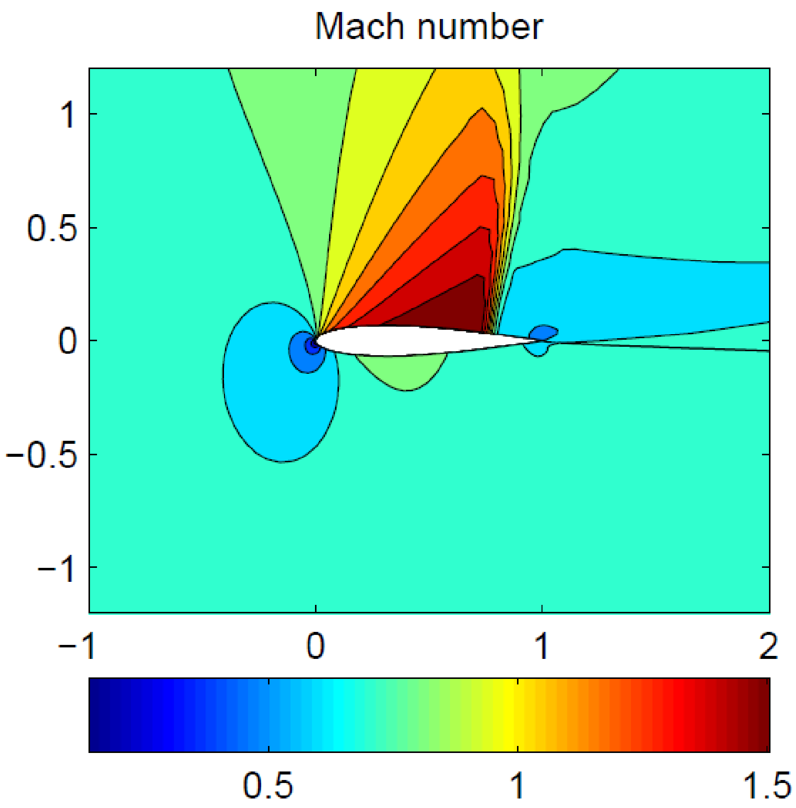
SESA3029 Aerothermodynamics



Lecture 4.2

Effect of Mach number on airfoil flow

Airfoil flows

- Objective: study the effect of Mach number on airfoil flow, including shock pattern
- Method: CFD solving the Euler equations
  - excludes viscous (boundary-layer) effects
  - finite volume method with shock waves captured by artificial dissipation

got to capture discontinuity without issues

we achive this using "artificial dissipation"

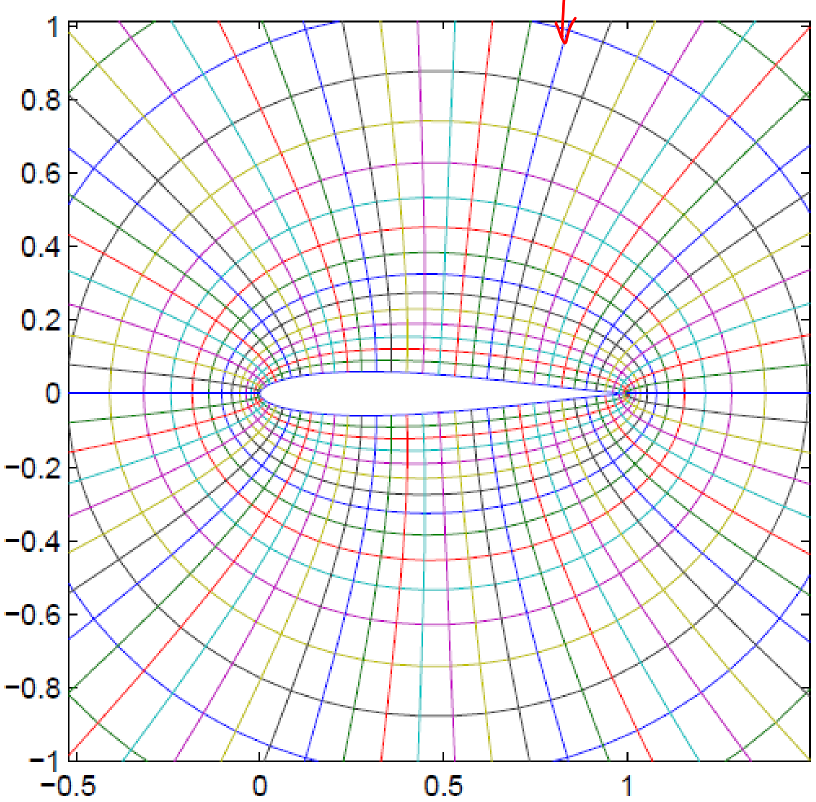
→ C-grid better for capturing wake

O-grid set up

12% thick Karman-Trefftz airfoil 51x101 grid points (grid generated by a conformal mapping)

