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!

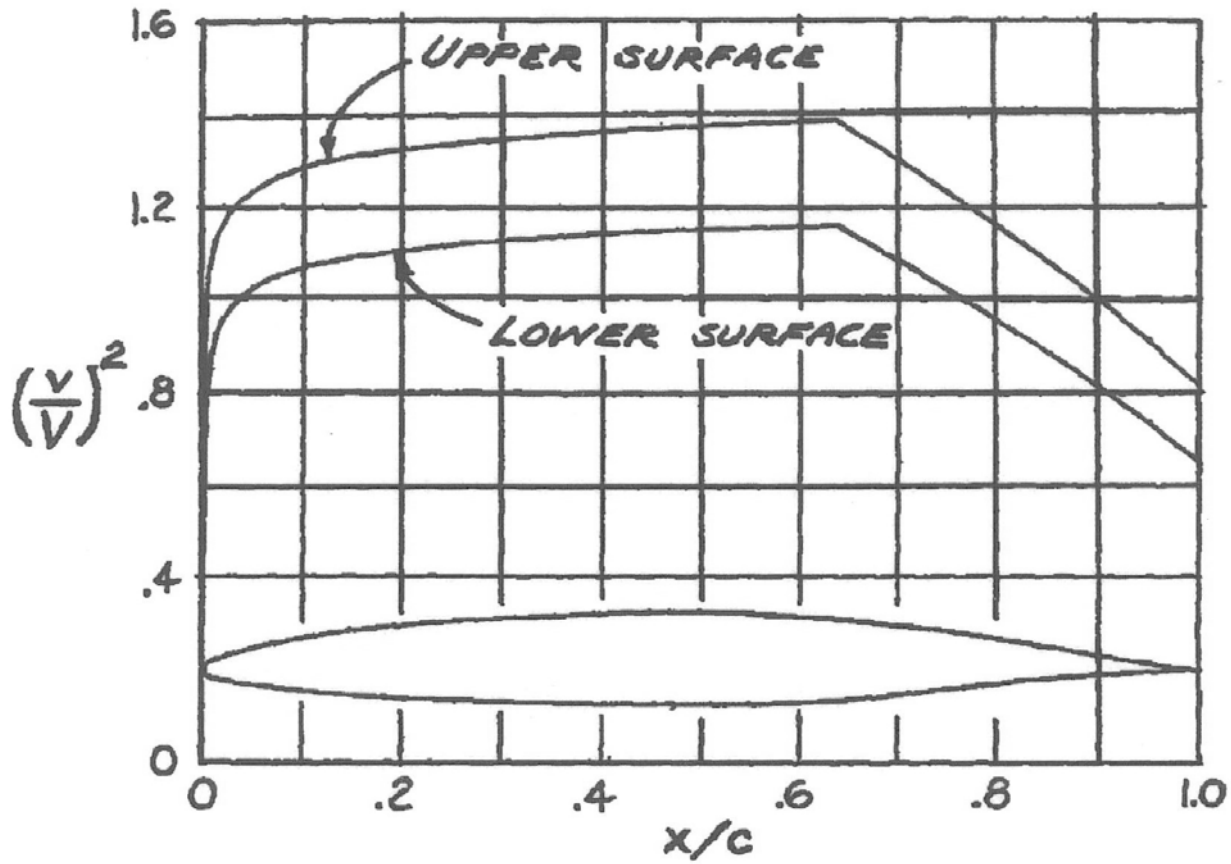
The P-38, and X-1 reveal transonic control problems/solutions

Airfoil Example: Transonic Characteristics

- From classical 6 series results

NACA TN No. 1396 Fig. 1

Subsonic design pressures



NACA 66-210 a = 1.0

Lift

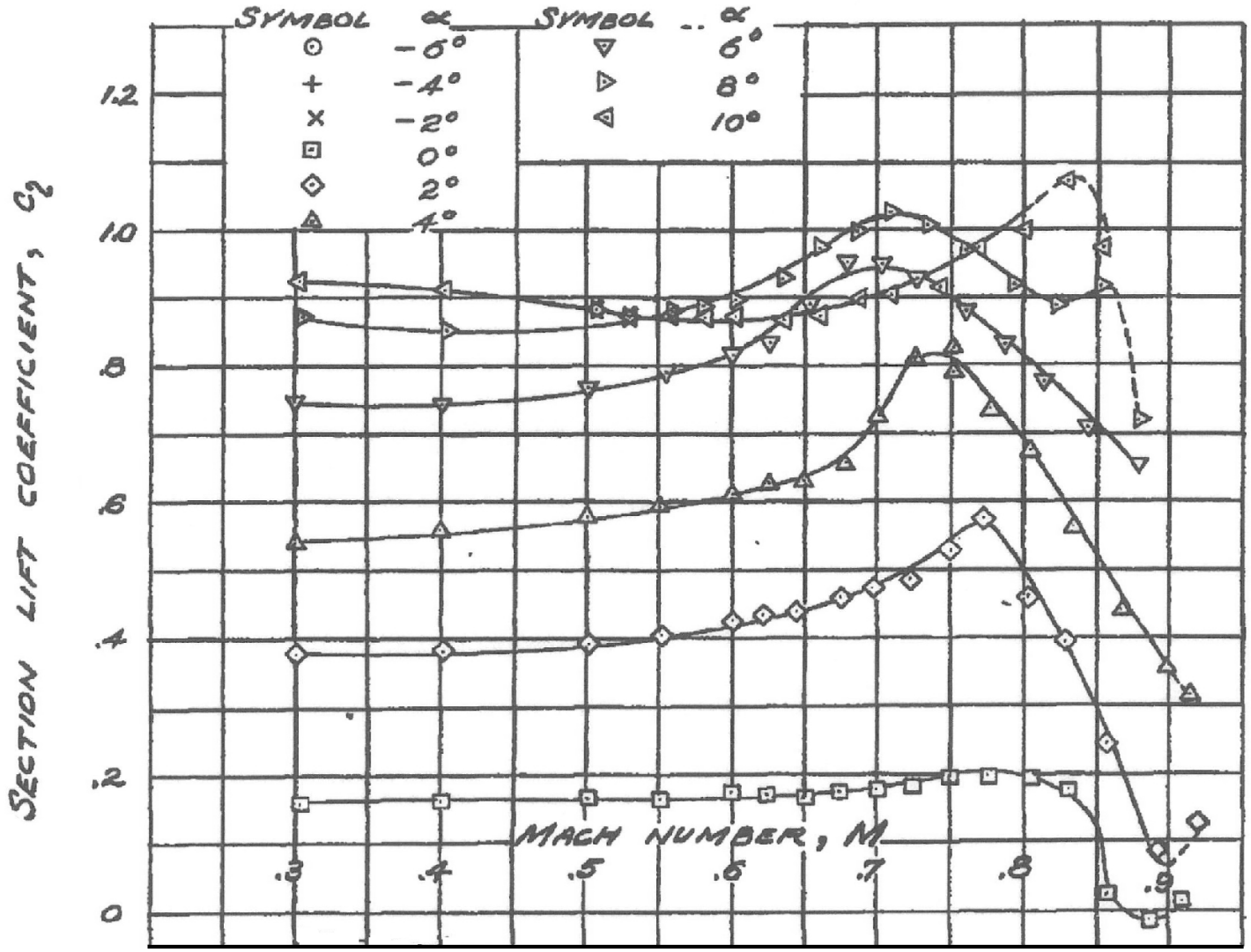


FIGURE 6:- THE VARIATION OF SECTION LIFT COEFFICIENT WITH MACH NUMBER AT VARIOUS ANGLES OF ATTACK FOR THE NACA 66-210 AIRFOIL.

From NACA TN 1396, by Donald Graham, Aug. 1947

Drag

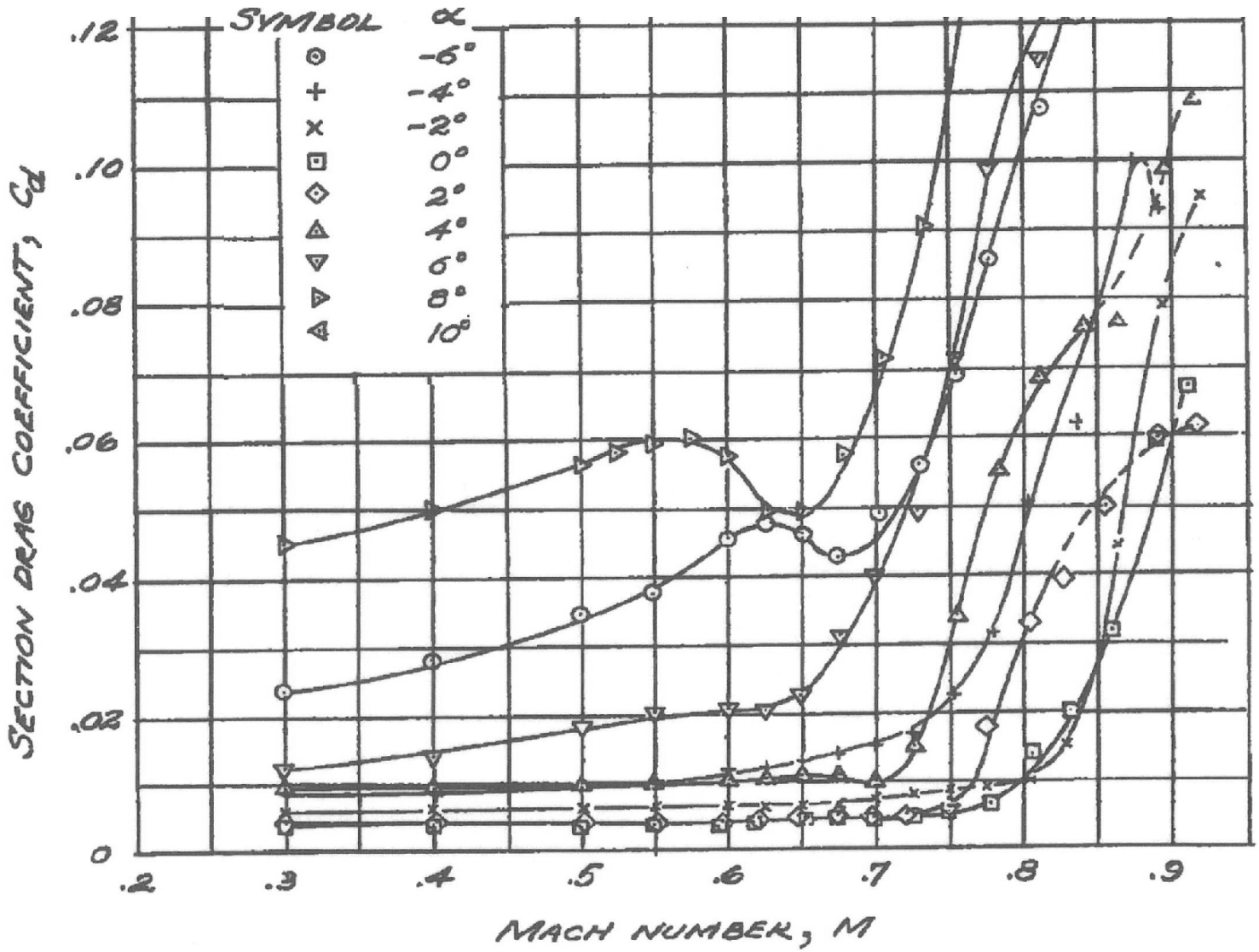


FIGURE 7.- THE VARIATION OF SECTION DRAG COEFFICIENT WITH MACH NUMBER AT VARIOUS ANGLES OF ATTACK FOR THE NACA 66-210 AIRFOIL.

From NACA TN 1396, by Donald Graham, Aug. 1947

Pitching Moment: a major problem!

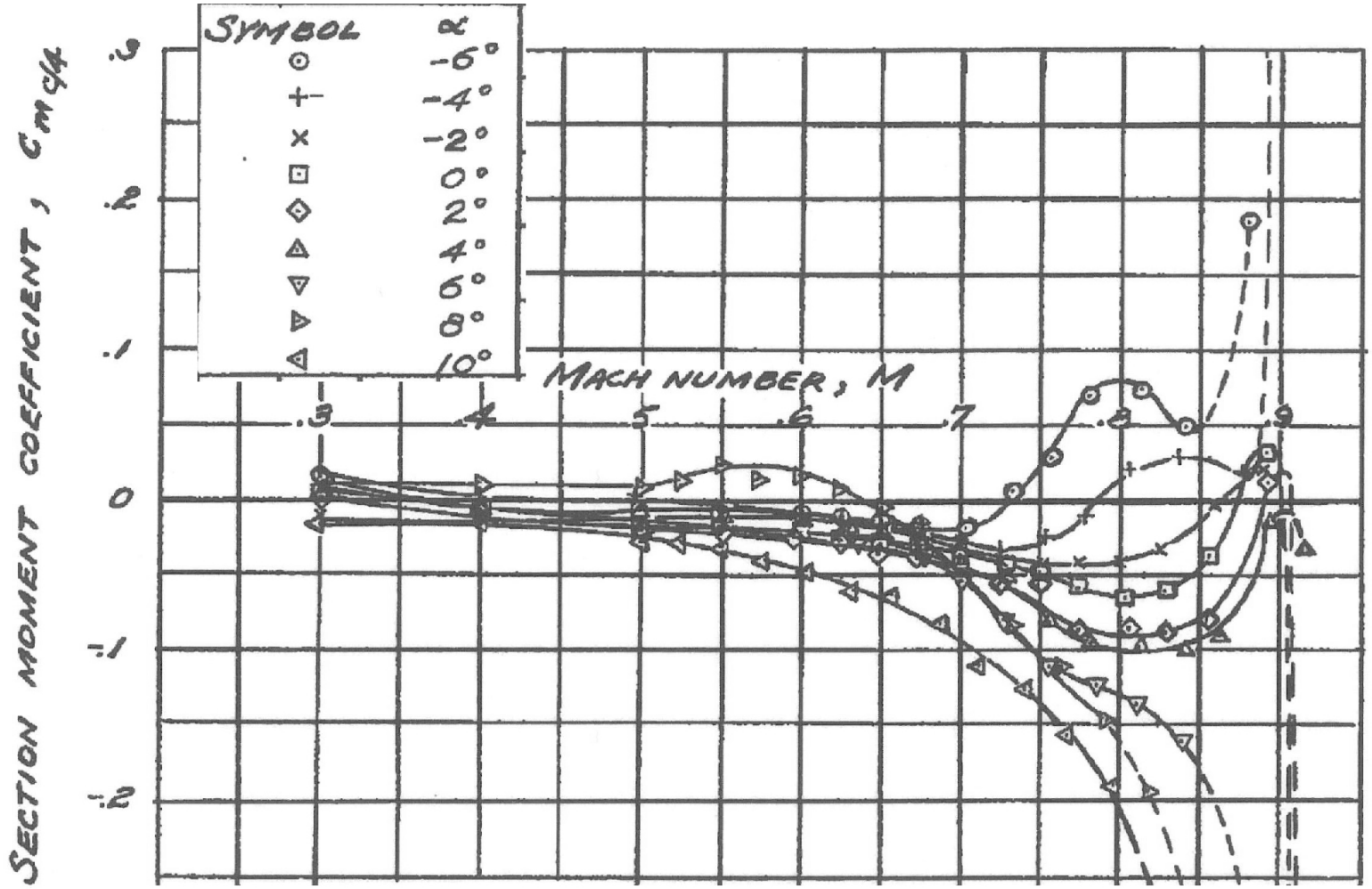
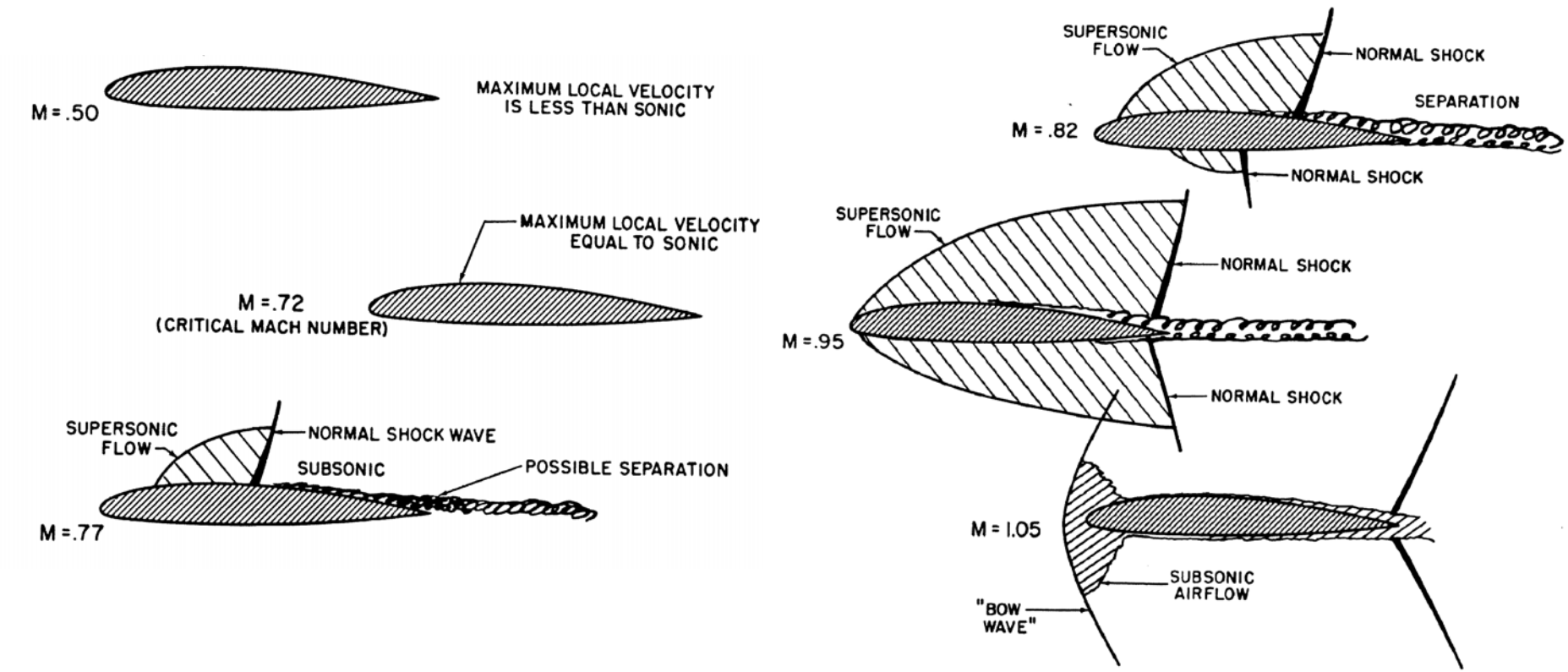


FIGURE 8.- THE VARIATION OF SECTION MOMENT COEFFICIENT WITH MACH NUMBER AT VARIOUS ANGLES OF ATTACK FOR THE NACA 66-210 AIRFOIL.

From NACA TN 1396, by Donald Graham, Aug. 1947

What's going on?
The flow development illustration



From *Aerodynamics for Naval Aviators* by Hurt

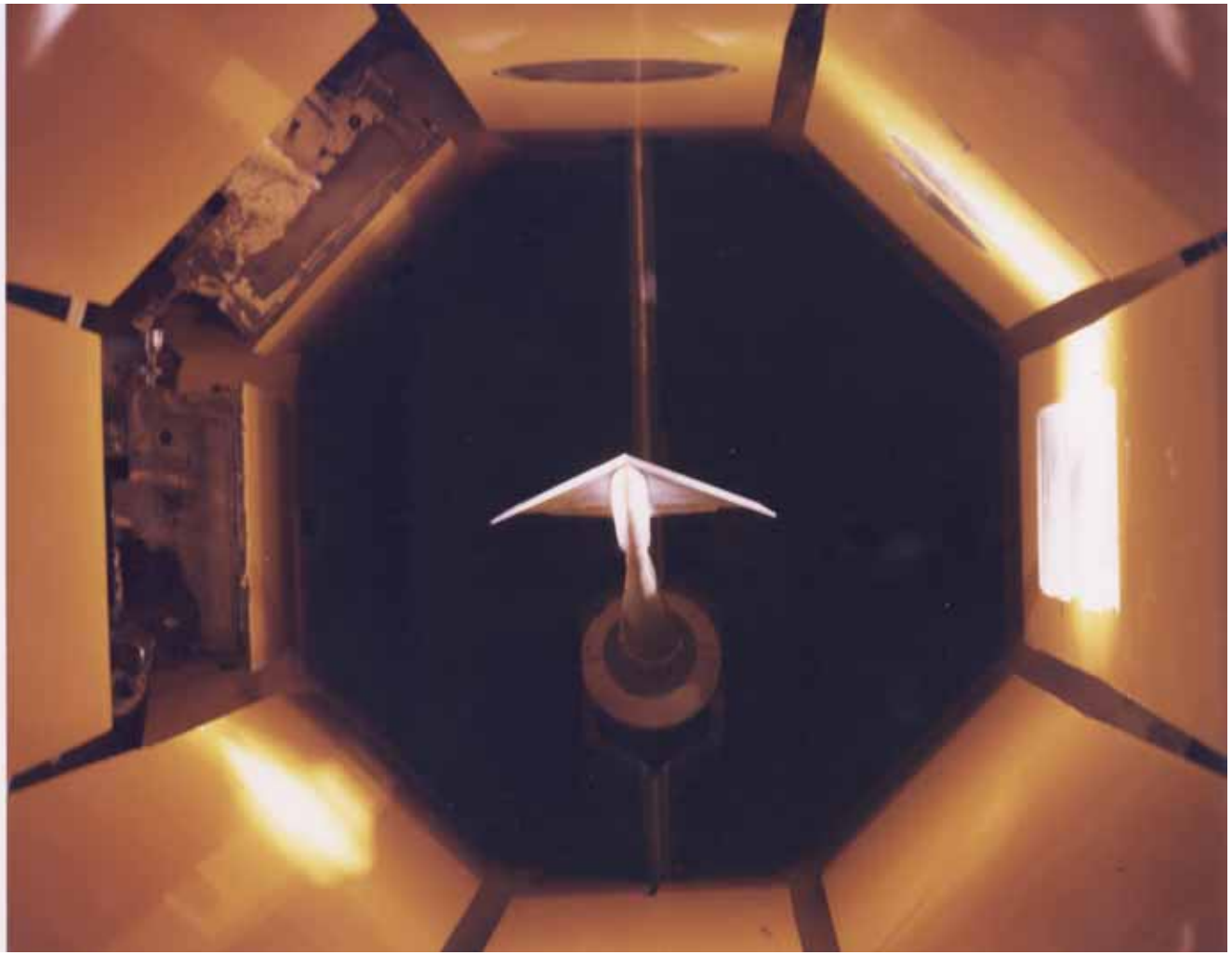
Addressing the Testing Problem

- 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

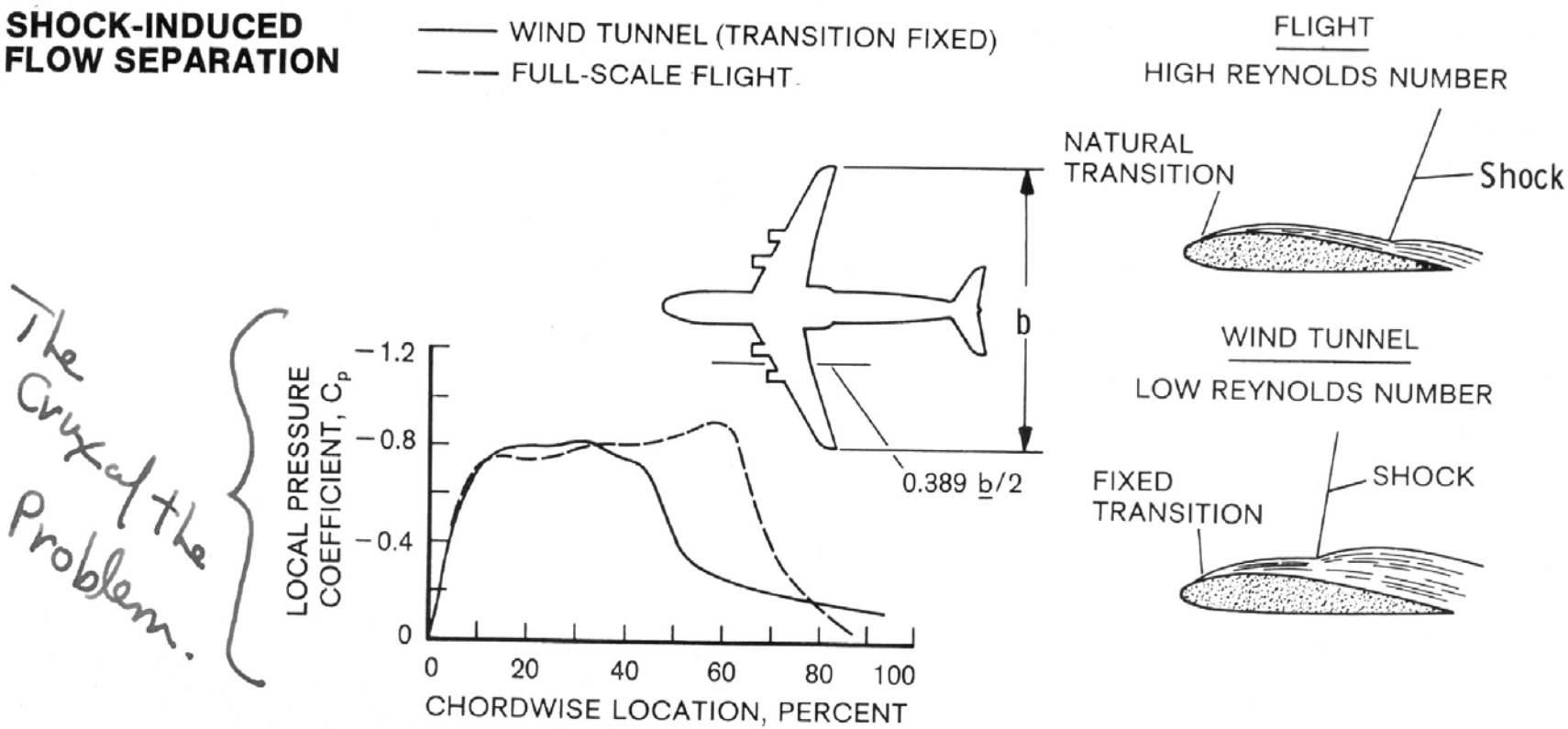
Wall Interference Solution 2: Porous Wall

