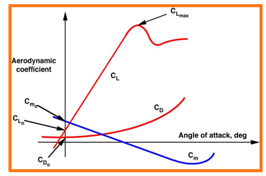
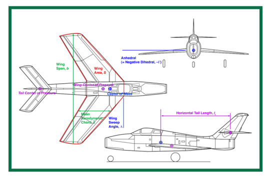
Low-Speed Aerodynamics

Robert Stengel, Aircraft Flight Dynamics, MAE 331, 2018

Learning Objectives

- **2D lift and drag**
- **Reynolds number effects**
- **Relationships between airplane shape and aerodynamic characteristics**
- **2D and 3D lift and drag**
- **Static and dynamic effects of aerodynamic control surfaces**





Reading:
Flight Dynamics
Aerodynamic Coefficients, 65-84

http://www.princeton.edu/~stengel/MAE331.html
http://www.princeton.edu/~stengel/FlightDynamics.html

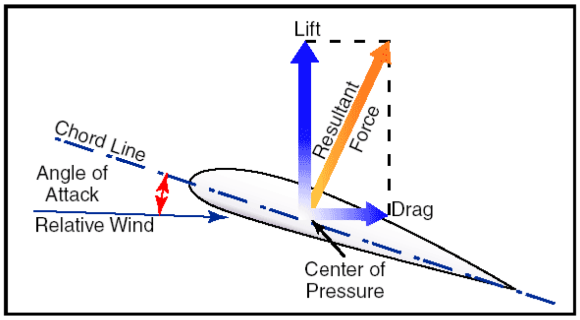
1

2-Dimensional Aerodynamic Lift and Drag

2

Wing Lift and Drag

- **Lift: Perpendicular to free-stream airflow**
- **Drag: Parallel to the free-stream airflow**



3

Longitudinal Aerodynamic Forces

Non-dimensional force coefficients, C_L and C_D, are dimensionalized by

dynamic pressure, $\bar{q}$, N/m^2 or $lb/sq\ ft$

reference area, S, m^2 of ft^2

$$Lift = C_L \bar{q}\, S = C_L \left(\frac{1}{2} \rho V^2 \right) S$$

$$Drag = C_D \bar{q}\, S = C_D \left(\frac{1}{2} \rho V^2 \right) S$$

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Circulation of Incompressible Air Flow About a 2-D Airfoil

Bernoulli's equation (inviscid, incompressible flow)

(Motivational, but not the whole story of lift)

$$p_{static} + \frac{1}{2}\rho V^2 = \text{constant along streamline} = p_{stagnation}$$

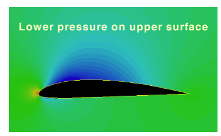
Vorticity at point *x*

$$V_{upper}(x) = V_\infty + \Delta V(x)/2$$
$$V_{lower}(x) = V_\infty - \Delta V(x)/2$$

$$\gamma_{2-D}(x) = \frac{\Delta V(x)}{\Delta z(x)}$$

Circulation about airfoil

$$\Gamma_{2-D} = \int_0^c \gamma_{2-D}(x)dx = \int_0^c \frac{\Delta V(x)}{\Delta z(x)}dx$$



Relationship Between Circulation and Lift

Differential pressure along chord section

$$\Delta p(x) = \left[p_{static} + \frac{1}{2}\rho_\infty\left(V_\infty + \Delta V(x)/2\right)^2\right] - \left[p_{static} + \frac{1}{2}\rho_\infty\left(V\infty - \Delta V(x)/2\right)^2\right]$$
$$= \frac{1}{2}\rho_\infty\left[\left(V_\infty + \Delta V(x)/2\right)^2 - \left(V_\infty - \Delta V(x)/2\right)^2\right]$$
$$= \rho_\infty V_\infty \Delta V(x) = \rho_\infty V_\infty \Delta z(x)\gamma_{2-D}(x)$$

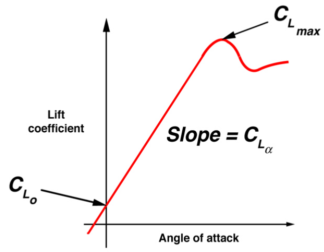
2-D Lift (inviscid, incompressible flow)

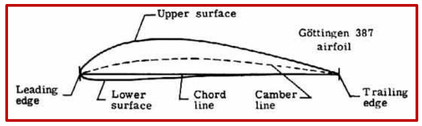
$$(Lift)_{2-D} = \int_0^c \Delta p(x)dx = \rho_\infty V_\infty \int_0^c \gamma_{2-D}(x)dx = \rho_\infty V_\infty (\Gamma)_{2-D}$$

$$\simeq \frac{1}{2}\rho_\infty V_\infty^{\ 2} c(2\pi\alpha)[\text{thin, symmetric airfoil}] + \rho_\infty V_\infty (\Gamma_{camber})_{2-D}$$
$$\simeq \frac{1}{2}\rho_\infty V_\infty^{\ 2} c\left(C_{L_\alpha}\right)_{2-D}\alpha + \rho_\infty V_\infty (\Gamma_{camber})_{2-D}$$

Lift vs. Angle of Attack

2-D Lift (inviscid, incompressible flow)





$$(\text{Lift})_{2\text{-}D} \simeq \left[\frac{1}{2}\rho_\infty V_\infty^{\,2} c \left(C_{L_\alpha}\right)_{2-D} \alpha\right] + \left[\rho_\infty V_\infty \left(\Gamma_{camber}\right)_{2-D}\right]$$

= [Lift due to angle of attack]

+ [Lift due to camber]

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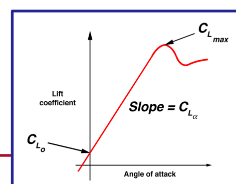
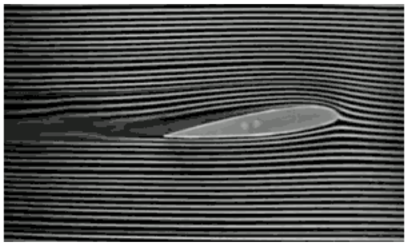
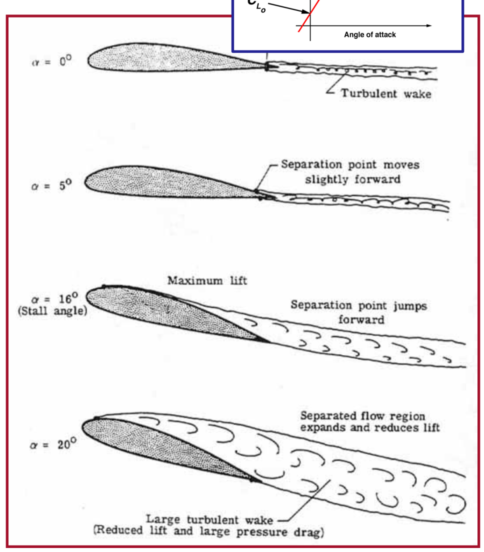
Typical Flow Variation with Angle of Attack



- At higher angles,
  - flow separates
  - wing loses lift
- Flow separation produces stall





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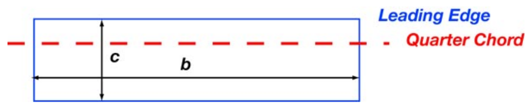
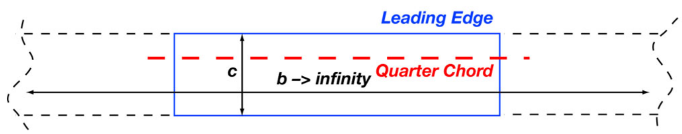
What Do We Mean by 2-Dimensional Aerodynamics?