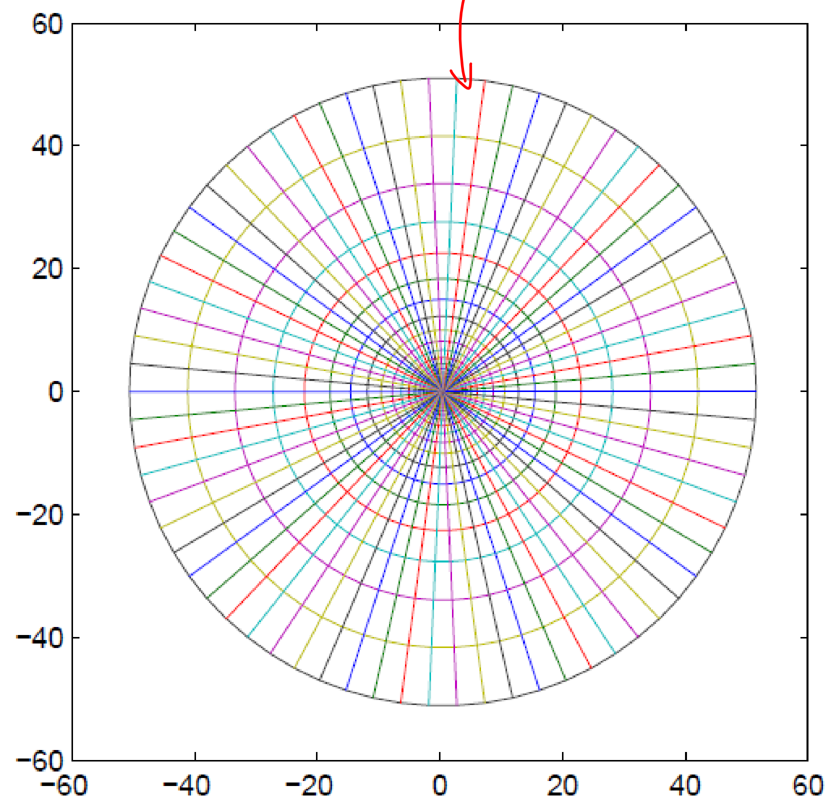
(a) close-up of airfoil

all gridlines are orthogonal



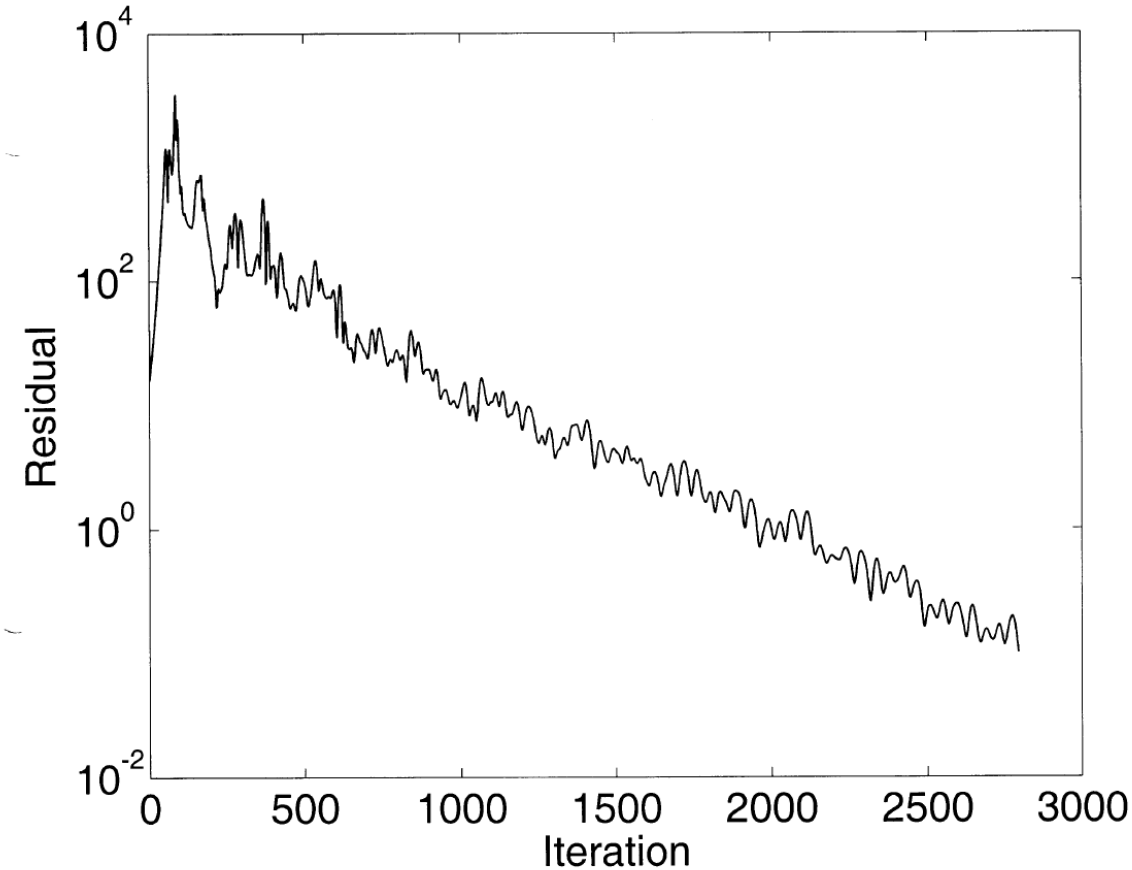
(b) whole grid

boundary is 50 chords away.



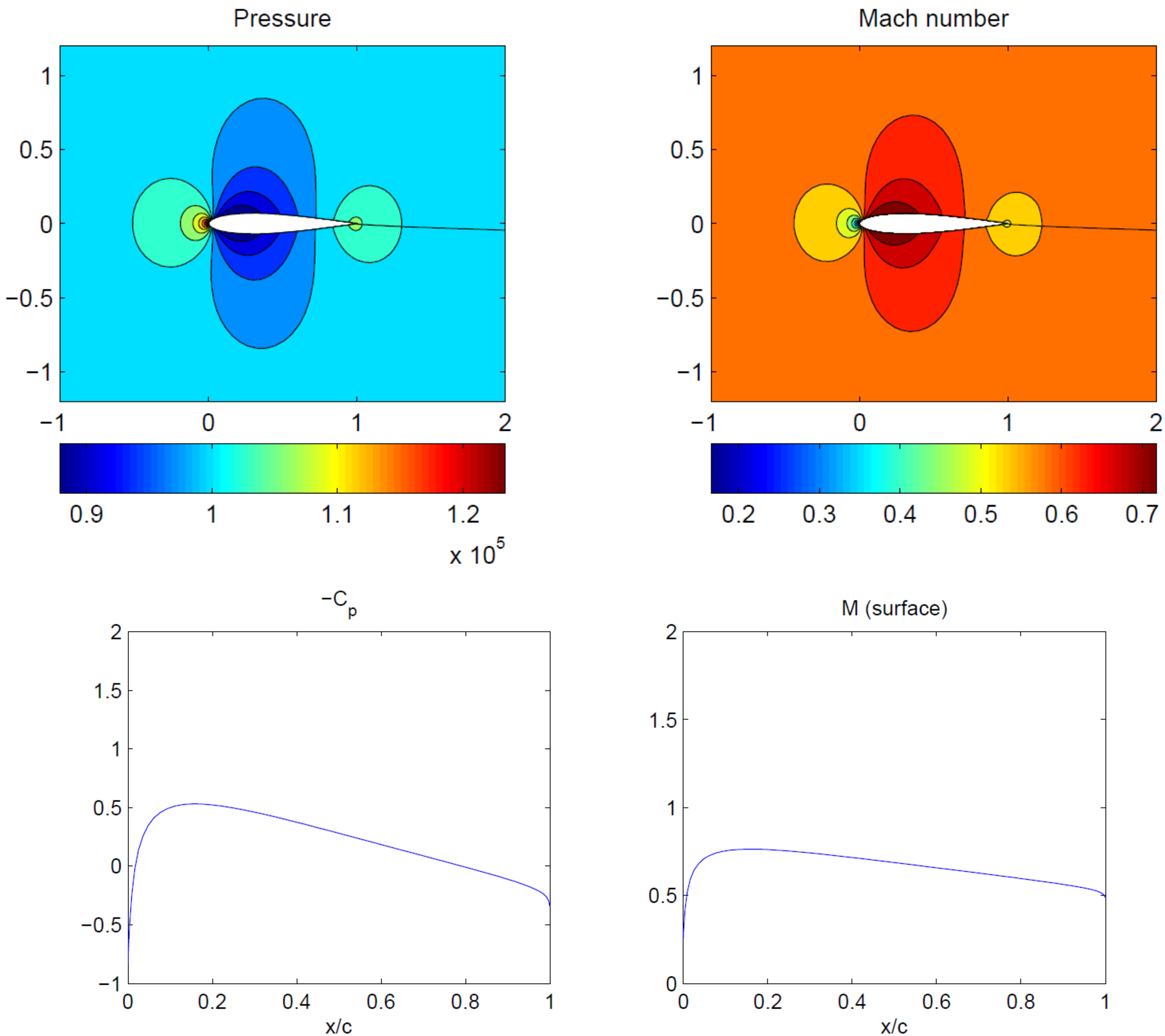
Iterate to convergence
(residual error → zero)

