COMPRESSIBLE FLOW OVER AN AIRFOIL

INCOMPRESSIBLE FLOW: $\rho = \rho_\infty = \boldsymbol{const}$

: LIFT GENERATION, DRAG GENERATION, PRESSURE DISTRIBUTION …
$C_L(\alpha)$, $C_D(Cz)$, $Cmy(C_L)$, aerodynamic center ….

COMPRESSIBLE FLOW: $\rho = \boldsymbol{var}$

?

COMPRESSIBLE FLOW OVER AN AIRFOIL

Continuity equation:

$$\frac{\partial(\rho U)}{\partial \boldsymbol{x}} + \frac{\partial(\rho V)}{\partial \boldsymbol{y}} = \nabla \cdot \left(\rho \vec{\boldsymbol{V}}\right) = \boldsymbol{0}$$

INCOMPRESSIBLE FLOW: $\rho = \rho_\infty = \boldsymbol{const}$

$$\frac{\partial(\rho_\infty U)}{\partial \boldsymbol{x}} + \frac{\partial(\rho_\infty V)}{\partial \boldsymbol{y}} = \nabla \cdot \left(\rho_\infty \vec{\boldsymbol{V}}\right) = \rho_\infty \boldsymbol{div} \vec{\boldsymbol{V}} = \boldsymbol{0}$$

$$\boldsymbol{div} \vec{\boldsymbol{V}} = \boldsymbol{0}$$

COMPRESSIBLE FLOW: $\rho = \boldsymbol{var}$

$$\frac{\partial(\rho U)}{\partial \boldsymbol{x}} + \frac{\partial(\rho V)}{\partial \boldsymbol{y}} = \nabla \cdot \left(\rho \vec{\boldsymbol{V}}\right) = \rho\, \boldsymbol{div} \vec{\boldsymbol{V}} + \vec{\boldsymbol{V}} \cdot \boldsymbol{grad}\ \rho = \boldsymbol{0}$$

$$\boldsymbol{div} \vec{\boldsymbol{V}} = -\frac{\vec{\boldsymbol{V}} \cdot \boldsymbol{grad}\ \rho}{\rho} \neq \boldsymbol{0}$$

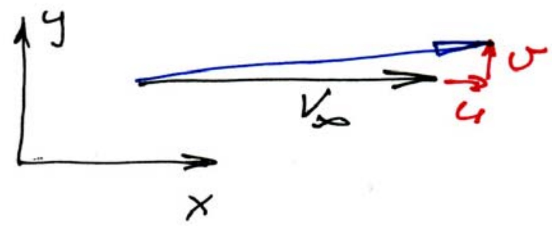
$$\boldsymbol{div\vec{V}} \quad :$$

$$\iiint_{\Omega} div\vec{V}\ d\Omega = \oiint_{\delta\Omega} \vec{V}\cdot\vec{n}\ dS$$

$$div\vec{V} = \frac{VOLUME\ \ OUTFLOW\ \ RATE\ \ FROM\ THE\ \ CONTROL\ \ ELEMENT}{VOLUME\ OF\ THE\ CONTROL\ ELEMENT}$$

RELATIVE GAS VOLUME RATE OF CHANGE IN CONTROL ELEMENT

SUBSONIC FLOW OVER AN AIRFOIL. SMALL DISTURBANCE THEORY



$$u, v \ll U_\infty$$

Potential flow: φ - perturbation velocity potential

$$U = U_\infty + u = U_\infty + \frac{\partial \varphi}{\partial x}$$

$$V = \qquad v = \qquad \frac{\partial \varphi}{\partial y}$$

PROCEDURE:

Solve continuity eq. + boundary + additional conditions → flow field
Find pressure field using Bernouli eq. (momentum eq.!)
Find aerodynamic forces and moments (pressure forces)

Continuity equation:

$$\frac{\partial(\rho U)}{\partial x}+\frac{\partial(\rho V)}{\partial y}=0=(U_\infty+\cancel{u})\frac{\partial\rho}{\partial x}+\rho\frac{\partial u}{\partial x}+\cancel{v}\frac{\partial\rho}{\partial y}+\rho\frac{\partial v}{\partial y}\simeq$$

$$U_\infty\frac{\partial\rho}{\partial x}+\rho\frac{\partial u}{\partial x}+\rho\frac{\partial v}{\partial y}=U_\infty\frac{\partial\rho}{\partial x}+\rho\, div\,\vec{v}=$$

$$\left|S=const\right|=U_\infty\left.\frac{\partial\rho}{\partial p}\right|_\infty\frac{\partial p}{\partial x}+\rho\, div\,\vec{v}=$$

$$\left|\begin{array}{l}\textbf{Euler eq.}(x):\quad \rho U\dfrac{\partial U}{\partial x}+\rho V\dfrac{\partial U}{\partial y}=-\dfrac{\partial p}{\partial x}\\[2em] \qquad\qquad \rho(U_\infty+\cancel{u})\dfrac{\partial u}{\partial x}+\rho\cancel{v}\dfrac{\partial u}{\partial y}=-\dfrac{\partial p}{\partial x}\quad\rightarrow\quad\dfrac{\partial p}{\partial x}\simeq-\rho U_\infty\dfrac{\partial u}{\partial x}\end{array}\right|$$

$$=-\rho\frac{U_\infty^2}{a_\infty^2}\frac{\partial u}{\partial x}+\rho\, div\,\vec{v}=-\rho\frac{U_\infty^2}{a_\infty^2}\frac{\partial u}{\partial x}+\rho\frac{\partial u}{\partial x}+\rho\frac{\partial v}{\partial y}=\rho\left(\left(1-Ma_\infty^2\right)\frac{\partial u}{\partial x}+\frac{\partial v}{\partial y}\right)=0$$

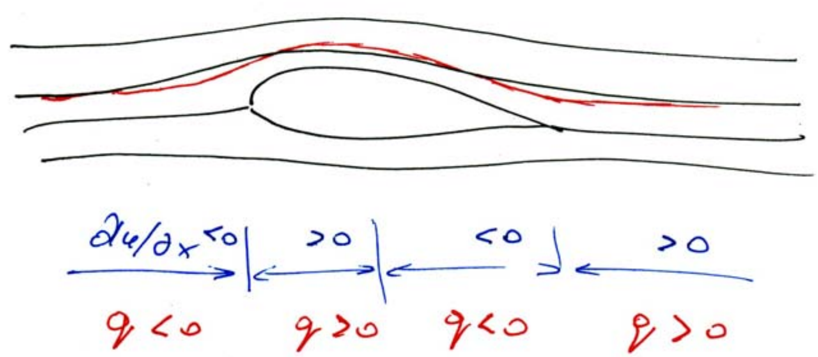
$$\left(1 - Ma_\infty^2\right)\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = \left(1 - Ma_\infty^2\right)\frac{\partial^2 \varphi}{\partial x^2} + \frac{\partial^2 \varphi}{\partial y^2} =$$

$$\boxed{\beta^2 \frac{\partial^2 \varphi}{\partial x^2} + \frac{\partial^2 \varphi}{\partial y^2} = 0 \;; \qquad \beta = \sqrt{1 - Ma_\infty^2}}$$

Prandtl-Glauert eq. (..continuity)

or:

$$div\vec{V} = q = Ma_\infty^2 \frac{\partial u}{\partial x}$$



BERNOULI EQUATION – SMALL DISTURBANCES

$$\textbf{\textit{Euler Eq.(x)}}: \quad \rho U \frac{\partial U}{\partial x} + \rho V \frac{\partial U}{\partial y} = -\frac{\partial p}{\partial x}$$

$$\left(\rho_\infty + \cancel{\rho'}\right)\left(U_\infty + \cancel{u}\right)\frac{\partial u}{\partial x} + \left(\rho_\infty + \cancel{\rho'}\right)\cancel{v}\frac{\partial u}{\partial y} = -\frac{\partial p}{\partial x} \quad \rightarrow \quad \frac{\partial p}{\partial x} \simeq -\rho_\infty U_\infty \frac{\partial u}{\partial x}$$

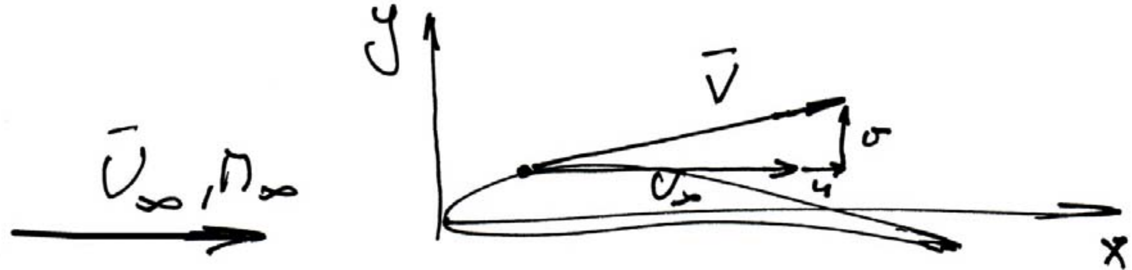
Integrating

$$\rho_\infty U_\infty \underbrace{\Delta U}_{U-U_\infty} = -\underbrace{\Delta p}_{p-p_\infty}$$

$$p - p_\infty = \rho_\infty U_\infty u$$

$$\boxed{Cp = \frac{p - p_\infty}{\rho_\infty U_\infty^2 / 2} = \frac{\rho_\infty U_\infty u}{\rho_\infty U_\infty^2 / 2} = -2u / U_\infty}$$

linearized Bernouli equation



$$y_B(x) = F(x)$$

$$\beta^2 \frac{\partial^2 \varphi}{\partial x^2} + \frac{\partial^2 \varphi}{\partial y^2} = 0 \;; \qquad \frac{\partial \varphi}{\partial y} = v = (U_\infty + u)F' \simeq U_\infty F'$$

$$\downarrow$$

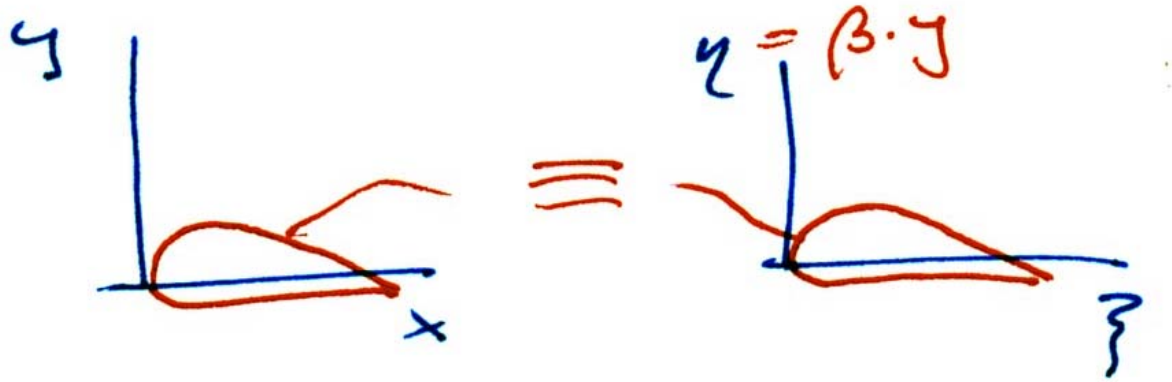
$$\varphi \rightarrow \vec{V} = \vec{V}_\infty + \nabla\varphi \rightarrow p \rightarrow Cp \rightarrow C_L, C_m$$

$$\xi = x\,, \quad \eta = \beta y\,, \quad \eta_B(\xi) = F(\xi)$$

$$\Phi = \lambda\,\varphi$$

$$\beta^2 \frac{\partial^2 \varphi}{\partial x^2} + \frac{\partial^2 \varphi}{\partial y^2} = 0 \;\rightarrow\; \frac{\partial^2 \Phi}{\partial \xi^2} + \frac{\partial^2 \Phi}{\partial \eta^2} = 0$$

$$\frac{\partial \varphi}{\partial y} = (U_\infty + u) F' \simeq U_\infty F' \quad \rightarrow \quad \frac{\partial \Phi}{\partial \eta} = (U_\infty + \tilde{u}) F' \simeq U_\infty F'$$

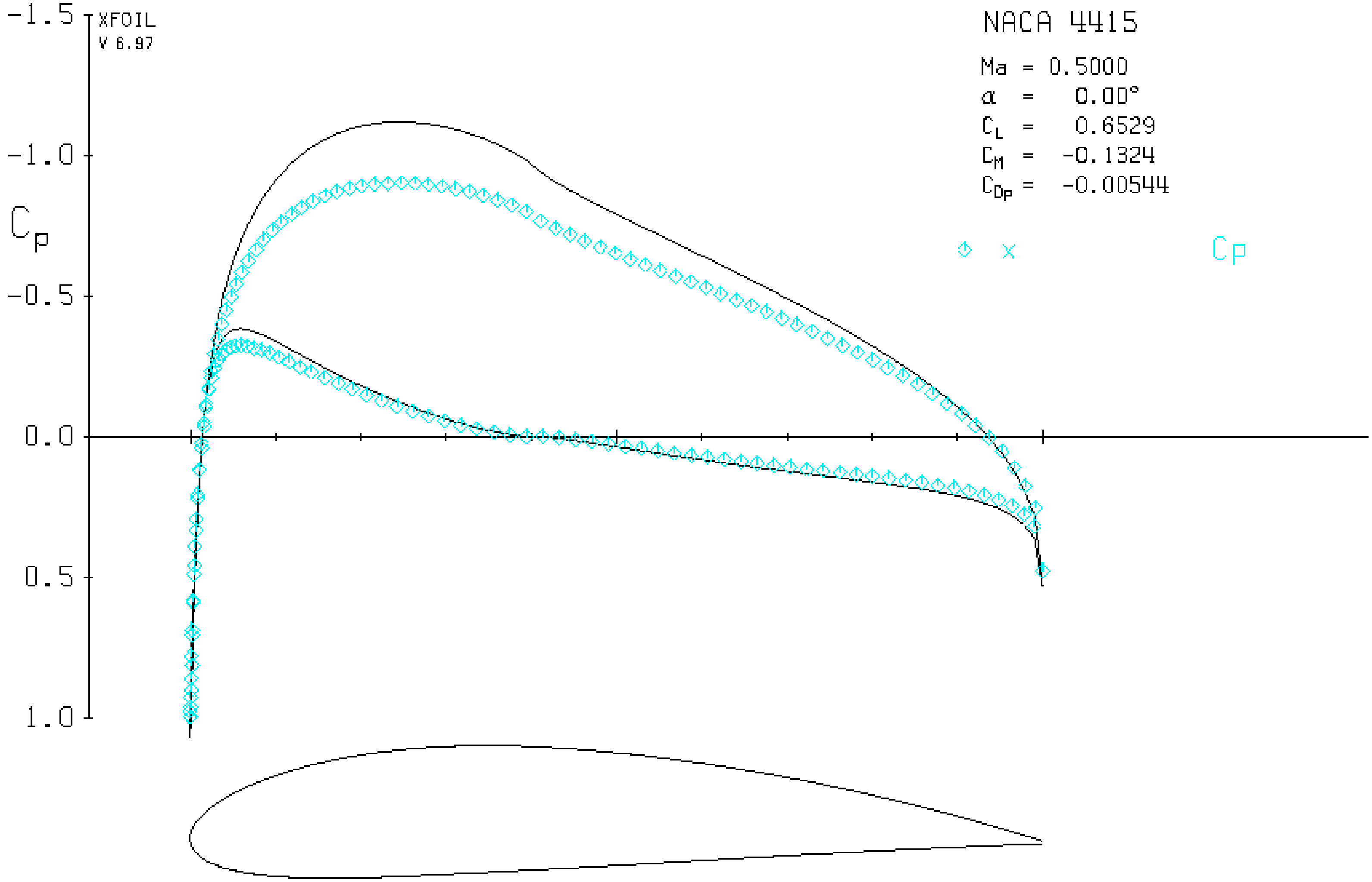


$$\frac{\partial \varphi}{\partial \boldsymbol{y}} = \left(\boldsymbol{U}_\infty + \boldsymbol{u}\right)\boldsymbol{F}' \simeq \boldsymbol{U}_\infty \boldsymbol{F}' \quad \rightarrow \quad \boldsymbol{U}_\infty \boldsymbol{F}' \simeq \frac{\partial \Phi}{\partial \eta} = \frac{\partial\left(\lambda \varphi\right)}{\partial\left(\beta \boldsymbol{y}\right)} = \underbrace{\left(\frac{\lambda}{\beta}\right)}_{\equiv 1}\frac{\partial \varphi}{\partial \boldsymbol{y}} \simeq \boldsymbol{U}_\infty \boldsymbol{F}'$$

$\lambda = \beta$ $\varphi = \Phi / \beta$

$$\boldsymbol{Cp} = -2\frac{\boldsymbol{u}}{\boldsymbol{U}_\infty} = -2\frac{\partial \varphi}{\partial \boldsymbol{x}} / \boldsymbol{U}_\infty = -2\frac{\partial\left(\Phi/\beta\right)}{\partial \xi} / \boldsymbol{U}_\infty = \frac{1}{\beta}\left(-\right)2\frac{\partial \Phi}{\partial \xi} / \boldsymbol{U}_\infty = \frac{1}{\beta}\boldsymbol{Cp}_{INC}$$

$$\boldsymbol{Cp} = \frac{1}{\beta}\boldsymbol{Cp}_{INC} \qquad \boldsymbol{C}_L = \frac{1}{\beta}\boldsymbol{C}_{L\ INC} \qquad \boldsymbol{Cm} = \frac{1}{\beta}\boldsymbol{Cm}_{INC}$$



-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
XFOIL
V 6.97
NACA 4415
Ma = 0.5000
α = 0.00°
CL = 0.6529
CM = -0.1324
CDp = -0.00544
Cp

$$U_\infty \quad p_\infty \quad \rho_\infty \quad T_\infty \quad a_\infty \quad Ma_\infty \quad \longrightarrow \quad U \quad p \quad \rho \quad T \quad a \quad Ma$$

$$S = const. \qquad p\left(1+\frac{k-1}{2}Ma^2\right)^{\frac{k}{k-1}} = p_0 = const.$$

$$p\left(1+\frac{k-1}{2}Ma^2\right)^{\frac{k}{k-1}} = p_\infty\left(1+\frac{k-1}{2}Ma_\infty^2\right)^{\frac{k}{k-1}} \quad \rightarrow \quad \frac{p}{p_\infty} = \left(\frac{1+\frac{k-1}{2}Ma_\infty^2}{1+\frac{k-1}{2}Ma^2}\right)^{\frac{k}{k-1}}$$

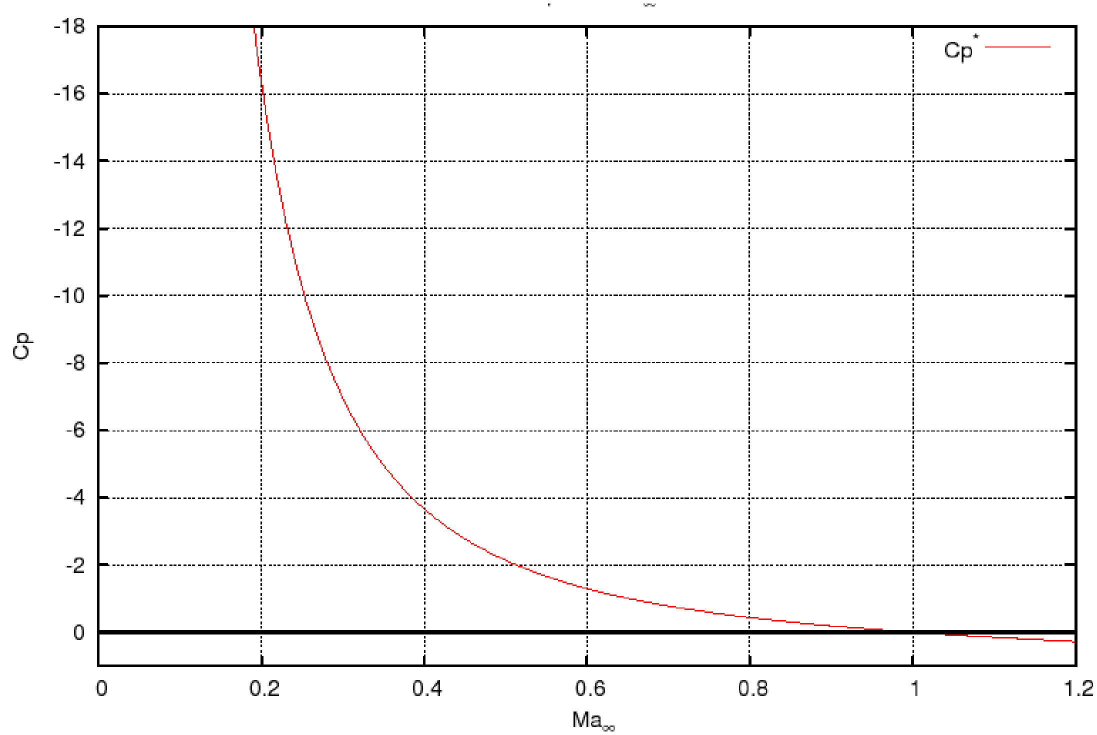
$$Cp = \frac{p-p_\infty}{\rho_\infty U_\infty^2/2} = \frac{p_\infty}{\rho_\infty}\frac{\left(\frac{p}{p_\infty}-1\right)}{\frac{1}{2}U_\infty^2} = kRT_\infty\frac{\left(\frac{1+\frac{k-1}{2}Ma_\infty^2}{1+\frac{k-1}{2}Ma^2}\right)^{\frac{k}{k-1}}-1}{\frac{k}{2}U_\infty^2} = \frac{\left(\frac{1+\frac{k-1}{2}Ma_\infty^2}{1+\frac{k-1}{2}Ma^2}\right)^{\frac{k}{k-1}}-1}{\frac{k}{2}Ma_\infty^2}$$

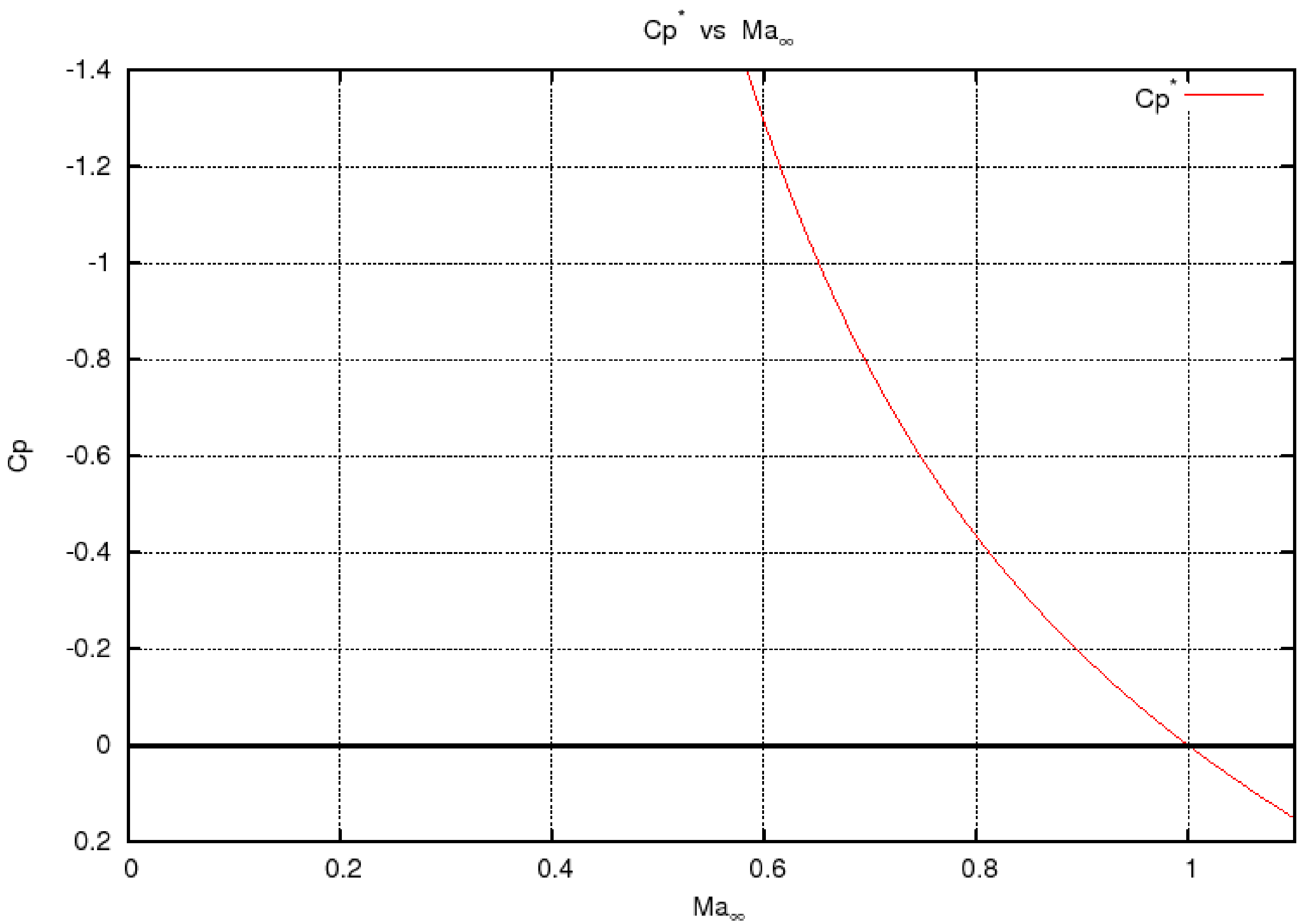
Ma_∞ → $Ma = 1$

p_∞ → p^*

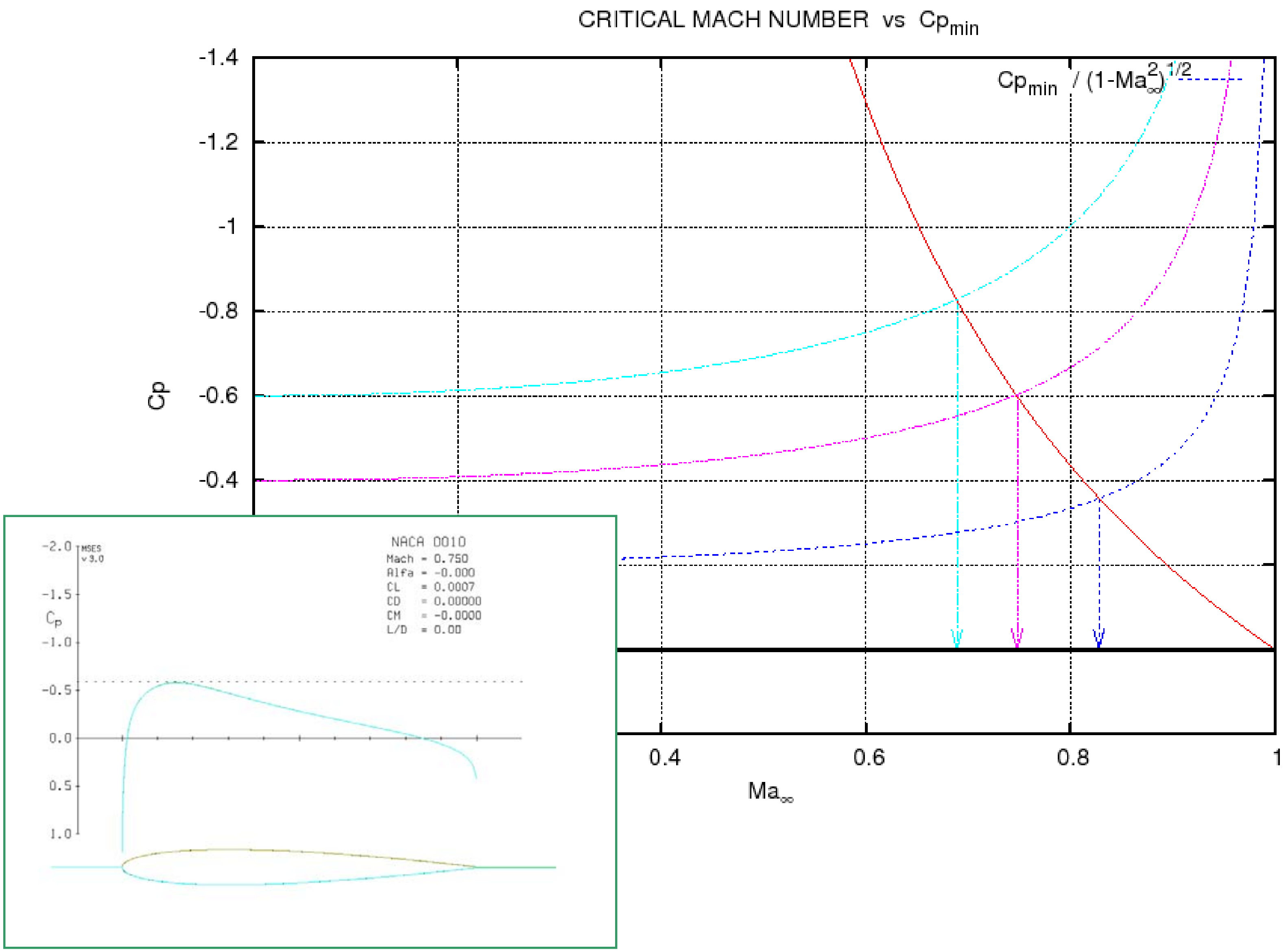
U_∞ → U^*

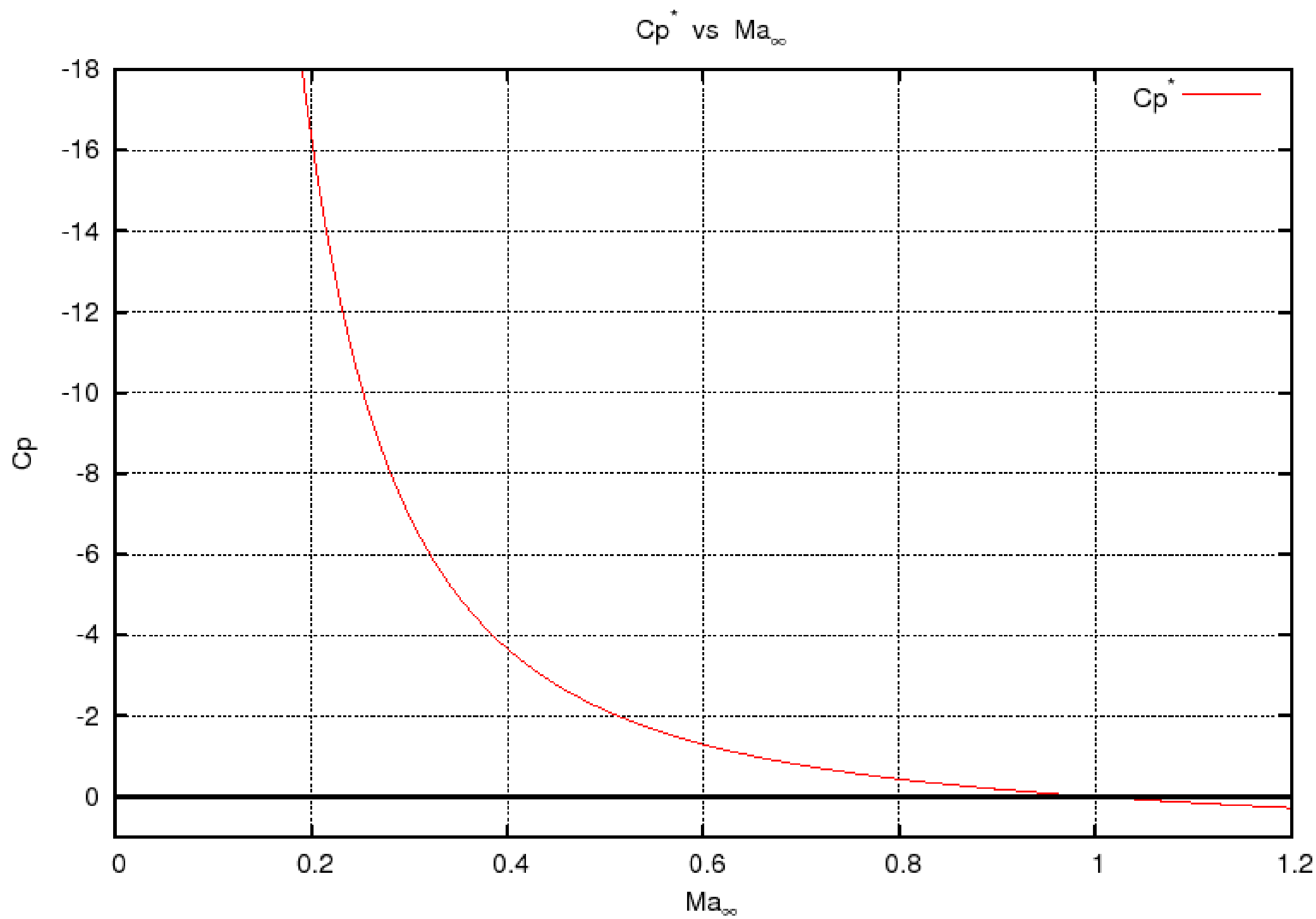
$$Cp^*\left(Ma_\infty\right) = \frac{\left(\dfrac{1+\frac{k-1}{2}Ma_\infty^2}{1+\frac{k-1}{2}}\right)^{\frac{k}{k-1}} - 1}{\dfrac{k}{2}Ma_\infty^2} = \frac{\left(\dfrac{1+0.2Ma_\infty^2}{1.2}\right)^{3.5} - 1}{0.7Ma_\infty^2}$$



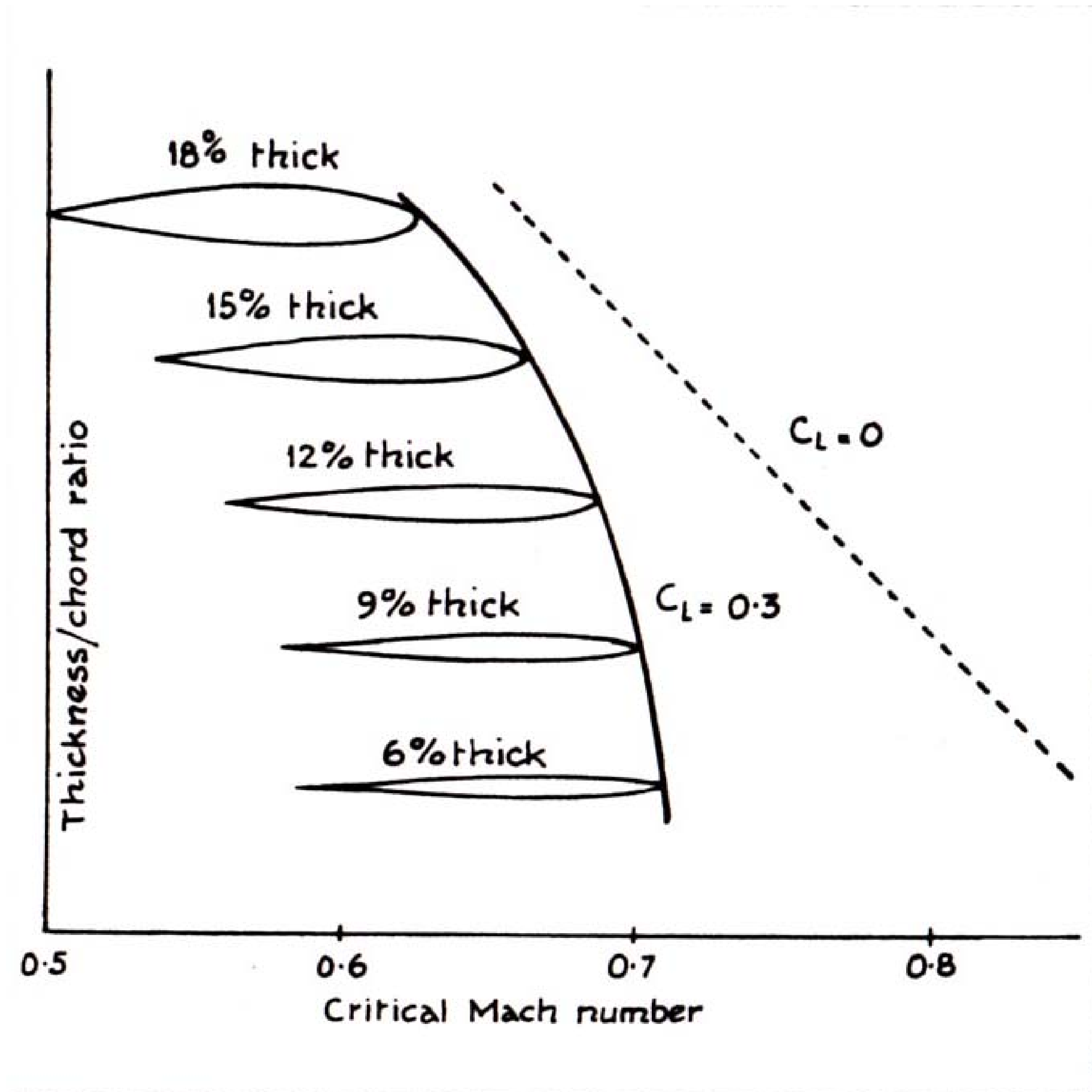


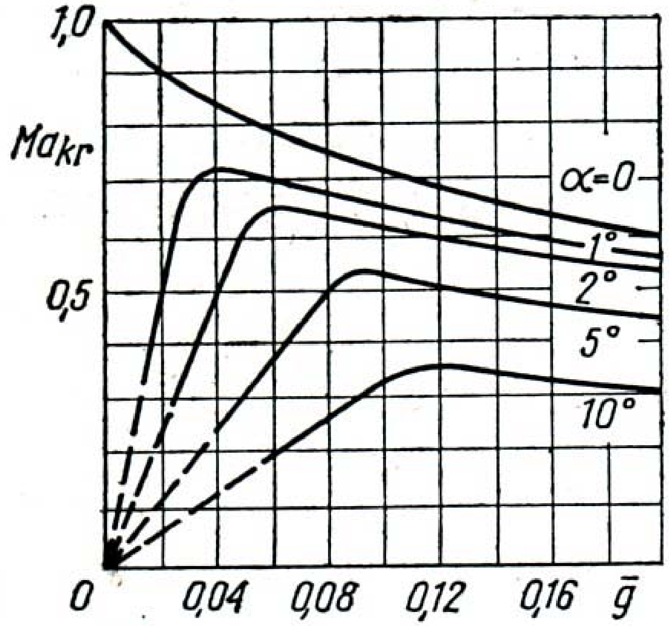
Cp* vs Ma∞
Cp*
-1.4
-1.2
-1
-0.8
-0.6
-0.4
-0.2
0
0.2
Cp
0
0.2
0.4
0.6
0.8
1
Ma∞

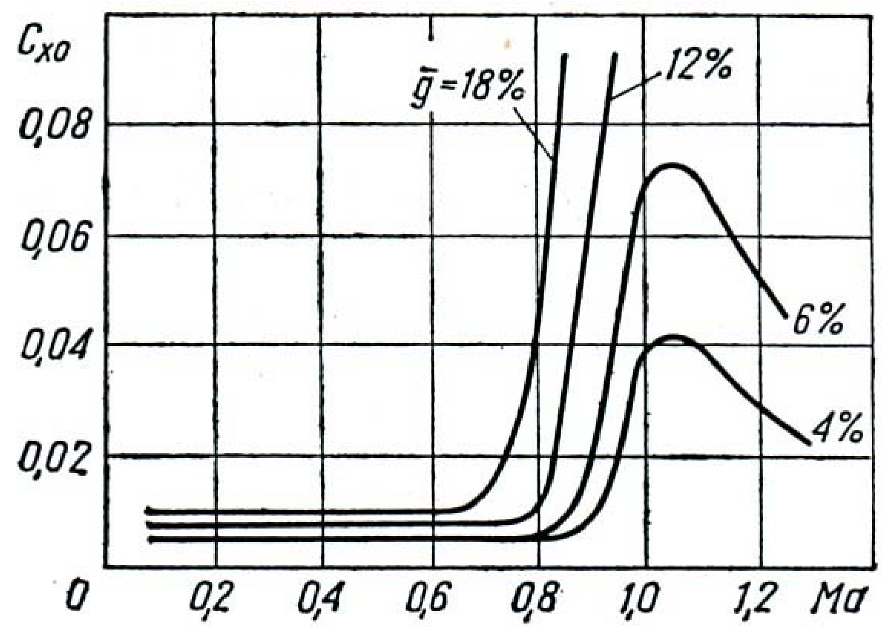

CRITICAL MACH NUMBER vs Cp_{min}
$Cp_{min} / (1-Ma_\infty^2)^{1/2}$
Cp
Ma∞
-1.4
-1.2
-1
-0.8
-0.6
-0.4
0.4
0.6
0.8
1
NACA 0010
Mach = 0.750
Alfa = -0.000
CL = 0.0007
CD = 0.00000
CM = -0.0000
L/D = 0.00
MSES v3.0
C_p
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0



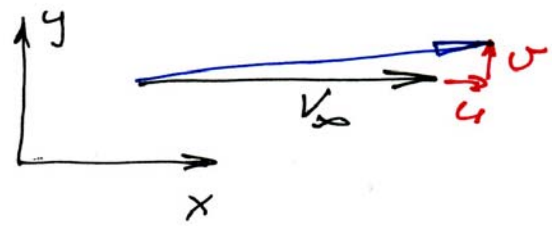
Cp* vs Ma∞
Cp*
Cp
-18
-16
-14
-12
-10
-8
-6
-4
-2
0
0
0.2
0.4
0.6
0.8
1
1.2
Ma∞


18% thick
15% thick
12% thick
9% thick
6% thick
$C_L = 0$
$C_L = 0.3$
Thickness/chord ratio
0·5
0·6
0·7
0·8
Critical Mach number