Finite-span wing –> finite aspect ratio

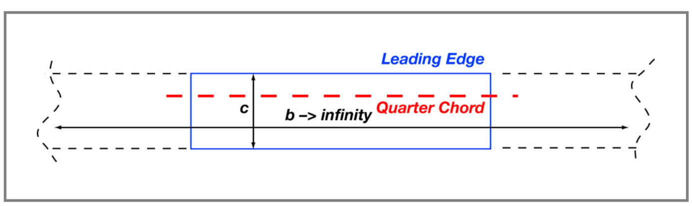


$$AR = \frac{b}{c} \quad \textit{rectangular wing}$$
$$= \frac{b \times b}{c \times b} = \frac{b^2}{S} \quad \textit{any wing}$$

Infinite-span wing –> infinite aspect ratio



What Do We Mean by 2-Dimensional Aerodynamics?



Assuming constant chord section, the "2-D Lift" is the same at any *y* station of the infinite-span wing

$$Lift_{3-D} = C_{L_{3-D}} \frac{1}{2} \rho V^2 S = C_{L_{3-D}} \frac{1}{2} \rho V^2 (bc) \quad \text{[Rectangular wing]}$$

$$\Delta\left(Lift_{3-D}\right) = C_{L_{3-D}} \frac{1}{2} \rho V^2 c \Delta y$$

$$\lim_{\Delta y \to \varepsilon > 0} \Delta\left(Lift_{3-D}\right) = \lim_{\Delta y \to \varepsilon > 0} \left(C_{L_{3-D}} \frac{1}{2} \rho V^2 c \Delta y \right) \Rightarrow \text{"2-D Lift"} \simeq C_{L_{2-D}} \frac{1}{2} \rho V^2 c$$

Effect of Sweep Angle on Lift

Unswept wing, symmetric airfoil, 2-D lift slope coefficient

Inviscid, incompressible flow

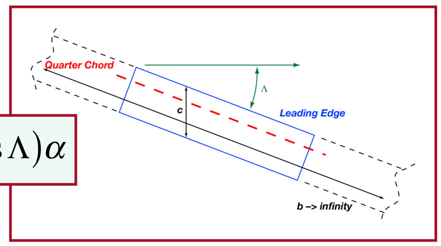
Referenced to chord length, *c*, rather than wing area

$$C_{L_{2-D}} = \alpha \left(\frac{\partial C_L}{\partial \alpha} \right)_{2-D} = \alpha \left(C_{L_\alpha} \right)_{2-D} = (2\pi)\alpha \quad \text{[Thin Airfoil Theory]}$$

Swept wing, 2-D lift slope coefficient

Inviscid, incompressible flow

$$C_{L_{2-D}} = \alpha \left(C_{L_\alpha} \right)_{2-D} = (2\pi \cos \Lambda)\alpha$$



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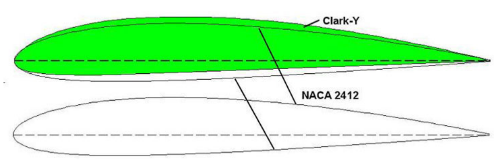
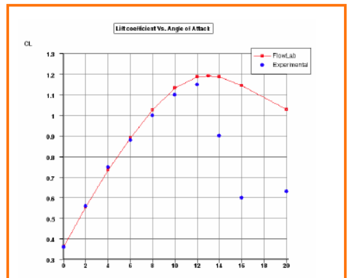
Classic Airfoil Profiles



- NACA 4-digit Profiles (*e.g., NACA 2412*)
  - Maximum camber as percentage of chord (2) = 2%
  - Distance of maximum camber from leading edge, (4) = 40%
  - Maximum thickness as percentage of chord (12) = 12%
- Clark Y (1922): Flat lower surface, 11.7% thickness
  - GA, WWII aircraft
  - Reasonable *L/D*
  - Benign theoretical stall characteristics
  - Experimental result is more abrupt



Fluent, Inc, 2007

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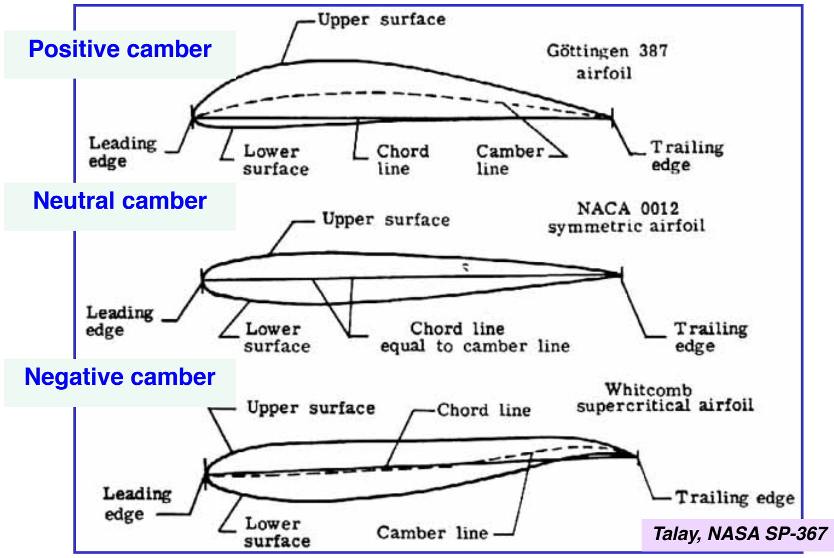
Typical Airfoil Profiles



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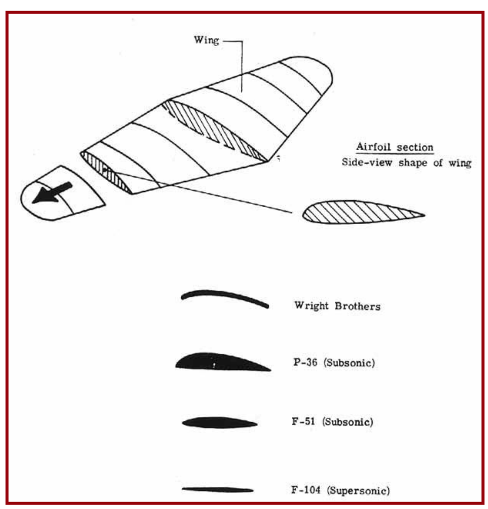
Airfoil Effects

- **Camber increases zero-*α* lift coefficient**
- **Thickness**
  - **increases *α* for stall and softens the stall break**
  - **reduces subsonic drag**
  - **increases transonic drag**
  - **causes abrupt pitching moment variation**
- **Profile design**
  - **can reduce center-of-pressure (static margin, *TBD*) variation with *α***
  - **affects leading-edge and trailing-edge flow separation**



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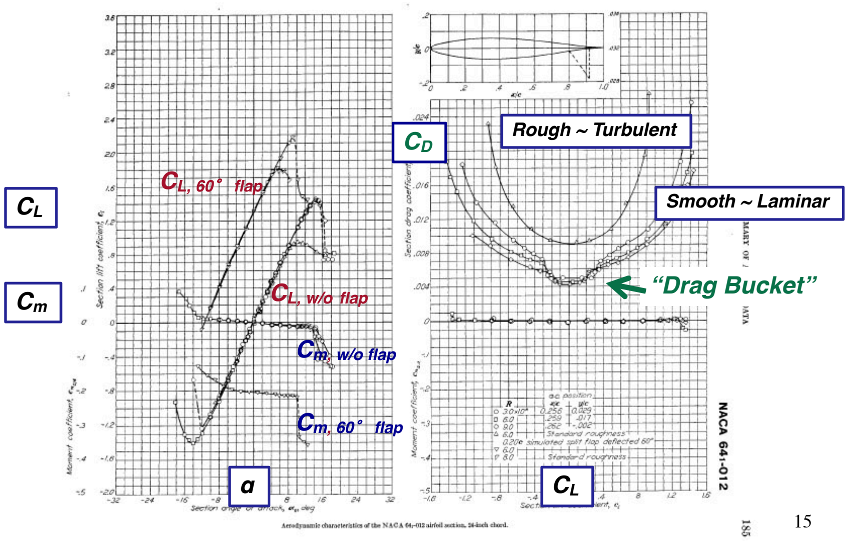
NACA 64_1-012 Chord Section Lift, Drag, and Moment (NACA TR-824)