→ change in solution between steps.

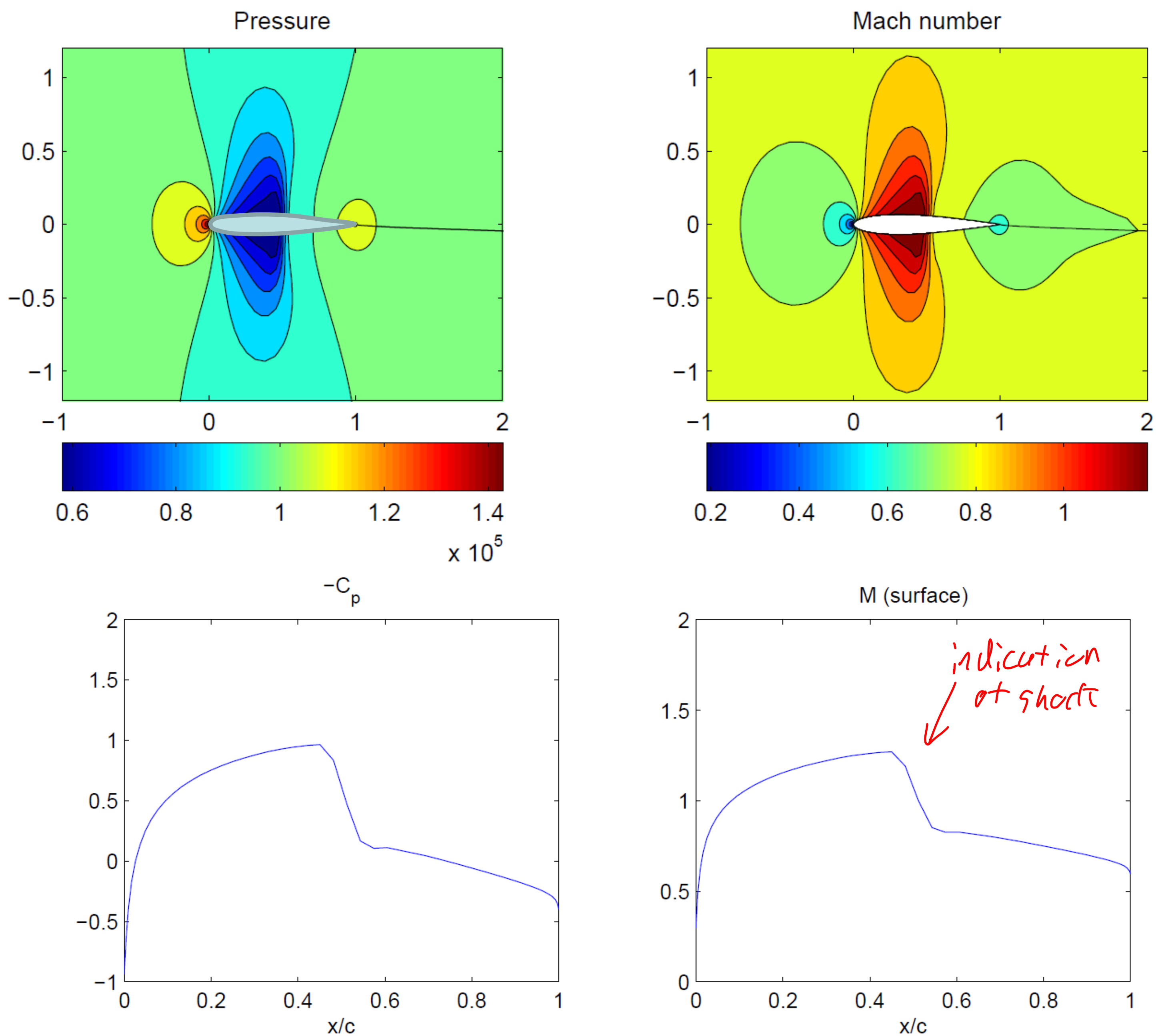


How many iterations do we need?