1,0
Ma_kr
0,5
α=0
1°
2°
5°
10°
0
0,04
0,08
0,12
0,16
g̅


C_xo
0,08
0,06
0,04
0,02
g̅ =18%
12%
6%
4%
0
0,2
0,4
0,6
0,8
1,0
1,2
Ma

SUPERCRITICAL FLOW OVER AN AIRFOIL



$$v \ll U_\infty$$

$$U = U_\infty + u = U_\infty + \frac{\partial \varphi}{\partial x}$$

$$V = \qquad v = \qquad \frac{\partial \varphi}{\partial y}$$

Continuity equation:

$$\frac{\partial(\rho \boldsymbol{U})}{\partial \boldsymbol{x}}+\frac{\partial(\rho \boldsymbol{V})}{\partial \boldsymbol{y}}=\mathbf{0}=(\boldsymbol{U}_{\infty}+\boldsymbol{u})\frac{\partial \rho}{\partial \boldsymbol{x}}+\rho\frac{\partial \boldsymbol{u}}{\partial \boldsymbol{x}}+\cancel{\boldsymbol{v}}\frac{\partial \rho}{\partial \boldsymbol{y}}+\rho\frac{\partial \boldsymbol{v}}{\partial \boldsymbol{y}}\simeq$$

$$\boldsymbol{U}\frac{\partial \rho}{\partial \boldsymbol{x}}+\rho\frac{\partial \boldsymbol{u}}{\partial \boldsymbol{x}}+\rho\frac{\partial \boldsymbol{v}}{\partial \boldsymbol{y}}=\boldsymbol{U}\frac{\partial \rho}{\partial \boldsymbol{x}}+\rho\, \boldsymbol{div}\ \vec{\boldsymbol{v}}=$$

$$\left|\boldsymbol{S}=\boldsymbol{const}\right|=\boldsymbol{U}\left.\frac{\partial \rho}{\partial \boldsymbol{p}}\right|_{L}\frac{\partial \boldsymbol{p}}{\partial \boldsymbol{x}}+\rho\, \boldsymbol{div}\ \vec{\boldsymbol{v}}=$$

$$\left|\begin{array}{l}\boldsymbol{r.\ Eulera}(\boldsymbol{x}):\quad \rho\boldsymbol{U}\frac{\partial \boldsymbol{U}}{\partial \boldsymbol{x}}+\rho\boldsymbol{V}\frac{\partial \boldsymbol{U}}{\partial \boldsymbol{y}}=-\frac{\partial \boldsymbol{p}}{\partial \boldsymbol{x}}\\ \\ \qquad\qquad \rho(\boldsymbol{U}_{\infty}+\boldsymbol{u})\frac{\partial \boldsymbol{u}}{\partial \boldsymbol{x}}+\rho\cancel{\boldsymbol{v}}\frac{\partial \boldsymbol{u}}{\partial \boldsymbol{y}}=-\frac{\partial \boldsymbol{p}}{\partial \boldsymbol{x}}\quad\rightarrow\quad\frac{\partial \boldsymbol{p}}{\partial \boldsymbol{x}}\simeq-\rho\boldsymbol{U}\frac{\partial \boldsymbol{u}}{\partial \boldsymbol{x}}\end{array}\right|$$

$$=-\rho\frac{\boldsymbol{U}^{2}}{\boldsymbol{a}^{2}}\frac{\partial \boldsymbol{u}}{\partial \boldsymbol{x}}+\rho\, \boldsymbol{div}\ \vec{\boldsymbol{v}}=\rho\left(\left(\mathbf{1}-\boldsymbol{Ma}_{L}^{2}\right)\frac{\partial \boldsymbol{u}}{\partial \boldsymbol{x}}+\frac{\partial \boldsymbol{v}}{\partial \boldsymbol{y}}\right)=\mathbf{0}$$

$$\left(1 - Ma_L^2\right)\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = 0$$

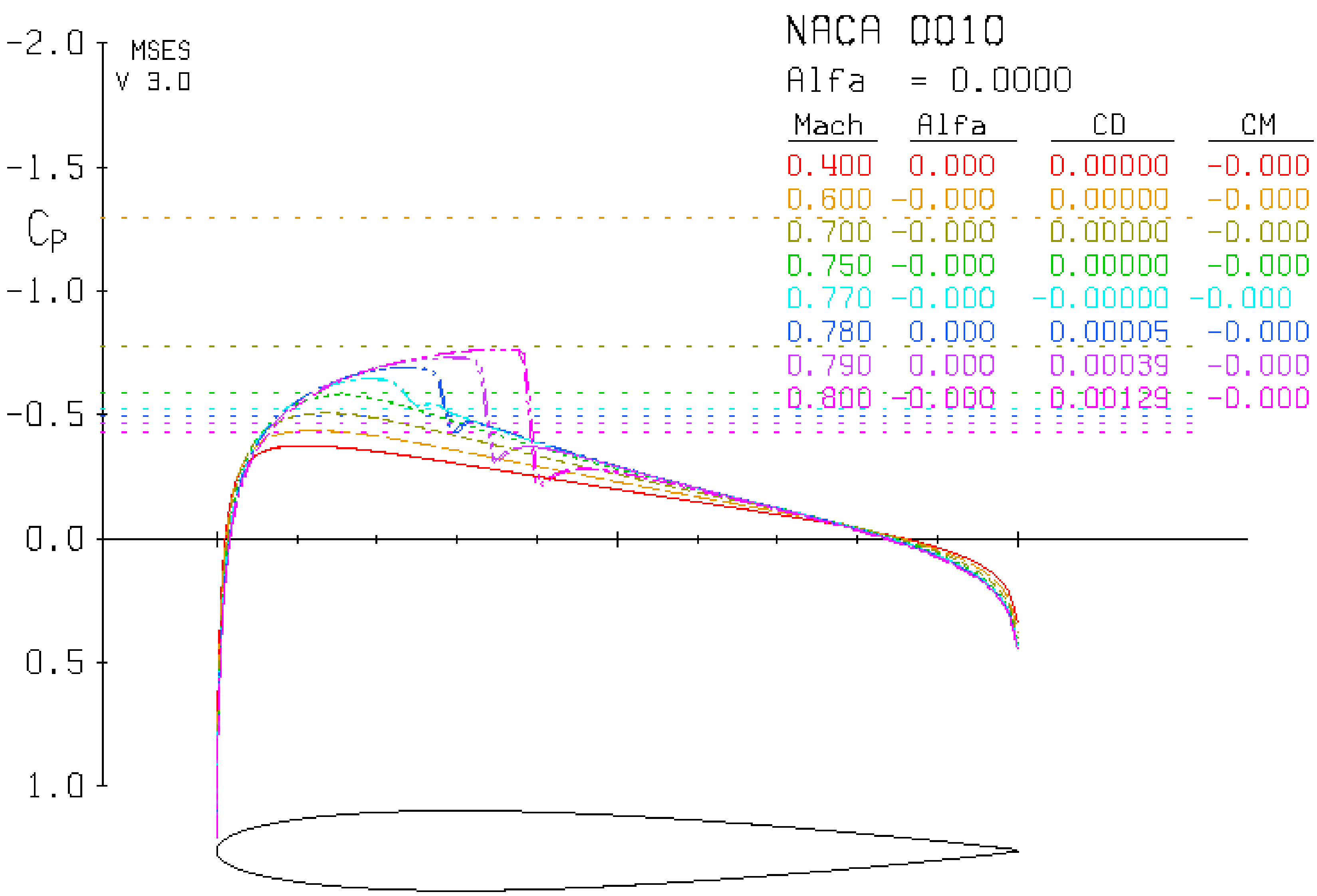
$$\boxed{\left(1 - Ma_L^2\right)\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = \left(1 - Ma_L^2\right)\frac{\partial^2 \varphi}{\partial x^2} + \frac{\partial^2 \varphi}{\partial y^2} = 0}$$

$$M_L < 1$$

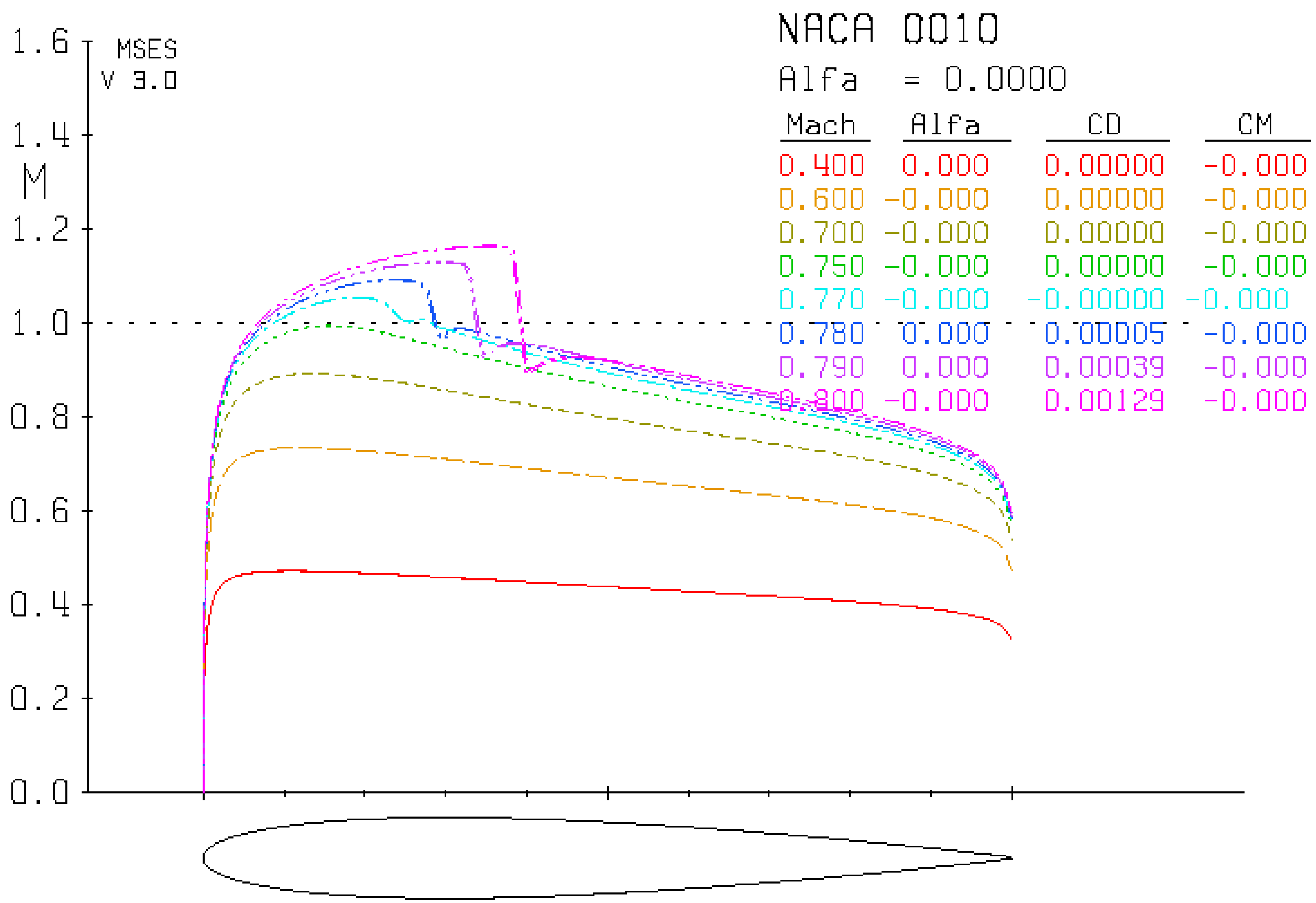
$$M_L = 1$$

$$M_L > 1$$

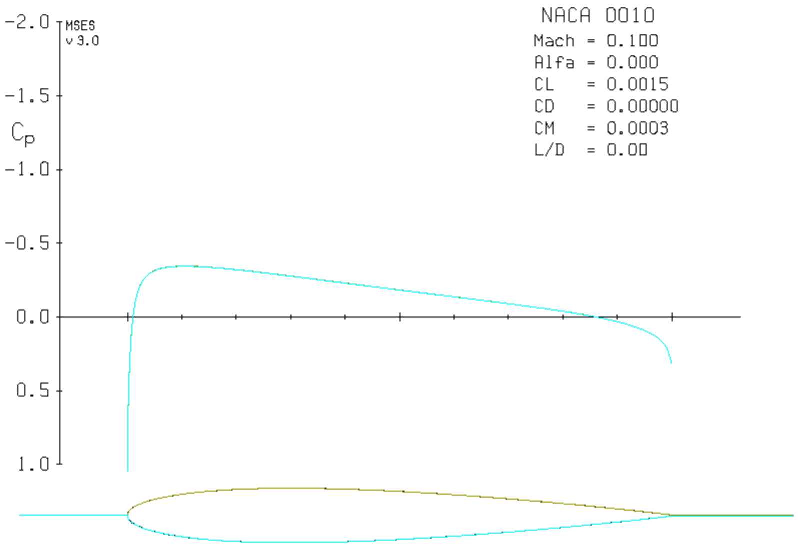
$$div\vec{V} = q = Ma_L^2 \frac{\partial u}{\partial x}$$



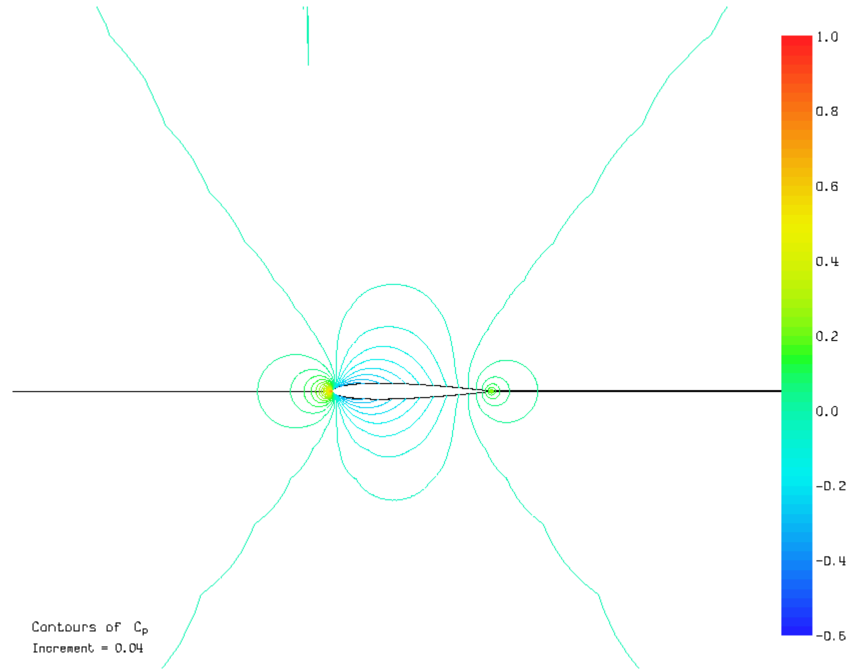
NACA 0010
Alfa = 0.0000
MSES
V 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Mach Alfa CD CM
0.400 0.000 0.00000 -0.000
0.600 -0.000 0.00000 -0.000
0.700 -0.000 0.00000 -0.000
0.750 -0.000 0.00000 -0.000
0.770 -0.000 -0.00000 -0.000
0.780 0.000 0.00005 -0.000
0.790 0.000 0.00039 -0.000
0.800 -0.000 0.00129 -0.000



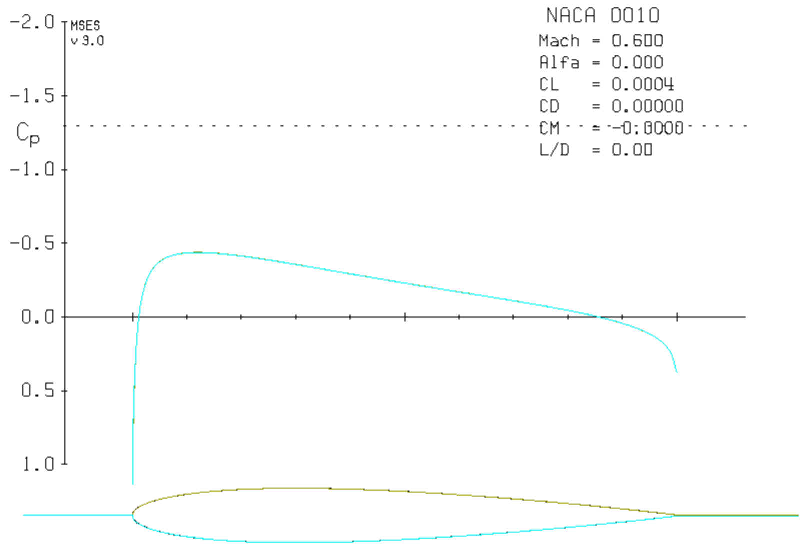
NACA 0010
Alfa = 0.0000
MSES
V 3.0
M
1.6
1.4
1.2
1.0
0.8
0.6
0.4
0.2
0.0
Mach Alfa CD CM
0.400 0.000 0.00000 -0.000
0.600 -0.000 0.00000 -0.000
0.700 -0.000 0.00000 -0.000
0.750 -0.000 0.00000 -0.000
0.770 -0.000 -0.00000 -0.000
0.780 0.000 0.00005 -0.000
0.790 0.000 0.00039 -0.000
0.800 -0.000 0.00129 -0.000

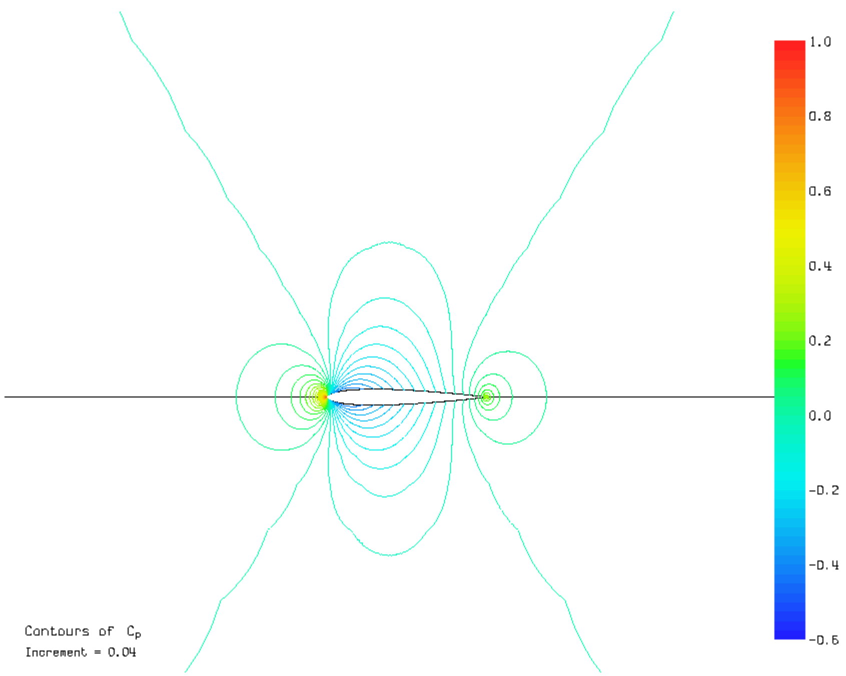


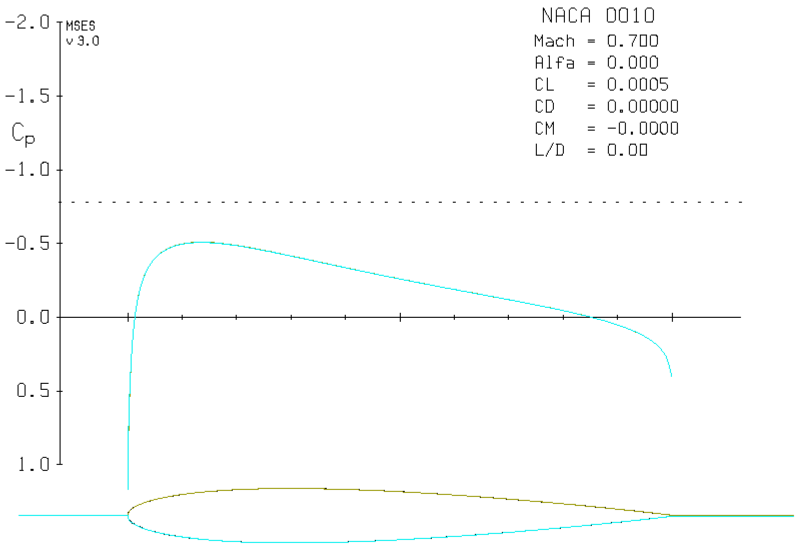
NACA 0010
Mach = 0.100
Alfa = 0.000
CL = 0.0015
CD = 0.00000
CM = 0.0003
L/D = 0.00
MSES
v 3.0
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

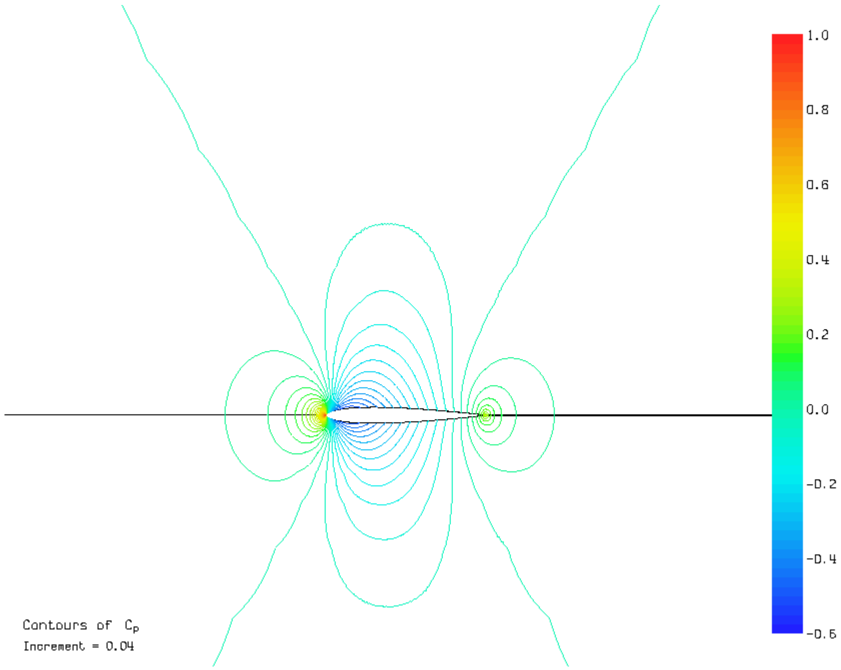


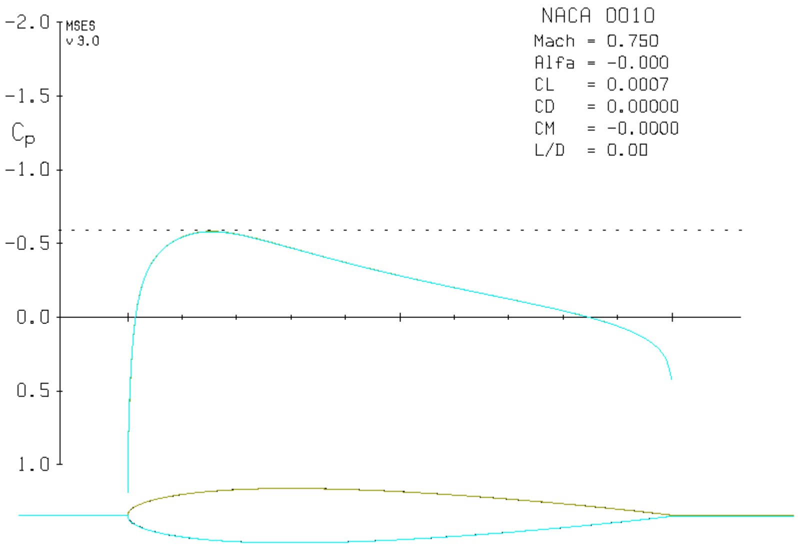
1.0
0.8
0.6
0.4
0.2
0.0
-0.2
-0.4
-0.6
Contours of Cp
Increment = 0.04


NACA 0010
Mach = 0.600
Alfa = 0.000
CL = 0.0004
CD = 0.00000
CM = -0.0000
L/D = 0.00
MSES
v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0

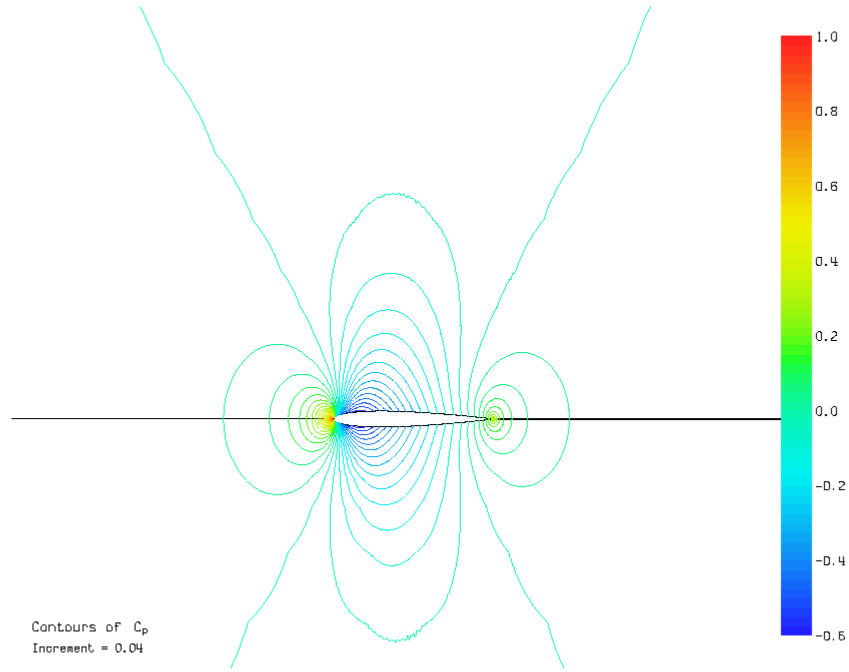

Contours of Cp
Increment = 0.04
1.0
0.8
0.6
0.4
0.2
0.0
-0.2
-0.4
-0.6


NACA 0010
Mach = 0.700
Alfa = 0.000
CL = 0.0005
CD = 0.00000
CM = -0.0000
L/D = 0.00
MSES
v 3.0
C_p
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0

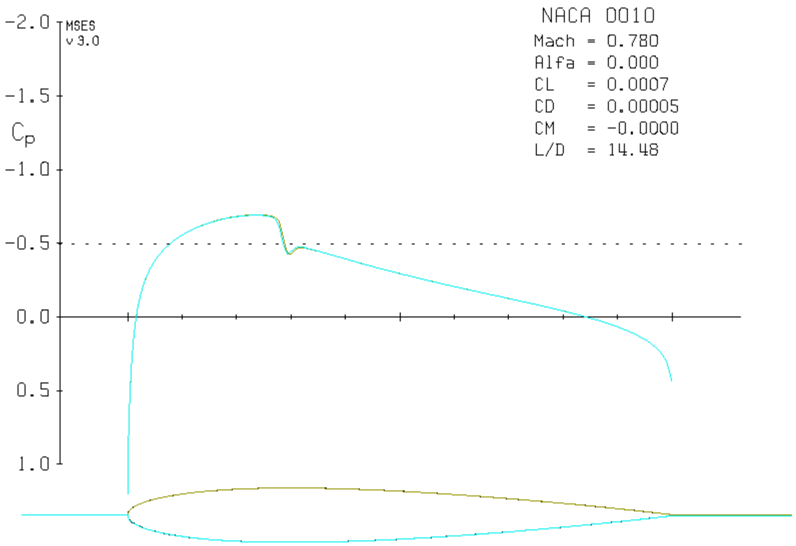

Contours of C_p
Increment = 0.04
1.0
0.8
0.6
0.4
0.2
0.0
-0.2
-0.4
-0.6

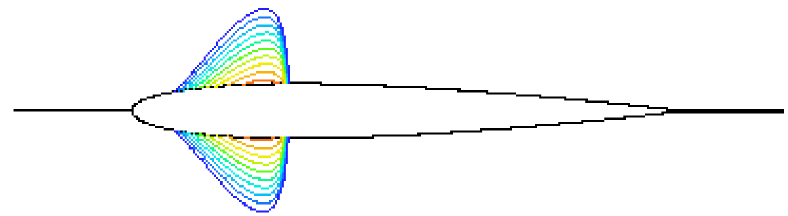


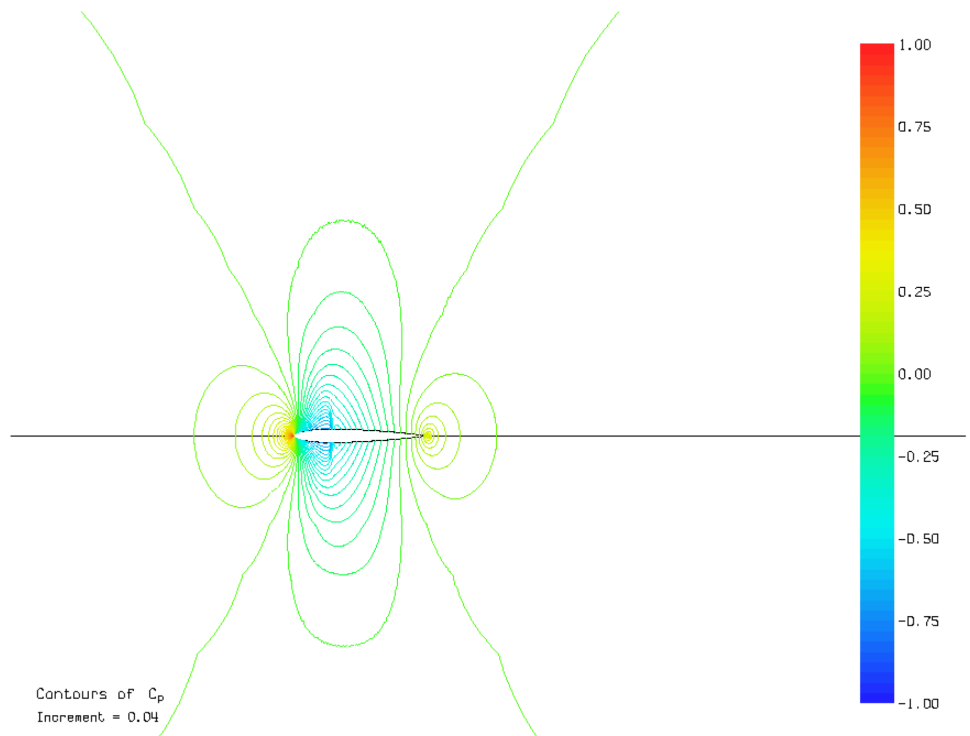
-2.0
MSES
v 3.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
NACA 0010
Mach = 0.750
Alfa = -0.000
CL = 0.0007
CD = 0.00000
CM = -0.0000
L/D = 0.00

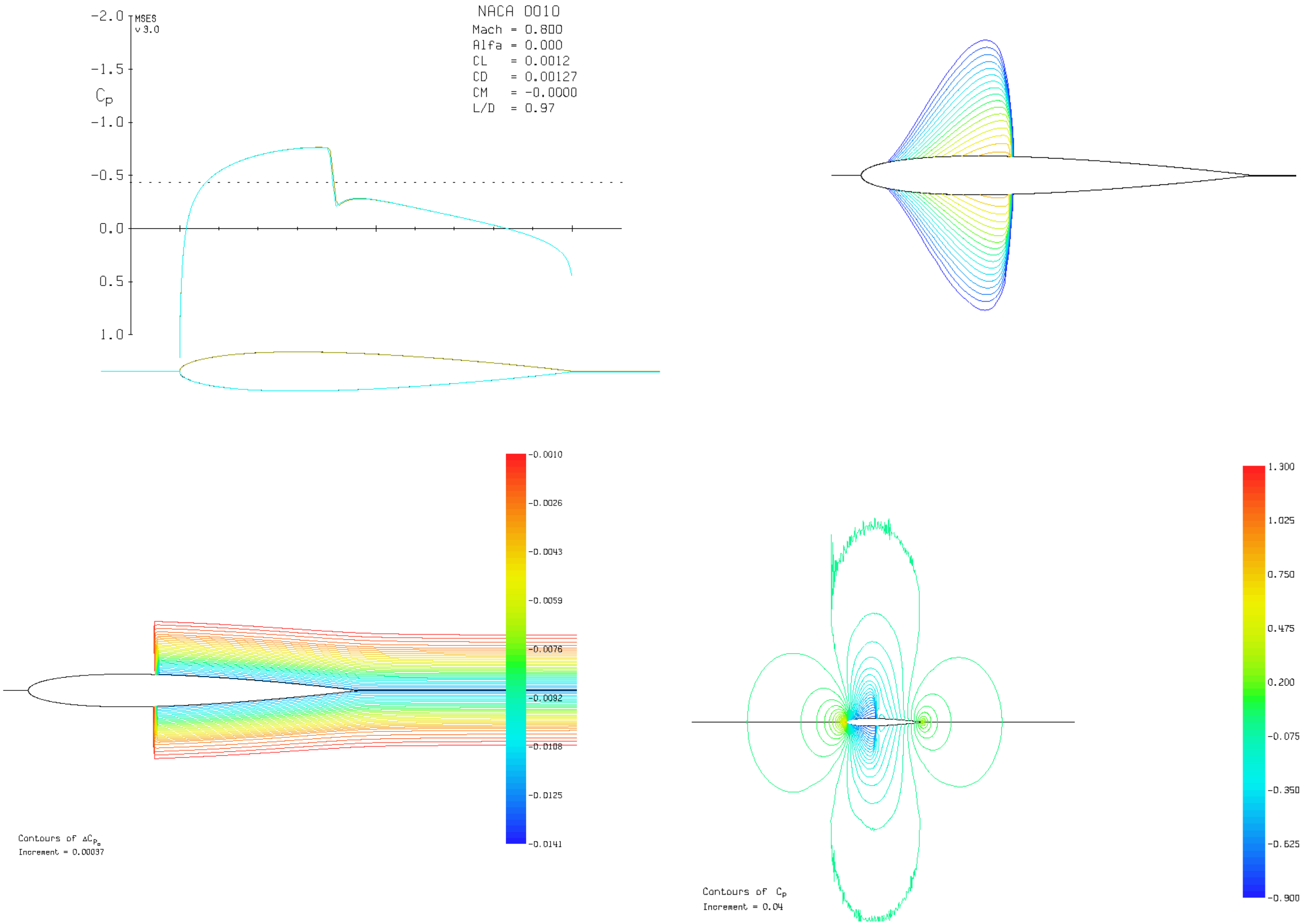


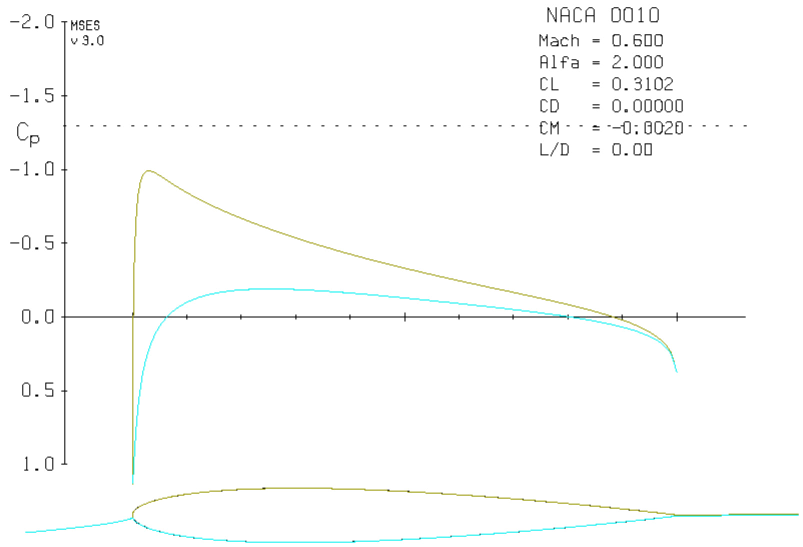
1.0
0.8
0.6
0.4
0.2
0.0
-0.2
-0.4
-0.6
Contours of Cp
Increment = 0.04


NACA 0010
Mach = 0.780
Alfa = 0.000
CL = 0.0007
CD = 0.00005
CM = -0.0000
L/D = 14.48
MSES
v 3.0
-2.0
-1.5
C_p
-1.0
-0.5
0.0
0.5
1.0

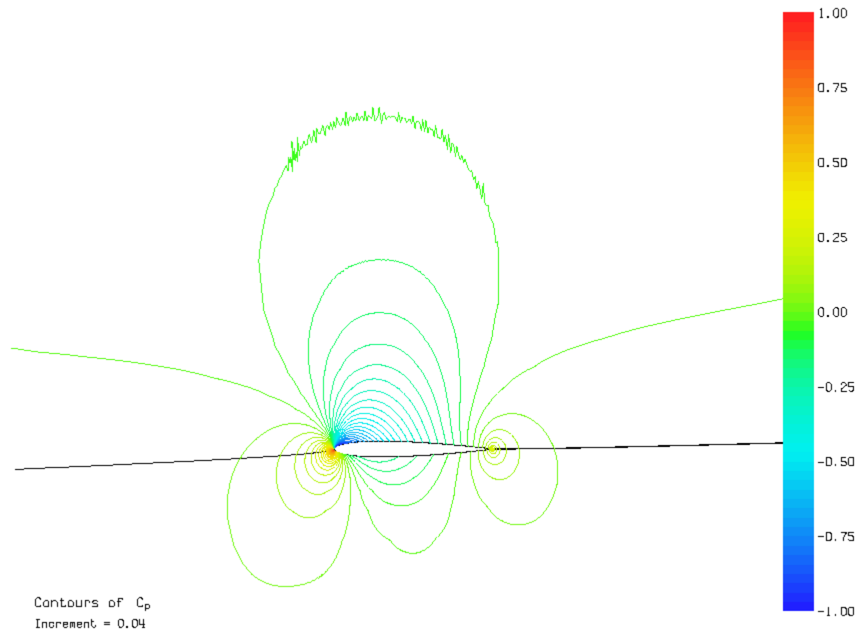



Contours of C_p
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00

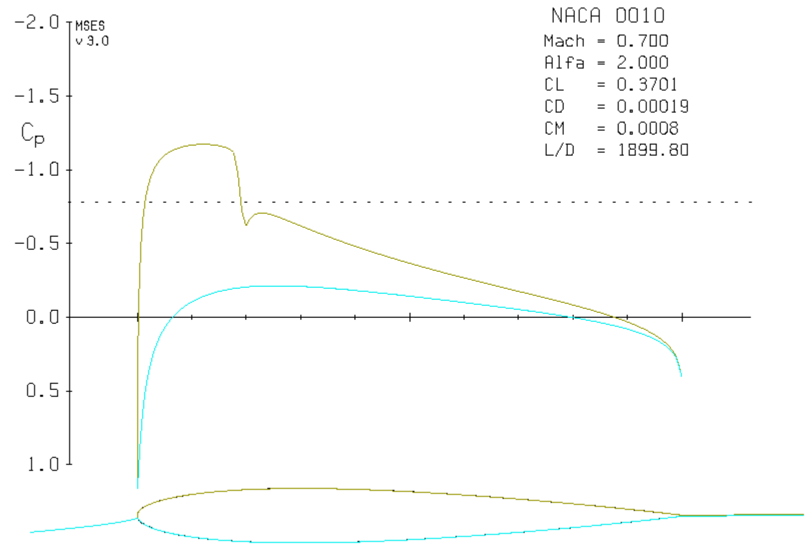

NACA 0010
Mach = 0.800
Alfa = 0.000
CL = 0.0012
CD = 0.00127
CM = -0.0000
L/D = 0.97
MSES
v 3.0
C_p
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of ΔC_{p_o}
Increment = 0.00037
-0.0010
-0.0026
-0.0043
-0.0059
-0.0076
-0.0092
-0.0108
-0.0125
-0.0141
Contours of C_p
Increment = 0.04
1.300
1.025
0.750
0.475
0.200
-0.075
-0.350
-0.625
-0.900

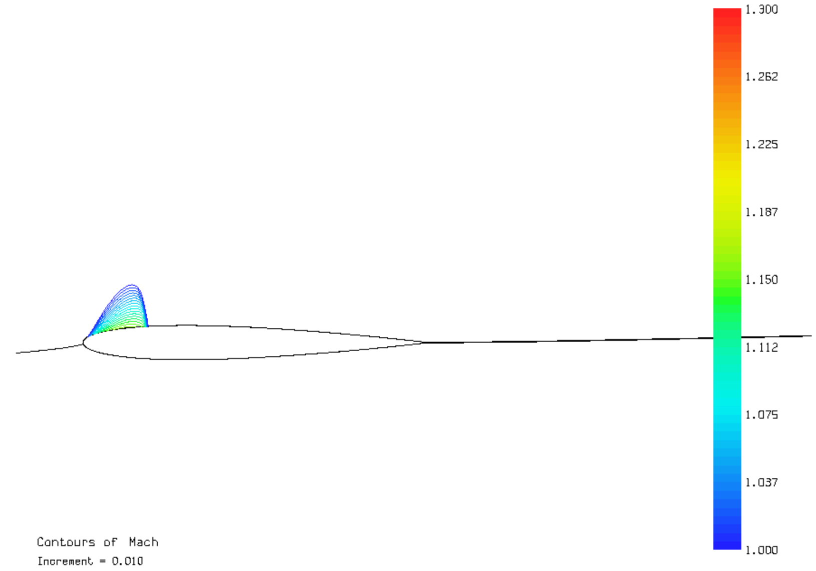


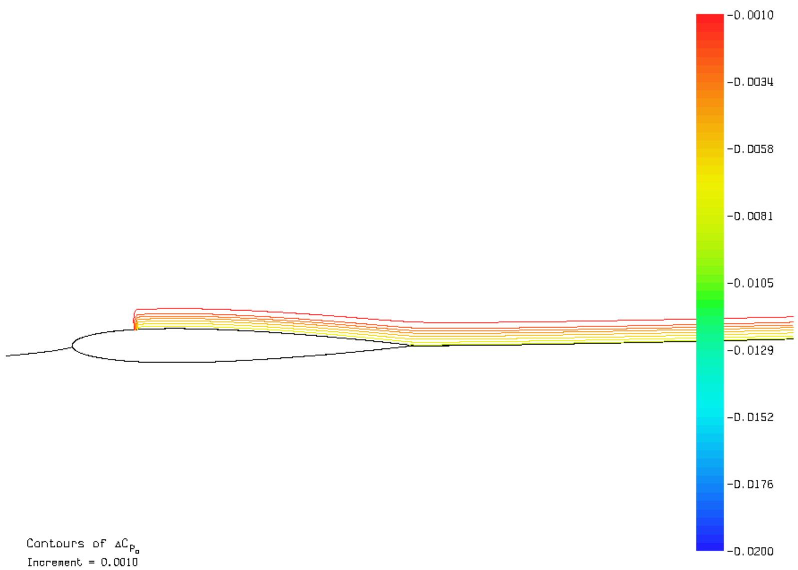
NACA 0010
Mach = 0.600
Alfa = 2.000
CL = 0.3102
CD = 0.00000
CM = -0.0020
L/D = 0.00
MSES
v 3.0
C_p
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0