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



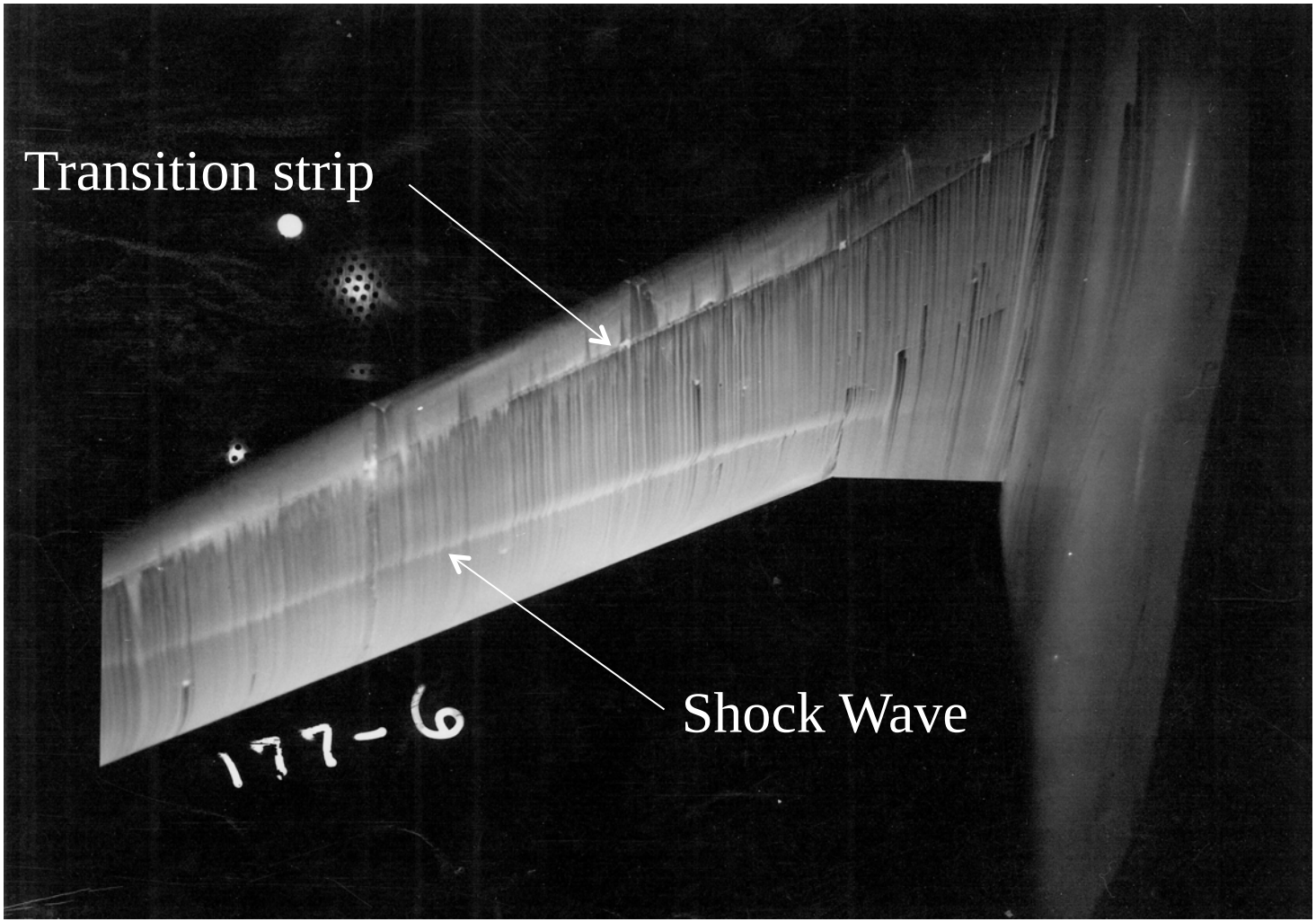
"The Need for developing a High Reynolds Number Transonic WT"
Astronautics and Aeronautics, April 1971, pp. 65-70

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?

$$Re = \frac{\rho V L}{\mu}$$

ρ: density, V: velocity, L: length, μ: viscosity,

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

$$\mathrm{Re} = \frac{\sqrt{\gamma}\, pML}{\sqrt{R}\, T^{1.4}}$$

Increase *Re* by increasing *p* or *L*, decreasing *T* or changing the gas

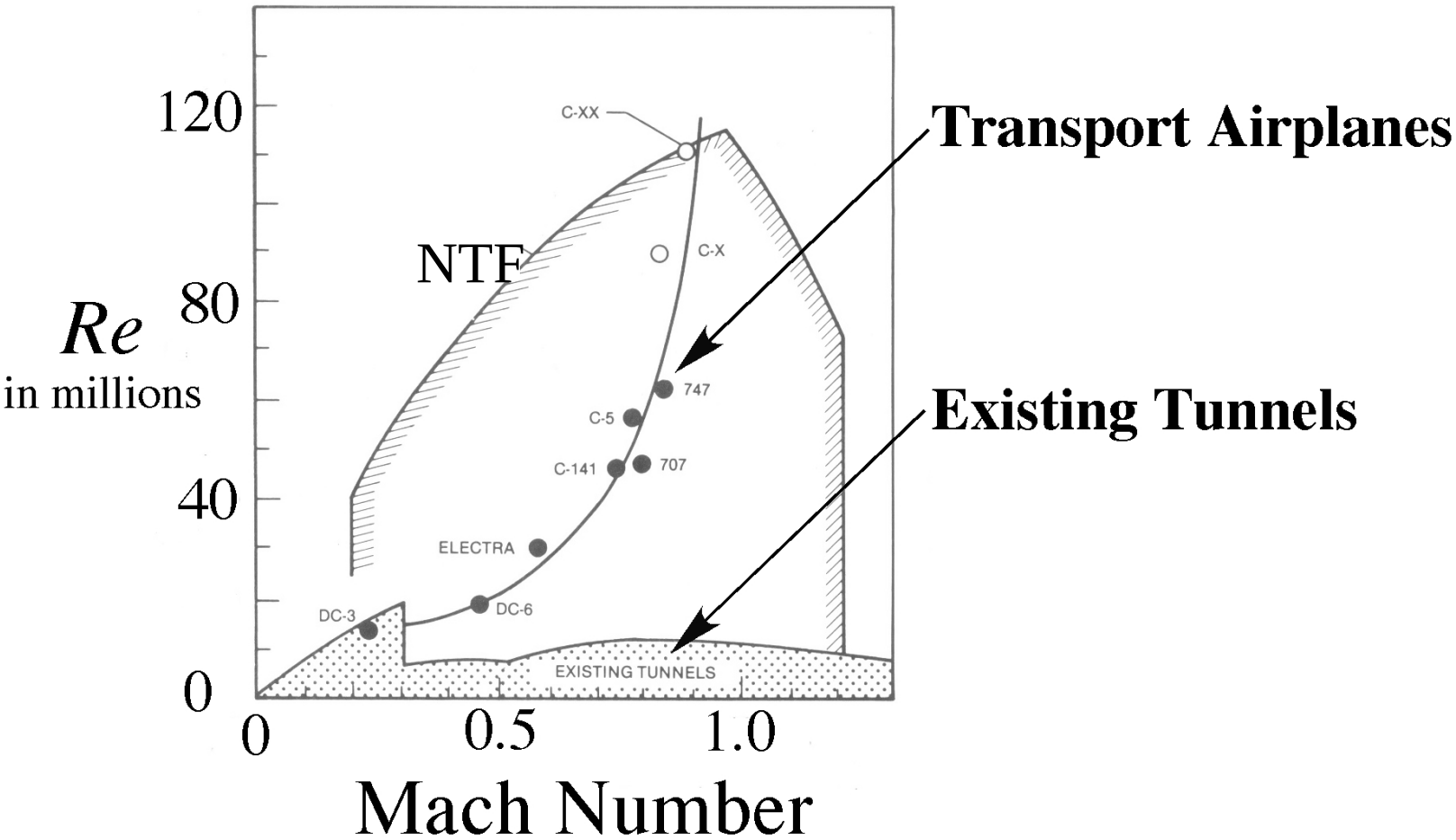
Balance forces are related to, say, $N = qSC_L$

$$q = \frac{\gamma}{2} pM^2$$

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

Feb. 1982

Performance: $M = 0.2$ to 1.20

$P_T = 1$ to 9 atm

$T_T = 77°$ to $350°$ Kelvin

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

$$\underbrace{\left[1 - M_\infty^2 - \left(\gamma + 1\right) M_\infty^2 \phi_x\right]}_{\substack{>0 \text{ locally subsonic, elliptic PDE} \\ <0 \text{ locally supersonic, hyperbolic PDE}}} \phi_{xx} + \phi_{yy} = 0$$

- Model problem assumes flow is primarily in the *x*-direction
- Nonlinearity allows math type to change locally

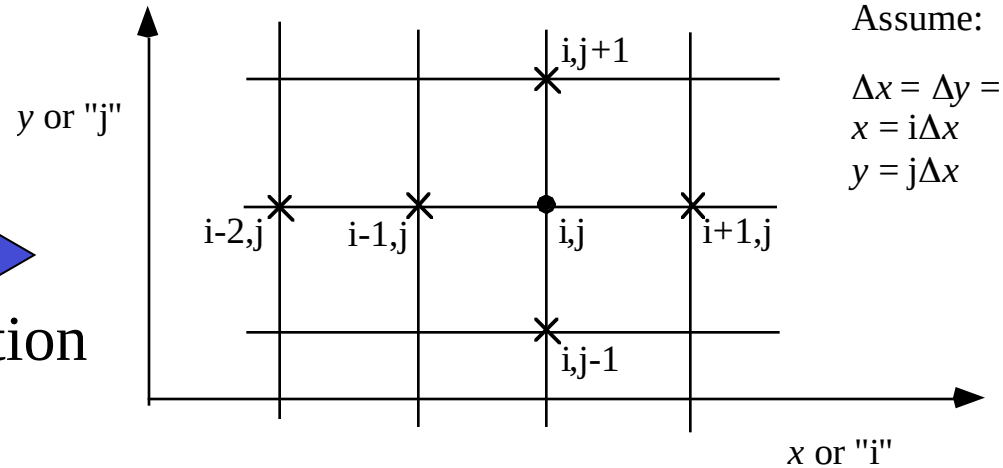
The Murman-Cole Idea

- 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 ϕ_{xx} with a central difference
  - Is locally supersonic, represent ϕ_{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)

A few more details

Using this notation for the grid:

Flow Direction



Assume:

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

For subsonic flow:

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

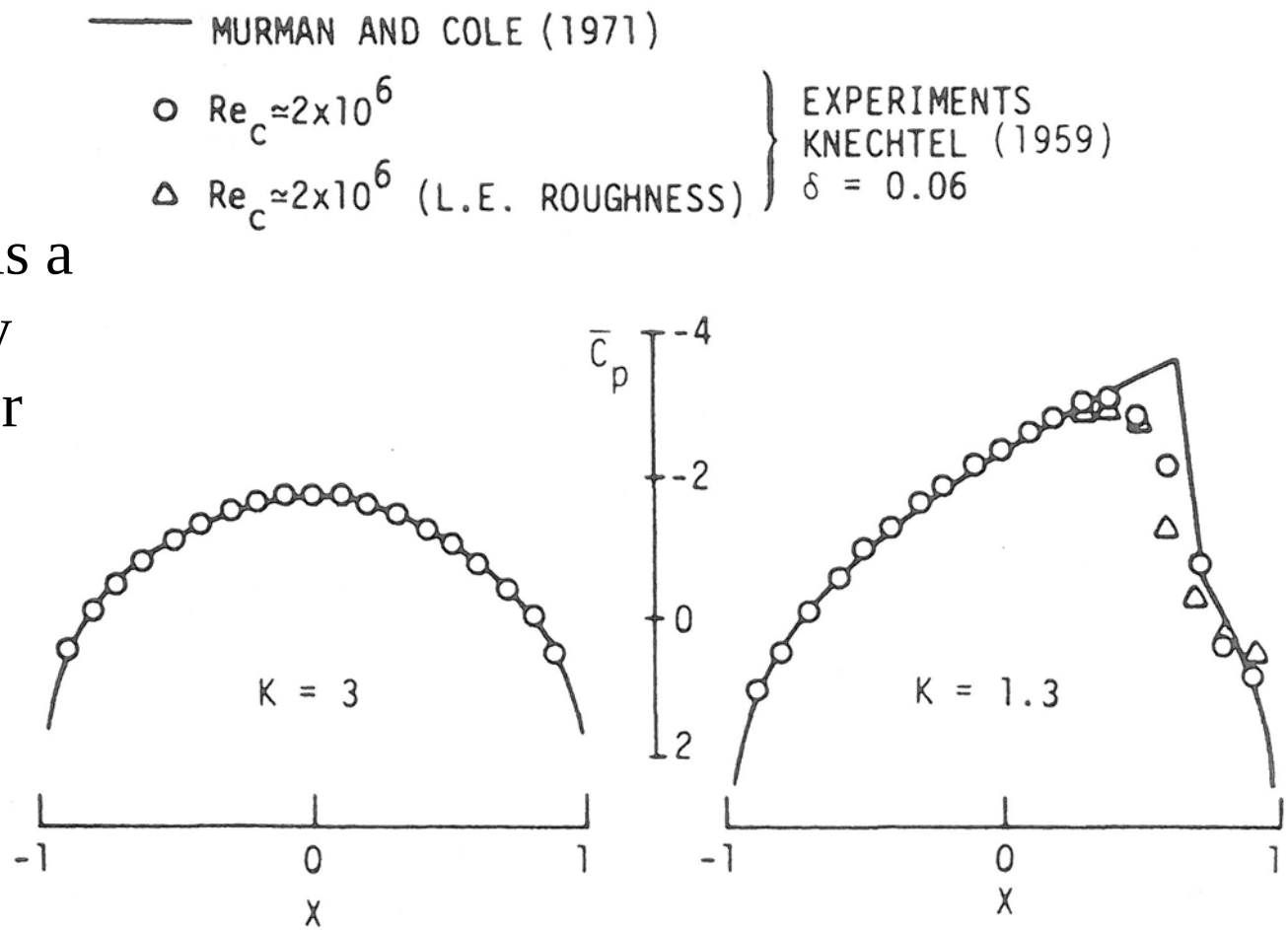
For supersonic flow:

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

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

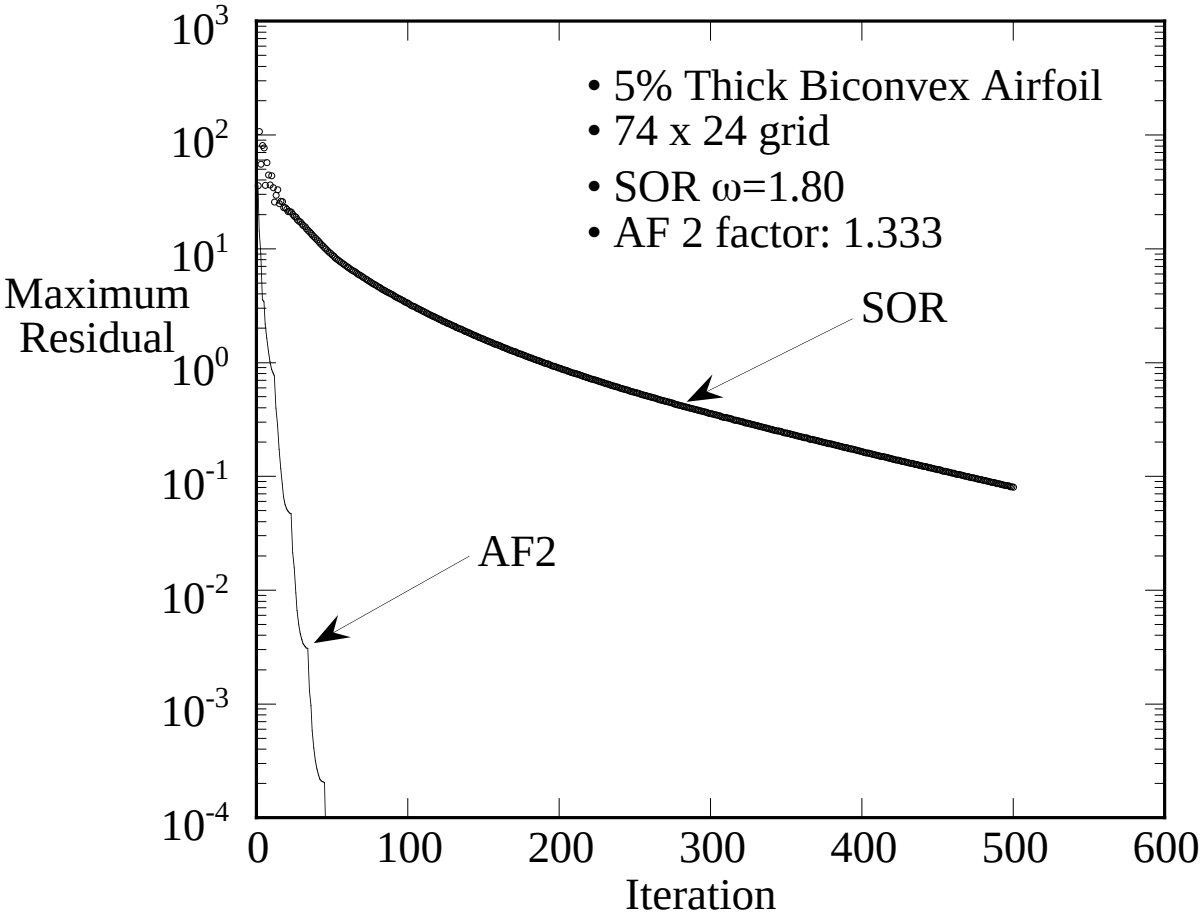


subcritical case supercritical case

from *Computational Fluid Mechanics and Heat Transfer*, 2nd Ed., by Tannehill, Anderson and Pletcher, Taylor and Francis, 1997.

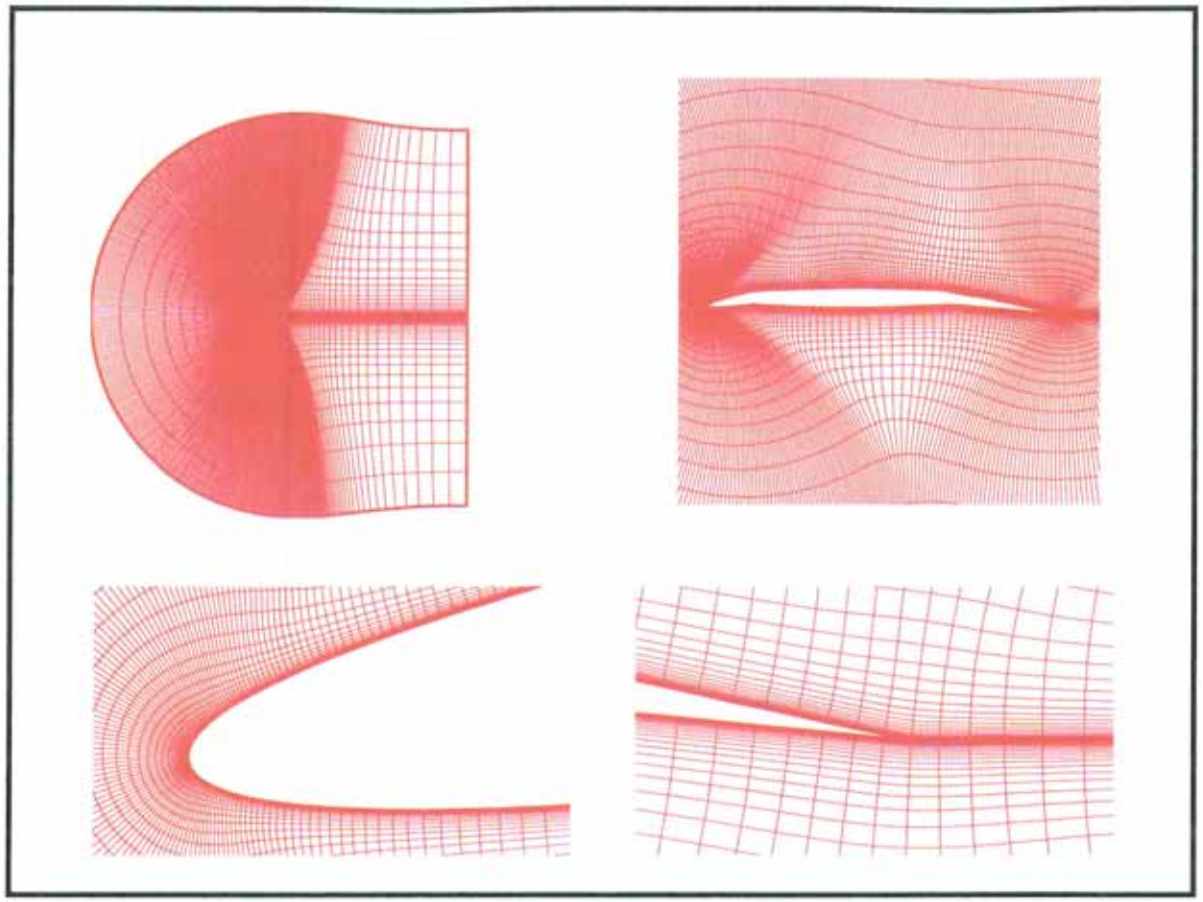
One convergence check in CFD

- "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



CFD Grids

Most of the work applying CFD is grid generation



Michael Henry, “Two-Dimensional Shock Sensitivity Analysis for Transonic Airfoils with Leading-Edge and Trailing-Edge Device Deflections" MS Thesis, Virginia Tech, August 2001, W.H. Mason, Adviser