Alpha=0 M=0.6

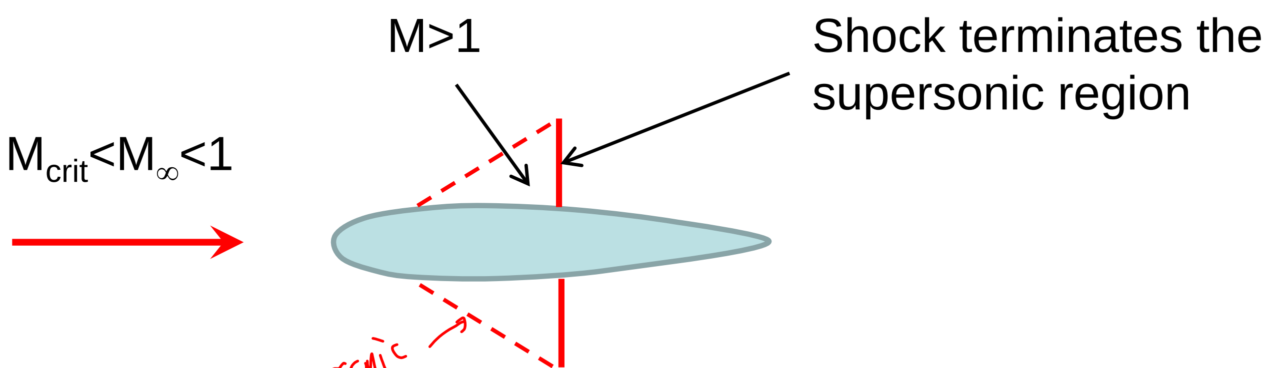


Alpha=0 M=0.8



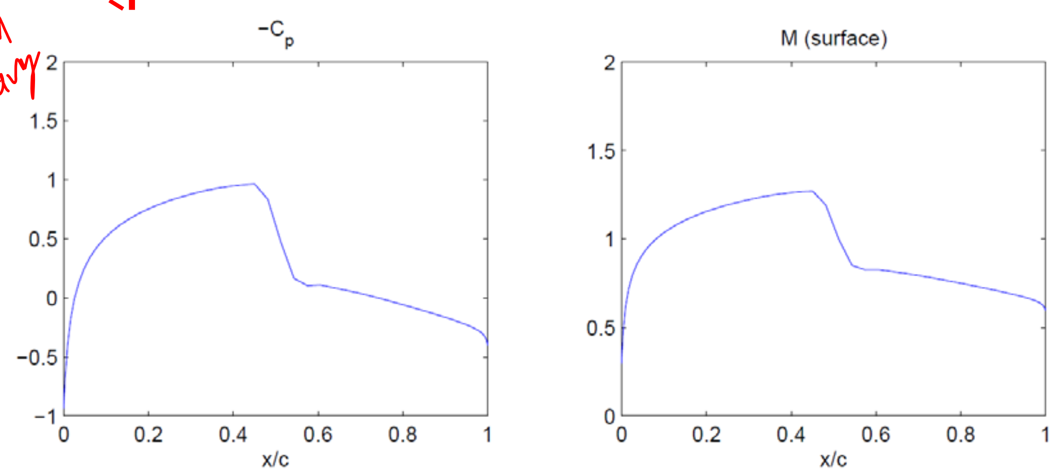
Pressure
Mach number
x 10^5
$-C_p$
M (surface)
x/c
x/c
indication of shock

In between the previous two cases, we have a conditions where the Mach number just reaches one on the surface: the critical M