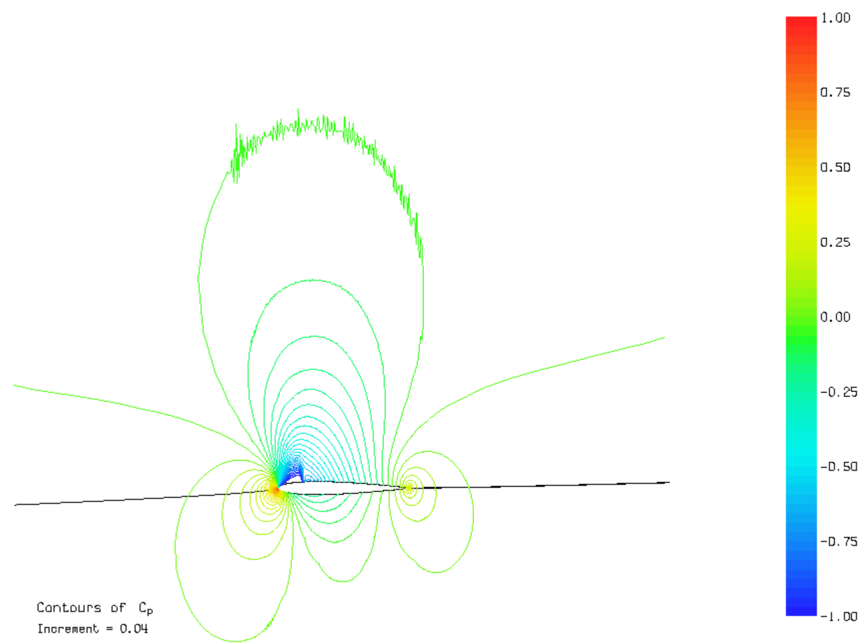


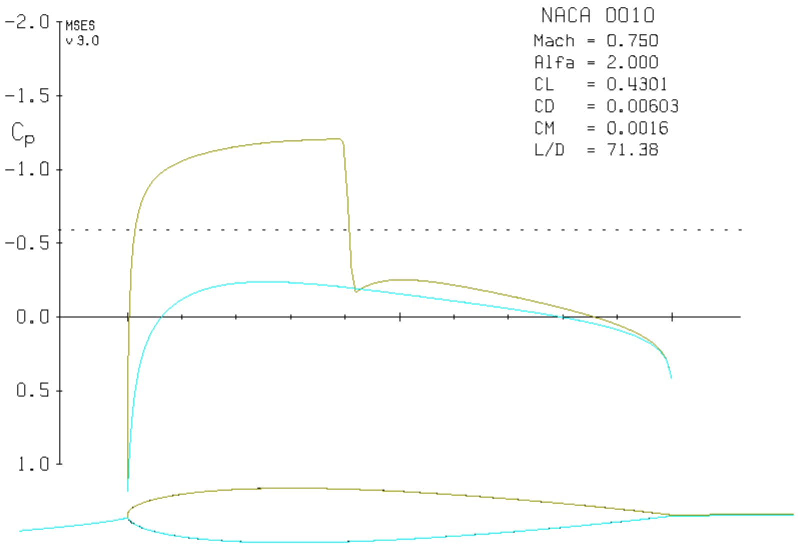
Contours of C_p
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00


NACA 0010
Mach = 0.700
Alfa = 2.000
CL = 0.3701
CD = 0.00019
CM = 0.0008
L/D = 1899.80
MSES
v 3.0
C_p
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0

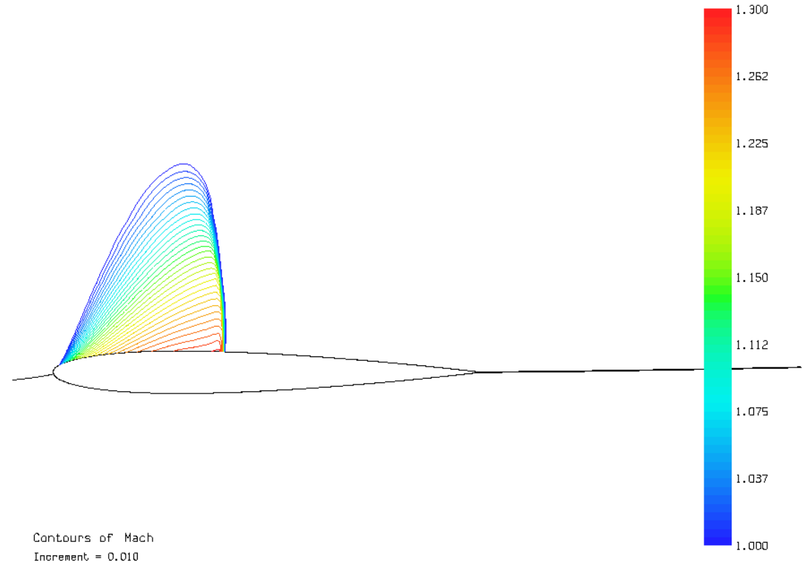

1.300
1.262
1.225
1.187
1.150
1.112
1.075
1.037
1.000
Contours of Mach
Increment = 0.010


-0.0010
-0.0034
-0.0058
-0.0081
-0.0105
-0.0129
-0.0152
-0.0176
-0.0200
Contours of ΔC_{p_o}
Increment = 0.0010

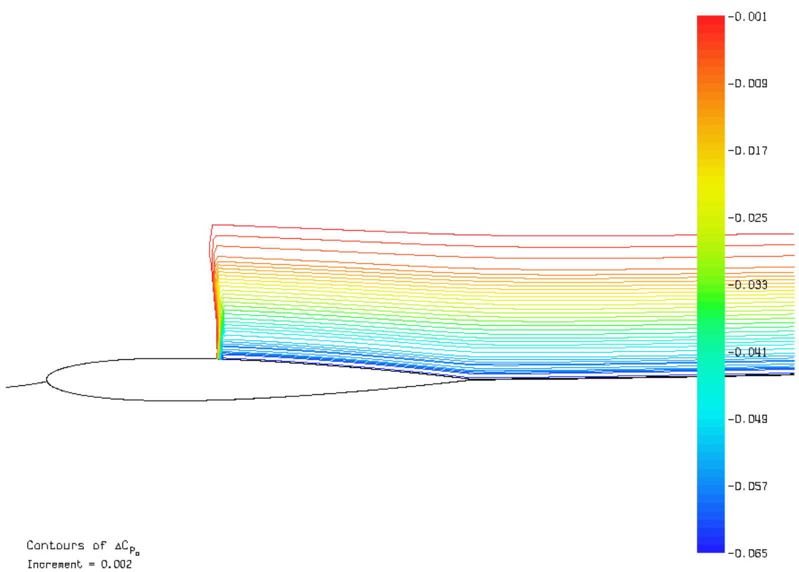

1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of C_p
Increment = 0.04



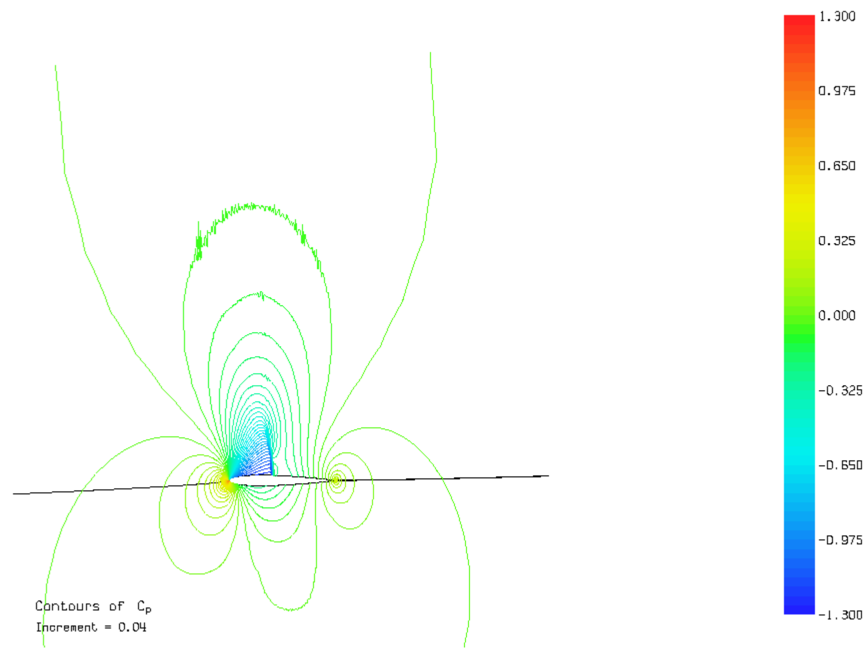
-2.0
MSES
v 3.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
NACA 0010
Mach = 0.750
Alfa = 2.000
CL = 0.4301
CD = 0.00603
CM = 0.0016
L/D = 71.38



1.300
1.262
1.225
1.187
1.150
1.112
1.075
1.037
1.000
Contours of Mach
Increment = 0.010

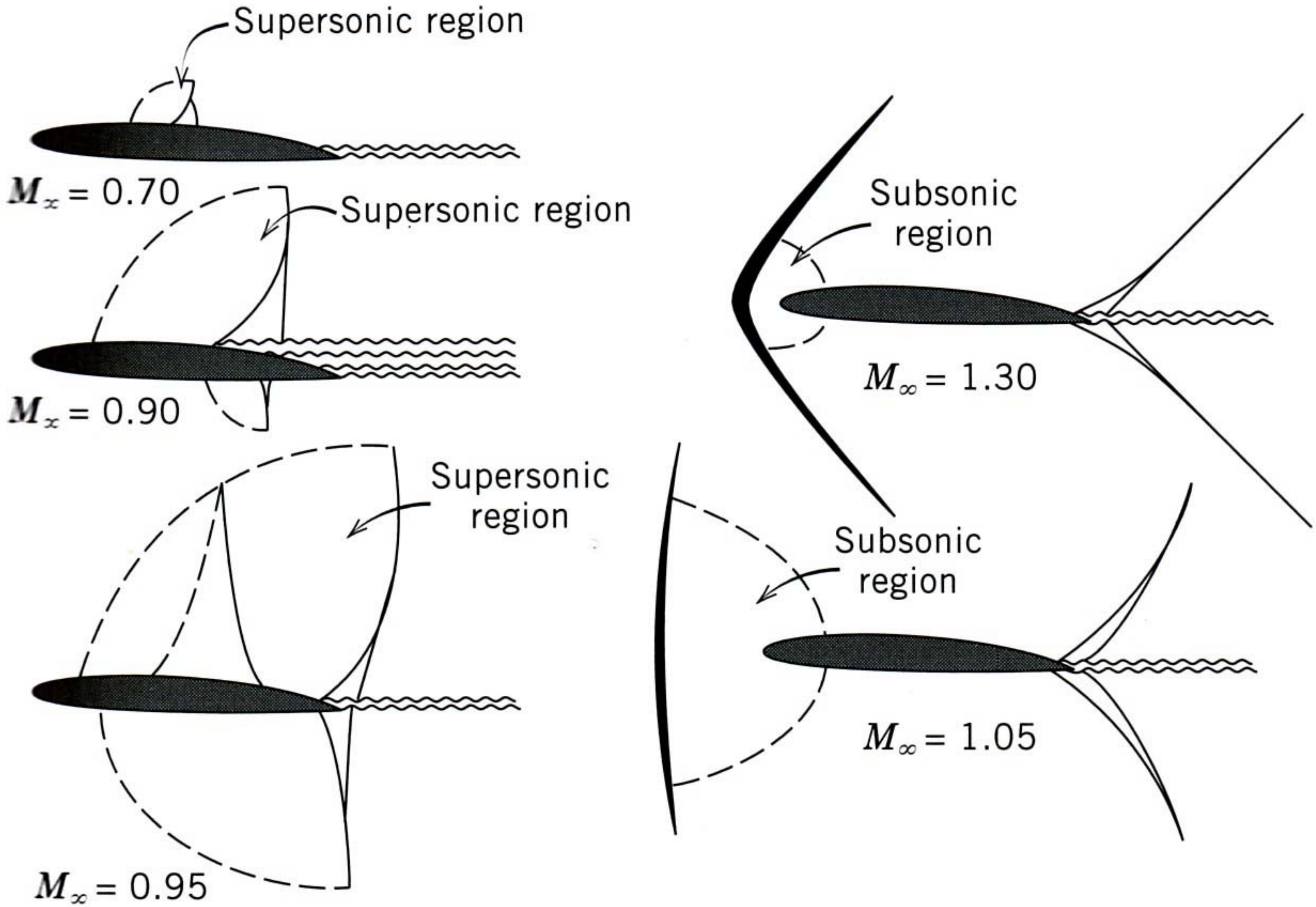


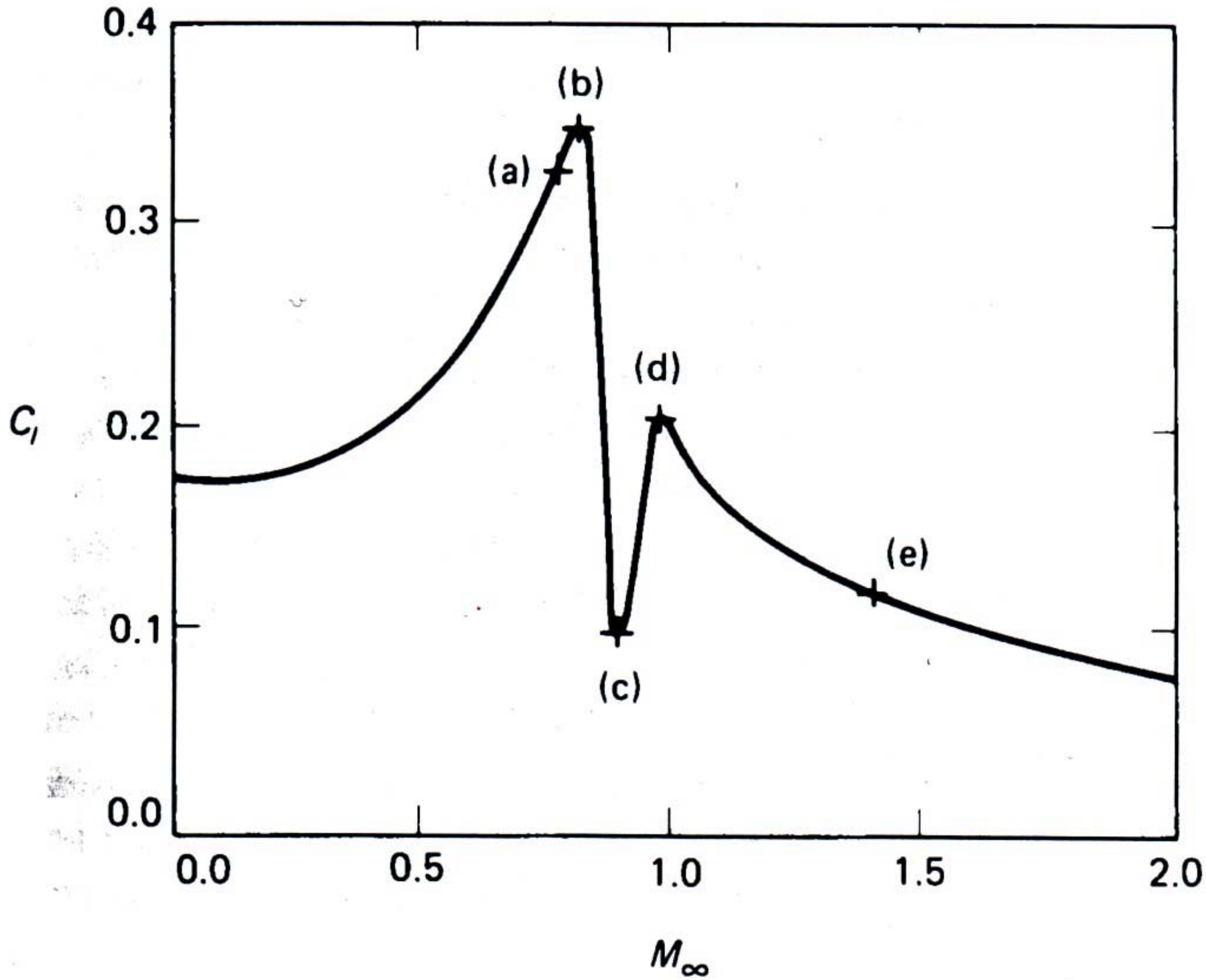
-0.001
-0.009
-0.017
-0.025
-0.033
-0.041
-0.049
-0.057
-0.065
Contours of ΔCp
Increment = 0.002



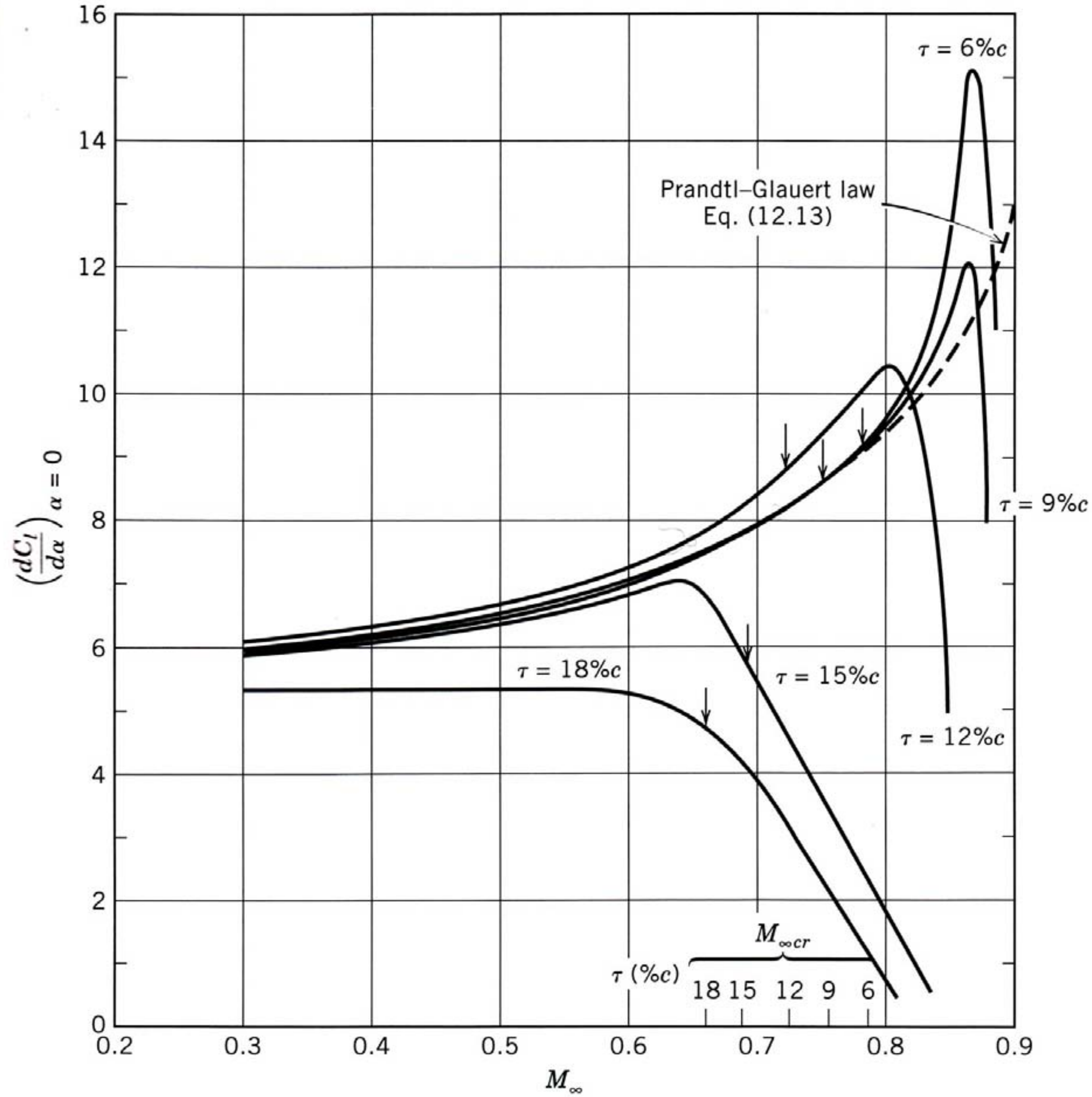
1.300
0.975
0.650
0.325
0.000
-0.325
-0.650
-0.975
-1.300
Contours of Cp
Increment = 0.04

REAL FLOW OVER AN AIRFOIL (VISCOUS FLUID)





0.4
0.3
0.2
0.1
0.0
C_l
(a)
(b)
(c)
(d)
(e)
0.0
0.5
1.0
1.5
2.0
M_∞



16
14
12
10
8
6
4
2
0
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
$\left(\frac{dC_l}{d\alpha}\right)_{\alpha=0}$
M_∞
τ = 6%c
Prandtl–Glauert law
Eq. (12.13)
τ = 9%c
τ = 12%c
τ = 15%c
τ = 18%c
$M_{\infty cr}$
τ (%c)
18 15 12 9 6

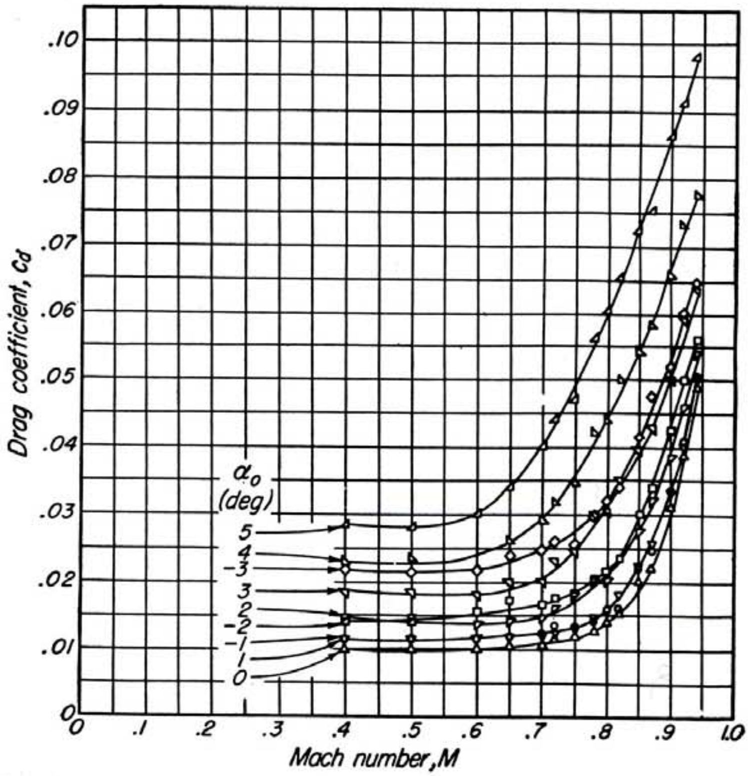


FIG. 176. Effect of compressibility on the drag of the NACA 2306 airfoil.

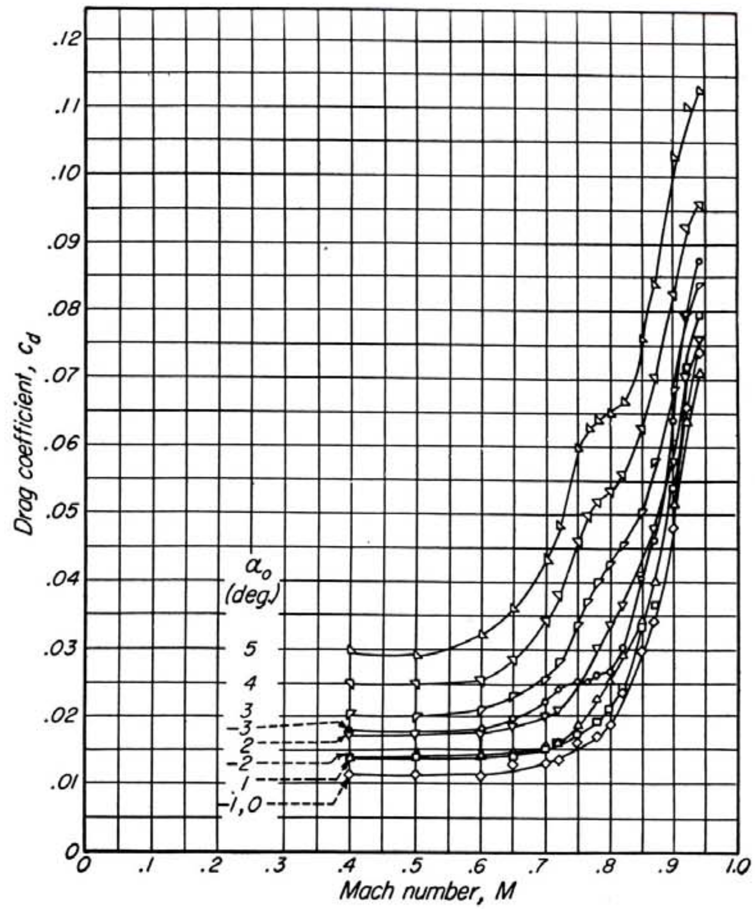
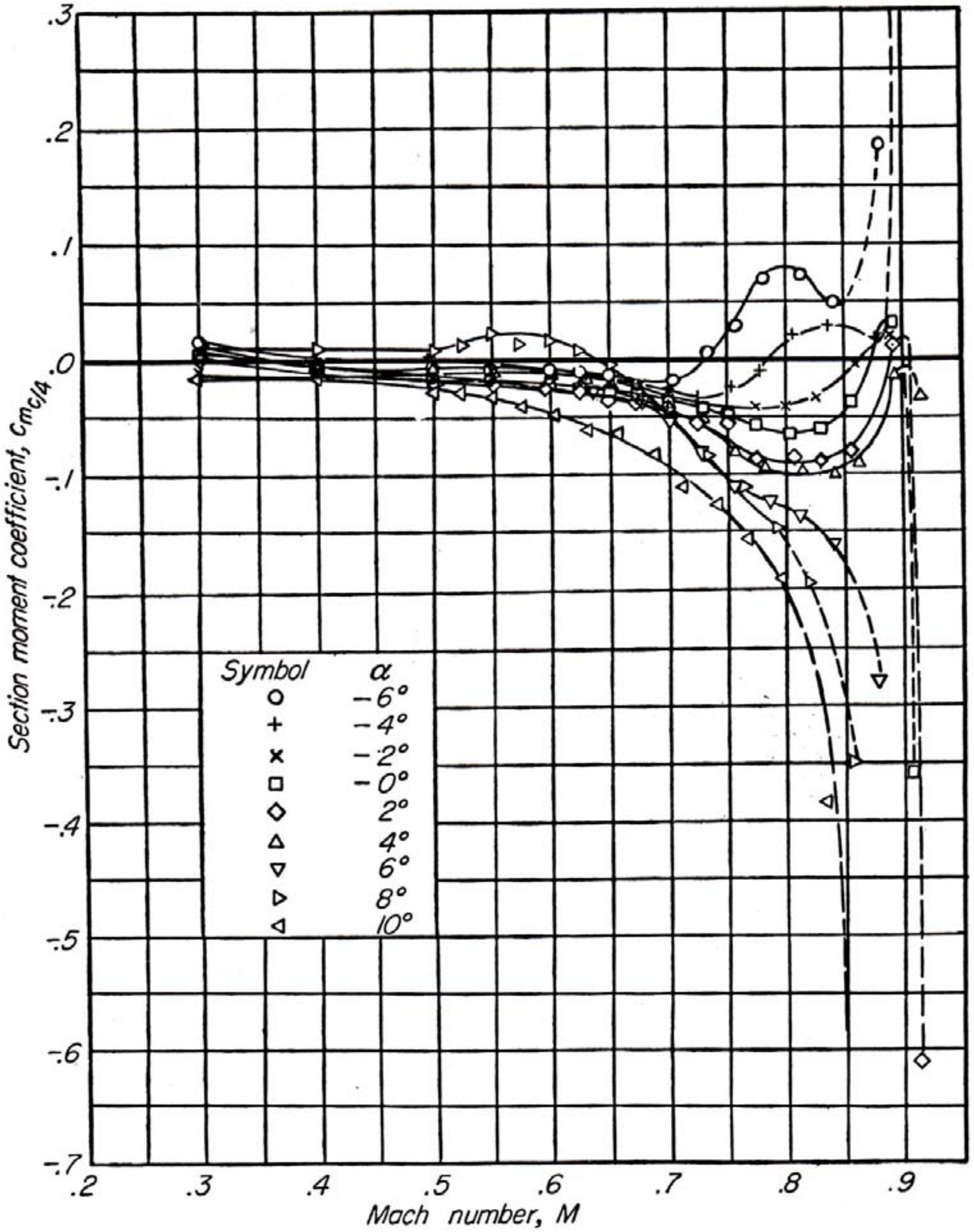
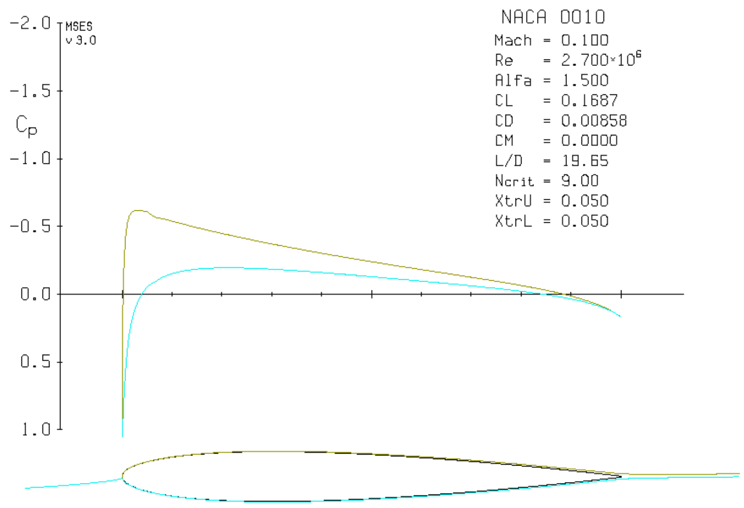


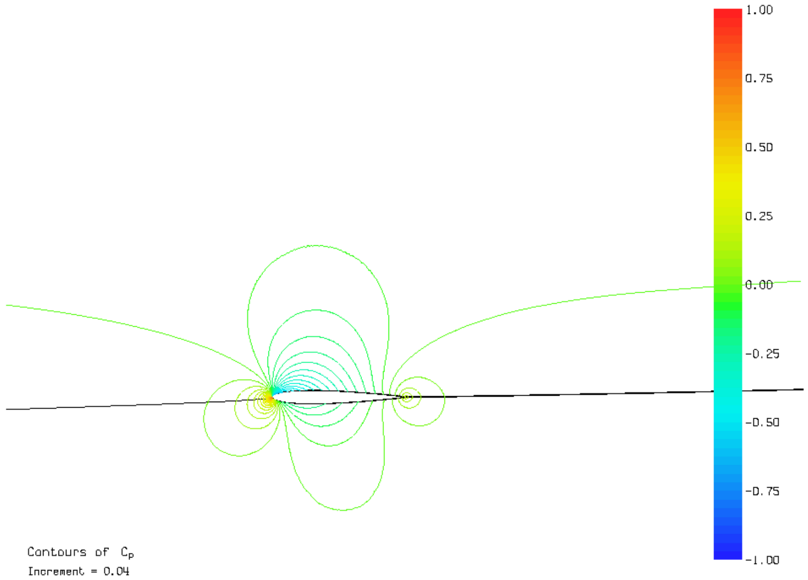
FIG. 177. Effect of compressibility on the drag of the NACA 2309 airfoil.



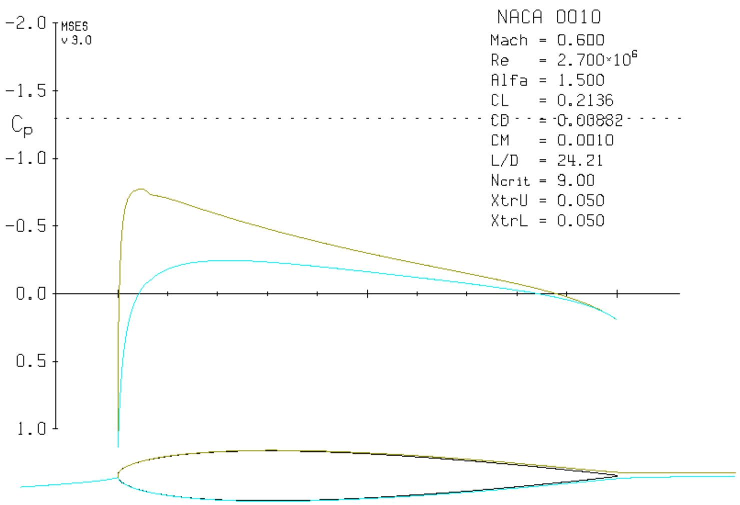
.3
.2
.1
.0
-.1
-.2
-.3
-.4
-.5
-.6
-.7
.2
.3
.4
.5
.6
.7
.8
.9
Section moment coefficient, $c_{m_{c/4}}$
Mach number, M
Symbol
α
−6°
−4°
−2°
−0°
2°
4°
6°
8°
10°



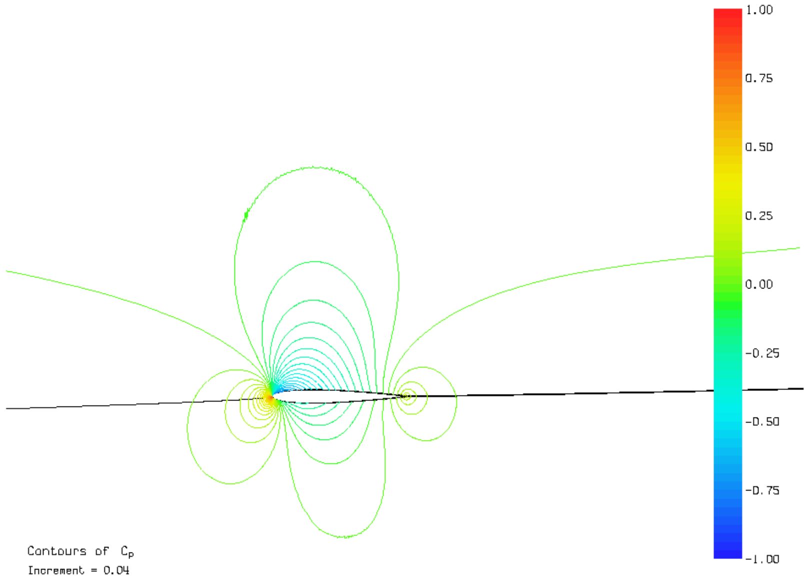
NACA 0010
Mach = 0.100
Re = 2.700×10^6
Alfa = 1.500
CL = 0.1687
CD = 0.00858
CM = 0.0000
L/D = 19.65
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES
v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0



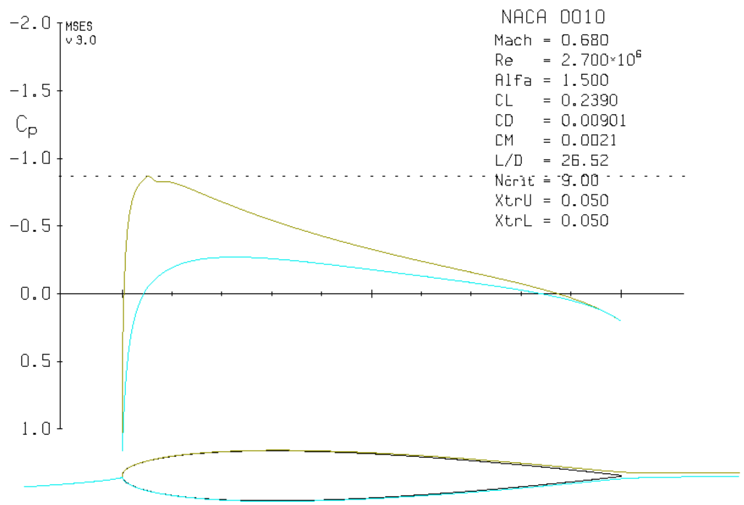
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Cp
Increment = 0.04



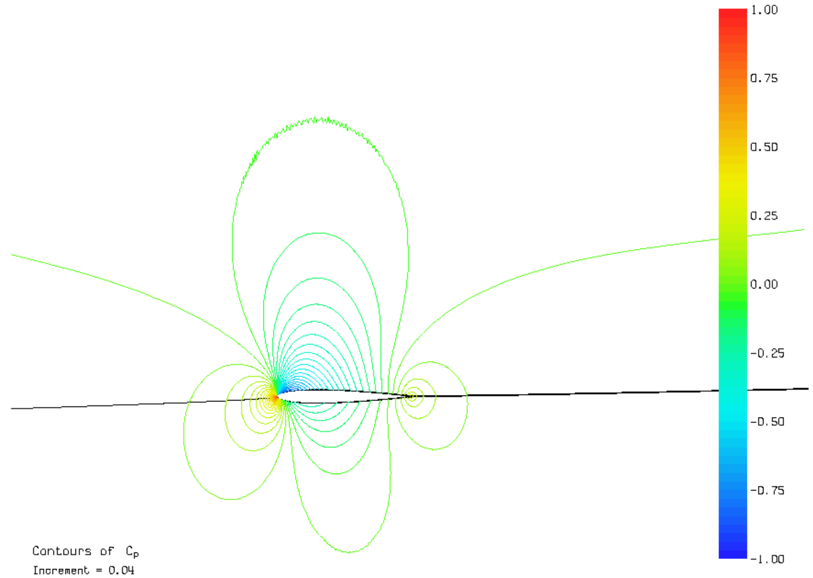
MSES
v 3.0
NACA 0010
Mach = 0.600
Re = 2.700×10^6
Alfa = 1.500
CL = 0.2136
CD = 0.00882
CM = 0.0010
L/D = 24.21
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0



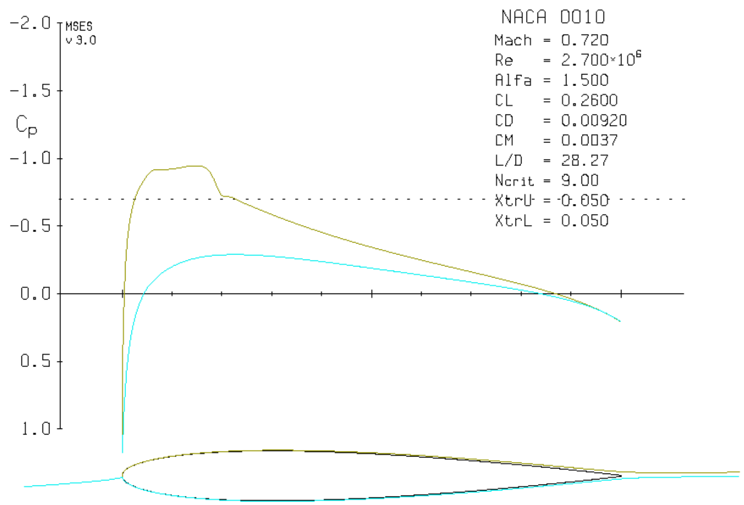
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Cp
Increment = 0.04

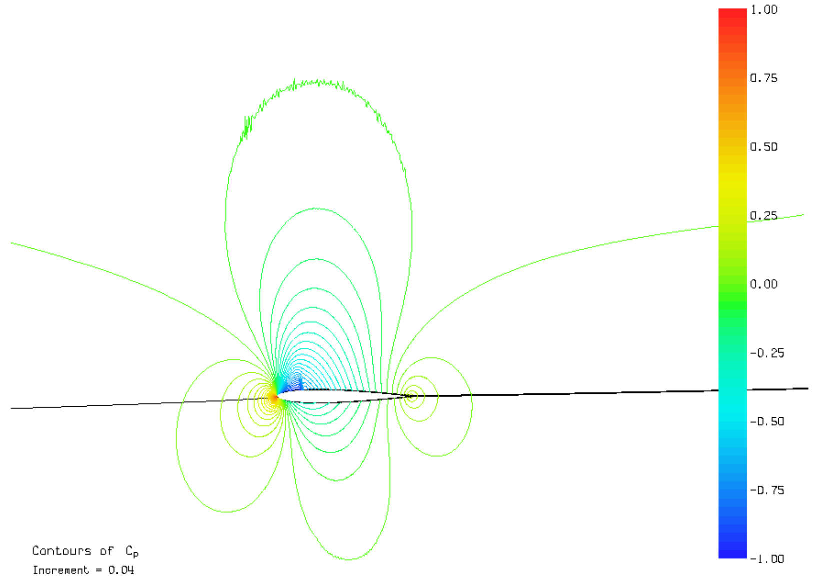


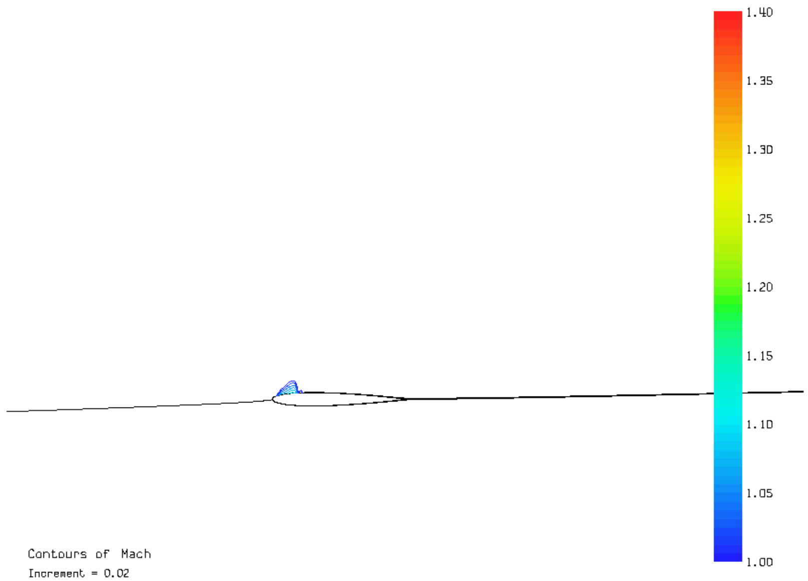
MSES v 3.0
NACA 0010
Mach = 0.680
Re = 2.700×10^6
Alfa = 1.500
CL = 0.2390
CD = 0.00901
CM = 0.0021
L/D = 26.52
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