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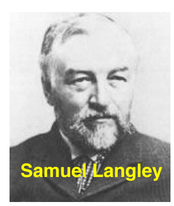
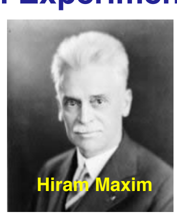
Historical Factoid

Measuring Lift and Drag with Whirling Arms and Early Wind Tunnels

Whirling Arm Experimentalists



Wind Tunnel Experimentalists



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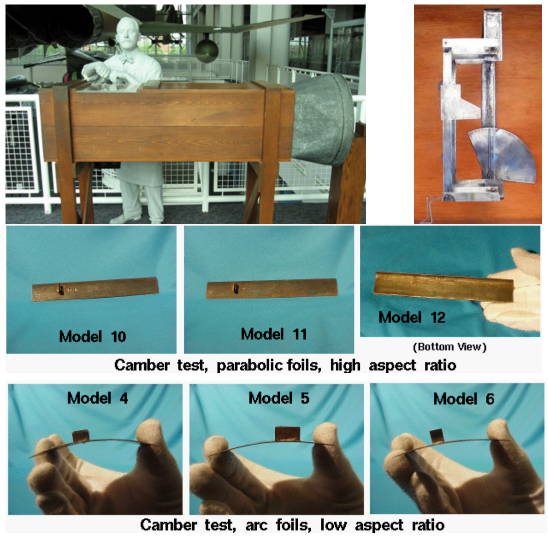
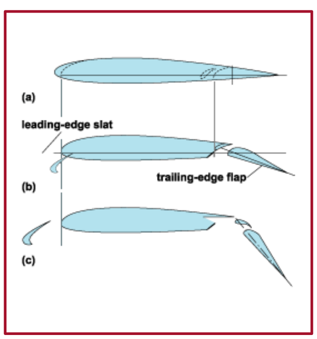
Historical Factoid

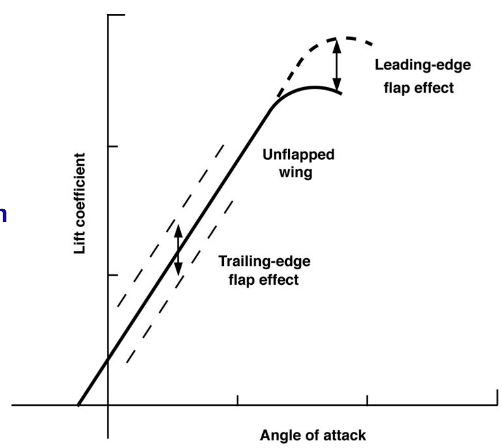
Wright Brothers Wind Tunnel





Flap Effects on Aerodynamic Lift

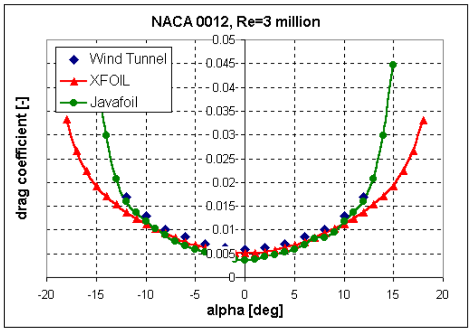
- Camber modification
- Trailing-edge flap deflection shifts C_L up and down
- Leading-edge flap (*slat*) deflection increases stall α
- Same effect applies for other control surfaces
  - Elevator (horizontal tail)
  - Ailerons (wing)
  - Rudder (vertical tail)



Aerodynamic Drag

$$Drag = C_D \frac{1}{2}\rho V^2 S \approx \left(C_{D_0} + \varepsilon C_L^2\right)\frac{1}{2}\rho V^2 S$$

$$\approx \left[C_{D_0} + \varepsilon\left(C_{L_o} + C_{L_\alpha}\alpha\right)^2\right]\frac{1}{2}\rho V^2 S$$



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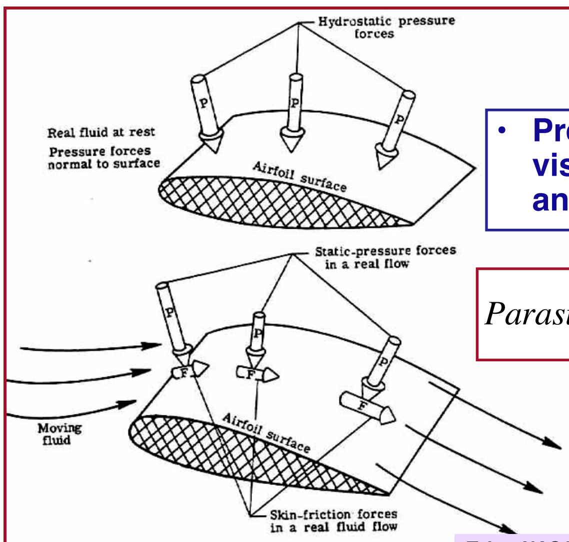
Parasitic Drag, C_{D_o}



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- Pressure differential, viscous shear stress, and separation

$$Parasitic\ Drag = C_{D_0}\frac{1}{2}\rho V^2 S$$

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Reynolds Number and Boundary Layer

$$Reynolds\ Number = \text{Re} = \frac{\rho V l}{\mu} = \frac{Vl}{\nu}$$

where

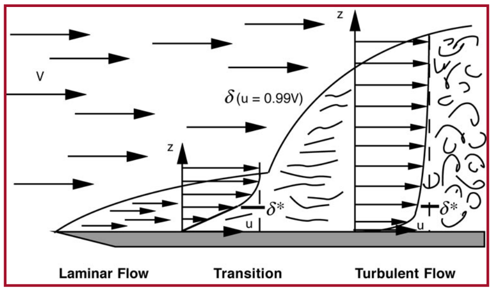
ρ = air density, kg/m^2

V = true airspeed, m/s

l = characteristic length, m

μ = absolute (dynamic) viscosity = $1.725 \times 10^{-5}\ kg/m \bullet s$

ν = kinematic viscosity (SL) = $1.343 \times 10^{-5}\ m/s^2$



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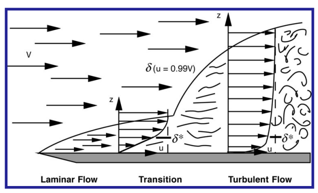
Reynolds Number, Skin Friction, and Boundary Layer



Skin friction coefficient for a flat plate

$$C_f = \frac{Friction\ Drag}{\bar{q} S_{wet}}$$

where S_{wet} = wetted area

<u>Wetted Area</u>: Total surface area of the wing or aircraft, subject to skin friction

