

Some Transonic Aerodynamics

W.H. Mason

Configuration Aerodynamics Class

Transonic Aerodynamics History

- Pre WWII propeller tip 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
 - Transonic flow is inherently nonlinear, there are no useful theoretical methods

The Sound Barrier!

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

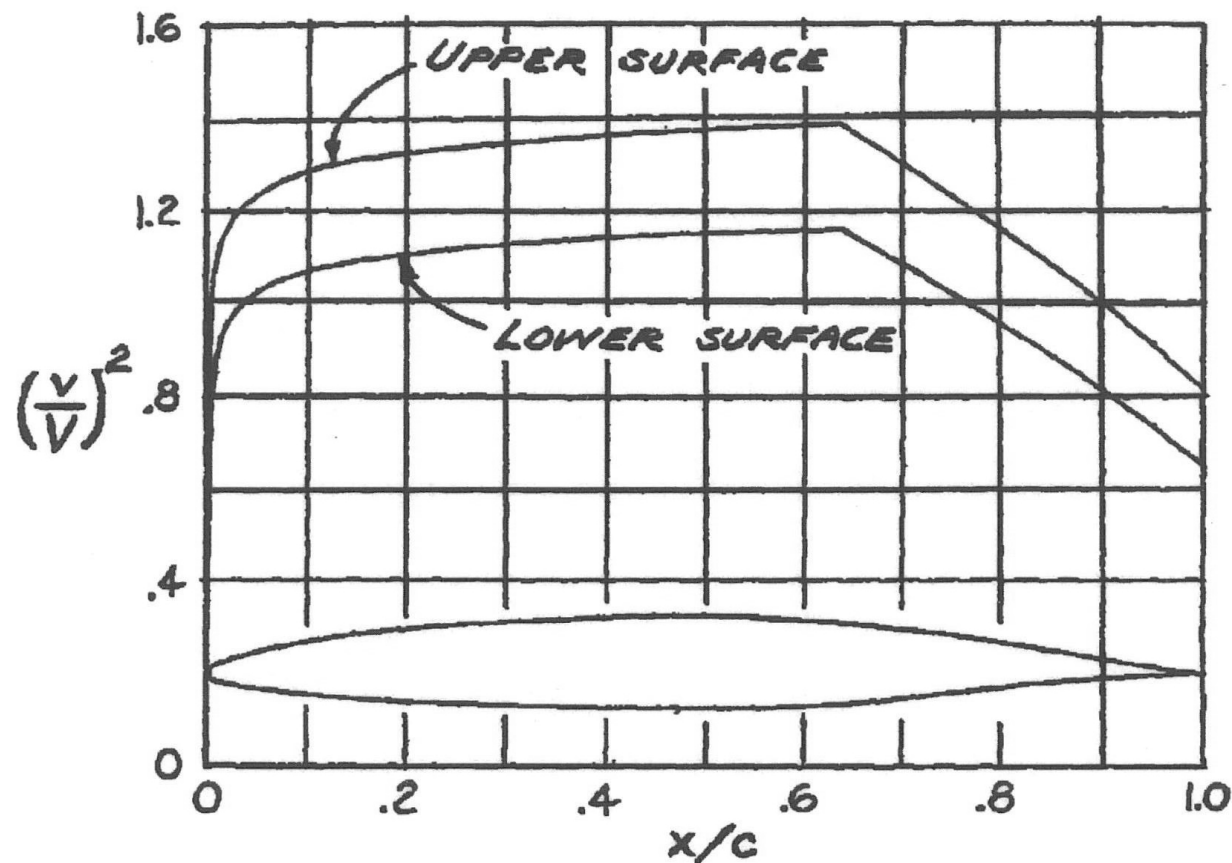
Airfoil Example: Transonic Mach Number Effects

- From classical 6 series results

NACA TN No. 1396

Fig. 1

Subsonic design pressures



NACA 66-210 $\alpha = 1.0$

Lift

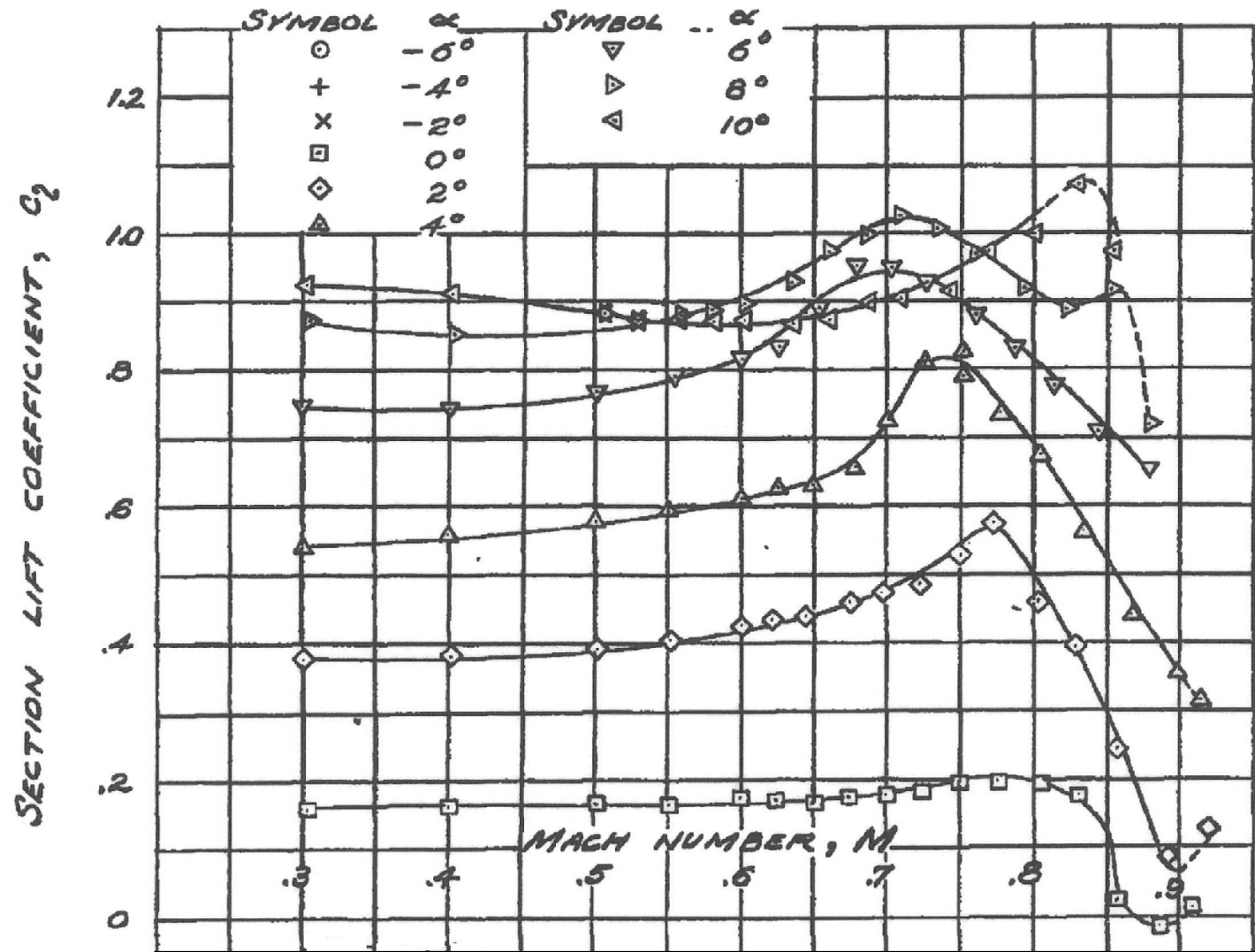


FIGURE 6: THE VARIATION OF SECTION LIFT COEFFICIENT WITH MACH NUMBER AT VARIOUS ANGLES OF ATTACK FOR THE NACA 66-210 AIRFOIL.

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

Re $\approx$ 2 Mill
 NACA Ames
 1 x 3.5 ft 2D WT
 6 inch chord foil

Drag

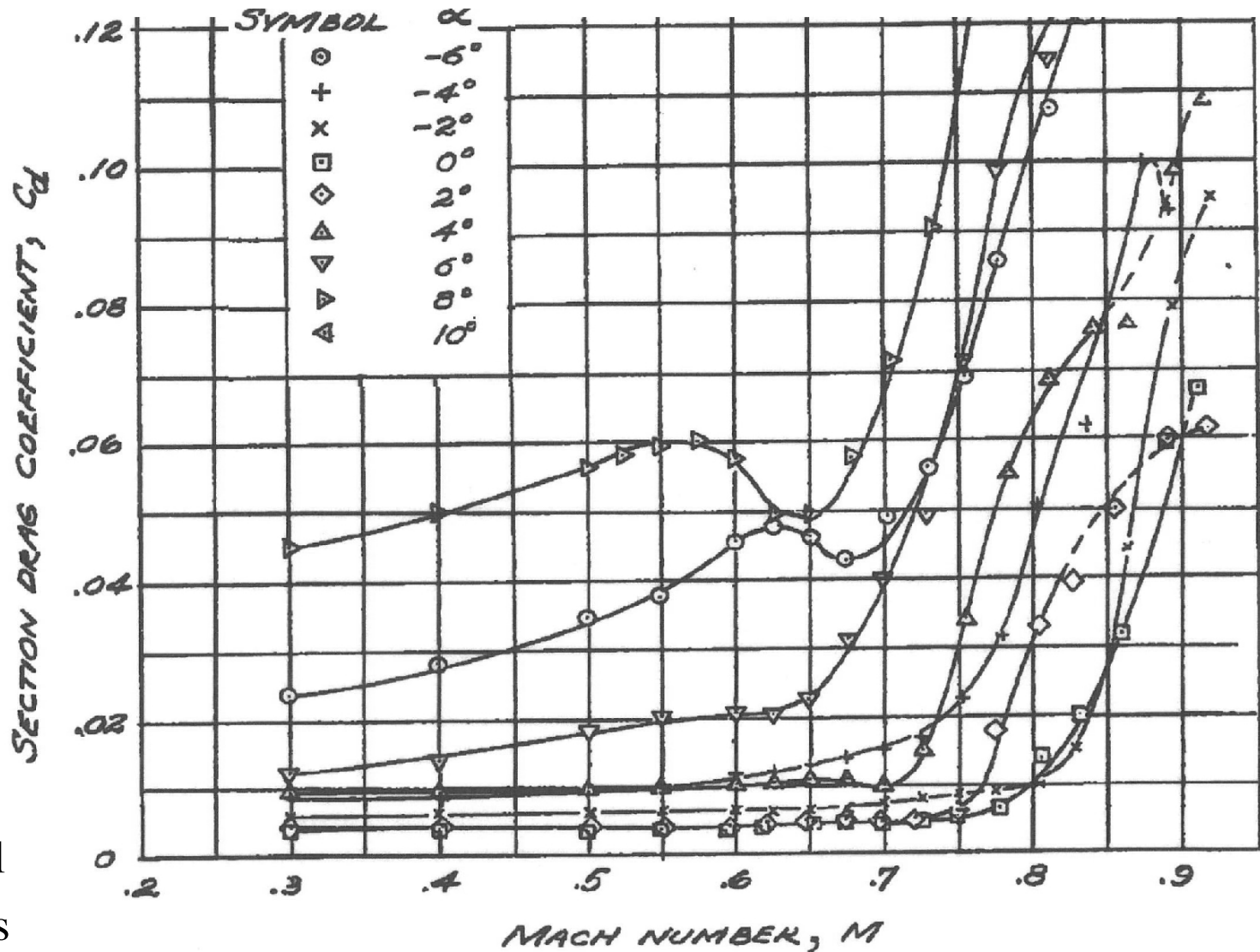


FIGURE 7.- THE VARIATION OF SECTION DRAG COEFFICIENT WITH MACH NUMBER AT VARIOUS ANGLES OF ATTACK FOR THE NACA 66-210 AIRFOIL.

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

Re $\approx$ 2 Mill
NACA Ames
1 x 3.5 ft 2D WT
6 inch chord foil

Pitching Moment: a major problem!

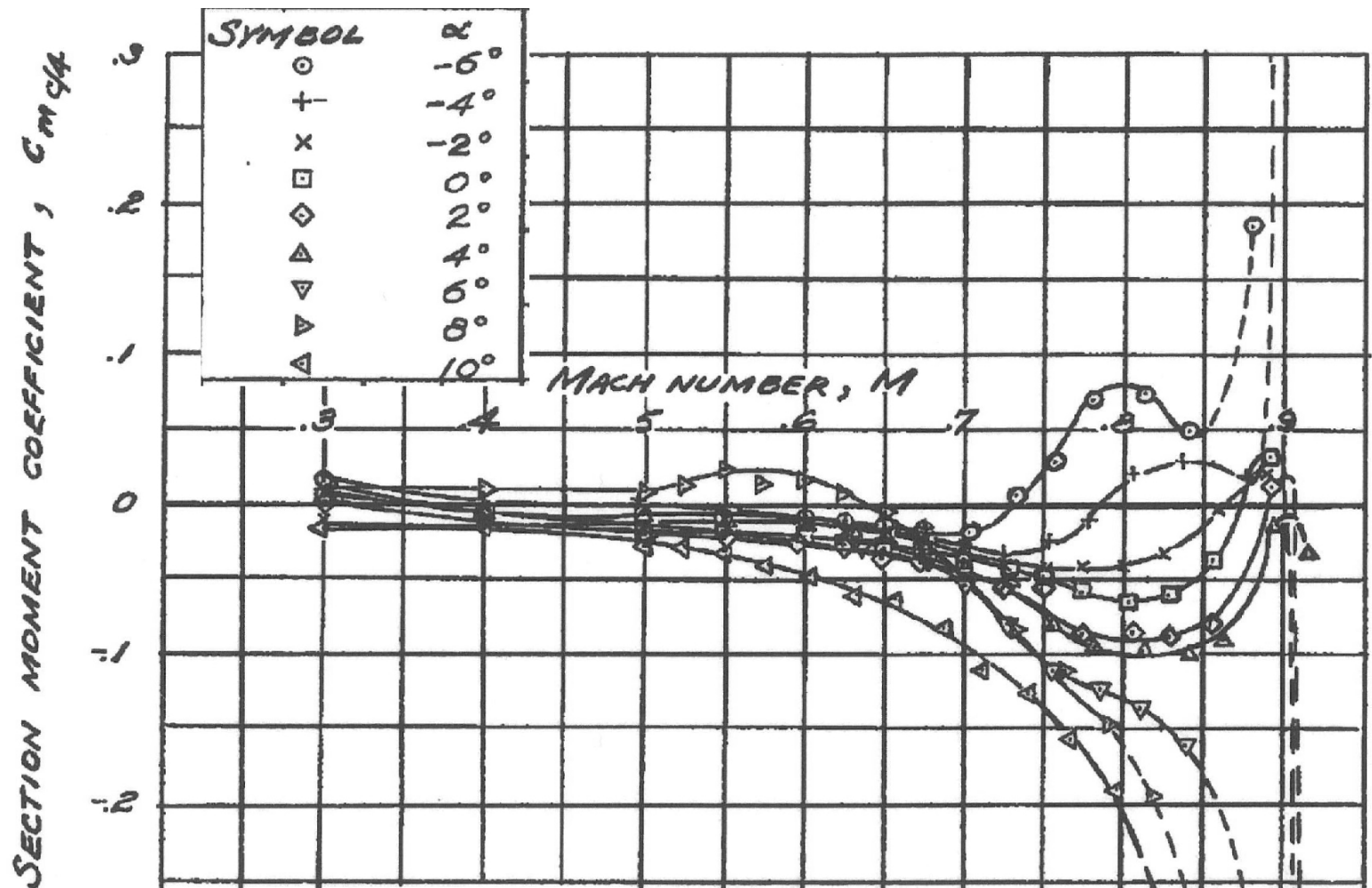


FIGURE 8.- THE VARIATION OF SECTION MOMENT COEFFICIENT WITH MACH NUMBER AT VARIOUS ANGLES OF ATTACK FOR THE NACA 66-210 AIRFOIL.

Re $\approx$ 2 Mill

NACA Ames

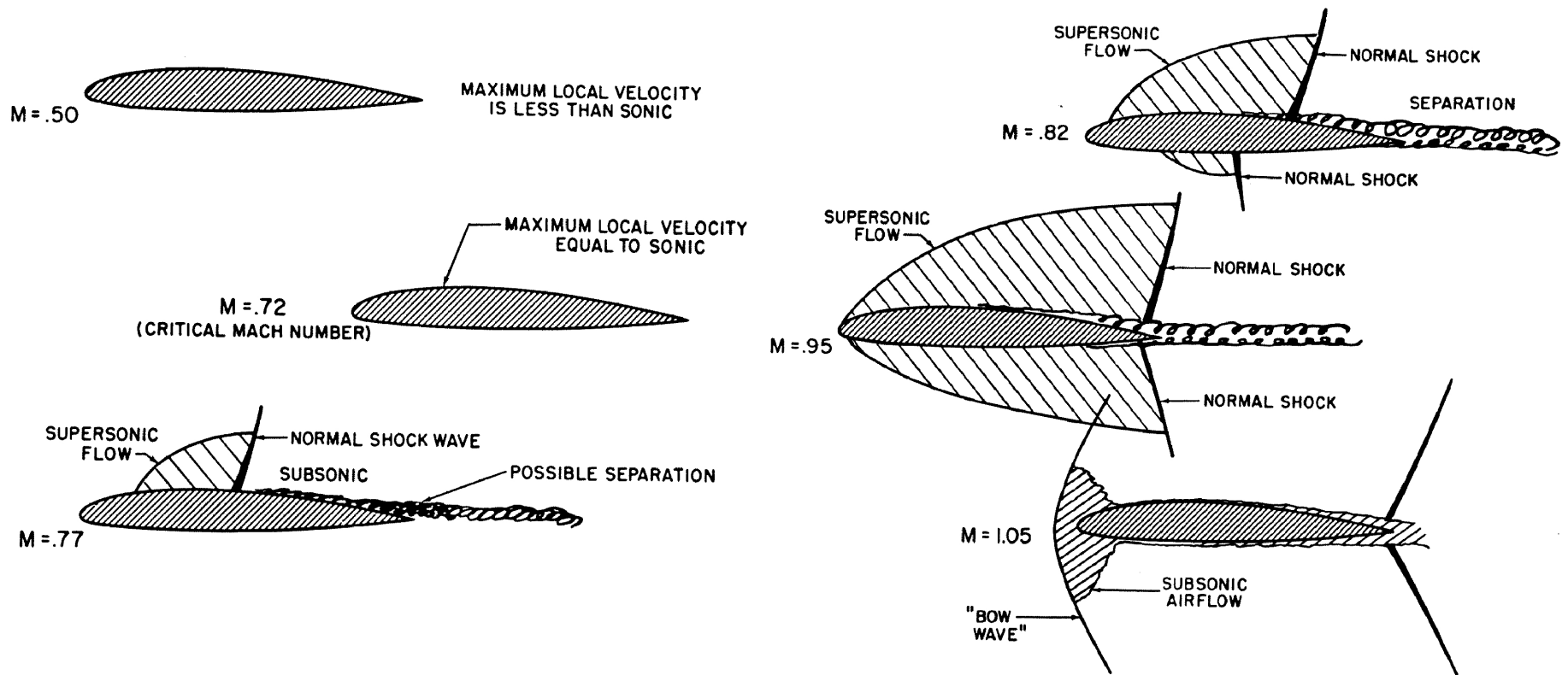
1 x 3.5 ft 2D WT

6 inch chord foil

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

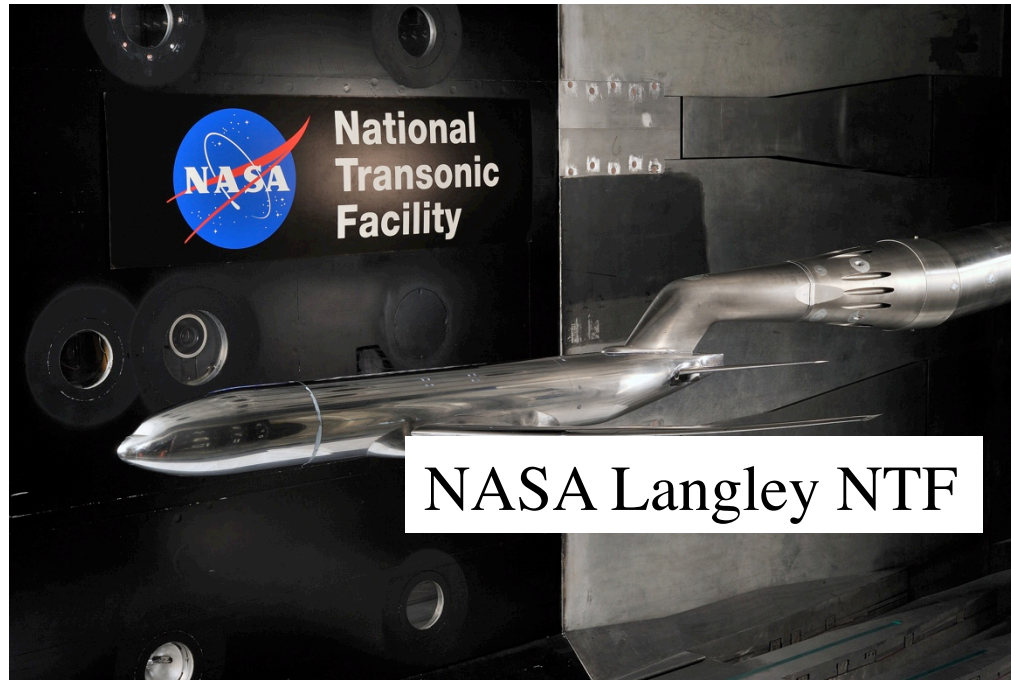
Wall Interference Solution 2: Porous Wall



The AEDC 4T, Tullahoma, TN

Another critical testing issue: Support Interference

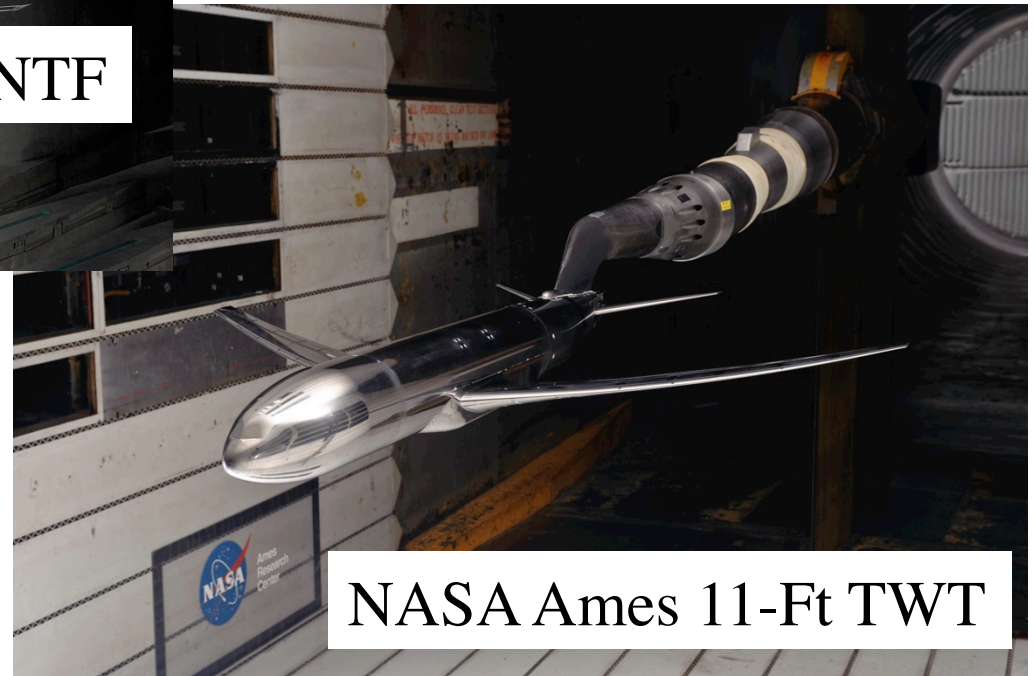
A Current focus: The NASA Common Research Model,
AIAA 2008-6919



NASA Langley NTF

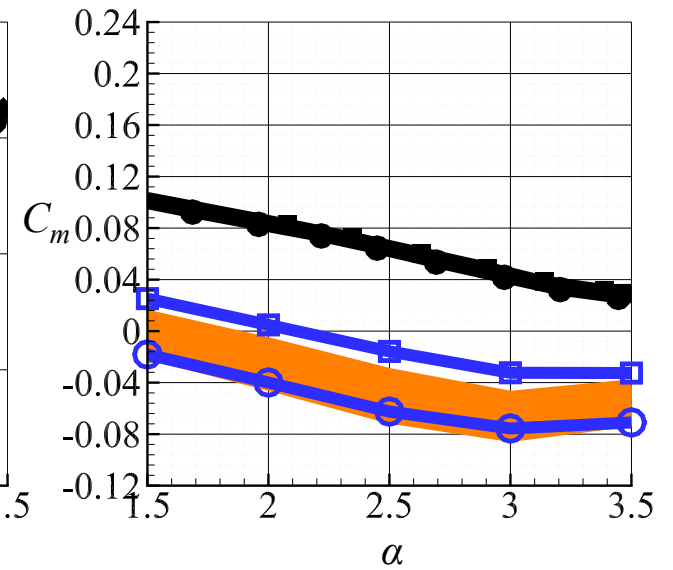
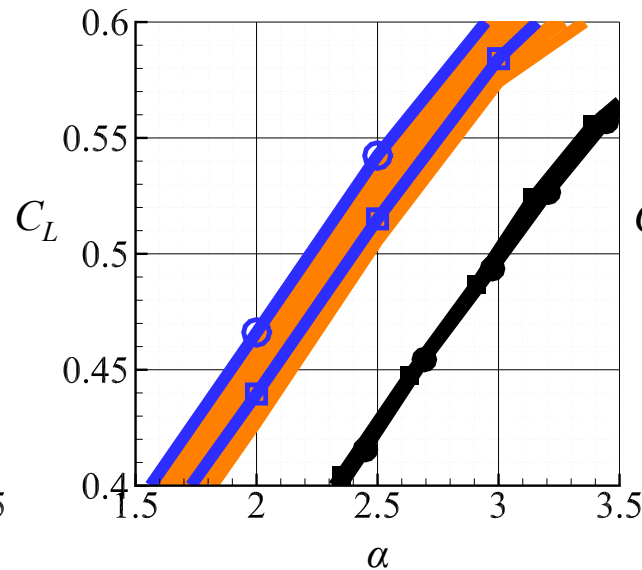
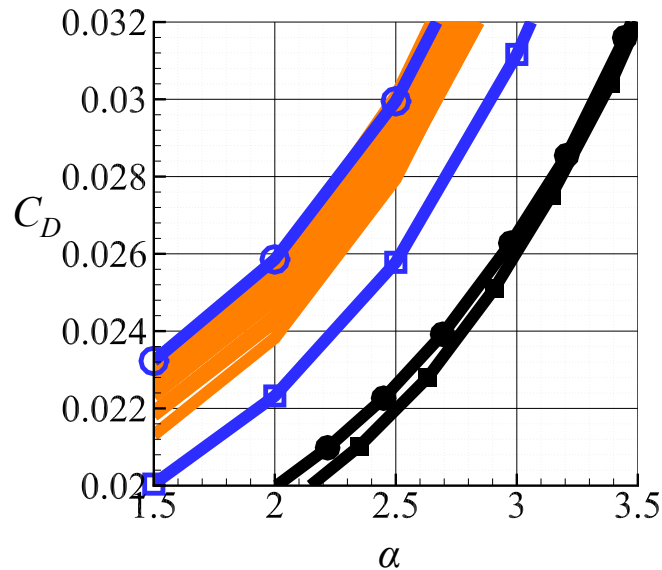
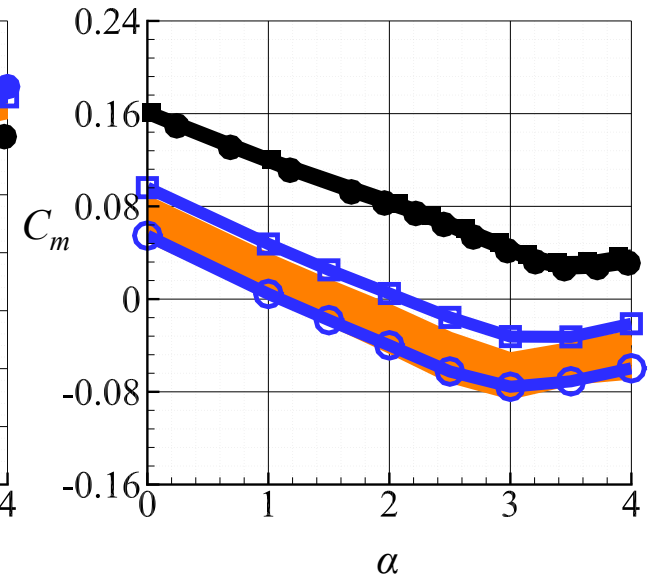
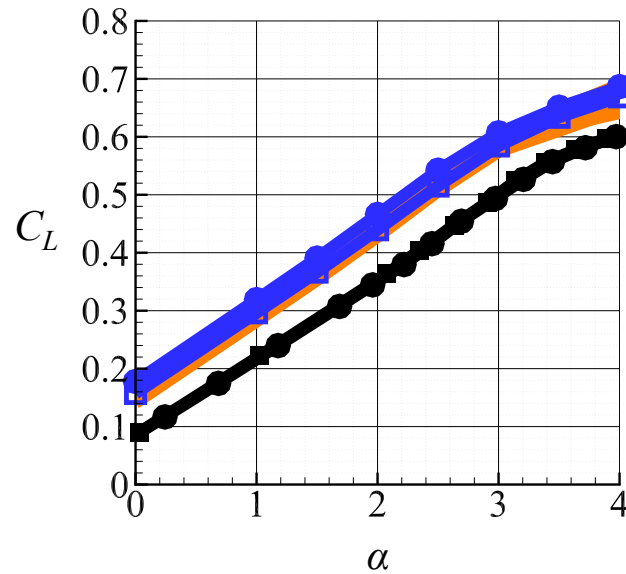
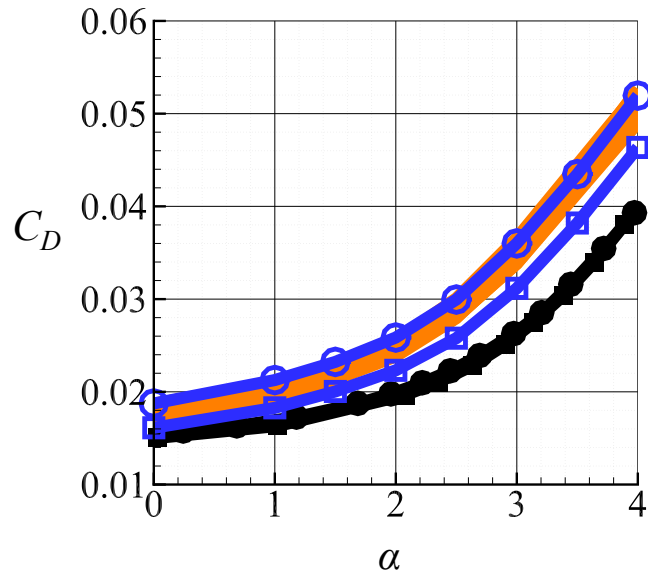
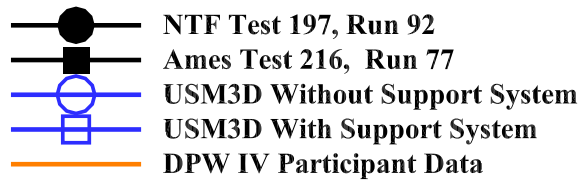
Note model support
system used to test the
model