Today

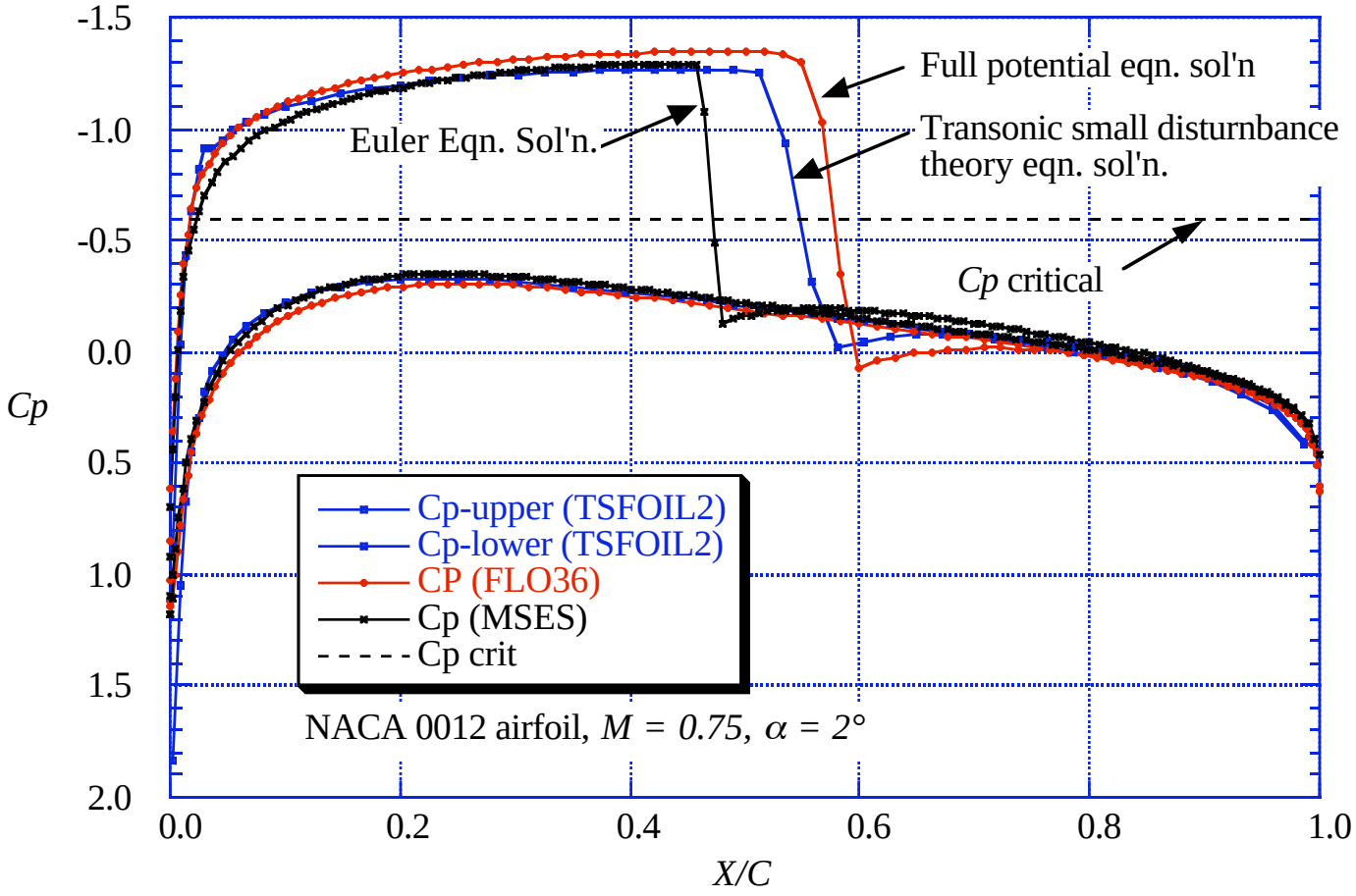
- CFD has been developed *way* past this snapshot
- For steady solutions we iterate *in time*, until a steady state value is obtained
  - the problem is always hyperbolic in time!
- Key issues you need to understand
  - Implementing boundary conditions
  - Solution stability
  - Discretization error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

One reference, *Computational Fluid Mechanics and Heat Transfer*, 2nd Ed., by Tannehill, Anderson and Pletcher Taylor and Francis, 1997.

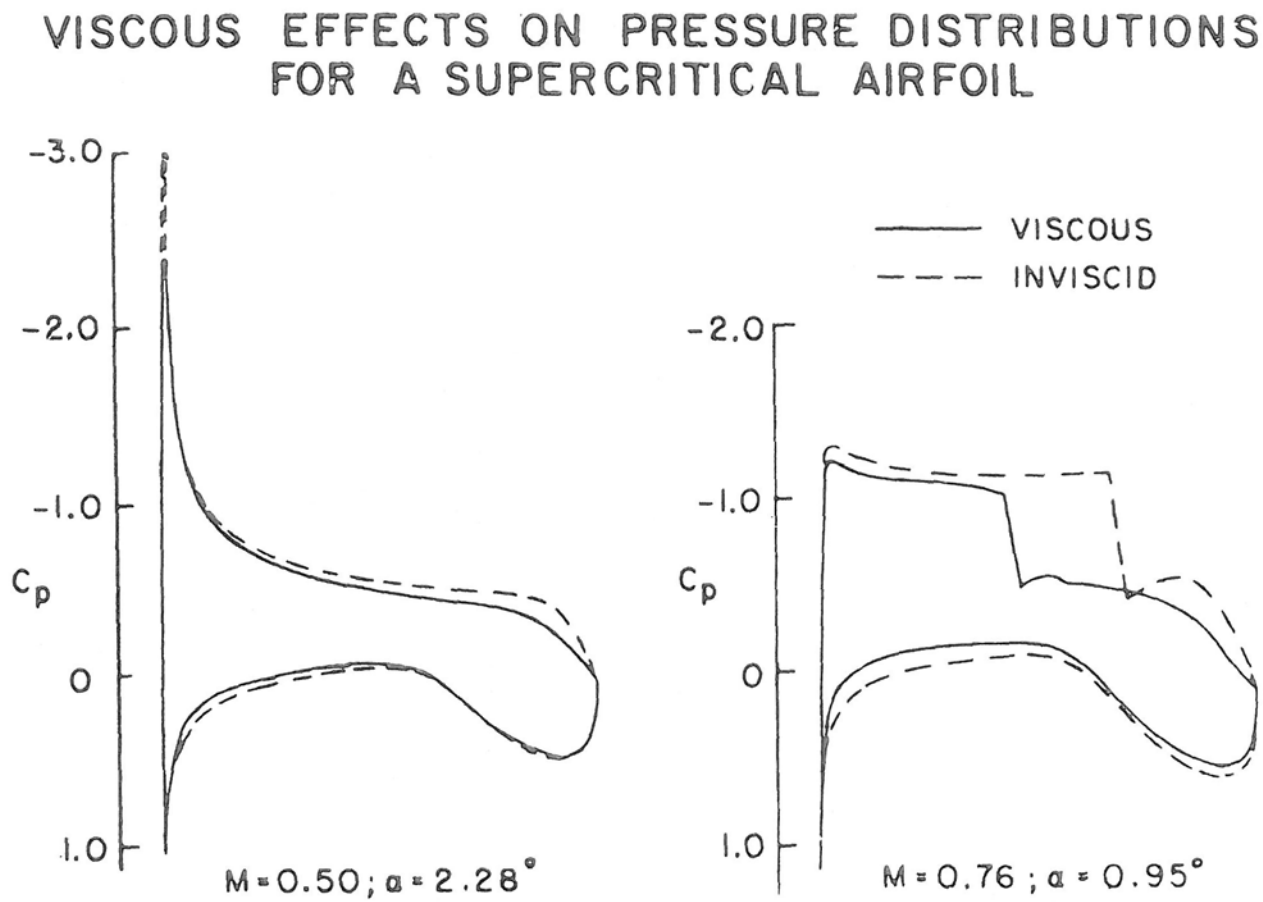
Can now use computational simulations as a numerical wind tunnel

- 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?
- **TSFOIL**2 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 -



An extra complication: Viscous effects are more important at Transonic Speeds



Whitcomb, "Review of NASA Supercritical Airfoils," ICAS 74-10, Aug. 1974

(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:

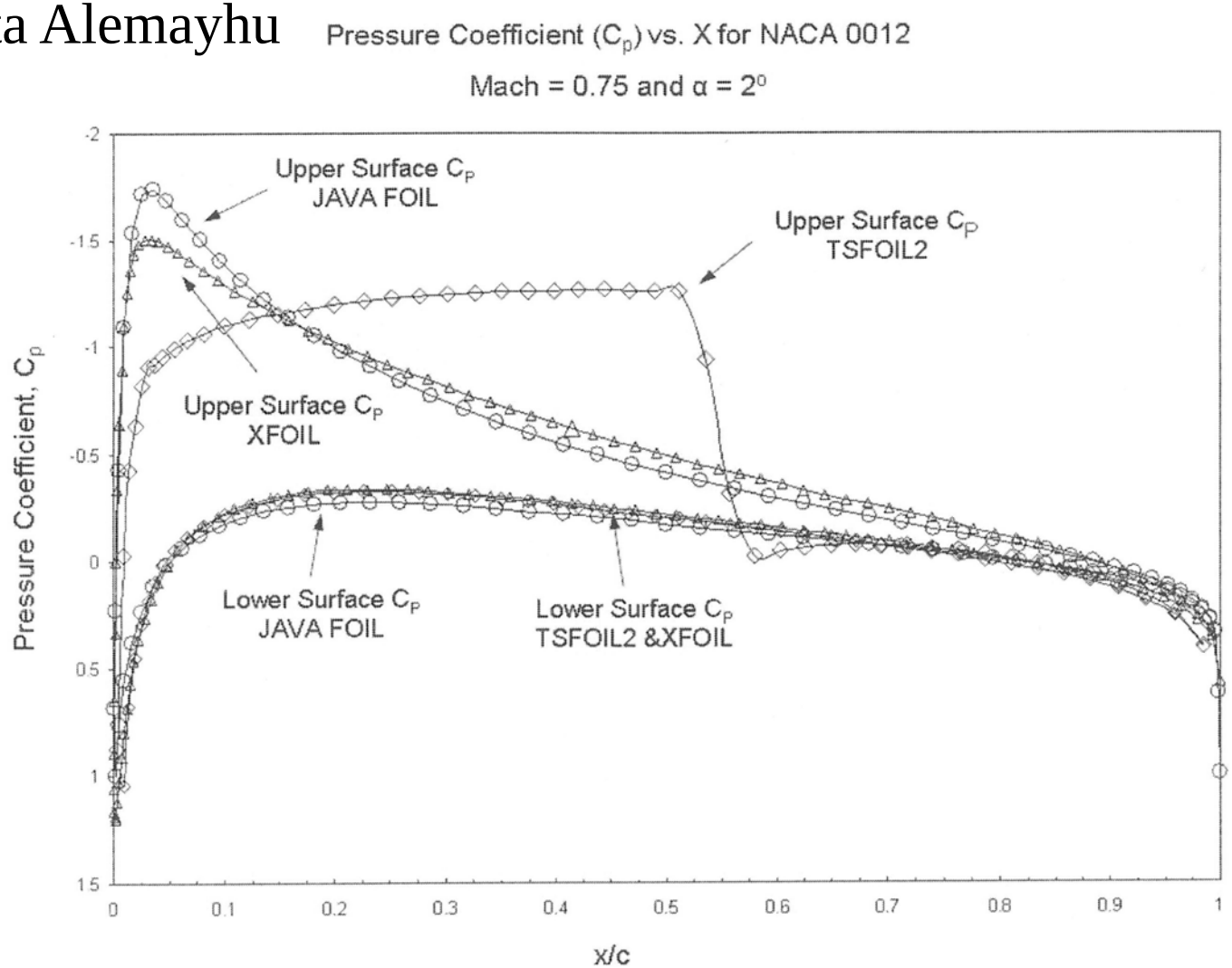
- Discussion of transonic airfoil characteristics and design goals

Primarily associated with
Richard Whitcomb,
Feb 21, 1921 – Oct. 13, 2009

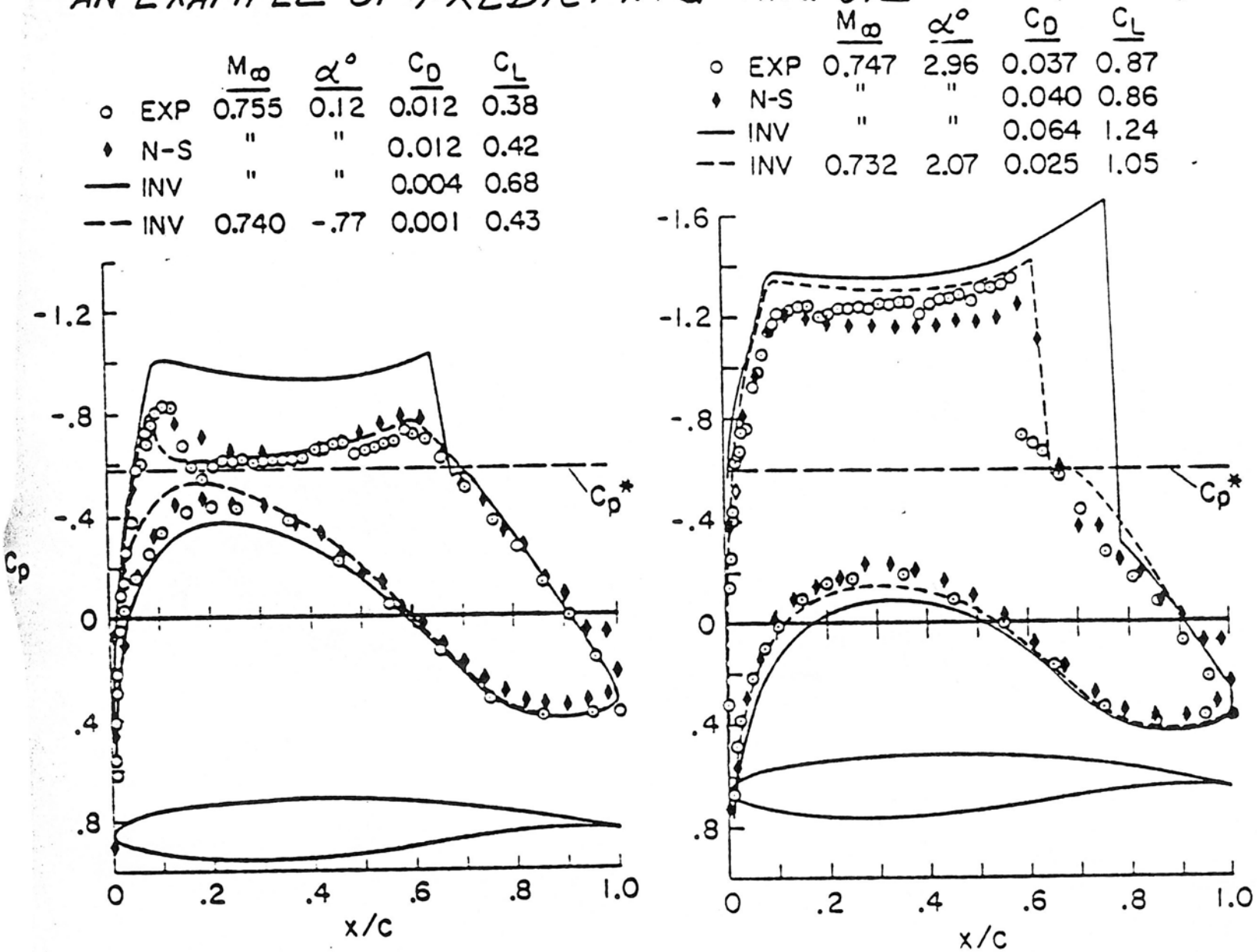
Richard Whitcomb with Supercritical Wing Model
NASA Langley Research Center 1/19/1970 Image # EL-2001-00478

Subsonic Linear Theory Can't Predict Transonic Flow!

From Desta Alemayhu



AN EXAMPLE OF PREDICTING AIRFOIL FLOWS (REF 3)

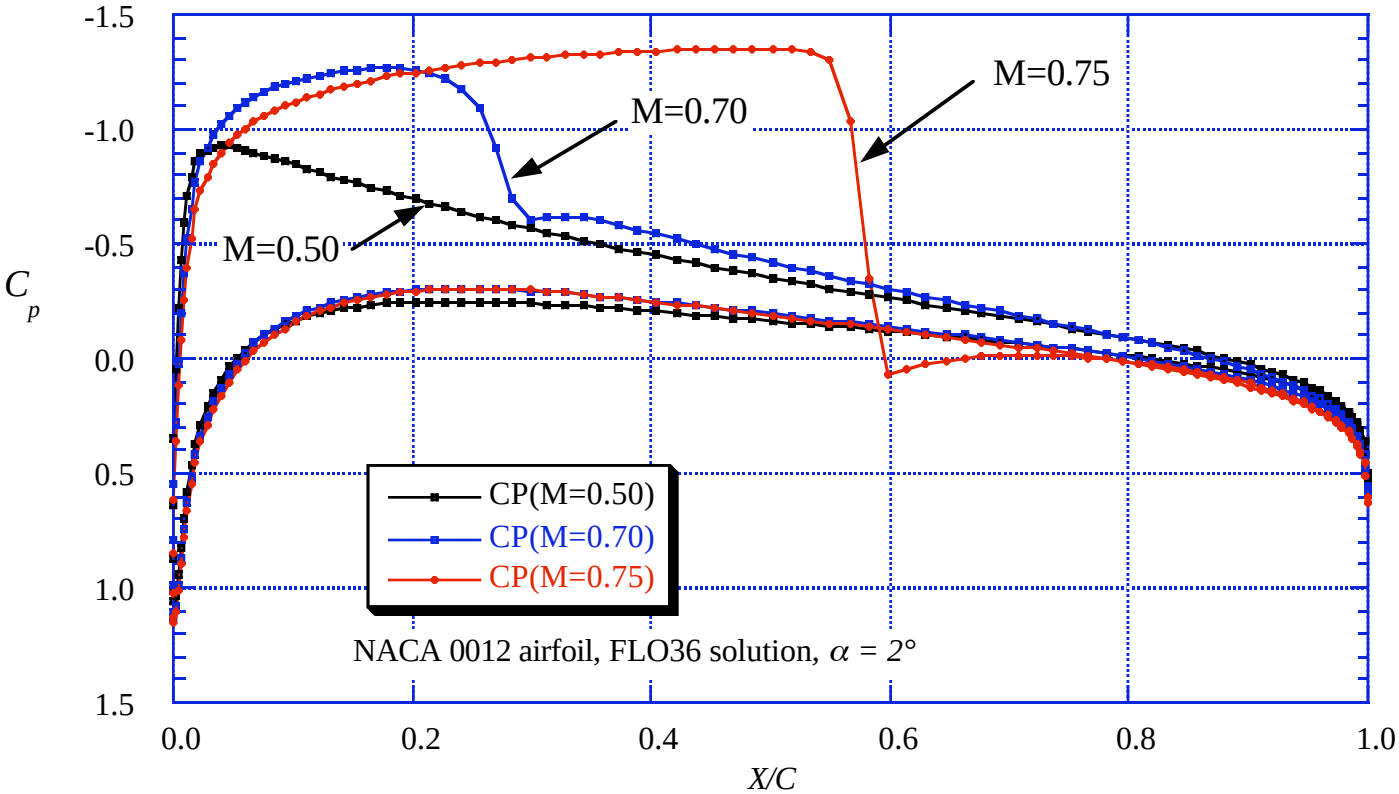


Illustrates “tricks” used to get calculations to agree with test data

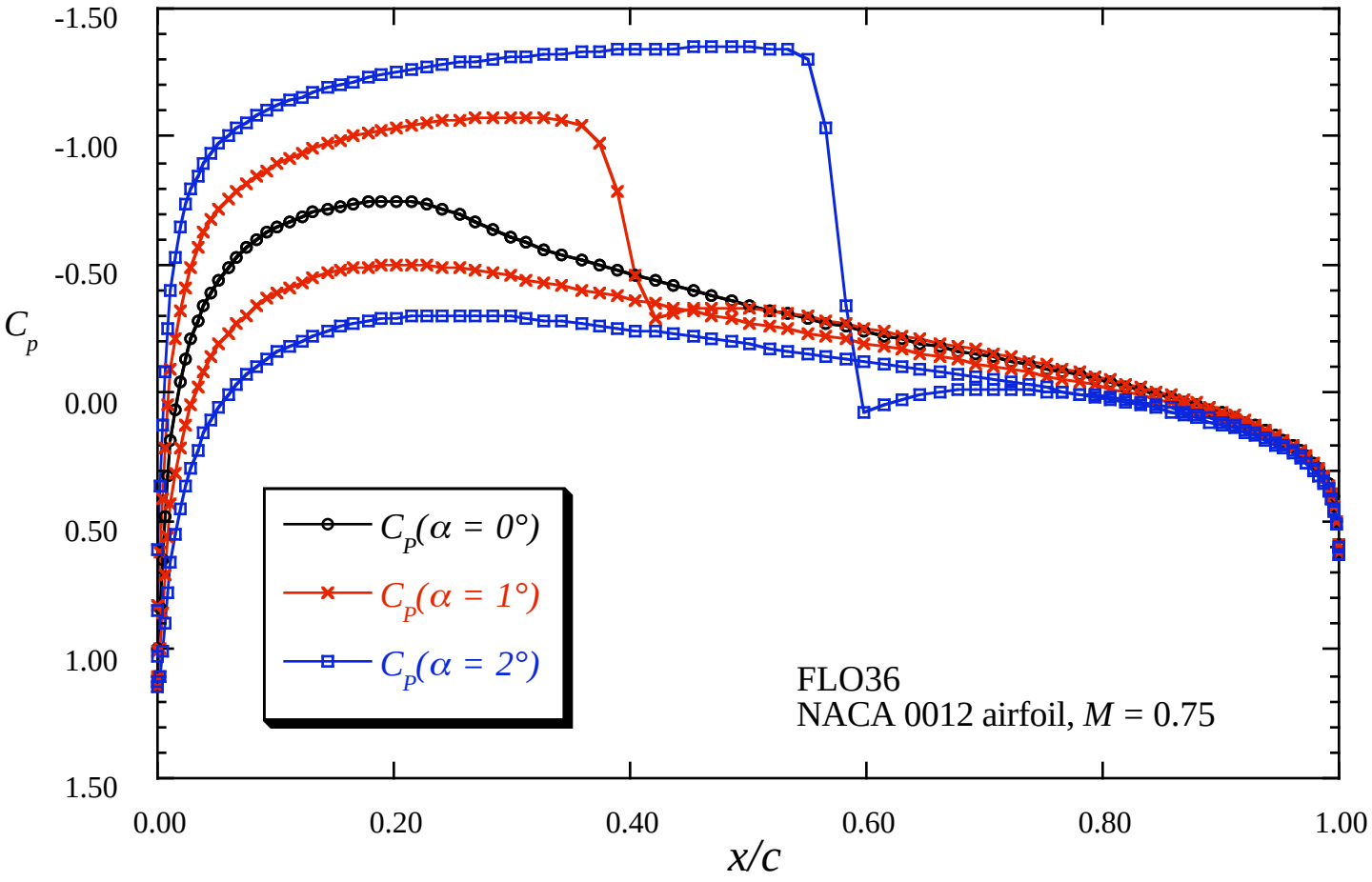
REVIEW: Obtaining CFD solutions

- 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

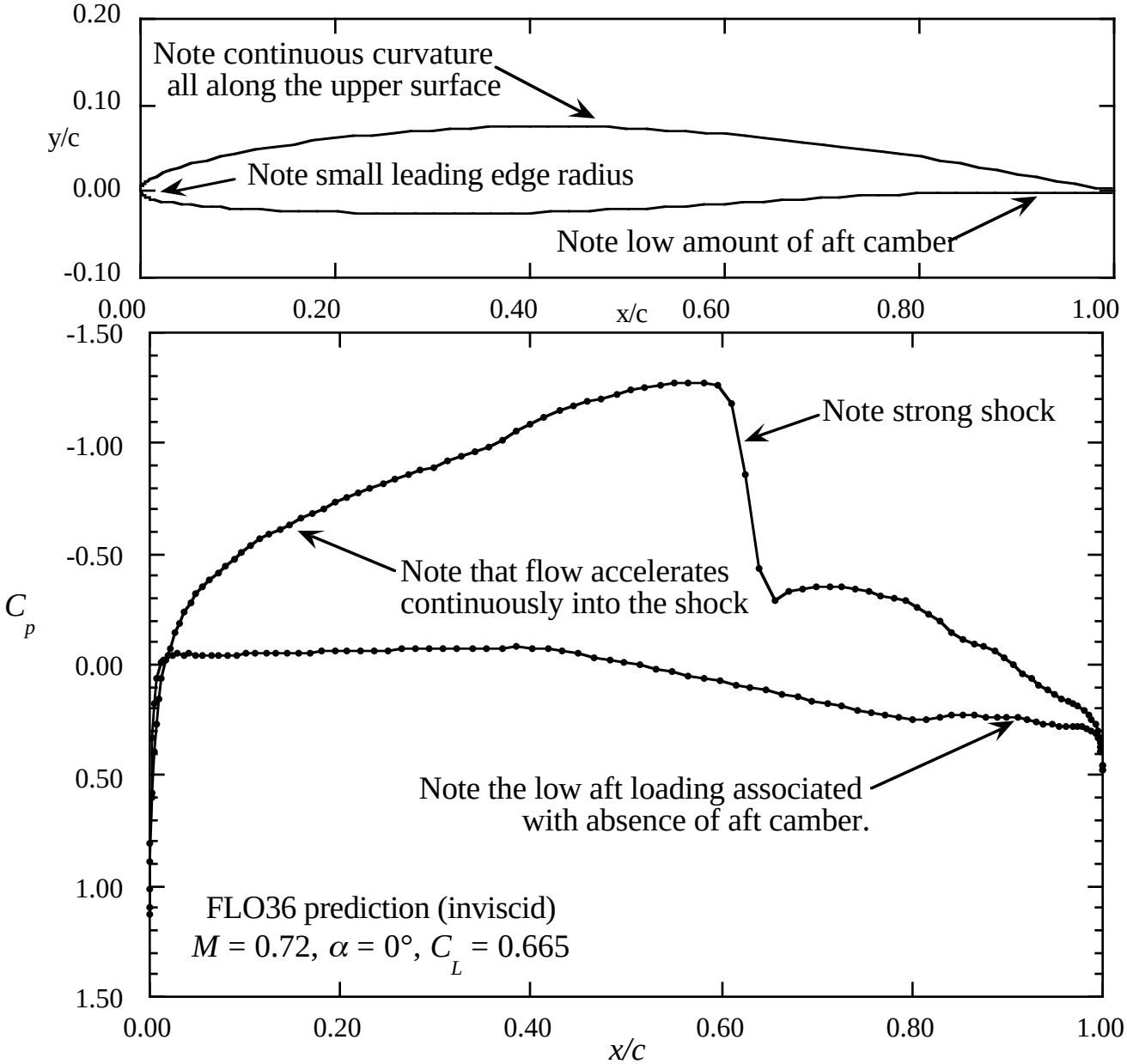
Airfoils
Mach number effects: NACA 0012



Angle of attack effects: NACA 0012



“Traditional” NACA 6-series airfoil



A “new” airfoil concept - from Whitcomb

1964

Slotted (1964)