Boundary layer thickens in transition, then thins in turbulent flow

$$C_f \approx 1.33 \text{Re}^{-1/2} \quad [laminar\ flow]$$

$$\approx 0.46 (\log_{10} \text{Re})^{-2.58} \quad [turbulent\ flow]$$

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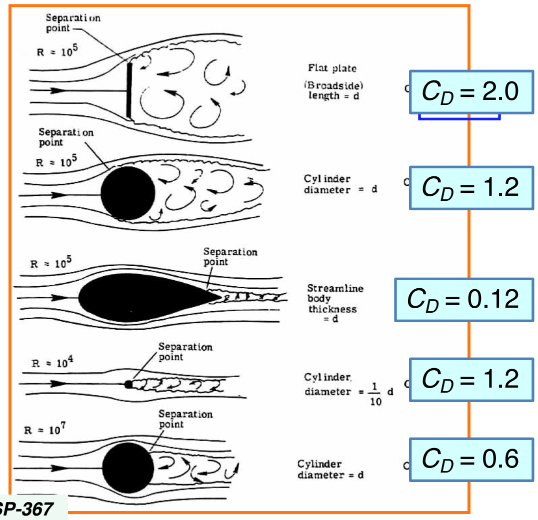
Effect of Streamlining on Parasitic Drag



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DRAG
https://www.youtube.com/watch?v=ylh1CPqBwEw

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Subsonic C_{D_o} Estimate (Raymer)

Table 12.3 Equivalent skin friction coefficients

$C_{D_0} = C_{fe} \dfrac{S_{wet}}{S_{ref}}$	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

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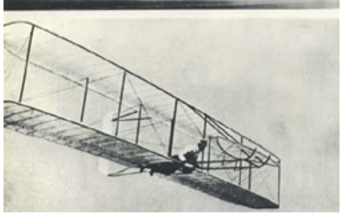
Historical Factoid

Wilbur *(1867-1912)* and Orville *(1871-1948)* Wright



- Bicycle mechanics from Dayton, OH
- Self-taught, empirical approach to flight
- Wind-tunnel, kite, and glider experiments
- Dec 17, 1903: Powered, manned aircraft flight ends in success

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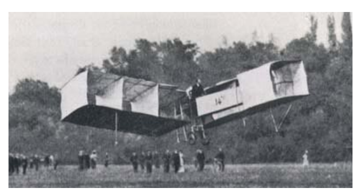
Historical Factoids

- 1906: 2nd successful aviator: Alberto Santos-Dumont, standing!
  - High dihedral, forward control surface
- Wrights secretive about results until 1908; few further technical contributions
- 1908: Glenn Curtiss *et al* incorporate ailerons
  - Separate aileron surfaces at right
  - Wright brothers sue for infringement of 1906 US patent (and win)
- 1909: Louis Bleriot's flight across the English Channel



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Description of Aircraft Configurations

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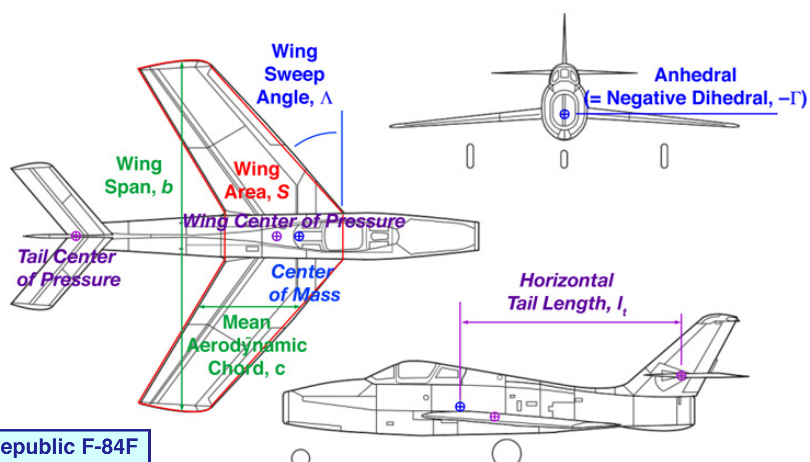
A Few Definitions



Republic F-84F
Thunderstreak

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Wing Planform Variables

Aspect Ratio

$$AR = \frac{b}{c} \quad \text{rectangular wing}$$

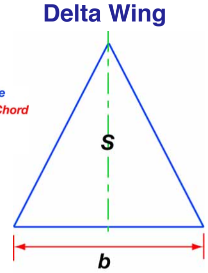
$$= \frac{b \times b}{c \times b} = \frac{b^2}{S} \quad \text{any wing}$$

Taper Ratio

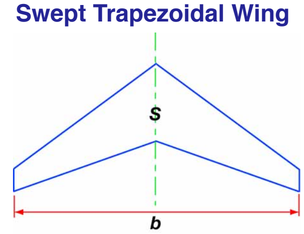
$$\lambda = \frac{c_{tip}}{c_{root}} = \frac{\text{tip chord}}{\text{root chord}}$$

Rectangular Wing

Delta Wing

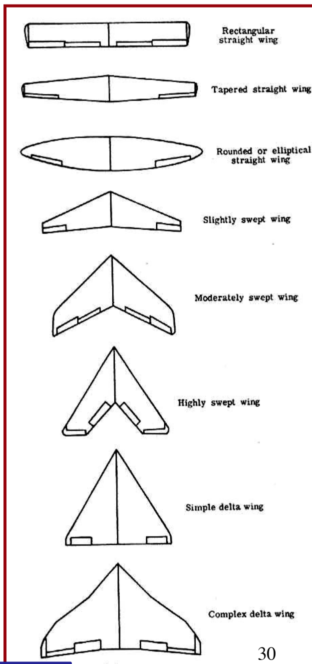


Swept Trapezoidal Wing



Wing Design Parameters

- **Planform**
  - Aspect ratio
  - Sweep
  - Taper
  - Complex geometries
  - Shapes at root and tip
- **Chord section**
  - Airfoils
  - Twist
- **Movable surfaces**
  - Leading- and trailing-edge devices
  - Ailerons
  - Spoilers
- **Interfaces**
  - Fuselage
  - Powerplants
  - Dihedral angle



(b) Examples of wing planform.

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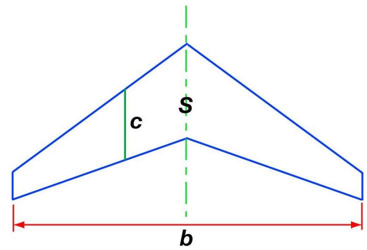
Mean Aerodynamic Chord, $\overline{c}$

Mean aerodynamic chord (m.a.c.) ~
mean geometric chord



$$\overline{c} = \frac{1}{S}\int_{-b/2}^{b/2} c^2(y)\,dy$$

$$= \left(\frac{2}{3}\right)\frac{1+\lambda+\lambda^2}{1+\lambda}c_{root} \quad \text{[for trapezoidal wing]}$$

$$\lambda = \frac{c_{tip}}{c_{root}} = \frac{\text{tip chord}}{\text{root chord}}$$

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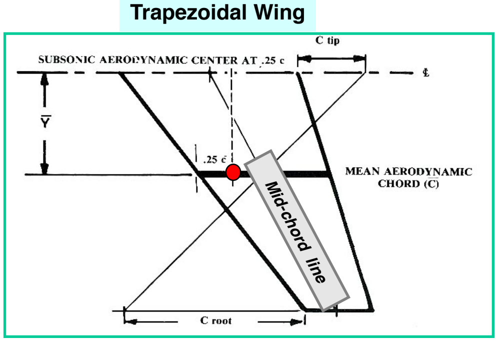
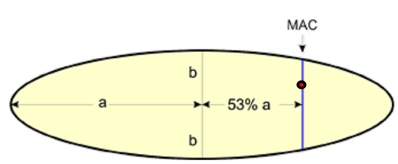
Location of Mean Aerodynamic Chord and Aerodynamic Center