Photos courtesy of Melissa
B. Rivers, NASA LaRC
AIAA 2012-0707



NASA Ames 11-Ft TWT

The effect of including the support system(?) on computational results



From AIAA 2012-0707, courtesy of Melissa B. Rivers

The Next Problem: Flow Similarity

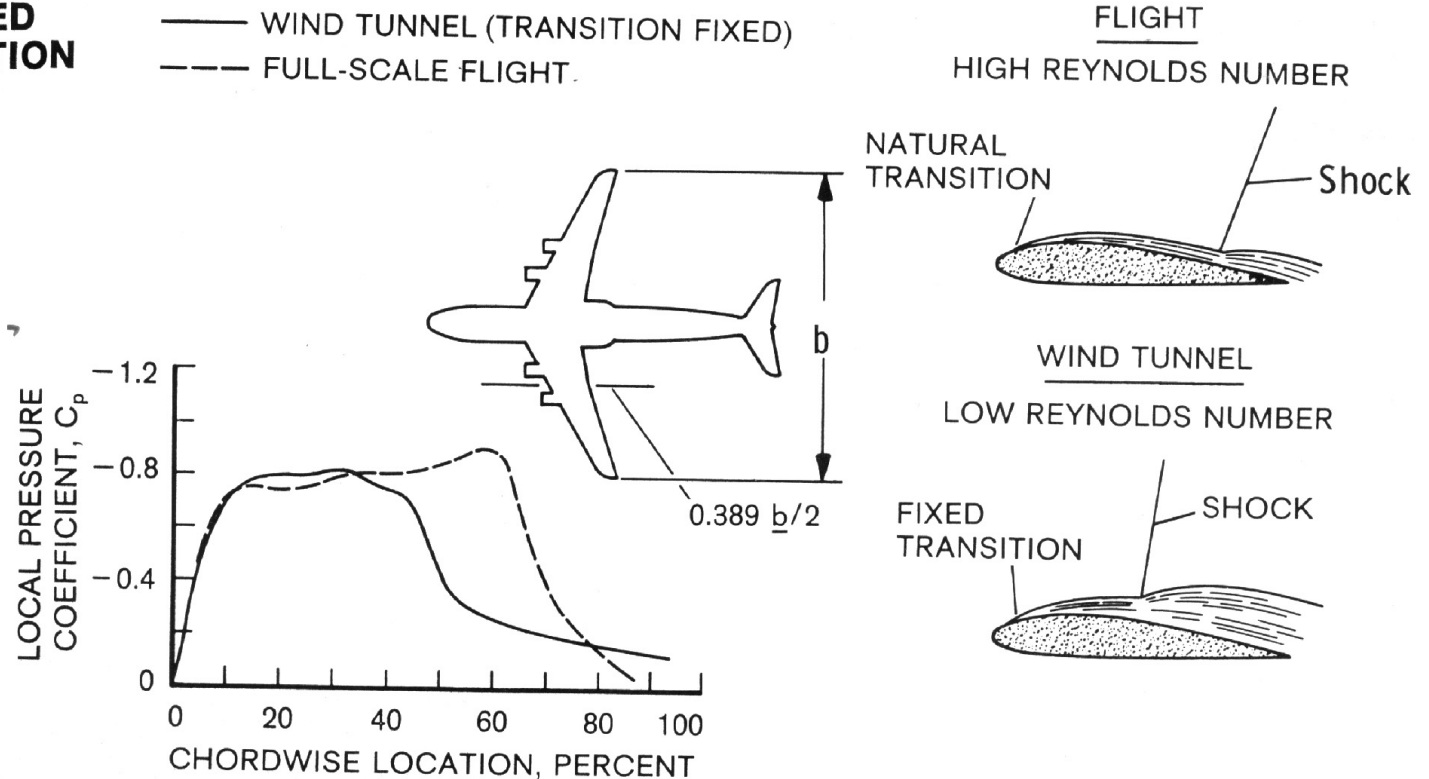
- particularly critical at transonic speed -

- *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

SHOCK-INDUCED FLOW SEPARATION

**The crux
of the
problem**



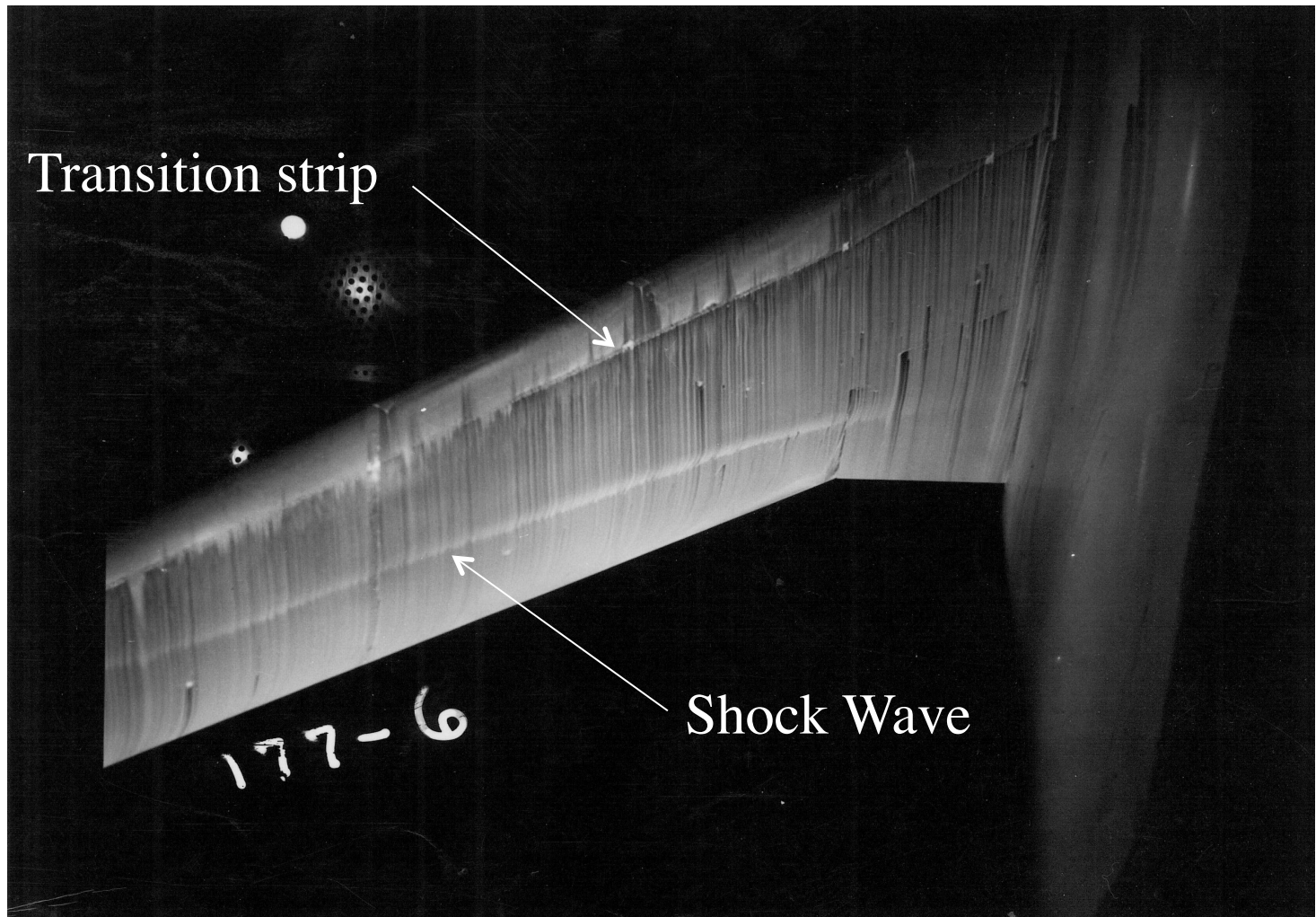
“The Need for developing a High Reynolds Number Transonic WT”
Astronautics and Aeronautics, April 1971, pp. 65-70

To Help Match Reynolds Number

- *Pressure* Tunnels
- *Cold* Tunnels
 - Keeps dynamic pressure “reasonable”
 - Implies acceptable balance forces
 - Also reduces tunnel 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 $\mu = T^{0.9}$

$$Re = \frac{\sqrt{\gamma} p M L}{\sqrt{R} T^{1.4}}$$

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

Balance forces are related to, say, $N = q S C_L$

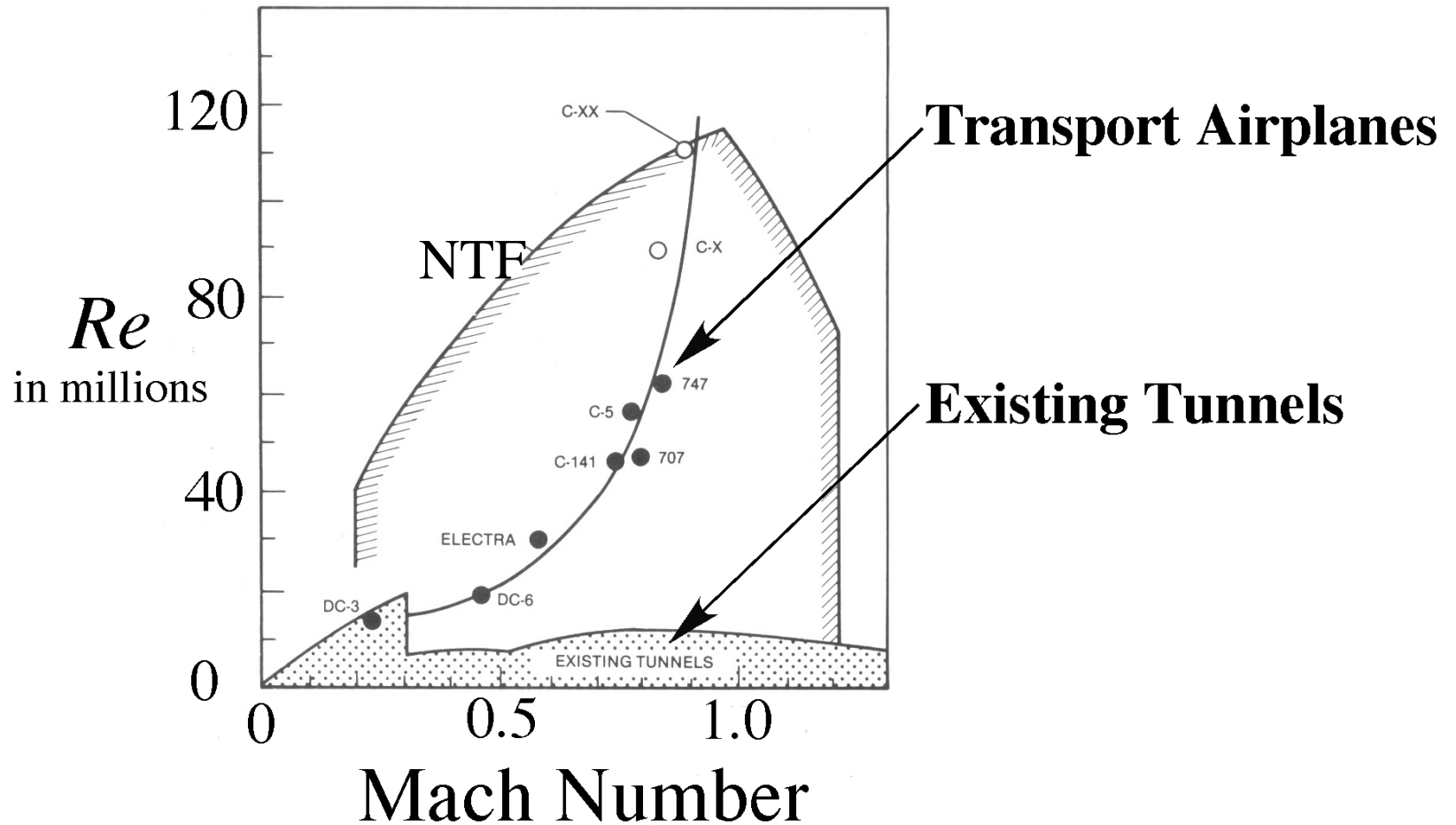
$$q = \frac{\gamma}{2} p M^2$$

Reducing T allows Re increase without huge balance forces
- note: q proportional to p , as shown above

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

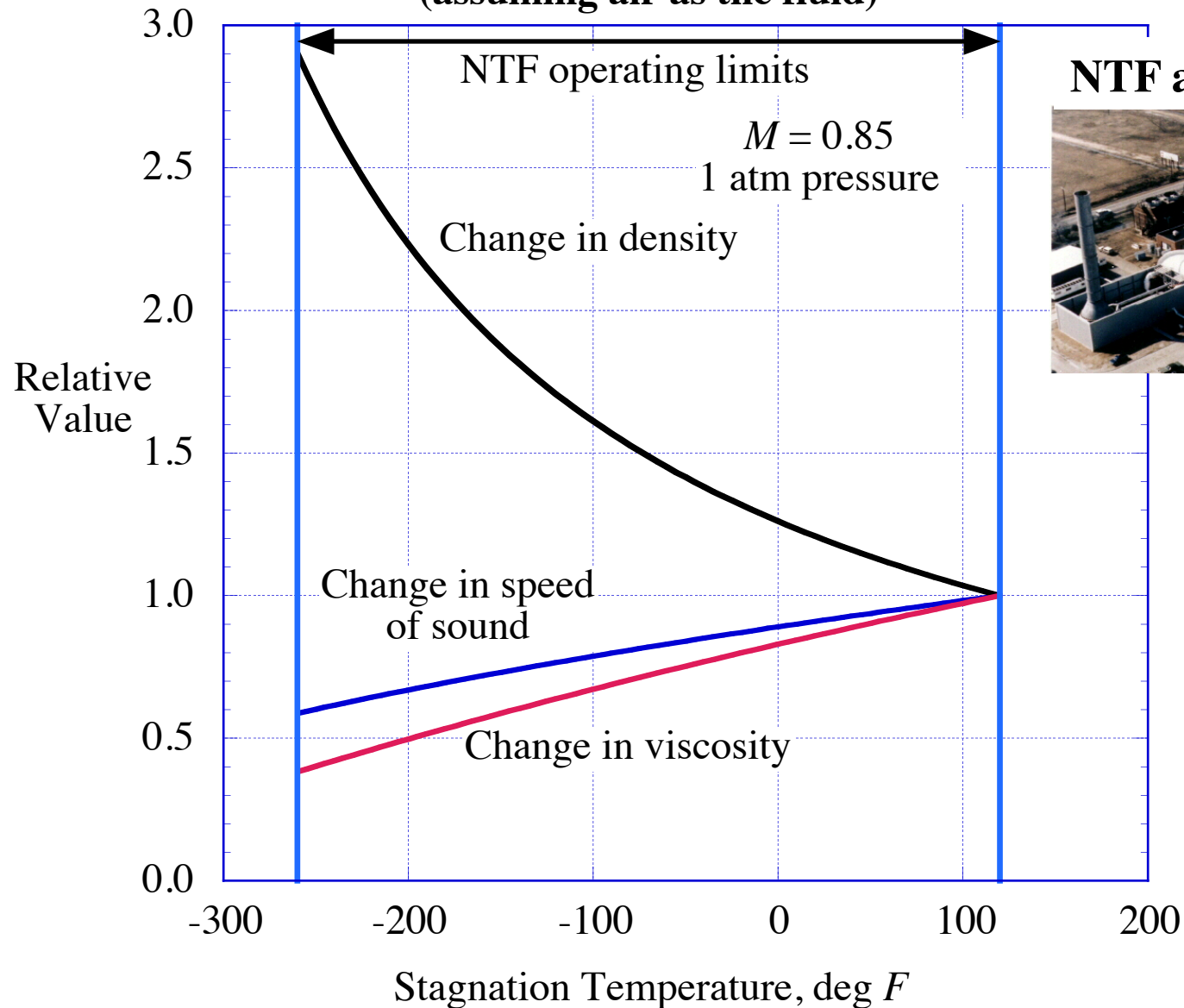


Feb. 1982

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

Cryo effects on fluid properties

Temperature effects on fluid properties
(assuming air as the fluid)

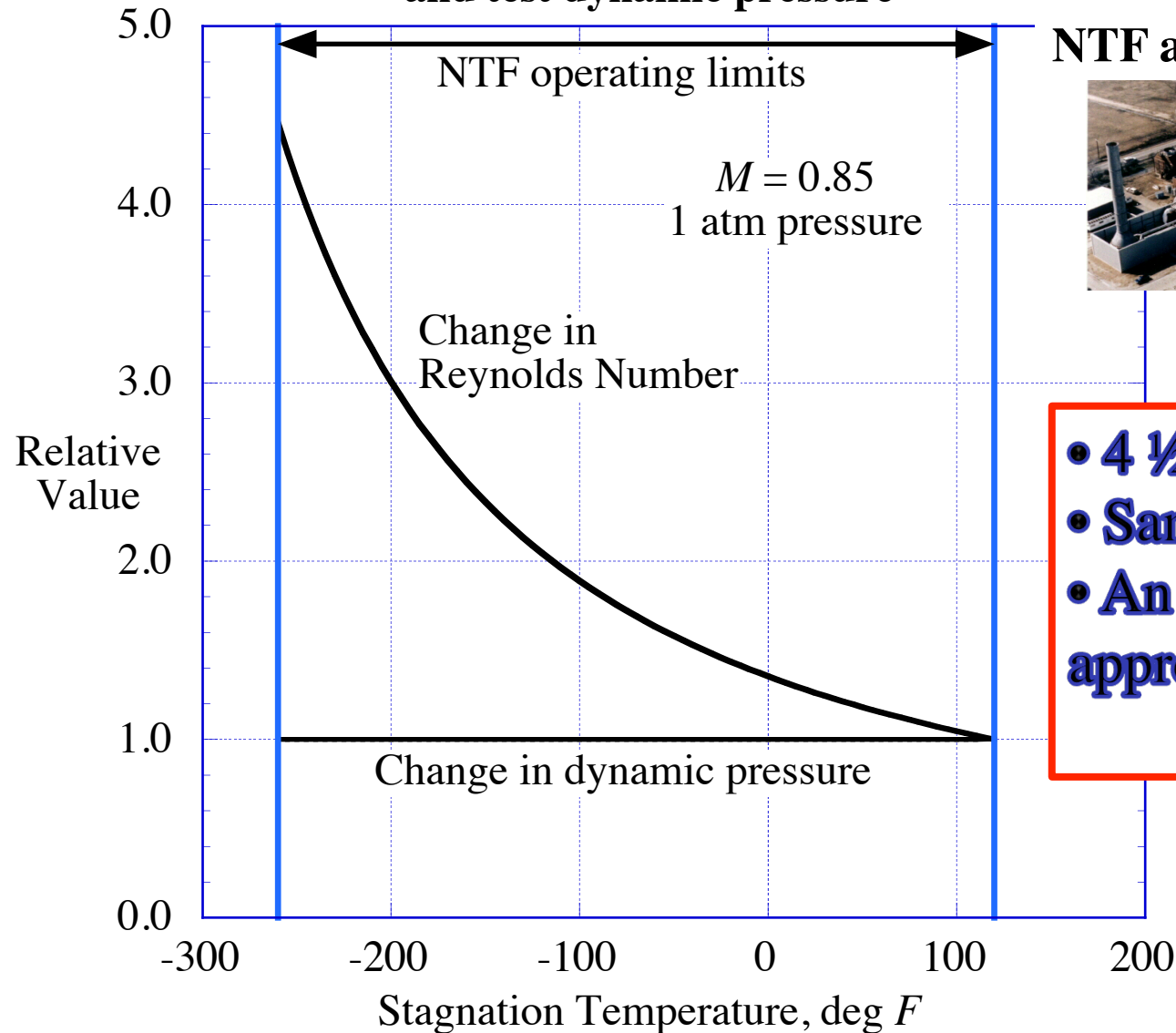


NTF at NASA LaRC



Cryo Effects on Re and q

Temperature effects on Reynolds Number
and test dynamic pressure



NTF at NASA LaRC



- 4 ½ times the Re
- Same q
- An enabling approach!

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

Murman and Cole, *AIAA J.*, Vol. 9, No. 1, Jan. 1971

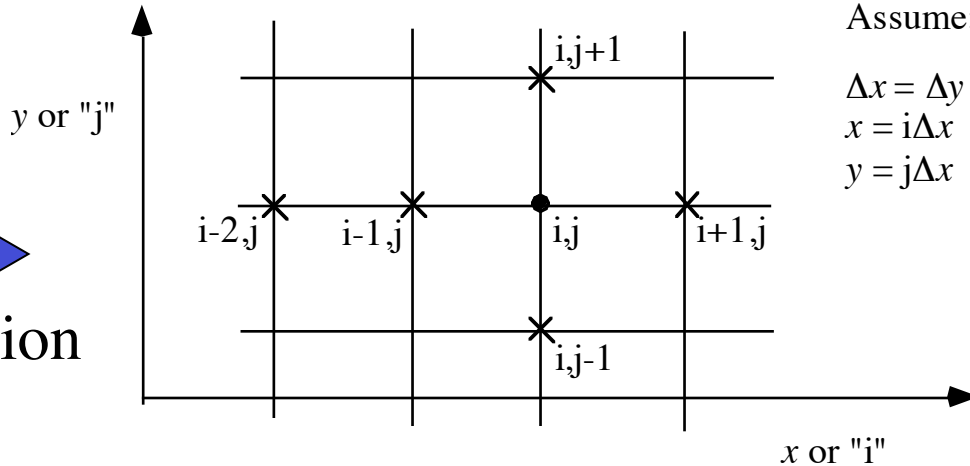
Presented at the Aerospace Sciences Mtg, NYC, Jan. 1970

The Murman-Cole Idea: Mixed Differencing

- 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
 - If 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)
 - Agreed with experimental data somewhat (and possibly fortuitously)

A few more details

Using this
notation
for the grid:



For subsonic flow:
(central difference)

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

For supersonic flow:
(upwind difference)

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

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

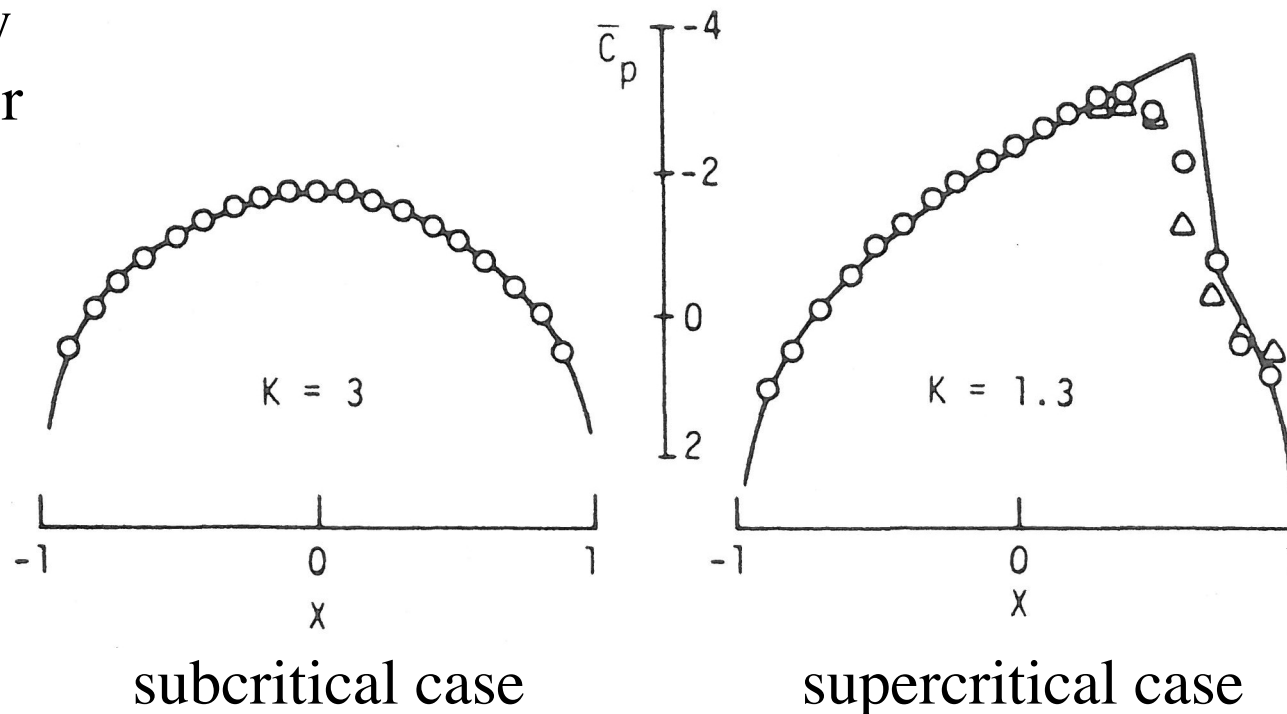
— MURMAN AND COLE (1971)

○ $Re_c \approx 2 \times 10^6$

△ $Re_c \approx 2 \times 10^6$ (L.E. ROUGHNESS)

EXPERIMENTS
KNECHTEL (1959)
 $\delta = 0.06$

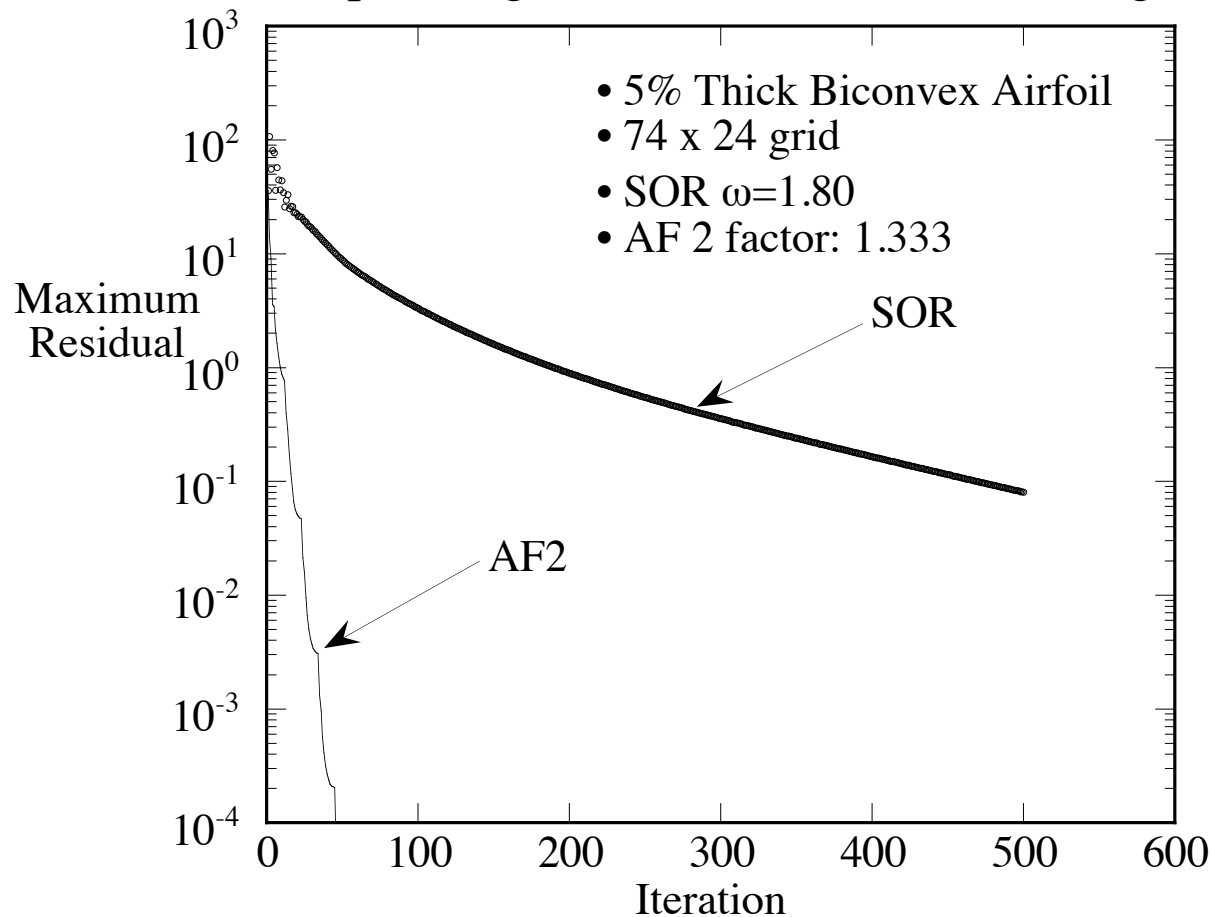
Note: K is a
similarity
parameter



from *Computational Fluid Mechanics and Heat Transfer*, 3rd Ed., by
Pletcher, Tannehill, and Anderson, CRC Press, 2013.