1966

Integral (1966)

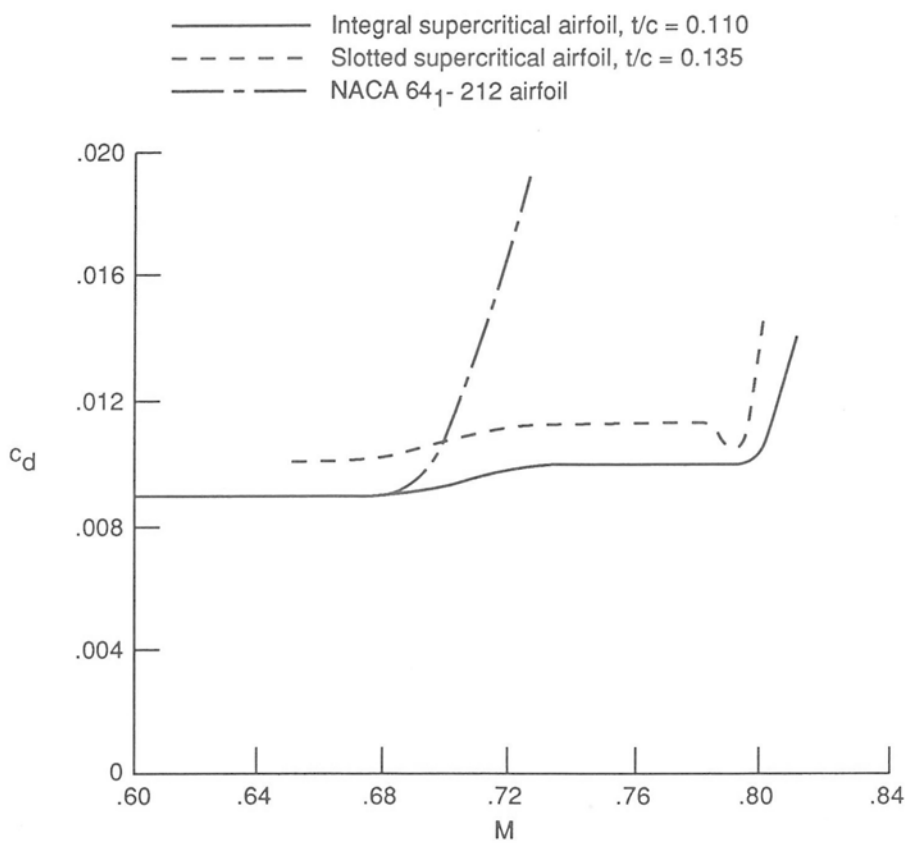
1968

Integral with thickened trailing edge (1968)

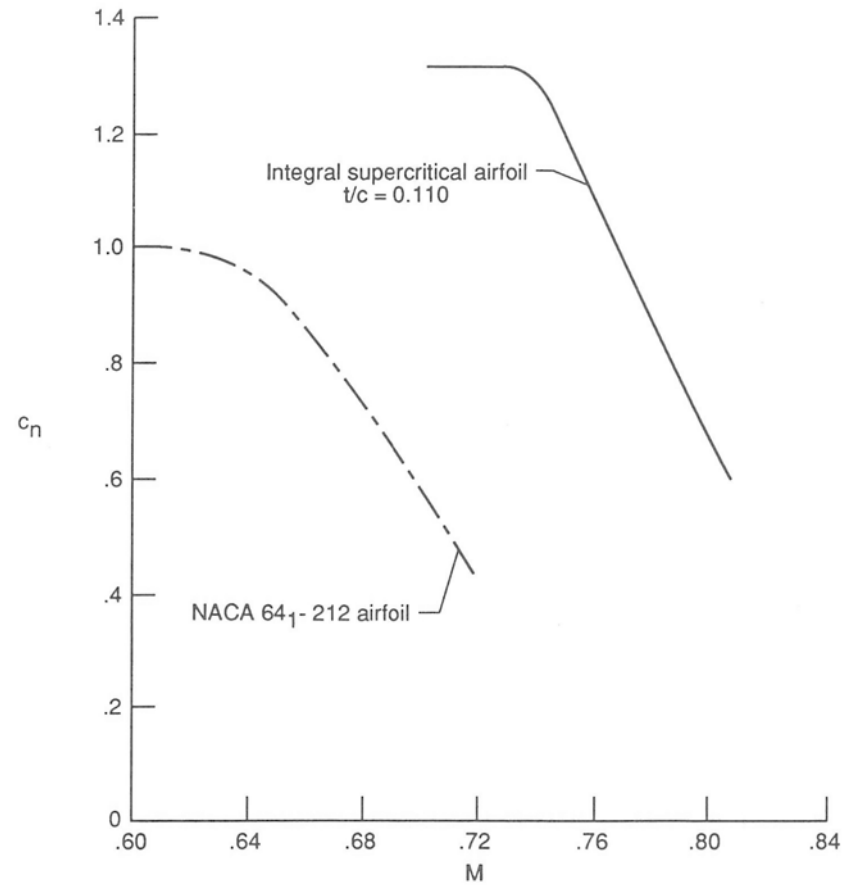
Progression of the Supercritical airfoil shape
“NASA Supercritical Airfoils,” by Charles D. Harris, NASA TP 2969, March 1990

What the supercritical concept achieved

Section drag at $C_N = 0.65$

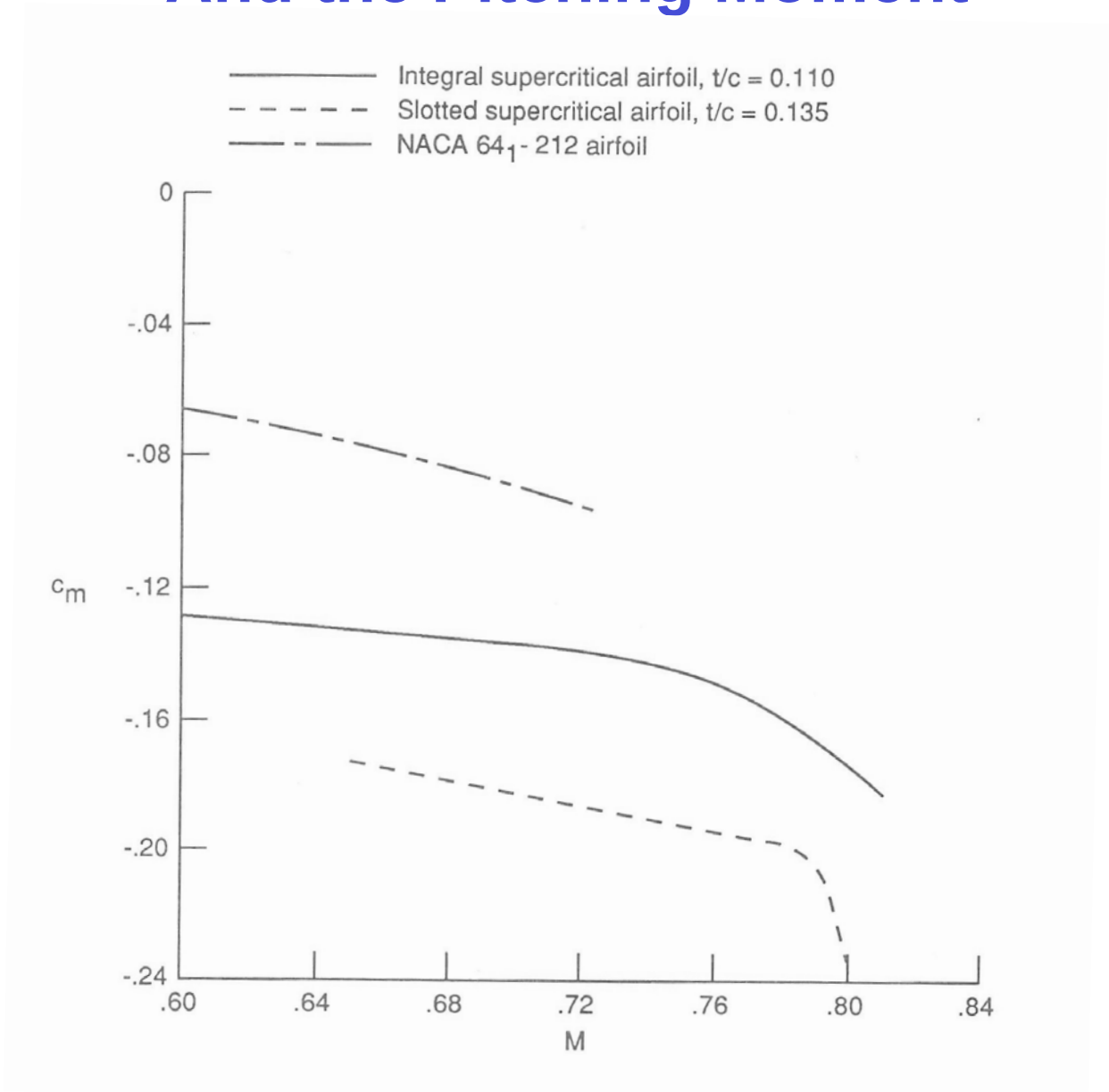


Force limit for onset of upper-surface boundary layer separation



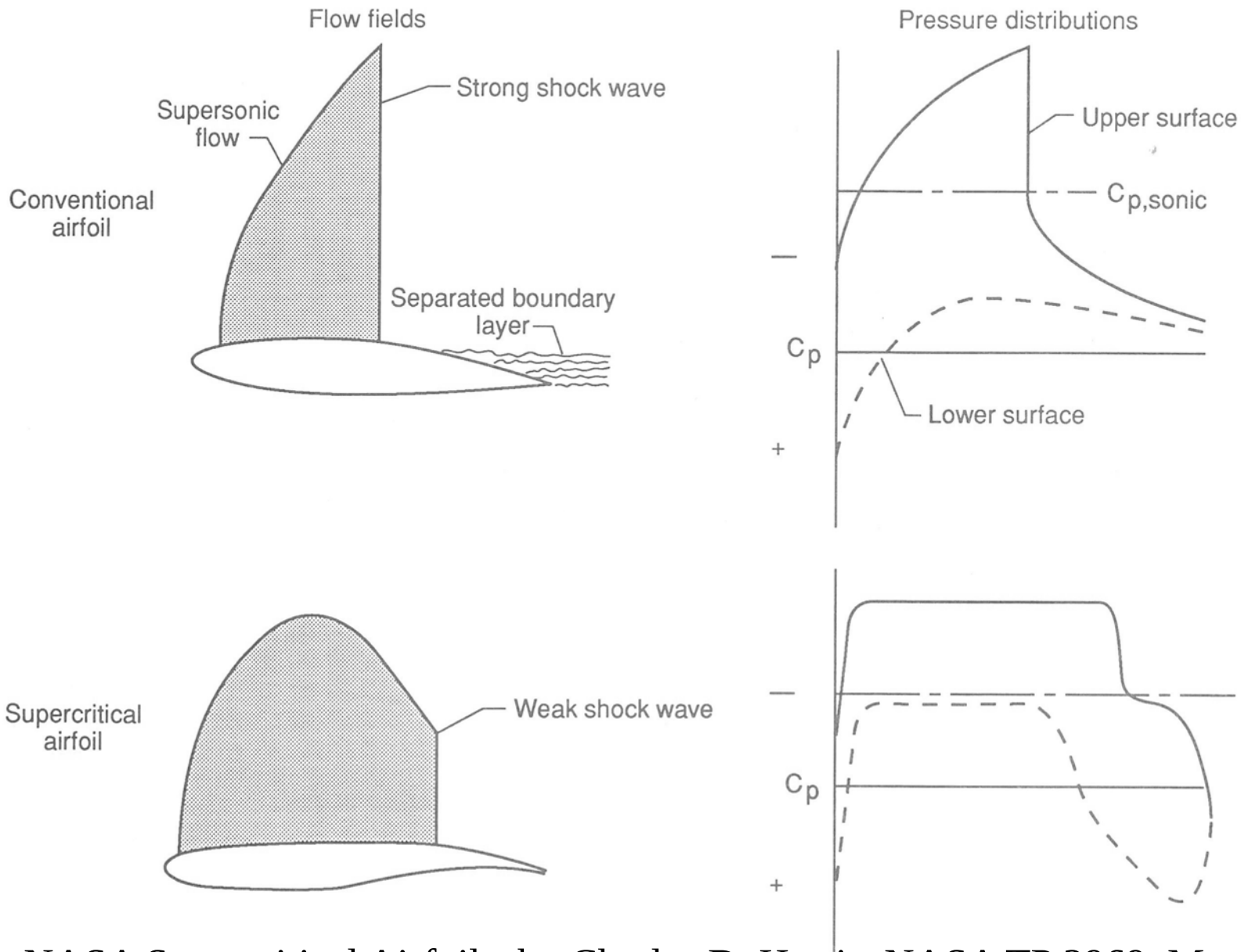
From "NASA Supercritical Airfoils," by Charles D. Harris, NASA TP 2969, March 1990

And the Pitching Moment



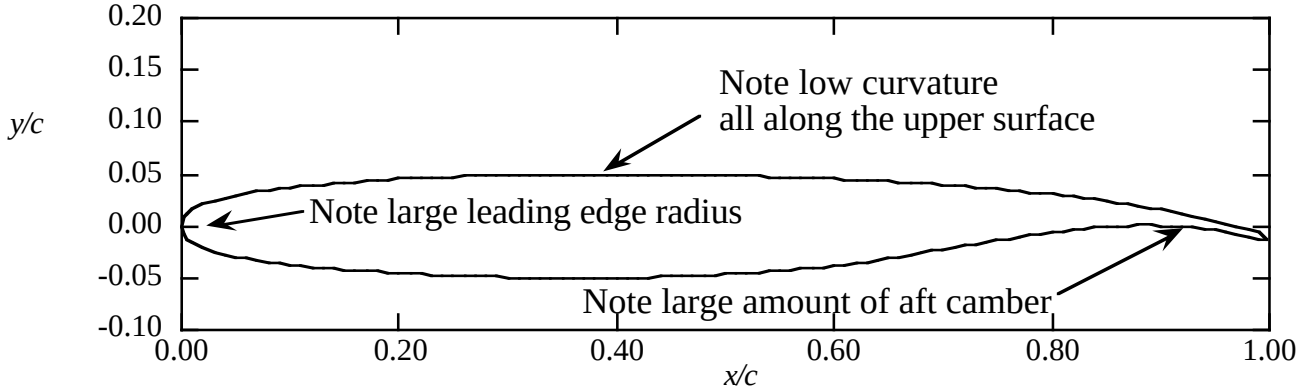
From NASA Supercritical Airfoils, by Charles D. Harris, NASA TP 2969, March 1990

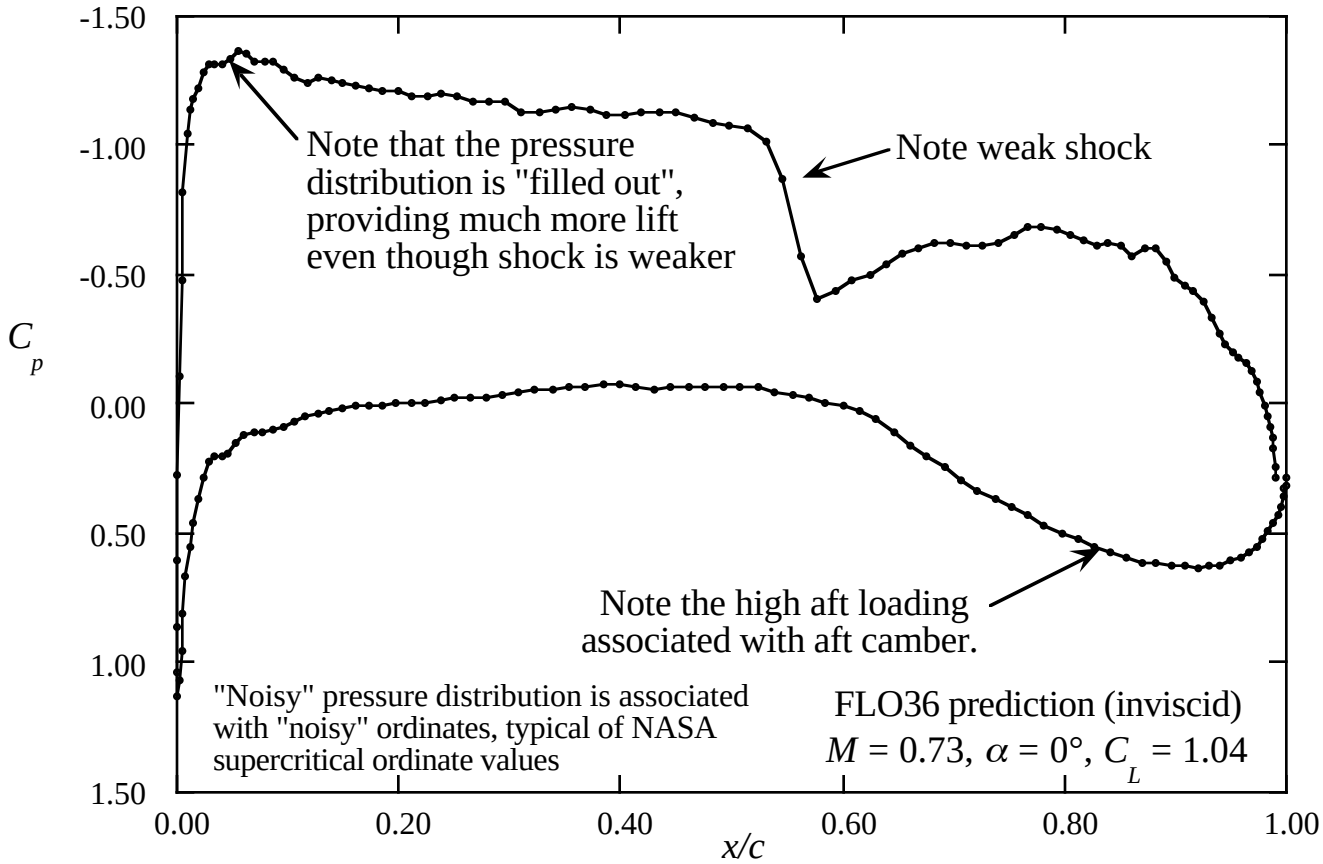
How Supercritical Foils are Different



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

“Supercritical” Airfoils





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

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

Example: Airfoils 31 and 33

10–PERCENT–THICK NASA SUPERCRITICAL AIRFOILS 31 AND 33

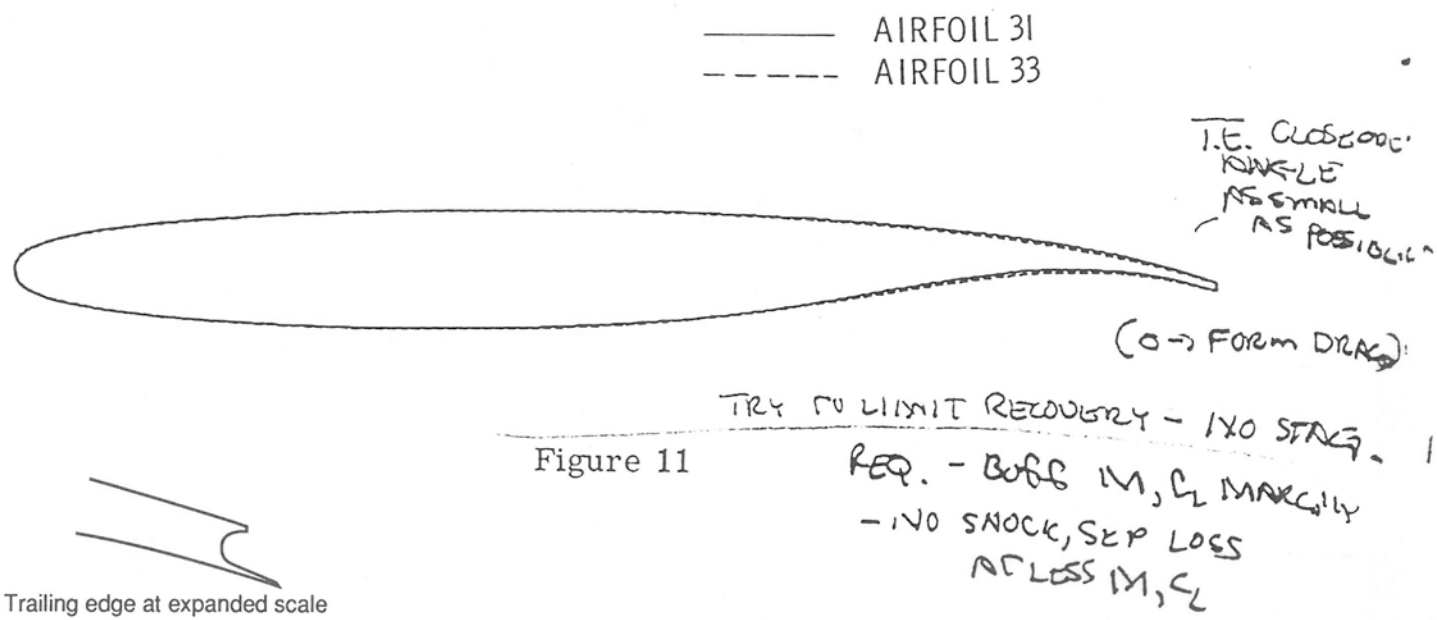
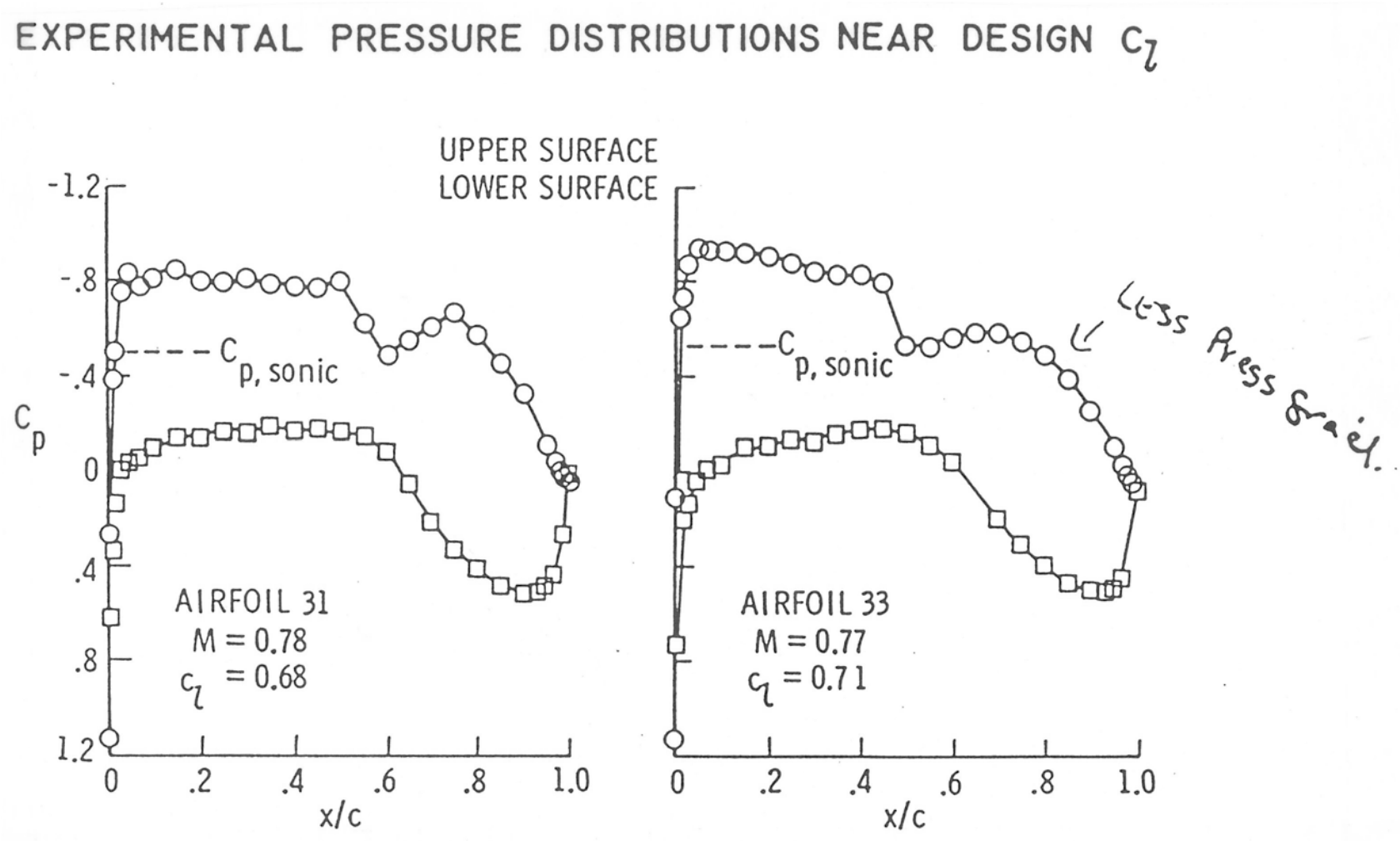