- **Axial location of the wing's subsonic aerodynamic center (a.c.)**
  - **Determine spanwise location of m.a.c.**
  - **Assume that aerodynamic center is at 25% m.a.c.**

Trapezoidal Wing



Elliptical Wing



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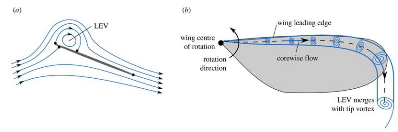
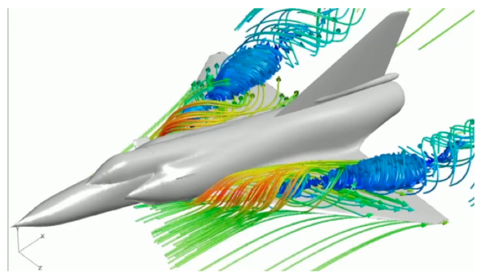
3-Dimensional Aerodynamic Lift and Drag

Insect Wing (flat plate)

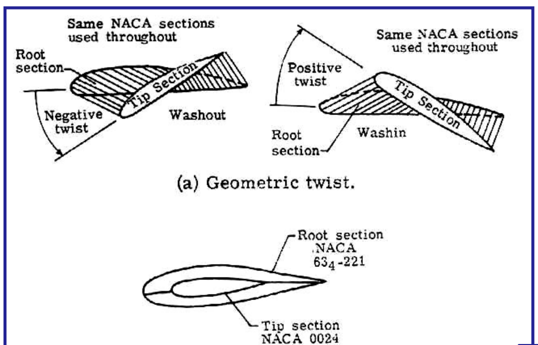


Delta Wing



Wing Twist Effects

- **Washout twist**
  - **reduces tip angle of attack**
  - **typical value: 2° - 4°**
  - **changes lift distribution (interplay with taper ratio)**
  - **reduces likelihood of tip stall**
  - **allows stall to begin at the wing root**
    - **separation"burble" produces buffet at tail surface, warning of stall**
  - **improves aileron effectiveness at high *α***



(a) Geometric twist.

(b) Aerodynamic twist.

Talay, NASA SP-367

Aerodynamic Strip Theory

- Airfoil section <u>may vary</u> from tip-to-tip
  - Chord length
  - Airfoil thickness
  - Airfoil profile
  - Airfoil twist
- **3-D Wing Lift**: Integrate 2-D lift coefficients of airfoil sections across finite span

Incremental lift along span

$$dL = C_{L_{2-D}}(y)c(y)\bar{q}dy = \frac{dC_{L_{3-D}}(y)}{dy}c(y)\bar{q}dy$$

3-D wing lift

$$L_{3-D} = \int_{-b/2}^{b/2} C_{L_{2-D}}(y)c(y)\bar{q}\,dy$$

Aero L-39 Albatros

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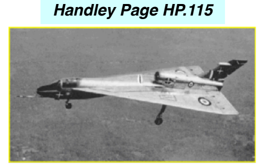
Bombardier Dash 8

Effect of Aspect Ratio on 3-Dimensional Wing Lift Slope Coefficient

(Incompressible Flow)

Handley Page HP.115



High Aspect Ratio (> 5) Wing

$$C_{L_\alpha} \triangleq \left(\frac{\partial C_L}{\partial \alpha}\right)_{3-D} = \frac{2\pi AR}{AR+2} = 2\pi\left(\frac{AR}{AR+2}\right)$$

Low Aspect Ratio (< 2) Wing

$$C_{L_\alpha} = \frac{\pi AR}{2} = 2\pi\left(\frac{AR}{4}\right)$$

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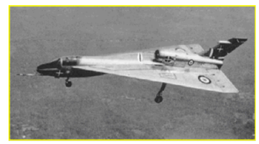
Effect of Aspect Ratio on 3-D Wing Lift Slope Coefficient

(Incompressible Flow)

All Aspect Ratios (Helmbold equation)

$$C_{L_\alpha} = \frac{\pi AR}{\left[1 + \sqrt{1 + \left(\frac{AR}{2}\right)^2}\right]}$$

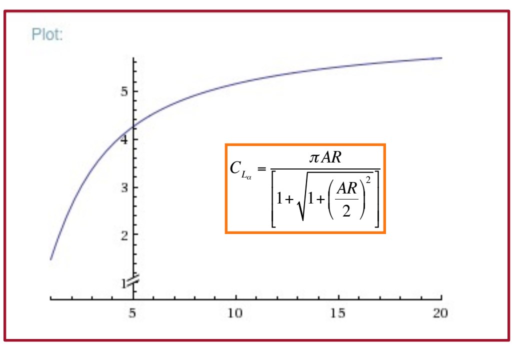


Effect of Aspect Ratio on 3-D Wing Lift Slope Coefficient

All Aspect Ratios (Helmbold equation)

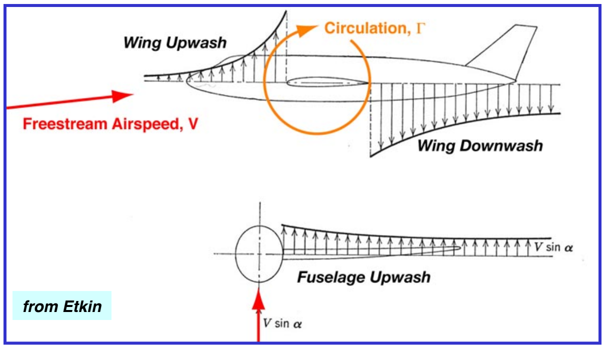
Wolfram Alpha (***https://www.wolframalpha.com/***)

plot(pi A / (1+sqrt(1 + (A / 2)^2)), A=1 to 20)



Wing-Fuselage Interference Effects

- Wing lift induces
  - Upwash in front of the wing affects canard
  - Downwash behind the wing affects aft tail
  - Local angles of attack modified, affecting net lift and pitching moment
- Flow around fuselage induces upwash on the wing, canard, and tail



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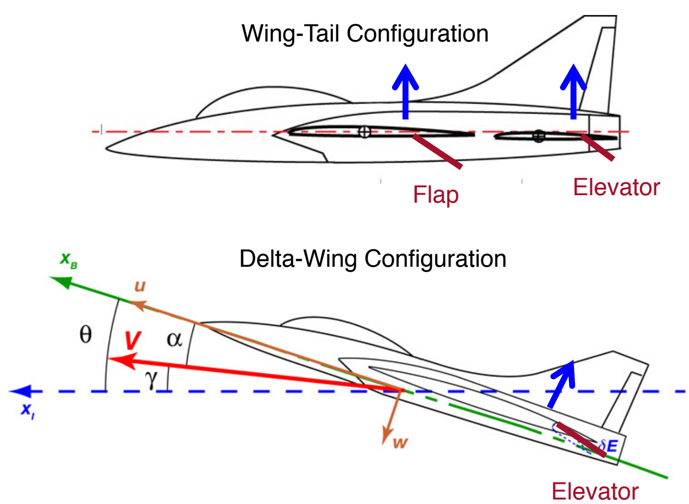
Longitudinal Control Surfaces



