Some Transonic Aerodynamics

W.H. Mason

Configuration Aerodynamics Class
Transonic Aerodynamics History

• Pre WWII Prop speeds limited airplane speed
  – Props *did* encounter transonic losses
• WWII Fighters started to encounter transonic effects
  – Dive speeds revealed loss of control/Mach “tuck”
• Invention of the jet engine revolutionized airplane design
• Now, supersonic flow occurred over the wing at cruise
• Aerodynamics couldn’t be predicted, so was *mysterious*!
  – Wind tunnels didn’t produce good data
  – Flow was inherently nonlinear, and there were no theoretical methods

The Sound Barrier!

The P-38, and X-1 reveal transonic control problems/solutions
Airfoil Example: Transonic Characteristics

- From classical 6 series results

NACA TN No. 1396

Subsonic design pressures

Fig. 1

\[ \left( \frac{v}{V} \right)^2 \]

\[ x/c \]

NACA 66-210  \( \alpha = 1.0 \)
Figure 6: The variation of section lift coefficient with Mach number at various angles of attack for the NACA 66-210 airfoil.

From NACA TN 1396, by Donald Graham, Aug. 1947
Figure 7.- The variation of section drag coefficient with Mach number at various angles of attack for the NACA 66-210 airfoil.
Pitching Moment: a major problem!

Figure 8: The variation of section moment coefficient with Mach number at various angles of attack for the NACA 66-210 airfoil.

From NACA TN 1396, by Donald Graham, Aug. 1947
What’s going on?
The flow development illustration

From *Aerodynamics for Naval Aviators* by Hurt
Addressing the Testing Problem

- The tunnels would choke, shocks reflected from walls!
- Initial solutions:
  - Bumps on the tunnel floor
  - Test on an airplane wing in flight
  - Rocket and free-fall tests
- At Langley (1946-1948):
  - Make the tunnel walls porous: slots
  - John Stack and co-workers: the Collier Trophy
- Later at AEDC, Tullahoma, TN:
  - Walls with holes!

Wall interference is still an issue - corrections and uncertainty

See Becker *The High Speed Frontier* for the LaRC tunnel story
Wall Interference Solution 1: Slotted Tunnel

Grumman blow-down pilot of Langley tunnel
Wall Interference Solution 2: Porous Wall

The AEDC 4T, Tullahoma, TN
The Next Problem: Flow Similarity
- particularly critical at transonic speed -

• *Reynolds Number* *(Re)*
  – To simulate the viscous effects correctly, match the Reynolds Number
  – Usually you can’t match the Reynolds number, we’ll show you why and what aeros do about the problem

• *Mach Number* *(M)*
  – To match model to full scale compressibility effects, test at the same Mach number, sub-scale and full scale
Example of the Re Issue: The C-141 Problem

“The Need for developing a High Reynolds Number Transonic WT”
Astronautics and Aeronautics, April 1971, pp. 65-70
To help Match Reynolds Number

- Pressure Tunnels
- Cold Tunnels
  • Also keeps dynamic pressure “reasonable”
  • Also reduces power requirements
- Big Wind Tunnels
- Games with the boundary layer
  • Force transition from laminar to turbulent flow: “trips”

- or a combination of the above -
Example: Oil Flow of a transport wing showing both the location of the transition strip and the shock at $M = 0.825$.
Matching the Reynolds Number?

\[ Re = \frac{\rho V L}{\mu} \]

\( \rho \): density, \( V \): velocity, \( L \): length, \( \mu \): viscosity,

Use perfect gas law, and \( \mu = T^{0.9} \)

\[ Re = \frac{\sqrt{\gamma p} ML}{\sqrt{RT^{1.4}}} \]

Increase \( Re \) by increasing \( p \) or \( L \), decreasing \( T \) or changing the gas

Balance forces are related to, say, \( N = qSC_L \)

\[ q = \frac{\gamma}{2} pM^2 \]

Reducing \( T \) allows \( Re \) increase without huge balance forces

AIAA 72-995 or Prog. in Aero. Sciences, Vol. 29, pp. 193-220, 1992
WT vs Flight
why the National Transonic Facility (NTF) was built

“The Large Second Generation of Cryogenic Tunnels”
Astronautics and Aeronautics, October 1971, pp. 38-51

Uses cryogenic nitrogen as the test gas
Trying to match flight Re using cryogenic nitrogen: The NTF at NASA Langley, Hampton, VA

Performance:  
M = 0.2 to 1.20  
P_T = 1 to 9 atm  
T_T = 77° to 350° Kelvin  

Feb. 1982
Example NTF Model: The NASA/Grumman Research Fighter Configuration (RFC)

- Special metal for cryo temps
- Extensive safety analysis req’d.