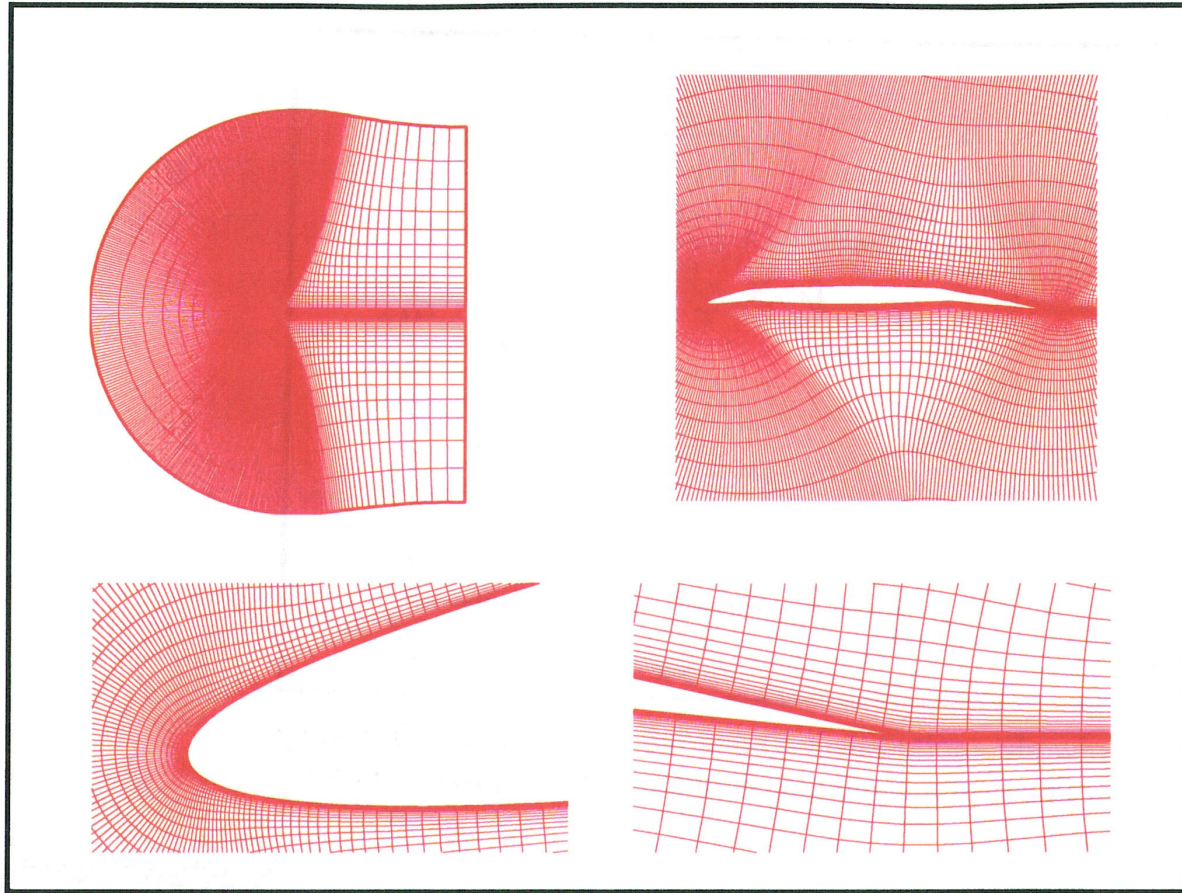
One convergence check in CFD

- “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
 - -Post processing solution visualization

References:

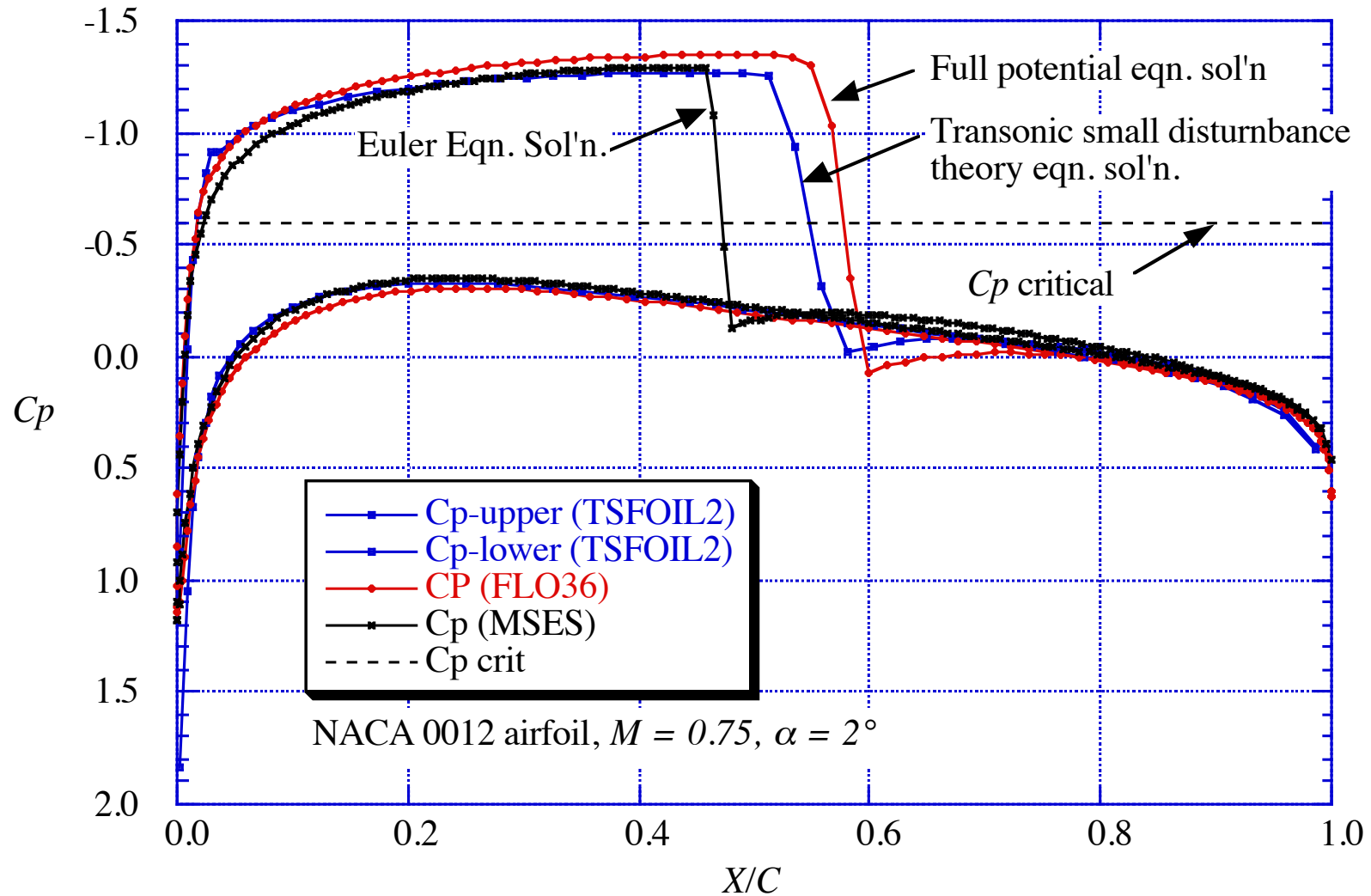
R. M. Cummings, W.H. Mason, S. A. Morton and D. R. McDaniel,
Applied Computational Aerodynamics, Cambridge Univ. Press, 2015

J. Blazek, *Computational Fluid Dynamics: Principles and Applications*,
Elsevier, 2005

Can now use computational simulations as a numerical wind tunnel

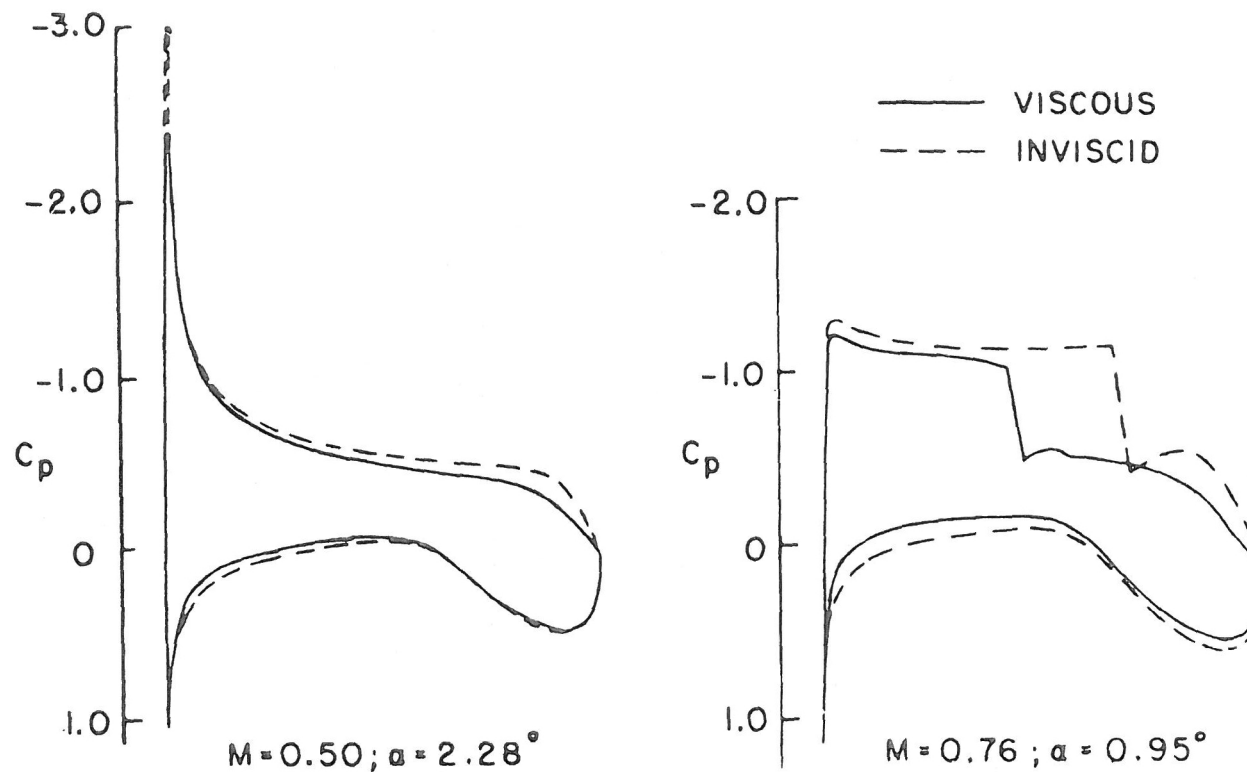
- Especially in 2D, many codes include grid generation, and can be used to investigate airfoil characteristics and modifications
- **TSFOIL2** is an available and free transonic code, the next chart illustrates its accuracy
- **FLO36** was the ultimate full potential code
 - By Antony Jameson, a key 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 -



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

VISCOUS EFFECTS ON PRESSURE DISTRIBUTIONS
FOR A SUPERCRITICAL AIRFOIL



Whitcomb, "Review of NASA Supercritical Airfoils," ICAS 74-10, Aug. 1974
(*First public description of supercritical airfoils*)

Transonic Applications

Review:

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

Now:

- 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

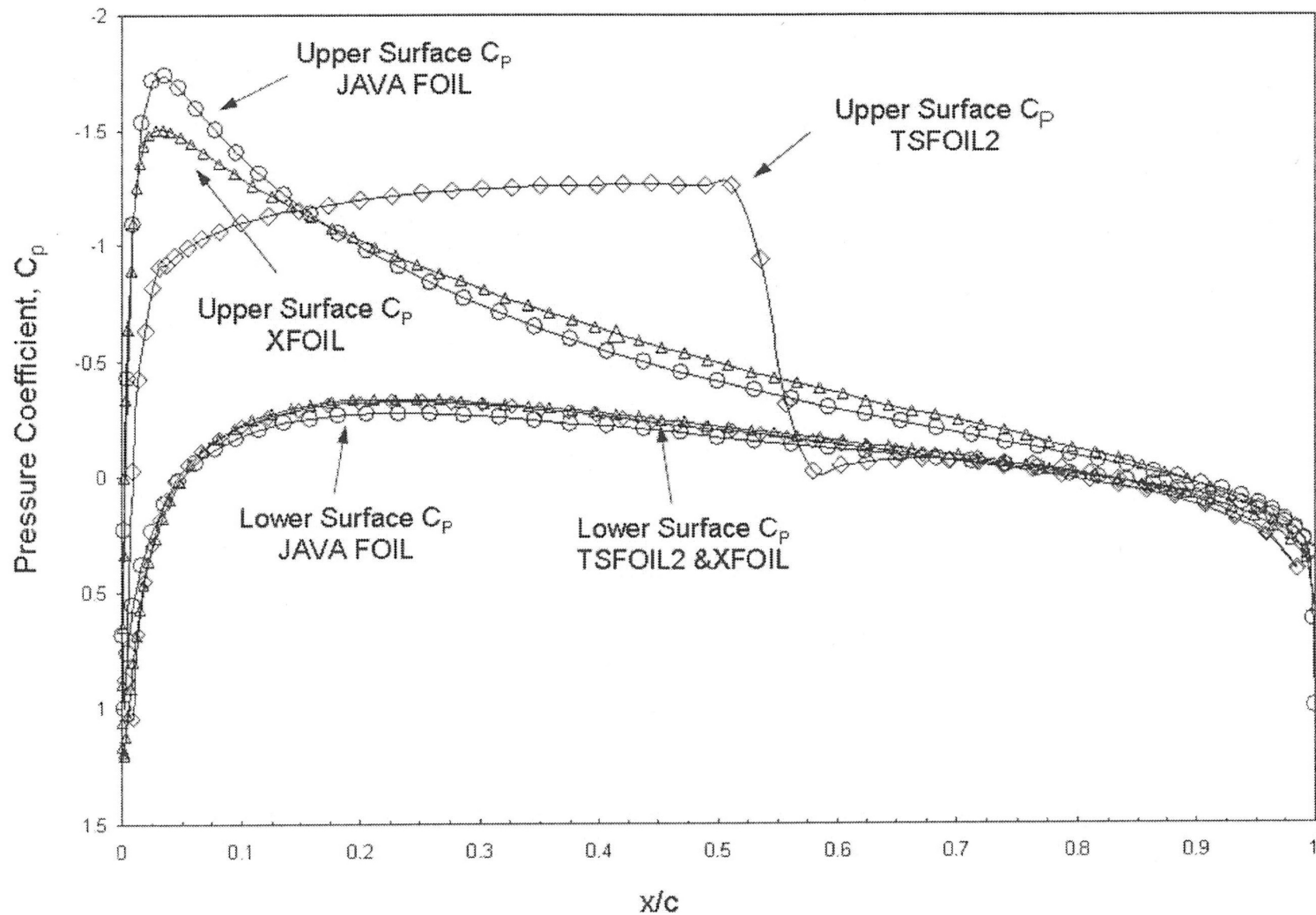
Image # EL-2001-00478

Subsonic Linear Theory, Even with the Compressibility Correction, Can't Predict Transonic Flow!

From Desta Alemayhu

Pressure Coefficient (C_p) vs. X for NACA 0012

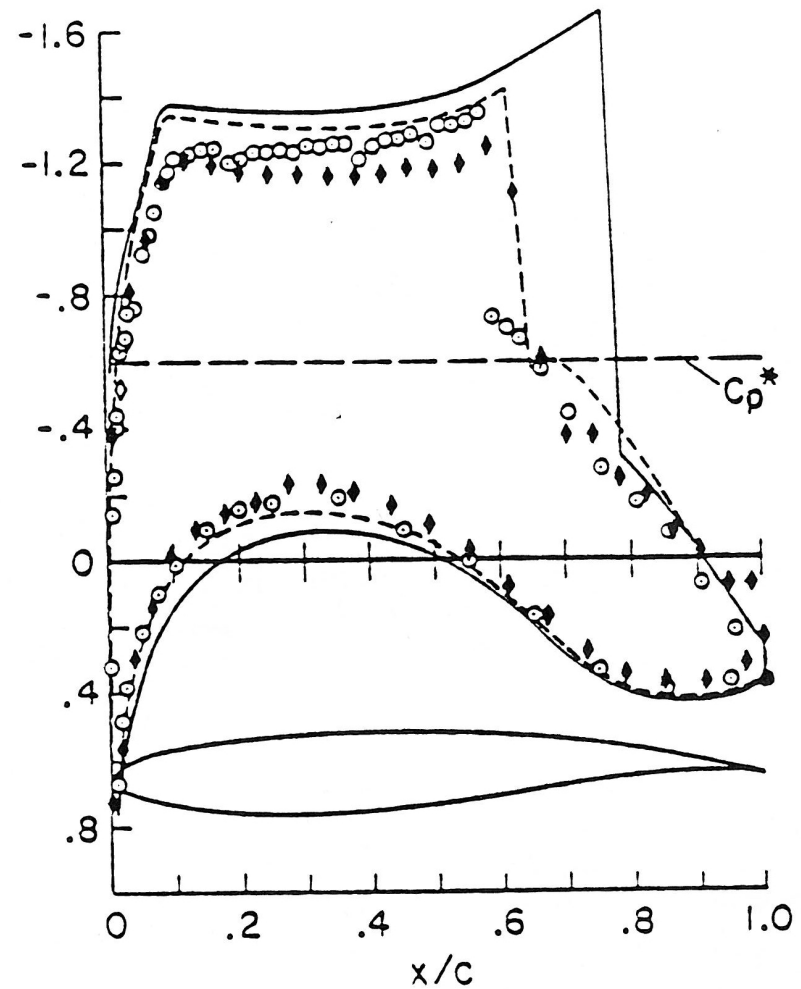
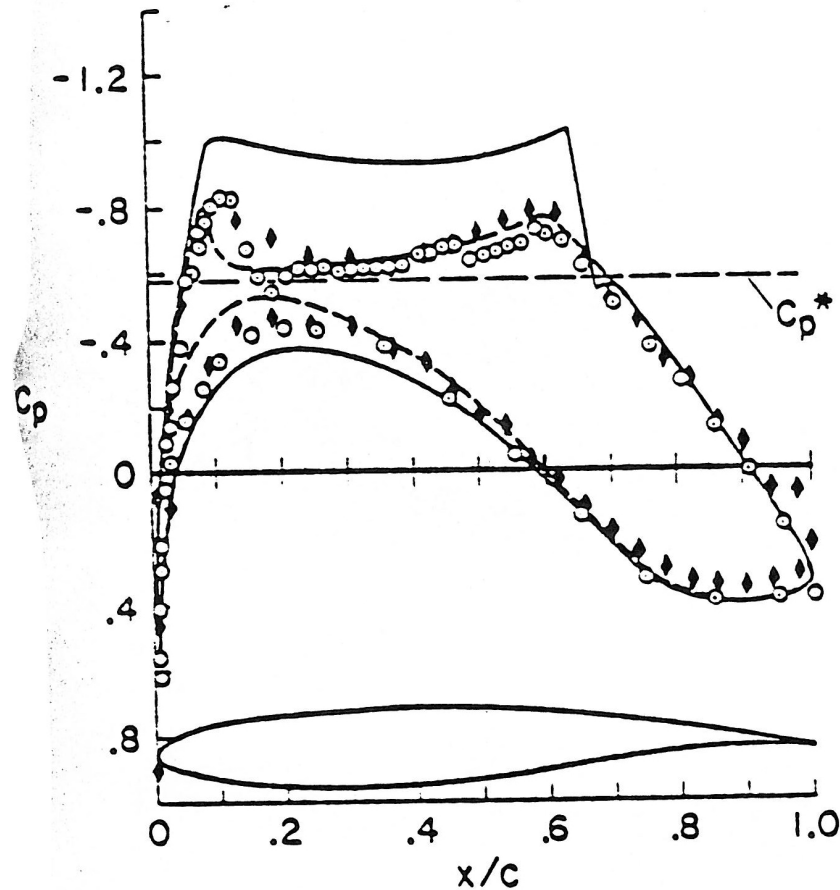
Mach = 0.75 and $\alpha = 2^\circ$



AN EXAMPLE OF PREDICTING AIRFOIL FLOWS (REF. 3)

	M_∞	α°	C_D	C_L
○ EXP	0.755	0.12	0.012	0.38
◆ N-S	"	"	0.012	0.42
— INV	"	"	0.004	0.68
-- INV	0.740	-0.77	0.001	0.43

	M_∞	α°	C_D	C_L
○ EXP	0.747	2.96	0.037	0.87
◆ N-S	"	"	0.040	0.86
— INV	"	"	0.064	1.24
-- INV	0.732	2.07	0.025	1.05



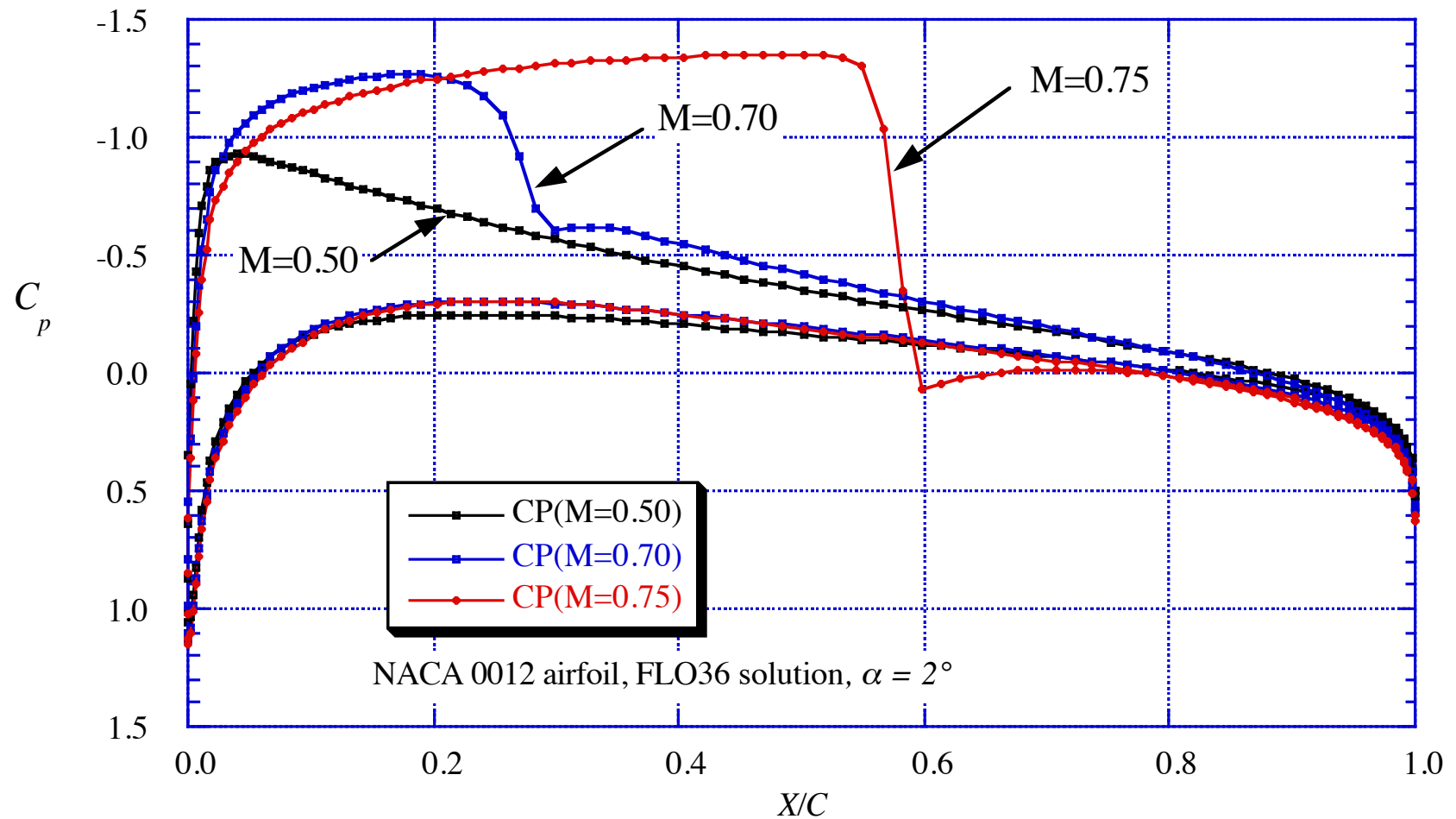
Illustrates "tricks" used to get calculations to agree with test data

REVIEW: Obtaining CFD solutions

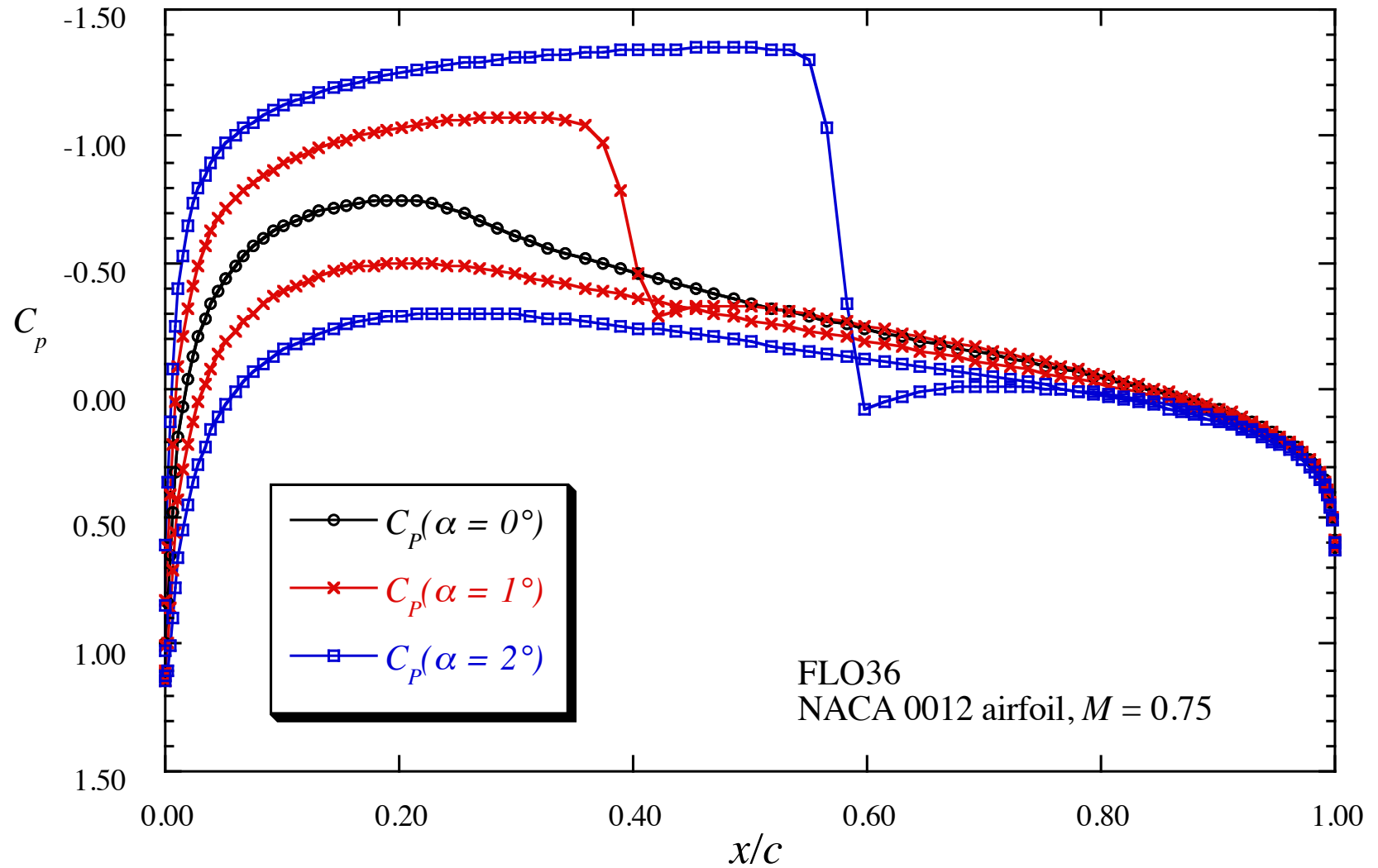
- Grid generation
- Flow solver
 - Typically solving 100,000s (or millions in 3D) of simultaneous *nonlinear* algebraic equations
 - An iterative procedure is required, and it's not even guaranteed to converge!
 - Requires more attention and skill than linear theory methods
- Flow visualization to examine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