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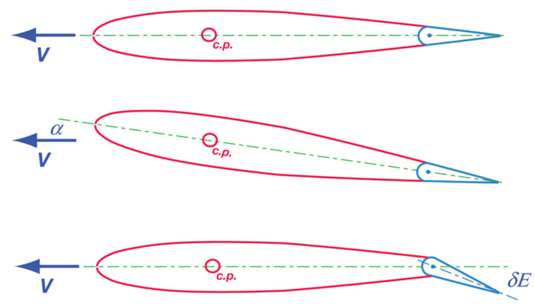
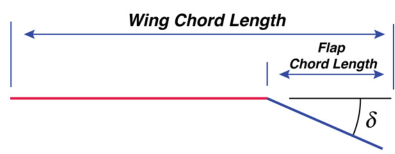
Angle of Attack and Control Surface Deflection

- **Horizontal tail with elevator control surface**
- **Horizontal tail at positive angle of attack**
- **Horizontal tail with positive elevator deflection**





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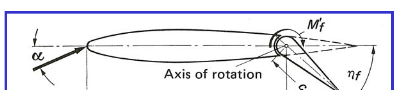
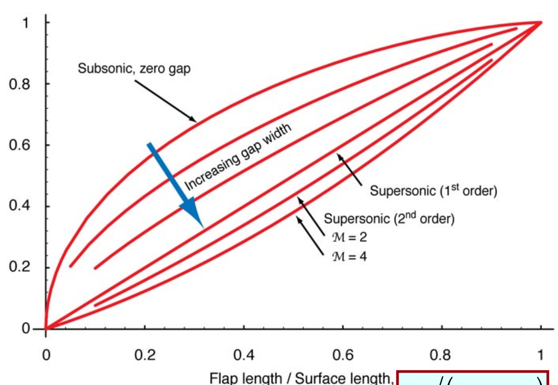
Control Flap Carryover Effect on Lift Produced By Total Surface



from Schlichting & Truckenbrodt

$$\frac{C_{L_{\delta E}}}{C_{L_\alpha}} \; vs. \; \frac{c_f}{x_f + c_f}$$



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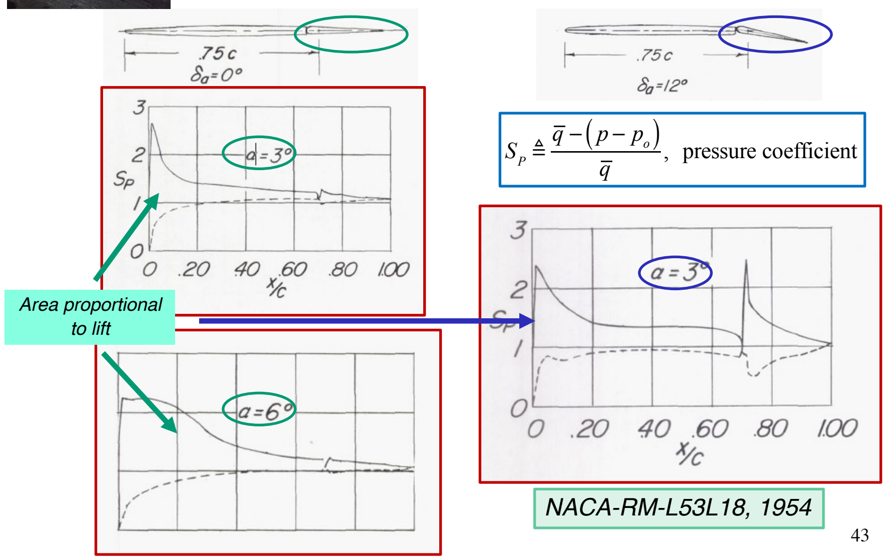
Bell X-1 Aileron Carryover Effect

$\mathcal{M} = 0.13$, $Re = 1.2 \times 10^6$



$$S_P \triangleq \frac{\bar{q} - (p - p_o)}{\bar{q}}, \text{ pressure coefficient}$$

Lift due to Elevator Deflection

Lift coefficient variation due to elevator deflection

$$C_{L_{\delta E}} \triangleq \frac{\partial C_L}{\partial \delta E} = \tau_{ht} \eta_{ht} \left(C_{L_\alpha}\right)_{ht} \frac{S_{ht}}{S}$$

$$\Delta C_L = C_{L_{\delta E}} \delta E$$

τ_{ht} = Carryover effect

η_{ht} = Tail efficiency factor

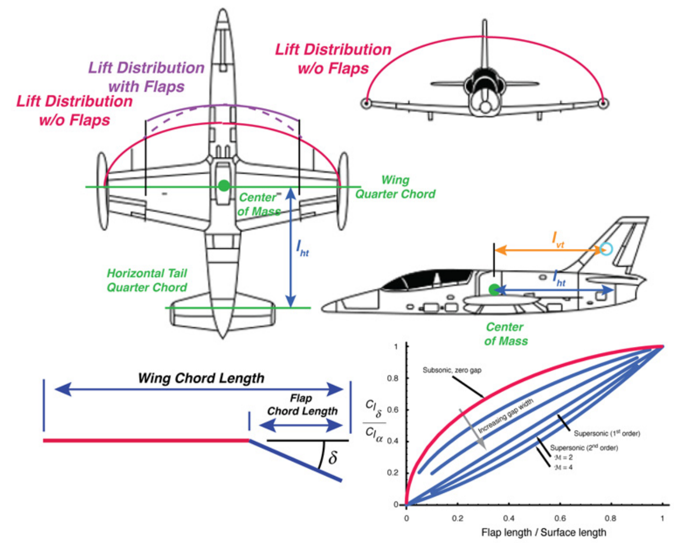
$\left(C_{L_\alpha}\right)_{ht}$ = Horizontal tail lift-coefficient slope

S_{ht} = Horizontal tail reference area

Lift variation due to elevator deflection

$$\Delta L = C_{L_{\delta E}} \bar{q} S \delta E$$

Example of Configuration and Flap Effects



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Next Time:
Induced Drag and High-Speed Aerodynamics

Reading:
Flight Dynamics
Aerodynamic Coefficients, 85-96
Airplane Stability and Control
Chapter 1

Learning Objectives

Understand drag-due-to-lift and effects of wing planform
Recognize effect of angle of attack on lift and drag coefficients
How to estimate Mach number (i.e., air compressibility) effects on aerodynamics
Be able to use Newtonian approximation to estimate lift and drag

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Supplementary Material

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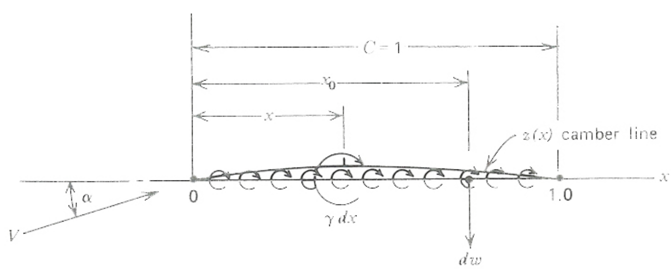
Thin Airfoil Theory



Downward velocity, *w*, at x_o due to vortex at *x*

Differential

$$dw(x_o) = \frac{\gamma(x)dx}{2\pi(x_o - x)}$$

Integral

$$w(x_o) = \frac{1}{2\pi}\int_0^1 \frac{\gamma(x)}{(x_o - x)}dx$$

Boundary condition: flow tangent to mean camber line

$$\frac{w(x_o)}{V} = \alpha - \left(\frac{dz}{dx}\right)_{x_o}$$

McCormick, 1995

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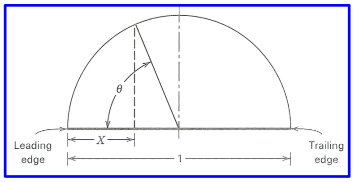
Thin Airfoil Theory