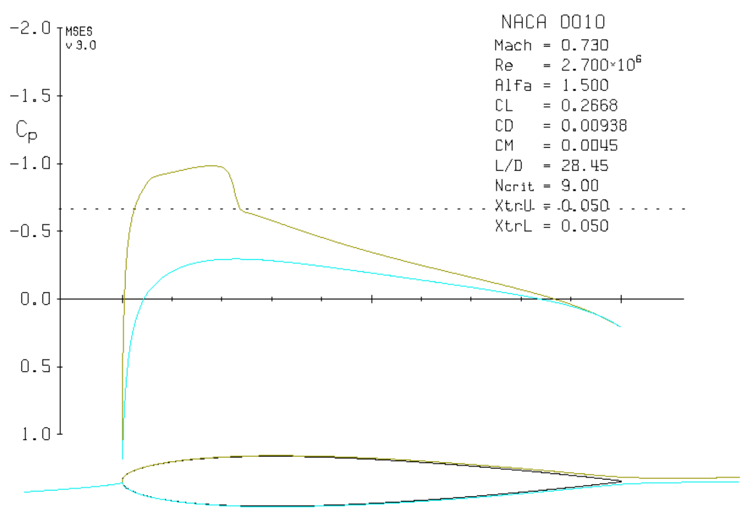


1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Cp
Increment = 0.04

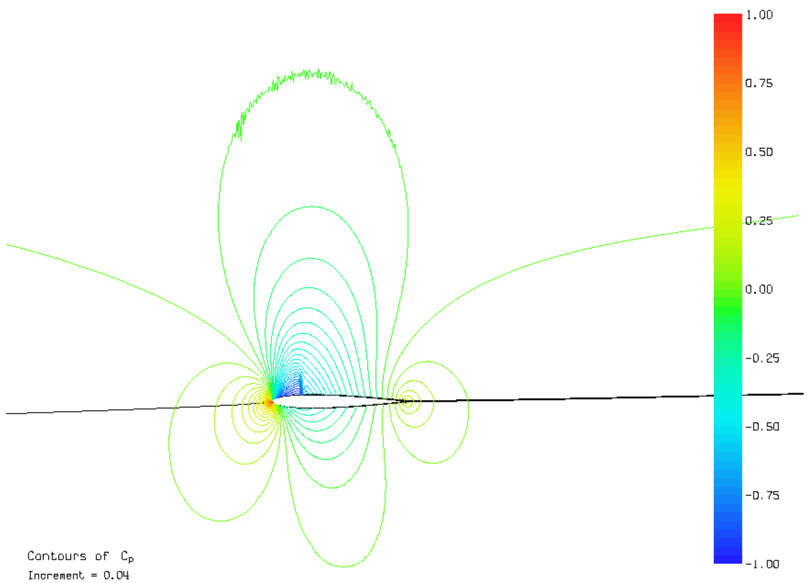

NACA 0010
Mach = 0.720
Re = 2.700×10^6
Alfa = 1.500
CL = 0.2600
CD = 0.00920
CM = 0.0037
L/D = 28.27
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES
v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0


1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Cp
Increment = 0.04

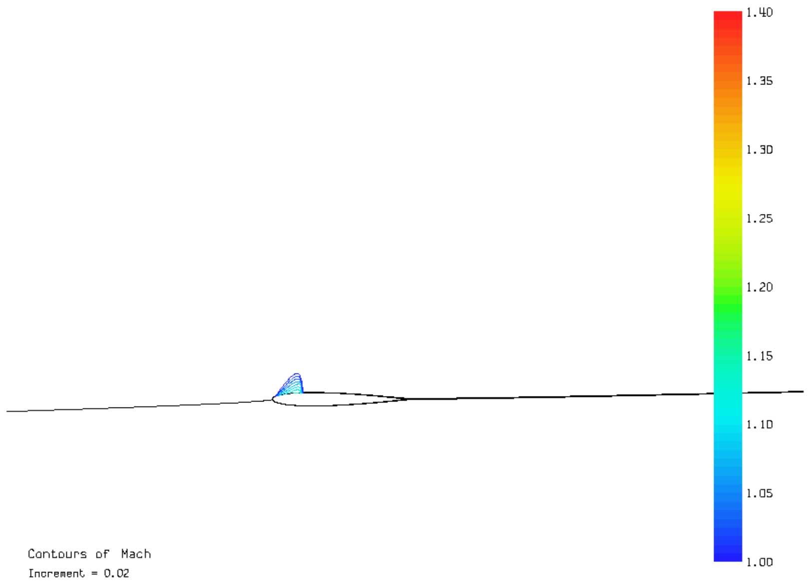

1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of Mach
Increment = 0.02



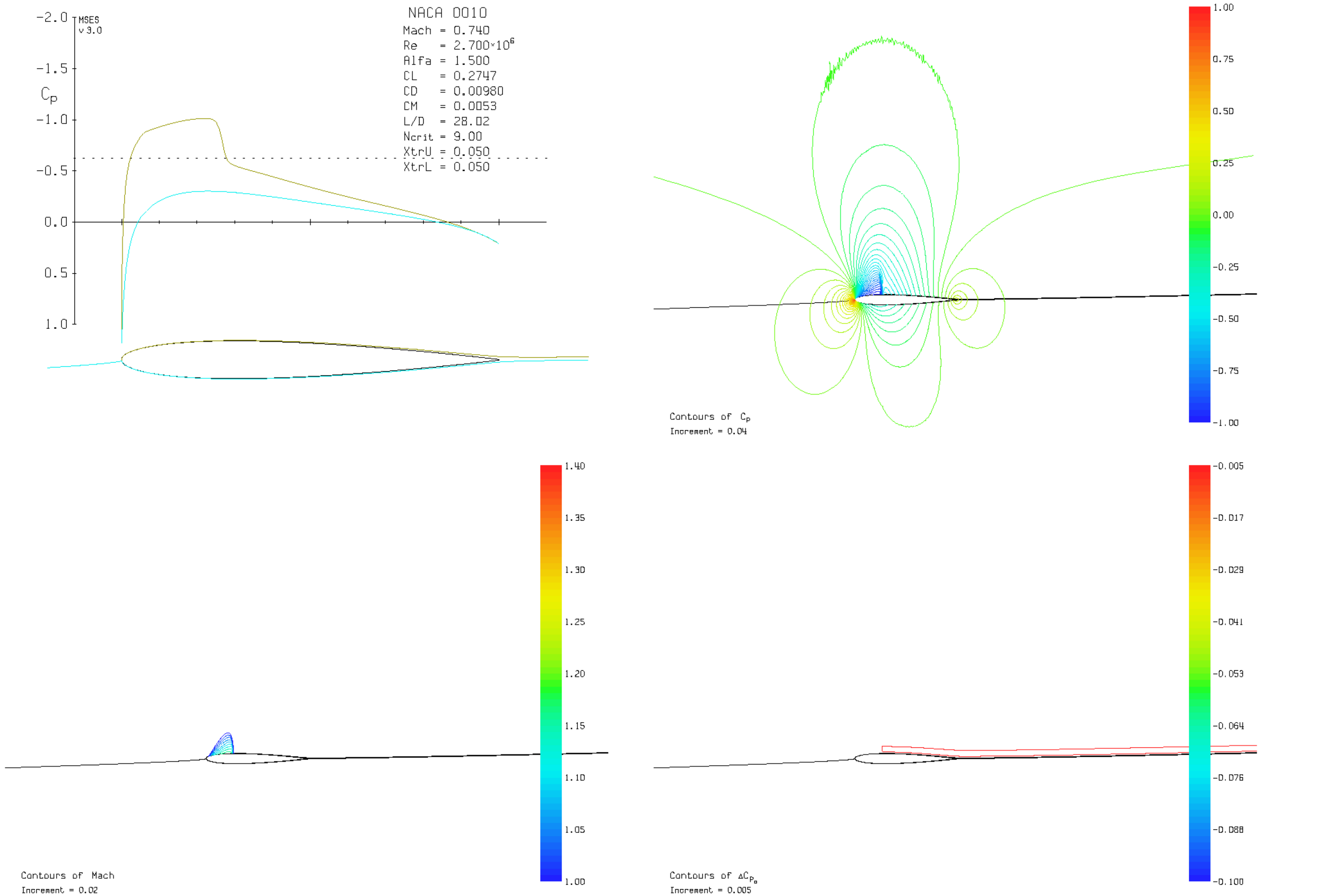
-2.0
MSES
v 3.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
NACA 0010
Mach = 0.730
Re = 2.700×10^6
Alfa = 1.500
CL = 0.2668
CD = 0.00938
CM = 0.0045
L/D = 28.45
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050



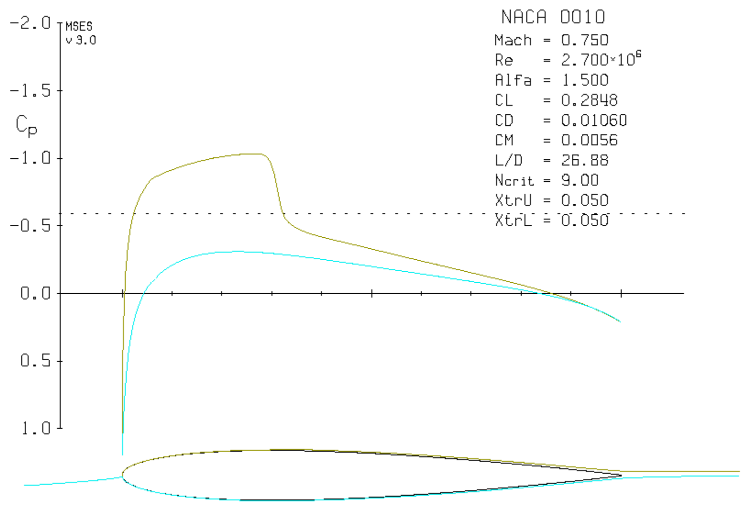
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Cp
Increment = 0.04



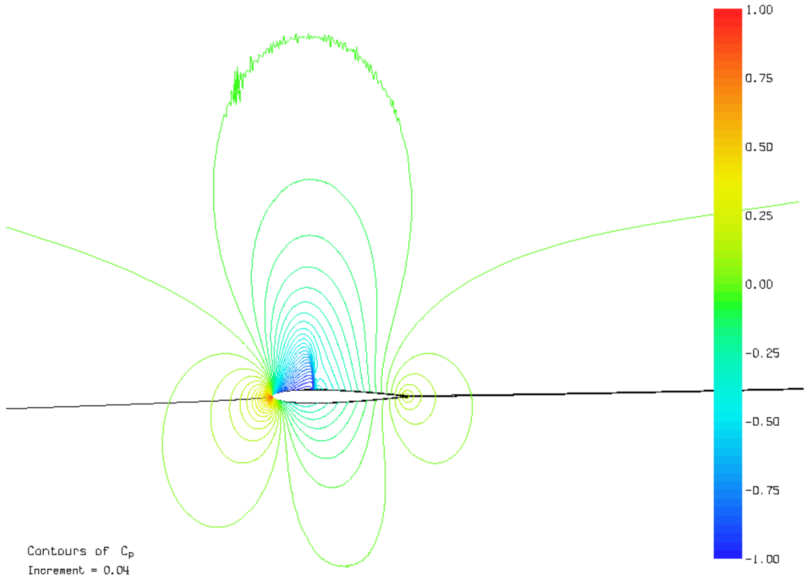
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of Mach
Increment = 0.02



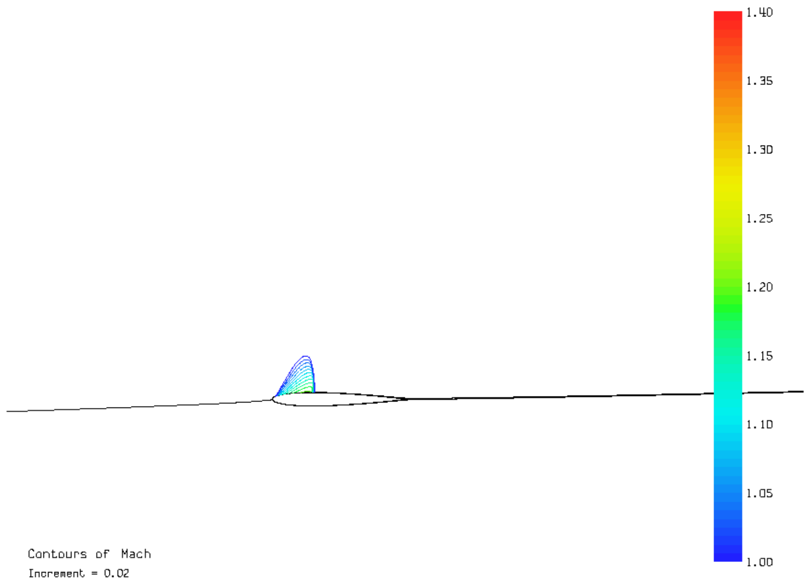
NACA 0010
Mach = 0.740
Re = 2.700×10^6
Alfa = 1.500
CL = 0.2747
CD = 0.00980
CM = 0.0053
L/D = 28.02
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of ΔCp0
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100



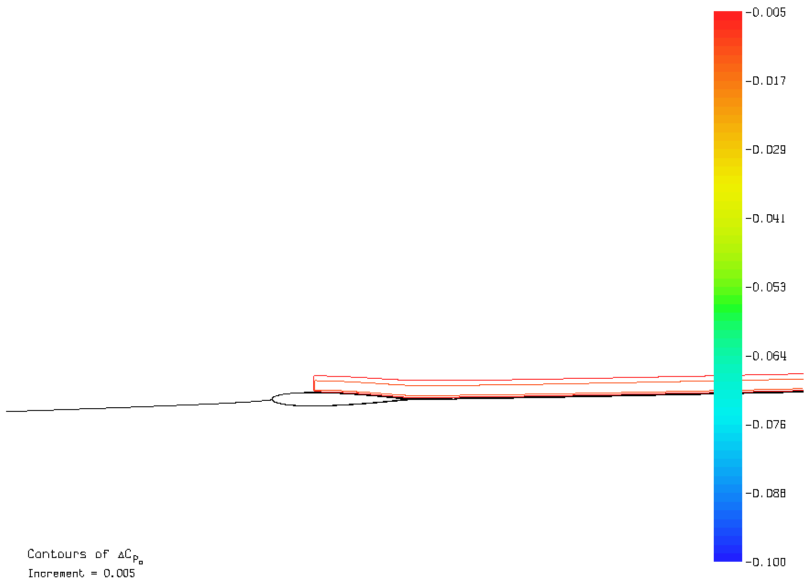
NACA 0010
Mach = 0.750
Re = 2.700×10^6
Alfa = 1.500
CL = 0.2848
CD = 0.01060
CM = 0.0056
L/D = 26.88
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES
v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0



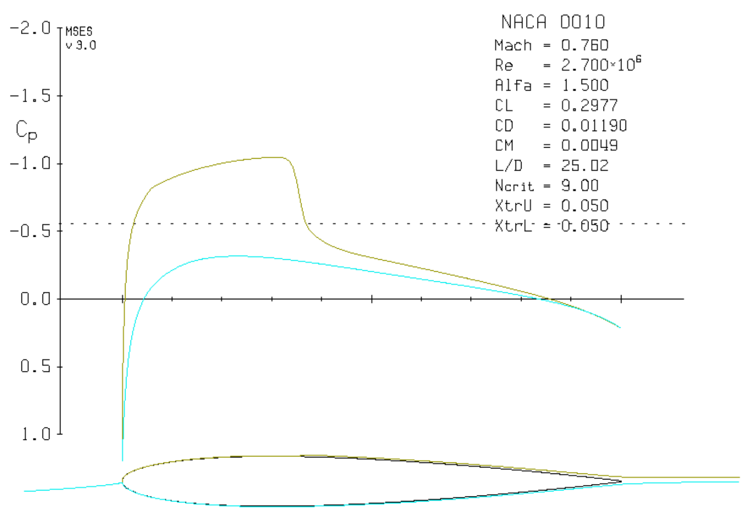
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Cp
Increment = 0.04

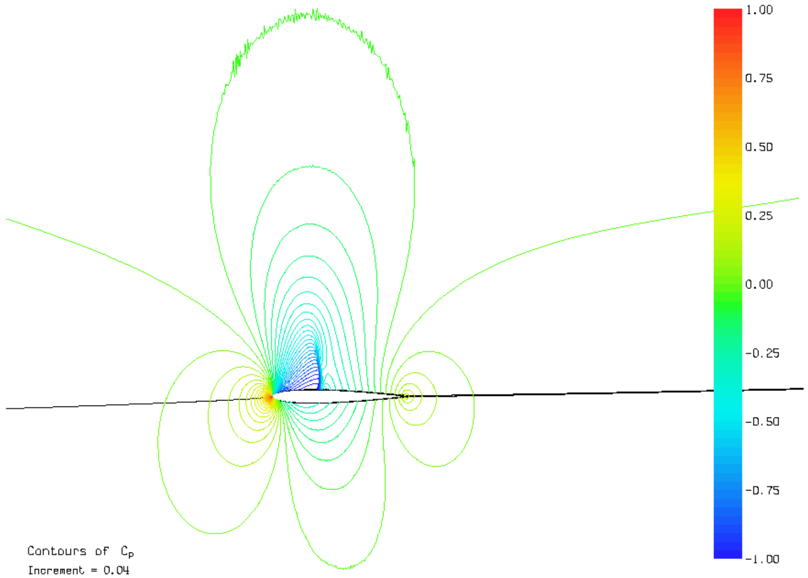


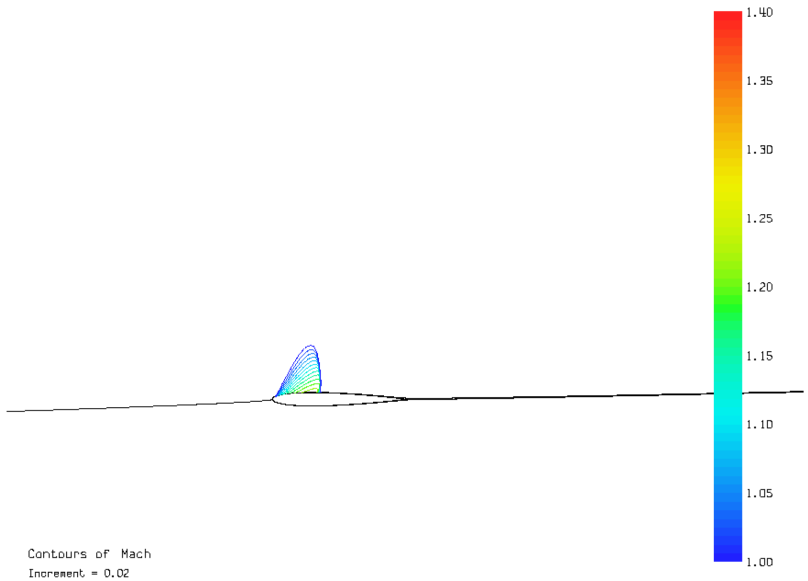
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of Mach
Increment = 0.02

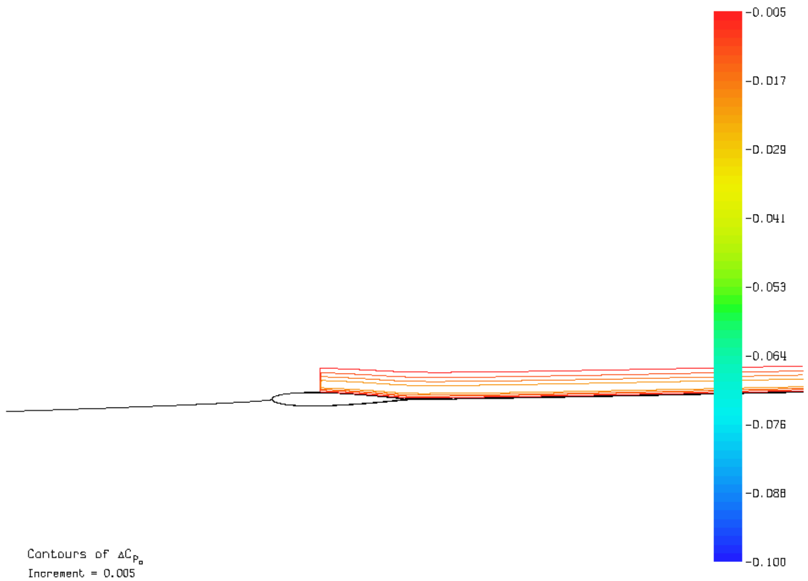


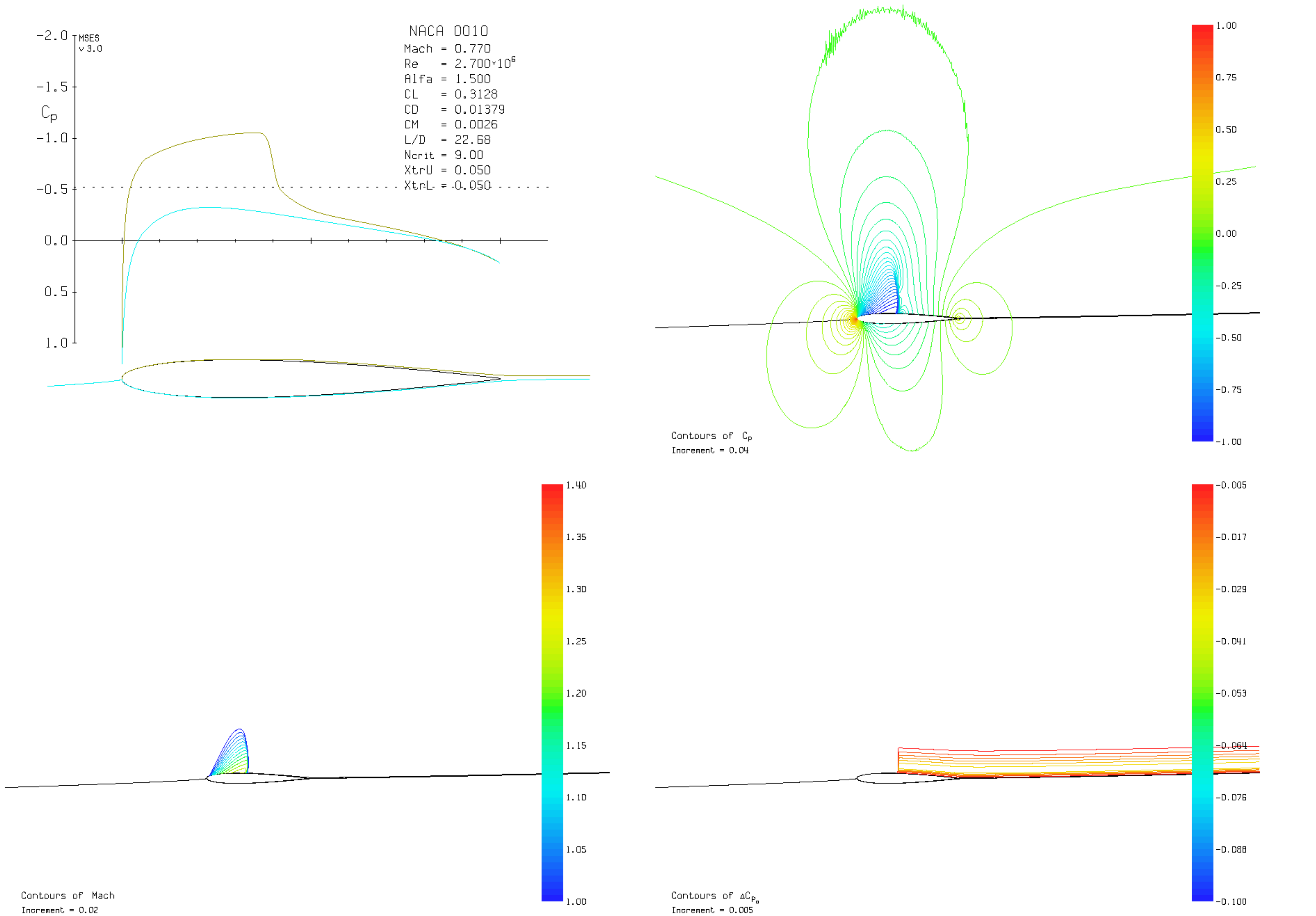
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100
Contours of ΔCp0
Increment = 0.005

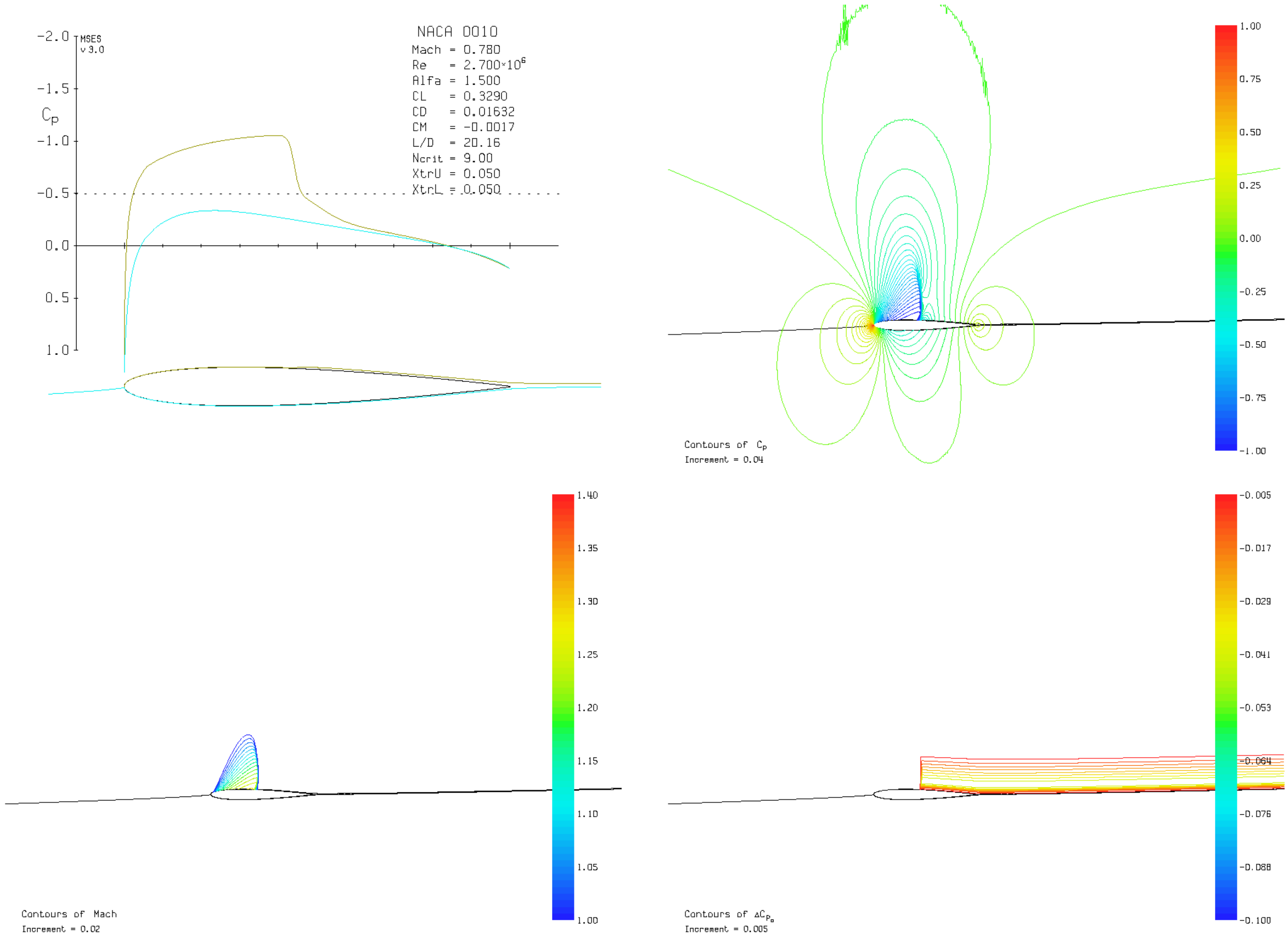

NACA 0010
Mach = 0.760
Re = 2.700×10^6
Alfa = 1.500
CL = 0.2977
CD = 0.01190
CM = 0.0049
L/D = 25.02
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0


Contours of C_p
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00

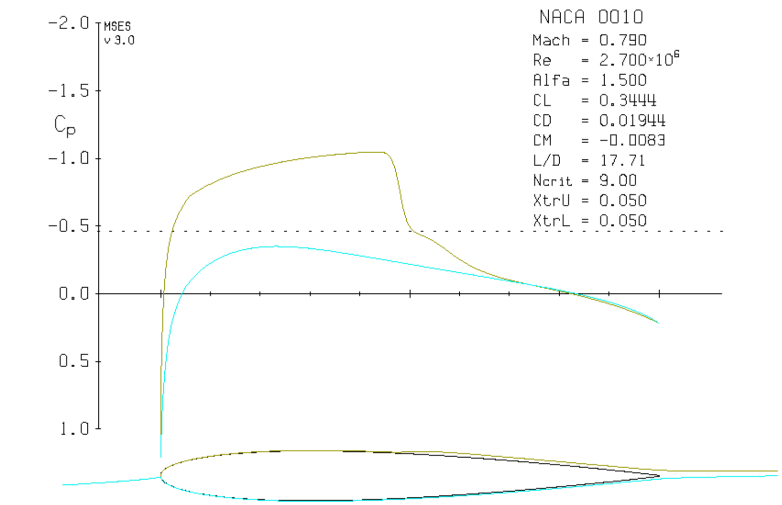

Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00


Contours of ΔC_{p_o}
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100

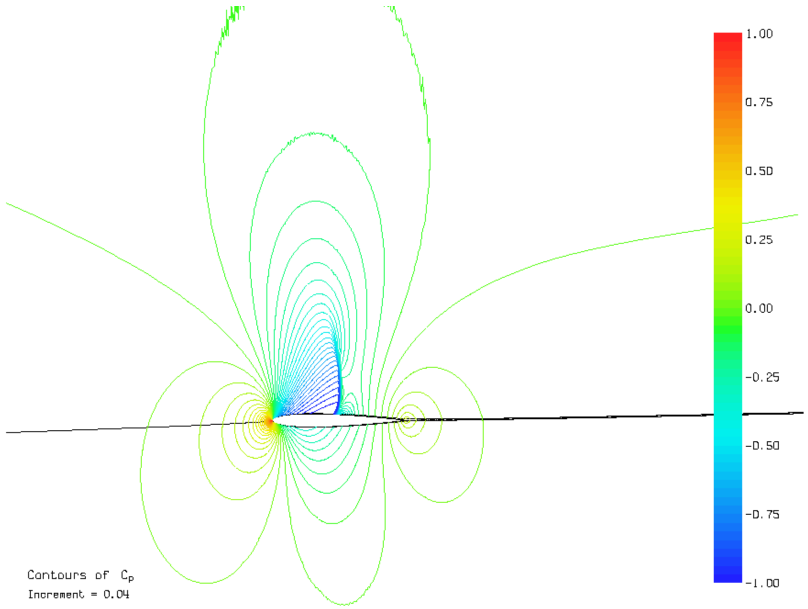

NACA 0010
Mach = 0.770
Re = 2.700×10^6
Alfa = 1.500
CL = 0.3128
CD = 0.01379
CM = 0.0026
L/D = 22.68
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of ΔCp0
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100



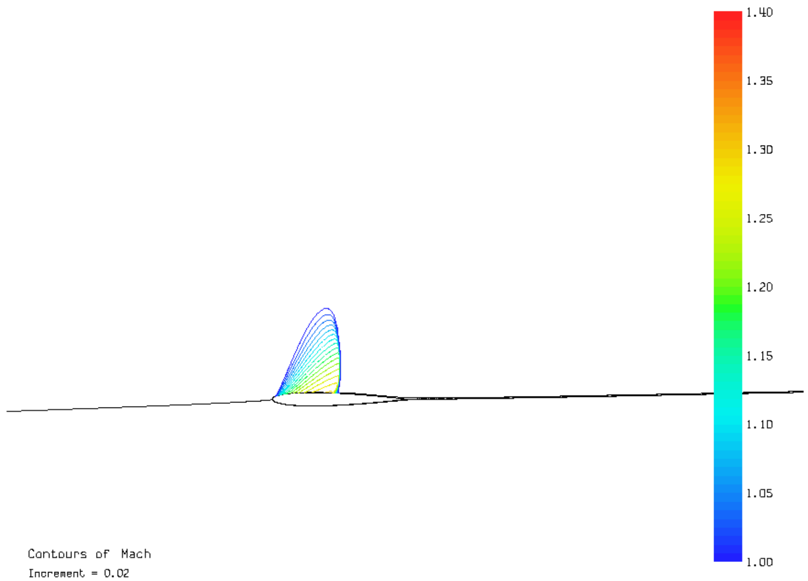
MSES v 3.0
NACA 0010
Mach = 0.780
Re = 2.700×10^6
Alfa = 1.500
CL = 0.3290
CD = 0.01632
CM = -0.0017
L/D = 20.16
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of Cp
Increment = 0.04
Contours of Mach
Increment = 0.02
Contours of ΔCp0
Increment = 0.005



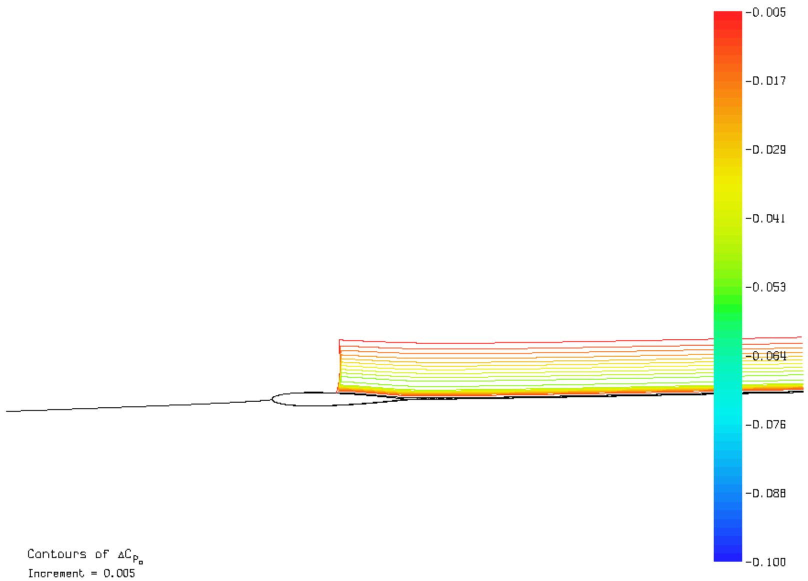
NACA 0010
Mach = 0.790
Re = 2.700×10^6
Alfa = 1.500
CL = 0.3444
CD = 0.01944
CM = -0.0083
L/D = 17.71
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0



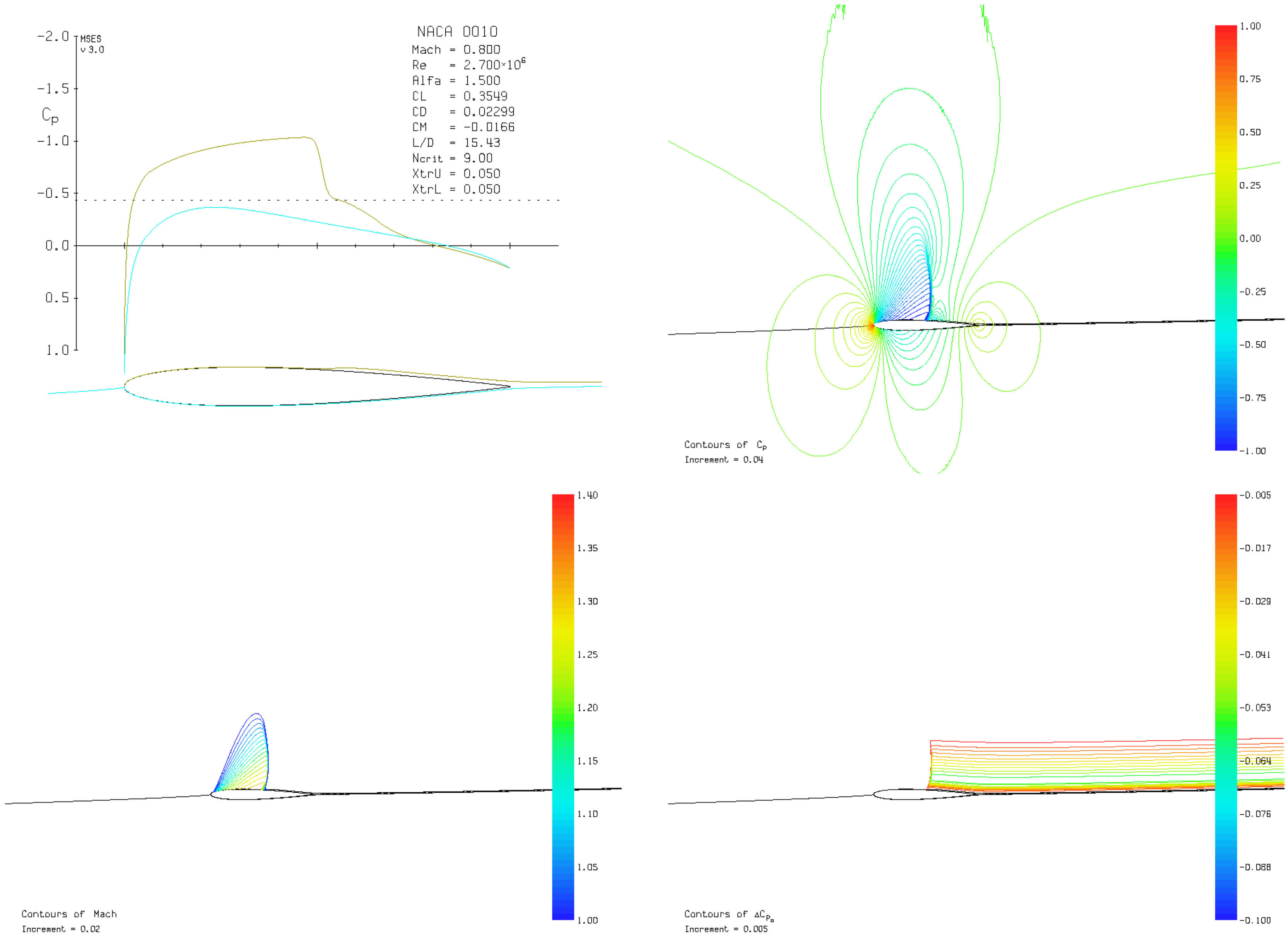
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00



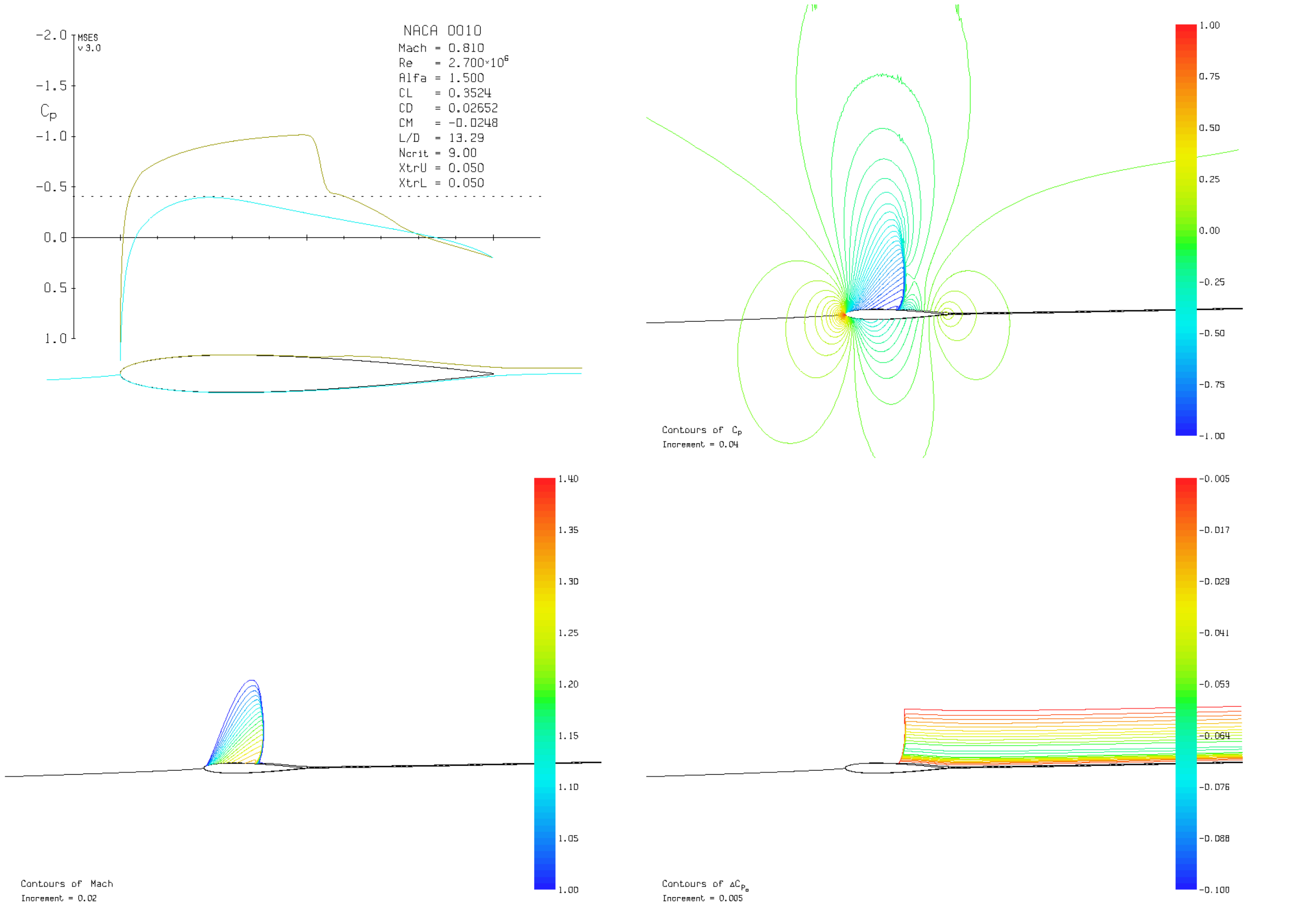
Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00

