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

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

2-Dimensional Aerodynamic Lift and Drag
Wing Lift and Drag

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

Longitudinal Aerodynamic Forces

Non-dimensional force coefficients, $C_L$ and $C_D$, are dimensionalized by
dynamic pressure, $\overline{q}$, N/m$^2$ or lb/sq ft
reference area, $S$, m$^2$ or ft$^2$

$$Lift = C_L \overline{q} S = C_L \left( \frac{1}{2} \rho V^2 \right) S$$
$$Drag = C_D \overline{q} S = C_D \left( \frac{1}{2} \rho V^2 \right) S$$
Circulation of Incompressible Air Flow About a 2-D Airfoil

Bernoulli’s equation (inviscid, incompressible flow)
(Motivational, but not the whole story of lift)

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

Vorticity at point \( x \)

\[
\begin{align*}
V_{\text{upper}}(x) &= V_{\infty} + \frac{\Delta V(x)}{2} \\
V_{\text{lower}}(x) &= V_{\infty} - \frac{\Delta V(x)}{2}
\end{align*}
\]

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

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

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

2-D Lift (inviscid, incompressible flow)

\[
(Lift)_{2-D} = \int_0^c \Delta p(x) \, dx = \rho_\infty V_{\infty} \int_0^c \gamma_{2-D}(x) \, dx = \rho_\infty V_{\infty} (\Gamma)_{2-D}
\approx \frac{1}{2} \rho_\infty V_{\infty}^2 c (2\pi\alpha) \left[ \text{thin, symmetric airfoil} \right] + \rho_\infty V_{\infty} (\Gamma_{\text{camber}})_{2-D}
\approx \frac{1}{2} \rho_\infty V_{\infty}^2 c (C_{L0})_{2-D} \alpha + \rho_\infty V_{\infty} (\Gamma_{\text{camber}})_{2-D}
\]
Lift vs. Angle of Attack

2-D Lift (inviscid, incompressible flow)

\[
(Lift)_{2-D} = \left[ \frac{1}{2} \rho_{\infty} V_{\infty}^2 c \left( C_{L_0} \right)_{2-D} \alpha \right] + \left[ \rho_{\infty} V_{\infty} \left( \Gamma_{\text{camber}} \right)_{2-D} \right]
\]

\[
= [\text{Lift due to angle of attack}]
+ [\text{Lift due to camber}]
\]

Typical Flow Variation with Angle of Attack

- At higher angles,
  - flow separates
  - wing loses lift
- Flow separation produces stall
What Do We Mean by 2-Dimensional Aerodynamics?

Finite-span wing $\rightarrow$ finite aspect ratio

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

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

Infinite-span wing $\rightarrow$ infinite aspect ratio

$$\Delta \text{Lift}_{3-D} = C_{L_{3-D}} \frac{1}{2} \rho V^2 \Delta y$$

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

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

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

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
Typical Airfoil Profiles

Positive camber

Neutral camber

Negative camber

Airfoil Effects

• Camber increases zero-$\alpha$ lift coefficient
• Thickness
  – increases $\alpha$ 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 $\alpha$
  – affects leading-edge and trailing-edge flow separation
**Historical Factoid**

**Measuring Lift and Drag with Whirling Arms and Early Wind Tunnels**

**Whirling Arm Experimentalists**
- Otto Lilienthal
- Hiram Maxim
- Samuel Langley

**Wind Tunnel Experimentalists**
- Frank Wenham
- Gustave Eiffel
- Hiram Maxim
- Wright Brothers
**Historical Factoids**

**Wright Brothers Wind Tunnel**

1. Camber modification
2. Trailing-edge flap deflection shifts $C_L$ up and down
3. Leading-edge flap (slat) deflection increases stall $\alpha$
4. Same effect applies for other control surfaces
   - Elevator (horizontal tail)
   - Ailerons (wing)
   - Rudder (vertical tail)
Aerodynamic Drag

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

\[ \approx C_{D_0} + \varepsilon \left( C_{L_0} + C_{L_\alpha} \alpha \right)^2 \frac{1}{2} \rho V^2 S \]

Parasitic Drag, \( C_{D_0} \)

- Pressure differential, viscous shear stress, and separation

\[ \text{Parasitic Drag} = C_{D_0} \frac{1}{2} \rho V^2 S \]
Reynolds Number and Boundary Layer

\[ \text{Reynolds Number} = \text{Re} = \frac{\rho V l}{\mu} = \frac{V l}{\nu} \]

where
- \( \rho \) = air density, \( \text{kg/m}^2 \)
- \( V \) = true airspeed, \( \text{m/s} \)
- \( l \) = characteristic length, \( \text{m} \)
- \( \mu \) = absolute (dynamic) viscosity = \( 1.725 \times 10^{-5} \text{ kg/m} \cdot \text{s} \)
- \( \nu \) = kinematic viscosity (SL) = \( 1.343 \times 10^{-5} \text{ m/s}^2 \)