Integral equation for vorticity

$$\frac{1}{2\pi V}\int_0^1 \frac{\gamma(x)}{(x_o - x)}dx = \alpha - \left(\frac{dz}{dx}\right)_{x_o}$$

Coordinate transformation

$$x = \frac{1}{2}(1-\cos\theta)$$



Solution for vorticity

$$\gamma = 2V\left[A_0\frac{1+\cos\theta}{\sin\theta} + \sum_{n=1}^{\infty} A_n \sin n\theta\right]$$

Coefficients

$$A_0 = \alpha - \frac{1}{\pi}\int_0^{\pi}\frac{dz}{dx}d\theta$$

$$A_n = \frac{2}{\pi}\int_0^{\pi}\frac{dz}{dx}\cos n\theta\, d\theta$$

McCormick, 1995

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Thin Airfoil Theory

Lift, from Kutta-Joukowski theorem

$$L = \int_0^1 \rho V\gamma(x)dx = 2\pi A_0 + \pi A_1$$

For thin airfoil with circular arc

$$A_0 = \alpha, \quad A_1 = 4z_{\max}$$

$$C_{L_{2-D}} = 2\pi\alpha + 4\pi z_{\max} = C_{L_\alpha}\alpha + C_{L_o} \quad \text{[Circular arc]}$$

$$= C_{L_\alpha}\alpha \quad \text{[Flat plate]}$$

$$C_{L_\alpha} = \frac{\partial C_L}{\partial\alpha} = 2\pi$$

McCormick, 1995

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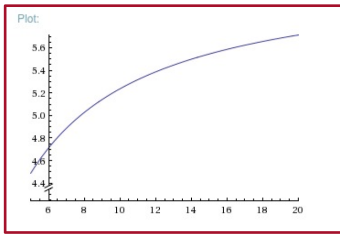
Effect of Aspect Ratio on 3-Dimensional Wing Lift Slope Coefficient

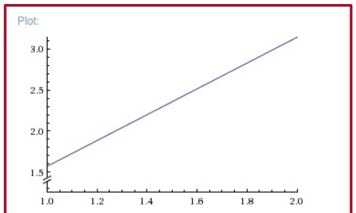
- **High Aspect Ratio (> 5) Wing**
  - Wolfram Alpha

plot(2 pi (a/(a+2)), a=5 to 20)



- **Low Aspect Ratio (< 2) Wing**
  - Wolfram Alpha

plot(2 pi (a / 4), a=1 to 2)



Aerodynamic Stall, Theory and Experiment

- **Flow separation produces stall**
- **Straight rectangular wing, AR = 5.536, NACA 0015**
- **Hysteresis for increasing/decreasing *α***

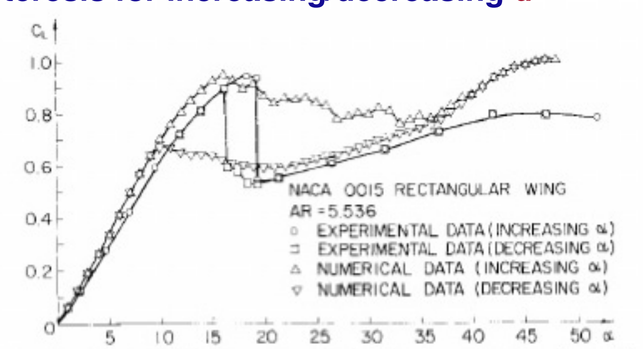


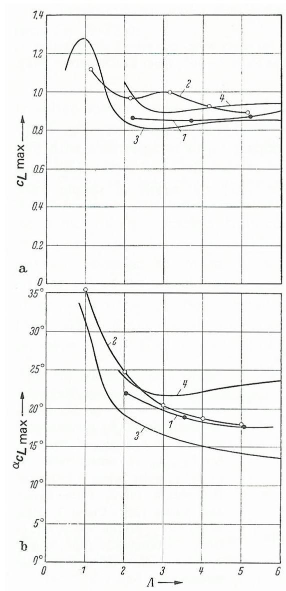
Fig. 3 Lift coefficient vs angle of attack for a rectangular wing with an NACA 0015 airfoil; comparison between experiment[7] and the present numerical technique.

Anderson et al, 1980

Maximum Lift of Rectangular Wings

Maximum Lift Coefficient, $C_{L\ max}$

Angle of Attack for $C_{L\ max}$



Aspect Ratio

Schlicting & Truckenbrodt, 1979

Figure 3-53 Maximum lift coefficients of rectangular wings ($\varphi = 0$) and swept-back wings of constant chord ($\varphi \neq 0$), Reynolds number $Re \approx 10^6$. (*a*) Maximum lift coefficient $c_{L\max}$ vs. aspect ratio Λ. (*b*) angle of attack α for $c_{L\ \max}$ vs. aspect ratio Λ. Curve 1, $\varphi = 0°$; profile NACA 0015, from Bussmann and Kopfermann [25]. Curve 2, $\varphi = 45°$; profile NACA 0012, from Truckenbrodt [85]. Curve 3, $\varphi = 0°$; $\delta \approx 0.10$, mean values of various measurements. Curve 4, $\varphi = 35°$; $\delta \approx 0.10$, mean values of various measurements.

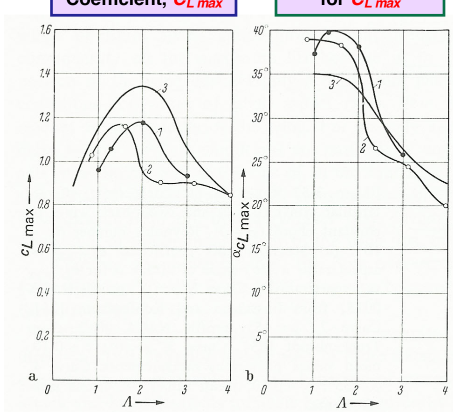
φ : Sweep angle

δ: Thickness ratio

Maximum Lift of Delta Wings with Straight Trailing Edges

Maximum Lift Coefficient, $C_{L\ max}$

Angle of Attack for $C_{L\ max}$



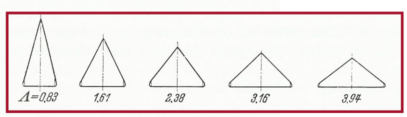


Figure 3-55 Maximum lift coefficients of delta wings, Reynolds number $Re \approx 10^6$. (*a*) Maximum lift coefficient $c_{L\max}$ vs. aspect ratio Λ. (*b*) Angle of attack α for $c_{L,\max}$ vs. aspect ratio Λ. Curve 1, delta wing; $\lambda = 0$; profile NACA 0012, from Lange and Wacke [25]. Curve 2, delta wing; $\lambda = \frac{1}{8}$; profile NACA 0012, from Truckenbrodt [85]. Curve 3, mean values of various measurements.