Figure 11

Airfoils 31 and 33



Off Design

EXPERIMENTAL SONIC PLATEAU PRESSURE DISTRIBUTIONS

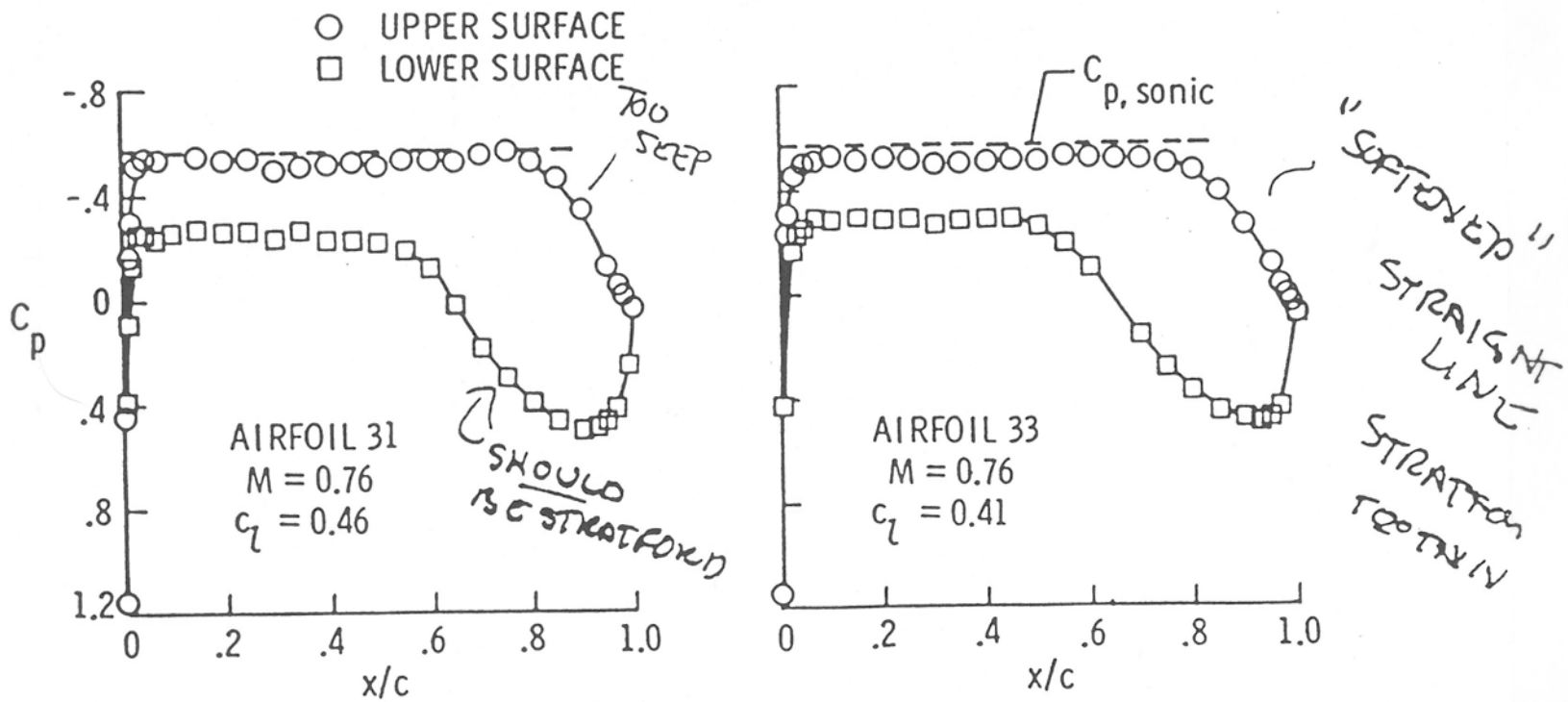


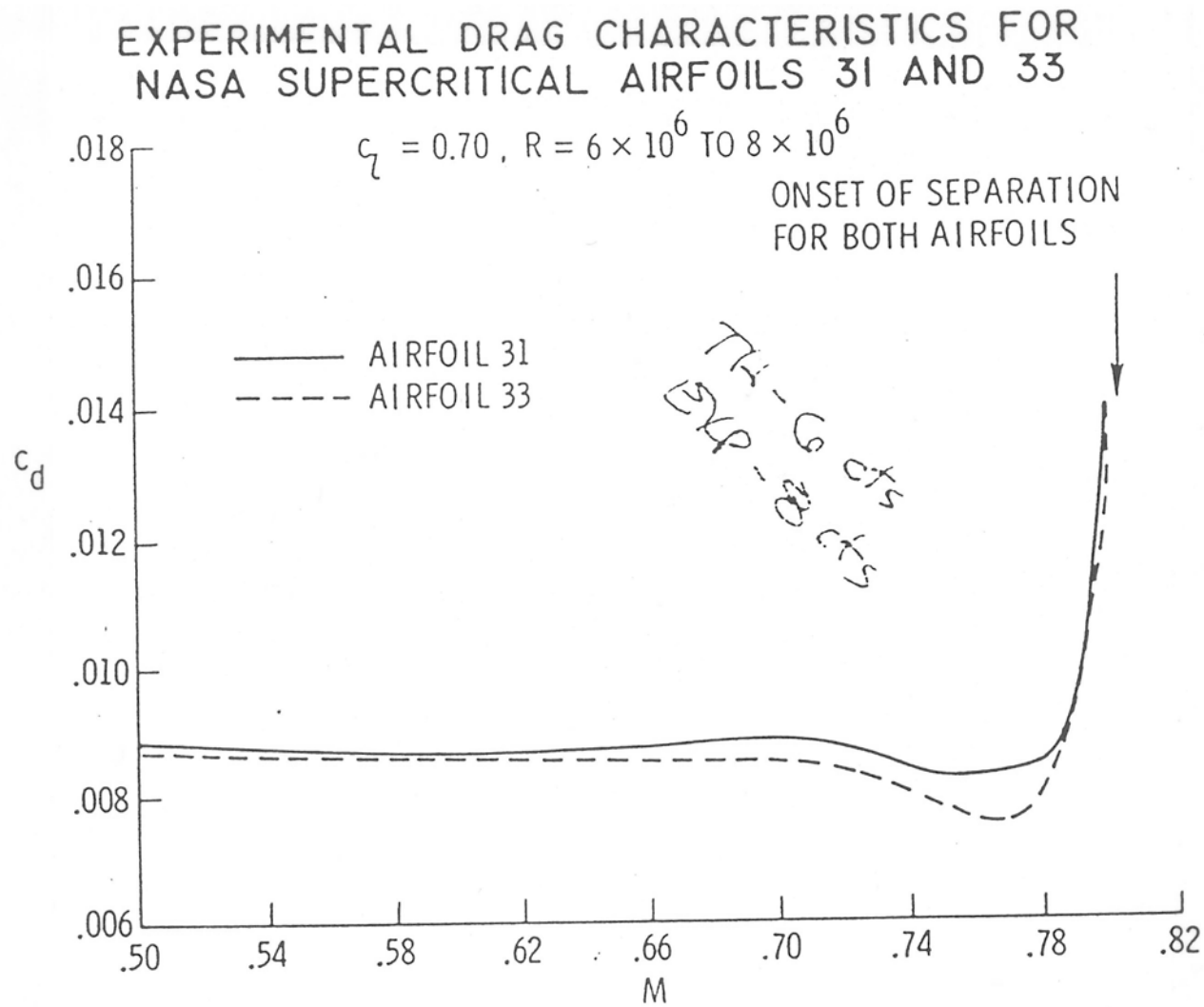
Figure 12

DROP M ~ .025

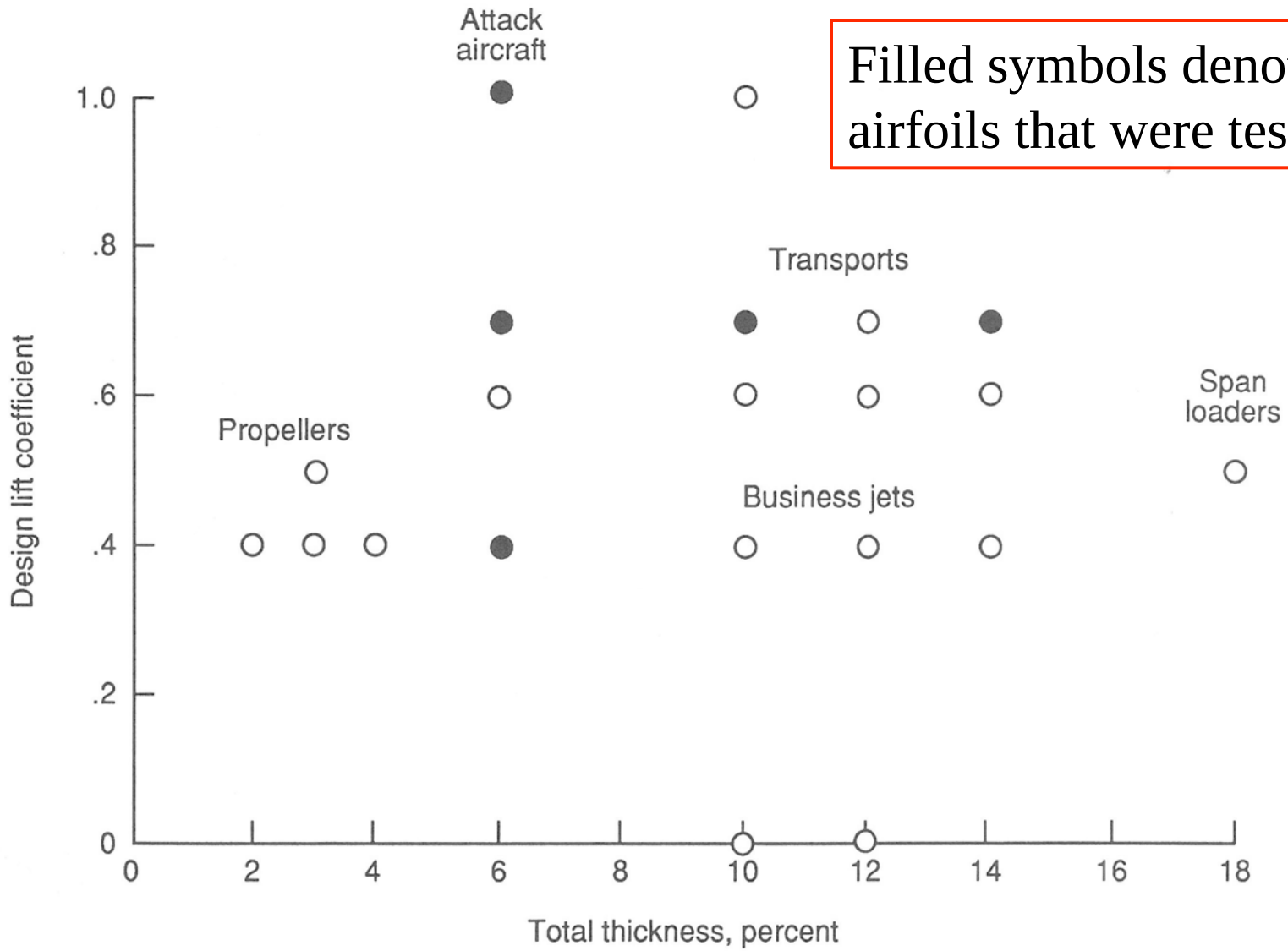
-GET SONIC PLAT.

IMP. FOR OFF DESIGN

Foils 31 and 33 Drag



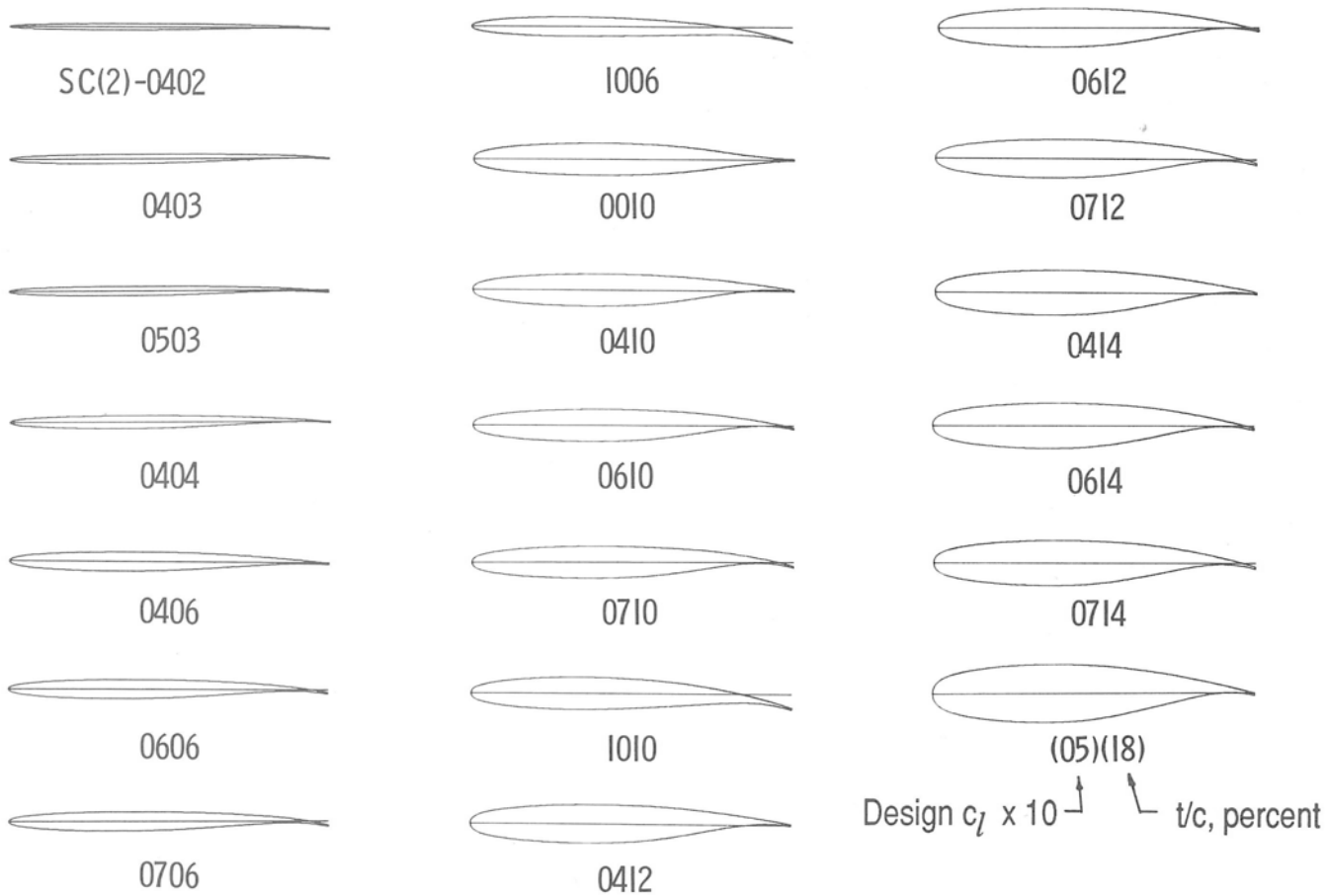
NASA Airfoils Developed Using the Guidelines



Filled symbols denote airfoils that were tested

from“NASA Supercritical Airfoils,” by Charles D. Harris, NASA TP 2969, March 1990

NASA Airfoil Catalog



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

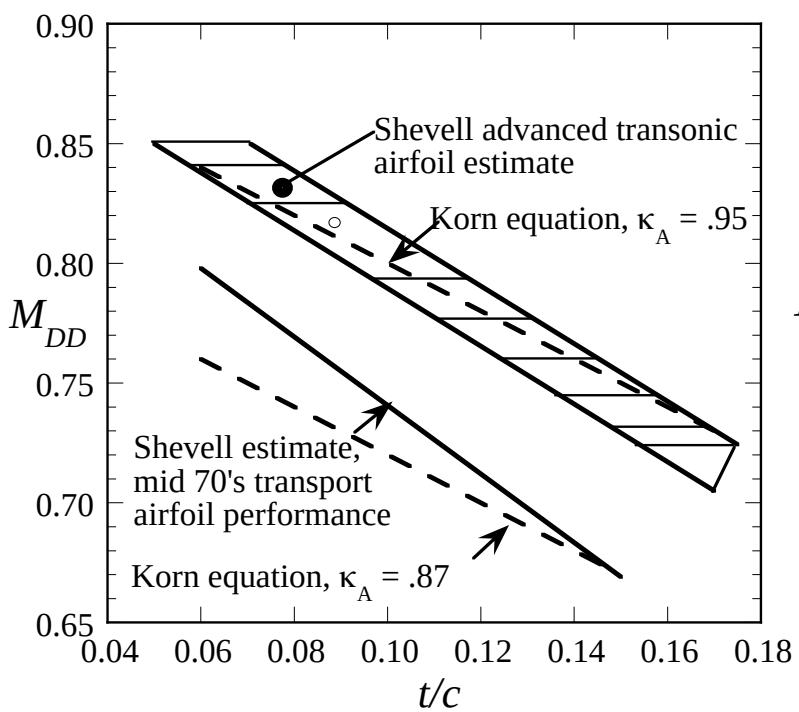
$$M_{DD} + \frac{C_L}{10} + \left(\frac{t}{c}\right) = \kappa_A$$

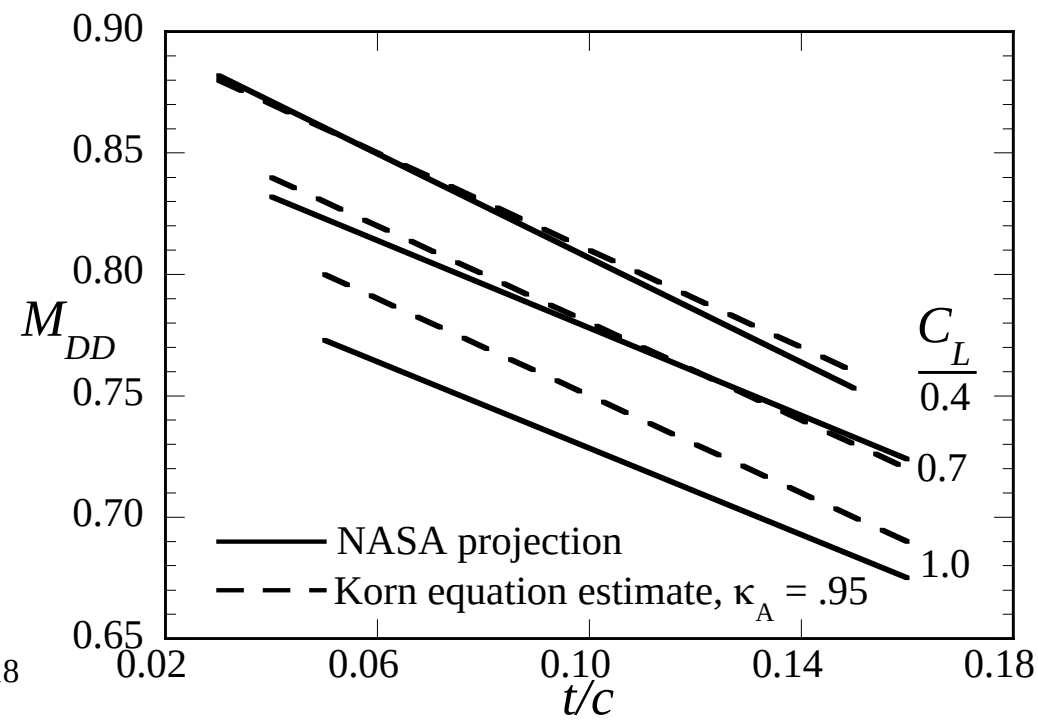
$\kappa_A = 0.87$ for conventional airfoils (6 series)

$\kappa_A = 0.95$ for supercritical airfoils

Note: the equation is sensitive to κ_A

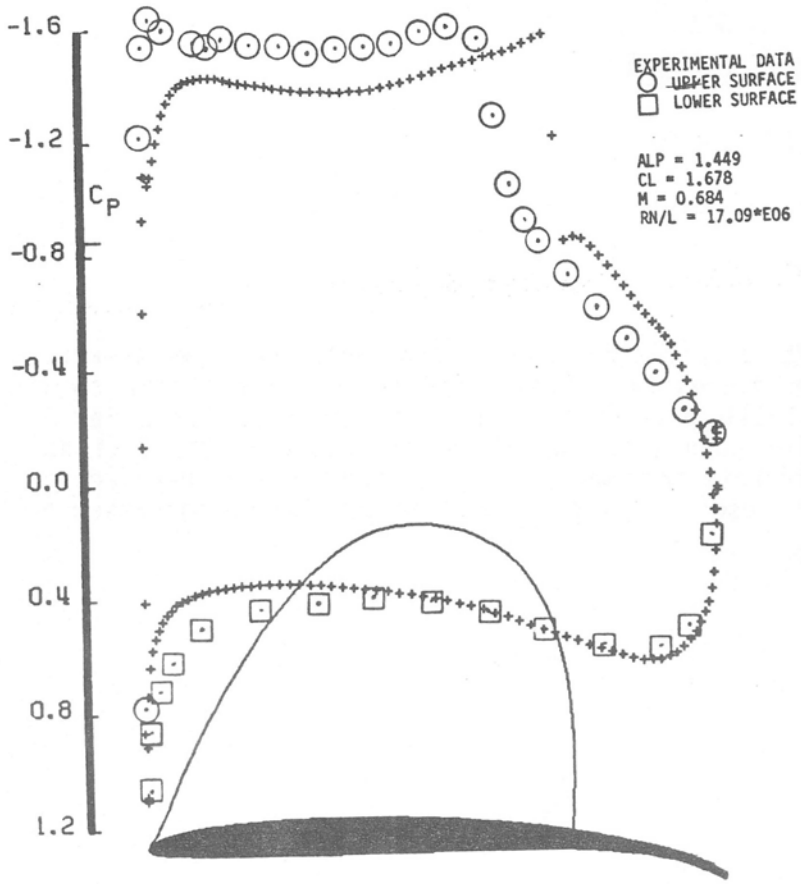
Airfoil Limits
Shevell and NASA Projections Compared to the Korn Equation

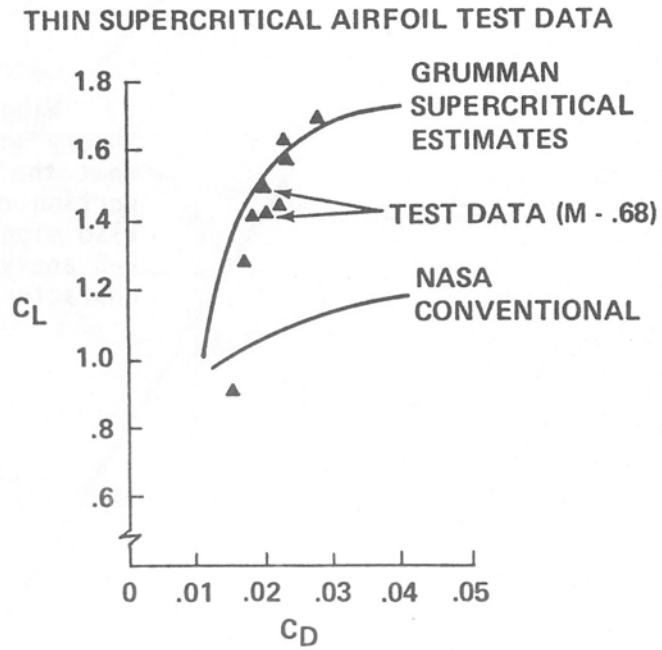




In W.H. Mason, “Analytic Models for Technology Integration in Aircraft Design,” AIAA Paper 90-3262, September 1990.

