Most people don’t realize that mankind can be divided into two great classes: those who take airfoil selection seriously, and those who don’t.

Typical Subsonic Methods: Panel Methods

- For subsonic inviscid flow, the flowfield can be found by solving an integral equation for the potential on the surface.
- This is done assuming a distribution of singularities along the surface, and finding the “strengths” of the singularities.
- The airfoil is represented by a series of (typically) straight line segments between “nodes”, and the nonpenetration boundary condition is typically satisfied at control points.
- Some version of a Kutta condition is required to close the system of equations.

See my Applied Computation Aero page for the derivation.
Comparison of Panel Method Pressure Distribution with Exact Conformal Transformation Results

XFOIL’s inviscid calculations use a panel method
The conformal mapping solution is from Antony Jameson
Convergence with increasing numbers of panels

NACA 0012 Airfoil, $\alpha = 8^\circ$
How to examine convergence: Lift

NACA 0012 Airfoil, $\alpha = 8^\circ$
Convergence with Panels: Moment

NACA 0012 Airfoil, $\alpha = 8^\circ$
Convergence with Panels: Drag

NACA 0012 Airfoil, $\alpha = 8^\circ$
Pressures: 20 and 60 panels

NACA 0012 airfoil, $\alpha = 8^\circ$
Pressures: 60 and 100 panels

NACA 0012 airfoil, $\alpha = 8^\circ$
Comparison with WT Data: Lift
- recall: panel methods are inviscid!

![Graph showing lift coefficient ($C_L$) vs. angle of attack ($\alpha$) for NACA 0012 and NACA 4412 airfoils. The graph compares the panel method predictions with experimental data.](image)
Comparison with Data: Pitching Moment
- about the quarter chord -

\[ C_m, \text{ NACA 0012 - PANEL} \]
\[ C_m, \text{ NACA 4412 - PANEL} \]
\[ C_m, \text{ NACA 0012 - exp. data} \]
\[ C_m, \text{ NACA 4412 - exp. data} \]
For Completeness: Drag Data
Effect of Camber

Re = 6 million

NACA 4412
NACA 0012

data from Abbott and von Doehhoff
A Sidebar: Can you use thin airfoil theory?
Camber Effects: compared to WT Data

Camber effect on Alpha Zero Lift

WT Data From Hemke, *Elementary Applied Aerodynamics*
Thin airfoil theory from Houghton and Carpenter, pg 252
WT Data used in Comparison

From Hemke, *Elementary Applied Aerodynamics*

---

### Table 7-4: Fundamental Airfoil Characteristics

\[ N_R = \frac{\rho V c}{\mu} = \text{Reynolds' Number} = 8.0 \times 10^6 \]

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<th>( \alpha_{Le} ) (deg.)</th>
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*First number: Position of A.C. ahead of \( \varepsilon \) in \% \varepsilon; Second number: Position of A.C. above \( \varepsilon \) in \% \varepsilon.*

---

Redo our previous comparisons

How Well Does Linear Theory Work?

*Pretty Well!*

\[ C_L \text{ vs } \alpha \]

- \( C_L \), NACA 0012 - PANEL
- \( C_L \), NACA 4412 - PANEL
- \( C_L \), NACA 4412 - exp. data
- \( C_L \), NACA 0012 - exp. data
- \( CL \) 4412 LT
- \( CL \) 0012 LT
And the Pitching Moment

![Graph showing the pitching moment for NACA 0012 and 4412 panels, with lines and data points indicating the relationship between pitching moment and angle of attack.]
Comparison of inviscid prediction with WT Pressure Distribution

Note viscous “relief” (loss) of full pressure recovery at the trailing edge
XFOIL: the code for subsonic airfoils

• Panel Methods: Inviscid!
• Couple with a BL analysis to include viscous effects
• The single element viscous subsonic airfoil analysis method of choice: XFOIL – by Prof. Mark Drela at MIT
• XFOIL also has a “inverse” option
• Link available from my software site
Airfoil pressures: What to look for

- Trailing edge pressure recovery
- Leading edge stagnation point
- Expansion/recovery around leading edge (minimum pressure or max velocity, first appearance of sonic flow)
- Rapidly accelerating flow, favorable pressure gradient
- Upper surface pressure recovery (adverse pressure gradient)

NACA 0012 airfoil, $\alpha = 4^\circ$
Effect of Angle of Attack

NACA 0012 airfoil
Inviscid calculation from PANEL

$CP$ vs $x/c$ for different angles of attack ($\alpha = 0^\circ$, $\alpha = 4^\circ$, $\alpha = 8^\circ$).
Thickness Effects
Comparison of NACA 4-Digit Airfoils 0006, 0012, 0018

4-digit series fixes LE radius in relation to $t/c_{\text{max}}$, Modified 4-Digit allow you to control separately
Thickness Effects on Airfoil Pressures
Zero Lift Case

Inviscid calculation from PANEL

- NACA 0006, $\alpha = 0^\circ$
- NACA 0012, $\alpha = 0^\circ$
- NACA 0018, $\alpha = 0^\circ$
Thickness Effects on Airfoil Pressures, $C_L = 0.48$

![Graph showing $C_p$ vs. $x/c$ for NACA 0006, 0012, and 0018 airfoils with $\alpha = 4^\circ$.]