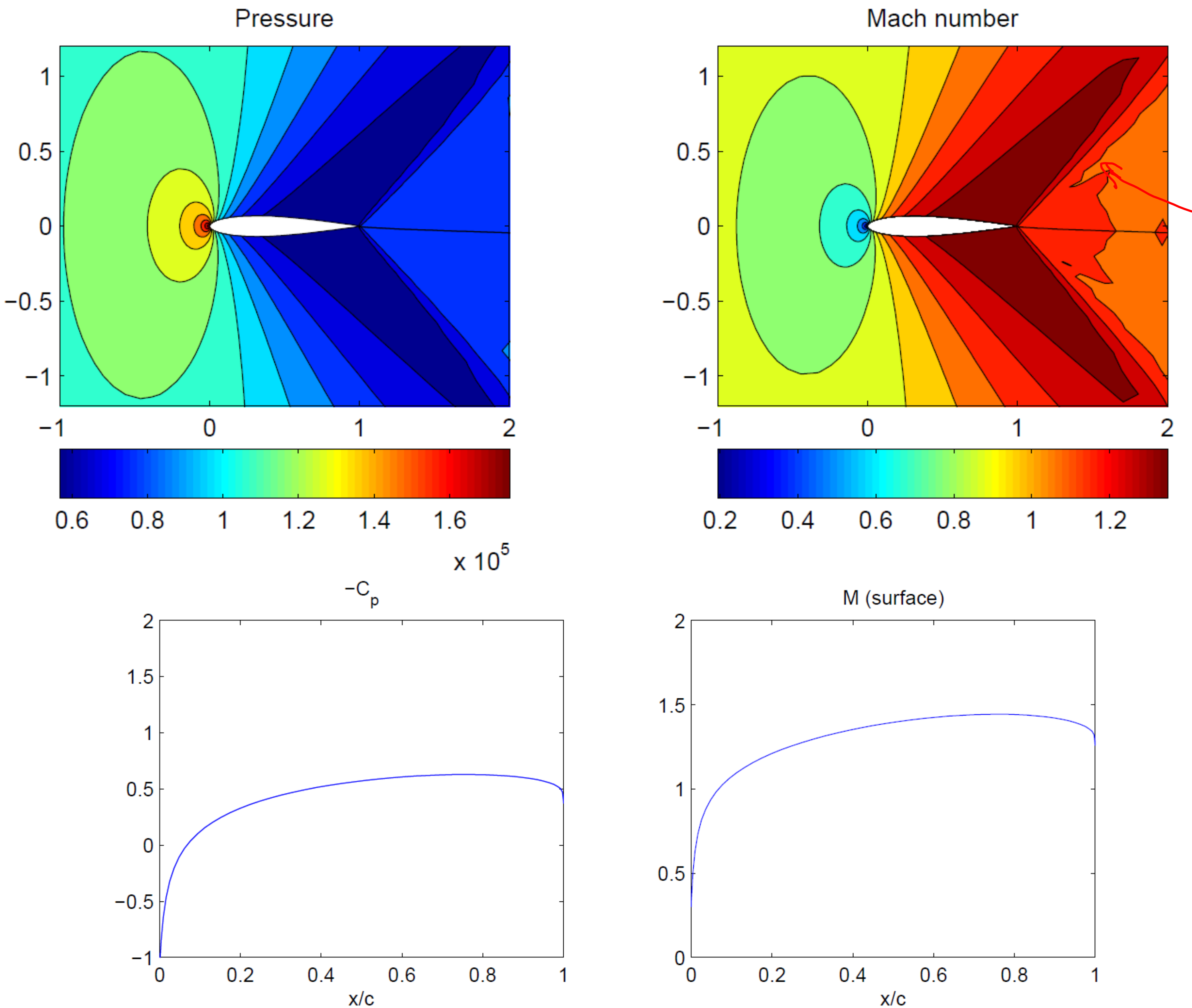


supersonic + rarefaction boundary

Transonic C_p and $M_{surface}$

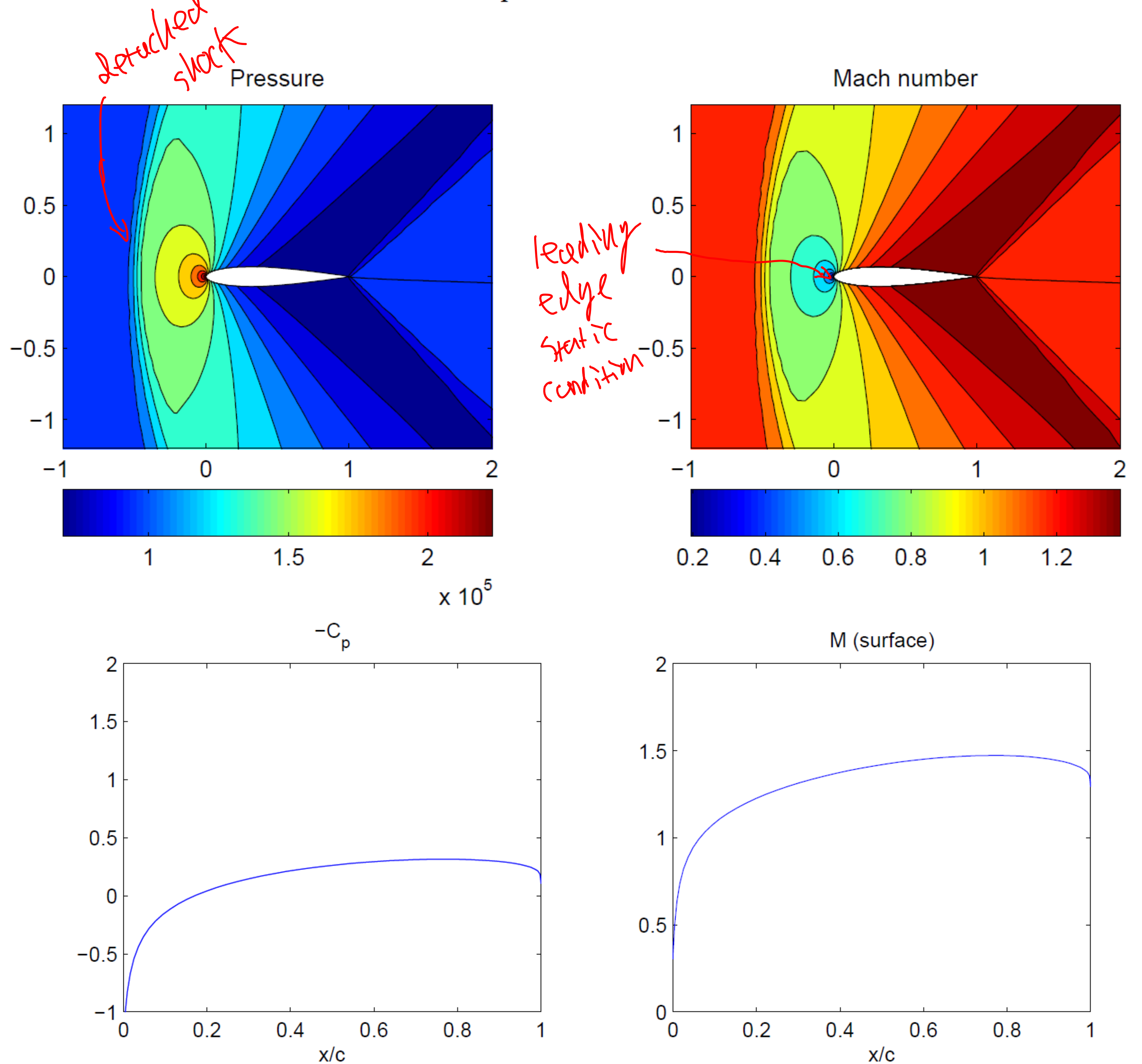


Alpha=0 M=1.0



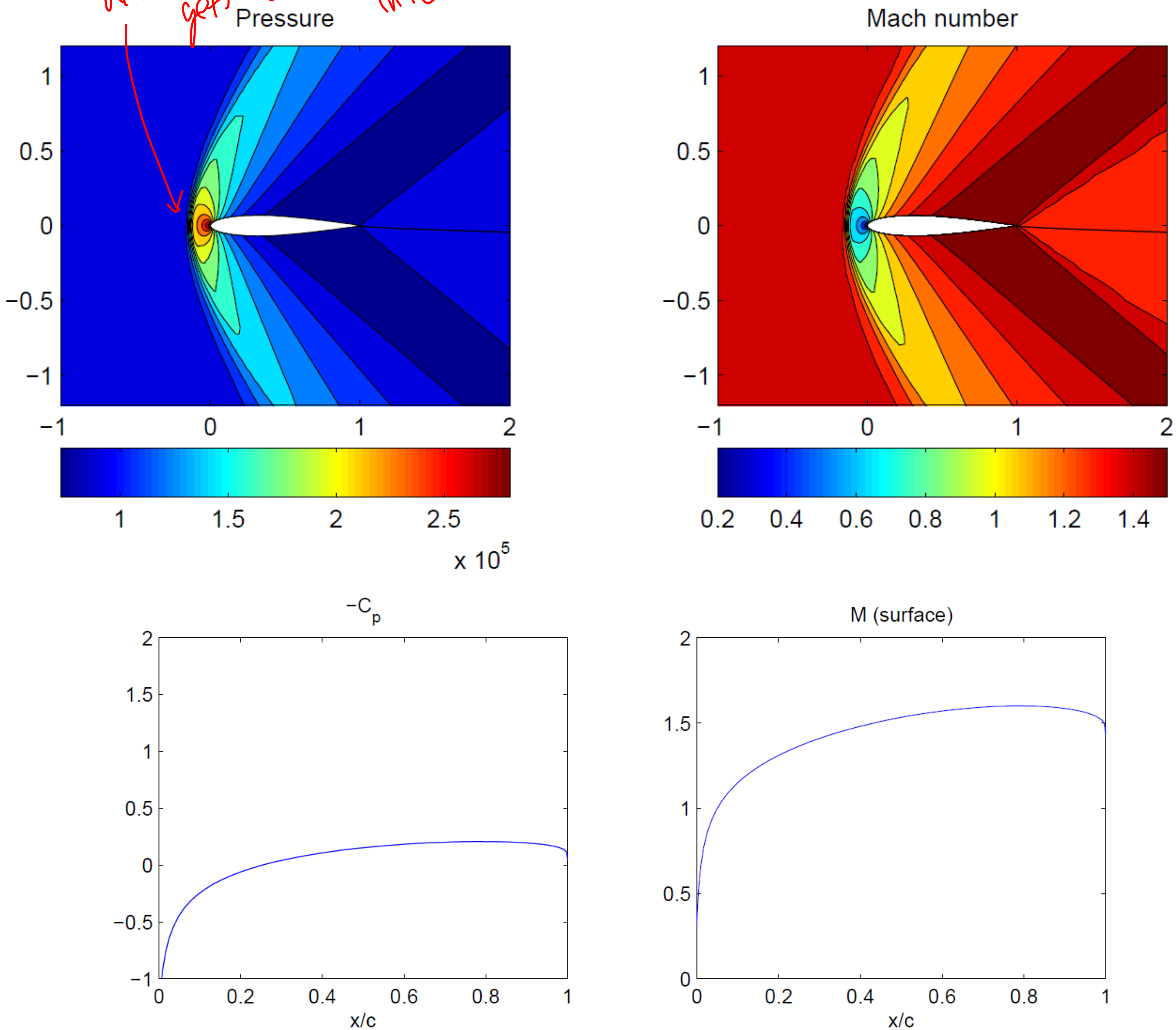