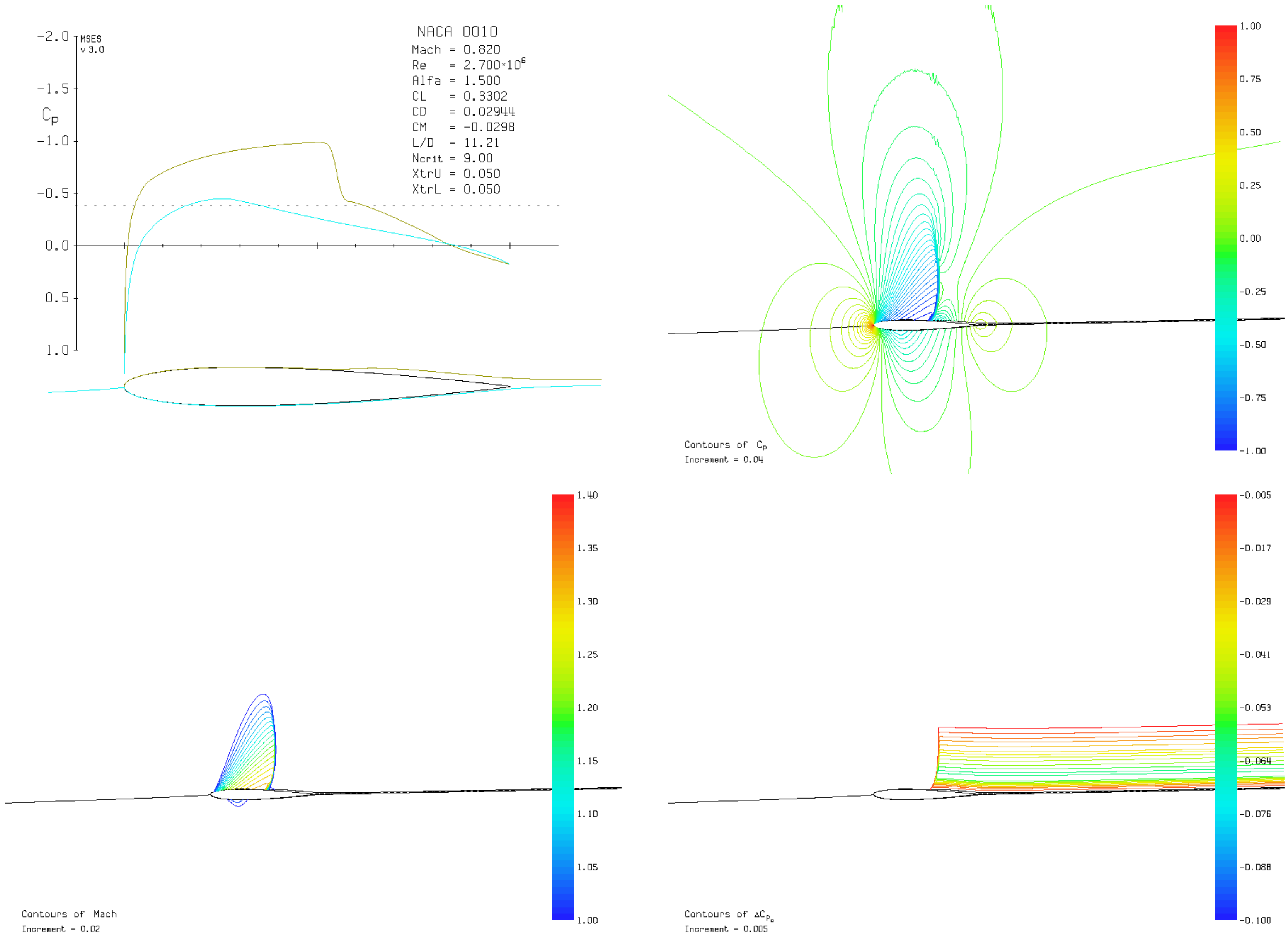


Contours of ΔCp0
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100

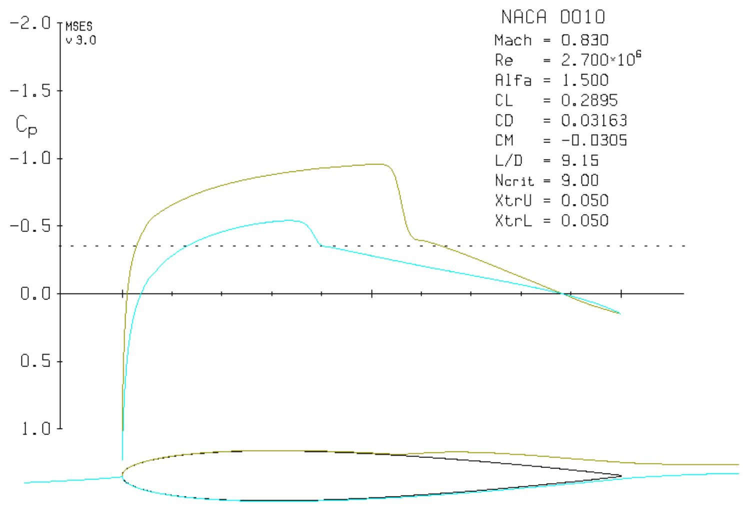


NACA 0010
Mach = 0.800
Re = 2.700×10^6
Alfa = 1.500
CL = 0.3549
CD = 0.02299
CM = -0.0166
L/D = 15.43
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Cp
Increment = 0.04
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of Mach
Increment = 0.02
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100
Contours of ΔCp0
Increment = 0.005

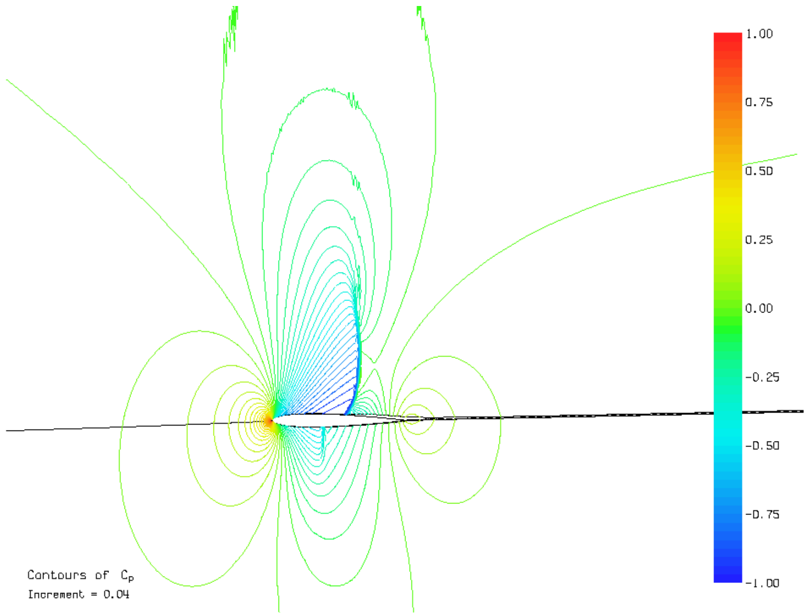

MSES
v 3.0
NACA 0010
Mach = 0.810
Re = 2.700×10^6
Alfa = 1.500
CL = 0.3524
CD = 0.02652
CM = -0.0248
L/D = 13.29
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of ΔCp0
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100



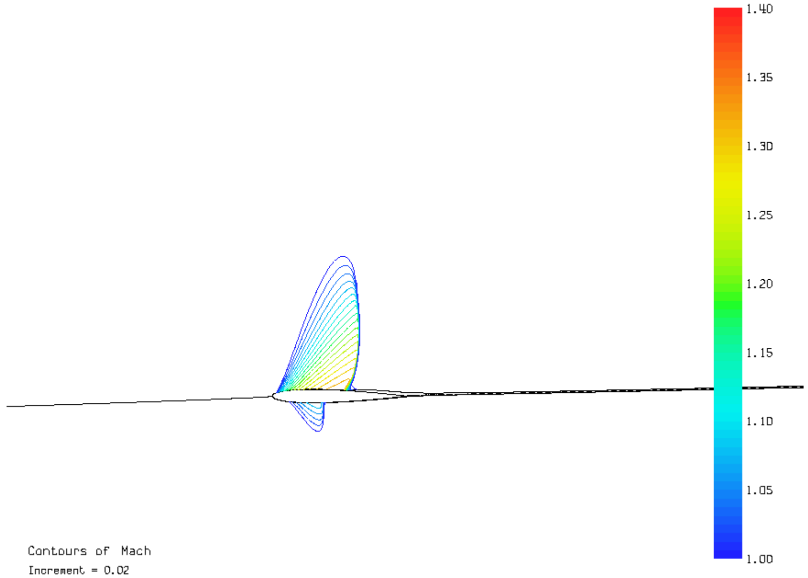
NACA 0010
Mach = 0.820
Re = 2.700×10^6
Alfa = 1.500
CL = 0.3302
CD = 0.02944
CM = -0.0298
L/D = 11.21
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of ΔCp0
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100



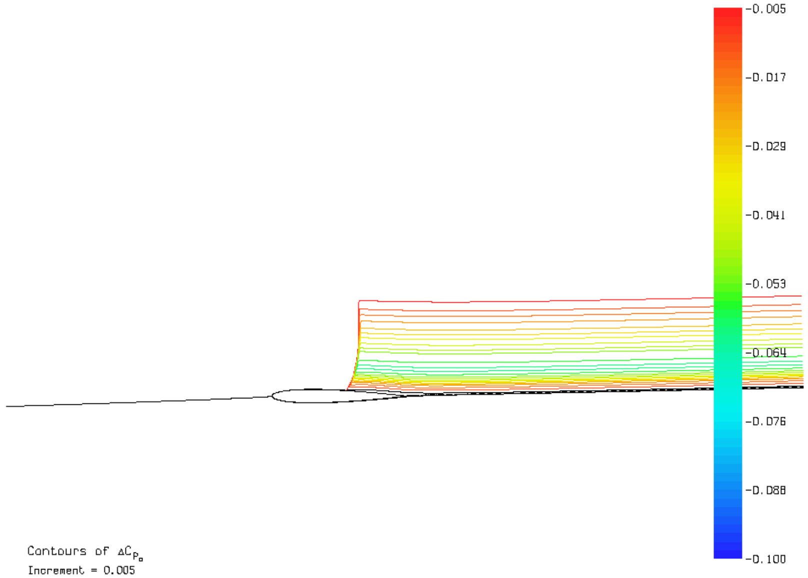
NACA 0010
Mach = 0.830
Re = 2.700×10^6
Alfa = 1.500
CL = 0.2895
CD = 0.03163
CM = -0.0305
L/D = 9.15
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0



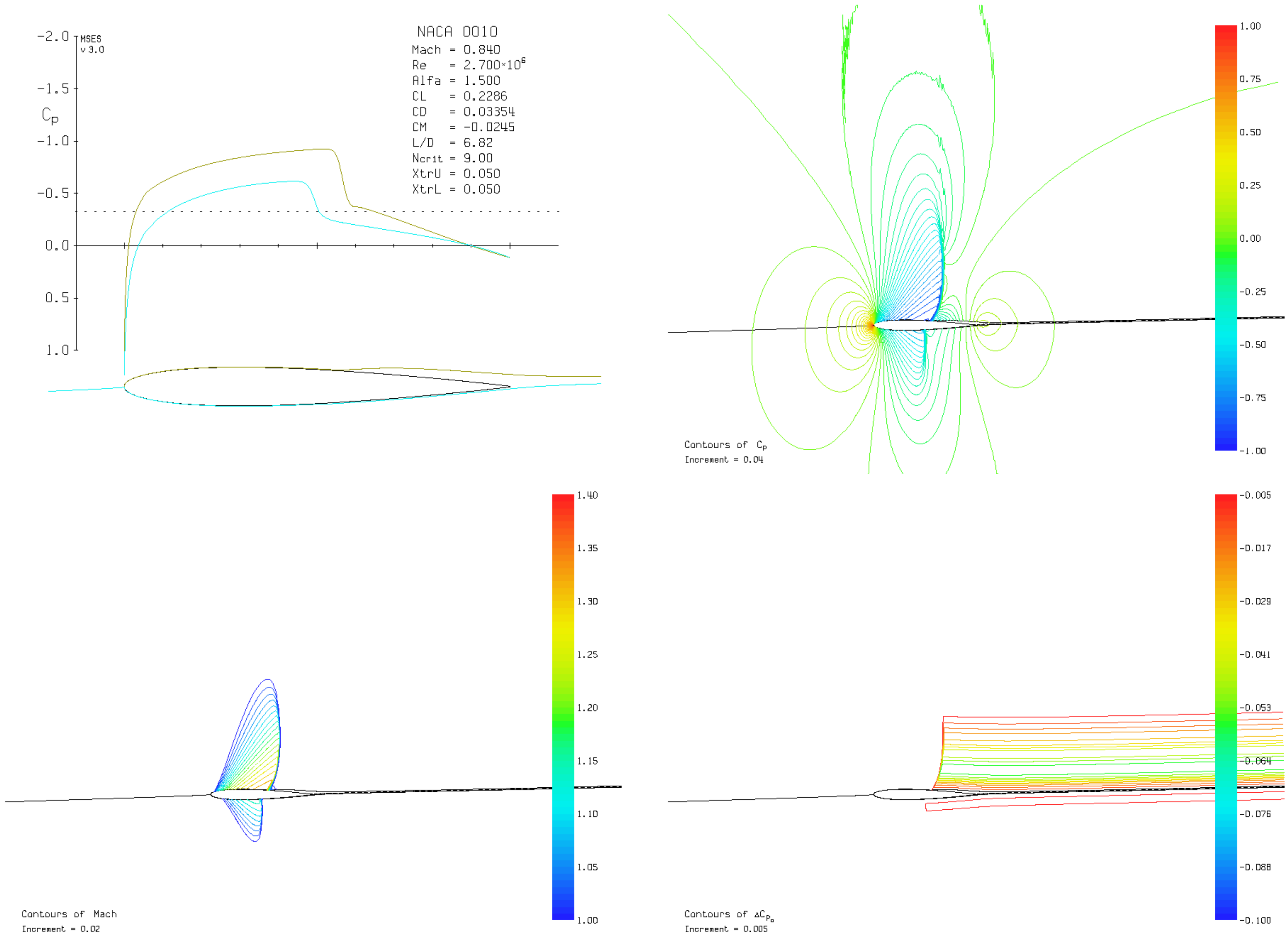
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00

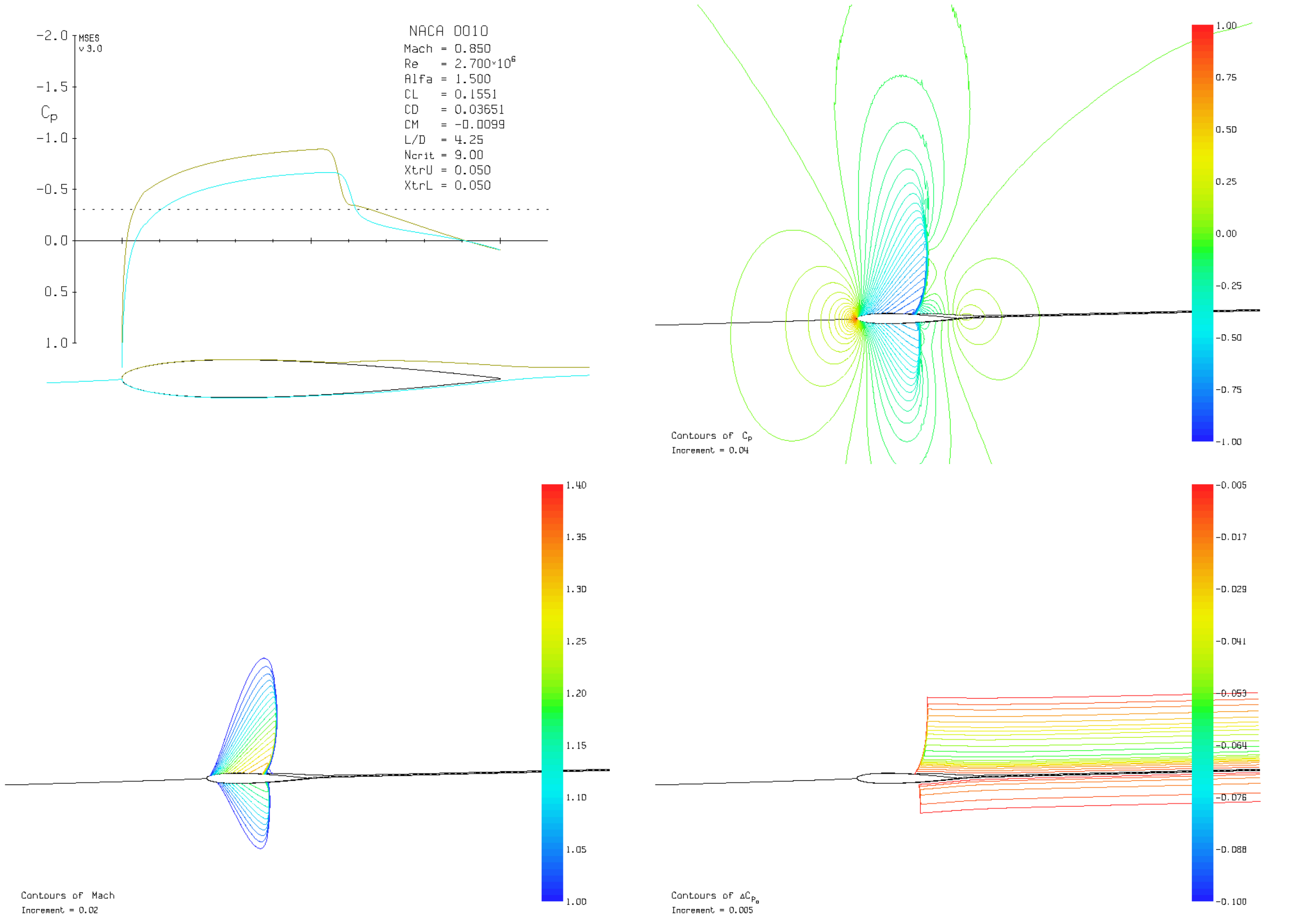


Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00

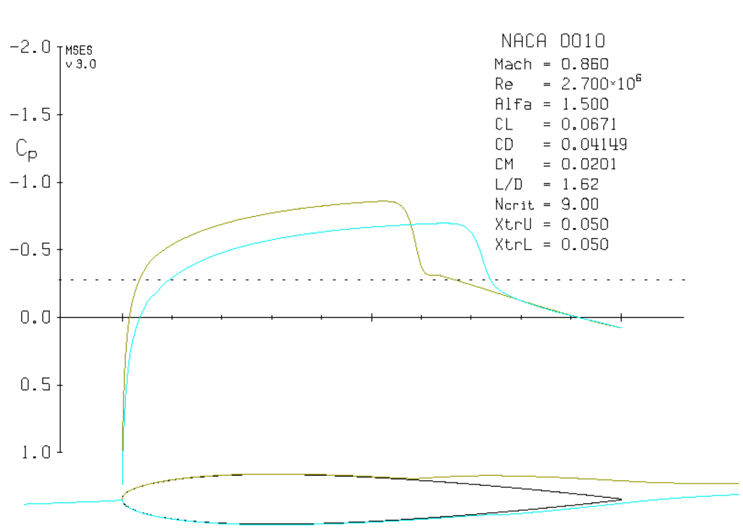


Contours of ΔCpo
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100

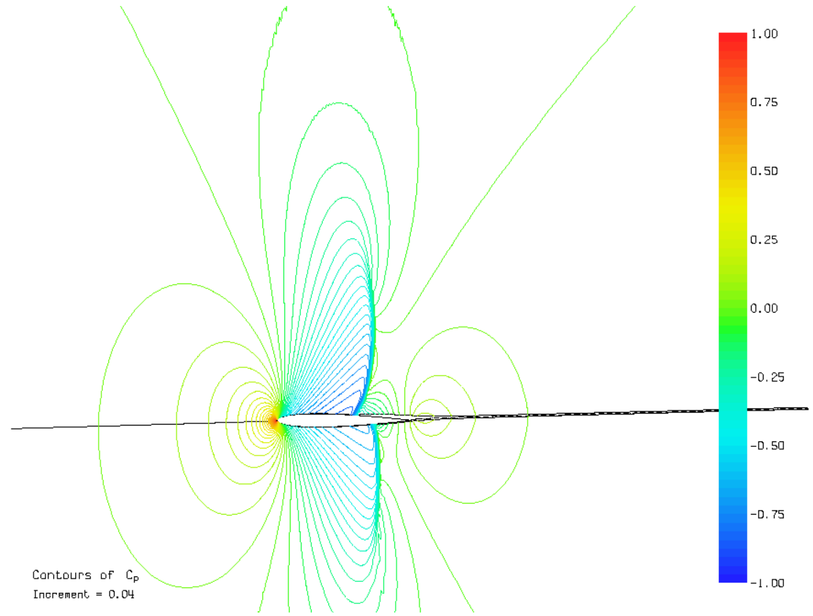

NACA 0010
Mach = 0.840
Re = 2.700×10^6
Alfa = 1.500
CL = 0.2286
CD = 0.03354
CM = -0.0245
L/D = 6.82
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES
v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of ΔCpo
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100



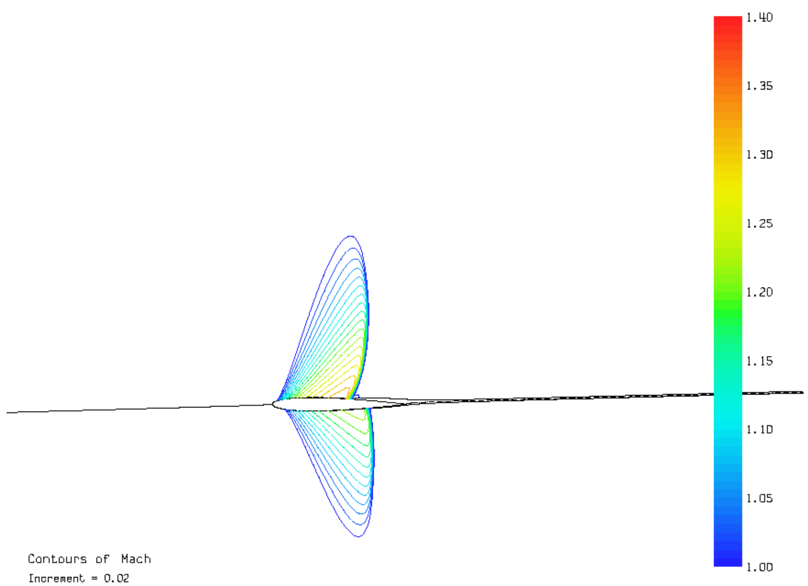
MSES v 3.0
NACA 0010
Mach = 0.850
Re = 2.700×10^6
Alfa = 1.500
CL = 0.1551
CD = 0.03651
CM = -0.0099
L/D = 4.25
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
Cp
Contours of Cp
Increment = 0.04
Contours of Mach
Increment = 0.02
Contours of ΔCp0
Increment = 0.005



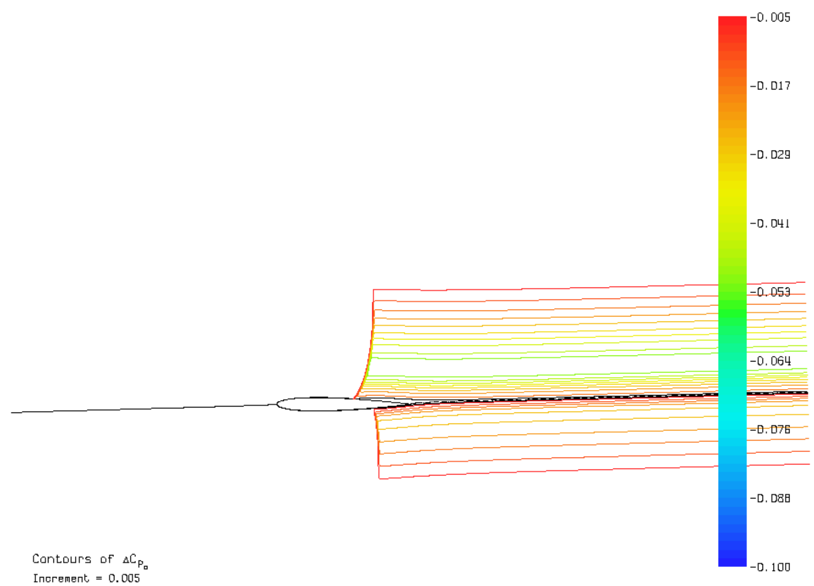
-2.0
MSES
v 3.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
NACA 0010
Mach = 0.860
Re = 2.700×10^6
Alfa = 1.500
CL = 0.0671
CD = 0.04149
CM = 0.0201
L/D = 1.62
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050



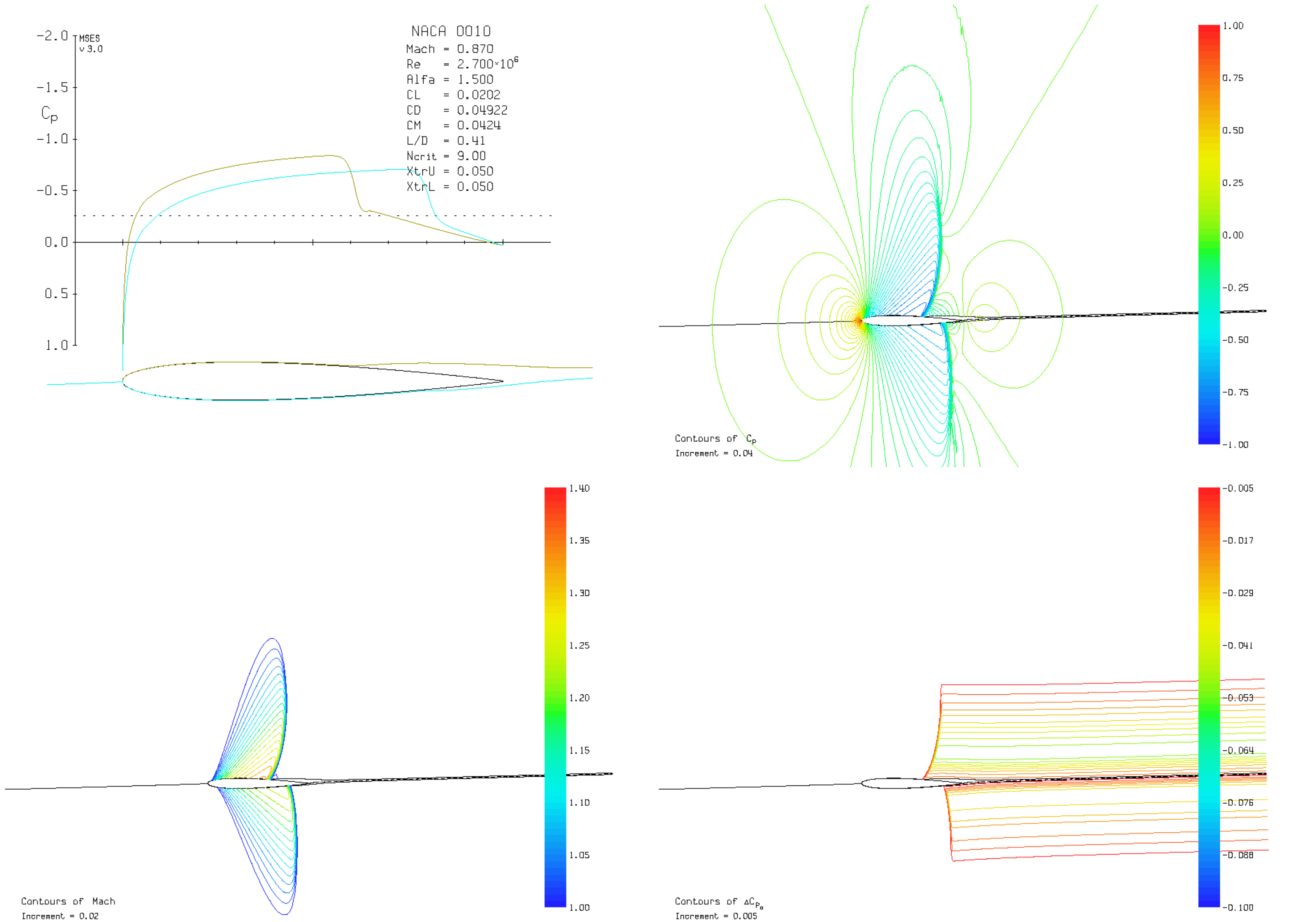
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Cp
Increment = 0.04



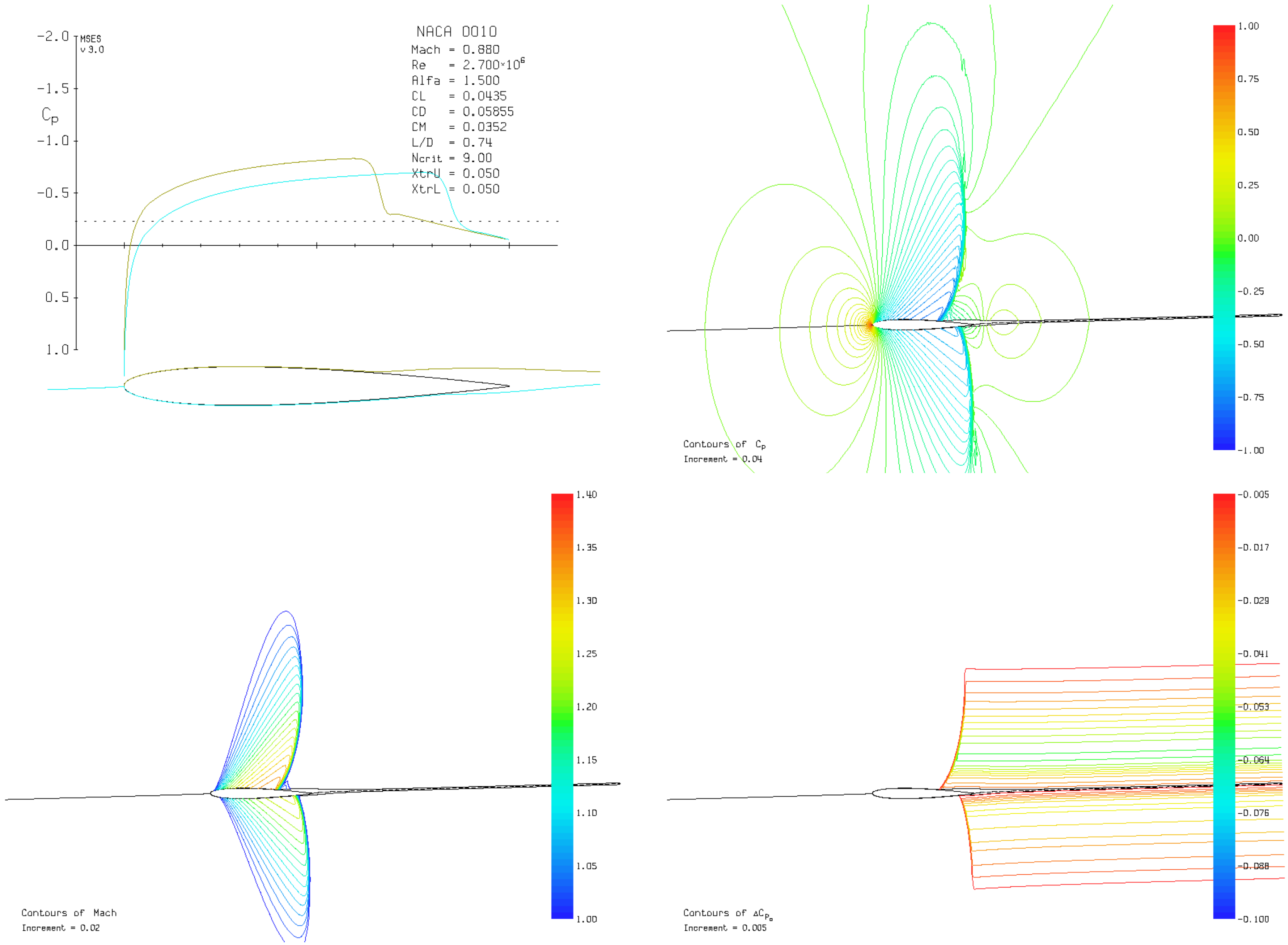
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of Mach
Increment = 0.02



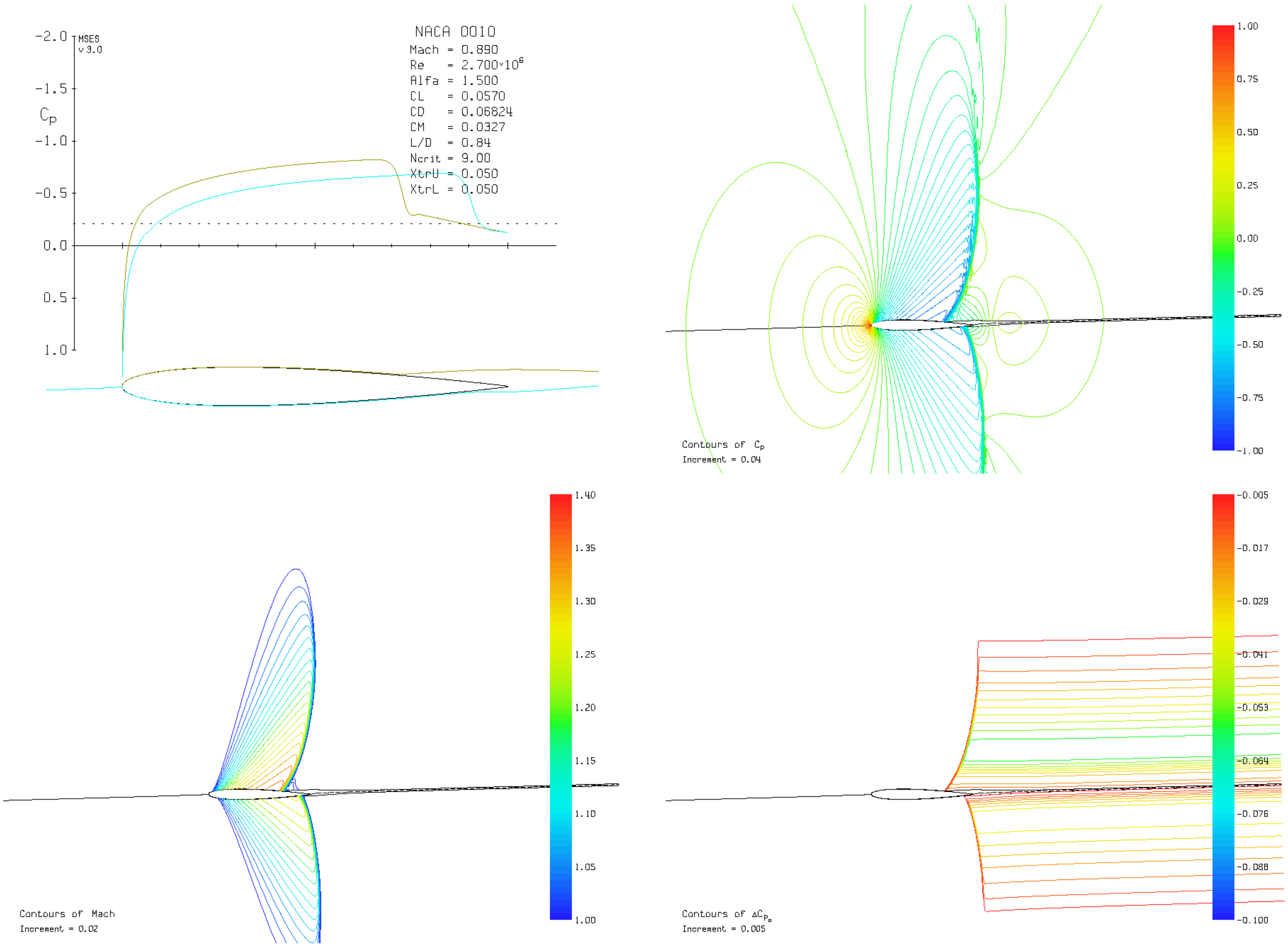
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100
Contours of ΔCp0
Increment = 0.005



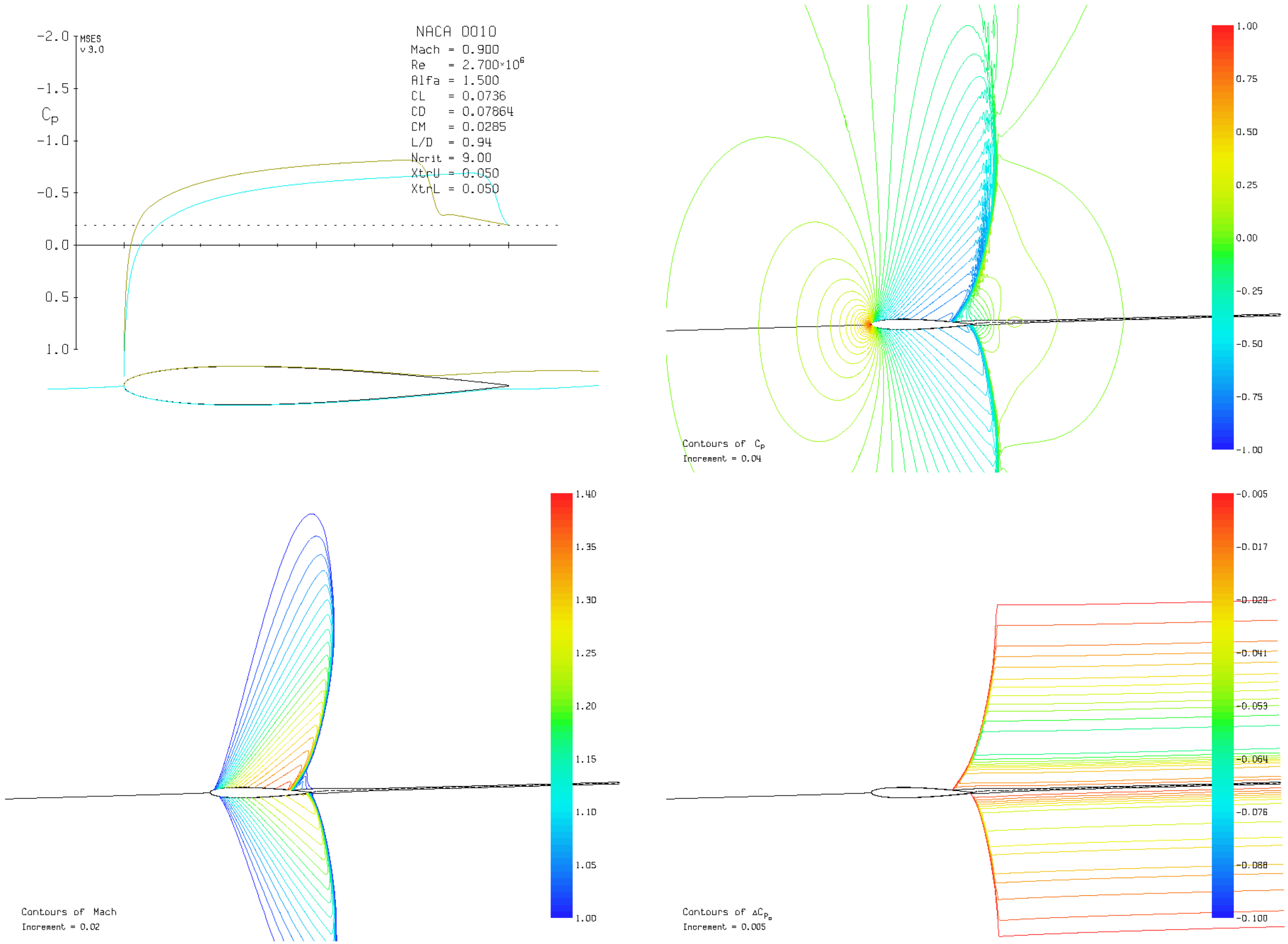
NACA 0010
Mach = 0.870
Re = 2.700×10^6
Alfa = 1.500
CL = 0.0202
CD = 0.04922
CM = 0.0424
L/D = 0.41
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of ΔCp0
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100