Small LE Radius leads to high acceleration around the LE
Camber Effects
Comparison of NACA 4-Digit Airfoils
the 0012 and 4412

\( y/c \)

\( x/c \)

- NACA 0012 (max t/c = 12%)
- NACA 4412 foil (max t/c = 12%)
Highly Cambered Airfoil Pressure Distribution
- NACA 4412 -

Note: For a comparison of cambered and uncambered pressure distributions at the same lift, see Fig. 18. Inviscid calculation from PANEL
Camber Effects on Airfoil Pressures, $C_L = 0.48$

Inviscid calculation from PANEL

- NACA 0012, $\alpha = 4^\circ$
- NACA 4412, $\alpha = 0^\circ$
Camber Effects on Airfoil Pressures, $C_L = 0.96$

Inviscid calculations from PANEL

- NACA 0012, $\alpha = 8^\circ$
- NACA 4412, $\alpha = 4^\circ$
Camber Effects on Airfoil Pressures, $C_L = 1.43$

Inviscid calculations from PANEL

- NACA 0012, $\alpha = 12^\circ$
- NACA 4412, $\alpha = 8^\circ$
For Completeness: Drag Data
Effect of Camber

Re = 6 million

NACA 4412
NACA 0012
data from Abbott and von Doehhoff
NACA 6712 Airfoil
- Heavy Aft Camber Geometry -

![Diagram of NACA 6712 Airfoil with heavy aft camber geometry](image)
NACA 6712 Airfoil
- Heavy Aft Camber, Pressure Distribution -

Inviscid calculations from PANEL

$\alpha = -0.6 \quad (C_L = 1.0)$
Whitcomb GA(W)-1 Airfoil

Note nearly parallel upper and lower surfaces at the trailing edge
Another Goal for Aerodynamics: Laminar Flow

For many years aerodynamicists have looked for ways to reduce skin friction drag by achieving laminar flow.

Werner Pfenninger: Active laminar flow X-plane

Northrop X-21: April 1963 - 1968

Active Laminar Flow Control via wing slots

Major re-work of Douglas WB-66D
AR = 7, LE Sweep = 30°
t/c = 0.10
Ultimately successful.
M = 0.745, Flights 120 and 121

Courtesy Tony Landis, personal collection
For Airfoils: NLF (Natural Laminar Flow)

Maintain a favorable pressure gradient for as long as possible, and then provide a pressure recovery as illustrated below.

\[ C_p \]

This is what the pressure recovery should look like.
Liebeck’s Hi-Lift Airfoil: Geometry and Lift
- note shape of pressure recovery -

From R.T. Jones, *Wing Theory*
Liebeck’s Hi-Lift Airfoil: Drag

From Bertin, Aerodynamics for Engineers
Camberline Design: DesCam

\[ \frac{(Z-Z_0)}{C} - \text{DesCam} \]

\[ \frac{Z}{C} - \text{from Abbott & vonDoenhoff} \]

Design Chord Loading

\[ \Delta C_p \]
Current Trend: “Morphing” Also called “adaptive” or “intelligent” Airfoil changes shape with flight condition

• Fighter have had scheduled LE and TE device deflections - since the 1970s (F-16 and F-18)
• Today transports are looking to “tweak” surfaces
• Example: the X-29 “Discrete Variable Camber”

AIAA 1983-1834, Mike Moore and Doug Frei
Effect on Lift and Drag

Note: Variable camber changes the lift curve slope

Results in an optimum envelope polar

AIAA 1983-1834, Mike Moore and Doug Frei
Airfoil Selection

Issues:

• Cruise $C_L$, and $C_{L_{\text{max}}}$, don’t forget $C_{m0}$
  - large LE radius?
  - Near parallel trailing edge closure
• Profile Drag: Laminar flow?
  – Tailor pressure distribution
• Thickness for low weight and internal volume
• Tails: often symmetric, 6 series foils picked

Study Abbott and von Doenhoff (both) as a start
To Conclude

You have the tools to do single element airfoil design