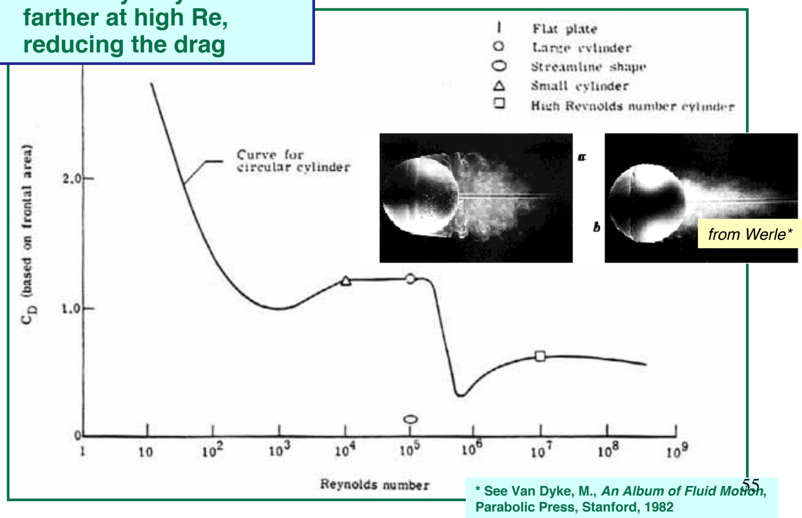
Aspect Ratio | Aspect Ratio

λ: Taper ratio

Schlicting & Truckenbrodt, 1979

Typical Effect of Reynolds Number on Parasitic Drag

- Flow may stay attached farther at high Re, reducing the drag



* See Van Dyke, M., *An Album of Fluid Motion*, Parabolic Press, Stanford, 1982

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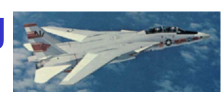
Aft Flap vs. All-Moving Control Surface



- Carryover effect of aft flap
  - Aft-flap deflection can be almost as effective as full surface deflection at subsonic speeds
  - Negligible at supersonic speed
- Aft flap
  - Mass and inertia lower, reducing likelihood of mechanical instability
  - Aerodynamic hinge moment is lower
  - Can be mounted on structurally rigid main surface

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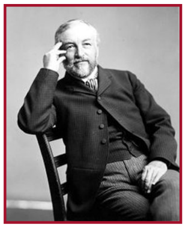
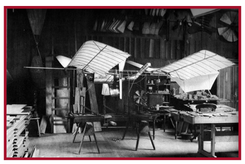
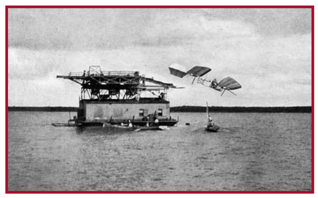


Historical Factoid

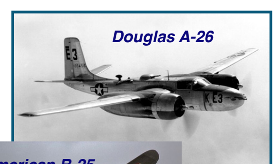
Samuel Pierpoint Langley *(1834-1906)*

- Astronomer supported by Smithsonian Institution
- Whirling-arm experiments
- **1896:** Langley's steam-powered ***Aerodrome*** model flies 3/4 mile
- **Oct 7 & Dec 8, 1903:** Manned aircraft flights end in failure

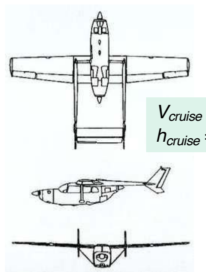

Multi-Engine Aircraft of World War II

- Large W.W.II aircraft had unpowered controls:
  - High foot-pedal force
  - Rudder stability problems arising from balancing to reduce pedal force
- Severe engine-out problem for twin-engine aircraft



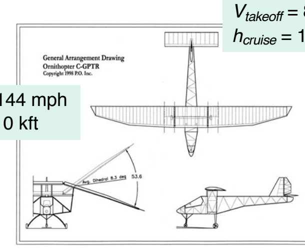
Medium to High Aspect Ratio Configurations

Cessna 337



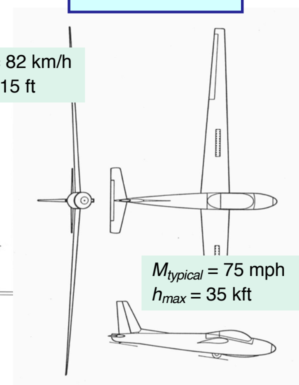
V_{cruise} = 144 mph
h_{cruise} = 10 kft

DeLaurier Ornithopter



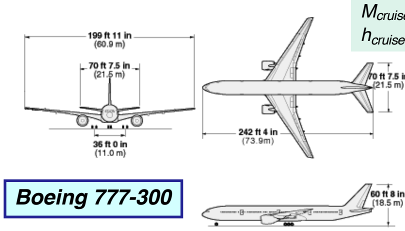
$V_{takeoff}$ = 82 km/h
h_{cruise} = 15 ft

Schweizer 2-32



$M_{typical}$ = 75 mph
h_{max} = 35 kft

Boeing 777-300



M_{cruise} = 0.84
h_{cruise} = 35 kft

- Typical for subsonic aircraft

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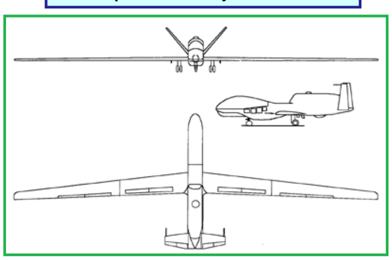
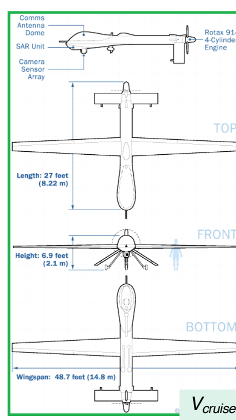
Uninhabited Air Vehicles

Northrop-Grumman/Ryan Global Hawk



V_{cruise} = 310 kt
h_{cruise} = 50 kft

General Atomics Predator



V_{cruise} = 70-90 kt
h_{cruise} = 25 kft

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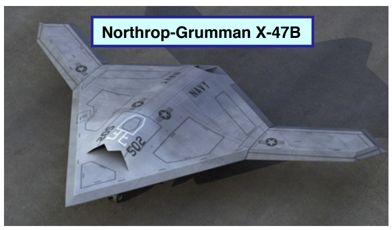
Stealth and Small UAVs

Lockheed-Martin RQ-170

General Atomics Predator-C (Avenger)

Northrop-Grumman X-47B

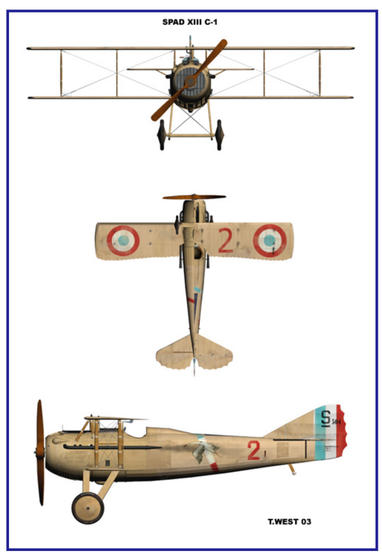


InSitu/Boeing ScanEagle

http://en.wikipedia.org/wiki/Stealth_aircraft

Subsonic Biplane

- Compared to monoplane
  – Structurally stiff (guy wires)
  – Twice the wing area for the same span
  – Lower aspect ratio than a single wing with same area and chord
  – Mutual interference
  – Lower maximum lift
  – Higher drag (interference, wires)
- Interference effects of two wings
  – Gap
  – Aspect ratio
  – Relative areas and spans
  – Stagger



Some Videos

Flow over a narrow airfoil, with downstream vortices

http://www.youtube.com/watch?v=zsO5BQA_CZk

Flow over transverse flat plate, with downstream vortices

http://www.youtube.com/watch?v=0z_hFZx7qvE

Laminar vs. turbulent flow

http://www.youtube.com/watch?v=WG-YCpAGgQQ&feature=related

Smoke flow visualization, wing with flap

http://www.youtube.com/watch?feature=fvwp&NR=1&v=eBBZF_3DLCU/

1930s test in NACA wind tunnel

http://www.youtube.com/watch?v=3_WgkVQWtno&feature=related