oblique trailing shockwave

Alpha=0 M=1.2

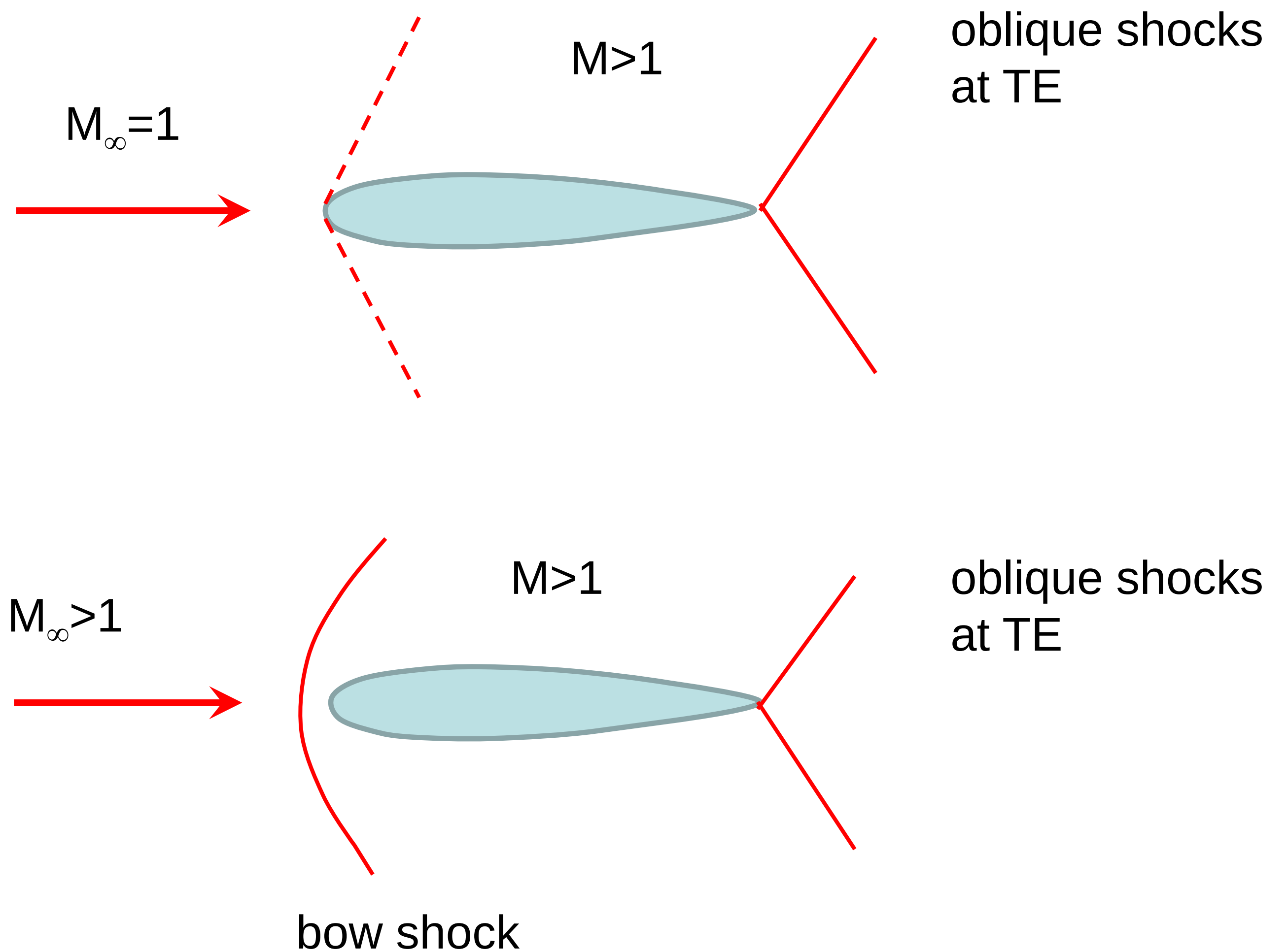

detached shock
Pressure
Mach number
leading edge static condition
x 10^5
−C_p
M (surface)
x/c
x/c