Airfoils

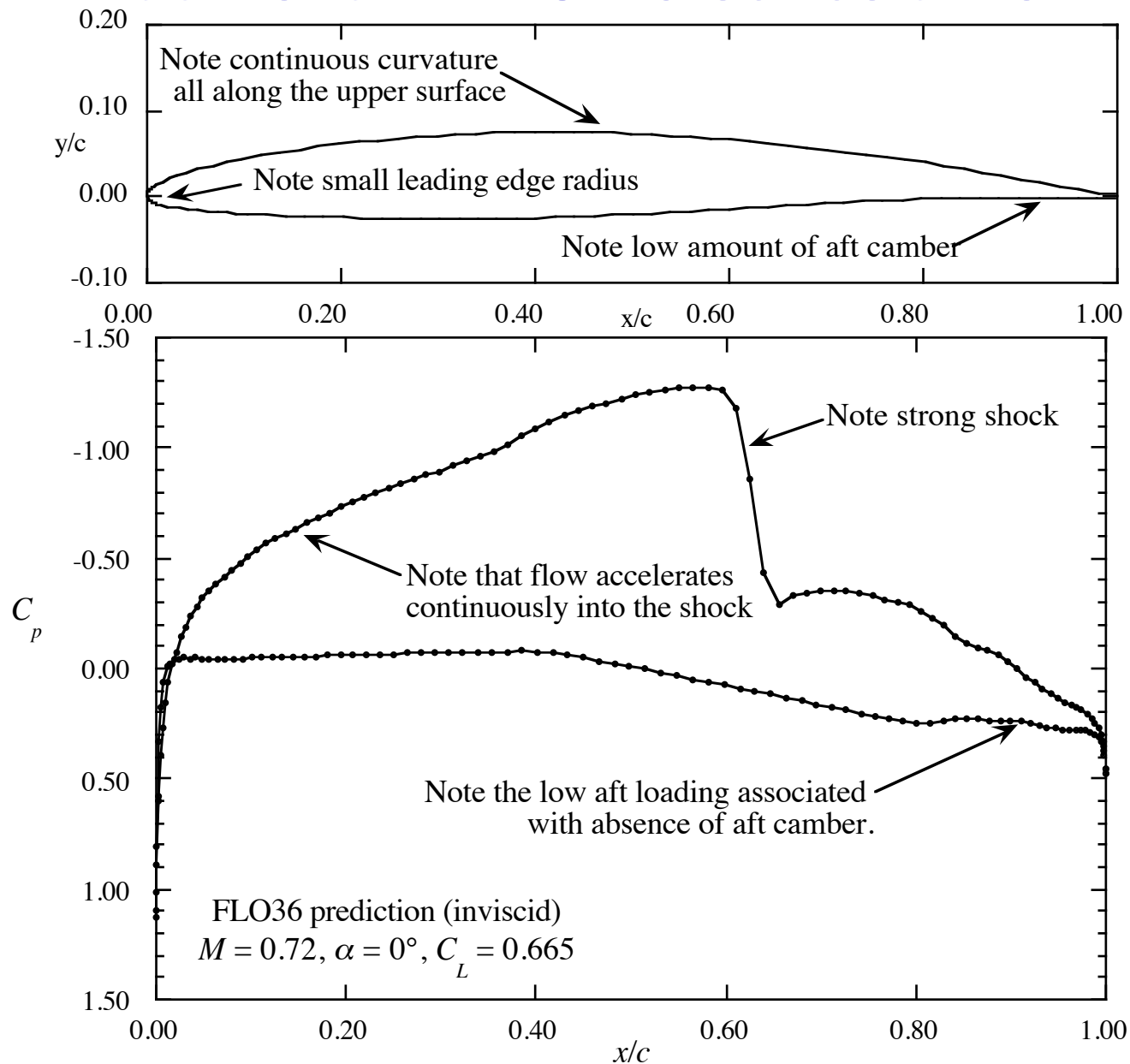
Mach number effects: NACA 0012



Angle of attack effects: NACA 0012

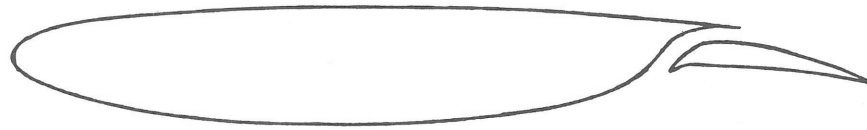


“Traditional” NACA 6-series airfoil



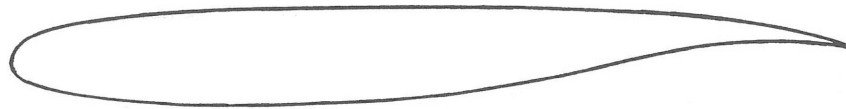
A “new” airfoil concept - from Whitcomb

1964



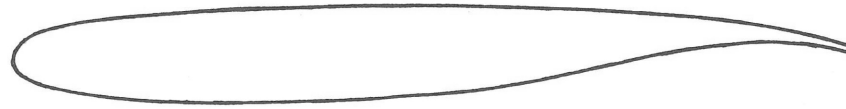
Slotted (1964)

1966



Integral (1966)

1968



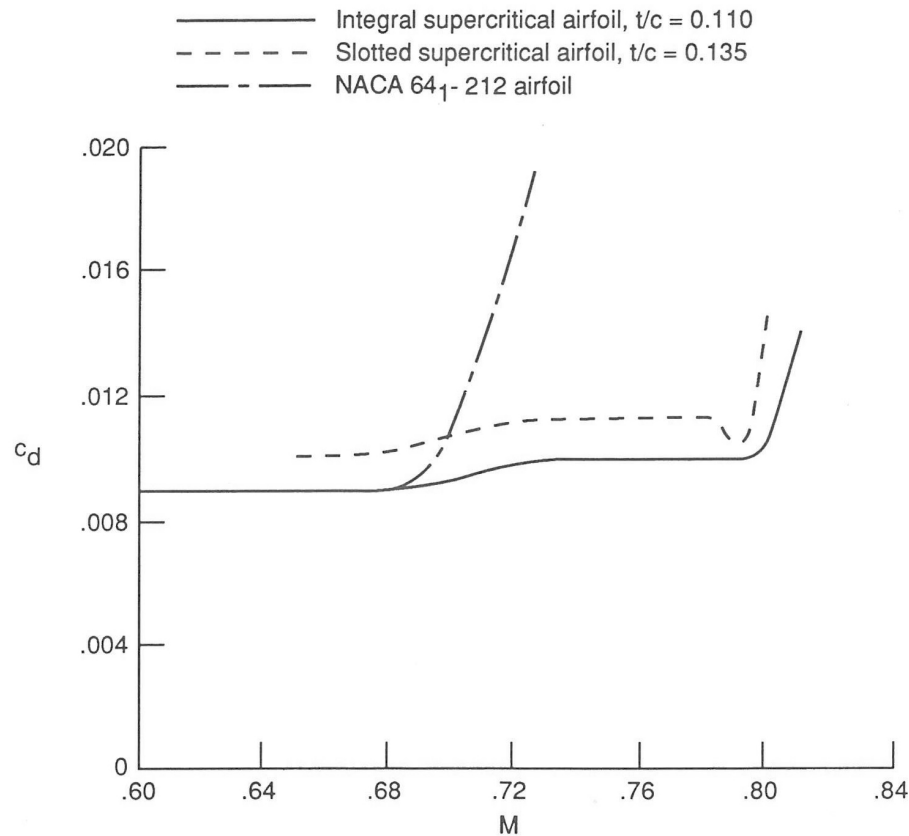
Integral with thickened trailing edge (1968)

Progression of the Supercritical airfoil shape

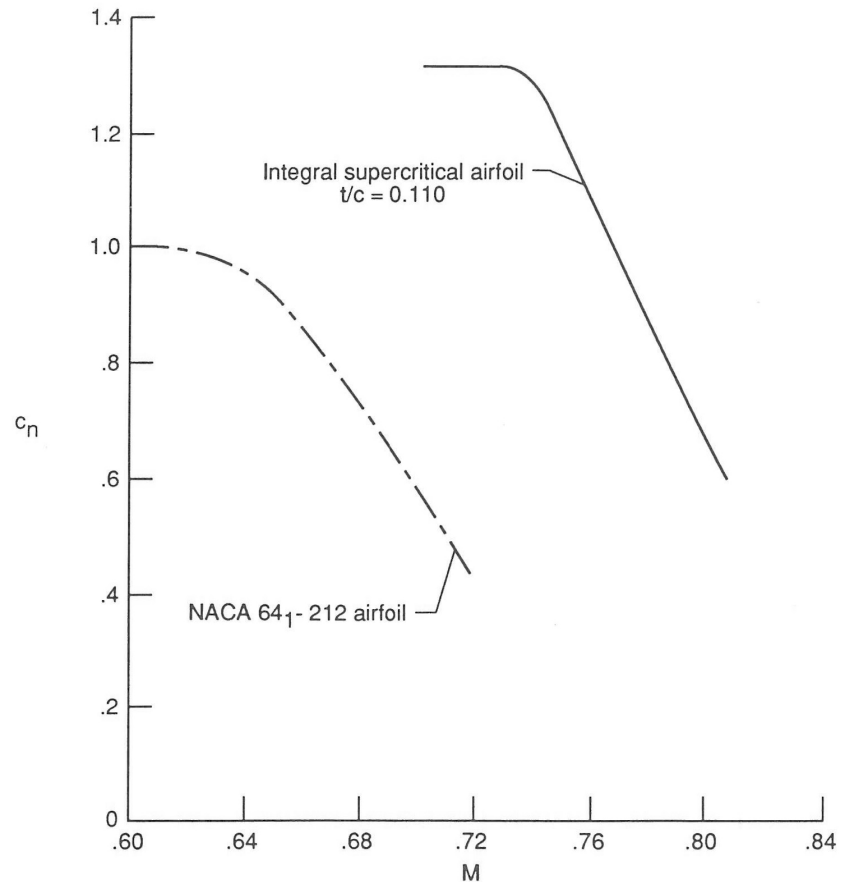
“NASA Supercritical Airfoils,” by Charles D. Harris, NASA TP 2969, March 1990

What the supercritical concept achieved

Section drag at $C_N = 0.65$

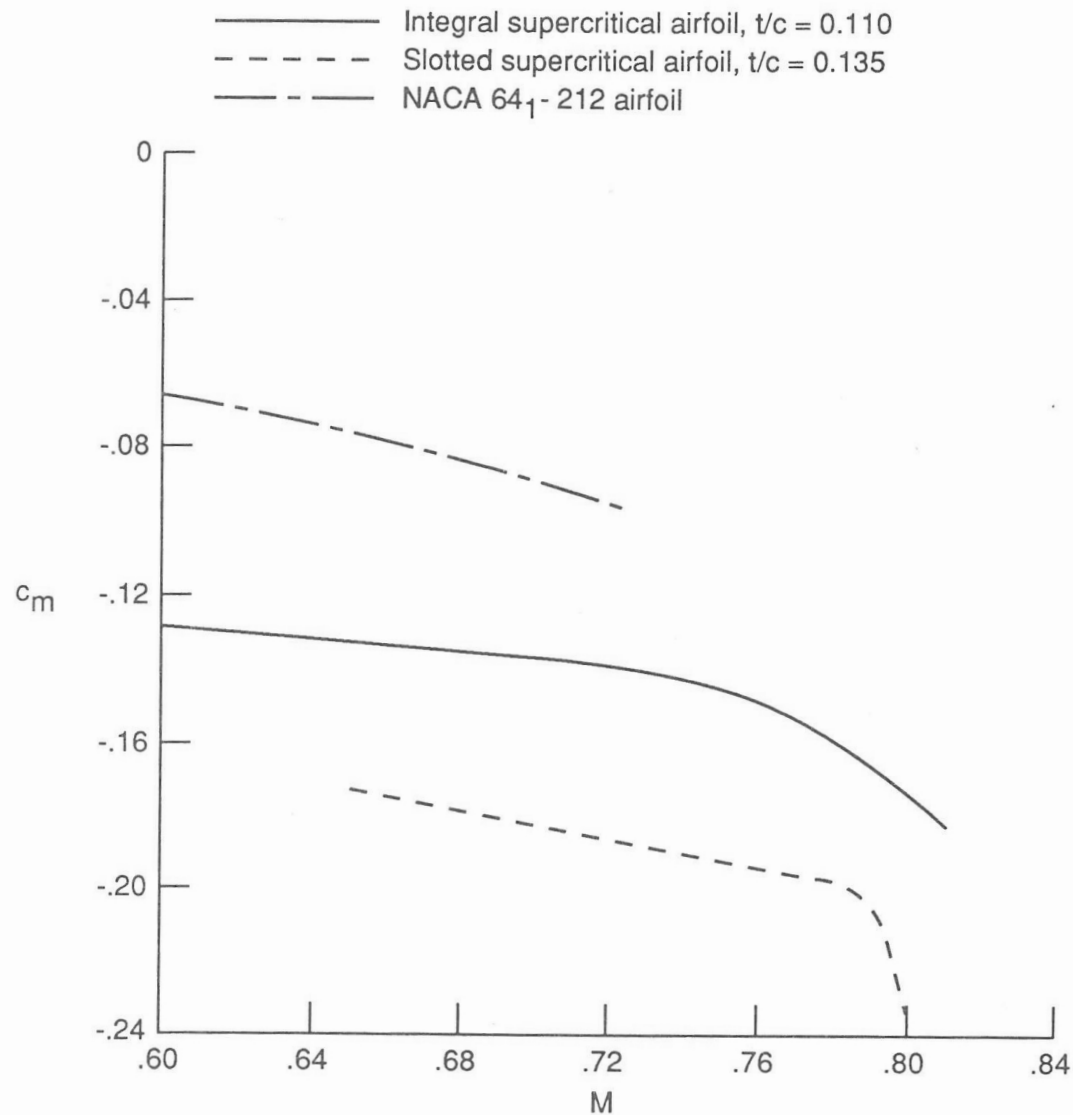


Force limit for onset of upper-surface boundary layer separation



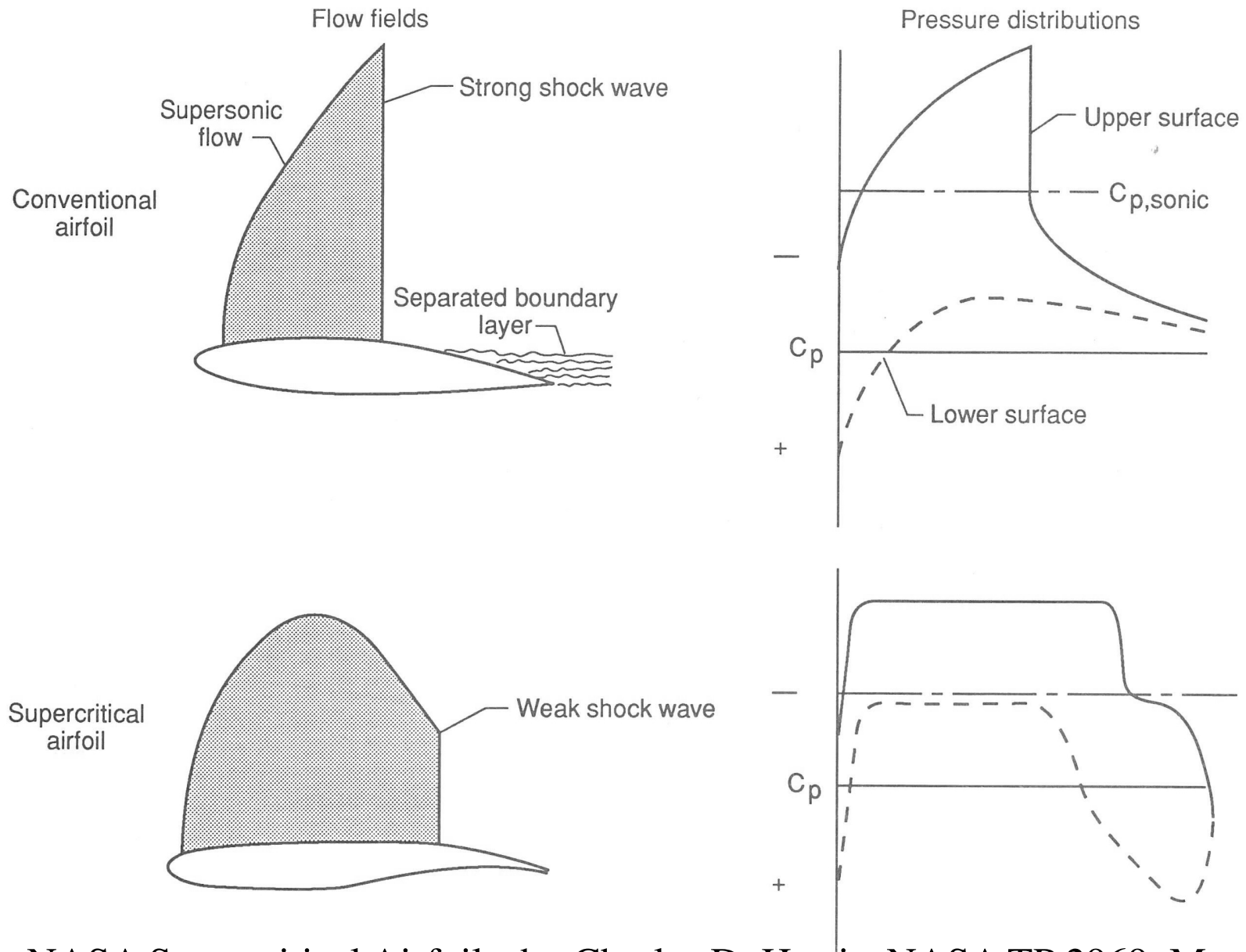
From "NASA Supercritical Airfoils," by Charles D. Harris, NASA TP 2969, March 1990

And the Pitching Moment



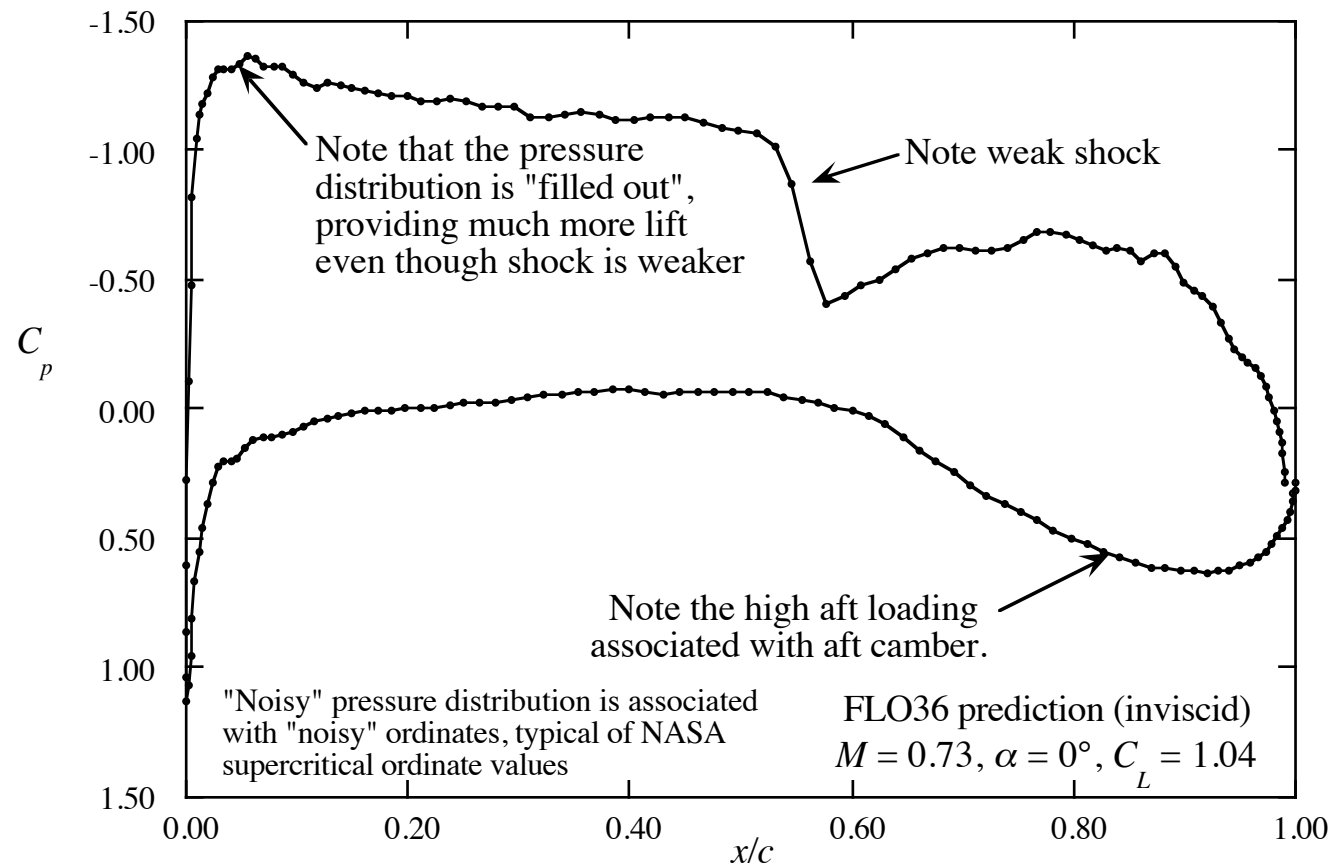
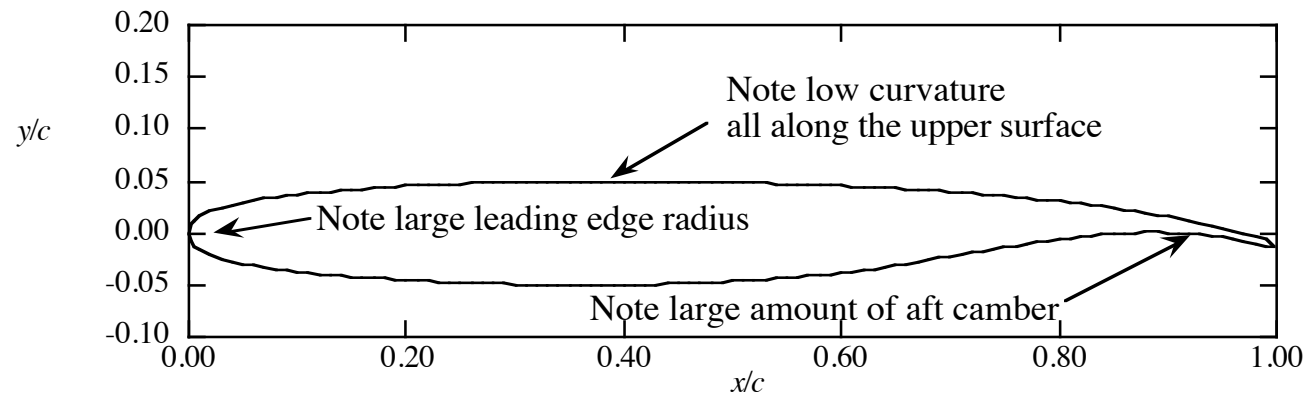
From NASA Supercritical Airfoils, by Charles D. Harris, NASA TP 2969, March 1990

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

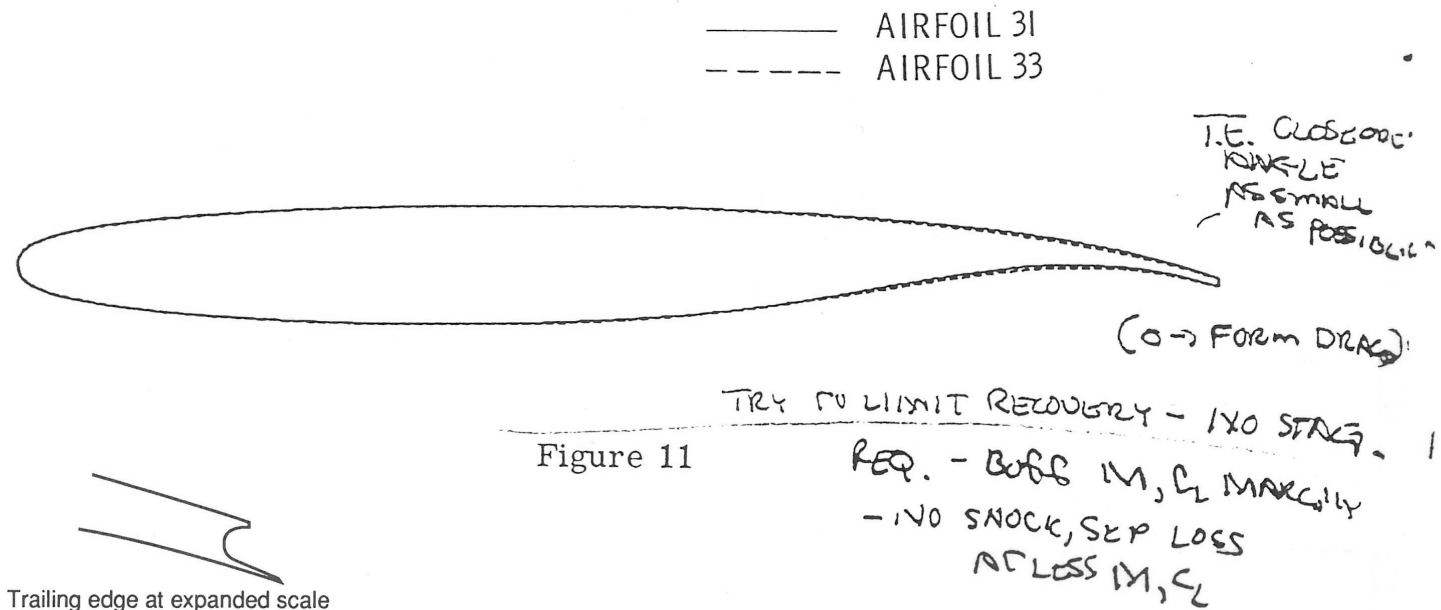
- An off-design criteria: a well behaved sonic plateau at $M = 0.025$ below the design M
- Gradient of pressure recovery gradual enough to avoid separation
 - in part: a thick TE, say 0.7% on a 10/11% thick foil
- Airfoil has aft camber so that design angle of attack is about zero, upper surface not 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)

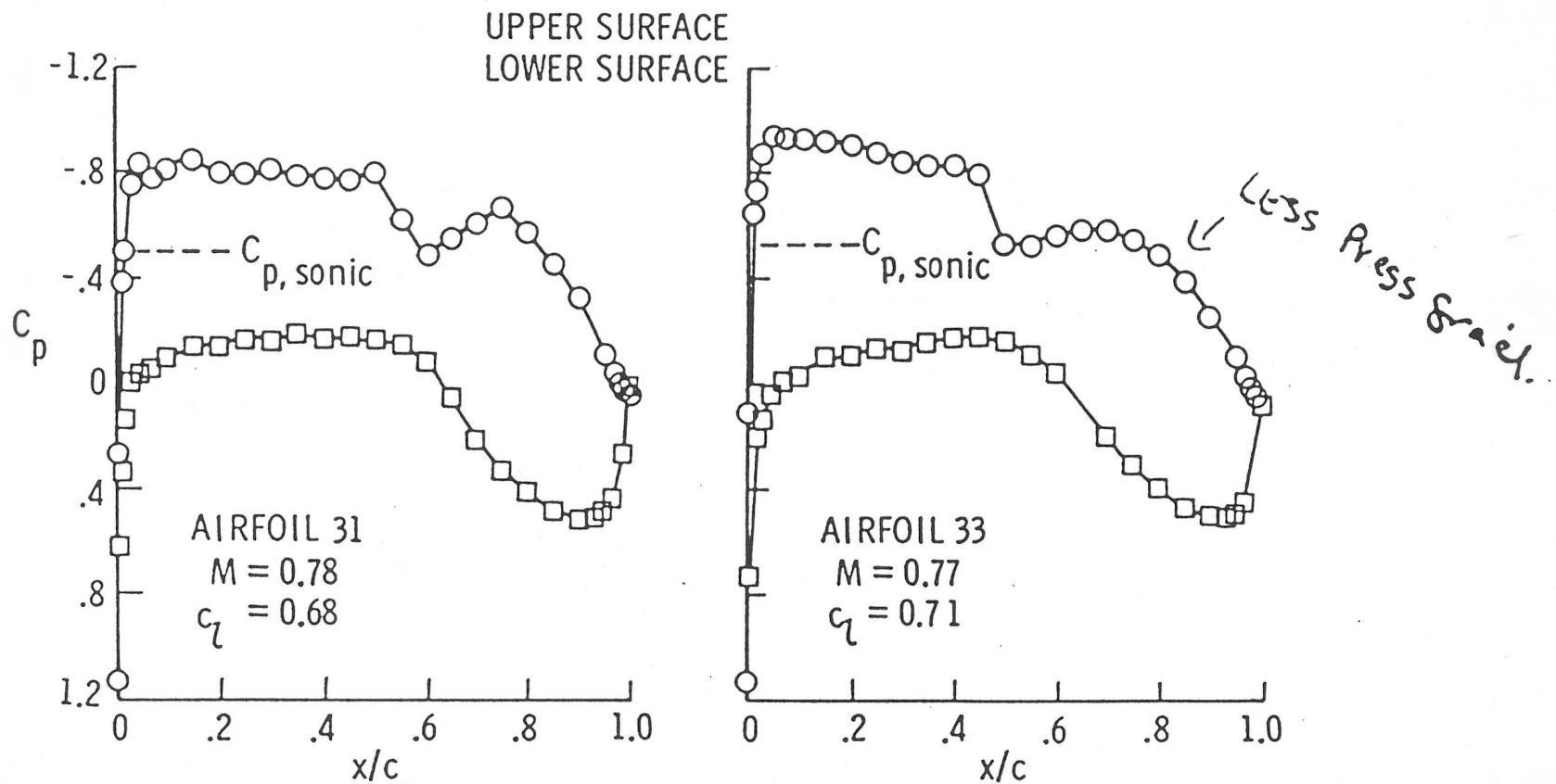
Example: Airfoils 31 and 33

10-PERCENT-THICK NASA SUPERCRITICAL AIRFOILS 31 AND 33



Airfoils 31 and 33

EXPERIMENTAL PRESSURE DISTRIBUTIONS NEAR DESIGN C_l



Off Design

EXPERIMENTAL SONIC PLATEAU PRESSURE DISTRIBUTIONS

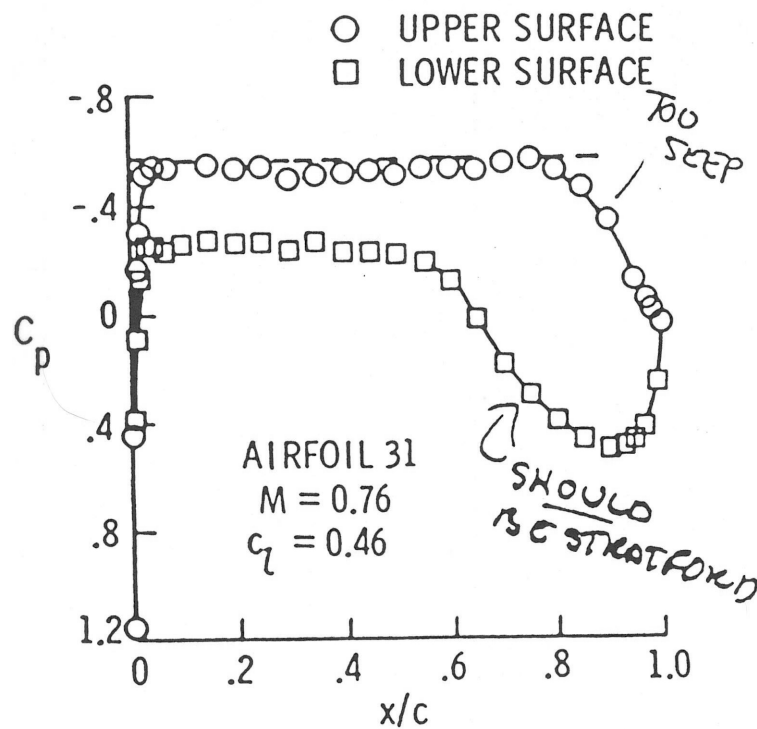
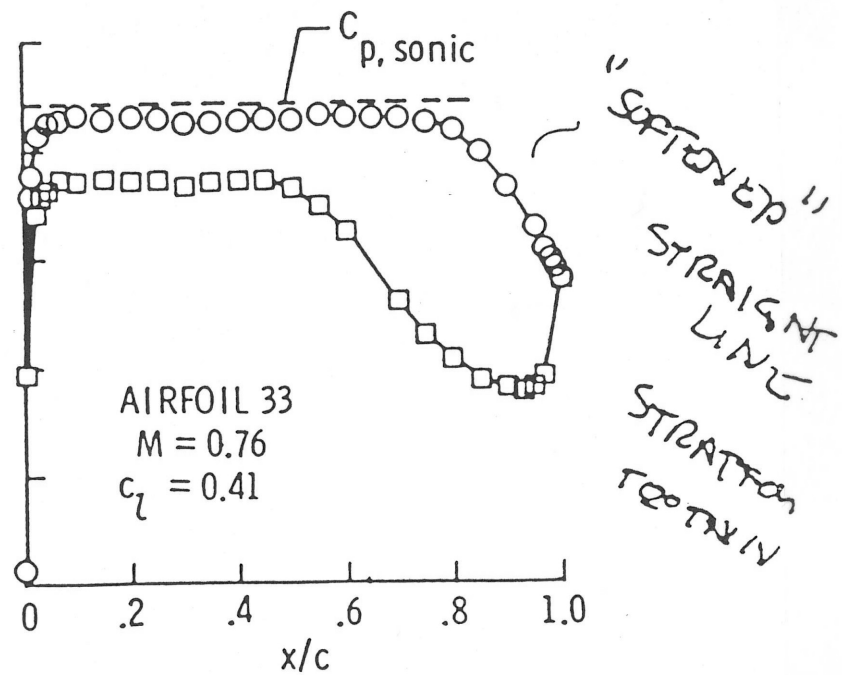