Mason did this at Grumman
Now: the Theoretical/Computational Problem

- Despite lots of effort: no practical theory
- Numerical methods: from about 1970
  - Practical inviscid methods about 1971-1980 till today
    - Transonic Small Disturbance Theory (inviscid)
      - TSFOIL2 - available on our software site
    - Inviscid -viscous interaction methods
      » Viscous effects become more important
- So called “Full Potential Theory” (inviscid)
- Euler Equations (inviscid)
- Reynolds Averaged Navier-Stokes (RANS) viscous
Computational Transonics
the birth of CFD for Airplane Aerodynamics

• Earll Murman and Julian Cole simply try an innovative and entirely new approach
  – Use the Transonic Small Disturbance Equation

\[
\left[ 1 - M_{\infty}^2 - (\gamma + 1) M_{\infty}^2 \phi_x \right] \phi_{xx} + \phi_{yy} = 0
\]

>0 locally subsonic, elliptic PDE
<0 locally supersonic, hyperbolic PDE

• Model problem assumes flow is primarily in the \( x \)-direction
• Nonlinearity allows math type to change locally
The Murman-Cole Idea

• Replace the PDE with finite difference formulas for the derivatives - a so-called finite difference representation of the PDE
• Make a grid of the flowfield: at each grid point, write down the algebraic equation representation of the PDE
• Solve the resulting system of simultaneous nonlinear algebraic equations using an iterative process (SOR, SLOR, etc.)

• The new part!
  – At each grid point, test to see if the flow is locally subsonic or supersonic
  – If locally subsonic, represent the $\phi_{xx}$ with a central difference
  – Is locally supersonic, represent $\phi_{xx}$ with an “upwind” difference

• The result
  – The numerical model represents the essential physics of the flow
  – If there are shocks, they emerge during the solution (are captured)
  – Agrees with experimental data somewhat (and possibly fortuitously)
Using this notation for the grid:

Flow Direction

\[ \phi_{xx} = \frac{\phi_{i+1,j} - 2\phi_{i,j} + \phi_{i-1,j}}{(\Delta x)^2} + O(\Delta x)^2 \]

For subsonic flow:

\[ \phi_{xx} = \frac{\phi_{i,j} - 2\phi_{i-1,j} + \phi_{i-2,j}}{(\Delta x)^2} + O(\Delta x) \]

For supersonic flow:

Assume:

\[ \Delta x = \Delta y = \text{const.} \]
\[ x = i\Delta x \]
\[ y = j\Delta x \]

For a few more details, read Chapter 8, Introduction to CFD, on my Applied Computational Aerodynamics web page: http://www.aoe.vt.edu/~mason/Mason_f/CAtxtTop.html
The Original Murman & Cole Result

Note: $K$ is a similarity parameter.

One convergence check in CFD

- “Residual” is the sum of the terms in the PDE: should be zero
- Note log scale for residual
- Two iterative methods compared
- This is for a specific grid, also need to check with grid refinement

- 5% Thick Biconvex Airfoil
- 74 x 24 grid
- SOR $\omega=1.80$
- AF 2 factor: 1.333
CFD Grids

Most of the work applying CFD is grid generation

Today

- CFD has been developed *way* past this snapshot
- For steady solutions we iterate *in time*, until a steady state value is obtained
  - the problem is always hyperbolic in time!
- Key issues you need to understand
  - Implementing boundary conditions
  - Solution stability
  - Discretization error
  - Types of grids and related issues
  - Turbulence models

Can now use computational simulations as a numerical wind tunnel

- Especially in 2D, many codes include grid generation, and can be used to investigate airfoil characteristics and modifications
  - Remember adding “bumps” to the subsonic airfoil?
- **TSFOIL2** is an available and free transonic code, the next chart illustrates its accuracy
- **FLO36** was the ultimate full potential code
  - By Antony Jameson, a key contributor to CFD for many years, and should have been mentioned earlier.
- **MSES** is the key Euler/Interacting BL code
  - By Prof. Drela, also author of **XFOIL** and **AVL**
Comparison of inviscid flow model predictions
- Airfoils -