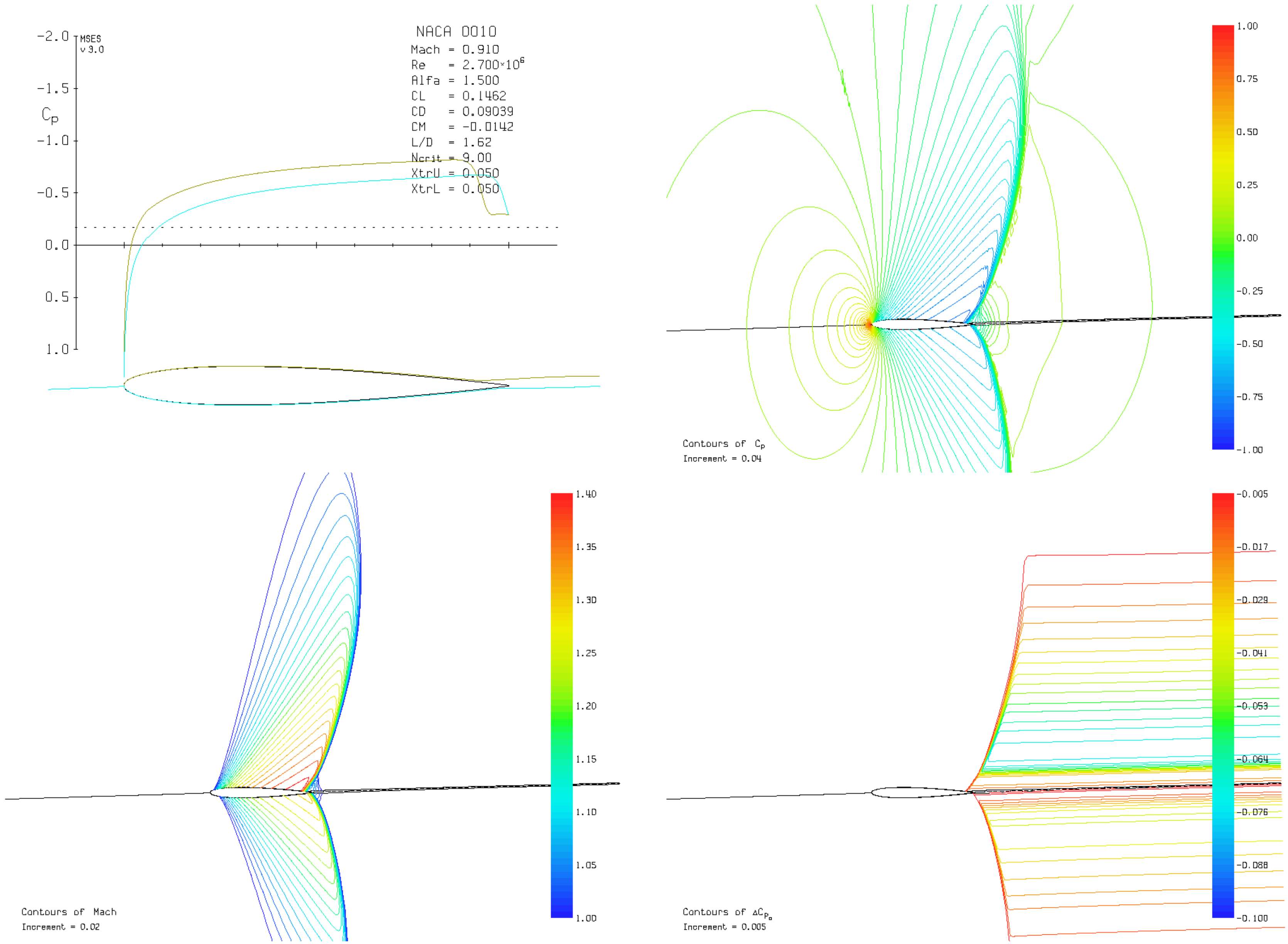
MSES
v 3.0
NACA 0010
Mach = 0.880
Re = 2.700×10^6
Alfa = 1.500
CL = 0.0435
CD = 0.05855
CM = 0.0352
L/D = 0.74
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of ΔCp0
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100

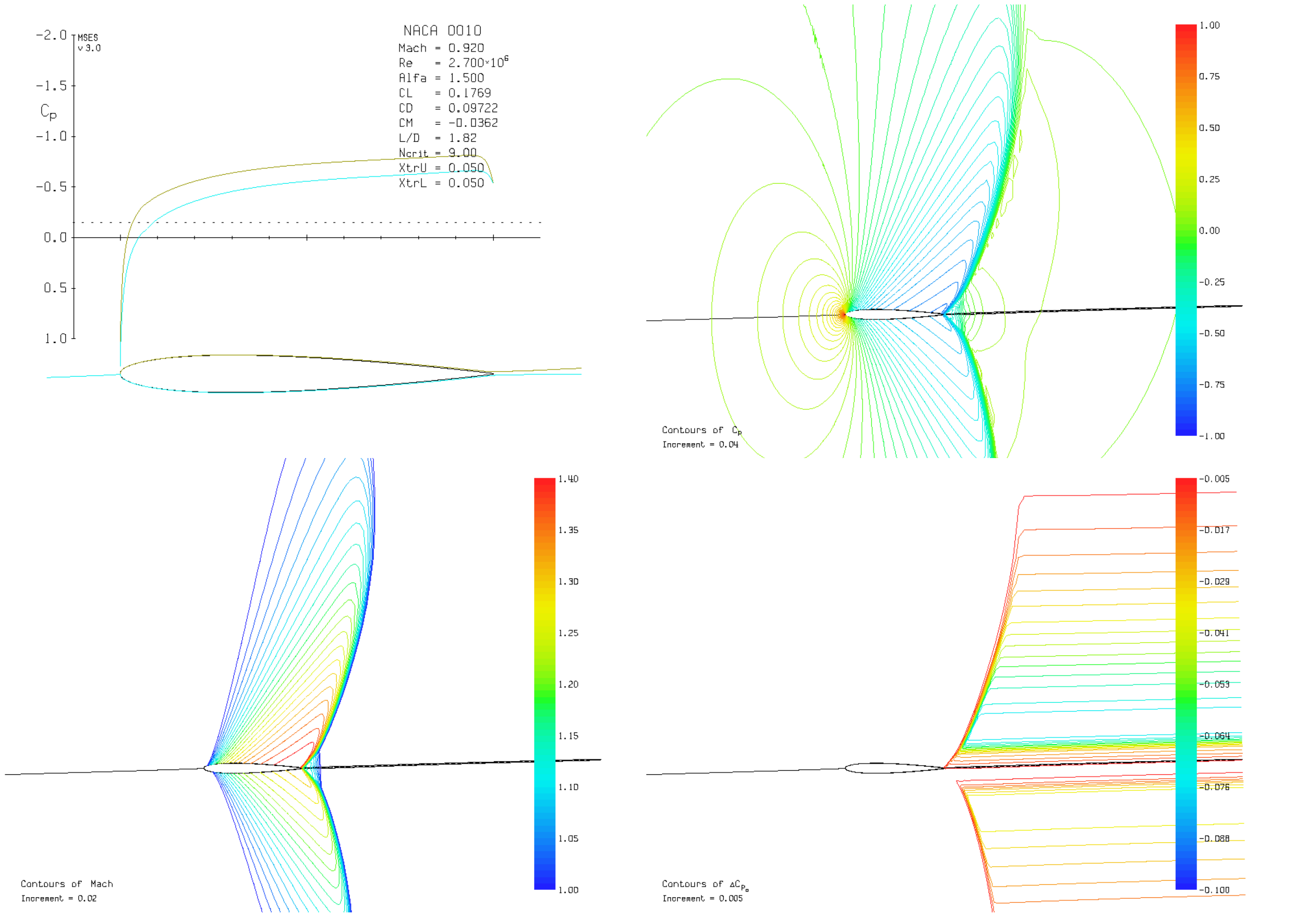


NACA 0010
Mach = 0.890
Re = 2.700×10^6
Alfa = 1.500
CL = 0.0570
CD = 0.06824
CM = 0.0327
L/D = 0.84
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
Contours of Cp
Increment = 0.04
Contours of Mach
Increment = 0.02
Contours of ΔCp0
Increment = 0.005

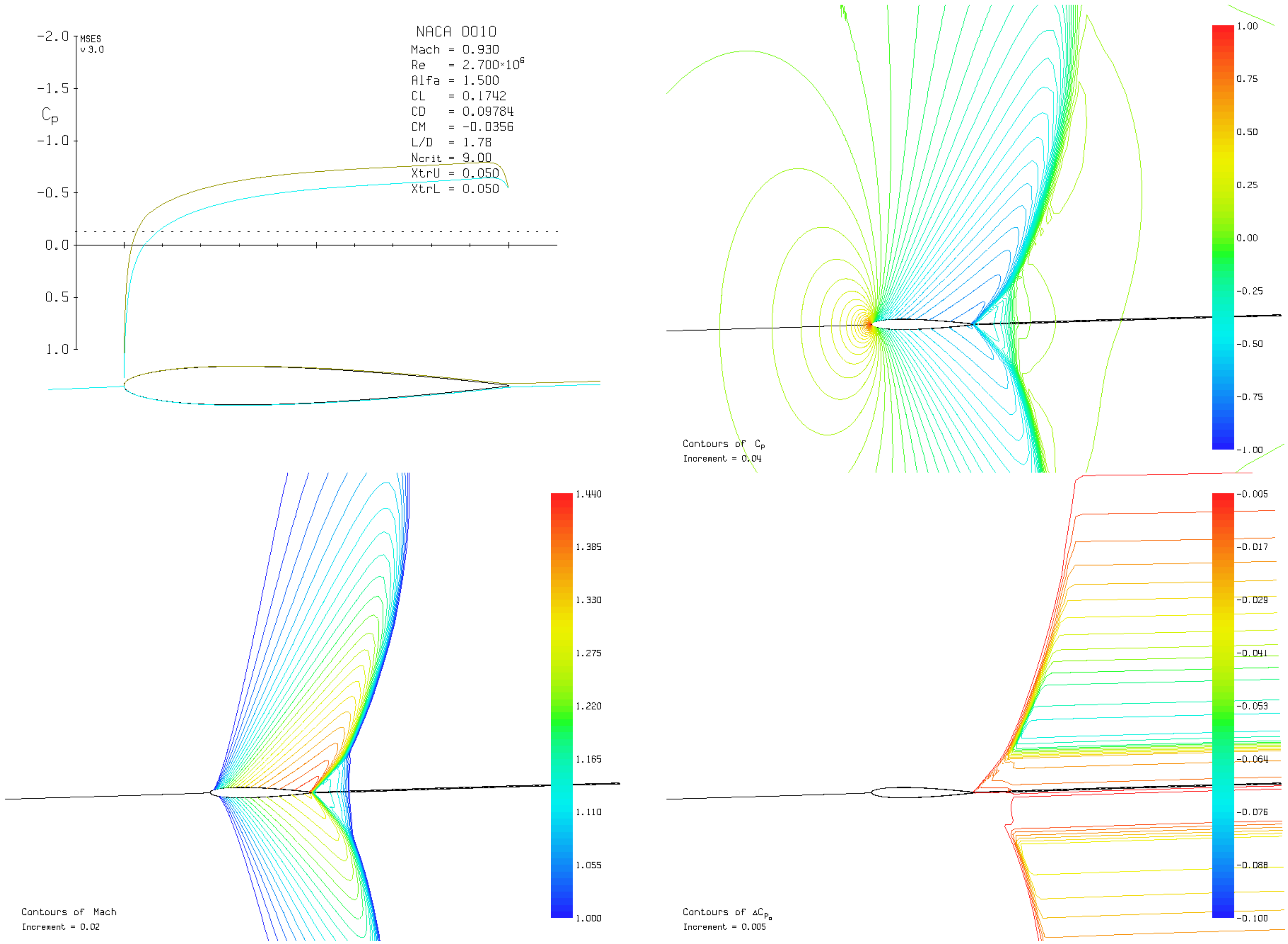


MSES
v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
NACA 0010
Mach = 0.900
Re = 2.700×10^6
Alfa = 1.500
CL = 0.0736
CD = 0.07864
CM = 0.0285
L/D = 0.94
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of ΔCp0
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100

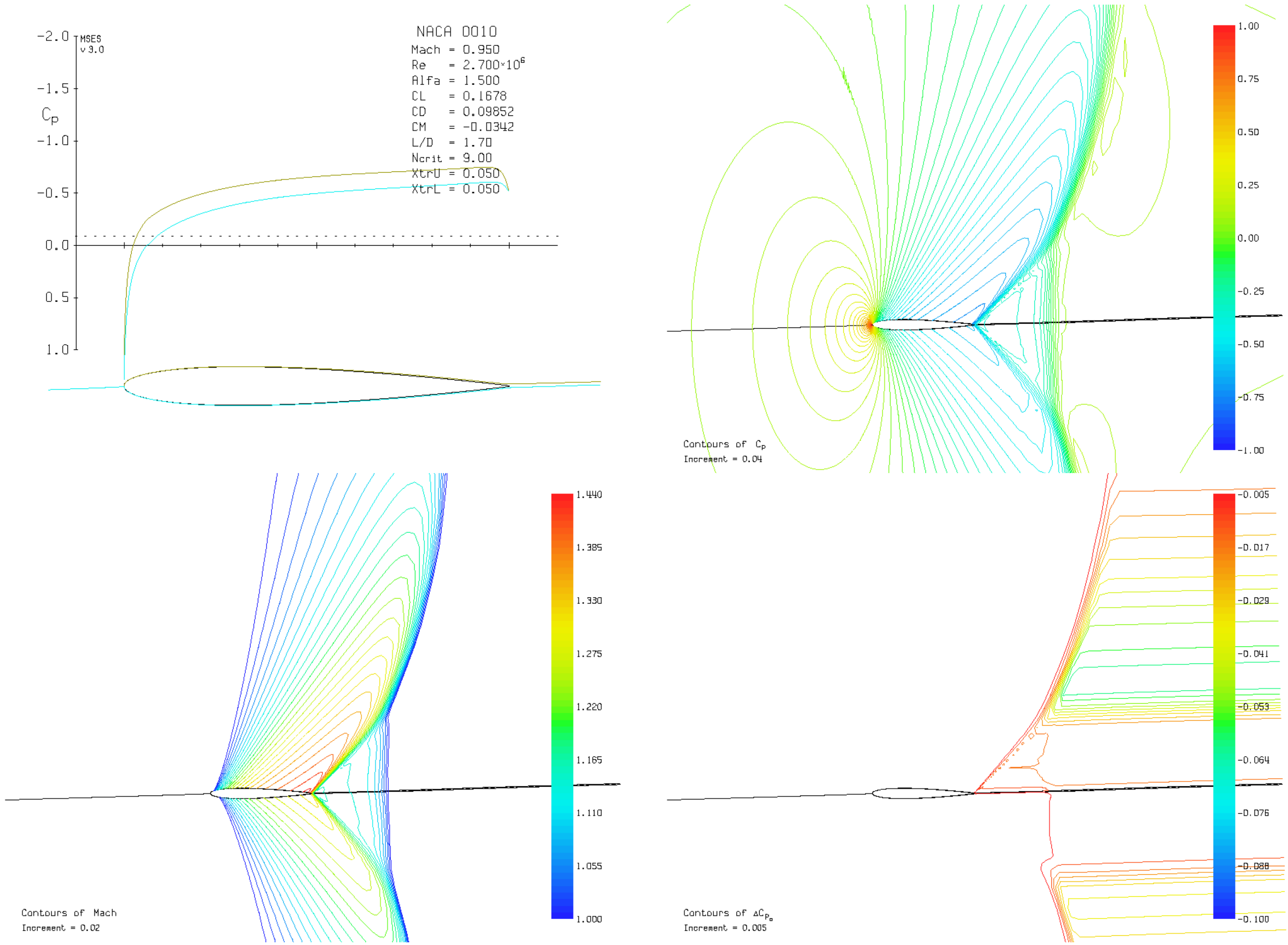

MSES
v 3.0
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
NACA 0010
Mach = 0.910
Re = 2.700×10^6
Alfa = 1.500
CL = 0.1462
CD = 0.09039
CM = -0.0142
L/D = 1.62
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Cp
Increment = 0.04
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of Mach
Increment = 0.02
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100
Contours of ΔCp0
Increment = 0.005



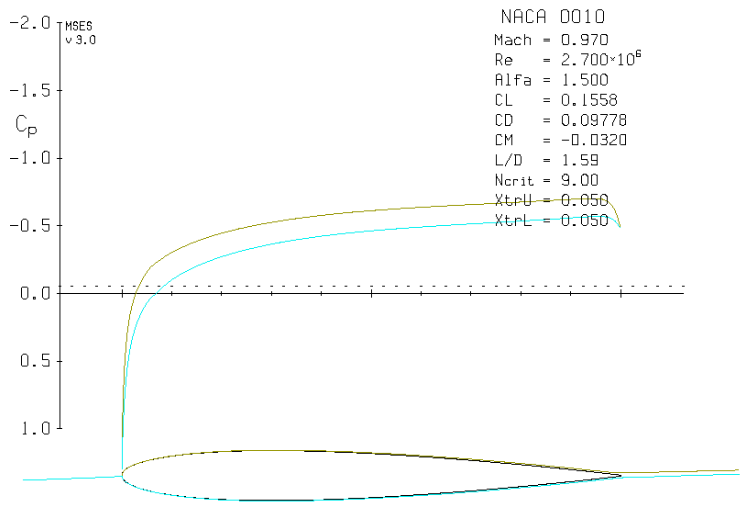
NACA 0010
Mach = 0.920
Re = 2.700×10^6
Alfa = 1.500
CL = 0.1769
CD = 0.09722
CM = -0.0362
L/D = 1.82
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Mach
Increment = 0.02
1.40
1.35
1.30
1.25
1.20
1.15
1.10
1.05
1.00
Contours of ΔCp0
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100

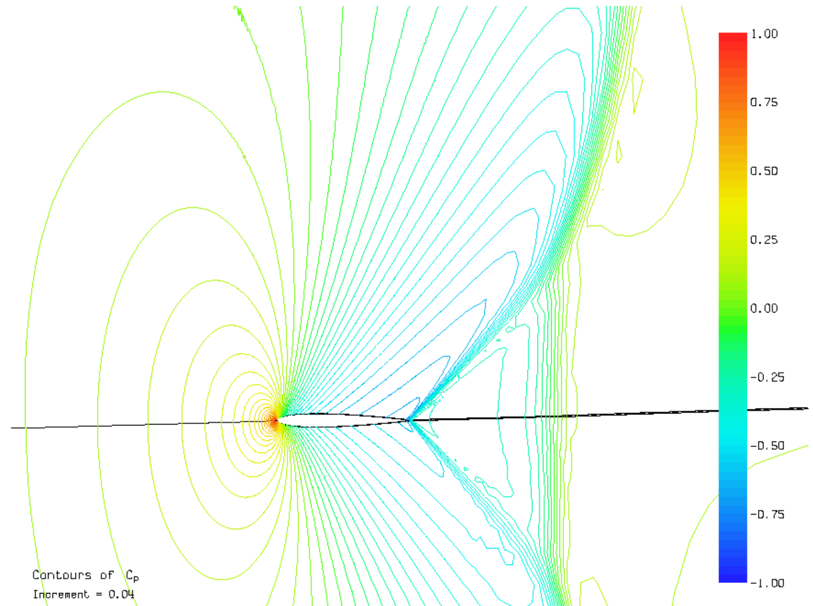


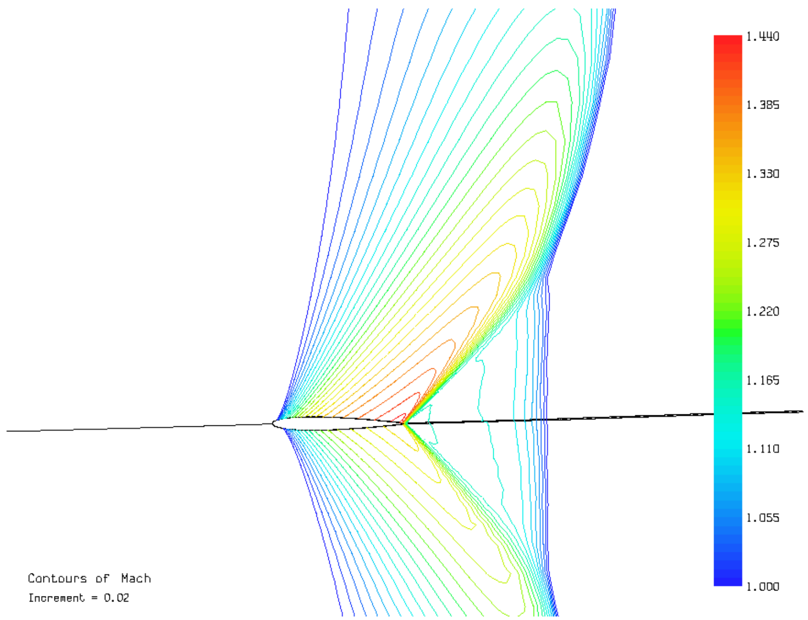
NACA 0010
Mach = 0.930
Re = 2.700×10^6
Alfa = 1.500
CL = 0.1742
CD = 0.09784
CM = -0.0356
L/D = 1.78
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Mach
Increment = 0.02
1.440
1.385
1.330
1.275
1.220
1.165
1.110
1.055
1.000
Contours of ΔCp0
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100



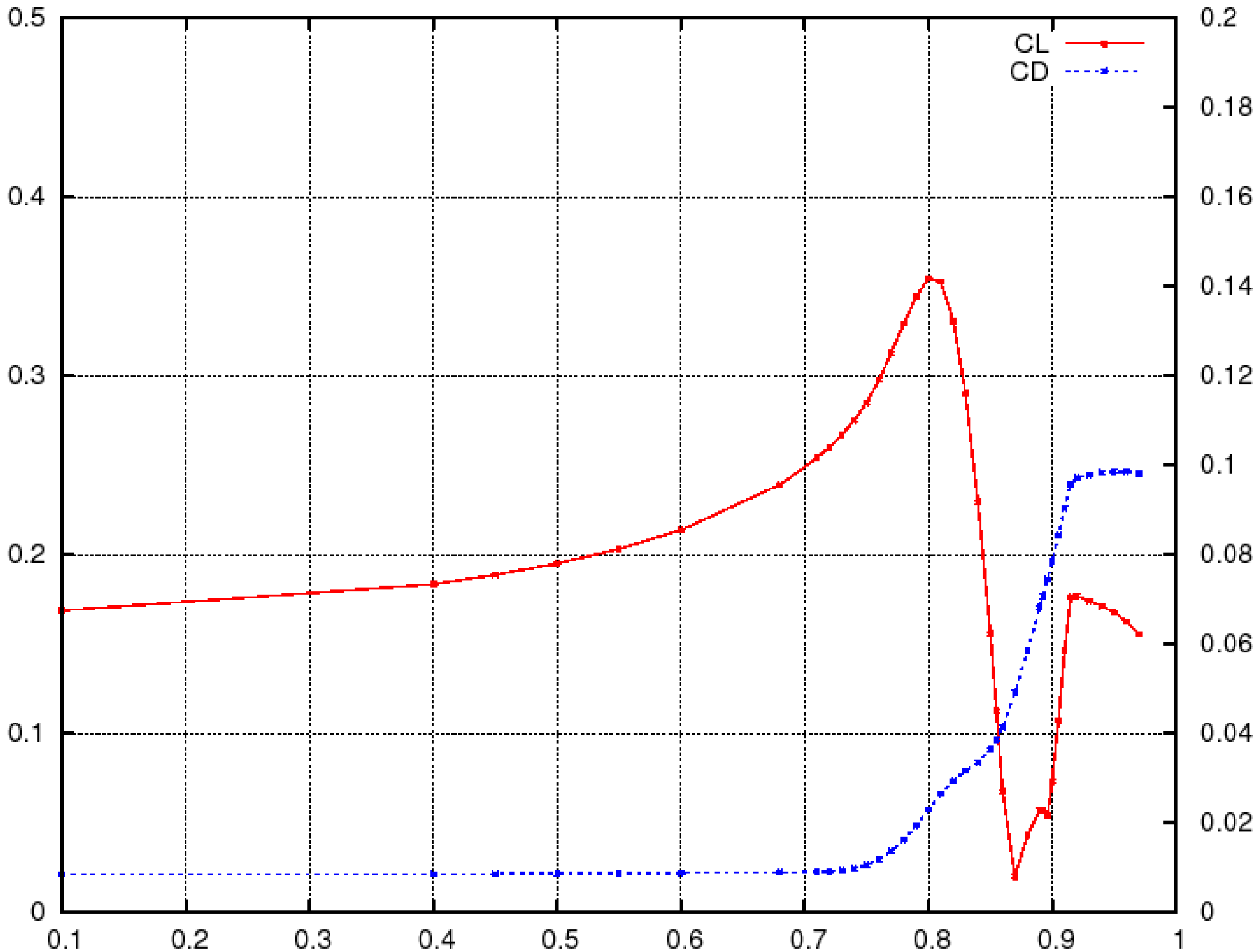
NACA 0010
Mach = 0.950
Re = 2.700×10^6
Alfa = 1.500
CL = 0.1678
CD = 0.09852
CM = -0.0342
L/D = 1.70
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00
Contours of Mach
Increment = 0.02
1.440
1.385
1.330
1.275
1.220
1.165
1.110
1.055
1.000
Contours of ΔCpo
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100


NACA 0010
Mach = 0.970
Re = 2.700×10^6
Alfa = 1.500
CL = 0.1558
CD = 0.09778
CM = -0.0320
L/D = 1.59
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
MSES
v 3.0
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0

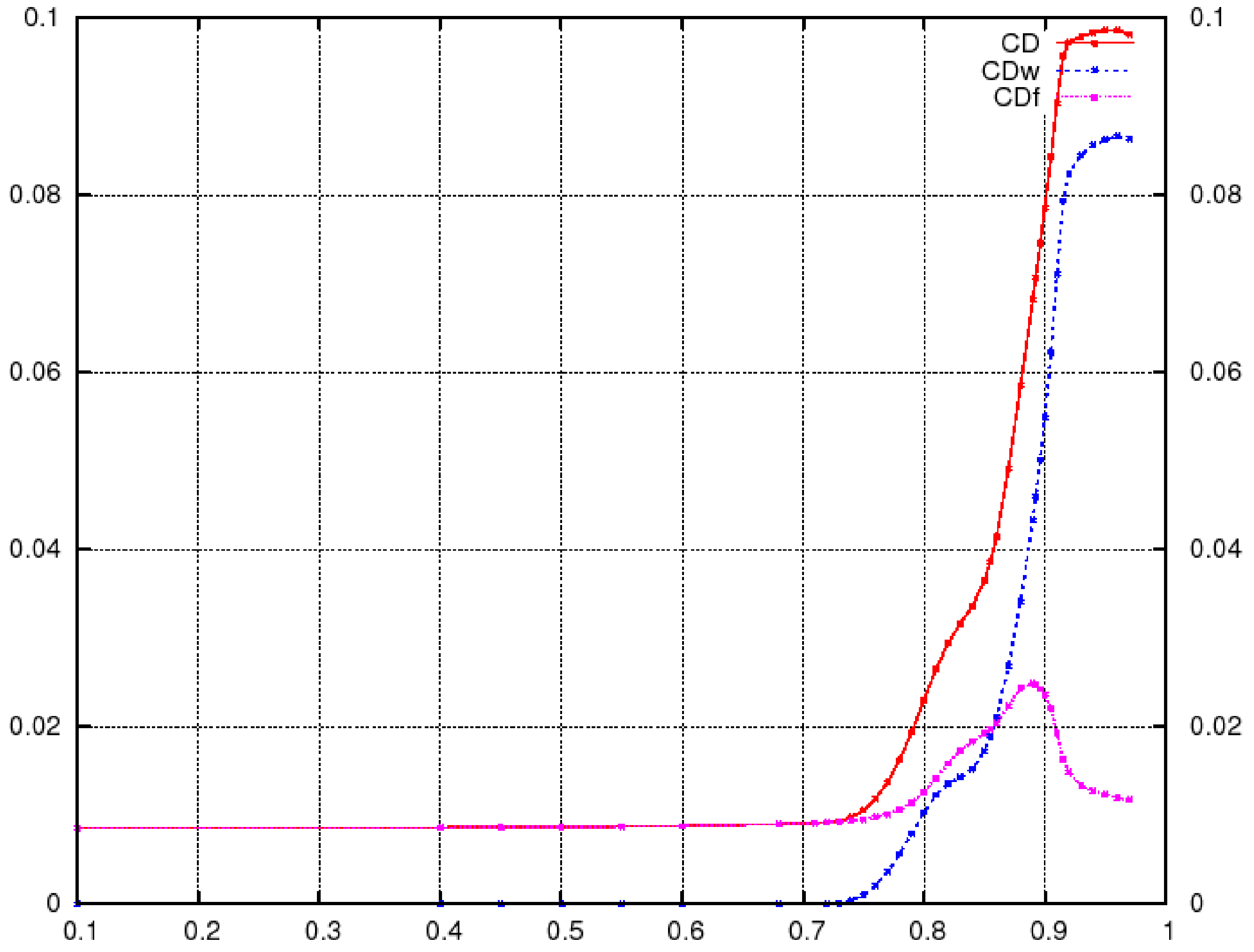

Contours of Cp
Increment = 0.04
1.00
0.75
0.50
0.25
0.00
-0.25
-0.50
-0.75
-1.00


Contours of Mach
Increment = 0.02
1.440
1.385
1.330
1.275
1.220
1.165
1.110
1.055
1.000

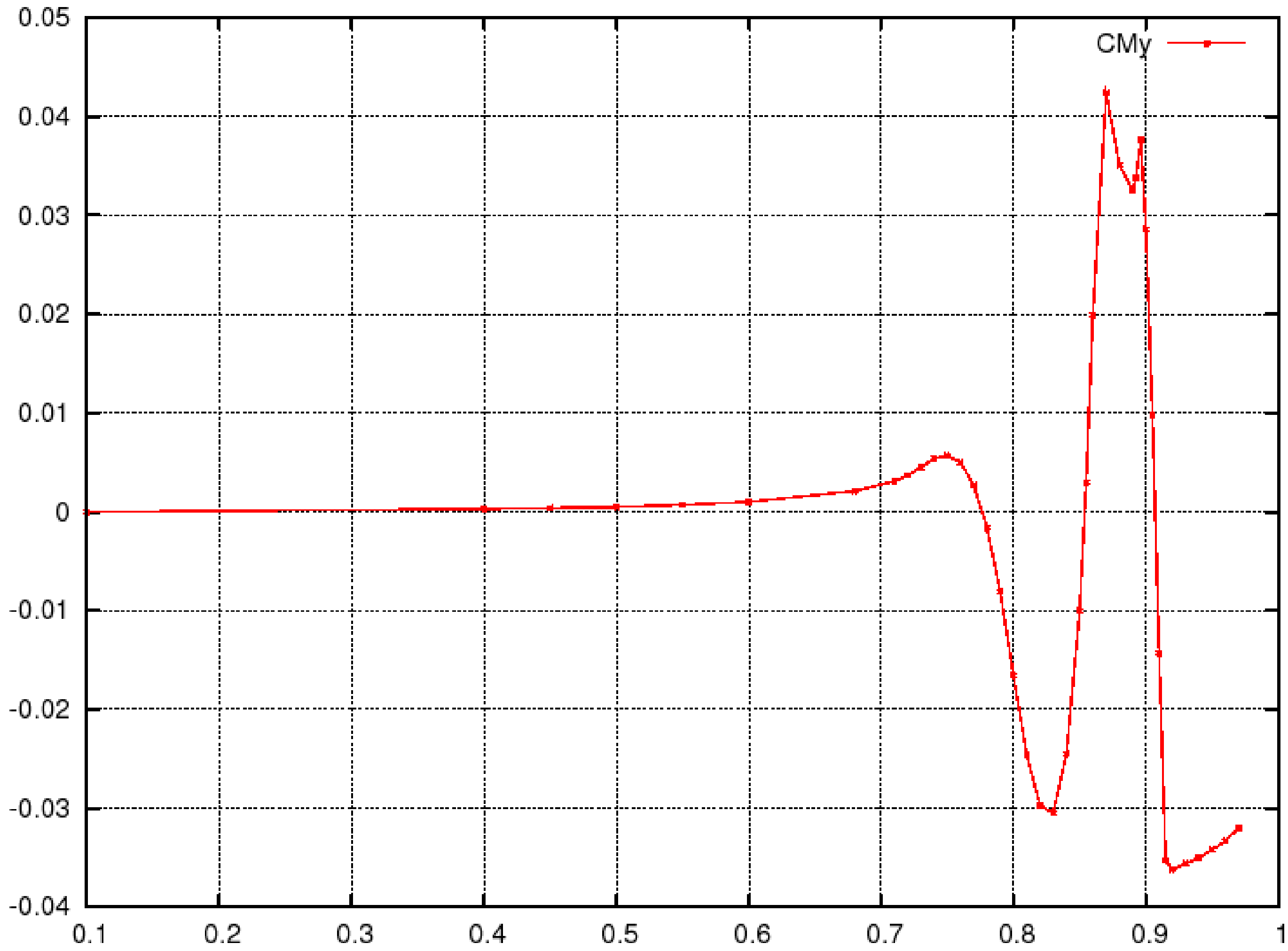

Contours of ΔCp0
Increment = 0.005
-0.005
-0.017
-0.029
-0.041
-0.053
-0.064
-0.076
-0.088
-0.100



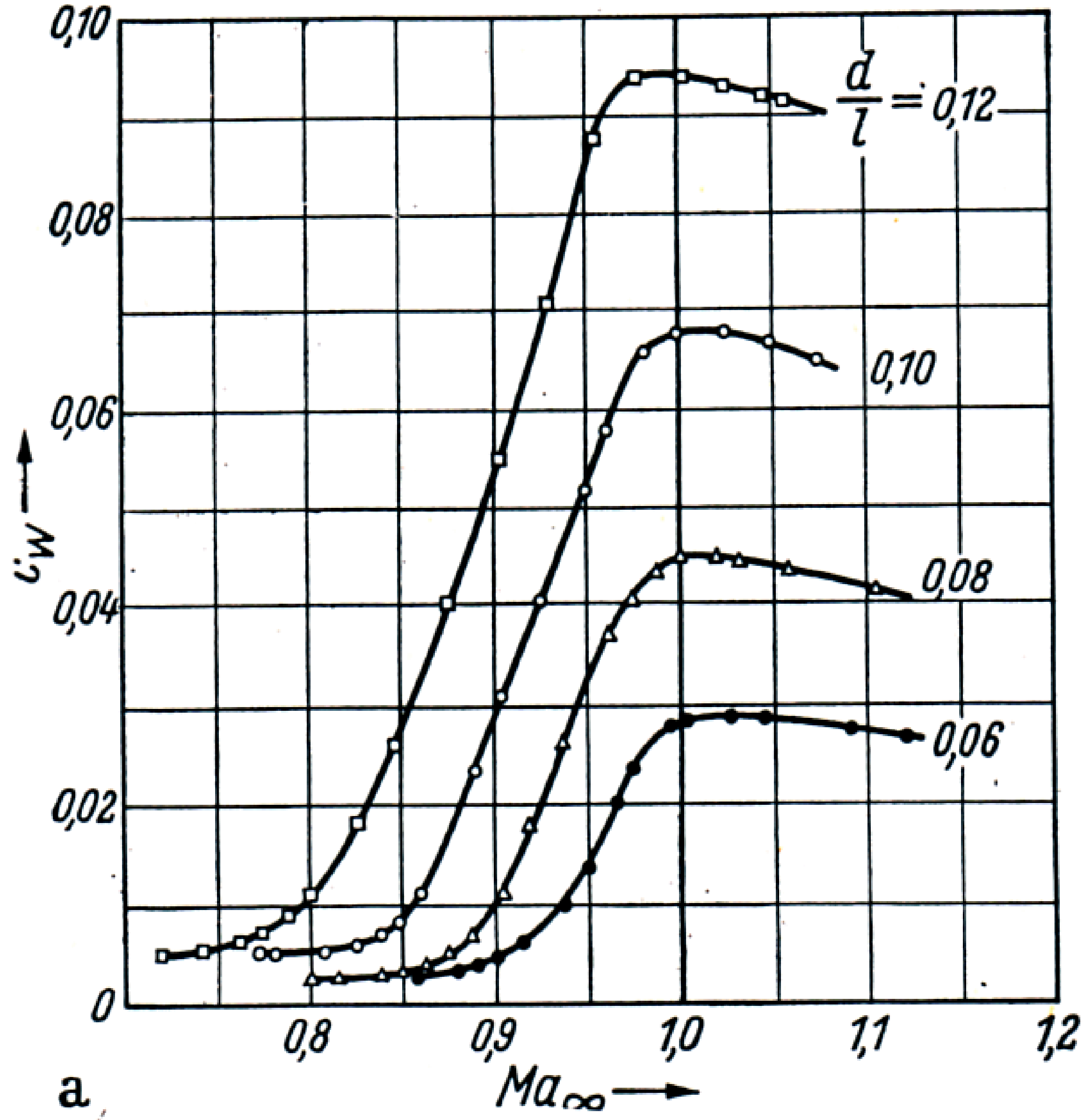
CL
CD
0.5
0.4
0.3
0.2
0.1
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1
0.2
0.18
0.16
0.14
0.12
0.1
0.08
0.06
0.04
0.02
0



0.1
0.08
0.06
0.04
0.02
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1
CD
CDw
CDf

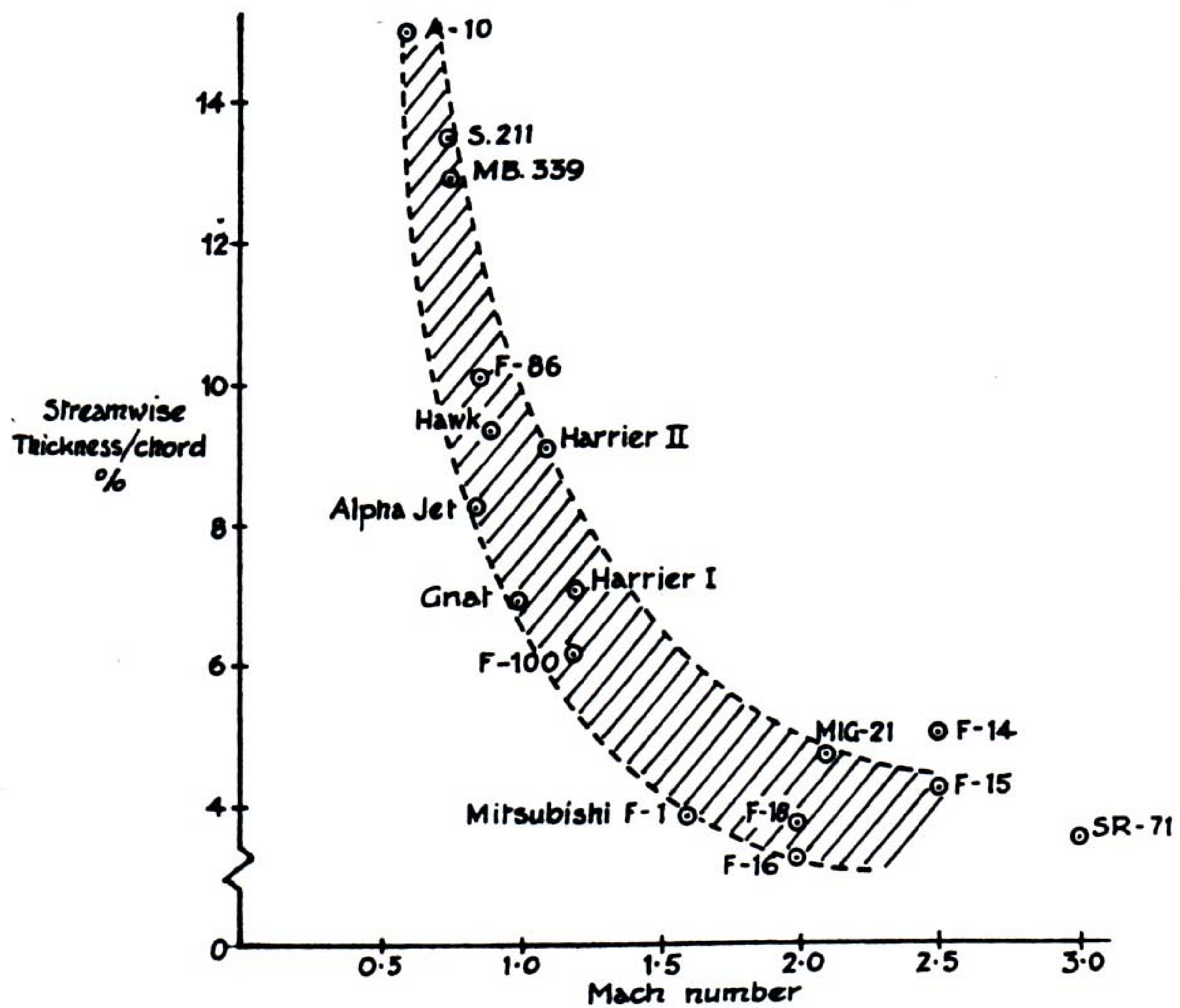


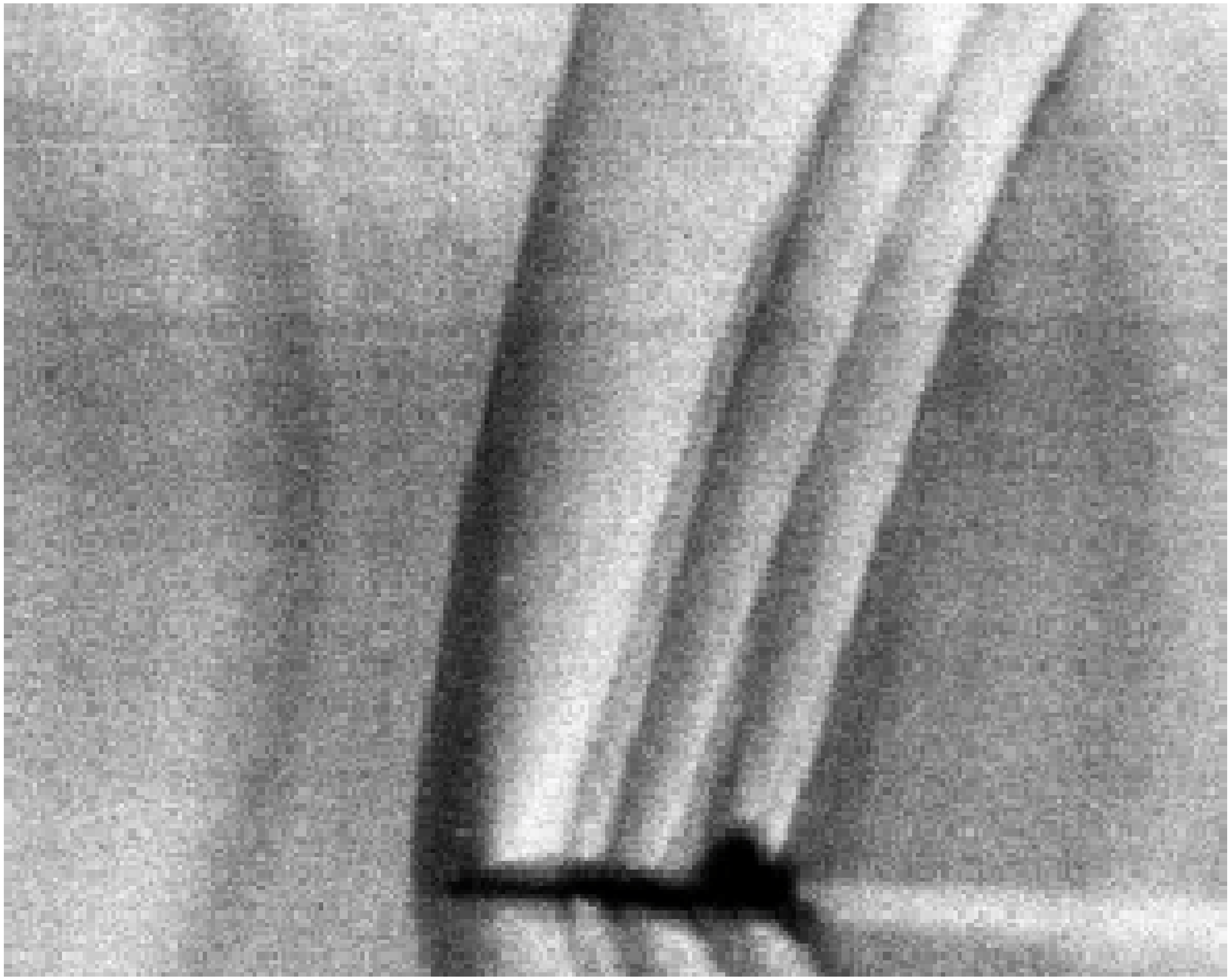
0.05
0.04
0.03
0.02
0.01
0
-0.01
-0.02
-0.03
-0.04
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1
CMy



0,10
0,08
0,06
0,04
0,02
0
c_W
0,8
0,9
1,0
1,1
1,2
Ma_∞
$\frac{d}{l}$ = 0,12
0,10
0,08
0,06

a


A-10
S.211
MB.339
F-86
Hawk
Harrier II
Alpha Jet
Harrier I
Gnat
F-100
MIG-21
F-14
F-15
Mitsubishi F-1
F-18
F-16
SR-71
Streamwise Thickness/chord %
14
12
10
8
6
4
0
0·5
1·0
1·5
2·0
2·5
3·0
Mach number



Schlieren photograph of shockwaves produced by a T-38 while flying Mach 1.1 at 13,000 feet.