Figure 12

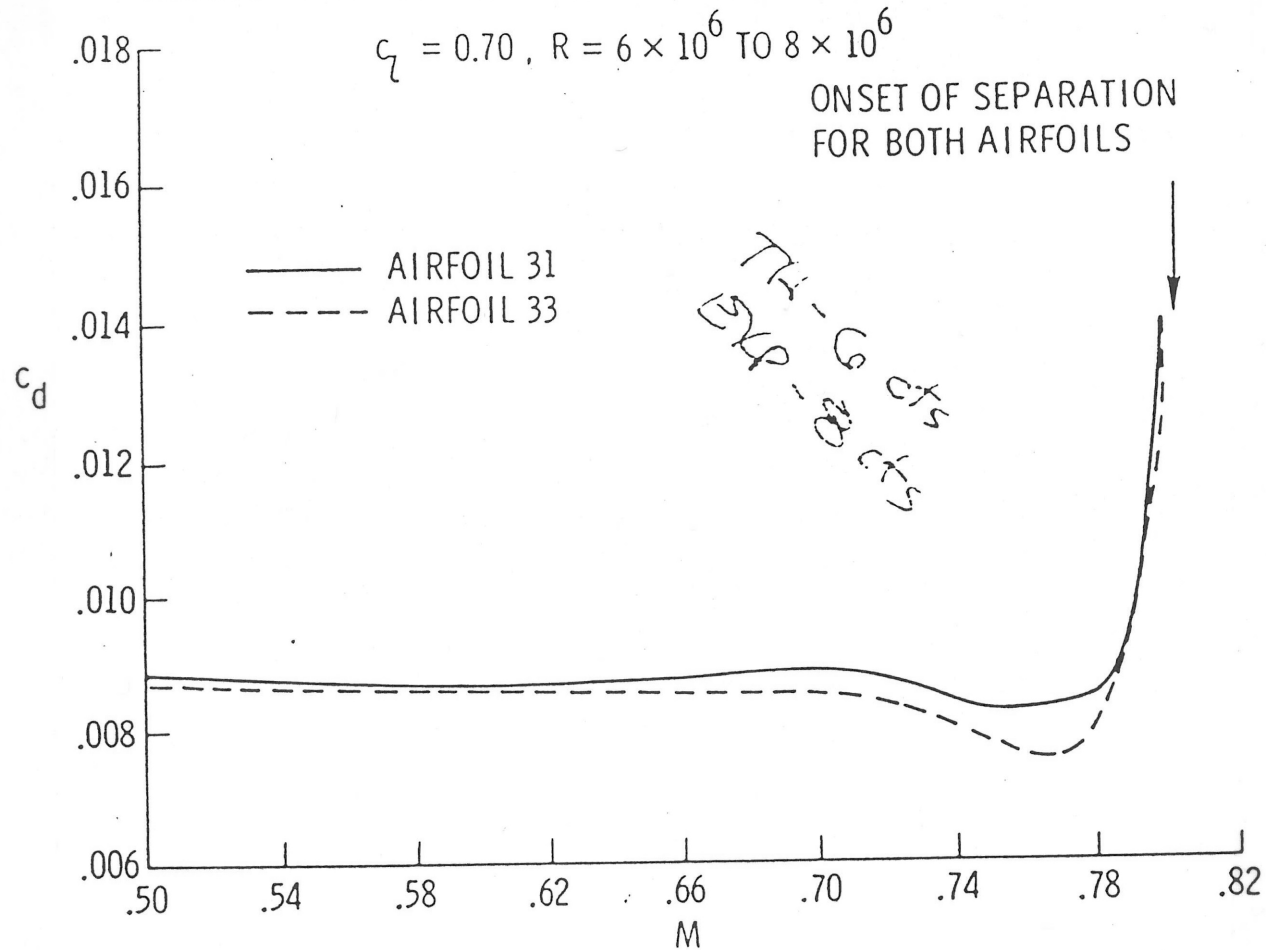


Drop in $M \sim .025$

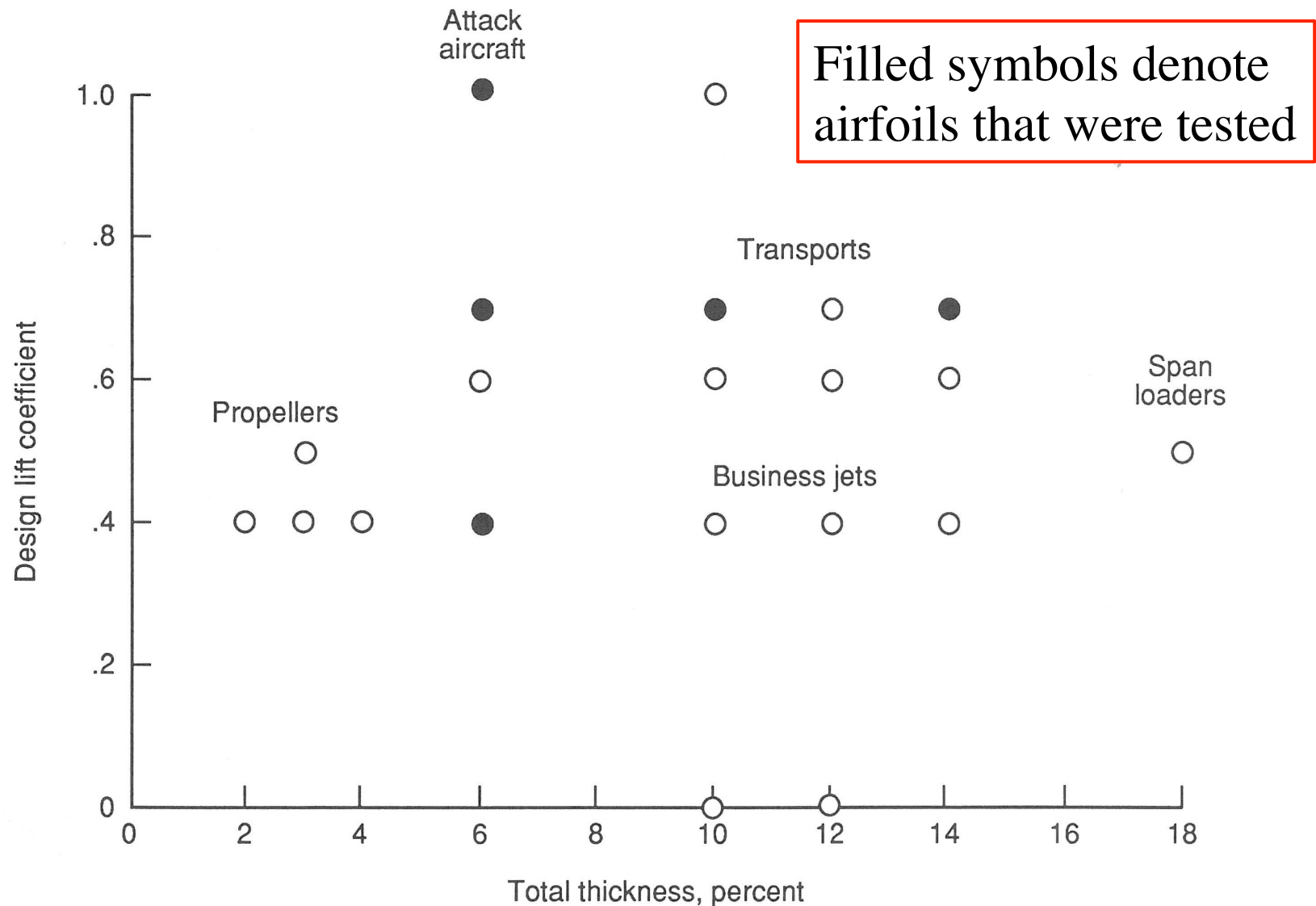
-GET SONIC PLAT.
IMP. FOR OFF DESIGN

Foils 31 and 33 Drag

EXPERIMENTAL DRAG CHARACTERISTICS FOR
NASA SUPERCritical AIRFOILS 31 AND 33

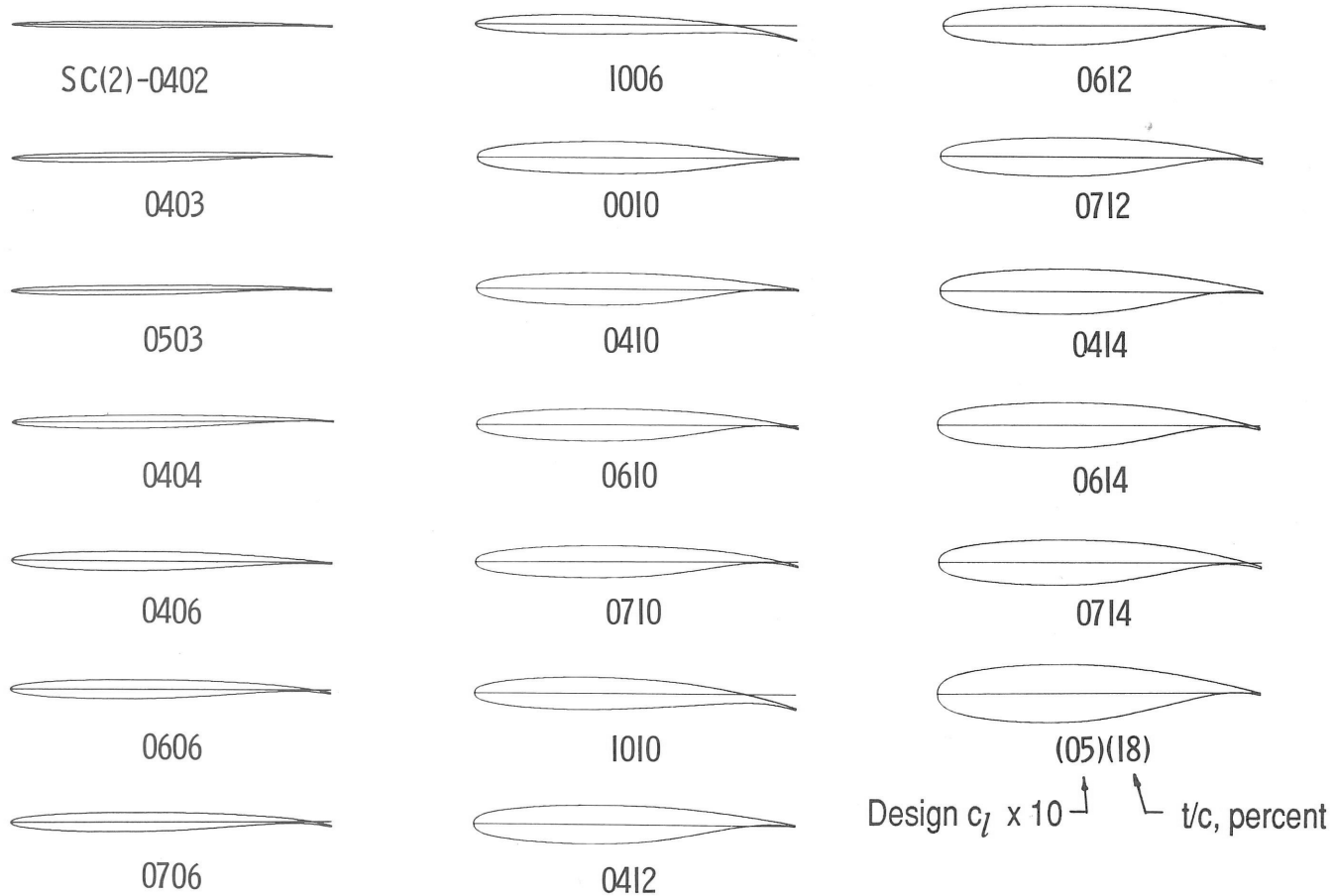


NASA Airfoils Developed Using the Guidelines



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

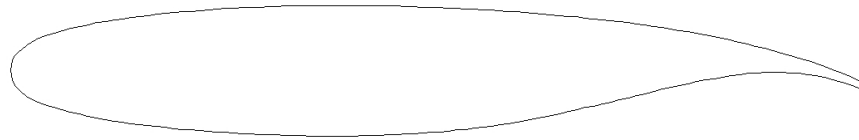
- clean wing low speed CL_{max}
- Mach divergence at moderate to high lift coefficients

Choice of aft camber

Design lift coefficient and flight Reynolds number

Increase favors

- clean wing low speed CL_{max}
- drag divergence Mach number at high lift coefficients



Decrease favors

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

Decrease favors

- trim drag by decreasing CM_0
- 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)

F.T. Lynch, "Commercial Transports—Aerodynamic Design for Cruise Performance Efficiency," in *Transonic Aerodynamics*, ed. by D. Nixon, AIAA, 1982.

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)

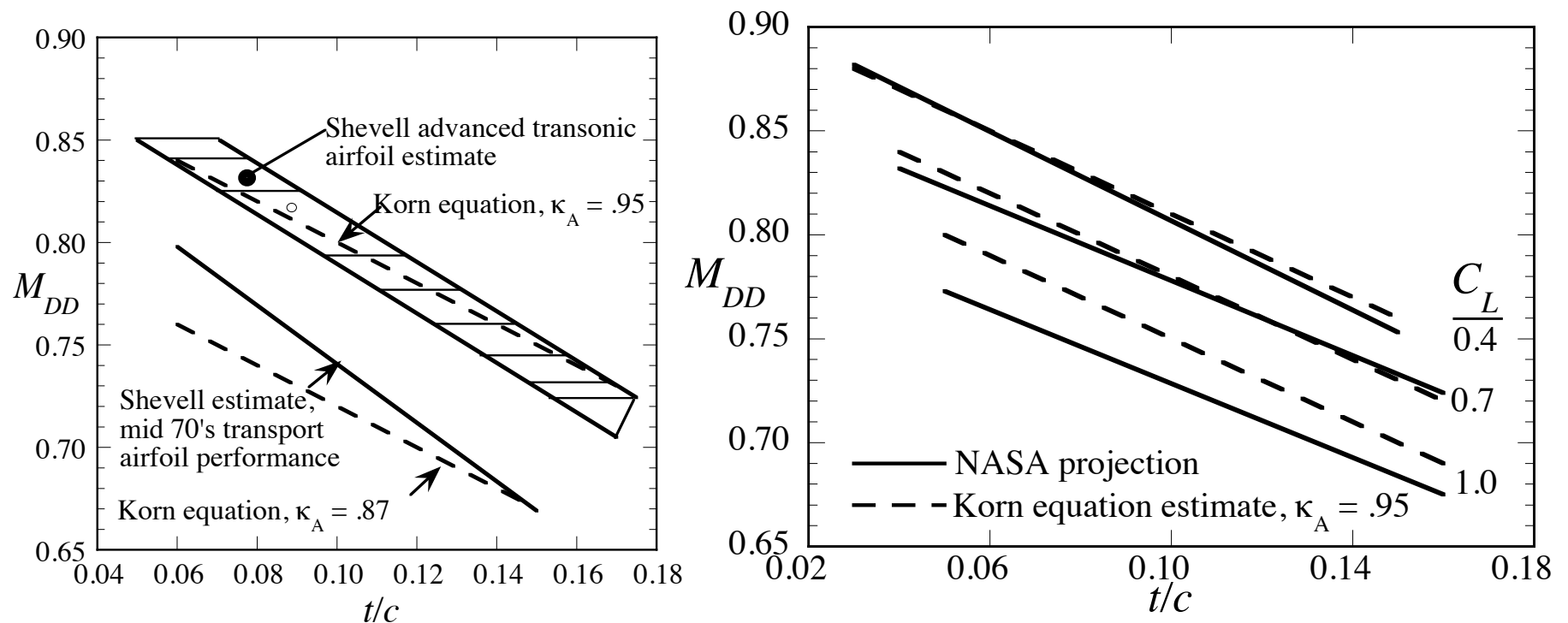
$\kappa_A = 0.95$ for supercritical airfoils

Note: the equation is sensitive to κ_A

This is an approximation until CFD or WT results arrive!

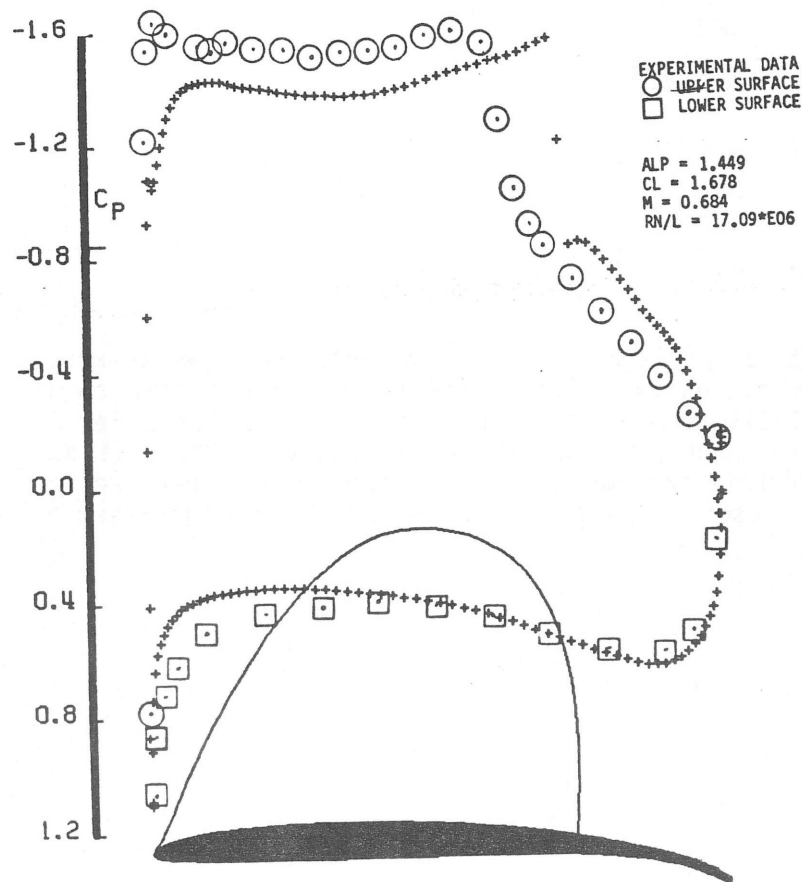
Airfoil Limits

Shevell and NASA Projections Compared to the Korn Equation

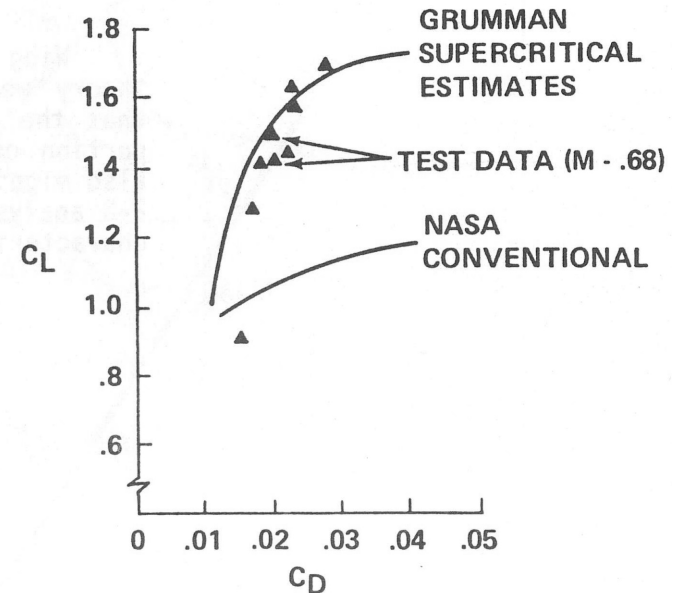


In W.H. Mason, "Analytic Models for Technology Integration in Aircraft Design," AIAA Paper 90-3262, September 1990.

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



THIN SUPERCRITICAL AIRFOIL TEST DATA



GRUMMAN K (EXP.) AIRFOIL
 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
 RN= 9.70 MILLION

Divergent Trailing Edge Airfoil

A NEW AIRFOIL DESIGN CONCEPT

by

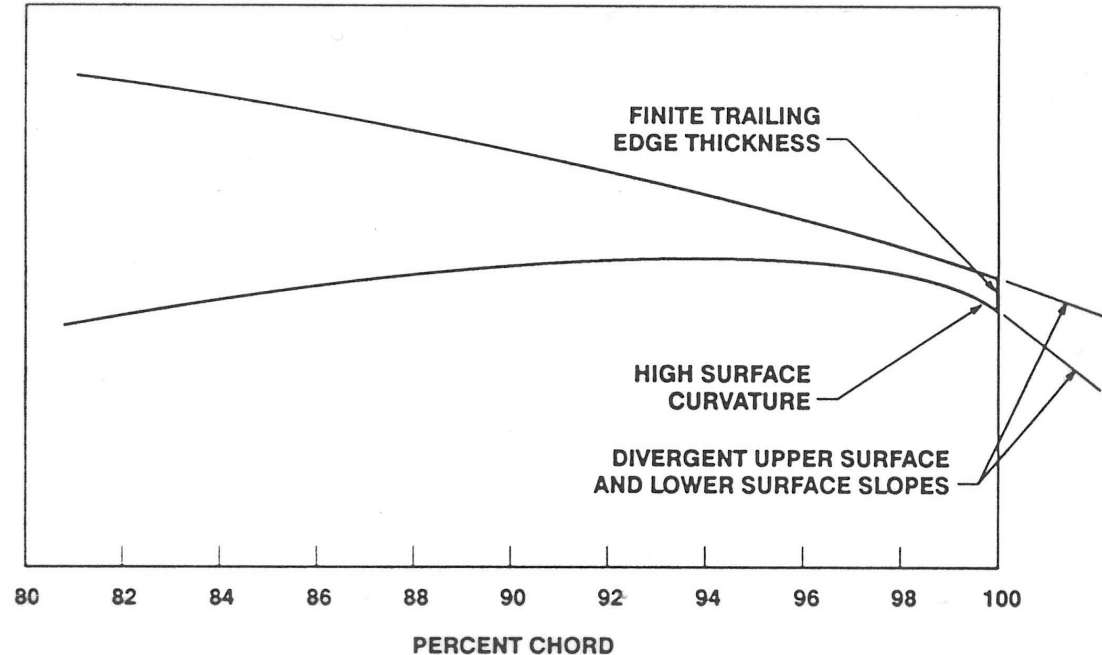
P. A. Henne and R. D. Gregg

Douglas Aircraft Company

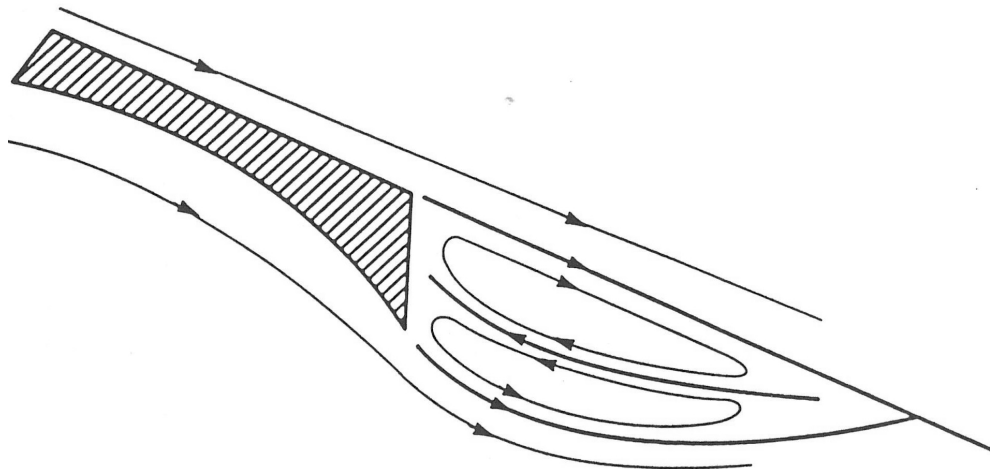
McDonnell Douglas Corporation

**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

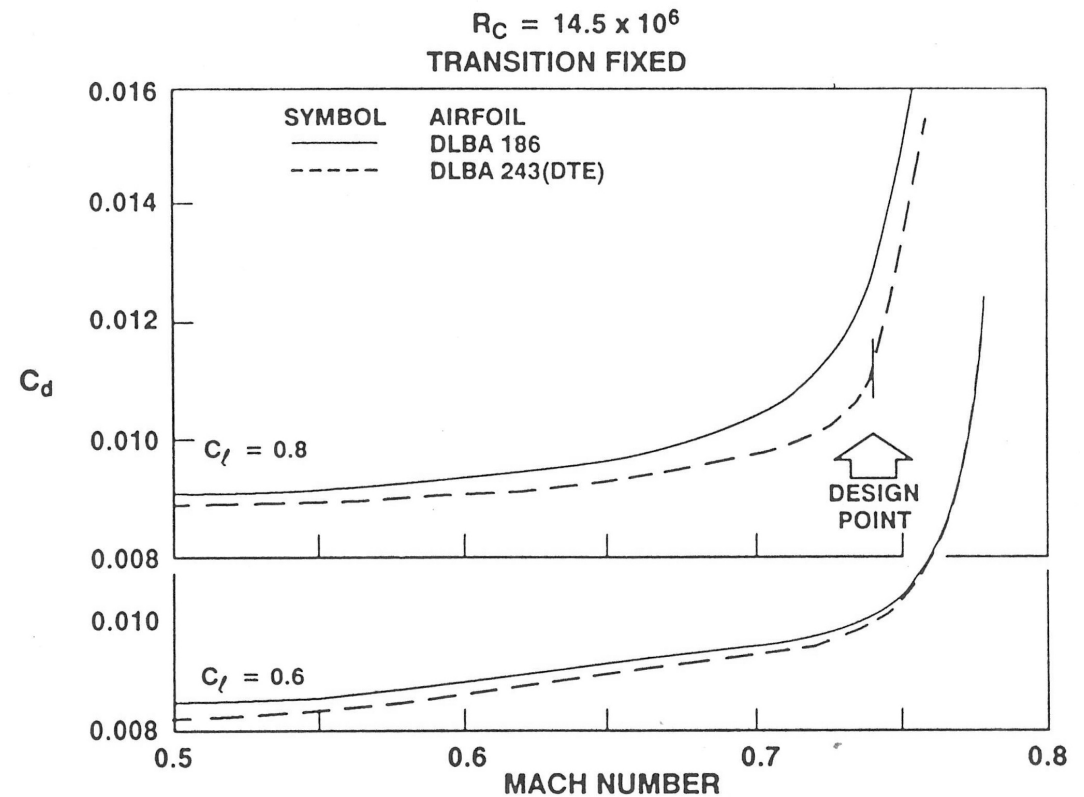


**Used on the MD-11
resisted in Seattle!**



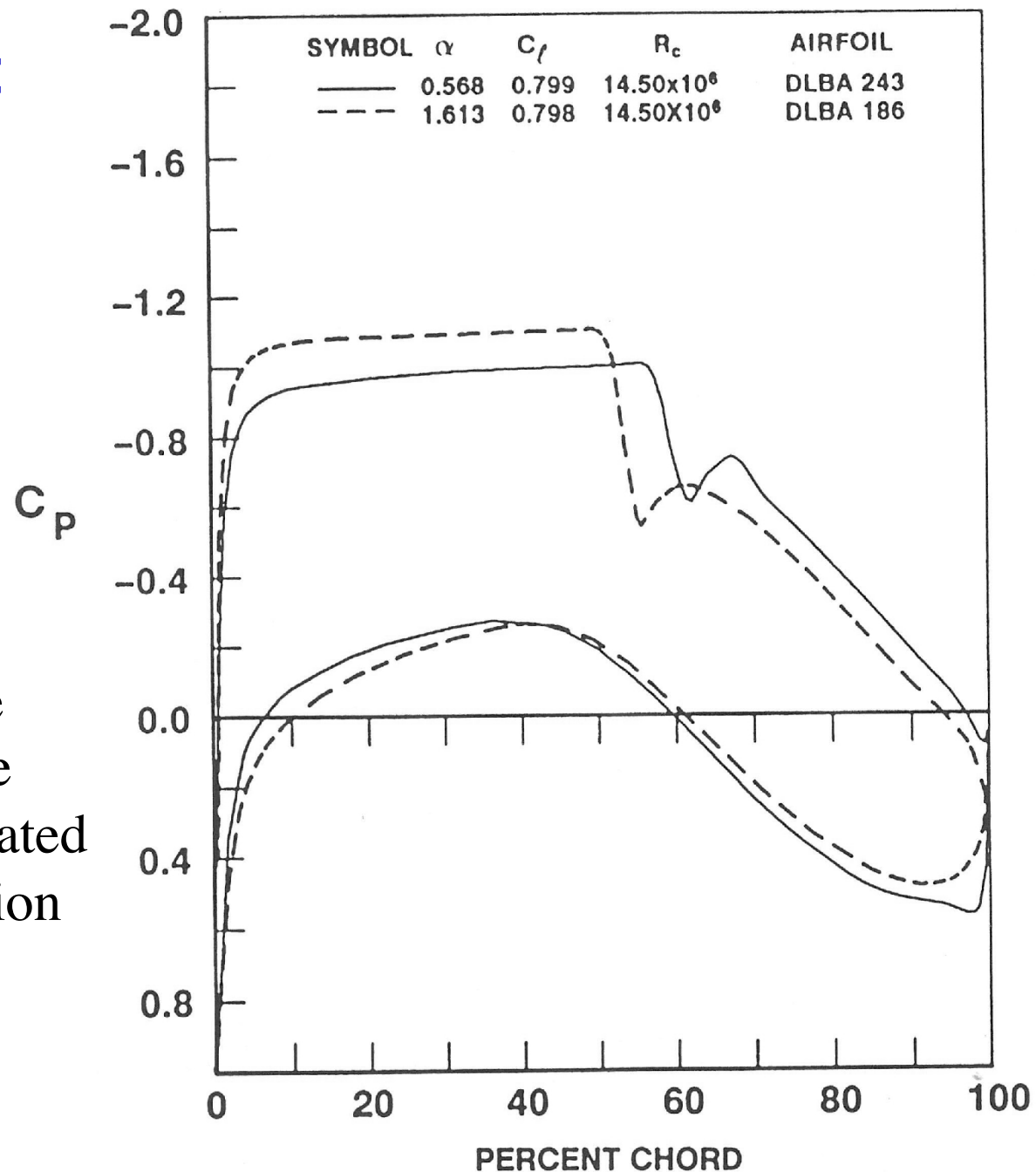
**Hypothesized Flow Closure for Divergent
Trailing Edge Airfoil**

**Comparison of DLBA 243 and DLBA 186
Calculated Drag Rise Characteristics**



Take a Look at the Pressure Distribution

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



Boeing shows next Fighter

Boeing X-36 – in Air Force Museum



1st flight - 1997

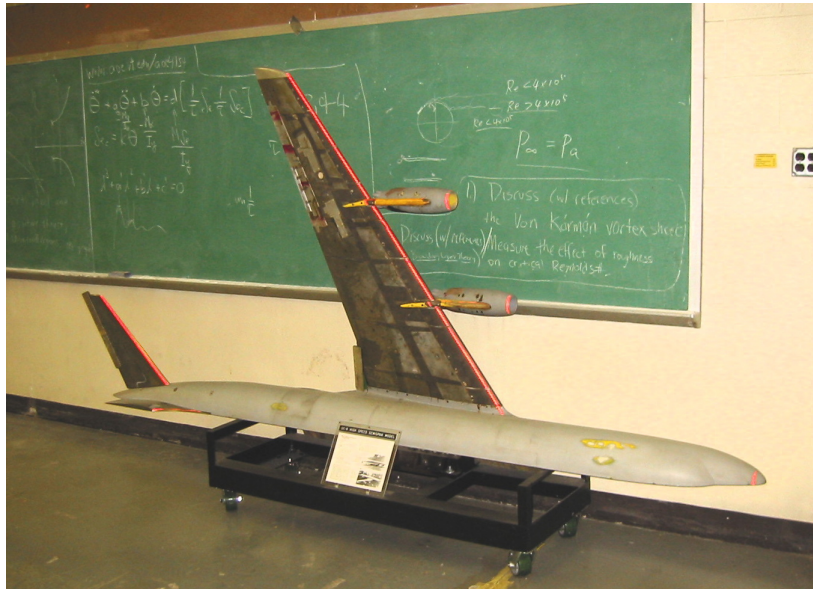
Companies usually
built on previous
experience

Found through
AIAA Daily Launch – Yesterday?