Alpha=0 M=1.4

detached shock gets closer and intensifies



Sonic and supersonic cases

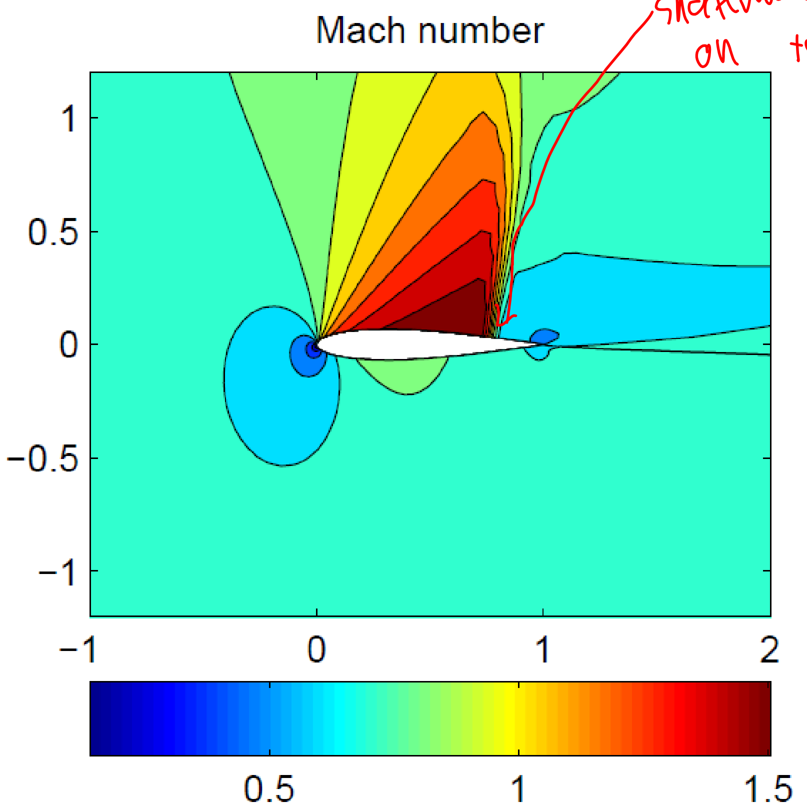


Notes (zero incidence)

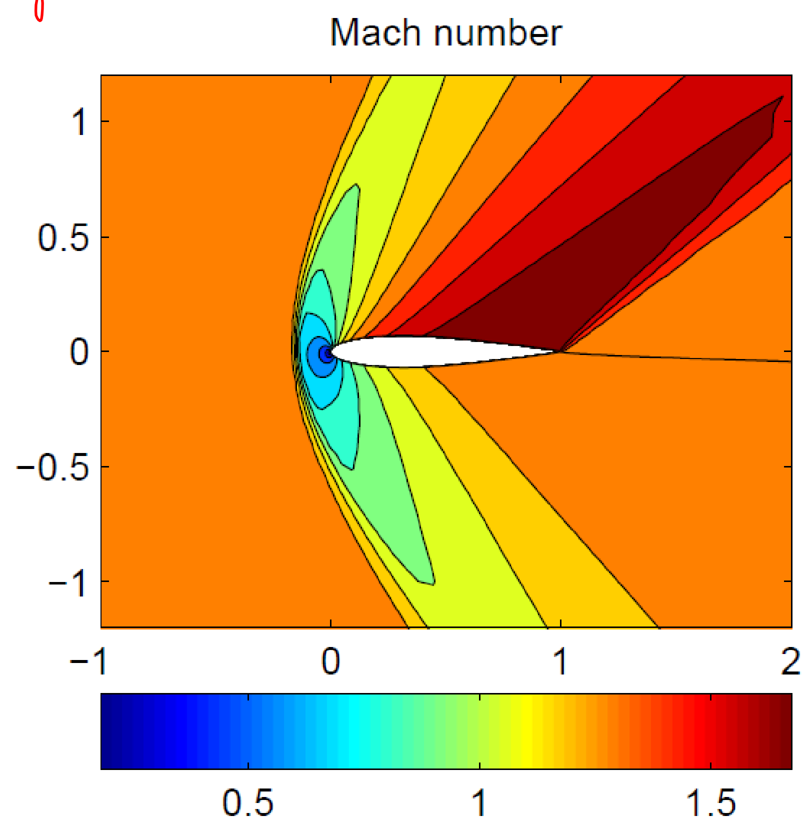
- $M_\infty < M_{crit}$ subsonic flow throughout
- $M_\infty = M_{crit}$ flow reaches M=1 at some point on the surface
- $M_{crit} < M_\infty < 1$ Supersonic flow region, terminated with shock wave
- $M_\infty > 1$ bow shock wave forms (closer to leading edge as M_∞ increases) and trailing edge shocks develop
  - Only region of subsonic flow is in front of leading edge

At 5.7° incidence

shockwave only on top.



Transonic flow (M_∞=0.8)