NACA 0012 airfoil, $M = 0.75$, $\alpha = 2^\circ$
An extra complication: Viscous effects are more important at Transonic Speeds

VISCOUS EFFECTS ON PRESSURE DISTRIBUTIONS
FOR A SUPERCritical AIRFOIL

(First public description of supercritical airfoils)
Finish for Today’s Lesson?
Next Lessons

Review:
• Transonic flowfields are inherently nonlinear
• Advances in both experimental and computational methods were required – and achieved

Today:
• Discussion of transonic airfoil characteristics and design goals

Subsonic Linear Theory Can't Predict Transonic Flow!

From Desta Alemayhu
Illustrates “tricks” used to get calculations to agree with test data
REVIEW: Obtaining CFD solutions

• Grid generation
• Flow solver
  – Typically solving 100,000s (or millions in 3D) of simultaneous nonlinear algebraic equations
  – An iterative procedure is required, and it’s not even guaranteed to converge!
  – Requires more attention and skill than linear theory methods
• Flow visualization to examine the results
Airfoils

Mach number effects: NACA 0012

NACA 0012 airfoil, FLO36 solution, $\alpha = 2^\circ$
Angle of attack effects: NACA 0012

FLO36
NACA 0012 airfoil, $M = 0.75$
“Traditional” NACA 6-series airfoil

Note continuous curvature all along the upper surface
Note small leading edge radius
Note low amount of aft camber

FLO36 prediction (inviscid)
$M = 0.72, \alpha = 0^\circ, C_L = 0.665$

Note strong shock
Note that flow accelerates continuously into the shock
Note the low aft loading associated with absence of aft camber.
A “new” airfoil concept - from Whitcomb

Progression of the Supercritical airfoil shape
“NASA Supercritical Airfoils,” by Charles D. Harris, NASA TP 2969, March 1990
What the supercritical concept achieved

Section drag at $C_N = 0.65$

Force limit for onset of upper-surface boundary layer separation

From “NASA Supercritical Airfoils,” by Charles D. Harris, NASA TP 2969, March 1990
And the Pitching Moment

From NASA Supercritical Airfoils, by Charles D. Harris, NASA TP 2969, March 1990
How Supercritical Foils are Different

From NASA Supercritical Airfoils, by Charles D. Harris, NASA TP 2969, March 1990
"Supercritical" Airfoils

Note low curvature all along the upper surface

Note large leading edge radius

Note large amount of aft camber

Note that the pressure distribution is "filled out", providing much more lift even though shock is weaker

Note the high aft loading associated with aft camber.

"Noisy" pressure distribution is associated with "noisy" ordinates, typical of NASA supercritical ordinate values

FLO36 prediction (inviscid) $M = 0.73$, $\alpha = 0^\circ$, $C_L = 1.04$
Whitcomb’s Four Design Guidelines

• An off-design criteria: a well behaved sonic plateau at $M = 0.025$ below the design $M$

• Gradient of pressure recovery gradual enough to avoid separation
  – in part: a thick TE, say 0.7% on a 10/11% thick foil

• Airfoil has aft camber so that design angle of attack is about zero, upper surface not sloped aft

• Gradually decreasing velocity in the supercritical region, resulting in a weak shock

Read “NASA Supercritical Airfoils,” by Charles D. Harris, NASA TP 2969, March 1990, for the complete story
Example: Airfoils 31 and 33
10–PERCENT–THICK NASA SUPERCRITICAL AIRFOILS 31 AND 33

The following charts are from the 1978 NASA Airfoil Conference, w/Mason's notes scribbled as Whitcomb spoke (rapidly)
Airfoils 31 and 33

Experimental pressure distributions near design $C_\ell$

Upper surface
Lower surface

Airfoil 31
$M = 0.78$
$C_\ell = 0.68$

Airfoil 33
$M = 0.77$
$C_\ell = 0.71$
Off Design

EXPERIMENTAL SONIC PLATEAU PRESSURE DISTRIBUTIONS

- UPPER SURFACE
- LOWER SURFACE

AIRFOIL 31
M = 0.76
\( \frac{c_l}{\text{constant}} = 0.46 \)