Last time: Transonic Airfoils

Today: Wing

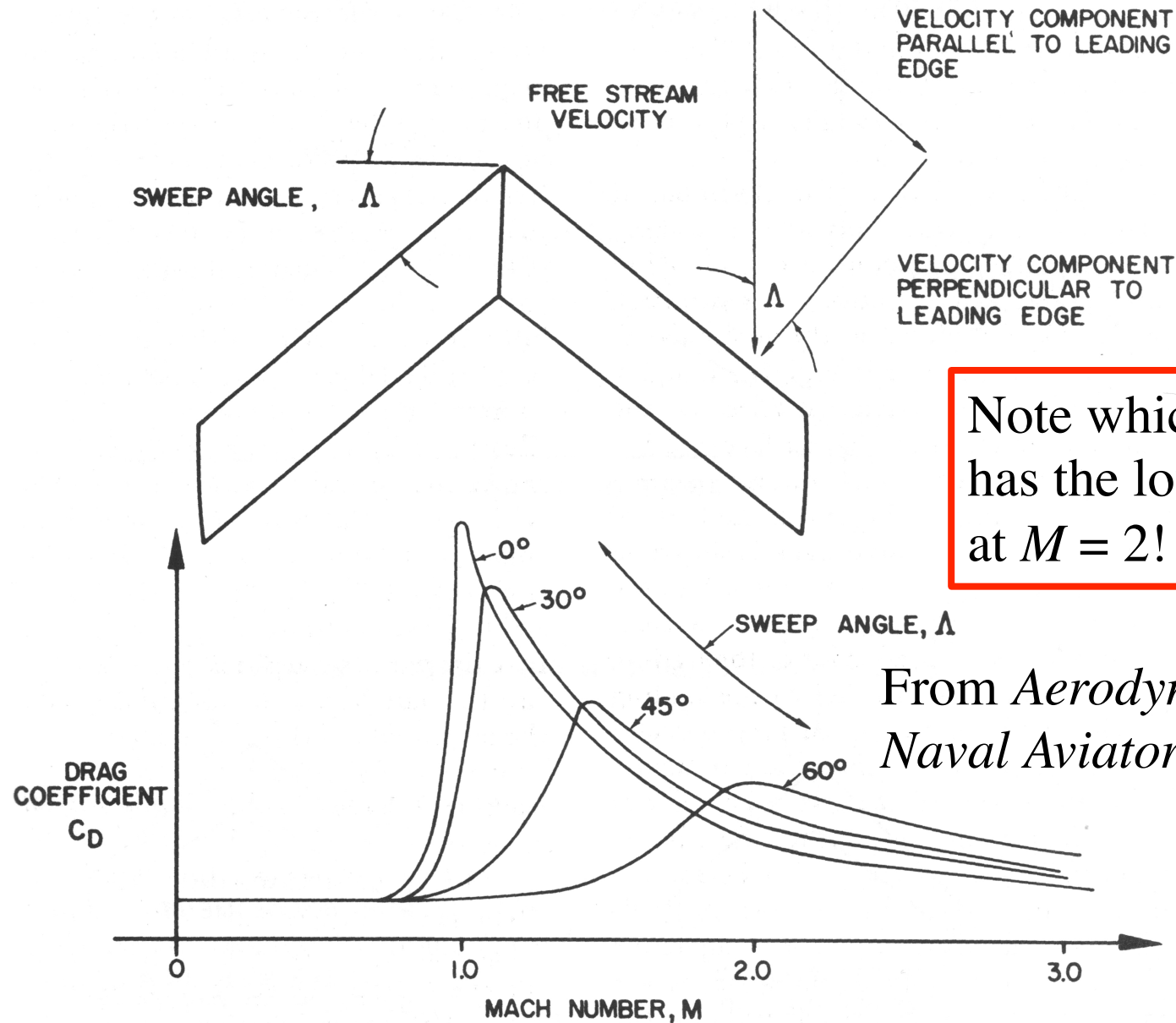


The original DC-8 reflection plane model, the wing is now in Prof. Philen's Lab in the basement of Randolph.

An advanced Douglas transport WT Model, also in the VT WT Lab



The Swept Wing – A Major Advance



Note which sweep has the lowest drag at $M = 2$!

From *Aerodynamics for Naval Aviators* by Hurt

The original swept wing model carved by George Shairer

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

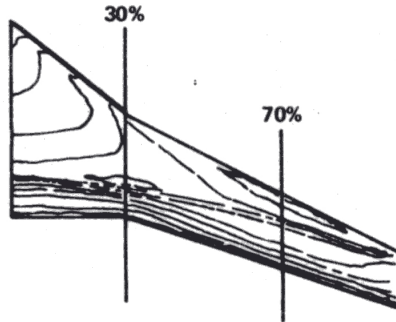


picture taken by Mason at the Museum of Flight, Seattle, WA

The problem of transonic wing design

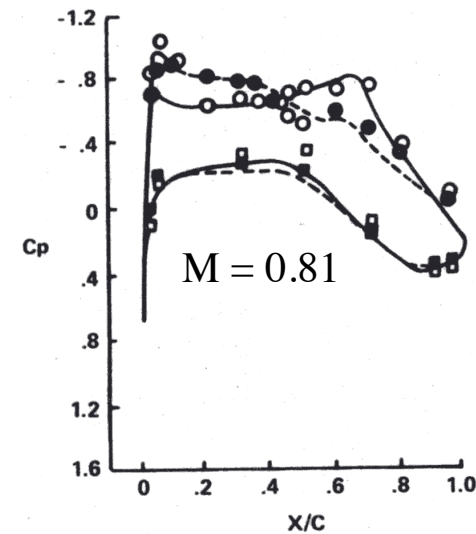
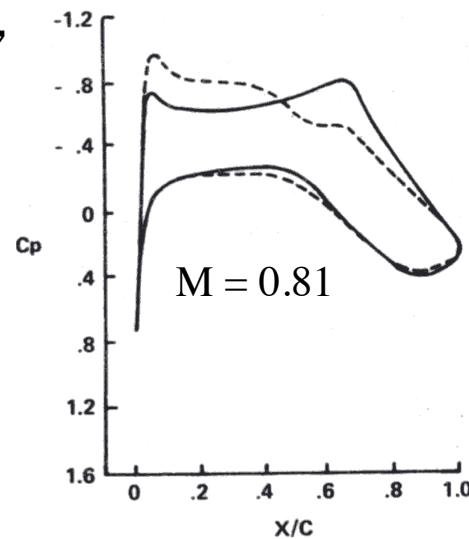
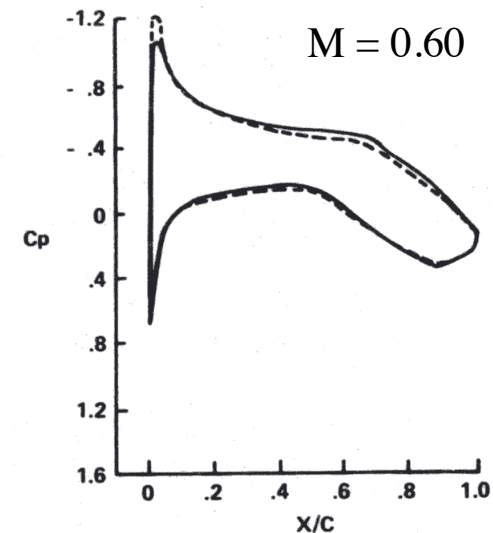
W. Mason, D.A. MacKenzie, M.A. Stern, and J.K. Johnson, "A Numerical Three-Dimensional Viscous Transonic Wing-Body Analysis and Design Tool," AIAA Paper 78-101, Jan. 1978

Remember the importance of good straight isobars!

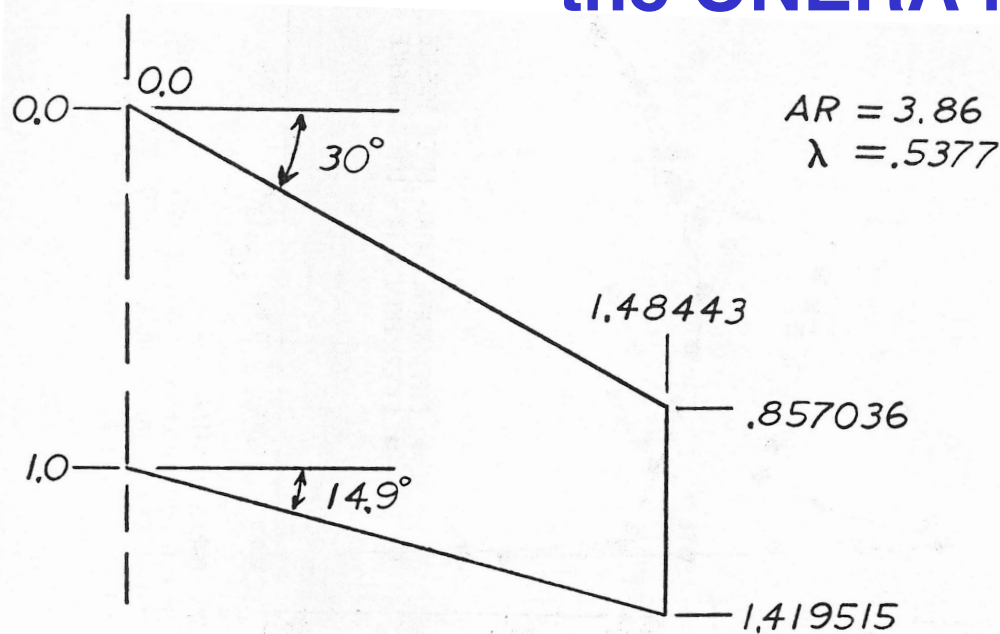


TRANSONIC CRUISE WING

DATA	PREDICTION
○ □	— 30% SPAN STATION
● ■	- - - 70% SPAN STATION

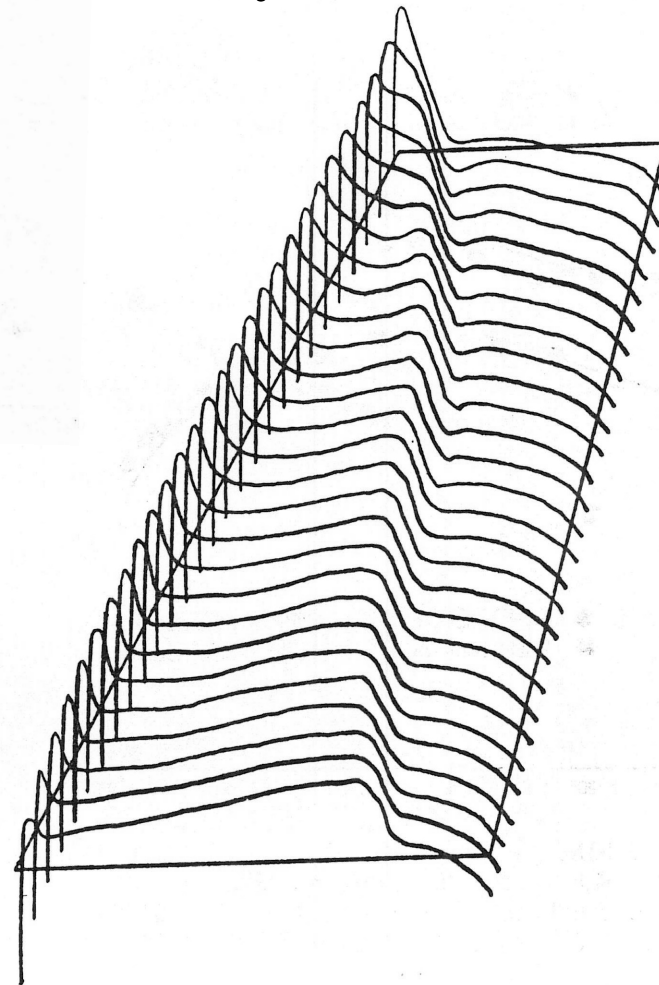


The Standard Transonic Test Case: the ONERA M6 Wing



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

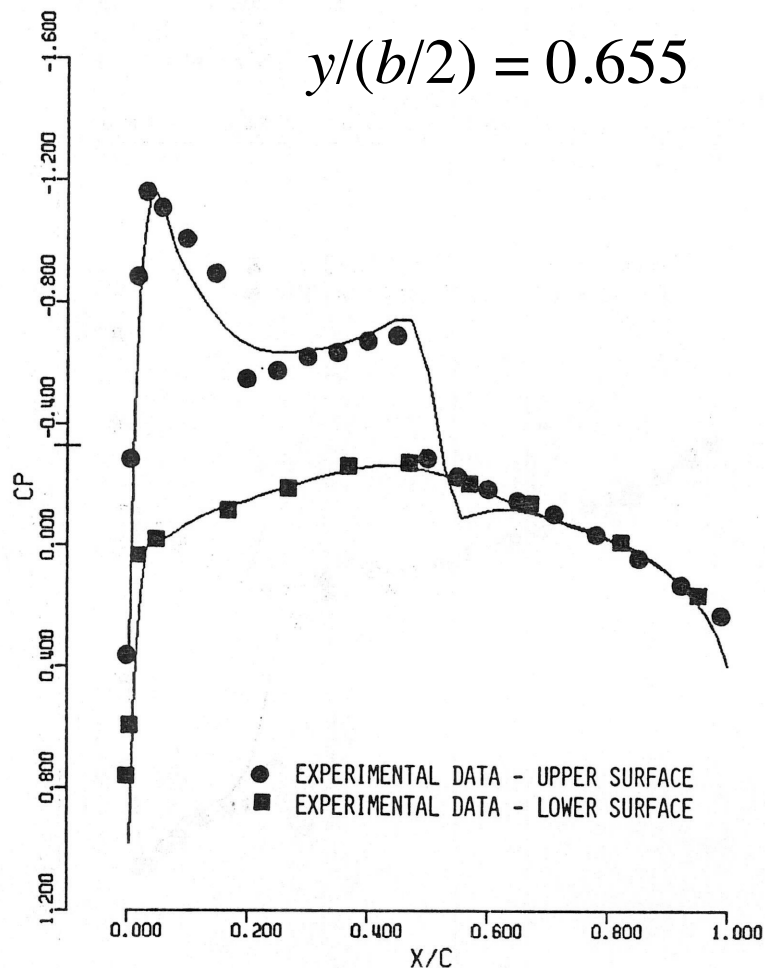


FIGURE 32E. ONERA M6 WING

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

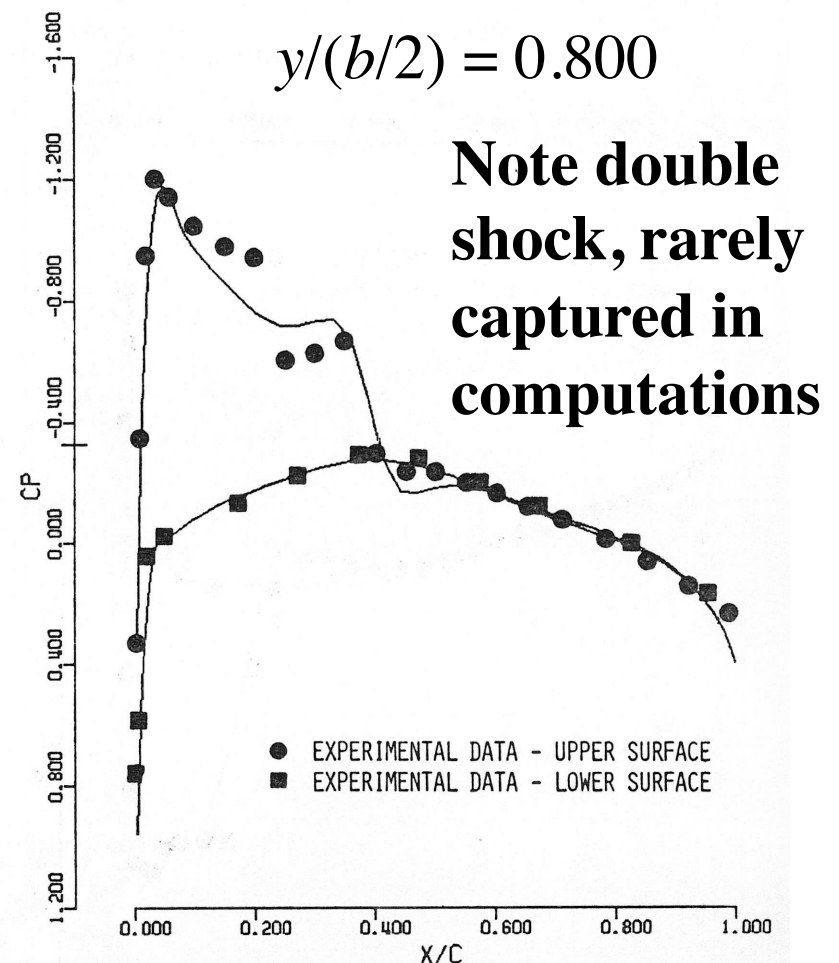


FIGURE 32F. ONERA M6 WING

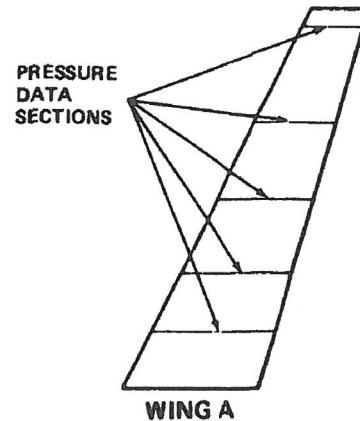
M = 0.840 ALF = 3.000 CONV = .327E-04
LOCAL CL = 0.258 LOCAL CM = -0.312 LOCAL CD = -.0089
SPAN STATION 23 Y/S = 0.800 44 MESH PTS ON CHD

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.

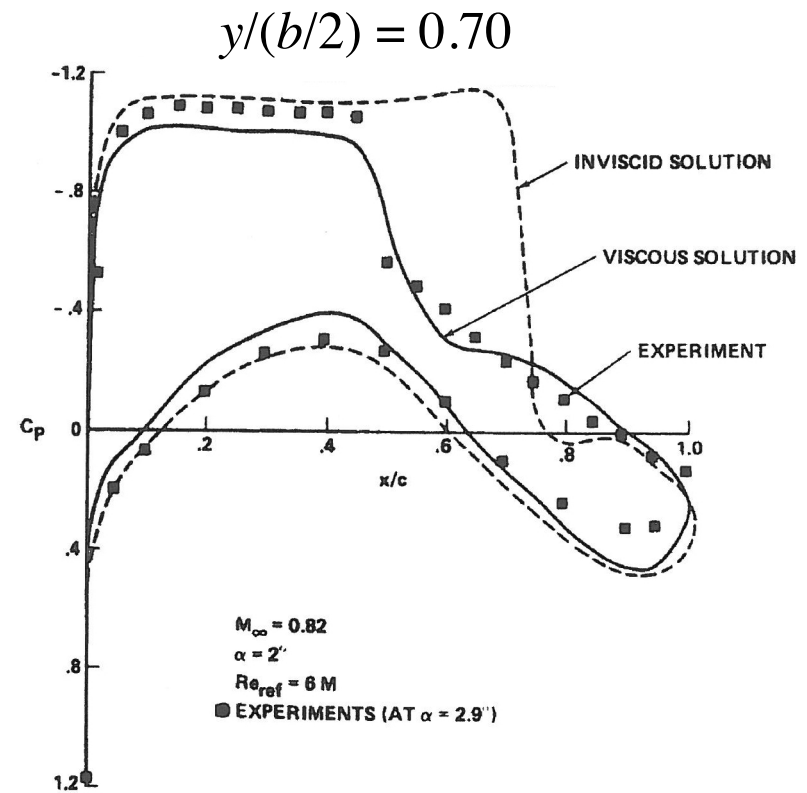
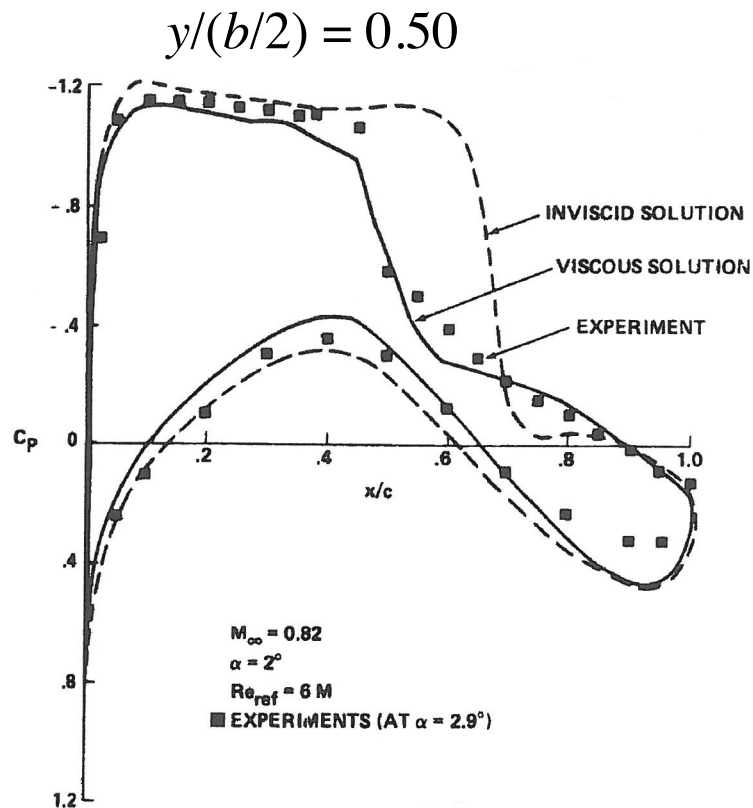
Viscous Effects Still Important

Lockheed Wing A

From NASA CR178156,
Oct. 1986

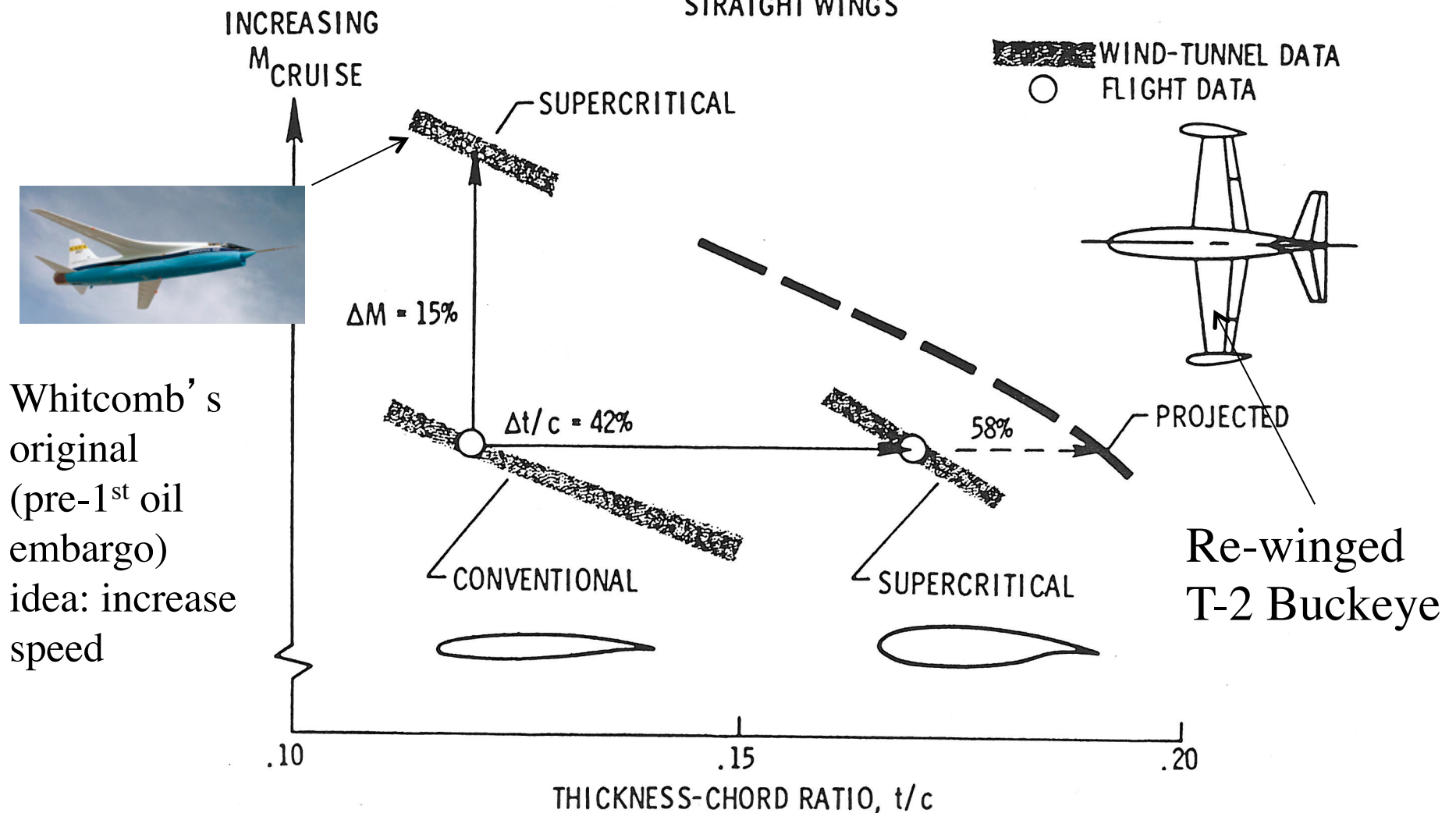


WING A AIRFOIL SECTIONS IN RIGGED POSITION



How to use advanced airfoil capability in wing design

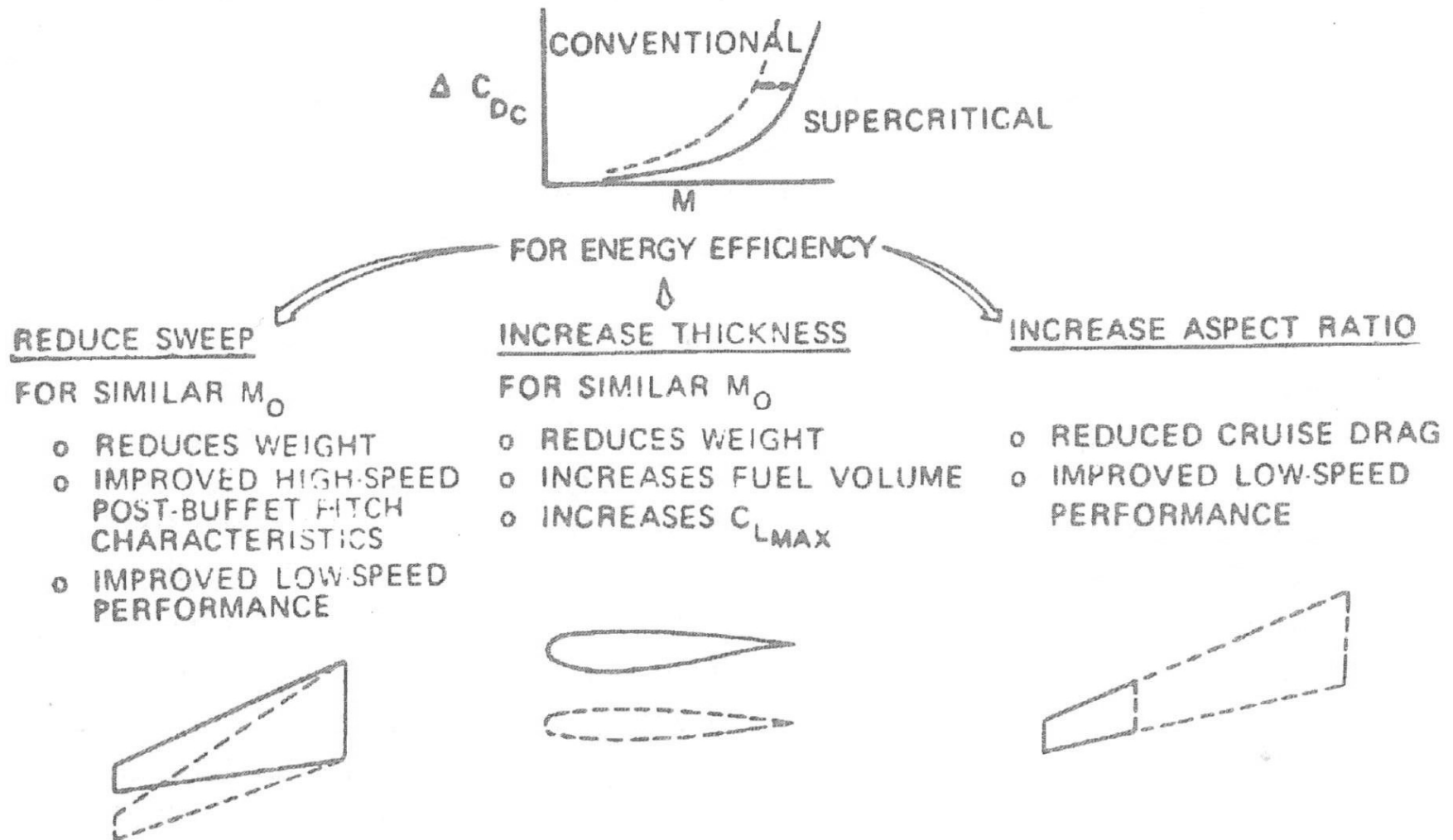
TWO WAYS OF USING SUPERCRITICAL AIRFOIL STRAIGHT WINGS



Ways to use Advanced Aerodynamics

Frank Lynch's Wing Chart

INSTEAD OF HIGHER SPEED FOR SAME SWEEP AND MAX THICKNESS



F.T. Lynch, "Commercial Transports—Aerodynamic Design for Cruise Performance Efficiency," in *Transonic Aerodynamics*, ed. By D. Nixon, AIAA, 1982.