Reynolds Number, Skin Friction, and Boundary Layer

Skin friction coefficient for a flat plate

\[ C_f = \frac{\text{Friction Drag}}{\frac{1}{2} q S_{\text{wet}}} \]

where \( S_{\text{wet}} \) = wetted area

\textbf{Wetted Area:} Total surface area of the wing or aircraft, subject to skin friction

Boundary layer thickens in transition, then thins in turbulent flow

\[ C_f \approx 1.33 \text{Re}^{-1/2} \quad \text{[laminar flow]} \]
\[ \approx 0.46 (\log_{10} \text{Re})^{-2.58} \quad \text{[turbulent flow]} \]
Effect of Streamlining on Parasitic Drag

C_D = 2.0
C_D = 1.2
C_D = 0.12
C_D = 1.2
C_D = 0.6

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

Subsonic $C_{D_0}$ Estimate (Raymer)

Table 12.3 Equivalent skin friction coefficients

<table>
<thead>
<tr>
<th>$C_{D_0} = C_{f_e} \frac{S_{wc}}{S_{ref}}$</th>
<th>$C_{f_e}$-subsonic</th>
</tr>
</thead>
<tbody>
<tr>
<td>Bomber and civil transport</td>
<td>0.0030</td>
</tr>
<tr>
<td>Military cargo (high upsweep fuselage)</td>
<td>0.0035</td>
</tr>
<tr>
<td>Air Force fighter</td>
<td>0.0035</td>
</tr>
<tr>
<td>Navy fighter</td>
<td>0.0040</td>
</tr>
<tr>
<td>Clean supersonic cruise aircraft</td>
<td>0.0025</td>
</tr>
<tr>
<td>Light aircraft – single engine</td>
<td>0.0055</td>
</tr>
<tr>
<td>Light aircraft – twin engine</td>
<td>0.0045</td>
</tr>
<tr>
<td>Prop seaplane</td>
<td>0.0065</td>
</tr>
<tr>
<td>Jet seaplane</td>
<td>0.0040</td>
</tr>
</tbody>
</table>
**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
Description of Aircraft Configurations

A Few Definitions
### Wing Planform Variables

#### Aspect Ratio

For a rectangular wing:

\[ AR = \frac{b}{c} \]

For any wing:

\[ AR = \frac{b \times b}{c \times b} = \frac{b^2}{S} \]

#### Taper Ratio

\[ \lambda = \frac{c_{\text{tip}}}{c_{\text{root}}} = \frac{\text{tip chord}}{\text{root chord}} \]

![Rectangular Wing](Image)

![Delta Wing](Image)

![Swept Trapezoidal Wing](Image)

### 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

[Talay, NASA SP-367]
Mean Aerodynamic Chord, $\bar{c}$

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

mean geometric chord

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

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

$= \left( \frac{2}{3} \right) \frac{1 + \lambda + \lambda^2}{1 + \lambda} c_{root}$

[for trapezoidal wing]

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.
3-Dimensional Aerodynamic Lift and Drag

- **Insect Wing**
  - (flat plate)

- **Delta Wing**

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**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 $\alpha$

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Talay, NASA SP-367
Aerodynamic Strip Theory

- Airfoil section may vary 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

\[ dL = C_{L_{2-D}}(y)c(y)\bar{q}dy \]

\[ = \frac{dC_{L_{2-D}}(y)}{dy}c(y)\bar{q}dy \]

3-D wing lift

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

Effect of Aspect Ratio on 3-Dimensional Wing Lift Slope Coefficient (Incompressible Flow)

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

\[ C_{L_{\alpha}} = \frac{\pi AR}{2} = 2\pi \left( \frac{AR}{4} \right) \]
Effect of Aspect Ratio on 3-D Wing Lift Slope Coefficient

*(Incompressible Flow)*

All Aspect Ratios *(Helmbold equation)*

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

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

Longitudinal Control Surfaces

- Wing-Tail Configuration
- Delta-Wing Configuration
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

Control Flap Carryover Effect on Lift Produced By Total Surface

\[ C_{L_{\text{CF}}} \] vs. \[ \frac{c_f}{C_{L_{\alpha}}} \times x_f + c_f \]
Bell X-1 Aileron Carryover Effect
\( M = 0.13, \ Re = 1.2 \times 10^6 \)

\[ S_p = \frac{q - (p - p_e)}{\bar{q}}, \] pressure coefficient

Area proportional to lift

NACA-RM-L53L18, 1954

Lift due to Elevator Deflection

Lift coefficient variation due to elevator deflection

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

\[ \Delta C_L = C_{LSE} \delta E \]