AIRFOIL 33
M = 0.76
\( \frac{c_l}{\text{constant}} = 0.41 \)

Figure 12

DROP IN M = 0.025
GET SONIC PLATE, IMPORTANCE FOR OFF DESIGN
Foils 31 and 33 Drag

EXPERIMENTAL DRAG CHARACTERISTICS FOR NASA SUPERCRITICAL AIRFOILS 31 AND 33

$C_l = 0.70$, $R = 6 \times 10^6$ TO $8 \times 10^6$

ONSET OF SEPARATION FOR BOTH AIRFOILS

$C_d$ vs. $M$

- AIRFOIL 31
- AIRFOIL 33

$70 \leq M \leq 78$
NASA Airfoils Developed Using the Guidelines

Filled symbols denote airfoils that were tested.

from “NASA Supercritical Airfoils,” by Charles D. Harris, NASA TP 2969, March 1990
Note: watch out for coordinates tabulated in NASA TP 2969!
**Frank Lynch’s Pro/Con Chart for supercritical airfoils**

### Choice of leading edge radius

- Airfoil thickness and high lift geometry (LE device?)

  **Increase favors**
  - clean wing low speed CLmax
  - Mach divergence at moderate to high lift coefficients

  **Decrease favors**
  - elimination of drag creep
  - drag divergence Mach numbers at low lift coefficients

### Choice of aft camber

- Design lift coefficient and flight Reynolds number

  **Increase favors**
  - clean wing low speed CLmax
  - drag divergence Mach number at high lift coefficients

  **Decrease favors**
  - trim drag by decreasing CM0
  - lower surface interference problems (flap hinge line fairings, etc.)
  - risk of premature separation at flight conditions
  - control surface hinge moments

### Other considerations

- spanwise location of airfoils on swept wings (root requires special treatment)
- chordwise distribution of thickness (determined by both aerodynamic and structural considerations)
Airfoil Limits: the Korn Eqn.

• We have a “rule of thumb” to let us estimate what performance we can achieve before drag divergence
  – By Dave Korn at NYU in the 70s

\[ M_{DD} + \frac{C_L}{10} + \left( \frac{t}{c} \right) = \kappa_A \]

\( \kappa_A = 0.87 \) for conventional airfoils (6 series)
\( \kappa_A = 0.95 \) for supercritical airfoils

Note: the equation is sensitive to \( \kappa_A \)
Airfoil Limits
Shevell and NASA Projections Compared to the Korn Equation

For the curious: the airfoil used on the X-29
Divergent Trailing Edge Airfoil
A NEW AIRFOIL DESIGN CONCEPT

by
P. A. Henne and R. D. Gregg
Douglas Aircraft Company
McDonnell Douglas Corporation

Just when we thought airfoil design was “finished”

Used on the MD-11 resisted in Seattle!

Hypothesized Flow Closure for Divergent Trailing Edge Airfoil

Comparison of DLBA 243 and DLBA 186 Calculated Drag Rise Characteristics

\[ R_c = 14.5 \times 10^6 \]

TRANSITION FIXED

<table>
<thead>
<tr>
<th>SYMBOL</th>
<th>AIRFOIL</th>
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<tbody>
<tr>
<td></td>
<td>DLBA 186</td>
</tr>
<tr>
<td></td>
<td>DLBA 243 (DTE)</td>
</tr>
</tbody>
</table>

\[ C_d \]

\[ C_f = 0.8 \]

\[ C_f = 0.6 \]

DESIGN POINT

MACH NUMBER

Aerospace and Ocean Engineering
Take a Look at the Pressure Distribution

Comparison of the DLBA 243 and the DLBA 186 Calculated Pressure Distribution at $M = 0.74$
Wings at Transonic Speed

The original DC-8 reflection plane model, in the VT WT Lab

An advanced Douglas transport WT Model, also in the VT WT Lab
The original model carved by George Shairer

Picture of B-52 made in Dayton Hotel over a weekend by Boeing engineers

picture taken by Mason at the Museum of Flight, Seattle, WA
The problem of transonic wing design
The Standard Transonic Test Case: the ONERA M6 Wing