Korn Eqn and Simple Sweep Theory

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

Extend Korn to swept wings using simple sweep theory

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

and then use Lock's approximation:

$$C_{D_{Wave}} = 20(M - M_{crit})^4 \quad M > M_{crit}$$

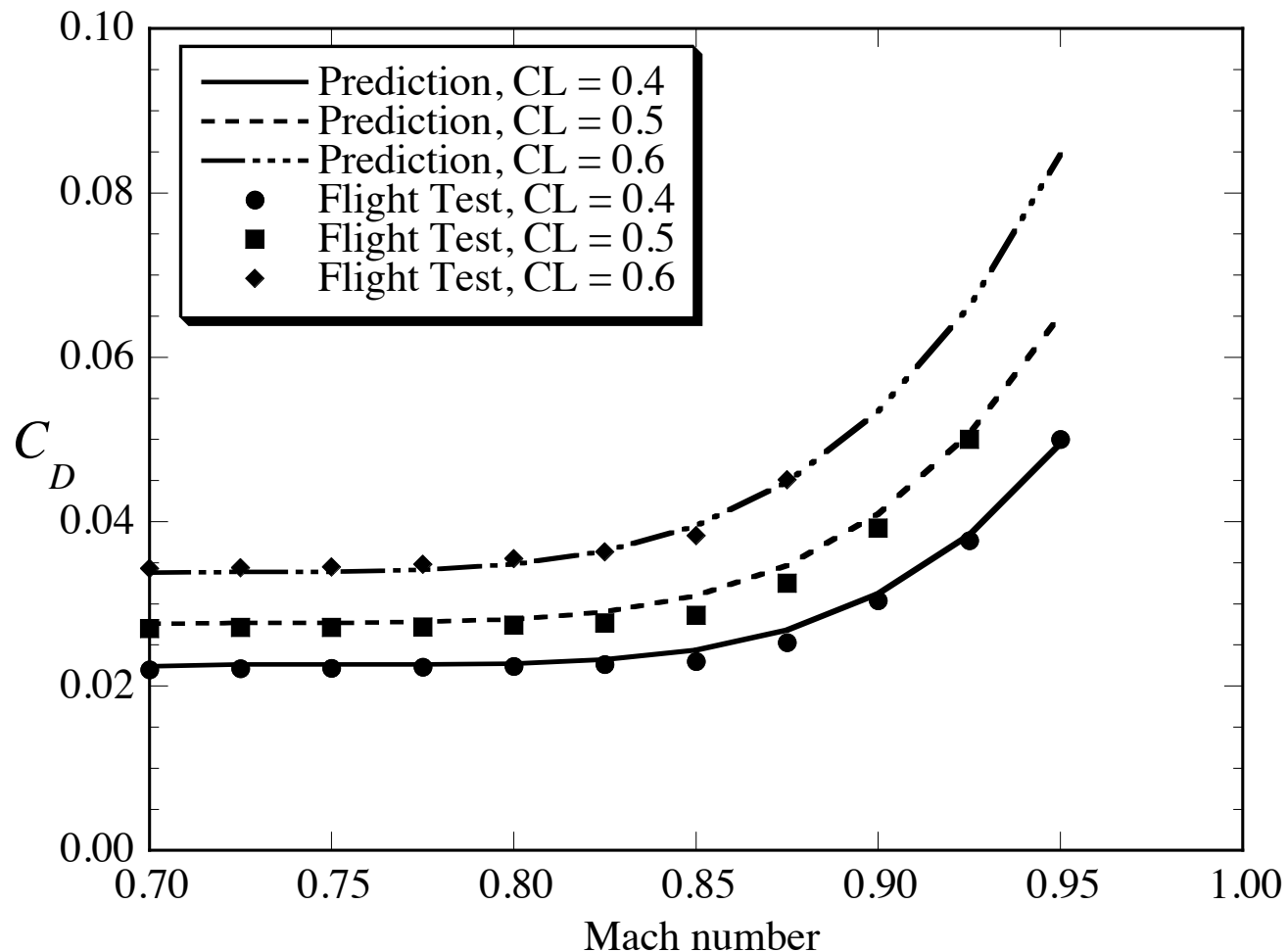
where using: $\frac{\partial C_D}{\partial M} = 0.1$, $\frac{\partial C_D}{\partial M} = 0.1 = 80(M - M_{crit})^3$

so that we find M_{crit} : $M_{crit} = M_{dd} - \left(\frac{0.1}{80}\right)^{1/3}$

Remember this is only a rough approximation!

Transonic Drag Rise Estimate: B747

Example computed by Joel Grasmeyer, $\kappa_A = 0.89$



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?

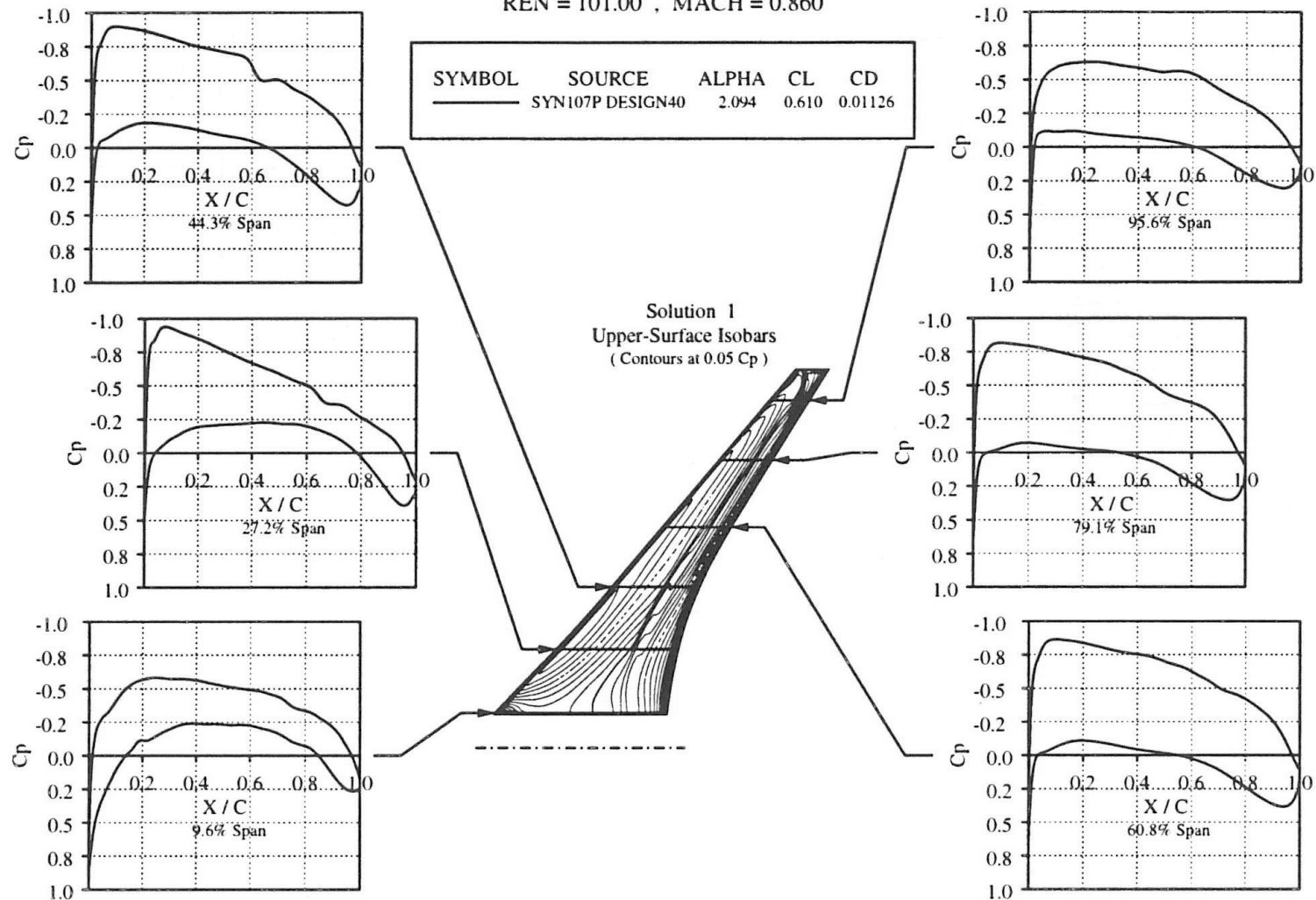
- Set the spanload in an MDO trade
- Surprisingly, linear theory gives a good first guess for the twist distribution
- Design the airfoils in 2D to start, and place in wing
- Finally, full 3D nonlinear analysis and design
 - use numerical optimization
 - Special class of CFD design methods – adjoints to get sensitivities
- 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

COMPARISON OF CHORDWISE PRESSURE DISTRIBUTIONS

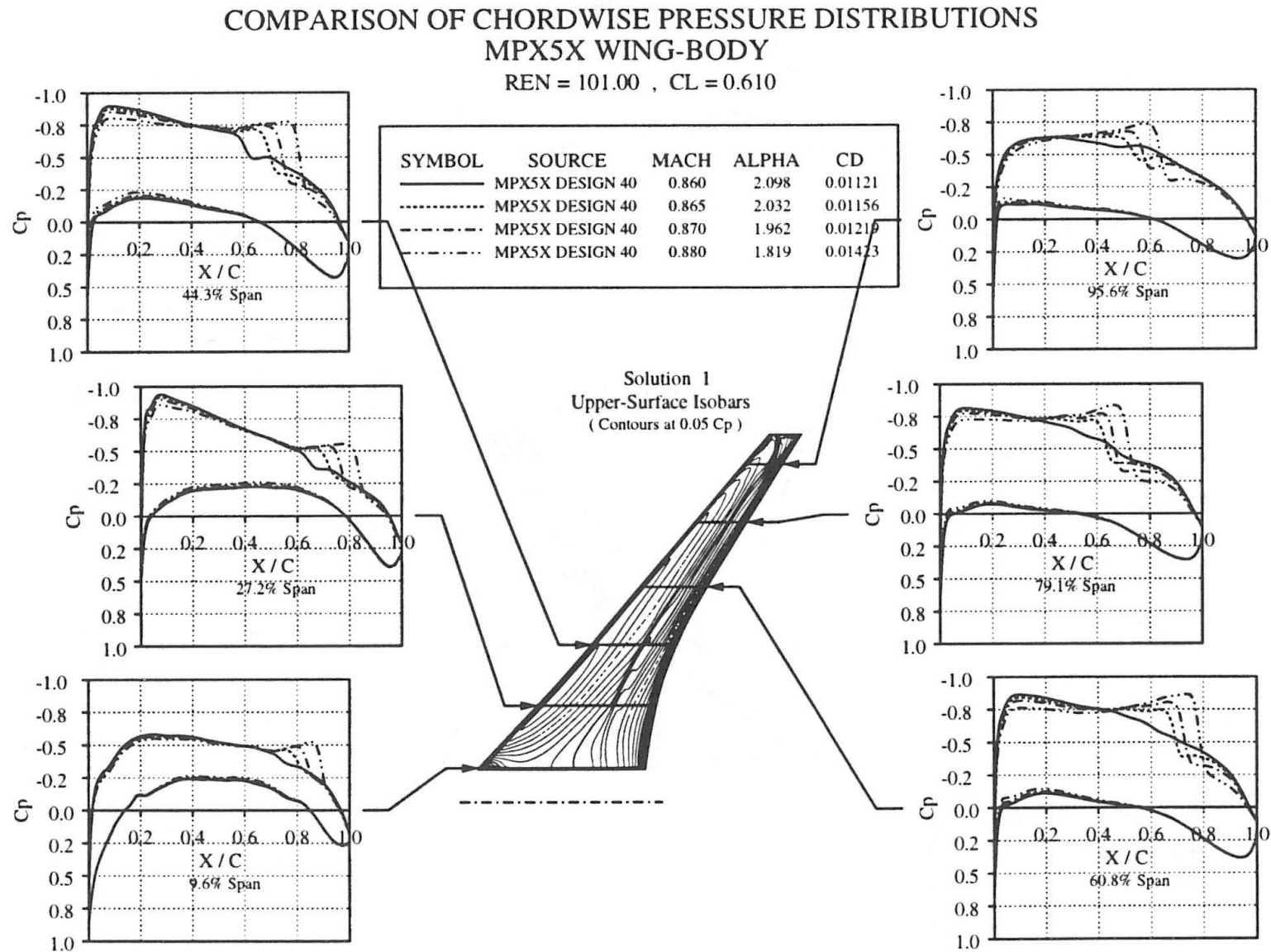
MPX5X WING-BODY

REN = 101.00 , MACH = 0.860



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

Other 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

Real Current Example, The A-380



Quarter Chord Sweep: 33.5°

Aspect Ratio: 7.67

Root chord: $0.132 \text{ } t/c$

Tip chord: $0.087 \text{ } t/c$

From *Aerodynamic Design of Transport Aircraft*, Ed Obert, Delft Press, 2009

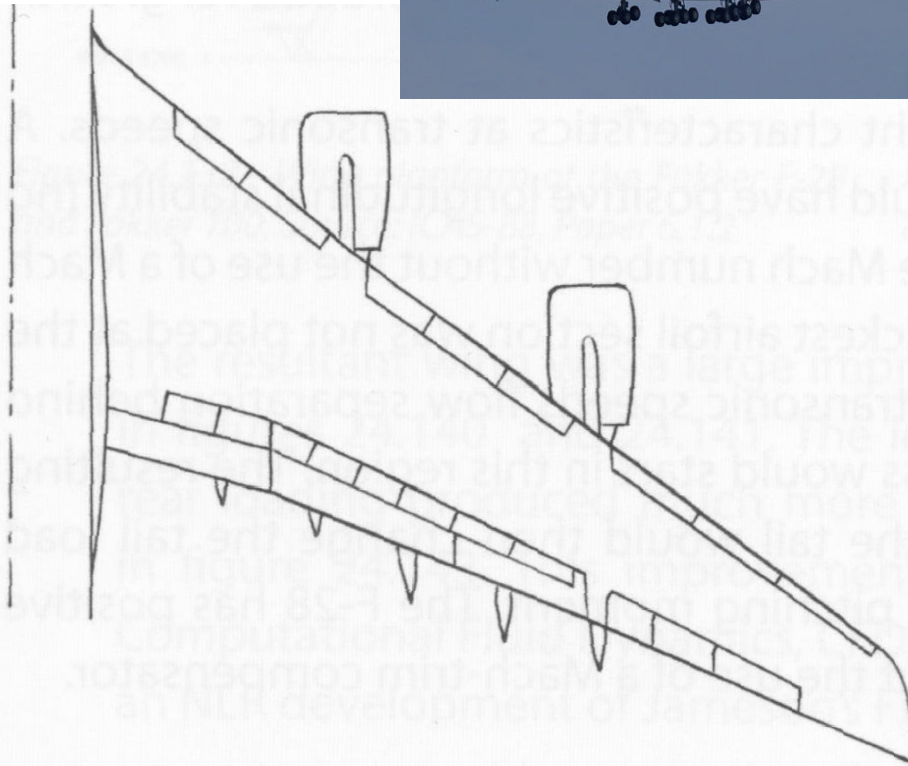


Figure 24.133 - Airbus A380 wing planform

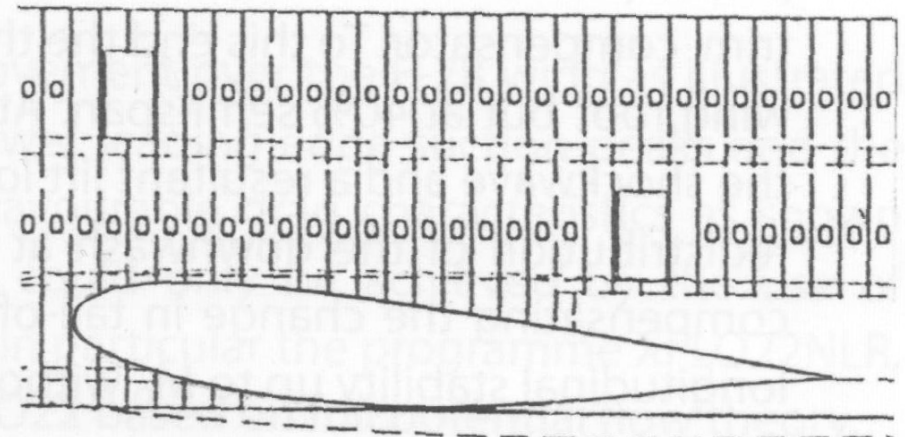
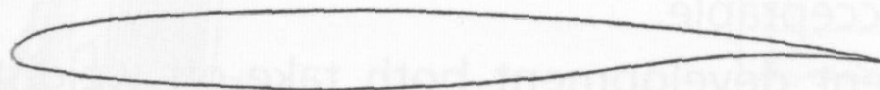
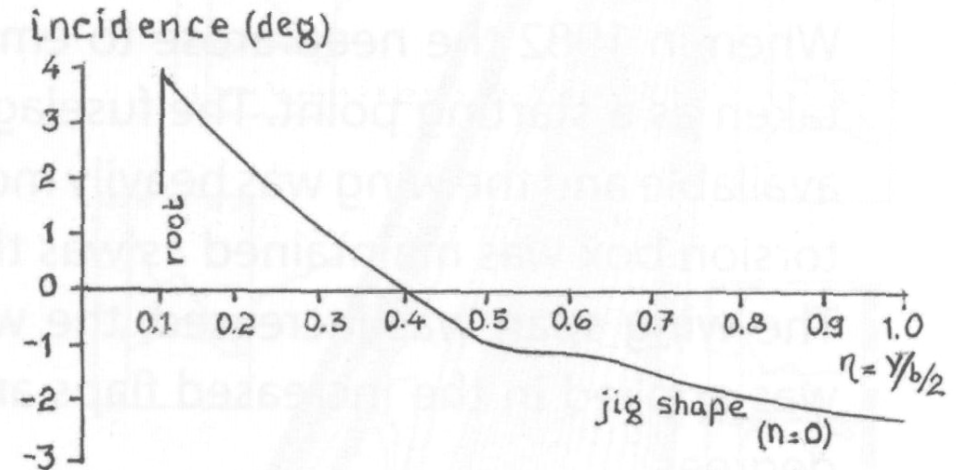
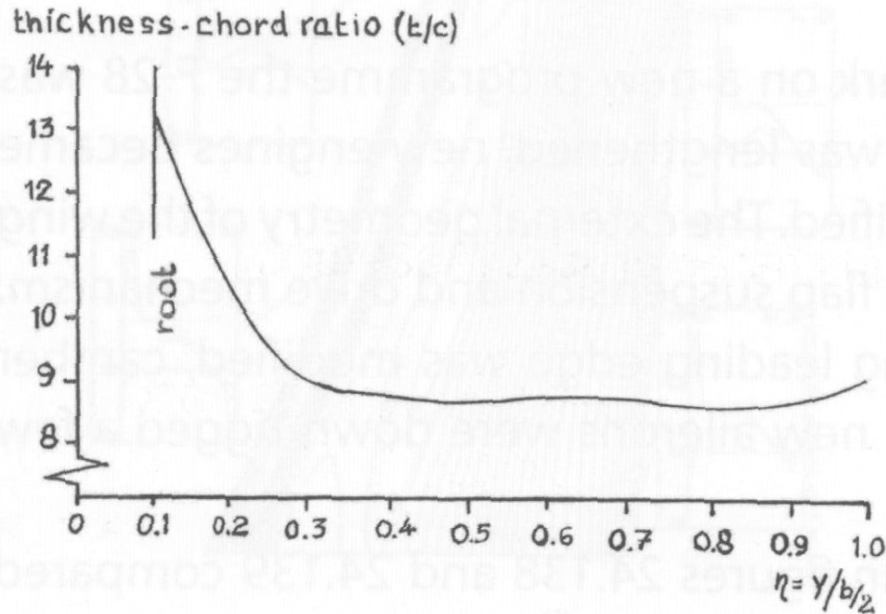


Figure 24.134 - Airbus A380 wing root section



A-380 Thickness and Twist Distributions

- a rare find -

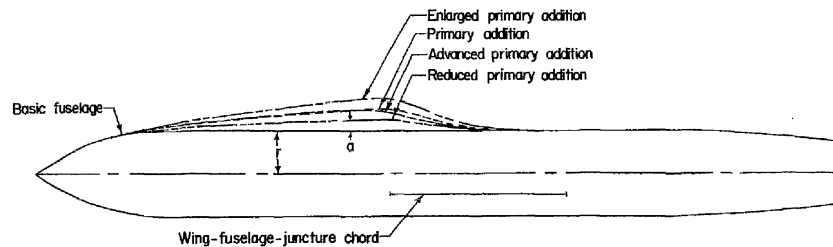


Note: “jig shape” means before the change in twist due to aerodynamic load

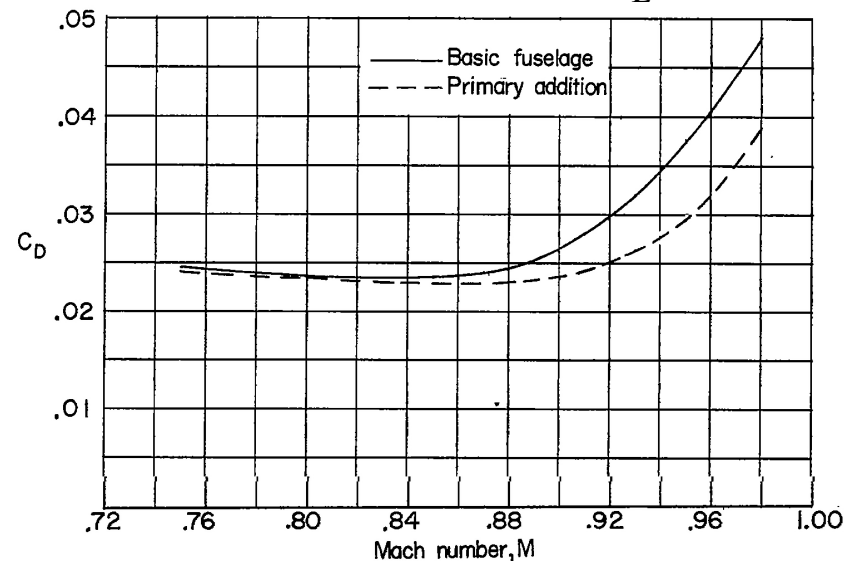
From *Aerodynamic Design of Transport Aircraft*, Ed Obert, Delft Press, 2009

Finally: the fuselage can effect drag

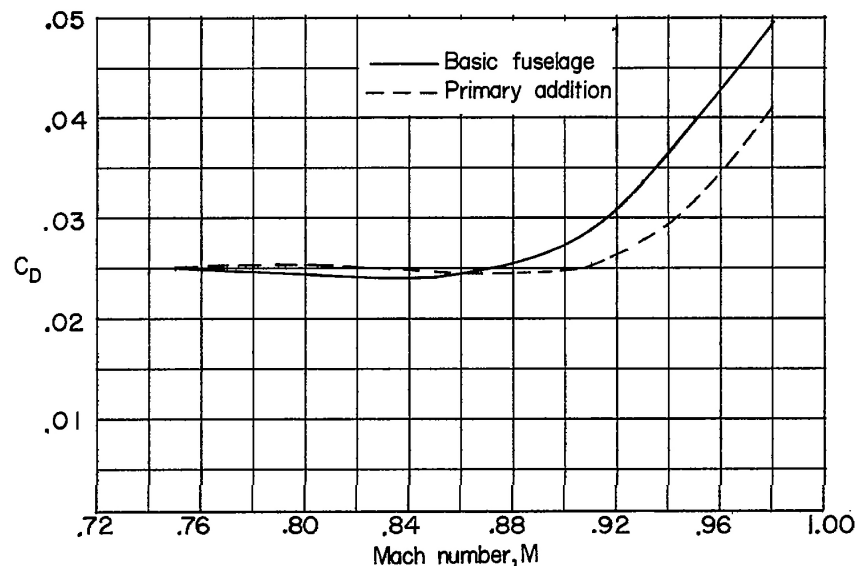
Another Whitcomb Contribution: Fuselage Area Addition



Low mounted wing, $C_L = 0.3$



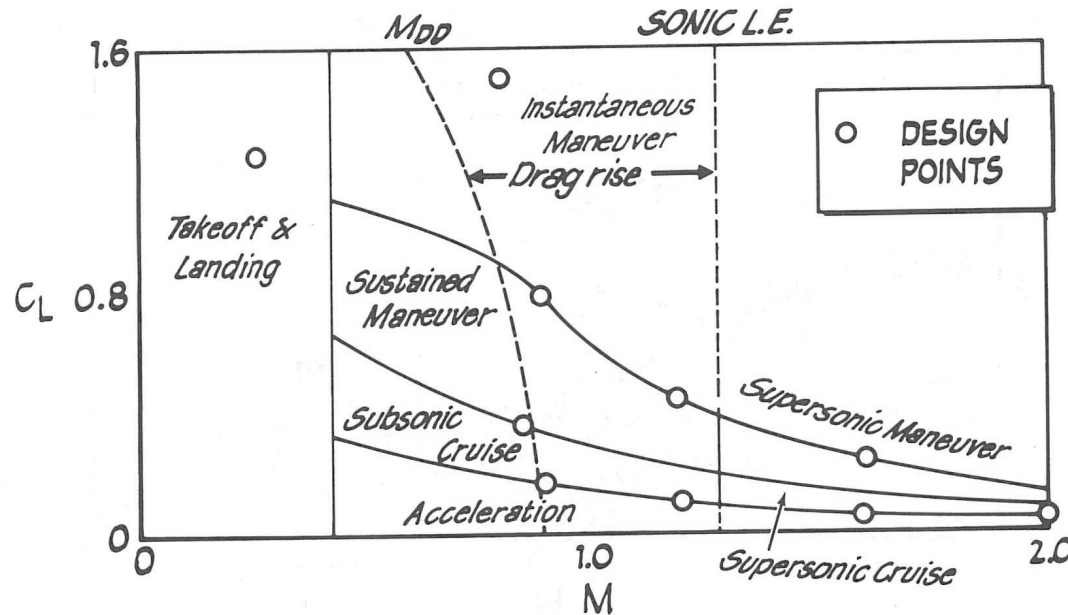
High mounted wing, $C_L = 0.3$



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

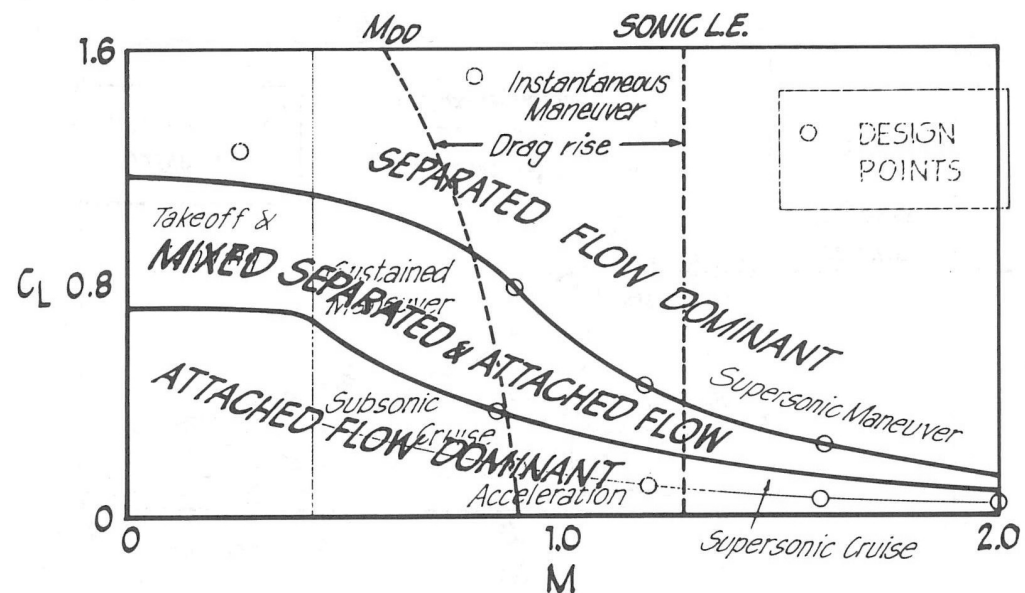
From Whitcomb NACA TN 4290, June 1958.