For the curious: the airfoil used on the X-29





GRUMMAN K (EXP.) AIRFOIL RN= 9.70 MILLION

M×N=160×30 NCY= 15 M=.683 ALP=-1.65 CL=1.634 CD=.0290

CM=-.3499 CDW=0.0116 CDF=0.0174 L/D= 56.37

UPPER SEP AT 0.9319 LOWER SEP AT 1.0000

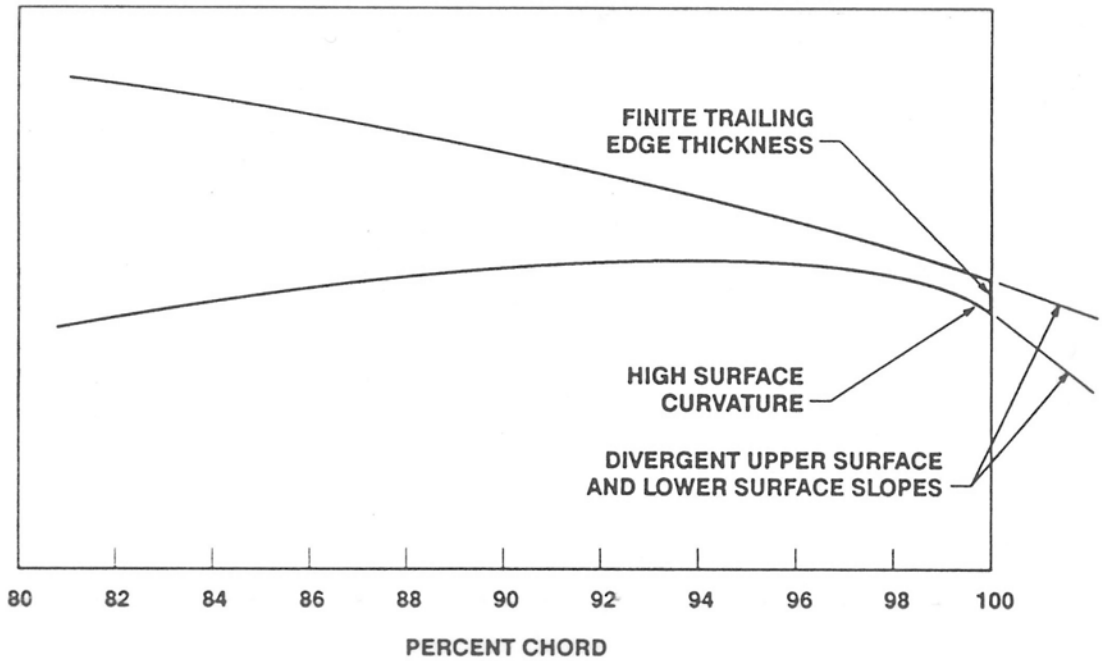
Divergent Trailing Edge Airfoil

A NEW AIRFOIL DESIGN CONCEPT

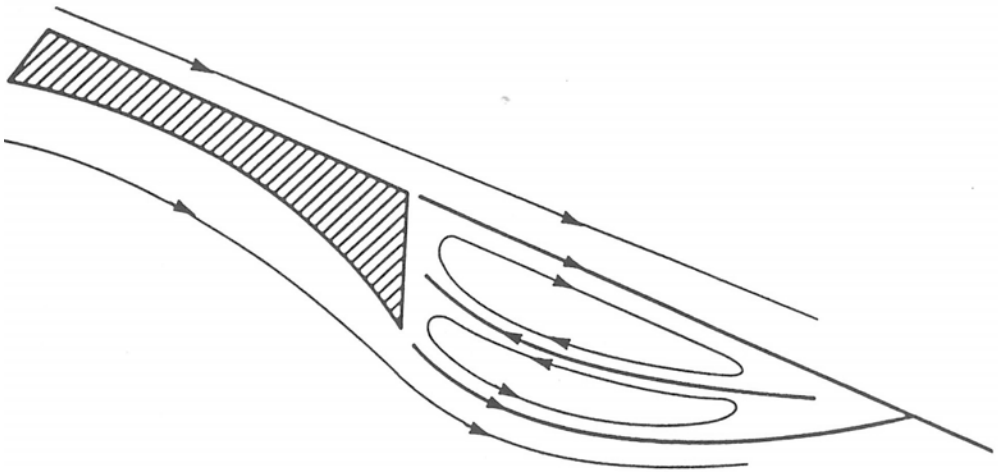
by
P. A. Henne and R. D. Gregg
Douglas Aircraft Company
McDonnell Douglas Corporation

Just when we thought airfoil design was "finished"

See Henne, "Innovation with Computational Aerodynamics: The Divergent Trailing Edge Airfoil," in *Applied Computational Aerodynamics*, P.A. Henne, ed., AIAA Progress in Aero Series, 1990

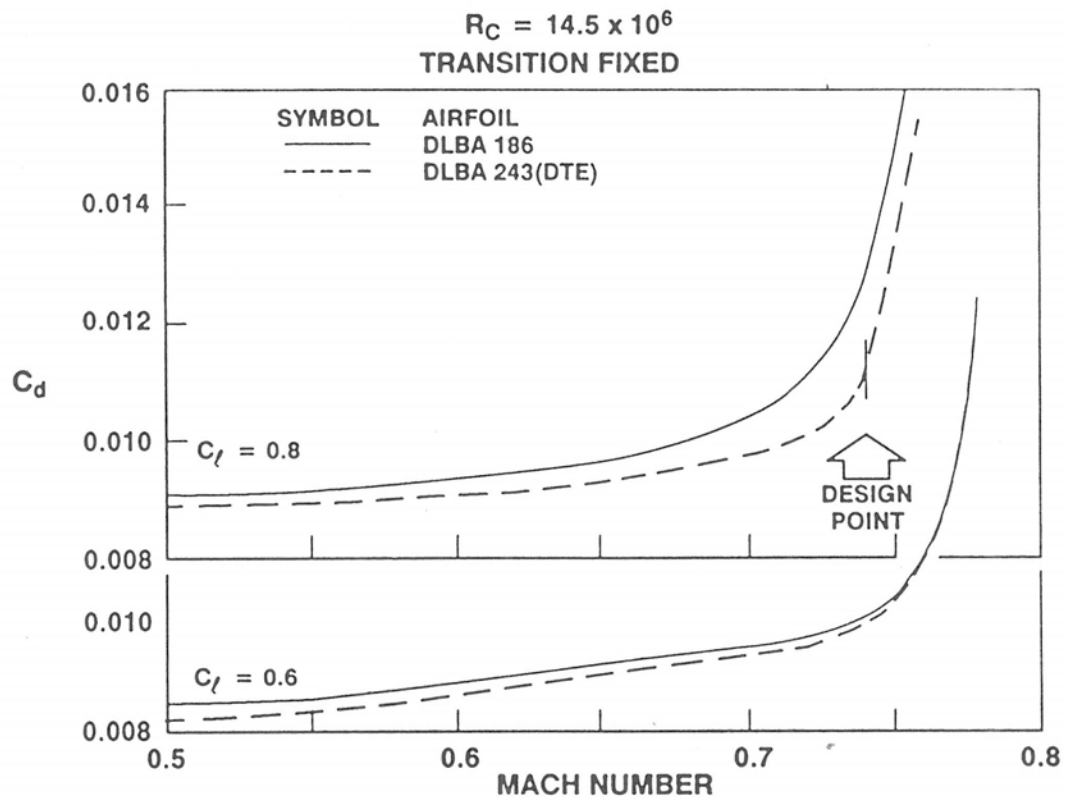


Used on the MD-11 resisted in Seattle!



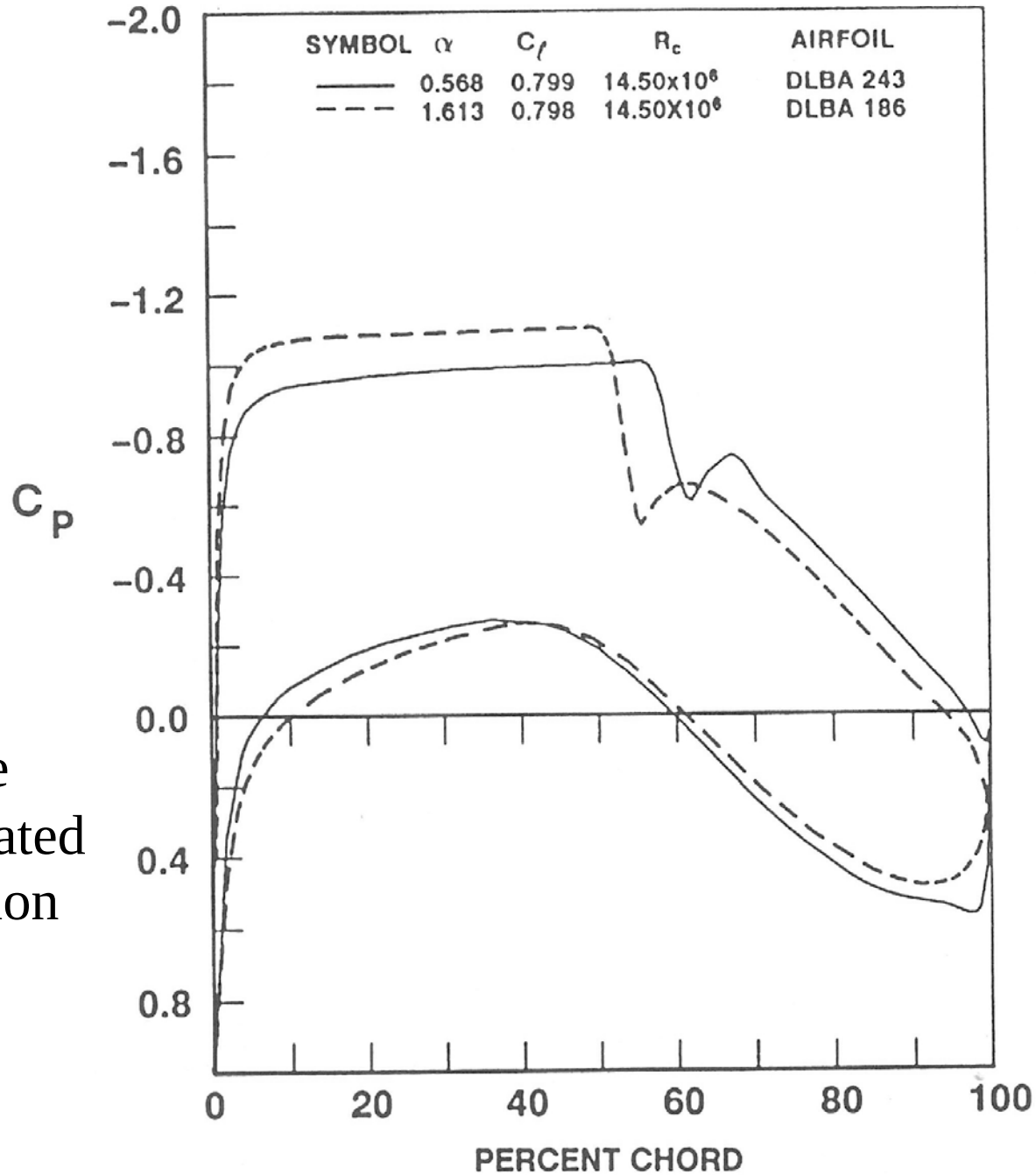
Hypothesized Flow Closure for Divergent Trailing Edge Airfoil

Comparison of DLBA 243 and DLBA 186 Calculated Drag Rise Characteristics



Aerospace and Ocean Engineering

Take a Look at the Pressure Distribution

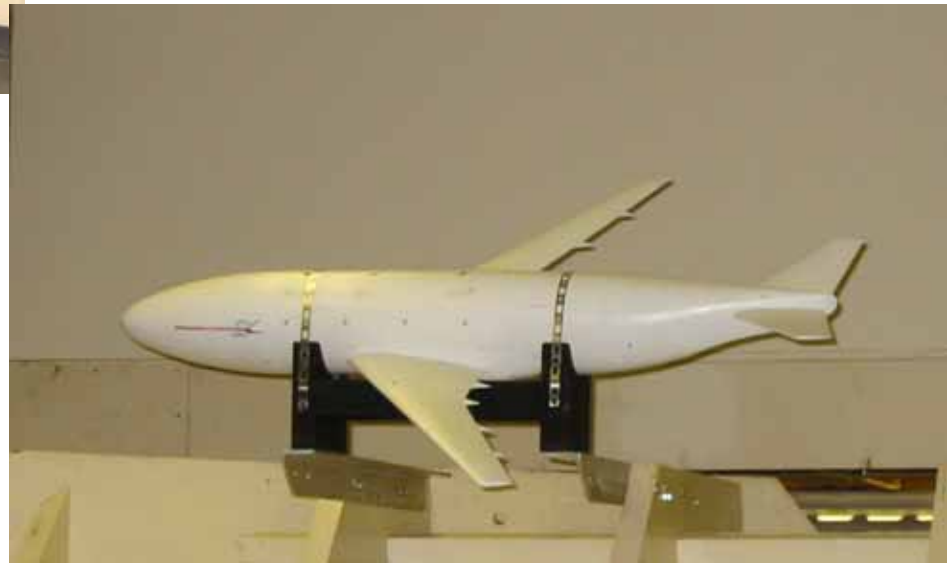


Comparison of the DLBA 243 and the DLBA 186 Calculated Pressure Distribution at M = 0.74

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

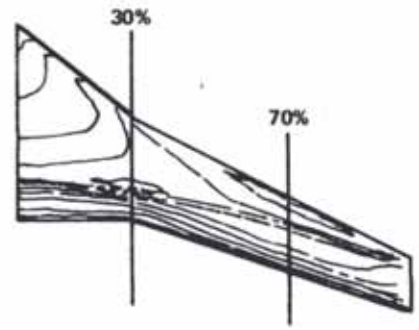


The original model carved by George Shairer

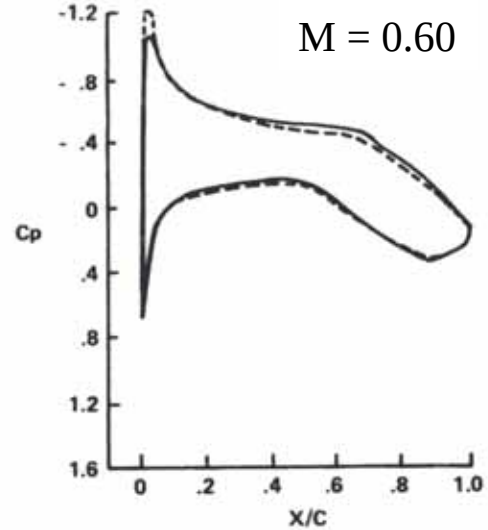
Picture of B-52 made in Dayton Hotel over a weekend by Boeing engineers

picture taken by Mason at the Museum of Flight, Seattle, WA

The problem of transonic wing design



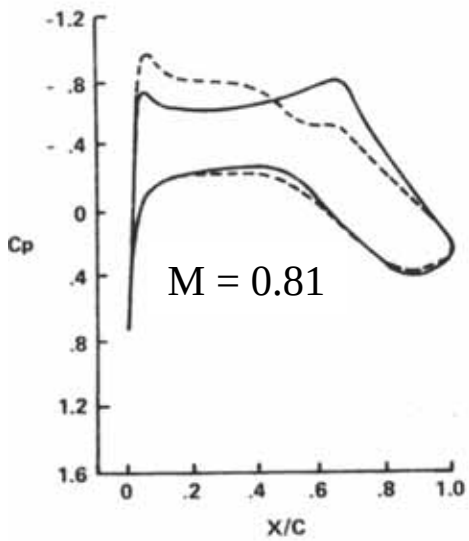
TRANSONIC CRUISE WING

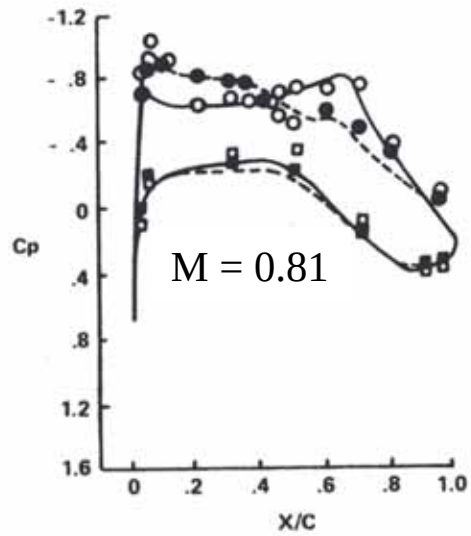


DATA PREDICTION

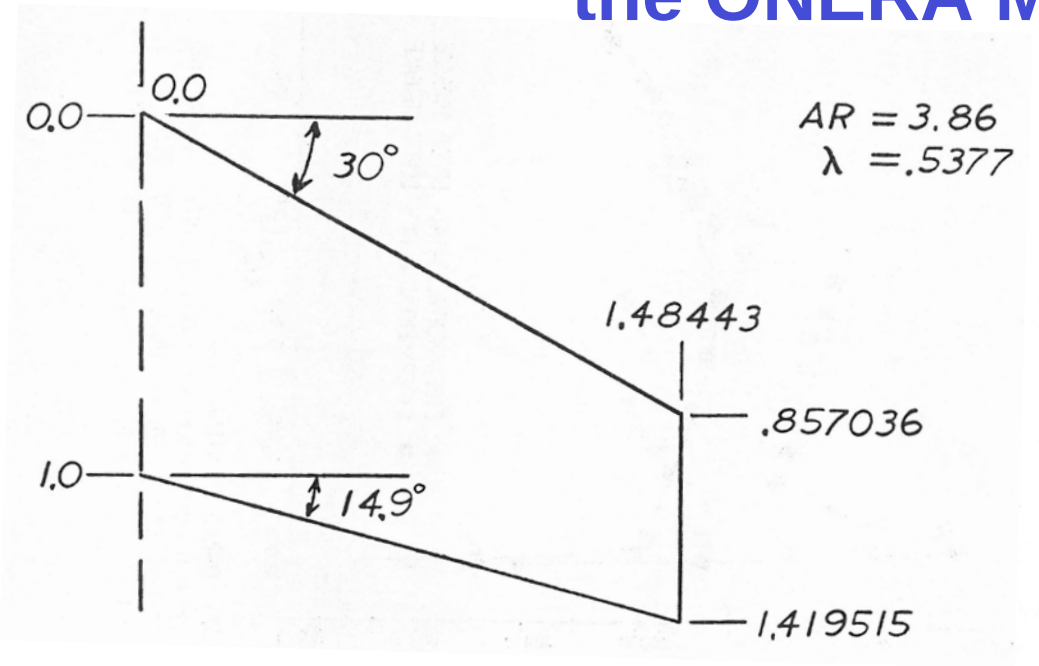
○ □ ——— 30% SPAN STATION

● ■ – – – 70% SPAN STATION

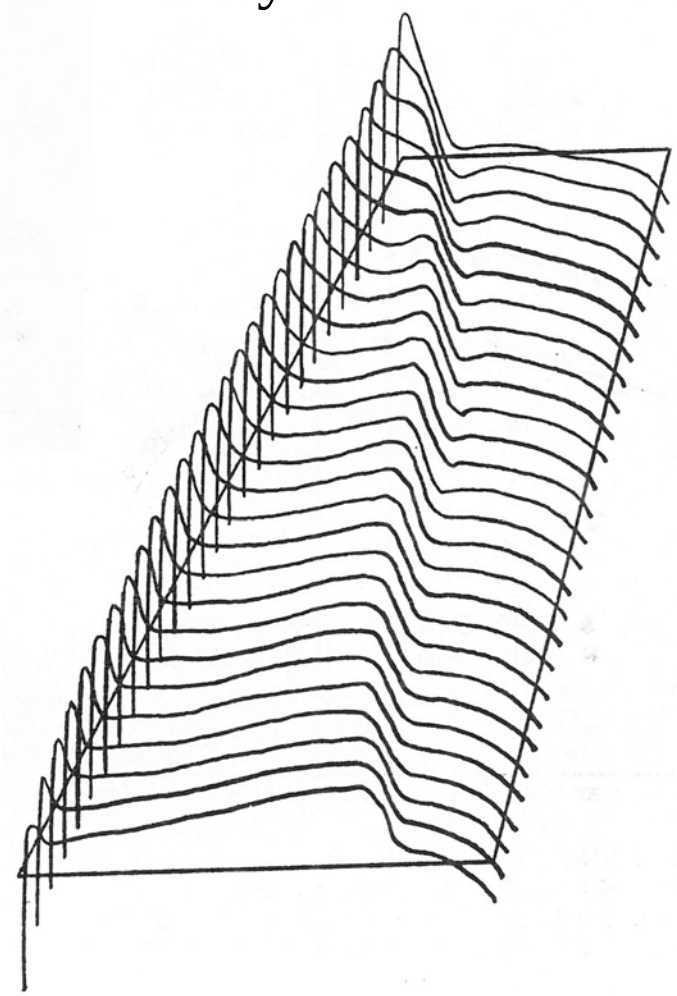




The Standard Transonic Test Case: the ONERA M6 Wing



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

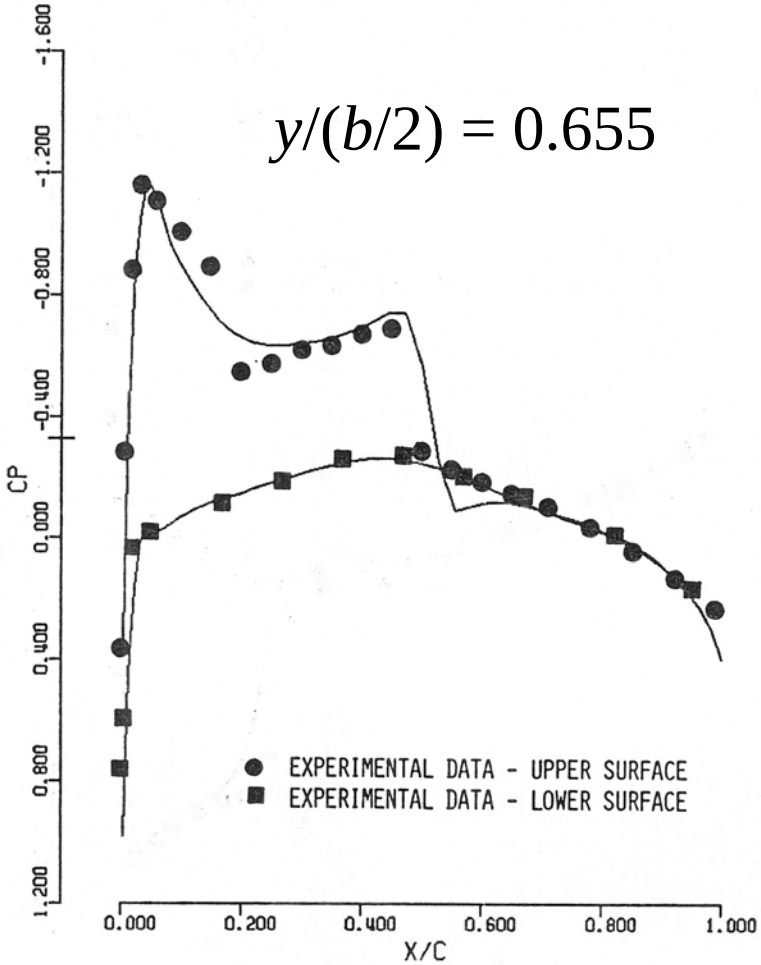


FIGURE 32E. ONERA M6 WING
M = 0.840 ALF = 3.000 CONV = .327E-04
LOCAL CL= 0.292 LOCAL CM=-0.288 LOCAL CD=-.0032
SPAN STATION 19 Y/S=0.655 44 MESH PTS ON CHD