A standard way to look at your results

The wing is symmetrical with no twist: note the unsweep of the shock and the build up of the pressure peak at the tip
Comparison: computations to data for the ONERA M6

\[ \frac{y}{b/2} = 0.655 \]

\[ \frac{y}{b/2} = 0.800 \]

Viscous Effects Still Important

Lockheed Wing A

From NASA CR178156, Oct. 1986

\[ \frac{y}{(b/2)} = 0.50 \]

\[ \frac{y}{(b/2)} = 0.70 \]
How to use advanced airfoil capability in wing design

**TWO WAYS OF USING SUPERCritical AIRFOIL**

**STRAIGHT WINGS**

- **Increasing $M_{Cruise}$**
  - SUPERCRITICAL
  - $\Delta M = 15\%$
  - $\Delta t/c = 42\%$
  - PROJECTED 58\%

- **Conventional**

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Re-winged T-2 Buckeye
Ways to use Advanced Aerodynamics
Frank Lynch’s Wing Chart

Instead of higher speed for same sweep and max thickness

\[ \Delta C_{DC} \]

\[ M \]

\[ \text{CONVENTIONAL} \]
\[ \text{SUPERCRITICAL} \]

For energy efficiency

\( \Delta \)

Reduce sweep

For similar \( M_0 \)
- Reduces weight
- Improved high-speed post-buffet pitch characteristics
- Improved low-speed performance

Increase thickness

For similar \( M_0 \)
- Reduces weight
- Increases fuel volume
- Increases \( C_{L_{\text{max}}} \)

Increase aspect ratio

- Reduced cruise drag
- Improved low-speed performance
Korn Eqn and Simple Sweep Theory

- a sort of Poor Man’s method (no CFD) -

Extend Korn to swept wings using simple sweep theory

\[ M_{dd} = \frac{\kappa_A}{\cos \Lambda} - \frac{(t / c)}{\cos^2 \Lambda} - \frac{c_l}{10 \cos^3 \Lambda} \]

and then use Lock’s approximation:

\[ C_D = 20 (M - M_{crit})^4 \]

where using: \( \frac{\partial C_D}{\partial M} = 0.1 \), \( \frac{\partial C_D}{\partial M} = 0.1 = 80 (M - M_{crit})^3 \)

so that we find \( M_{crit}: \)

\[ M_{crit} = M_{dd} - \left( \frac{0.1}{80} \right)^{1/3} \]
Transonic Drag Rise Estimate: B747

Example computed by Joel Grasmeyer

747-100 test data taken from Mair and Birdsall,
So how do you design the wing?

• Set the spanload in an MDO trade
• Linear theory gives a good first guess for the twist distribution
• Design the airfoils in 2D to start, and place in wing
• Finally, full 3D nonlinear analysis and design
• Special attention to root and tip mods to maintain isobar sweep
• Well known considerations:
  – Viscous effects are important
  – You must also model and include the fuselage
  – Need to address off design: buffet onset and overspeed
Jameson Wing Design for the MD-12

Comparision of Chordwise Pressure Distributions
MPX5X Wing-Body
REN = 101.00, MACH = 0.860

Solution 1
Upper-Surface Isobars
(Contours at 0.05 Cp)

Off Design Characteristics of the Jameson Wing

Transonic Transport Wing Design
Key Lessons learned at Boeing

• You can increase the root thickness w/o a drag penalty
• Nacelle/pylon wing interference can be done with CFD with almost no penalty
Finally: the fuselage can effect drag

Another Whitcomb Contribution: Fuselage Area Addition

Low mounted wing, $C_L = 0.3$

High mounted wing, $C_L = 0.3$

Adding a fuselage bump reduces drag slightly above $M = 0.85$: the Boeing 747 used this idea, it was supposed to cruise at $M = 0.90$

Fighter Wings: A more complicated story

Design Points for the F-15

Grumman proposed HiMAT, $M = 0.90$

Fighter Wings: Vortex Flow effects

$E$ vs. Lift Coefficient

- Advantage for attached flow
- Advantage for vortex flow

- Attached flow concept
- Vortex flow concept
To Conclude

• We’ve given a broad overview
  – Stressed the fundamental physics and issues

• Thousands of papers written on
  – Computational analysis methods
  – Design and optimization methods
  – Applications and design Concepts

• You are prepared to start contributing