Fighter Wings: A more complicated story

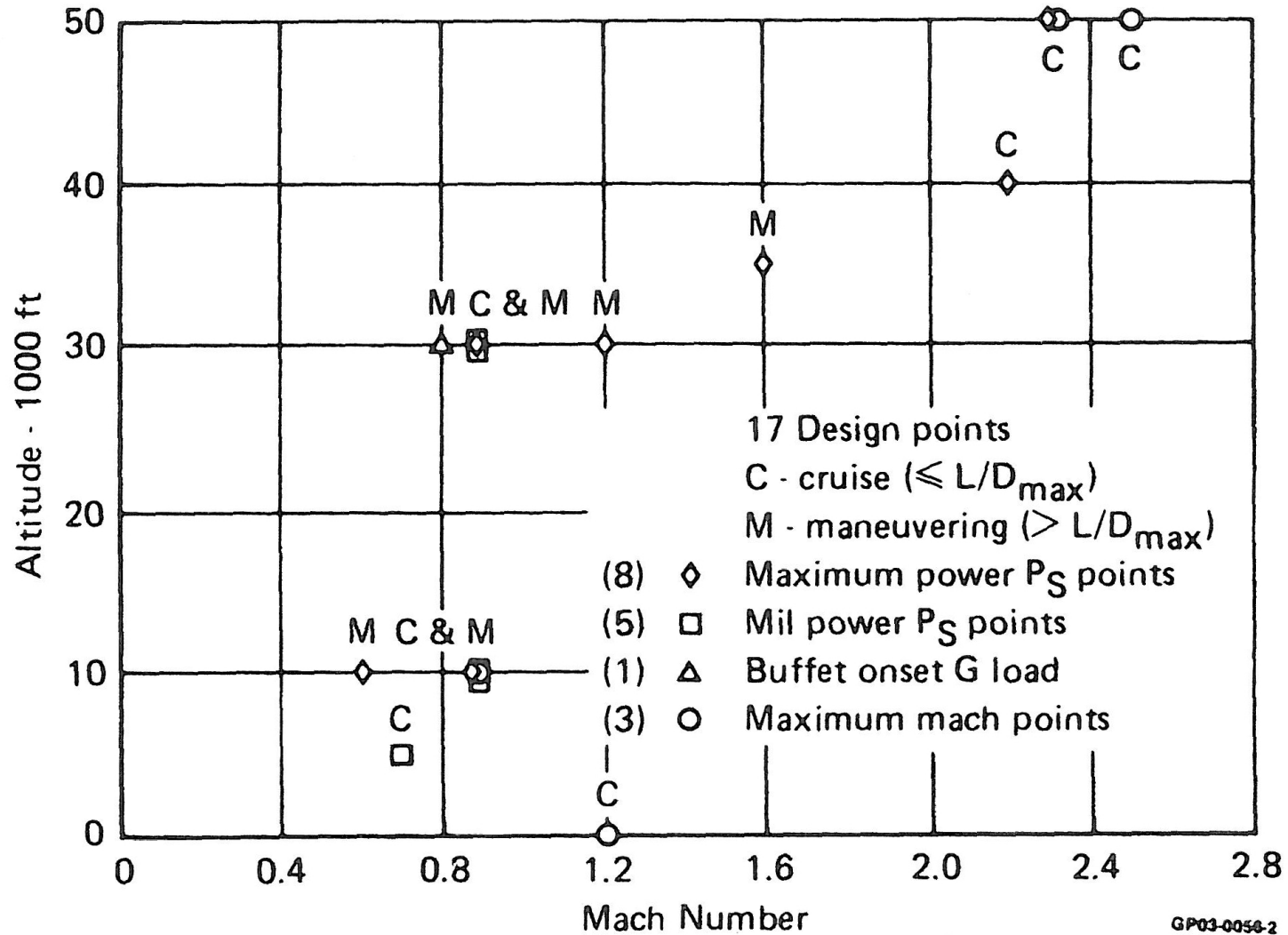


From Richard G. Bradley, "Practical Aerodynamic Problems-Military Aircraft" in *Transonic Aerodynamics*, AIAA 1982

Rather than cruise, fighter wings have to consider supersonics and low speed speed maneuvering as well as transonic cruise and maneuver

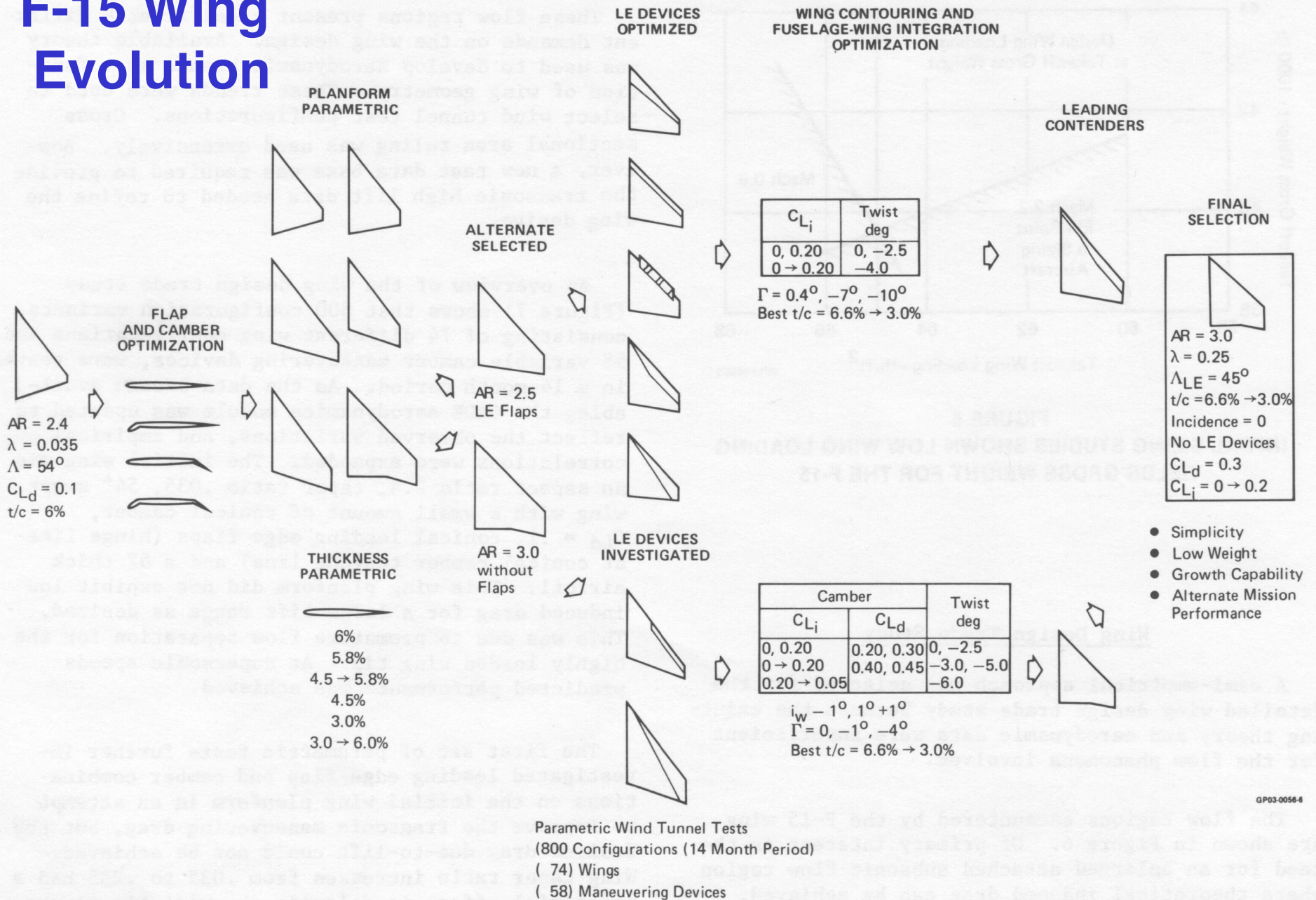


An Example: the F-15 - Design Points



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

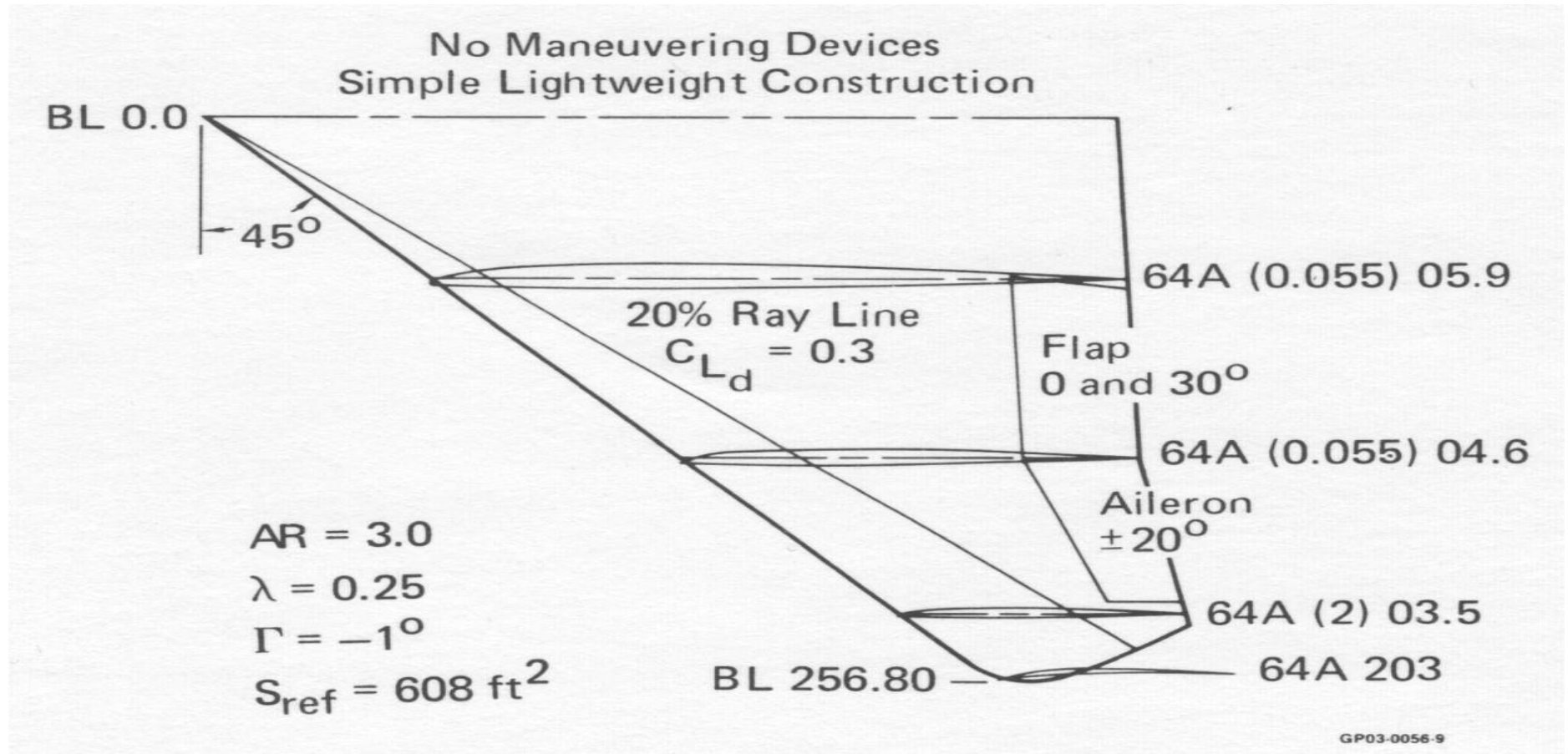
F-15 Wing Evolution



GP03-0056-6

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

F-15 Wing Lines



The wing has conical camber: hard to see in streamwise section cuts

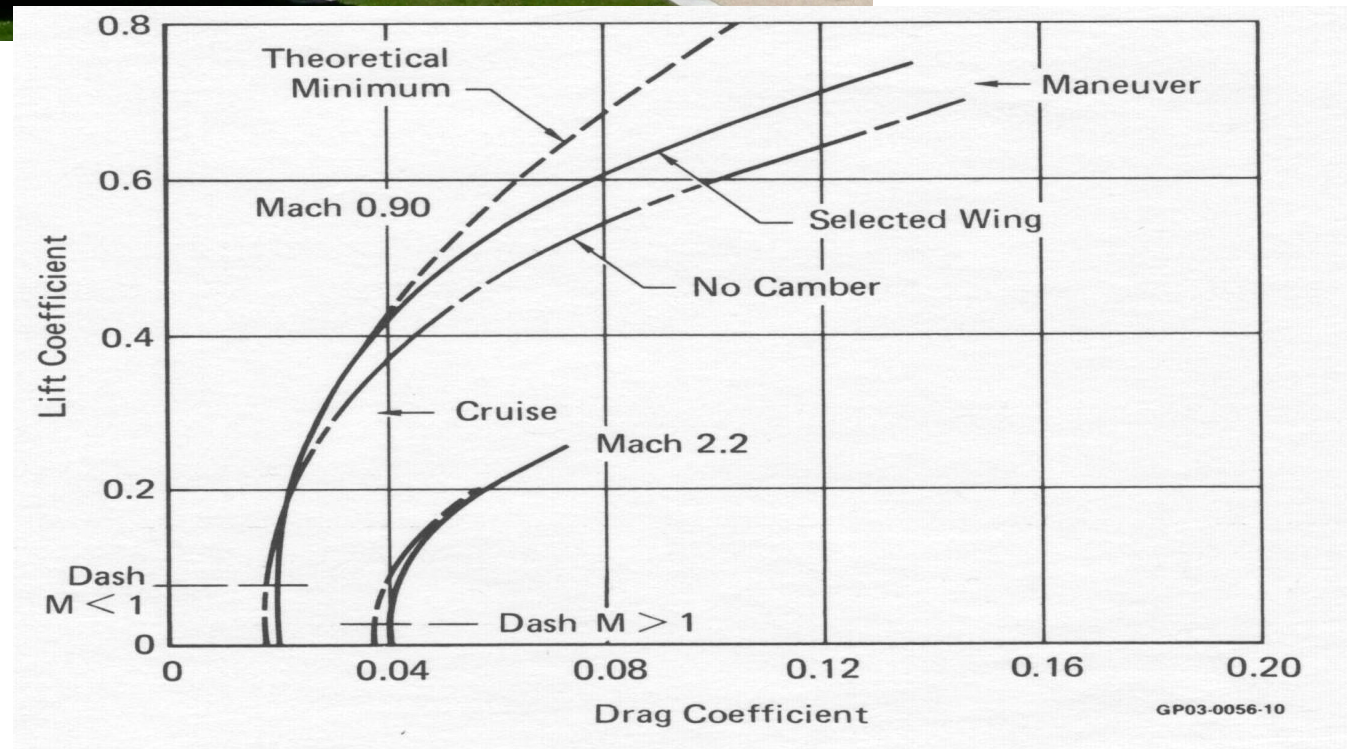
From Larry Niedling, "The F-15 Wing Development Program", AIAA Paper 80-3044

The final F-15

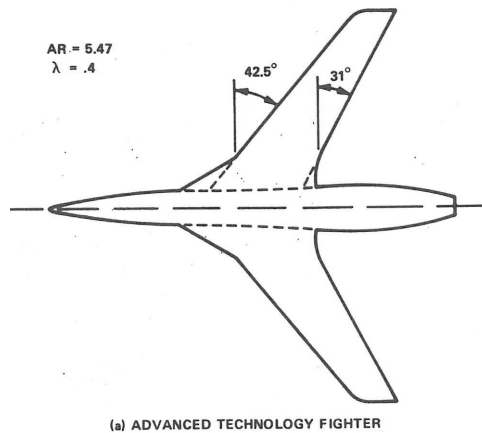


F-15 on display
at the U.S. Air
Force Academy

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



Example of attached flow fighter design



Grumman proposed
 HiMAT, $M = 0.90$

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.

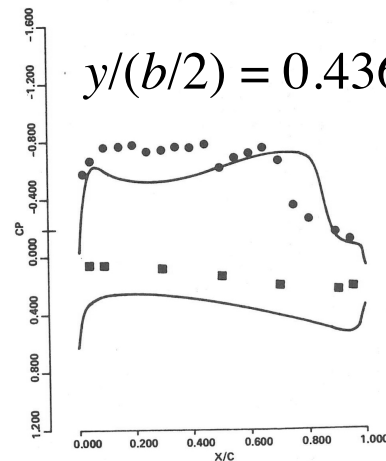


FIGURE 9b. ADVANCED FIGHTER
 $M = 0.900$ $ALF = 7.050$ $CONV = .886E-01$
 $LOCAL CL = 0.890$ $LOCAL CM = 1.306$ $LOCAL CD = 0.0585$
 SPAN STATION 13 $Y/S = 0.436$ 44 MESH PTS ON CHD

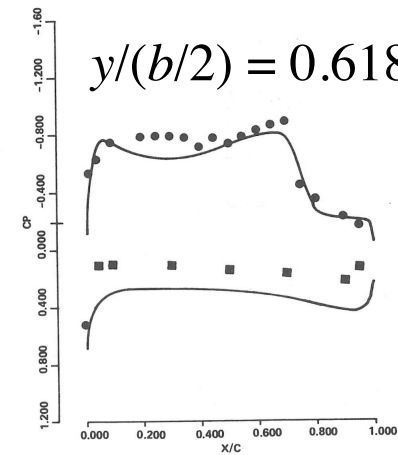


FIGURE 9c. ADVANCED FIGHTER
 $M = 0.900$ $ALF = 7.050$ $CONV = .886E-01$
 $LOCAL CL = 0.911$ $LOCAL CM = 1.218$ $LOCAL CD = 0.0357$
 SPAN STATION 18 $Y/S = 0.618$ 44 MESH PTS ON CHD

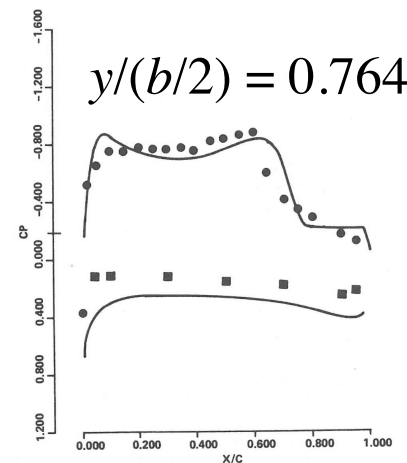


FIGURE 9d. ADVANCED FIGHTER
 $M = 0.900$ $ALF = 7.050$ $CONV = .886E-01$
 $LOCAL CL = 0.910$ $LOCAL CM = 1.075$ $LOCAL CD = 0.0206$
 SPAN STATION 22 $Y/S = 0.764$ 44 MESH PTS ON CHD

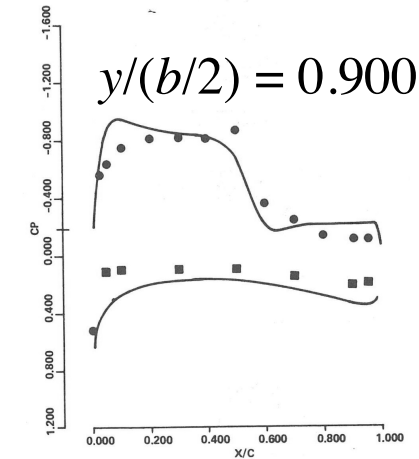
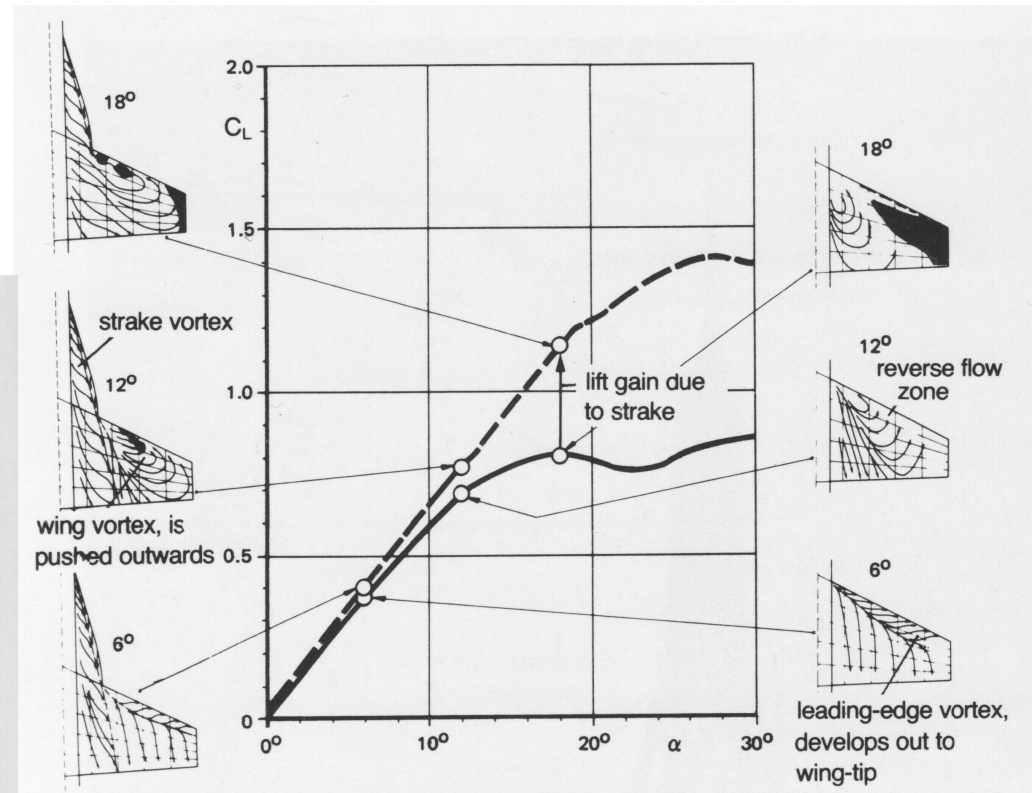
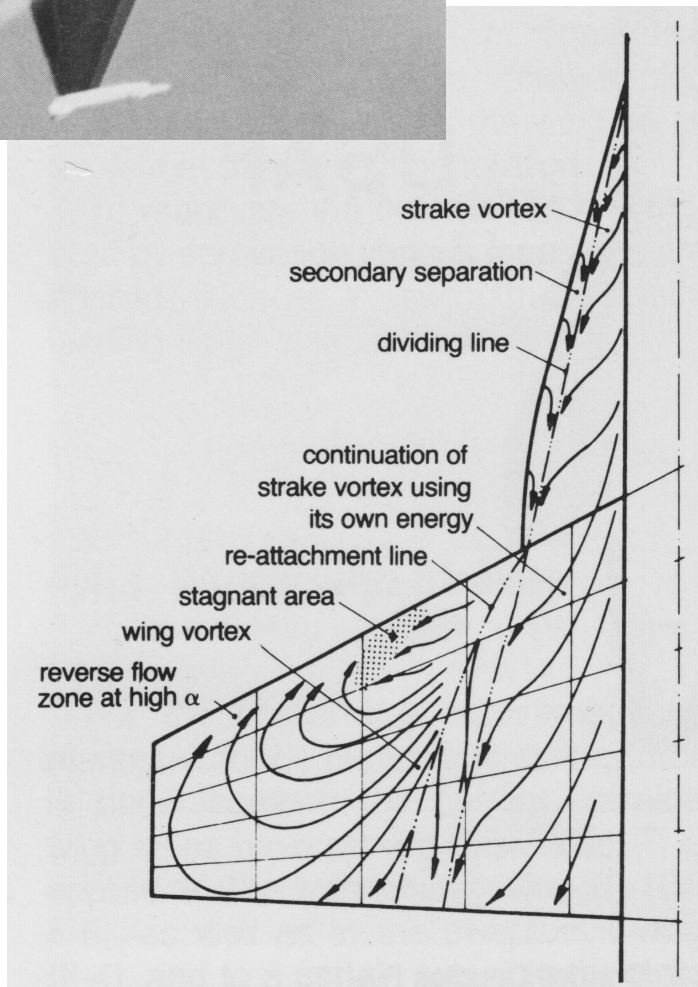


FIGURE 9e. ADVANCED FIGHTER
 $M = 0.900$ $ALF = 7.050$ $CONV = .886E-01$
 $LOCAL CL = 0.782$ $LOCAL CM = 0.766$ $LOCAL CD = -.0012$
 SPAN STATION 26 $Y/S = 0.909$ 44 MESH PTS ON CHD

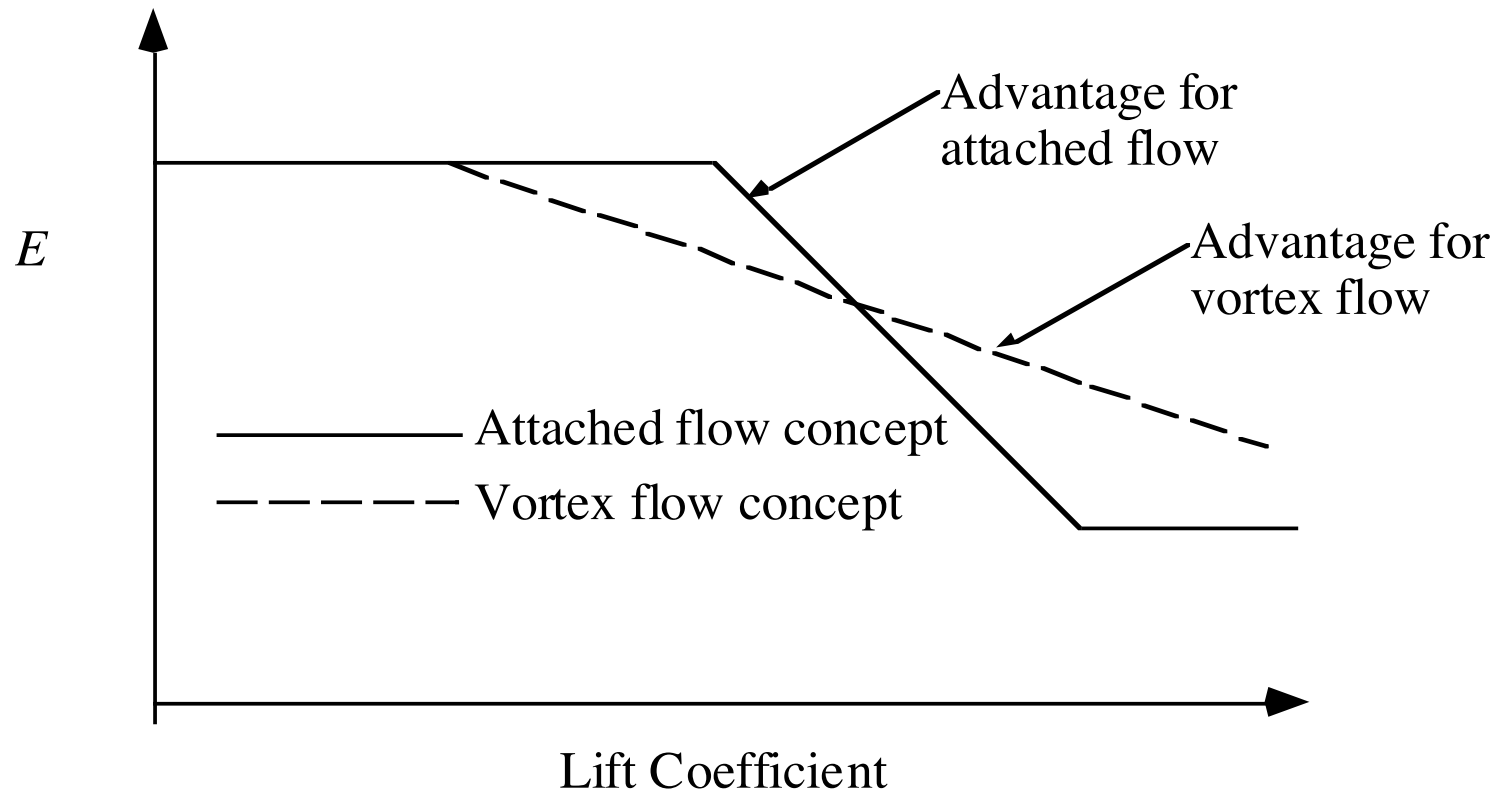
Hybrid Wings: the F-16 and F-18



A combined low aspect ratio strake and a traditional wing

From *Modern Combat Aircraft Design*, by Klaus Huenecke, 1987

Fighter Wings – Drag: Hybrid Wing Vortex Flow Effects Compared to Attached Flow Concepts



Vortex flow concepts have an advantage at high C_L s

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