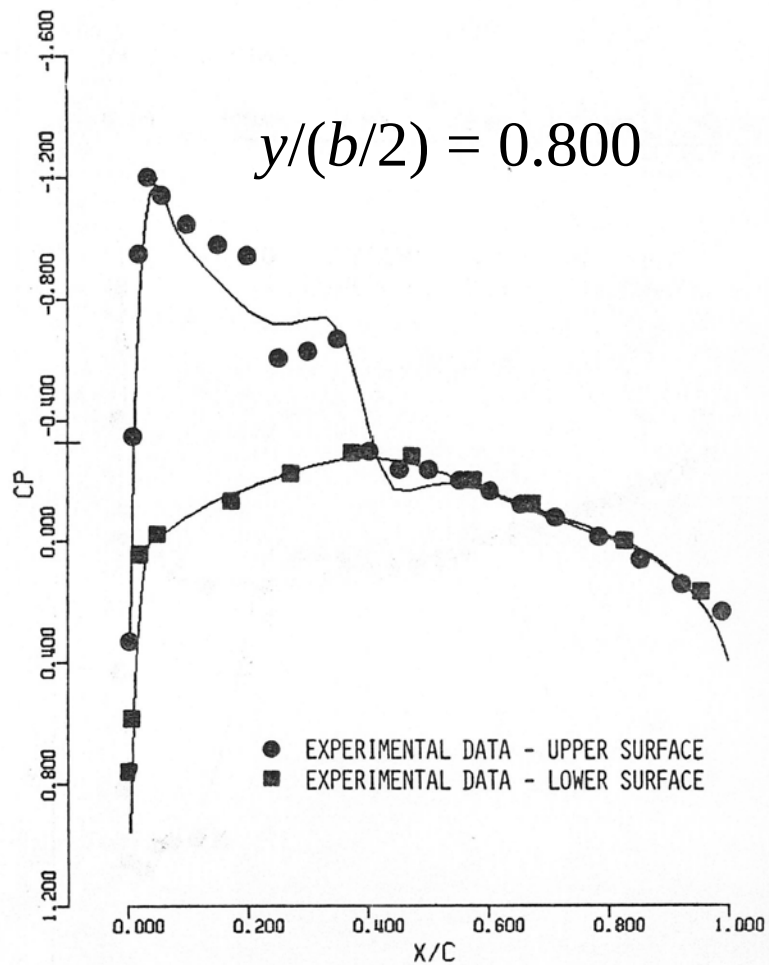


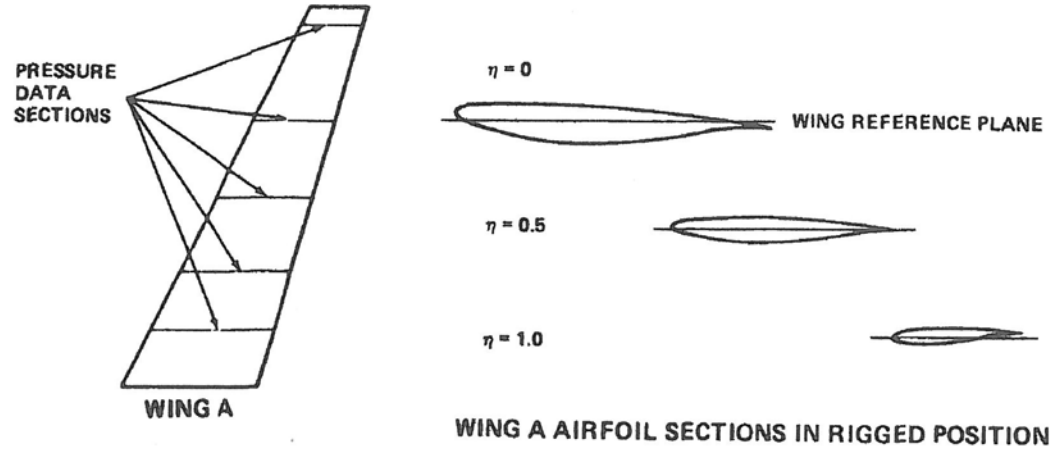
FIGURE 32F. ONERA M6 WING
M = 0.840 ALF = 3.000 CONV = .327E-04
LOCAL CL= 0.258 LOCAL CM=-0.312 LOCAL CD=-.0089
SPAN STATION 23 Y/S=0.800 44 MESH PTS ON CHD

W.H. Mason, D.A. MacKenzie, M.A. Stern, W.F Ballhaus Jr., and J. Frick, “Automated Procedure for Computing the Three-Dimensional Transonic Flow Over Wing-Body Combinations, including Viscous Effects,” AFFDL-TR-77-122, February 1978.

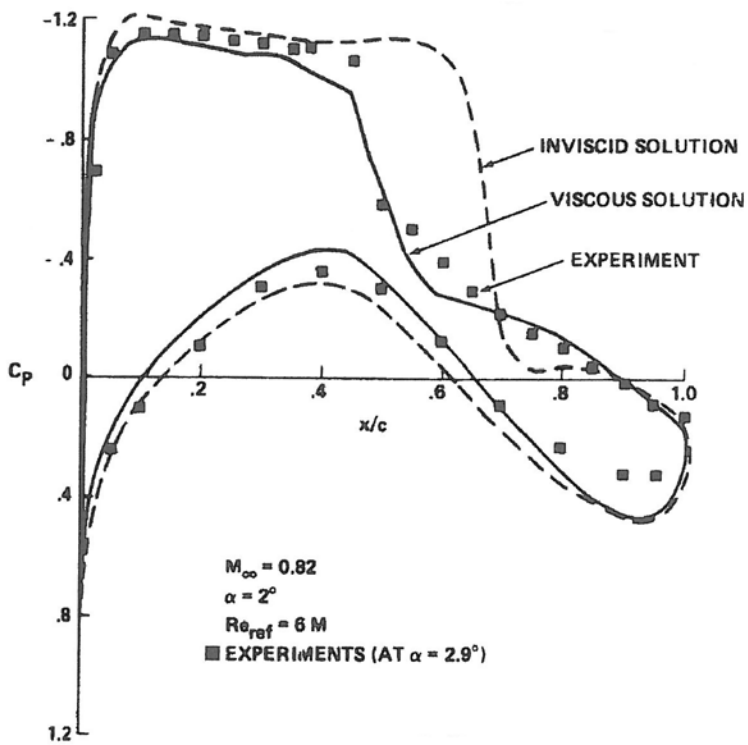
Viscous Effects Still Important

Lockheed Wing A

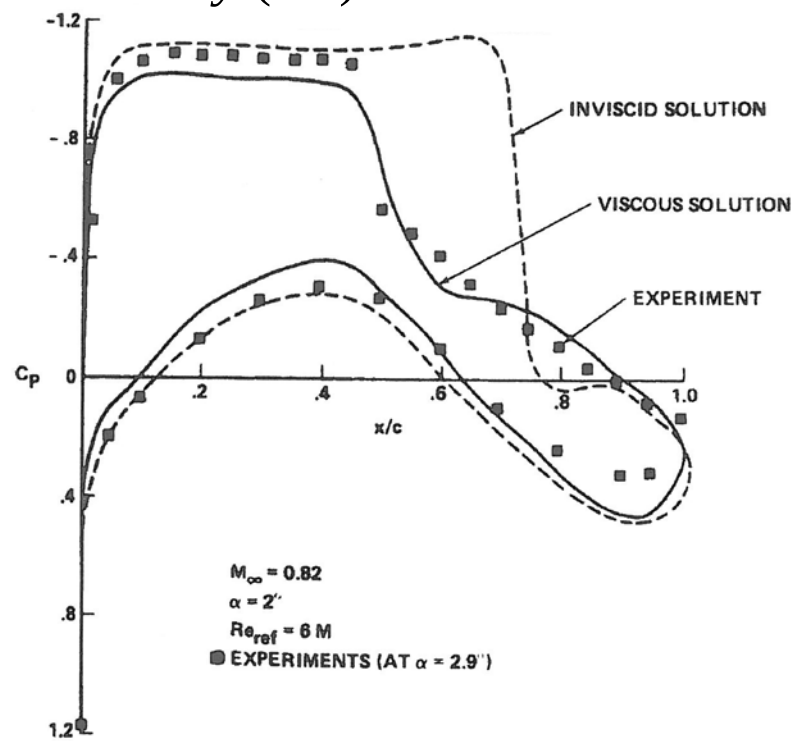
From NASA CR178156,
Oct. 1986



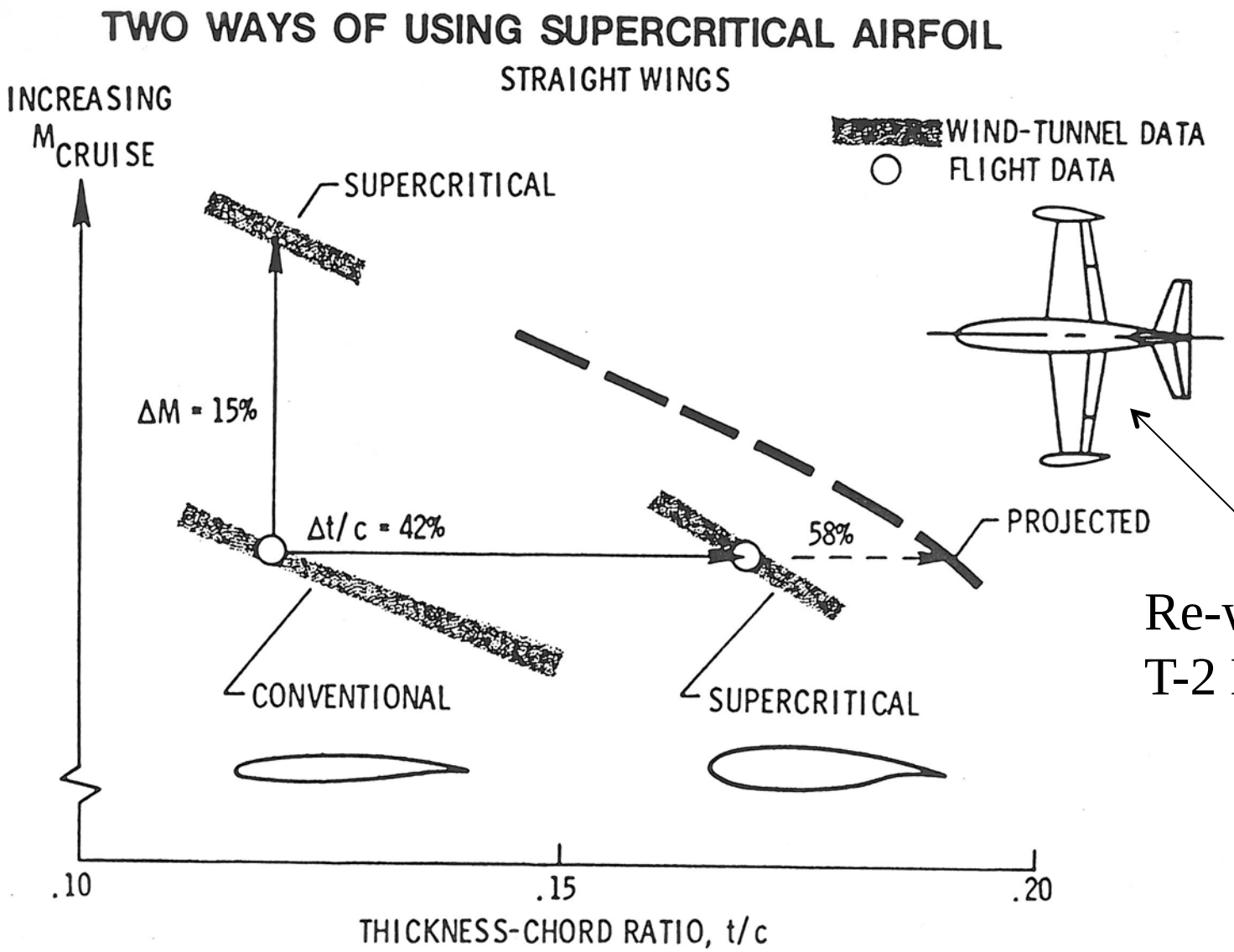
$y/(b/2) = 0.50$



$y/(b/2) = 0.70$



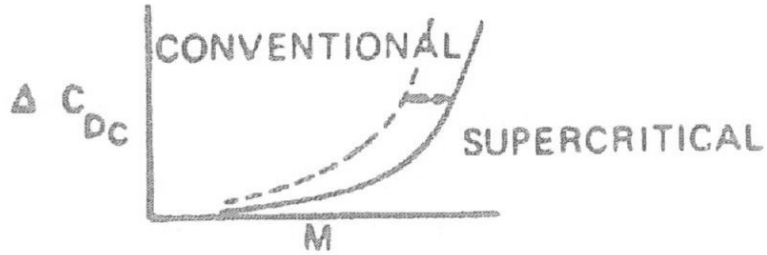
How to use advanced airfoil capability in wing design



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



FOR ENERGY EFFICIENCY

REDUCE SWEEP

FOR SIMILAR M_O

- REDUCES WEIGHT
- IMPROVED HIGH-SPEED POST-BUFFET PITCH CHARACTERISTICS
- IMPROVED LOW-SPEED PERFORMANCE

INCREASE THICKNESS

FOR SIMILAR M_O

- REDUCES WEIGHT
- INCREASES FUEL VOLUME
- INCREASES $C_{L_{MAX}}$

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:

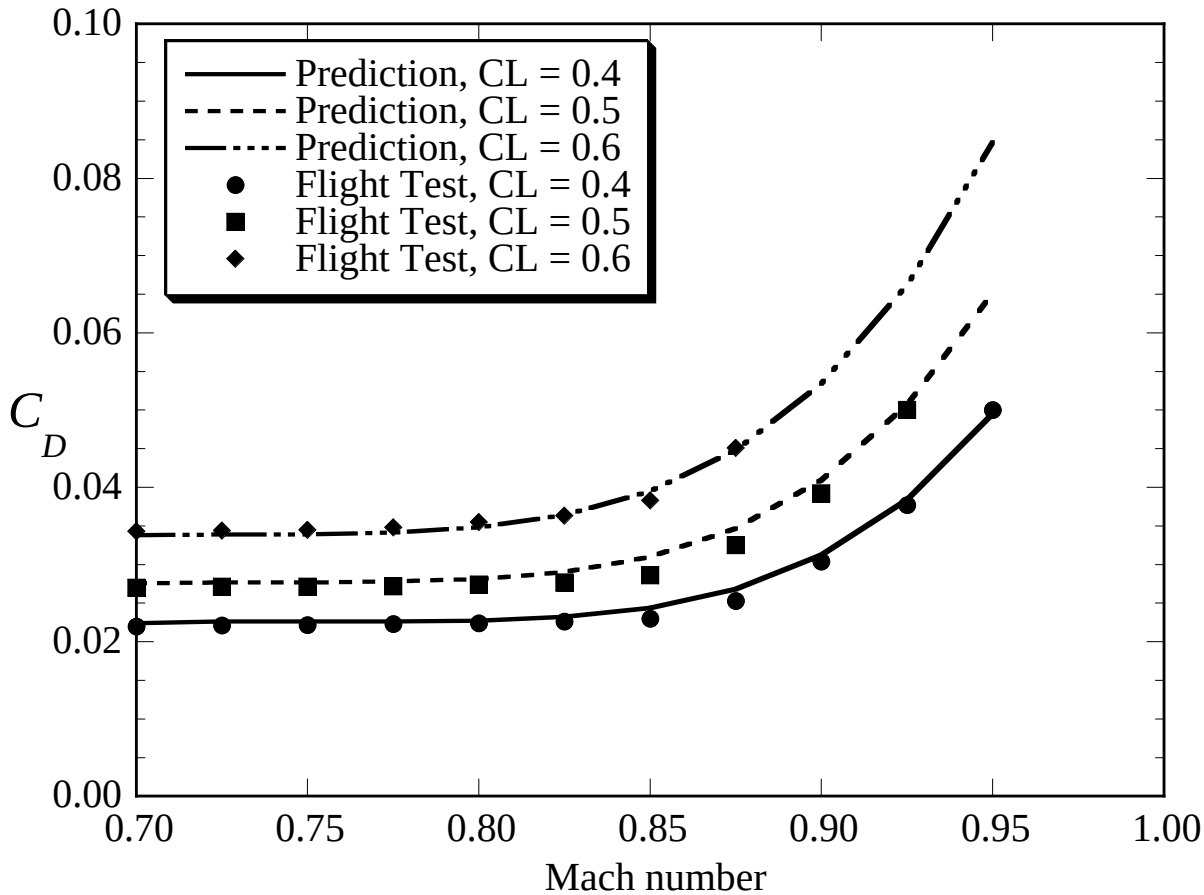
$$C_D = 20\,(M - M_{crit})^4$$

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

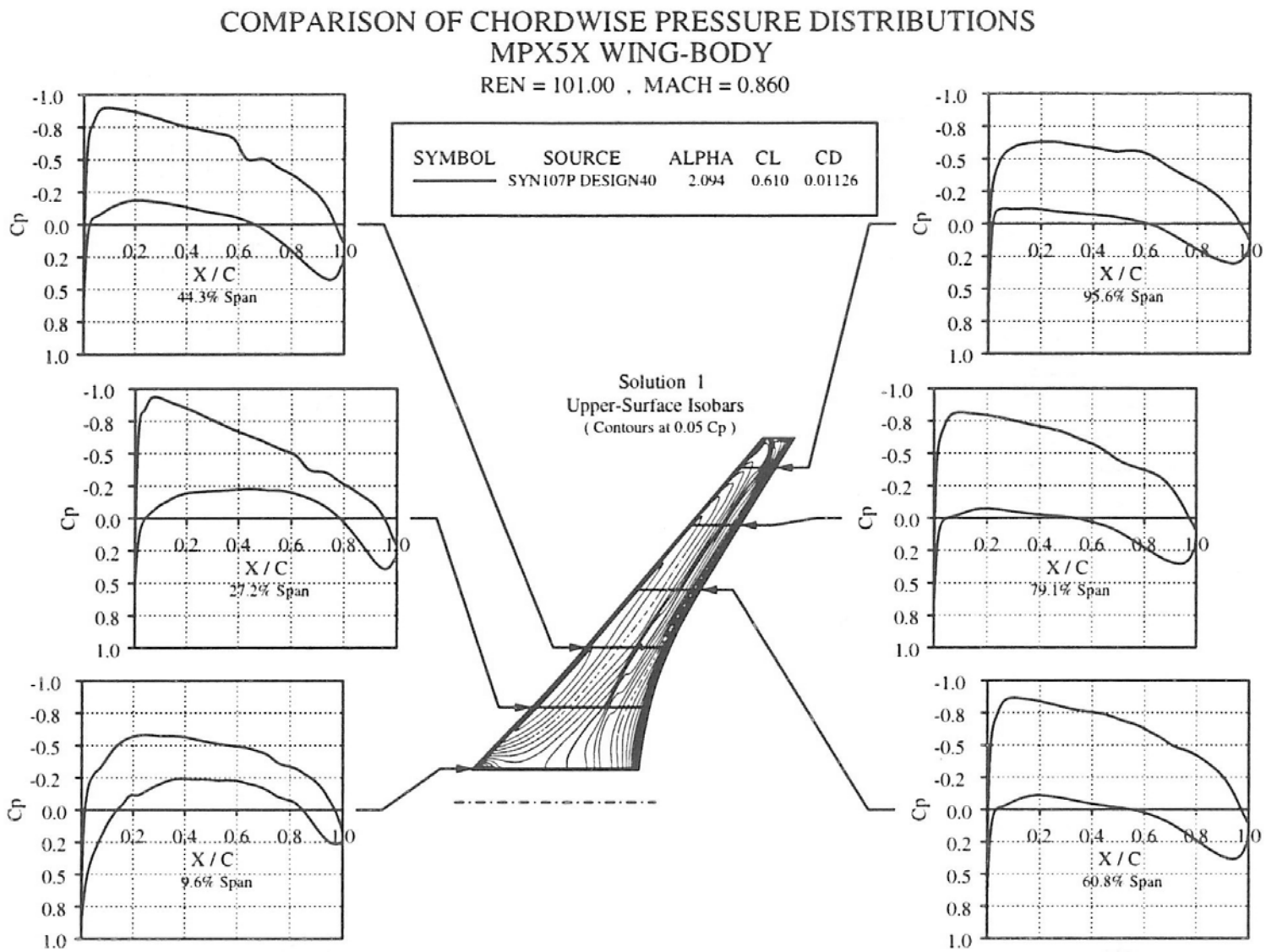


747-100 test data taken from Mair and Birdsall,
Aircraft Performance, Cambridge University Press, 1992, pp. 255-257

So how do you design the wing?