\( \tau_{ht} = \) Carryover effect

\( \eta_{ht} = \) 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

\[ \Delta L = C_{LSE} \bar{q} S \delta E \]
Example of Configuration and Flap Effects

Next Time:
Induced Drag and High-Speed Aerodynamics

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
**Supplementary Material**

**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**

\[ w(x_o) = \frac{1}{2\pi} \int_{x_o}^{1} \frac{\gamma(x)}{x_o - x} dx \]

Boundary condition: flow tangent to mean camber line

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

McCormick, 1995
Thin Airfoil Theory

Integral equation for vorticity

\[
\frac{1}{2\pi V} \int_{x_0}^{x} \gamma(x) \, dx = \alpha - \frac{dz}{dx}
\]

Coordinate transformation

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

Solution for vorticity

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

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

\[ A_0 = \alpha, \quad A_1 = 4z_{\text{max}} \]

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

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

\[ C_{L_o} = \frac{\partial C_{L}}{\partial \alpha} = 2\pi \]

McCormick, 1995
Effect of Aspect Ratio on 3-Dimensional Wing Lift Slope Coefficient

- **High Aspect Ratio (> 5) Wing**
  - Wolfram Alpha
  
  \[
  \text{plot}(2 \pi \left(\frac{a}{a+2}\right), \ a=5 \text{ to } 20)
  \]

- **Low Aspect Ratio (< 2) Wing**
  - Wolfram Alpha
  
  \[
  \text{plot}(2 \pi \left(\frac{a}{4}\right), \ a=1 \text{ to } 2)
  \]

Aerodynamic Stall, Theory and Experiment

- Flow separation produces stall
- Straight rectangular wing, AR = 5.536, NACA 0015
- Hysteresis for increasing/decreasing \( \alpha \)

![Aerodynamic Stall Diagram](image)
Maximum Lift of Rectangular Wings

Schlicting & Truckenbrodt, 1979

Figure 3-53 Maximum lift coefficients of rectangular wings, sweepback wings of constant chord, $\psi = 0$. Reynolds number $Re = 10^7$. (a) Maximum lift coefficient $c_{l, \text{max}}$ vs. aspect ratio $\lambda$. (b) Angle of attack $\alpha$ for $c_{l, \text{max}}$ vs. aspect ratio $\lambda$. Curve 1, $\psi = 0^\circ$, profile NACA 0012, from Rusmann and Kopfermann [25]. Curve 2, $\psi = 45^\circ$, profile NACA 0012, from Truckenbrodt [85]. Curve 3, $\psi = 0^\circ$, $\delta = 0.10$, mean values of various measurements. Curve 4, $\psi = 35^\circ$, $\delta = 0.10$, mean values of various measurements.

$\psi$: Sweep angle
$\delta$: Thickness ratio

Maximum Lift of Delta Wings with Straight Trailing Edges

Schlicting & Truckenbrodt, 1979

Figure 3-55 Maximum lift coefficients of delta wings, Reynolds number $Re = 10^7$. (a) Maximum lift coefficient $c_{l, \text{max}}$ vs. aspect ratio $\lambda$. (b) Angle of attack $\alpha$ for $c_{l, \text{max}}$ vs. aspect ratio $\lambda$. Curve 1, delta wing, $\lambda = 0$, profile NACA 0012, from Lange and Wacke [25]. Curve 2, delta wing, $\lambda = \frac{1}{4}$, profile NACA 0012, from Truckenbrodt [85]. Curve 3, mean values of various measurements.

$\lambda$: Taper ratio
Typical Effect of Reynolds Number on Parasitic Drag

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

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

**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

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**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
  - $V_{takeoff} = 144$ mph
  - $h_{takeoff} = 10$ kft
  - $M_{cruise} = 0.84$
  - $h_{cruise} = 35$ kft

- DeLaurier Ornithopter
  - $V_{takeoff} = 82$ km/h
  - $h_{takeoff} = 15$ ft

- Schweizer 2-32
  - $M_{cruise} = 75$ mph
  - $h_{cruise} = 35$ kft

- Boeing 777-300
  - Typical for subsonic aircraft

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
Stealth and Small UAVs

- Lockheed-Martin RQ-170
- General Atomics Predator-C (Avenger)
- Northrop-Grumman X-47B
- InSitu/Boeing ScanEagle


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=zs05BQA_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?v=3_WgkVQWtno&feature=related

1930s test in NACA wind tunnel
http://www.youtube.com/watch?v=eBBZF_3DLCU/