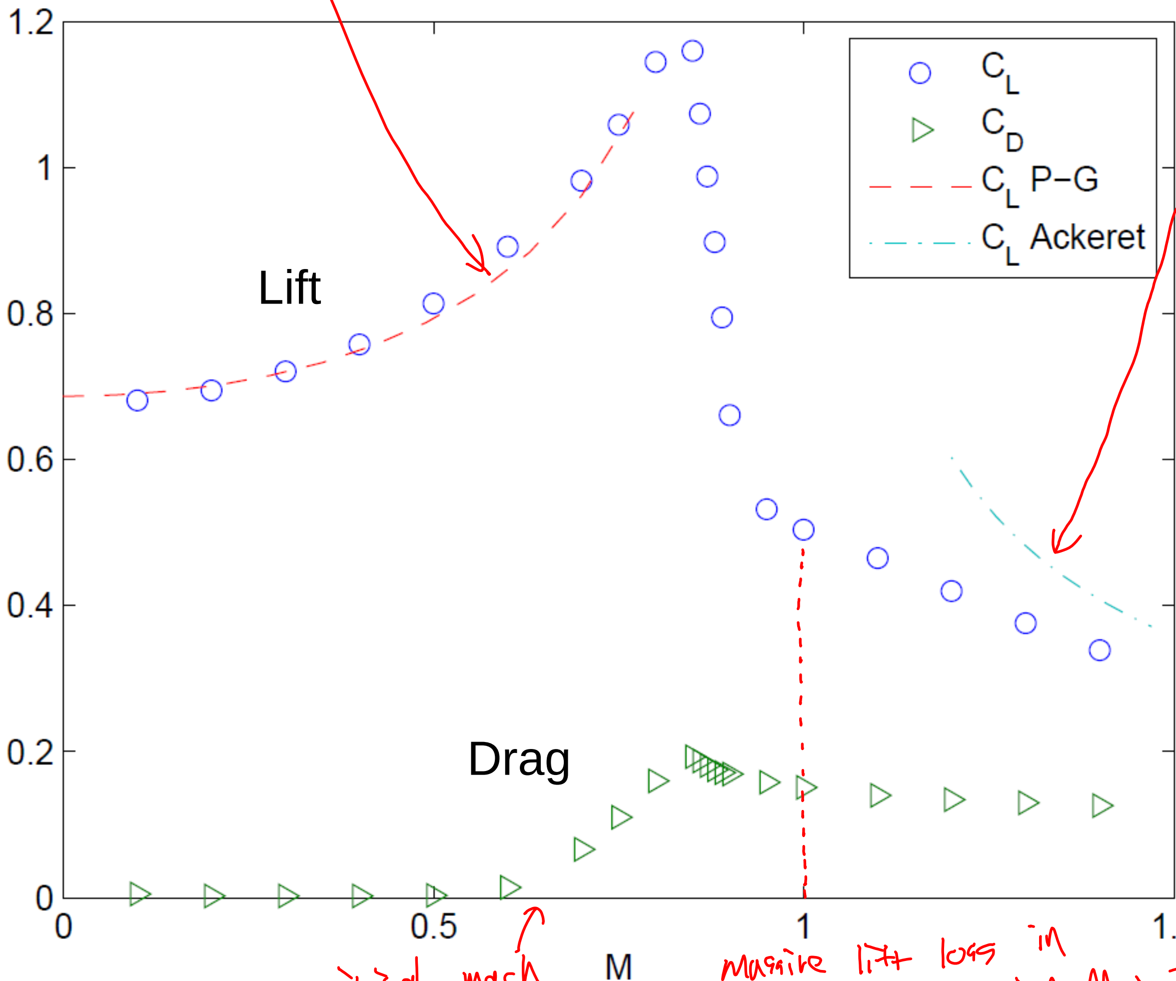


Supersonic flow (M_∞=1.4)

Lift and drag variations

Prandtl Glauert can predict decently

Prediction for thin aerofoils in supersonic foils. (this is thick which hurts accuracy)



critical mach number

massive lift loss in transonic region $M_{crit} \to 1$

Aerodynamic data at angle of attack 0.1 rad.

wave drag (invisid effect)

Lift rises in subsonic region – well predicted by Prandtl-Glauert theory (covered later in the module)

Lift falls in supersonic region, approximated by Ackeret theory for high M_∞

Drag-rise Mach number approximately 0.6.

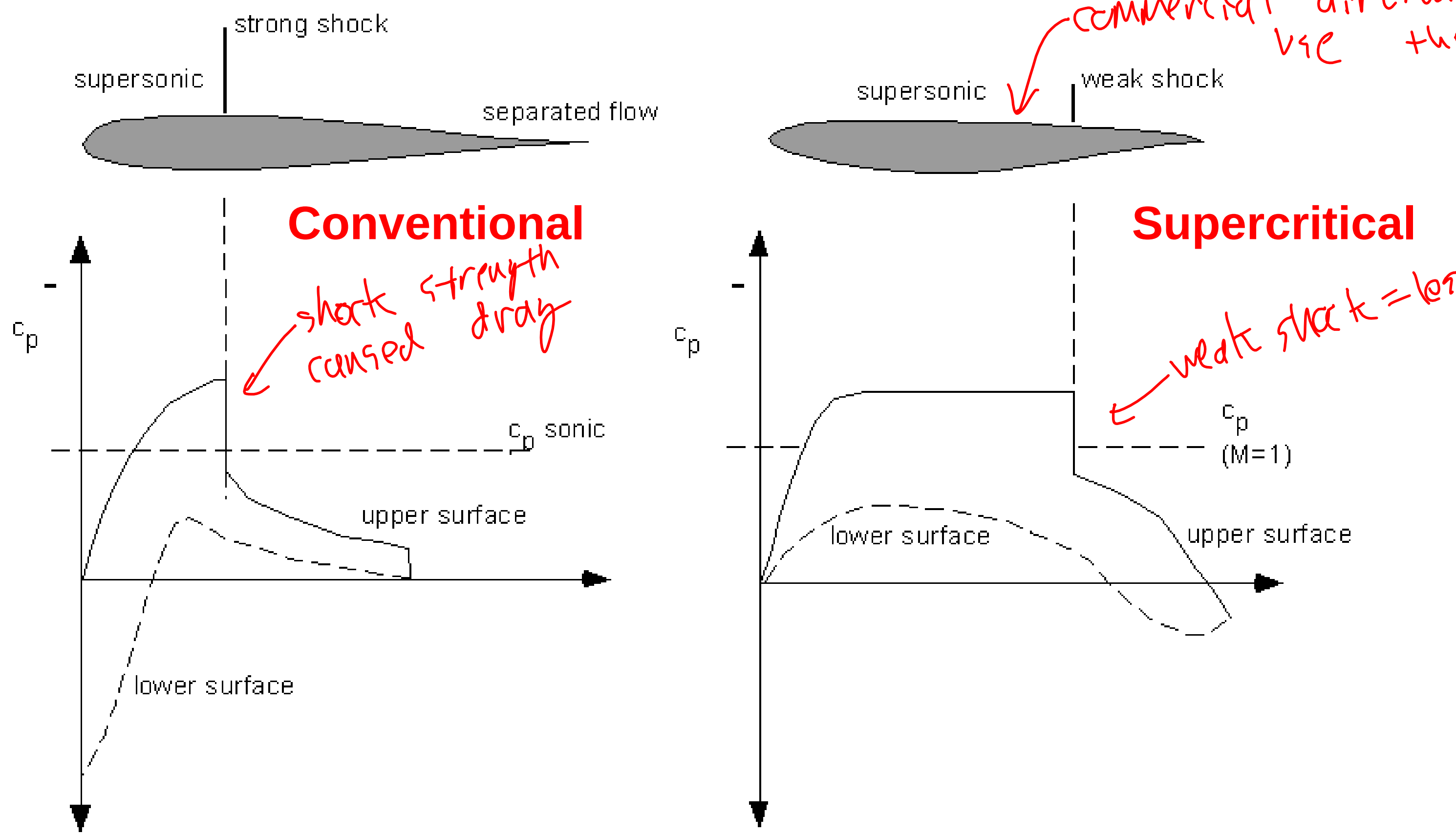
Wave drag rises steeply, peaking in transonic flow regime and reaching a plateau for $M_\infty > 1$

Drag reduction strategies

increase dragrise Mach

- Supercritical sections (for high subsonic flight)
  - Flat rooftop airfoil shape to reduce shock strength

commercial aircraft M~0.8 use these



shock strength caused drag

weak shock = less wave drag

From Bertin & Smith, 2nd Edition, p. 366

- Wing sweep

normal component to wing

  – Effective oncoming Mach number is $M_\infty \cos\beta$, where β is the leading edge sweep angle
  – Used in transonic and supersonic regions to reduce the effective Mach number seen by the airfoil section

F-111

- Area rule
  – Supersonic aircraft can reduce wave drag by keeping a smooth streamwise variation of total cross-sectional area (wings plus fuselage)

fuselage thins where wings start to reduce rate of cross section change

F-106