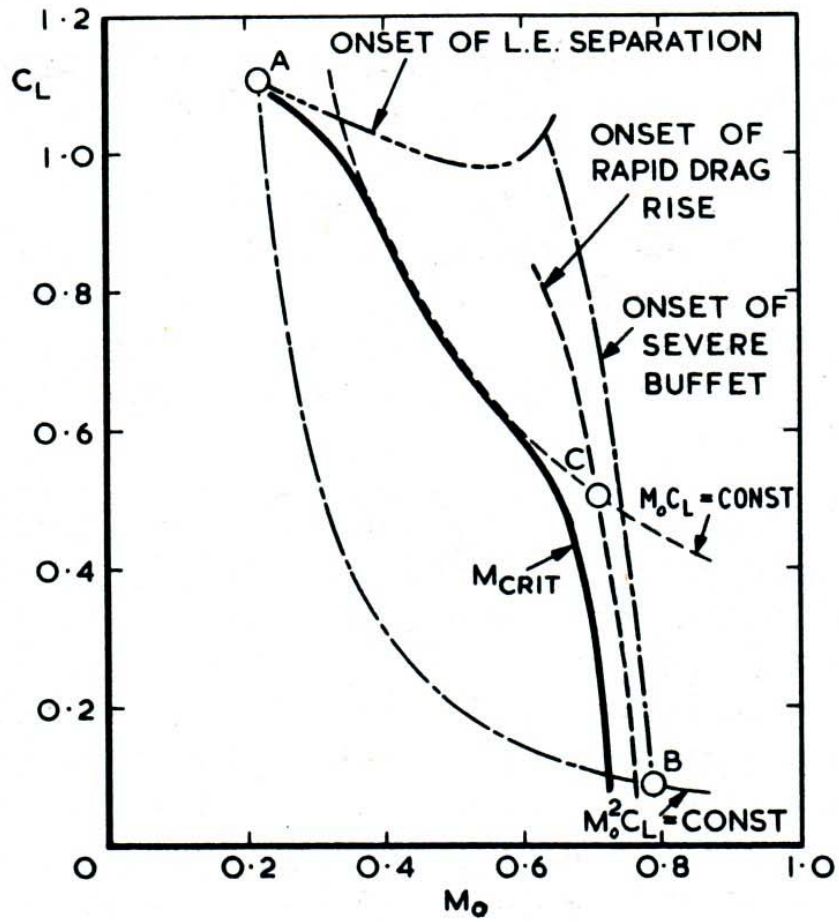


Fig. 4.59 Some significant flow boundaries on a twodimensional aerofoil. After Pearcey & Osborne (1970)

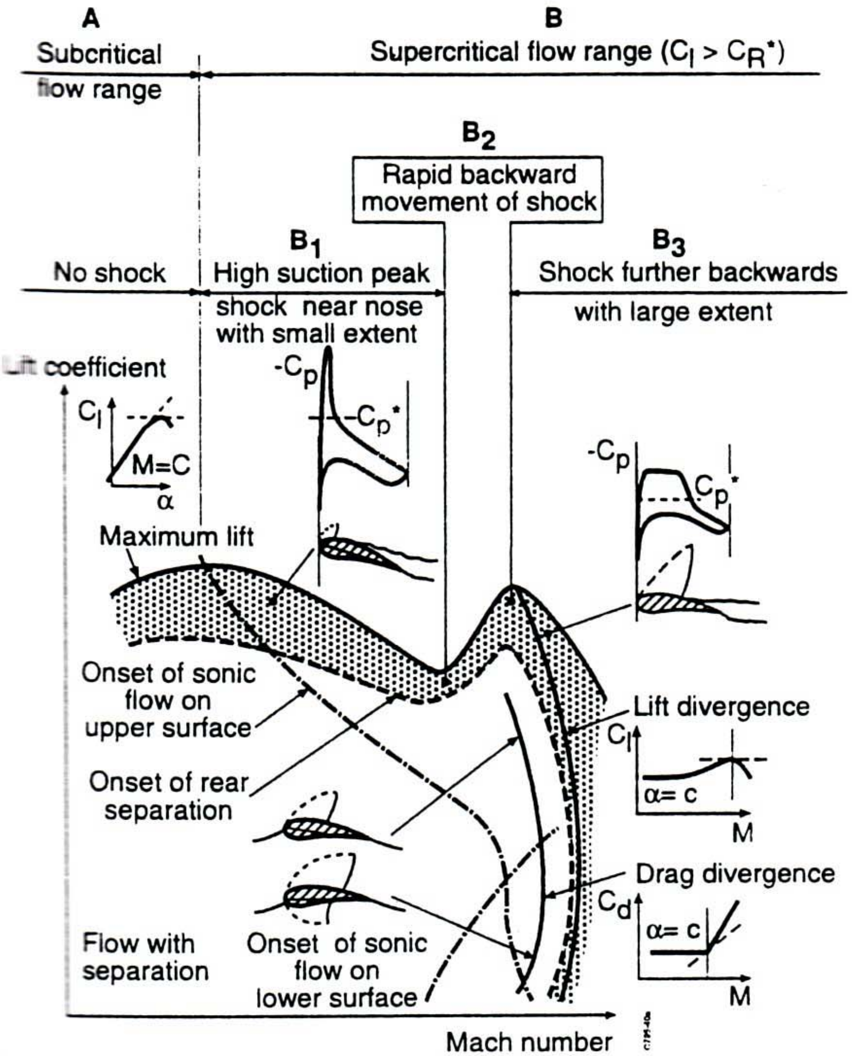
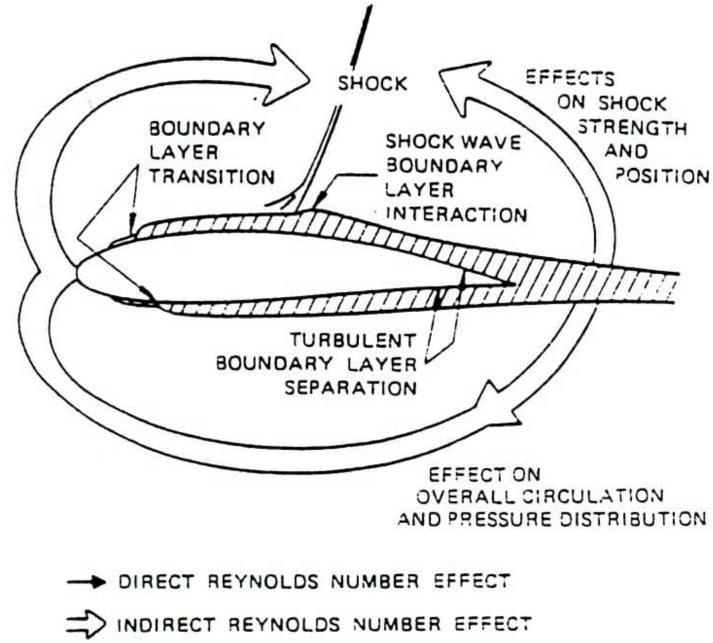
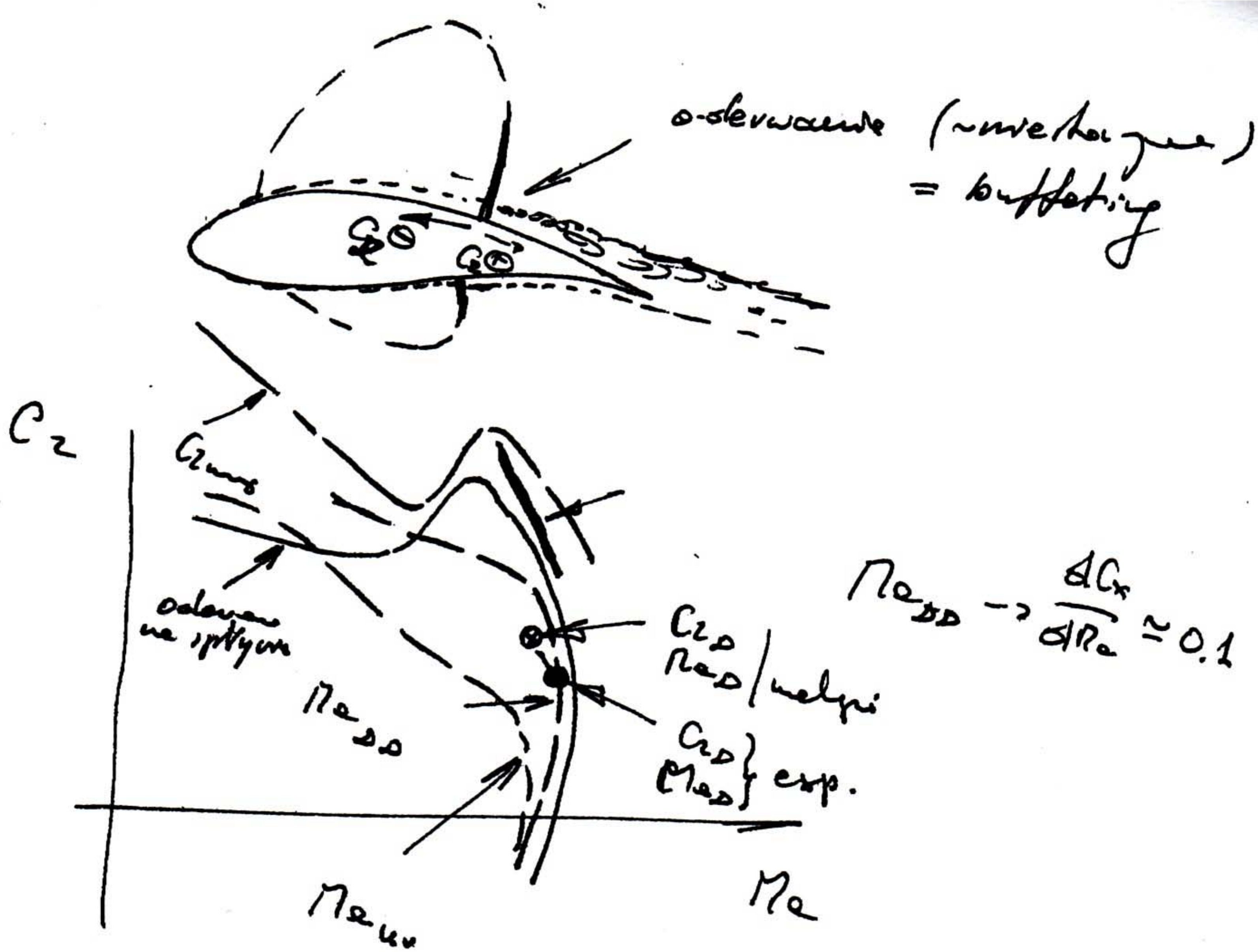


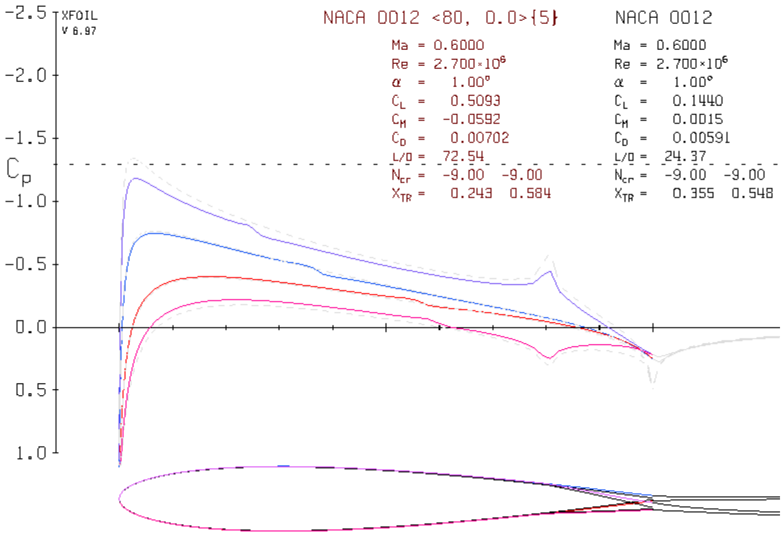
Figure 11: Airfoil characteristics in the C_L-Mach number plane.



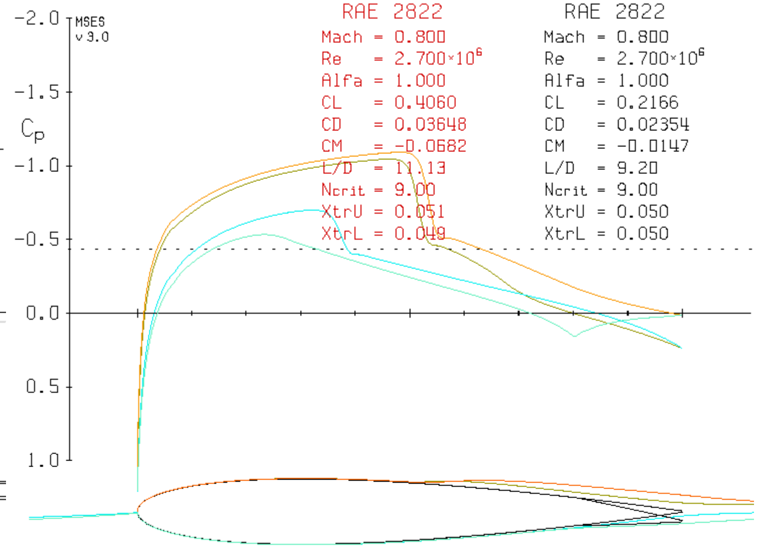
DOMINANT RE-NUMBER EFFECT		
CHARACTERISTIC	DIRECT	INDIRECT
LIFT AND PITCHING MOMENT		X
VISCOUS DRAG	X	
WAVE DRAG		X
DRAG DIVERGENCE		X
BOUNDARY LAYER SEPARATION	X	
BUFFET BOUNDARY	X	X

Figure 12: Schematic representation of direct and indirect Reynolds number effects (Elsenaar, 1988)

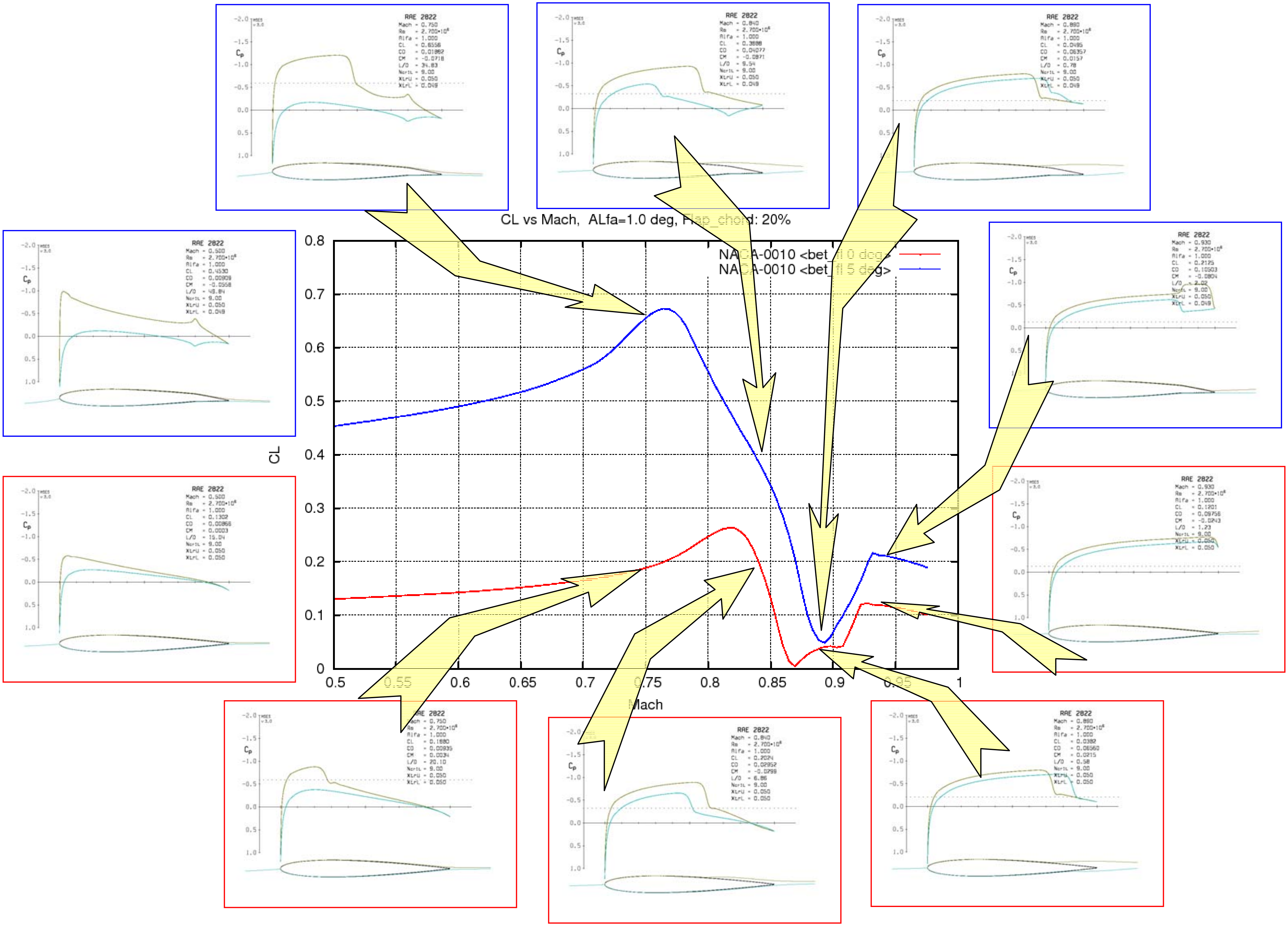

= buffeting
Cz
Ma
Ma_kr
Ma_DD
$Ma_{DD} \rightarrow \frac{dC_x}{dMa} \simeq 0.1$
exp.



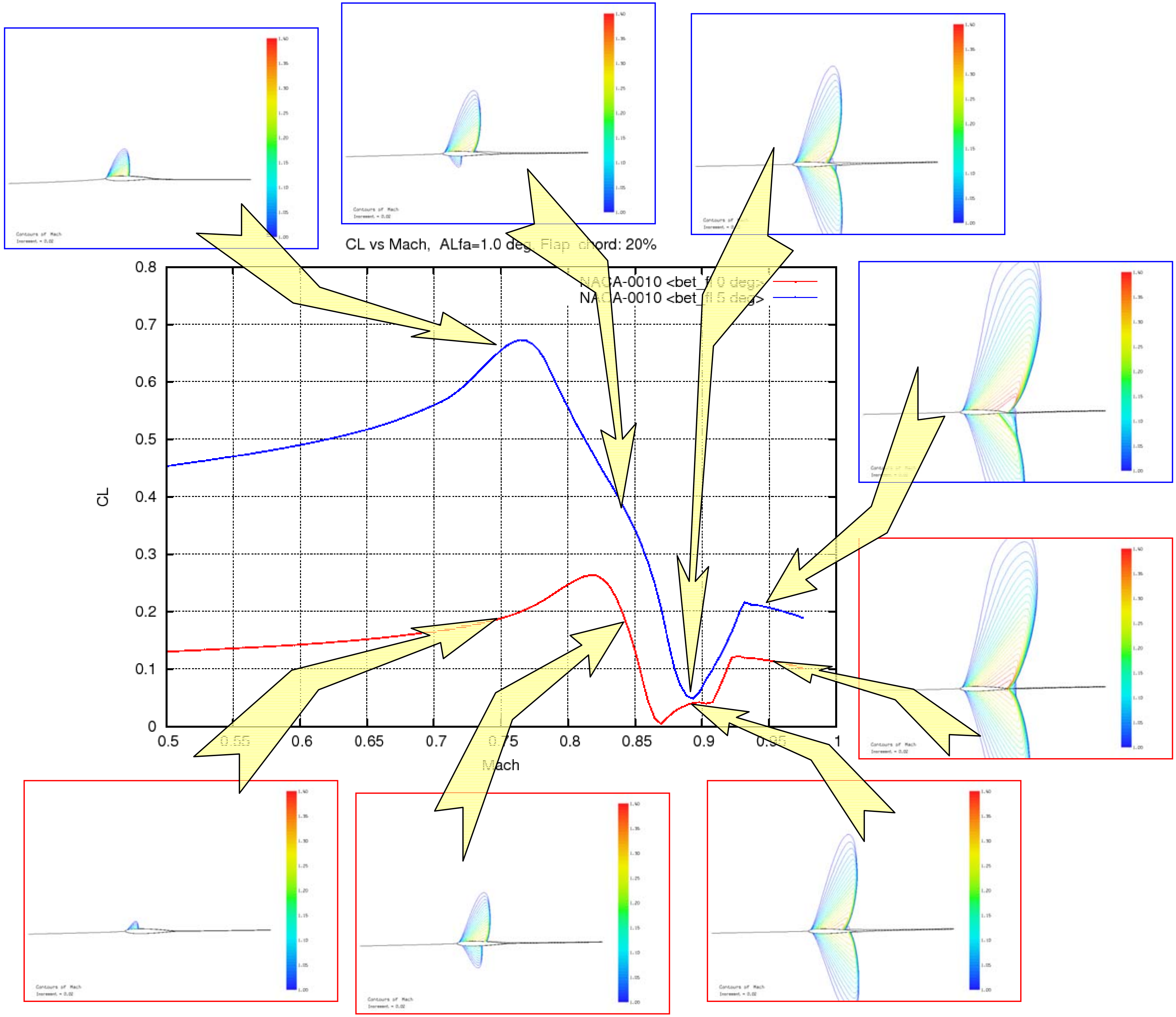
XFOIL
V 6.97
NACA 0012 <80, 0.0>{5}
Ma = 0.6000
Re = 2.700×10^6
α = 1.00°
CL = 0.5093
CM = -0.0592
CD = 0.00702
L/D = 72.54
Ncr = -9.00 -9.00
XTR = 0.243 0.584
NACA 0012
Ma = 0.6000
Re = 2.700×10^6
α = 1.00°
CL = 0.1440
CM = 0.0015
CD = 0.00591
L/D = 24.37
Ncr = -9.00 -9.00
XTR = 0.355 0.548
Cp
-2.5
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0

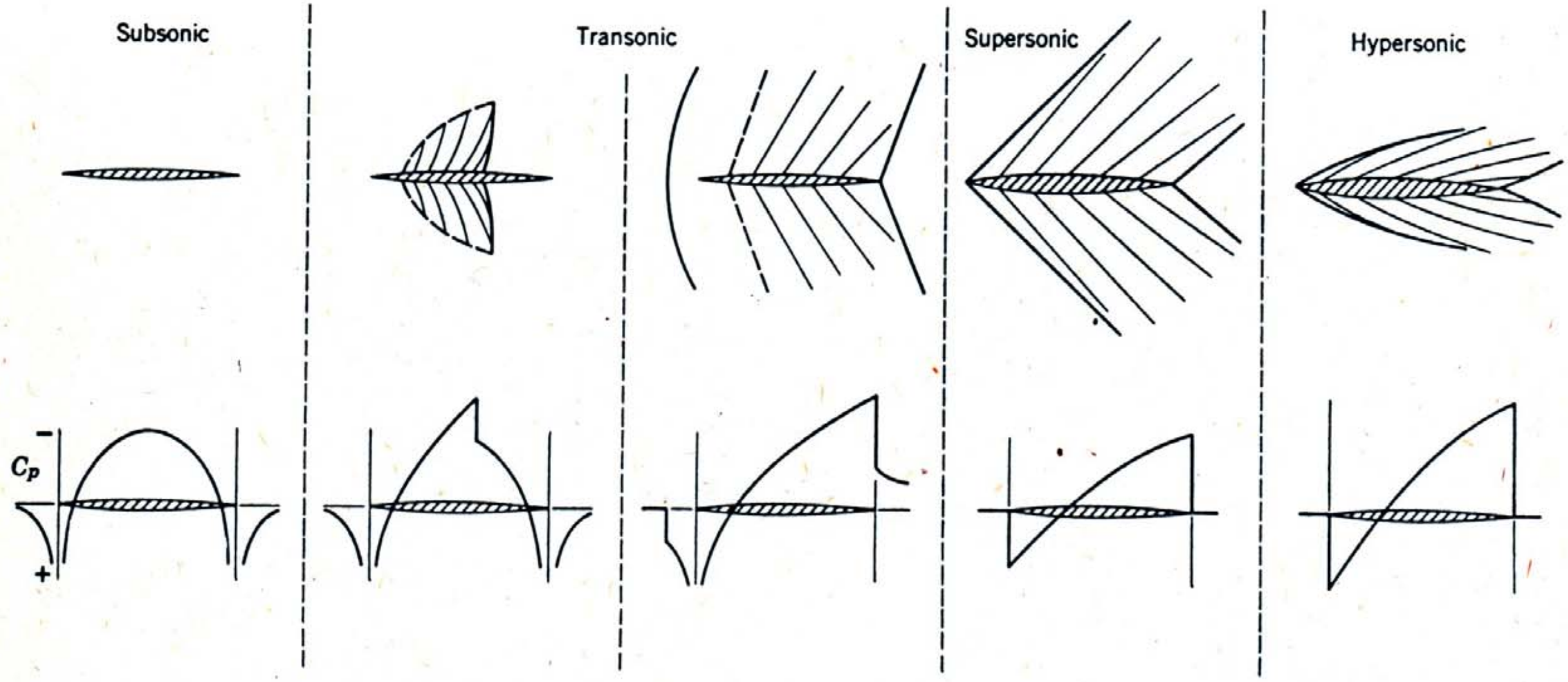


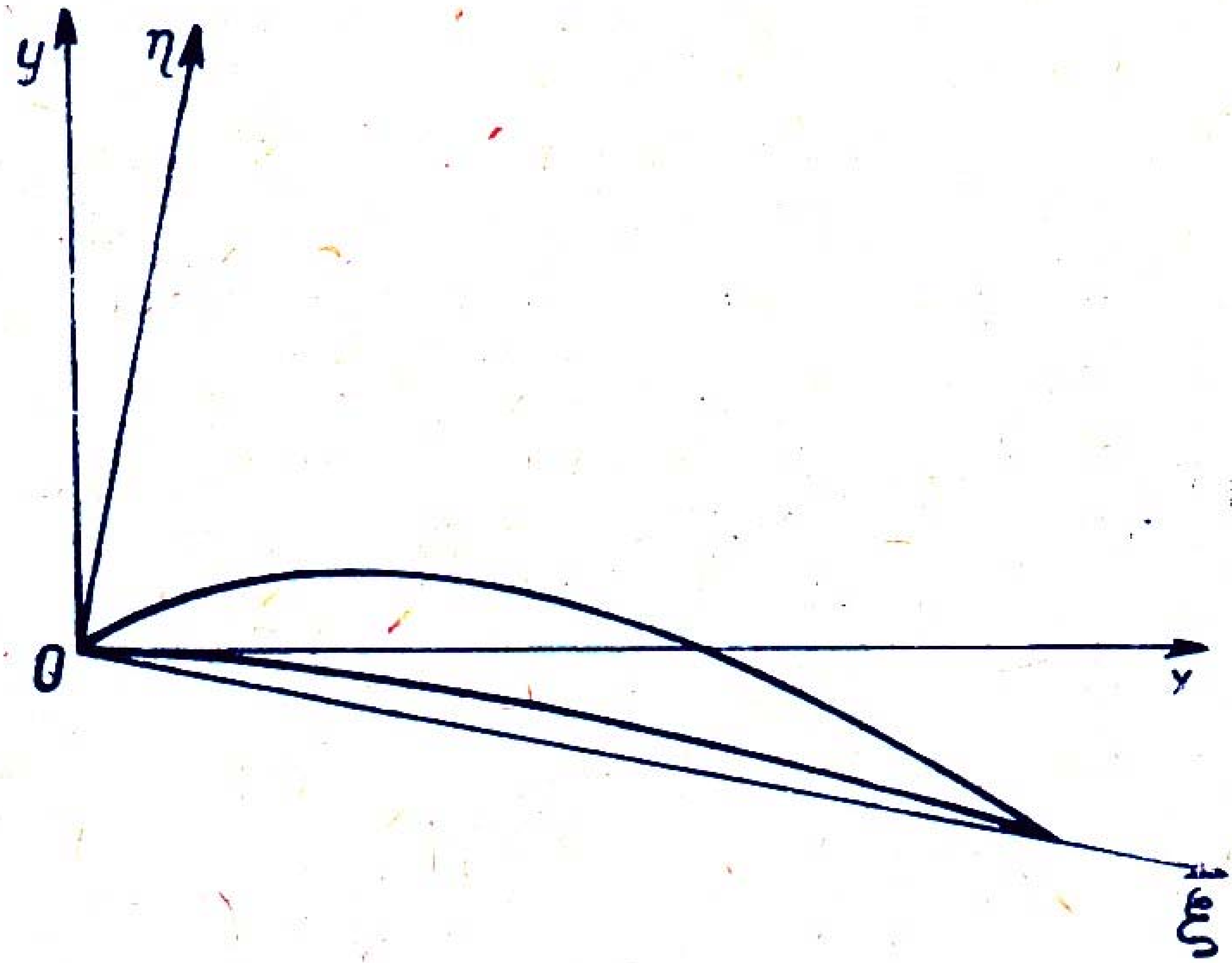
MSES
v 3.0
RAE 2822
Mach = 0.800
Re = 2.700×10^6
Alfa = 1.000
CL = 0.4060
CD = 0.03648
CM = -0.0682
L/D = 11.13
Ncrit = 9.00
XtrU = 0.051
XtrL = 0.049
RAE 2822
Mach = 0.800
Re = 2.700×10^6
Alfa = 1.000
CL = 0.2166
CD = 0.02354
CM = -0.0147
L/D = 9.20
Ncrit = 9.00
XtrU = 0.050
XtrL = 0.050
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0



CL vs Mach, ALfa=1.0 deg, Flap_chord: 20%
NACA-0010 <bet_fl 0 deg>
NACA-0010 <bet_fl 5 deg>
CL
Mach
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
0.5
0.55
0.6
0.65
0.7
0.75
0.8
0.85
0.9
0.95
1
RAE 2822

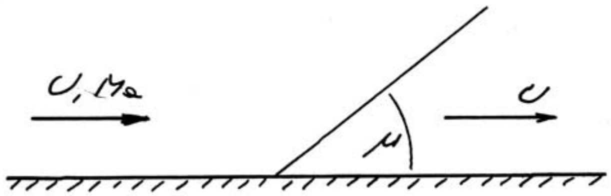

CL vs Mach, ALfa=1.0 deg, Flap chord: 20%
NACA-0010 <bet_fl 0 deg>
NACA-0010 <bet_fl 5 deg>
CL
Mach
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.5
0.55
0.6
0.65
0.7
0.75
0.8
0.85
0.9
0.95
1


Subsonic
Transonic
Supersonic
Hypersonic
C_p
−
+

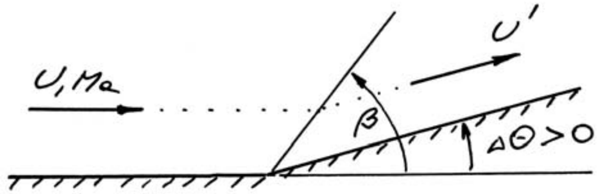


y
η
O
x
ξ

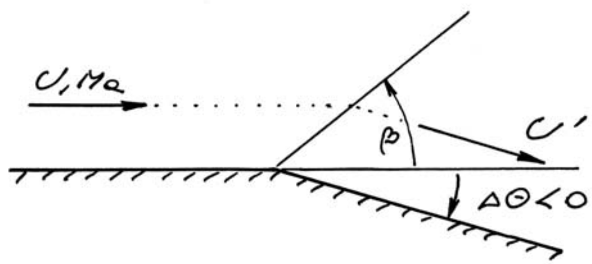
SUPERSONIC FLOW- SMALL DISTURBANCES



$$\mu = \arcsin\left(\frac{1}{Ma}\right) \qquad \Delta S = 0 \quad \beta = \mu$$



$$\Delta S > 0 \quad \beta > \mu$$



$$? \qquad \Delta S < 0 \quad \beta\,?$$

$$V_s \equiv V_s' = V\cos(\beta) = V'\cos(\beta-\theta)$$

$$V' = V + \Delta V$$

$$V\cos(\beta) = (V+\Delta V)\left(\cos(\beta)\underbrace{\cos(\Delta\theta)}_{\sim 1} + \sin(\beta)\underbrace{\sin(\Delta\theta)}_{\sim\Delta\theta}\right)$$

$$\Delta V\cos(\beta) + V\sin(\beta)\Delta\theta \simeq 0$$

$$\frac{\Delta V}{V} = -\tan(\beta)\cdot\Delta\theta$$

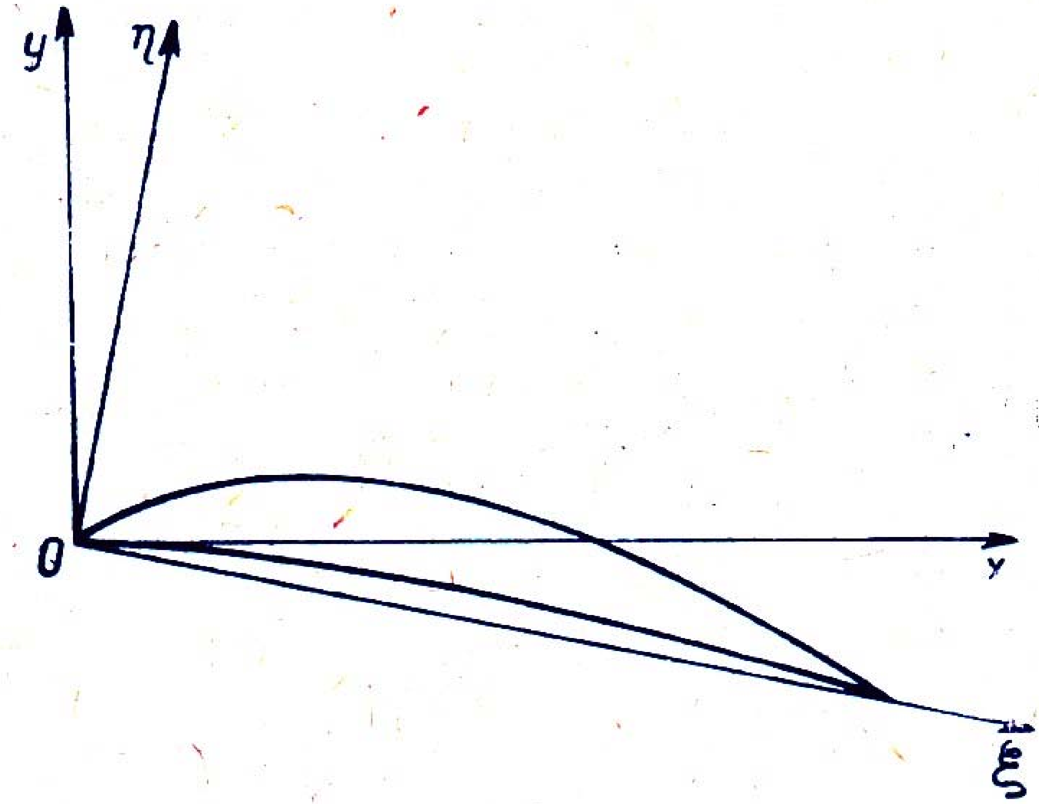
$$|\Delta\theta| \ll 1 \quad \rightarrow \quad \Delta S = 0 \quad \rightarrow \quad \beta = \mu = \arcsin\left(\frac{1}{Ma}\right)$$

$$\sin\mu = \frac{1}{Ma_\infty}$$

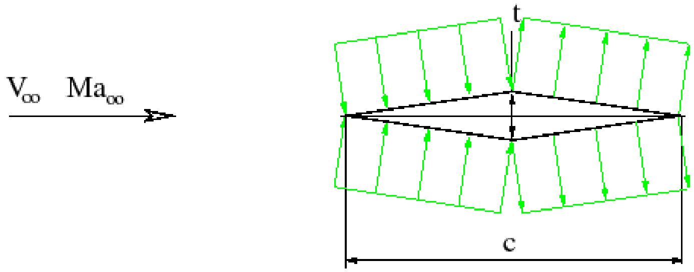
$$\cos\mu = \frac{\sqrt{Ma_\infty^2 - 1}}{Ma_\infty}$$

$$\tan\mu = \frac{1}{\sqrt{Ma_\infty^2 - 1}}$$

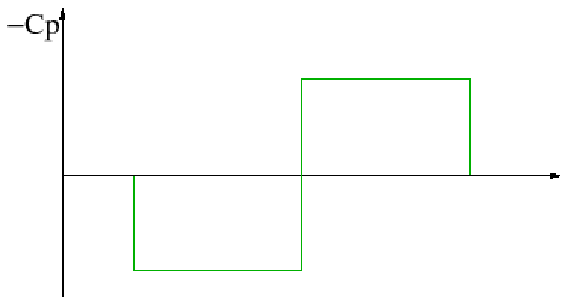
$$Cp = -2\frac{u}{U_\infty} \simeq -2\frac{\Delta V}{V} = 2\tan(\mu)\cdot\Delta\theta = \frac{2\cdot\Delta\theta}{\sqrt{Ma_\infty^2 - 1}}$$



$$Cp^{U/L} = \pm \frac{2y'}{\sqrt{Ma_{\infty}^{2} - 1}}$$



$$y' = \pm t / c = \pm \overline{t}$$

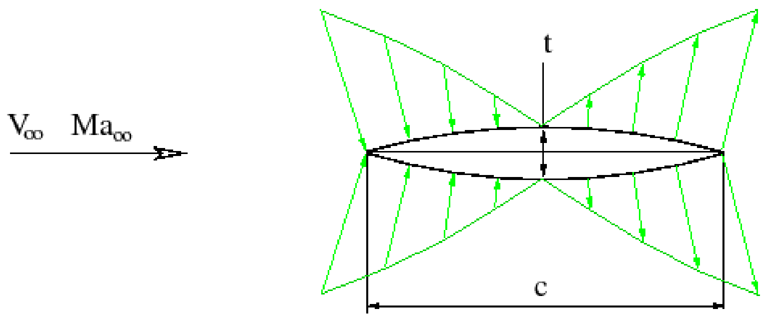


$$Cp^{U/L} = \pm \frac{2 \cdot \overline{t}}{\sqrt{Ma_\infty^2 - 1}}$$

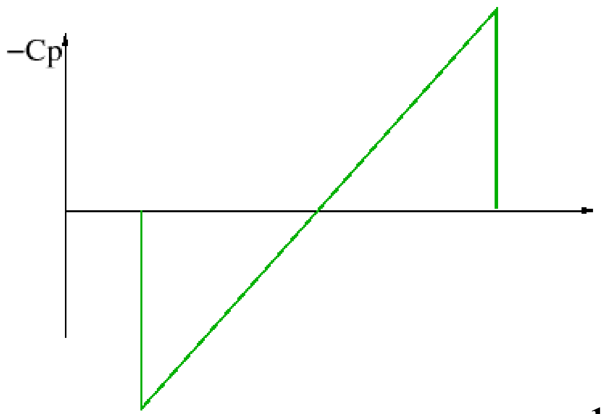
$$C_L = \frac{1}{c}\int_0^c \left(Cp^L - Cp^U\right) dx = \int_0^1 \left(Cp^L - Cp^U\right) d\left(x/c\right) = 0$$

$$C_D = \frac{1}{c}\int_0^c \left(Cp^U \left.\frac{dy}{dx}\right|_U - Cp^L \left.\frac{dy}{dx}\right|_L\right) dx = \int_0^1 \left(Cp^U \left.\frac{dy}{dx}\right|_U - Cp^L \left.\frac{dy}{dx}\right|_L\right) d\left(x/c\right) =$$

$$= \ldots = \frac{4 \cdot \overline{t}^2}{\sqrt{Ma_\infty^2 - 1}}$$



$$y' = \pm 2 \cdot \overline{t} \cdot \left(1 - 2\overline{x}\right)$$

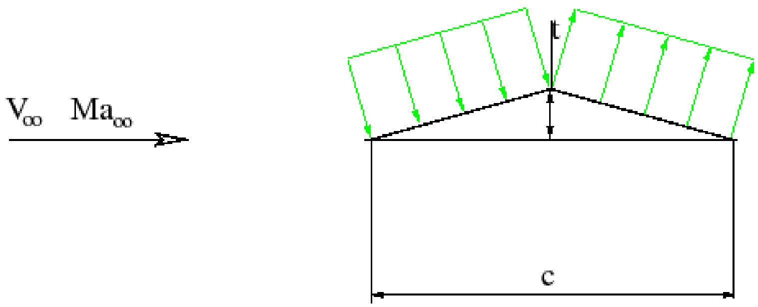


$$Cp^{U/L} = \pm \frac{4 \cdot \overline{t}\left(1 - 2\overline{x}\right)}{\sqrt{Ma_\infty^2 - 1}}$$