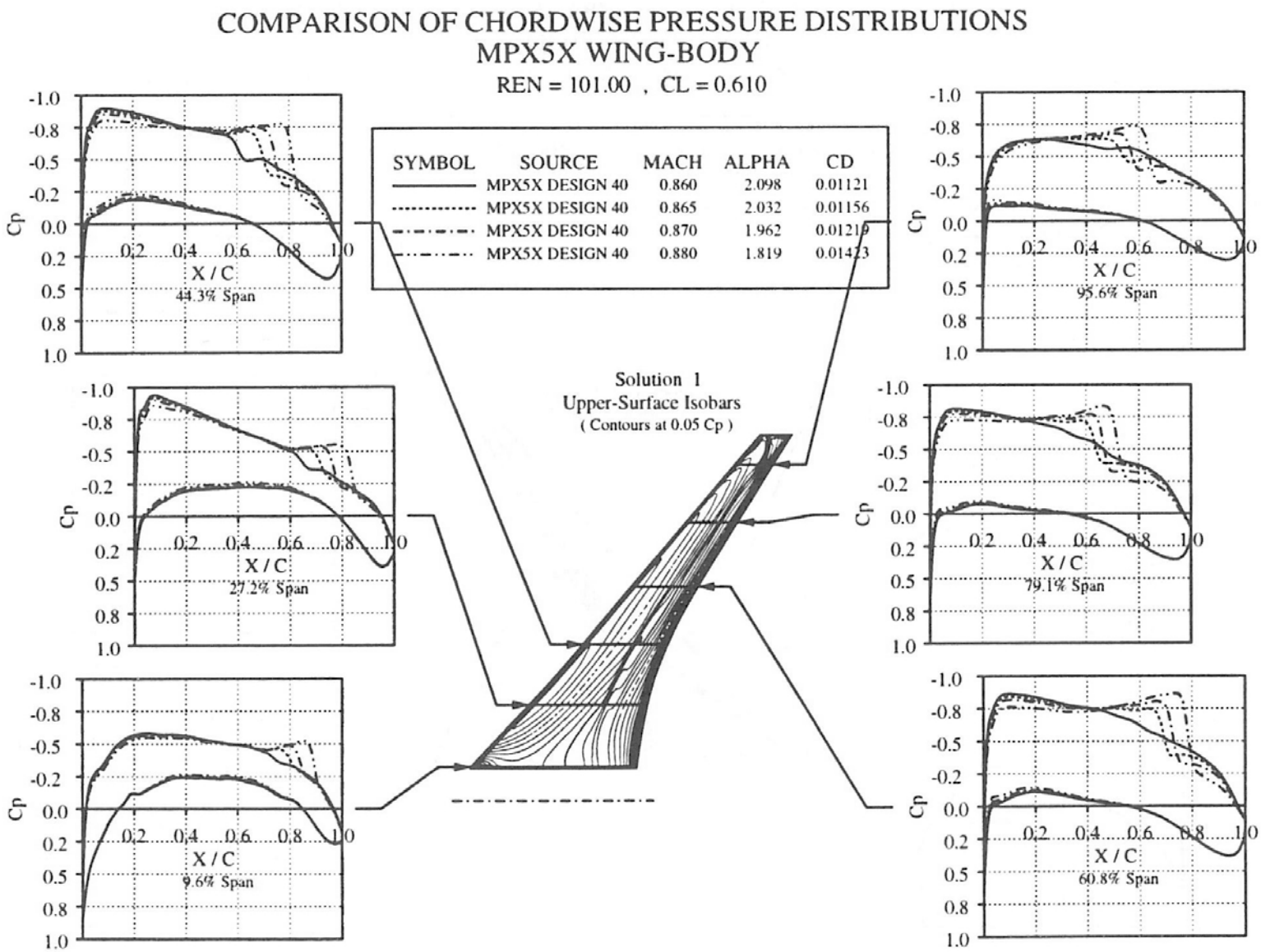
- Set the spanload in an MDO trade
- Linear theory gives a good first guess for the twist distribution
- Design the airfoils in 2D to start, and place in wing
- Finally, full 3D nonlinear analysis and design
- Special attention to root and tip mods to maintain isobar sweep
- Well known considerations:
  - 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



Antony Jameson, “Re-Engineering the Design Process through Computation” AIAA Paper 97-0641, January 1997

Off Design Characteristics of the Jameson Wing



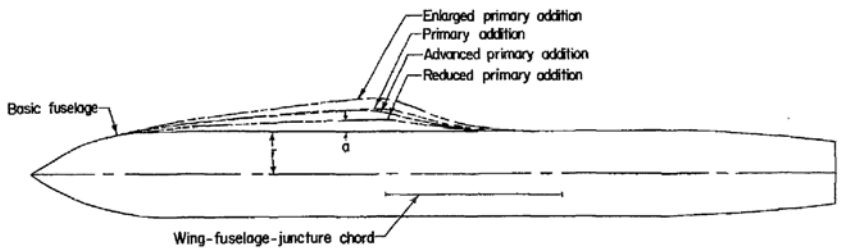
Antony Jameson, “Re-Engineering the Design Process through Computation” AIAA Paper 97-0641, January 1997

Transonic Transport Wing Design
Key Lessons learned at Boeing

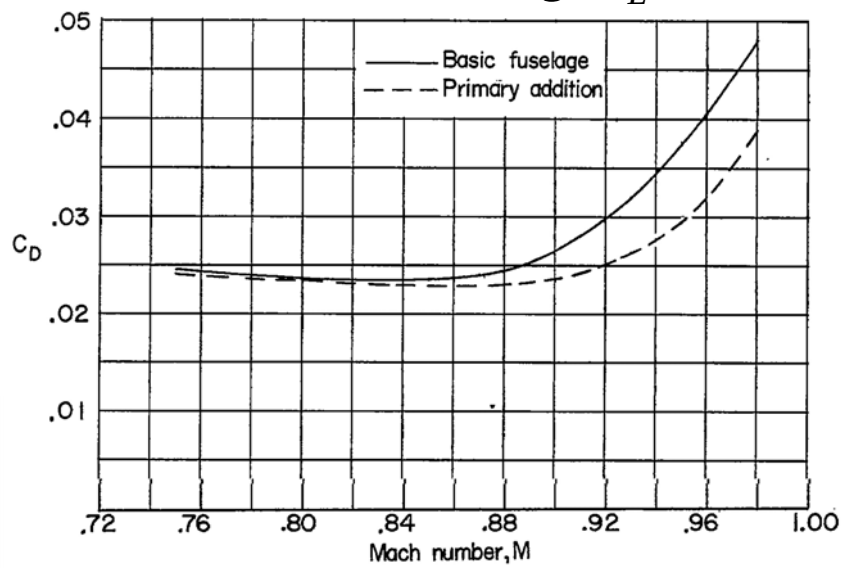
- You can increase the root thickness w/o a drag penalty
- Nacelle/pylon wing interference can be done with CFD with almost no penalty

Finally: the fuselage can effect drag

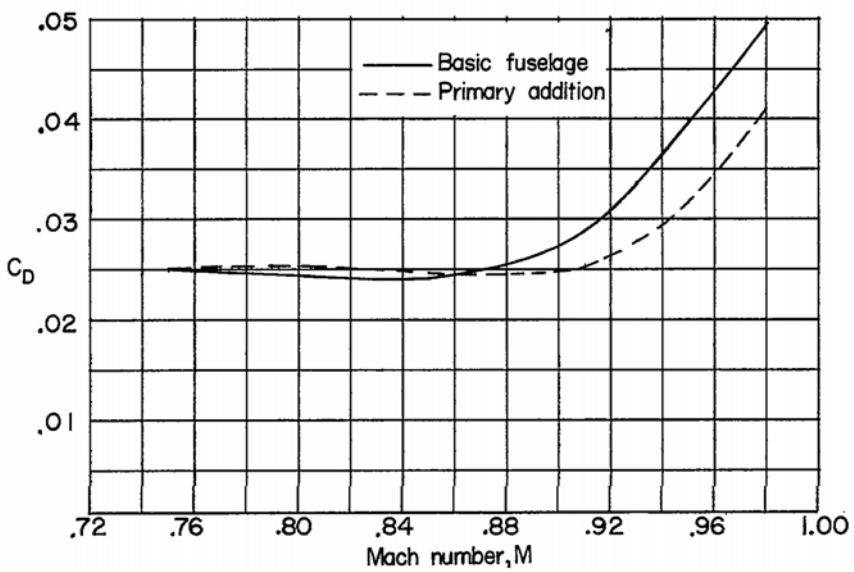
Another Whitcomb Contribution: Fuselage Area Addition



Low mounted wing, $C_L = 0.3$



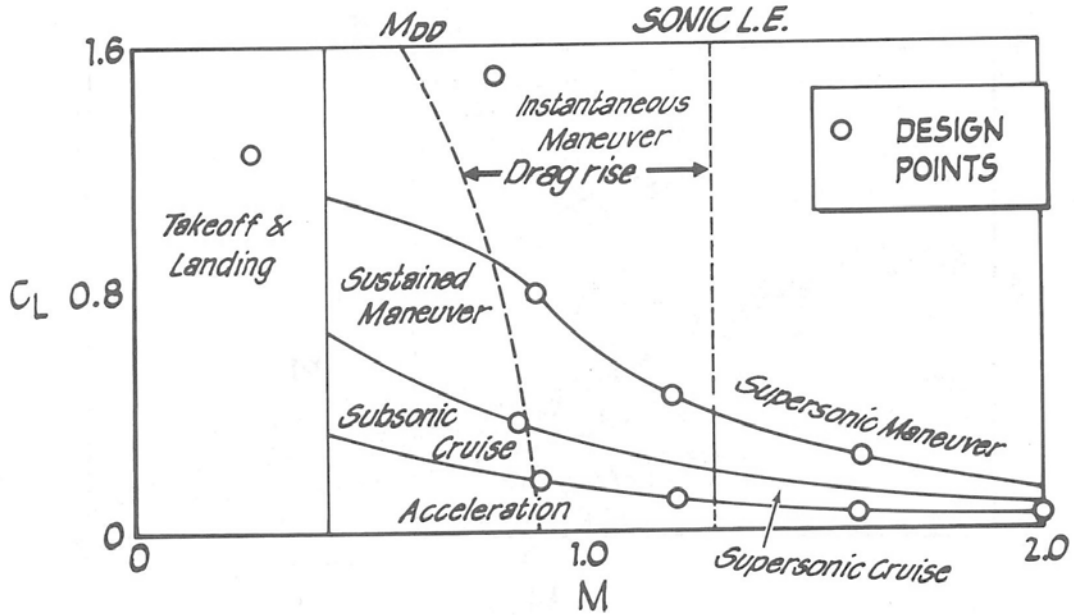
High mounted wing, $C_L = 0.3$



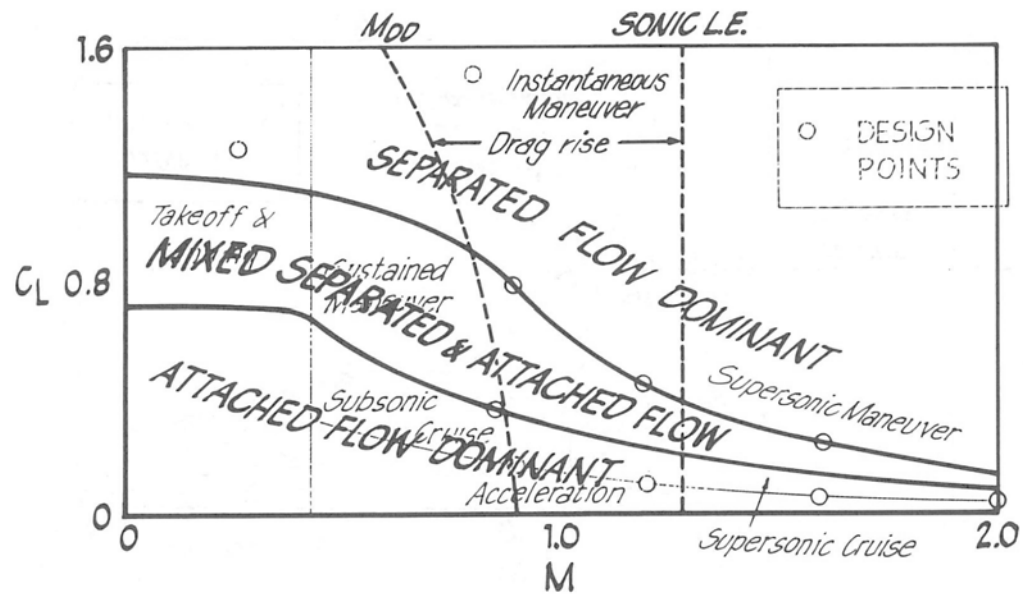
Adding a fuselage bump reduces drag slightly above $M = 0.85$: the Boeing 747 used this idea, it was supposed to cruise at M = 0.90

From Whitcomb NACA TN 4290, June 1958.

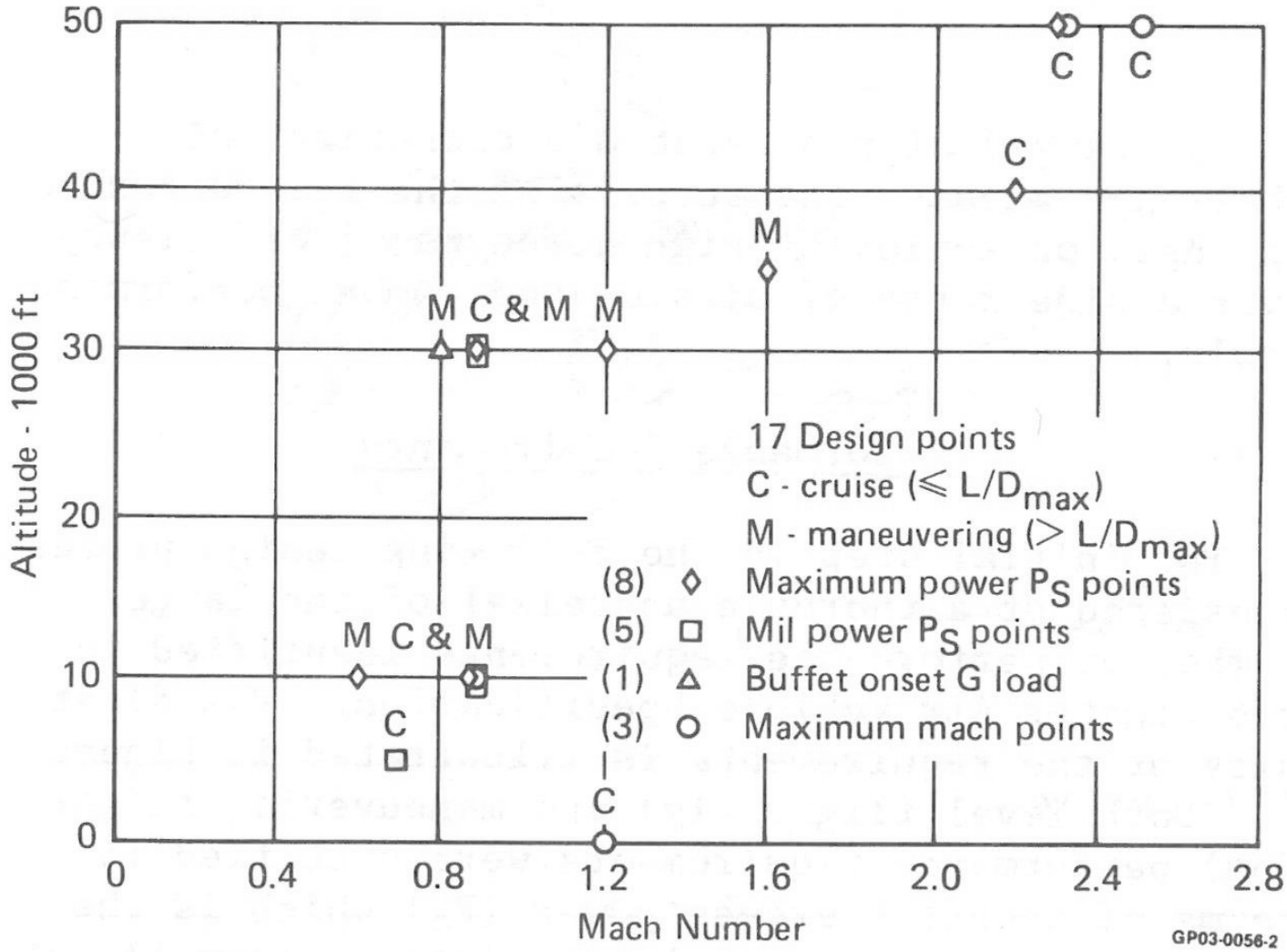
Fighter Wings: A more complicated story



From Richard G. Bradley, "Practical Aerodynamic Problems-Military Aircraft" in *Transonic Aerodynamics*, AIAA 1982

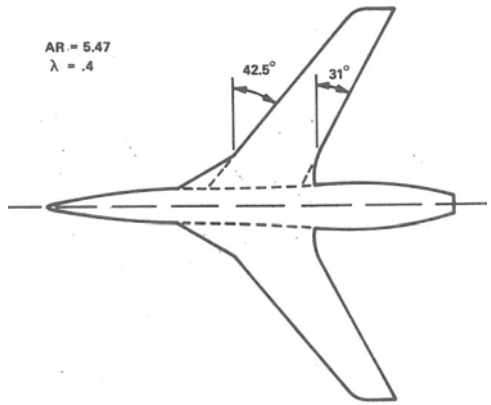


Design Points for the F-15



From Larry Niedling, “The F-15 Wing Development Program”, AIAA Paper 80-3044

Example of attached flow fighter design



(a) ADVANCED TECHNOLOGY FIGHTER

Grumman proposed HiMAT, $M = 0.90$

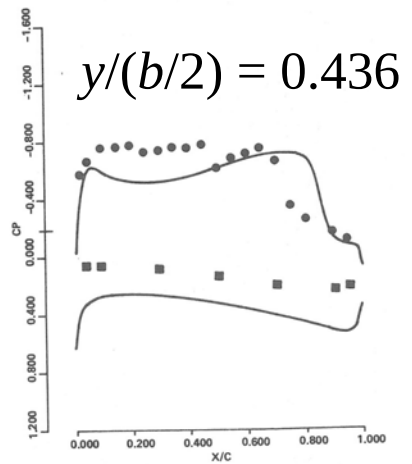


FIGURE 9b. ADVANCED FIGHTER
M = 0.900 ALF = 7.050 CONV = .886E-01
LOCAL CL = 0.890 LOCAL CM = 1.306 LOCAL CD = 0.0585
SPAN STATION 13 Y/S = 0.436 44 MESH PTS ON CHD

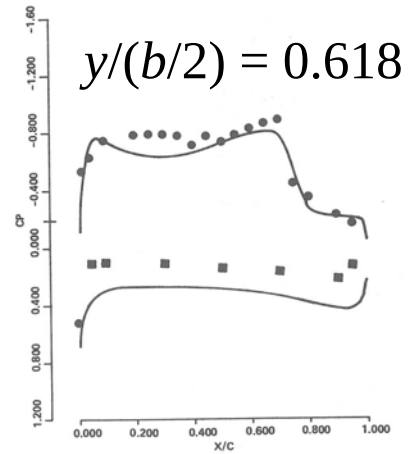


FIGURE 9c. ADVANCED FIGHTER
M = 0.900 ALF = 7.050 CONV = .886E-01
LOCAL CL = 0.911 LOCAL CM = 1.218 LOCAL CD = 0.0357
SPAN STATION 18 Y/S = 0.618 44 MESH PTS ON CHD

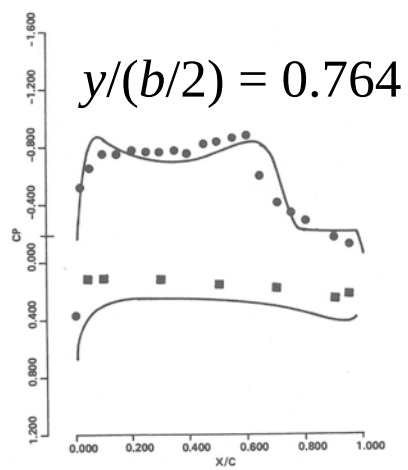


FIGURE 9d. ADVANCED FIGHTER
M = 0.900 ALF = 7.050 CONV = .886E-01
LOCAL CL = 0.910 LOCAL CM = 1.075 LOCAL CD = 0.0206
SPAN STATION 22 Y/S = 0.764 44 MESH PTS ON CHD

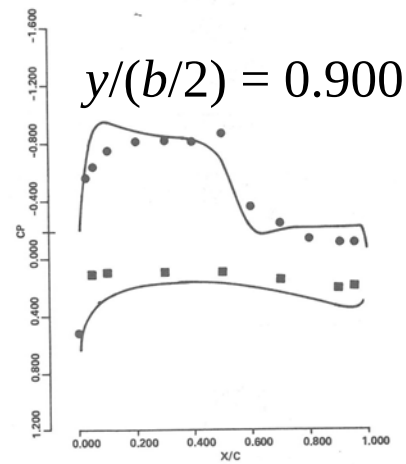
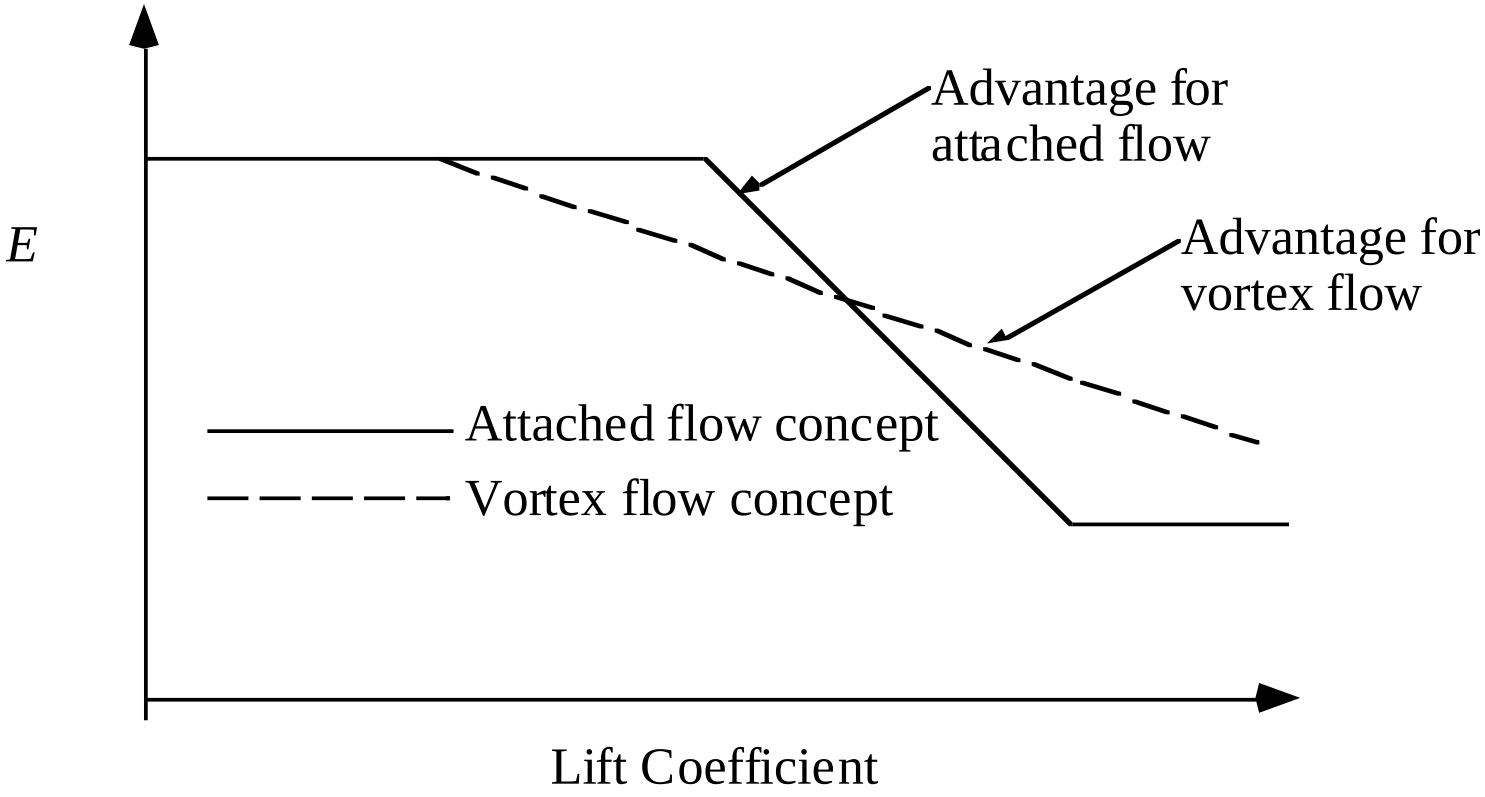


FIGURE 9e. ADVANCED FIGHTER
M = 0.900 ALF = 7.050 CONV = .886E-01
LOCAL CL = 0.782 LOCAL CM = 0.766 LOCAL CD = -.0012
SPAN STATION 26 Y/S = 0.909 44 MESH PTS ON CHD

W.H. Mason, D.A. MacKenzie, M.A. Stern, and J.K. Johnson, "A Numerical Three Dimensional Viscous Transonic Wing-Body Analysis and Design Tool," AIAA Paper 78-101, Jan. 1978.

Fighter Wings: Vortex Flow effects



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