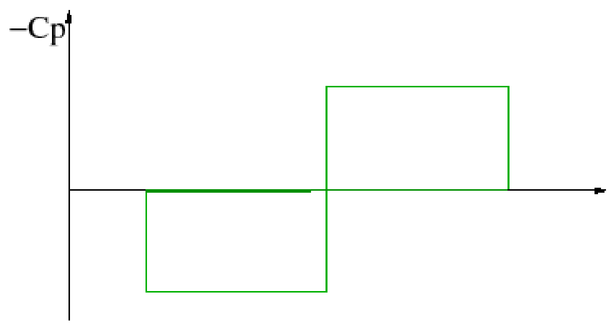
$$C_L = \frac{1}{c}\int_0^c \left(Cp^L - Cp^U\right) dx = \int_0^1 \left(Cp^L - Cp^U\right) d\left(x / c\right) = 0$$

$$C_D = \frac{1}{c}\int_0^c \left(Cp^U \left.\frac{dy}{dx}\right|_U - Cp^L \left.\frac{dy}{dx}\right|_L \right) dx = \int_0^1 \left(Cp^U \left.\frac{dy}{dx}\right|_U - Cp^L \left.\frac{dy}{dx}\right|_L \right) d\left(x / c\right) =$$

$$= \ldots = \frac{5.333 \cdot \overline{t}^{\,2}}{\sqrt{Ma_\infty^2 - 1}}$$



$$y' = \pm 2 \cdot t / c = \pm 2 \cdot \bar{t}$$

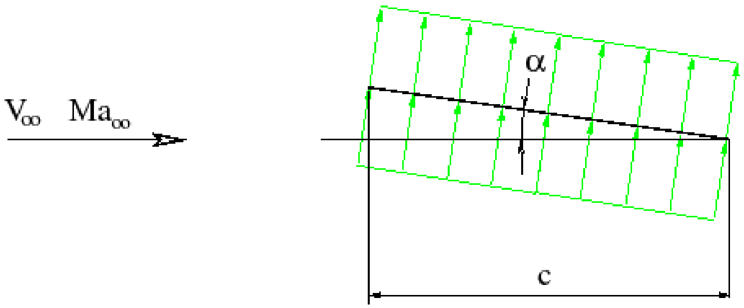


$$Cp^U = \pm \frac{4 \cdot \bar{t}}{\sqrt{Ma_\infty^2 - 1}}$$

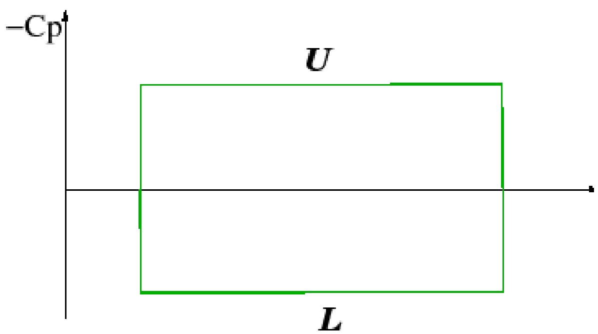
$$C_L = \frac{1}{c}\int_0^c \left(Cp^L - Cp^U\right) dx = \int_0^1 \left(Cp^L - Cp^U\right) d\left(x/c\right) = 0$$

$$C_D = \frac{1}{c}\int_0^c \left(Cp^U \left.\frac{dy}{dx}\right|_U - Cp^L \left.\frac{dy}{dx}\right|_L \right) dx = \int_0^1 \left(Cp^U \left.\frac{dy}{dx}\right|_U - Cp^L \left.\frac{dy}{dx}\right|_L \right) d\left(x/c\right) =$$

$$= \ldots = \frac{8 \cdot \bar{t}^2}{\sqrt{Ma_\infty^2 - 1}}$$



$$y' = \alpha$$

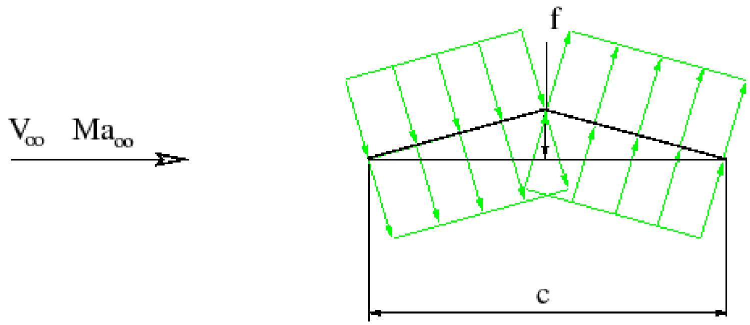


$$Cp^{U/L} = \pm \frac{2 \cdot \alpha}{\sqrt{Ma_\infty^2 - 1}}$$

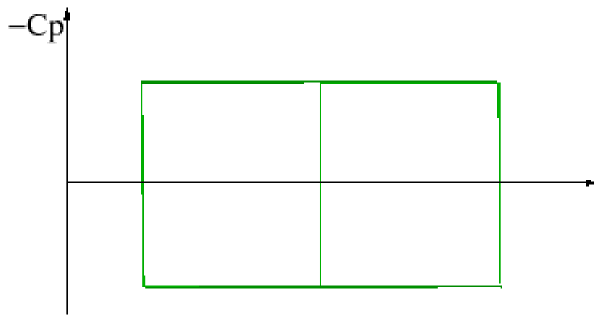
$$C_L = \frac{1}{c}\int_0^c \left(Cp^L - Cp^U\right) dx = \int_0^1 \left(Cp^L - Cp^U\right) d\left(x/c\right) = \frac{4\,\alpha}{\sqrt{Ma_\infty^2 - 1}}$$

$$C_D = \frac{1}{c}\int_0^c \left(Cp^U \left.\frac{dy}{dx}\right|_U - Cp^L \left.\frac{dy}{dx}\right|_L \right) dx = \int_0^1 \left(Cp^U \left.\frac{dy}{dx}\right|_U - Cp^L \left.\frac{dy}{dx}\right|_L \right) d\left(x/c\right) =$$

$$= \ldots = \frac{4\,\alpha^2}{\sqrt{Ma_\infty^2 - 1}} = \frac{\sqrt{Ma_\infty^2 - 1}}{4} \cdot C_L^2$$



$$y' = \pm 2 \cdot f / c = \pm 2 \cdot \bar{f}$$



$$Cp^{U/L} = \pm \frac{4 \cdot \bar{f}}{\sqrt{Ma_\infty^2 - 1}}$$

$$C_L = \frac{1}{c} \int_0^c \left(Cp^L - Cp^U \right) dx = \int_0^1 \left(Cp^L - Cp^U \right) d\left(x / c \right) = 0$$

$$C_D = \frac{1}{c} \int_0^c \left(Cp^U \left. \frac{dy}{dx} \right|_U - Cp^L \left. \frac{dy}{dx} \right|_L \right) dx = \int_0^1 \left(Cp^U \left. \frac{dy}{dx} \right|_U - Cp^L \left. \frac{dy}{dx} \right|_L \right) d\left(x / c \right) =$$

$$= \ldots = \frac{16 \cdot \bar{f}^2}{\sqrt{Ma_\infty^2 - 1}}$$

$$Cp^{U/L} = \pm \frac{2y'}{\sqrt{Ma_{\infty}^{2} - 1}}$$

$$y^{U/L} = \pm y_t + y_c - x \cdot \alpha$$

$$Cp^{U/L} = \frac{2}{\sqrt{Ma_{\infty}^{2} - 1}} \left(y_t' \pm y_c' \mp \alpha \right)$$

$$C_L = \frac{1}{c}\int_0^c \left(Cp^L - Cp^U\right)dx = \frac{2}{\sqrt{Ma_\infty^2 - 1}}\int_0^1 \left(-2y_c' + 2\alpha\right)d\left(x/c\right) =$$

$$= \frac{4\alpha}{\sqrt{Ma_\infty^2 - 1}} - \frac{4}{\sqrt{Ma_\infty^2 - 1}}\int_0^1 y_c' \cdot d\left(x/c\right) \equiv \frac{4\alpha}{\sqrt{Ma_\infty^2 - 1}}$$

$$C_D = \frac{1}{c}\int_0^c Cp^U y'_U \cdot dx - \frac{1}{c}\int_0^c Cp^L y'_L \cdot dx =$$

$$= \frac{2}{\sqrt{Ma_\infty^2 - 1}} \int_0^1 \left(y'_t + \left(y'_c - \alpha\right)\right)\left(y'_t + \left(y'_c - \alpha\right)\right) d\left(x/c\right)$$

$$- \frac{2}{\sqrt{Ma_\infty^2 - 1}} \int_0^1 \left(y'_t - \left(y'_c - \alpha\right)\right)\left(-y'_t + \left(y'_c - \alpha\right)\right) d\left(x/c\right) =$$

$$= \frac{2}{\sqrt{Ma_\infty^2 - 1}} \int_0^1 \left\{\left[y'_t + \left(y'_c - \alpha\right)\right]^2 + \left[y'_t - \left(y'_c - \alpha\right)\right]^2\right\} d\left(x/c\right) =$$

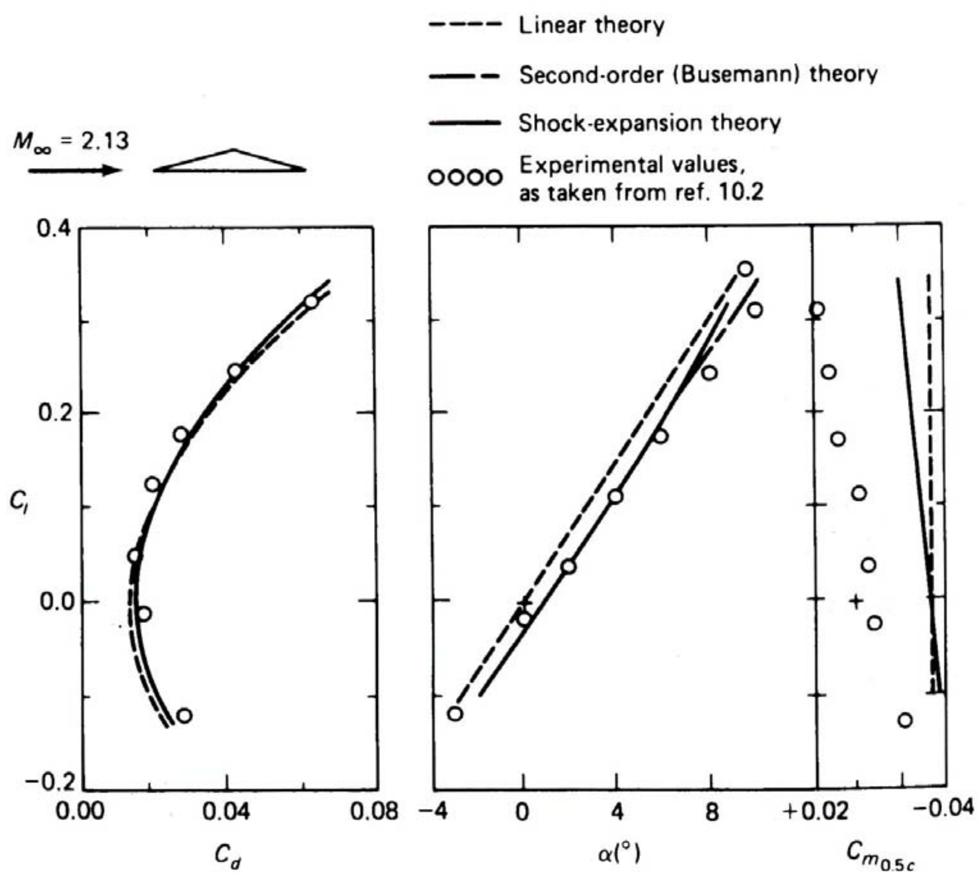
$$C_D = \ldots = \frac{2}{\sqrt{Ma_\infty^2 - 1}} \int_0^1 \left\{ \left[y_t' + \left(y_c' - \alpha \right) \right]^2 + \left[y_t' - \left(y_c' - \alpha \right) \right]^2 \right\} d\left(x/c \right) =$$

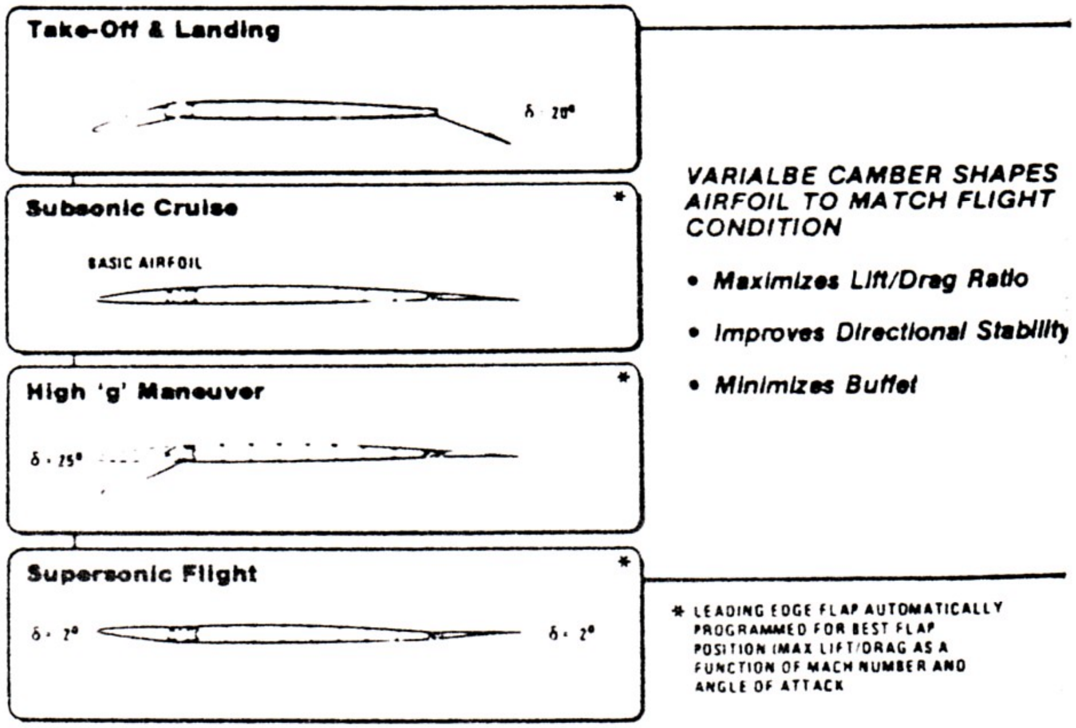
$$= \frac{4}{\sqrt{Ma_\infty^2 - 1}} \int_0^1 \left(y_t' \right)^2 d\left(x/c \right) + \frac{4}{\sqrt{Ma_\infty^2 - 1}} \int_0^1 \left(y_c' - \alpha \right)^2 d\left(x/c \right) =$$

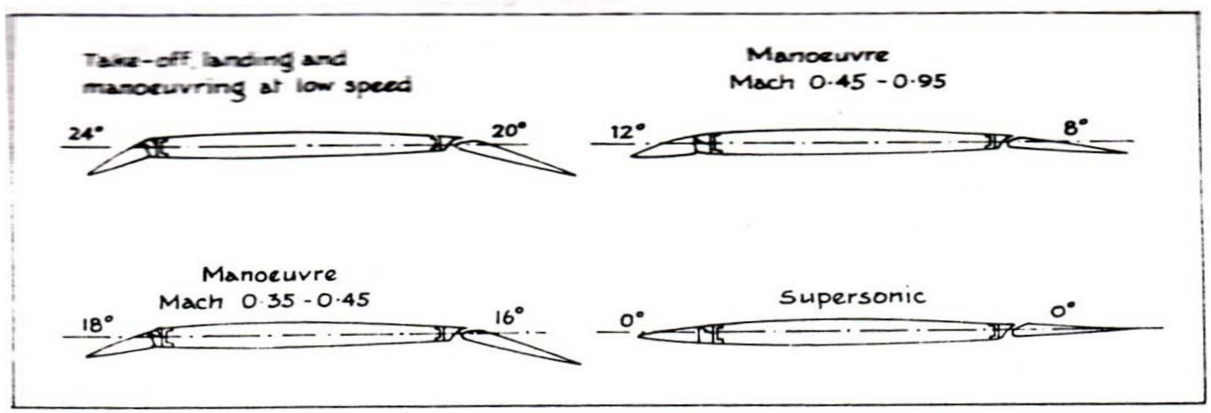
$$= \frac{4}{\sqrt{Ma_\infty^2 - 1}} \int_0^1 \left[\left(y_t' \right)^2 + \left(y_c' \right)^2 \right] d\left(x/c \right) + \frac{4\alpha^2}{\sqrt{Ma_\infty^2 - 1}} =$$

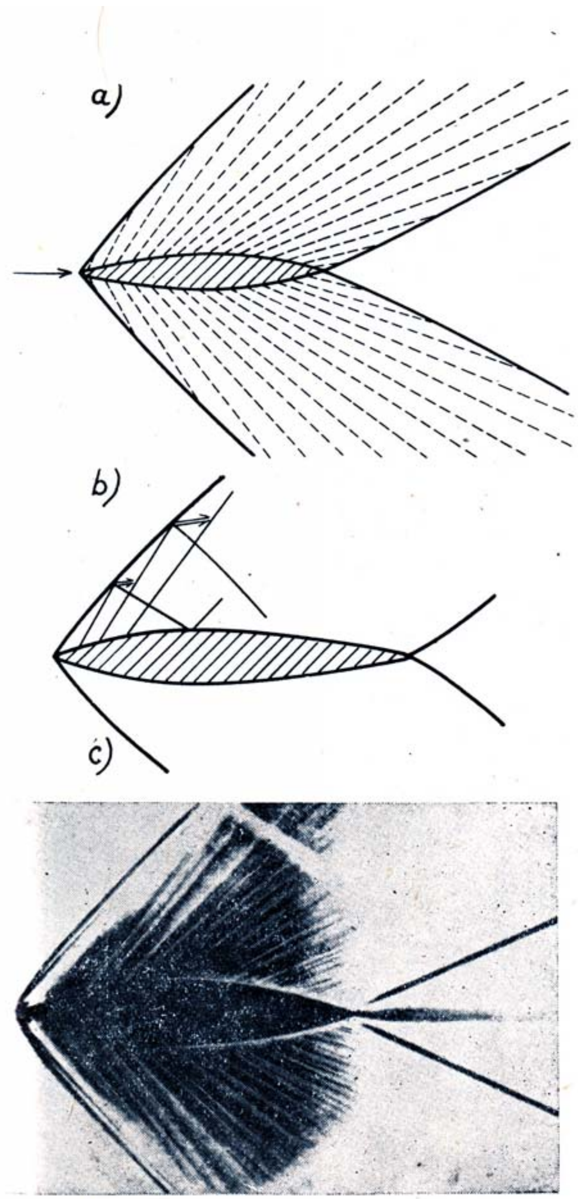
$$= \underbrace{\frac{4}{\sqrt{Ma_\infty^2 - 1}} \int_0^1 \left[\left(y_t' \right)^2 + \left(y_c' \right)^2 \right] d\left(x/c \right)}_{C_{D_{W0}}} + \underbrace{\frac{\sqrt{Ma_\infty^2 - 1}}{4} C_L^2}_{C_{D_{WL}}}$$

$$Cp = \frac{2}{\sqrt{Ma_{\infty}^2 - 1}} y' + \left[\frac{(k+1) Ma_{\infty}^4 - 4(Ma_{\infty}^2 - 1)}{2(Ma_{\infty}^2 - 1)^2} \right] (y')^2$$

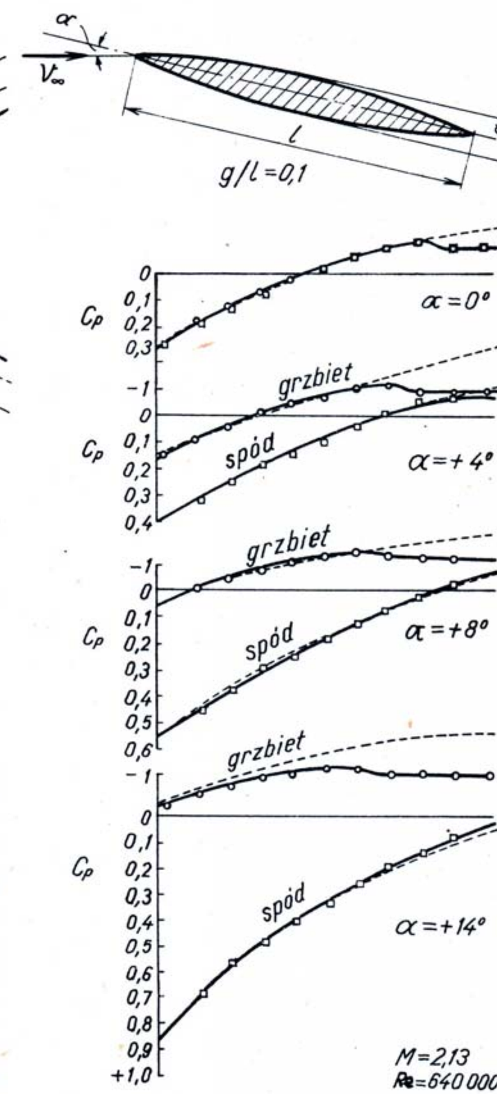



Take-Off & Landing
δ = 20°
Subsonic Cruise
*
BASIC AIRFOIL
High 'g' Maneuver
*
δ = 25°
Supersonic Flight
*
δ = 2°
δ = 2°
VARIALBE CAMBER SHAPES AIRFOIL TO MATCH FLIGHT CONDITION
• Maximizes Lift/Drag Ratio
• Improves Directional Stability
• Minimizes Buffet
* LEADING EDGE FLAP AUTOMATICALLY PROGRAMMED FOR BEST FLAP POSITION (MAX LIFT/DRAG AS A FUNCTION OF MACH NUMBER AND ANGLE OF ATTACK

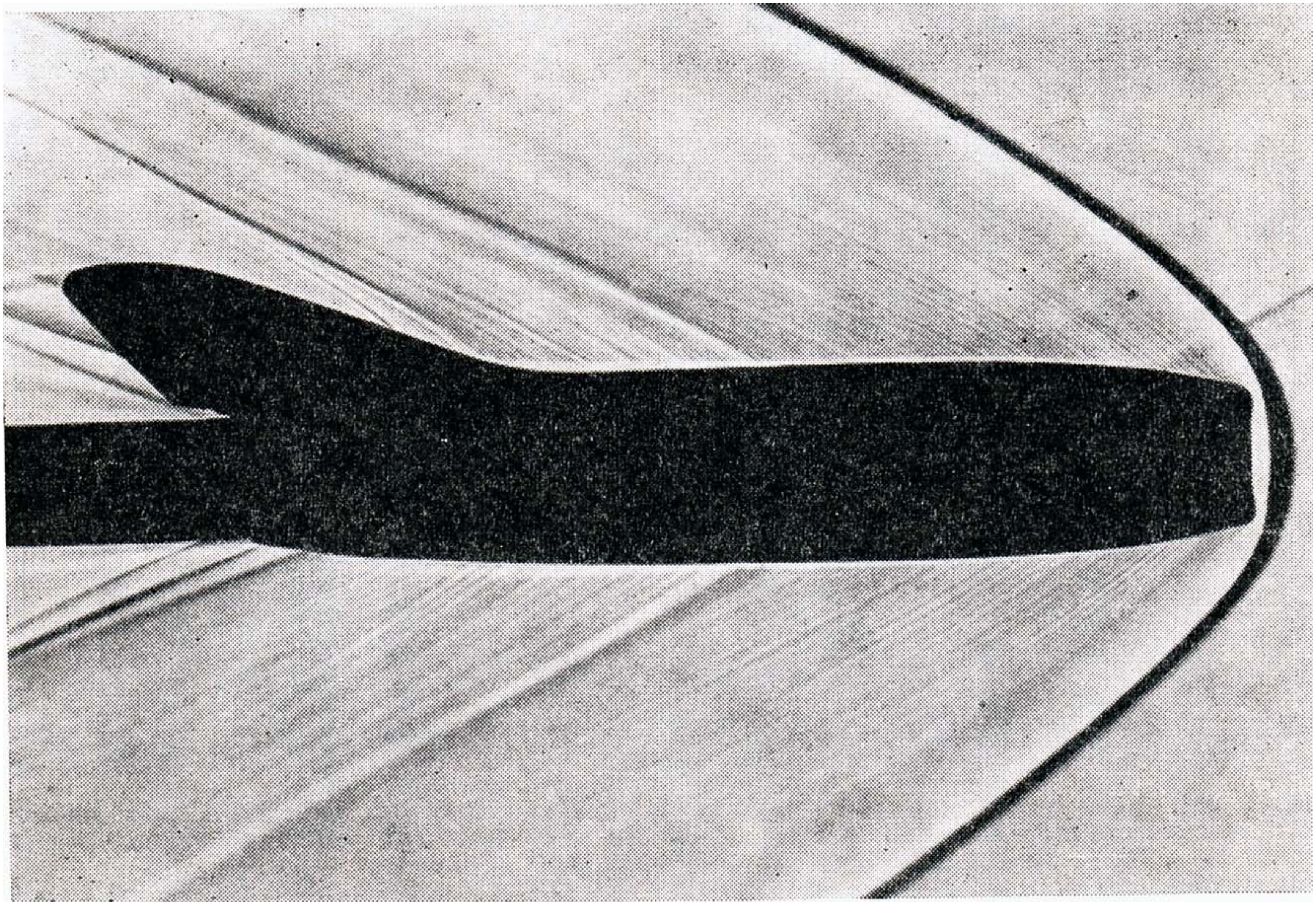

Take-off, landing and manoeuvring at low speed
24°
20°
Manoeuvre
Mach 0·45 - 0·95
12°
8°
Manoeuvre
Mach 0·35 - 0·45
18°
16°
Supersonic
0°
0°

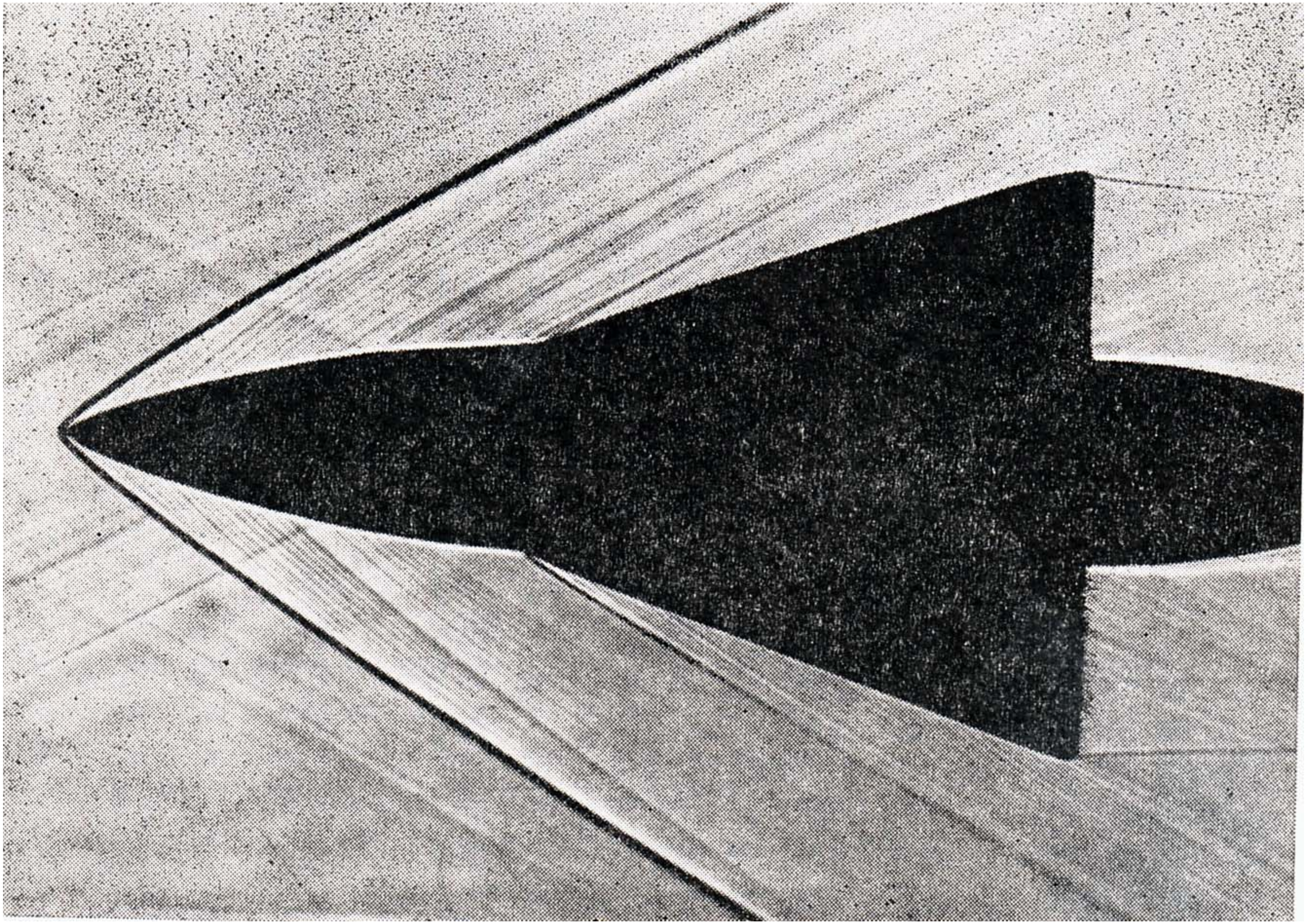


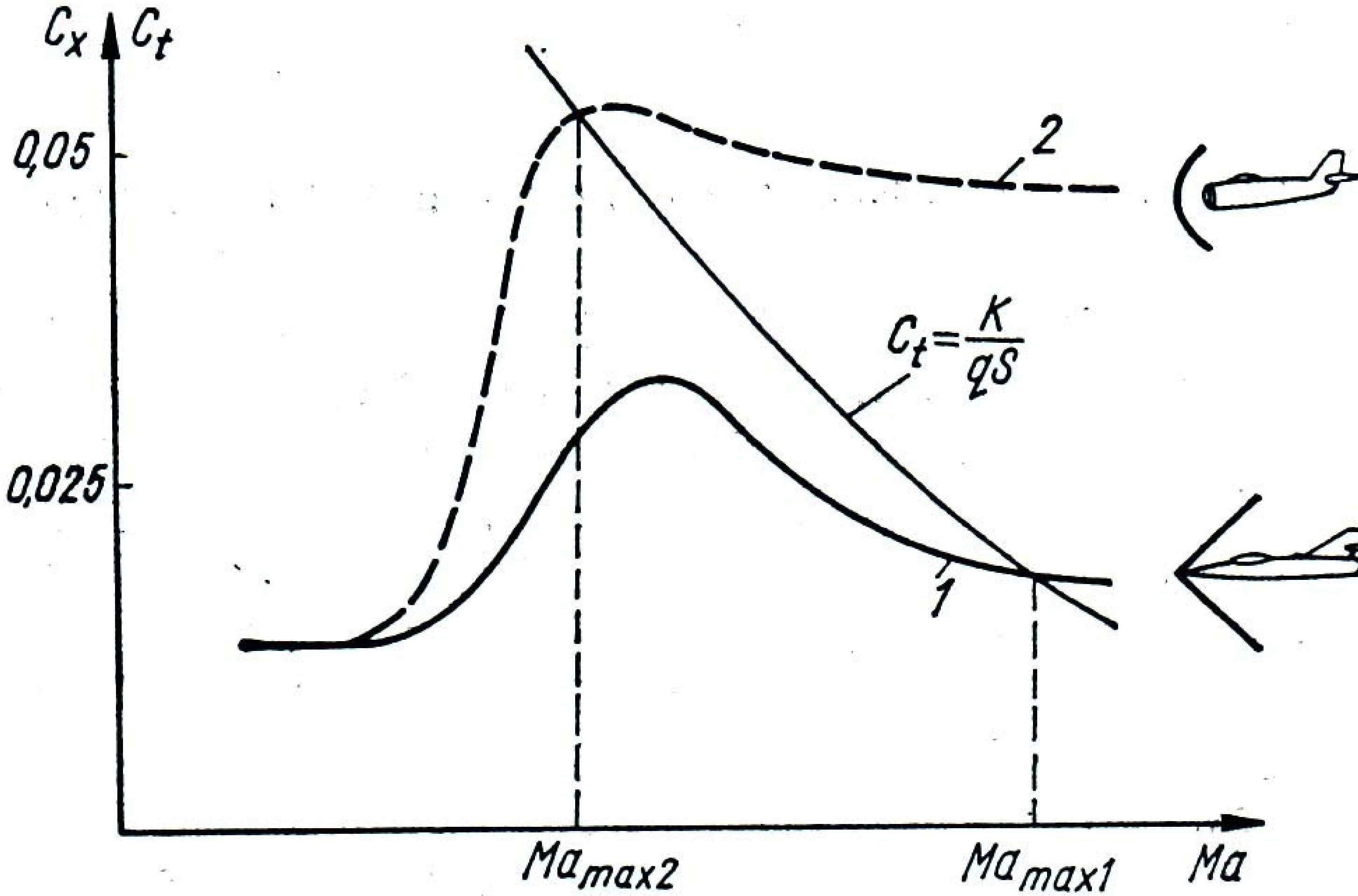
Rys.VI-71



Rys.VI-72







c_x c_t
0,05
0,025
2
1
$c_t = \frac{K}{qS}$
Ma_{max2}
Ma_{max1}
Ma

$$L = \frac{\rho_\infty V_\infty^2}{2} \cdot S \cdot C_L = k\,p_\infty \frac{\rho_\infty}{k\,p_\infty} \frac{V_\infty^2}{2} \cdot S \cdot C_L = \frac{k\,p_\infty}{2} V_\infty^2 / a_\infty^2 \cdot S \cdot C_L = \frac{k\,p_\infty}{2} \cdot Ma_\infty^2 \cdot S \cdot C_L$$

$$\frac{L / S}{p_\infty} = \frac{k}{2} \cdot \left(Ma_\infty^2 \cdot C_L \right)$$

$$Cp = \frac{1}{\frac{k}{2} Ma_\infty^2} \left[\left(\frac{1 + \frac{k-1}{2} Ma_\infty^2}{1 + \frac{k-1}{2} Ma^2} \right)^{\frac{k}{k-1}} - 1 \right]$$

$$Cp_{VAC} = \left| Ma \to \infty \right| = -\frac{2}{k\,Ma_\infty^2}$$

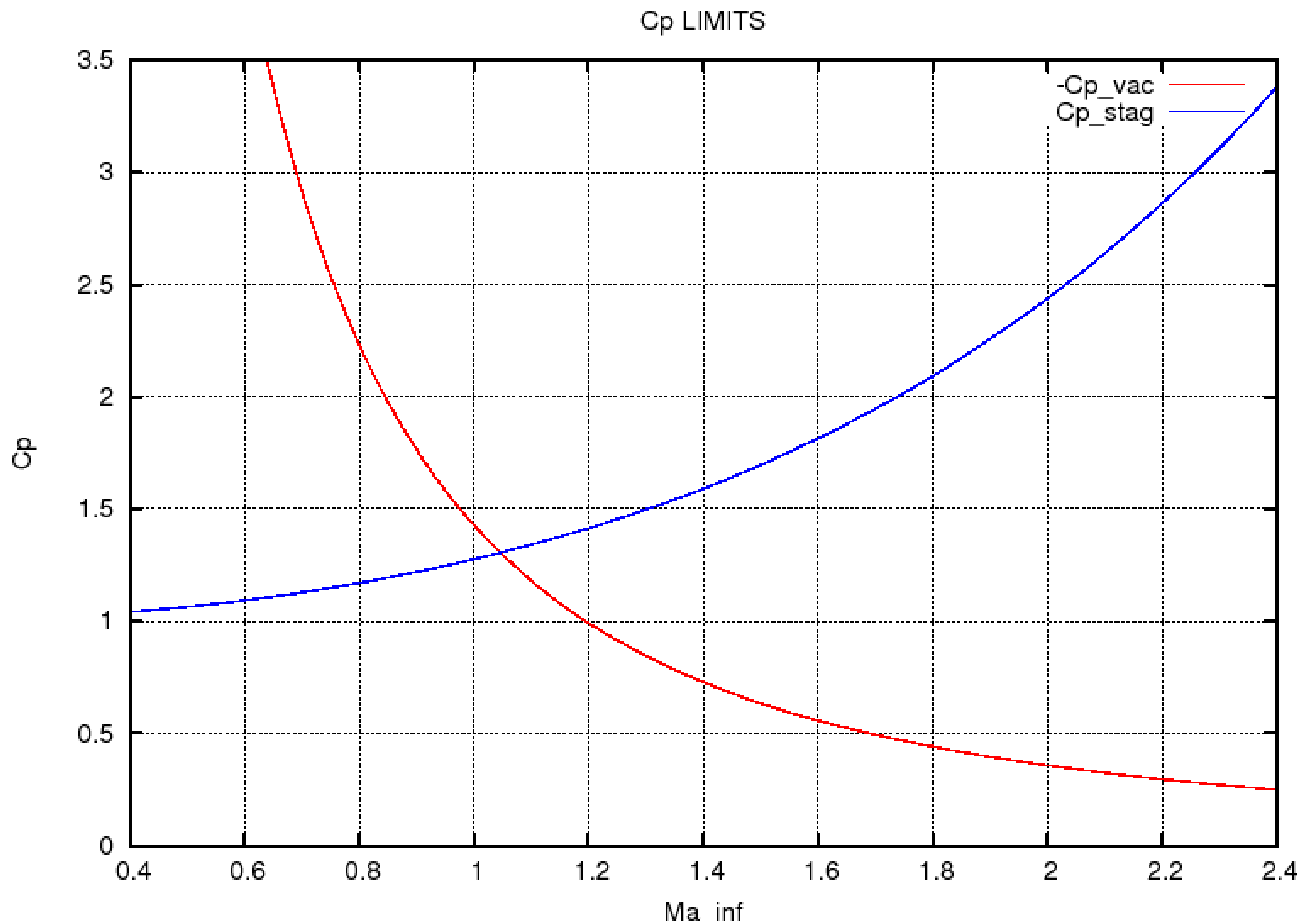
$$Cp_{STAG} = \left| Ma \to 0 \right| = \frac{2}{k\,Ma_\infty^2} \left[\left(1 + \frac{k-1}{2} Ma_\infty^2 \right)^{\frac{k}{k-1}} - 1 \right]$$

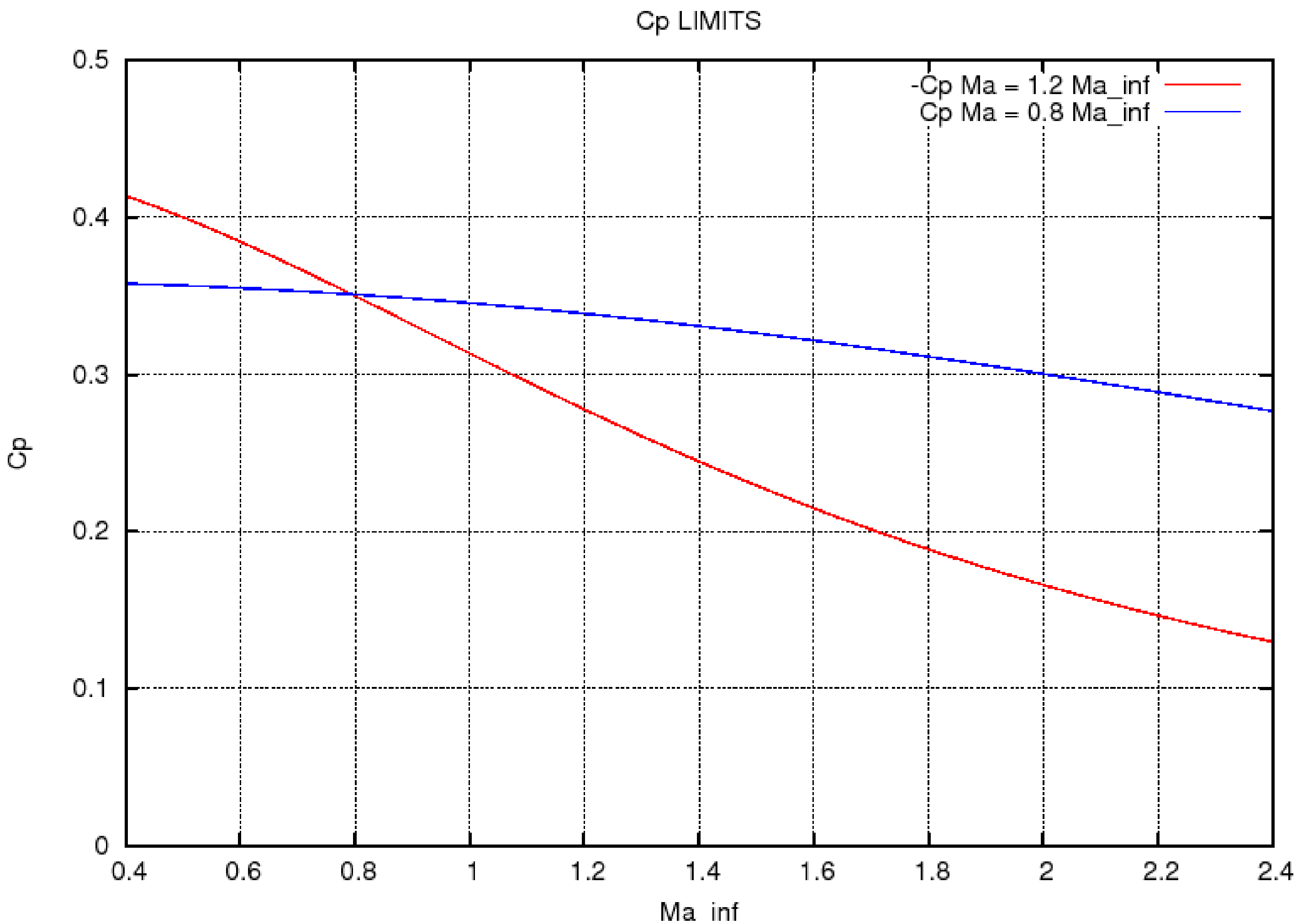
$$C_{L_{MAX}} = Cp_{STAG} - Cp_{VAC} = \frac{2}{k\,Ma_\infty^2}\left(1+\frac{k-1}{2}Ma_\infty^2\right)^{\frac{k}{k-1}}$$

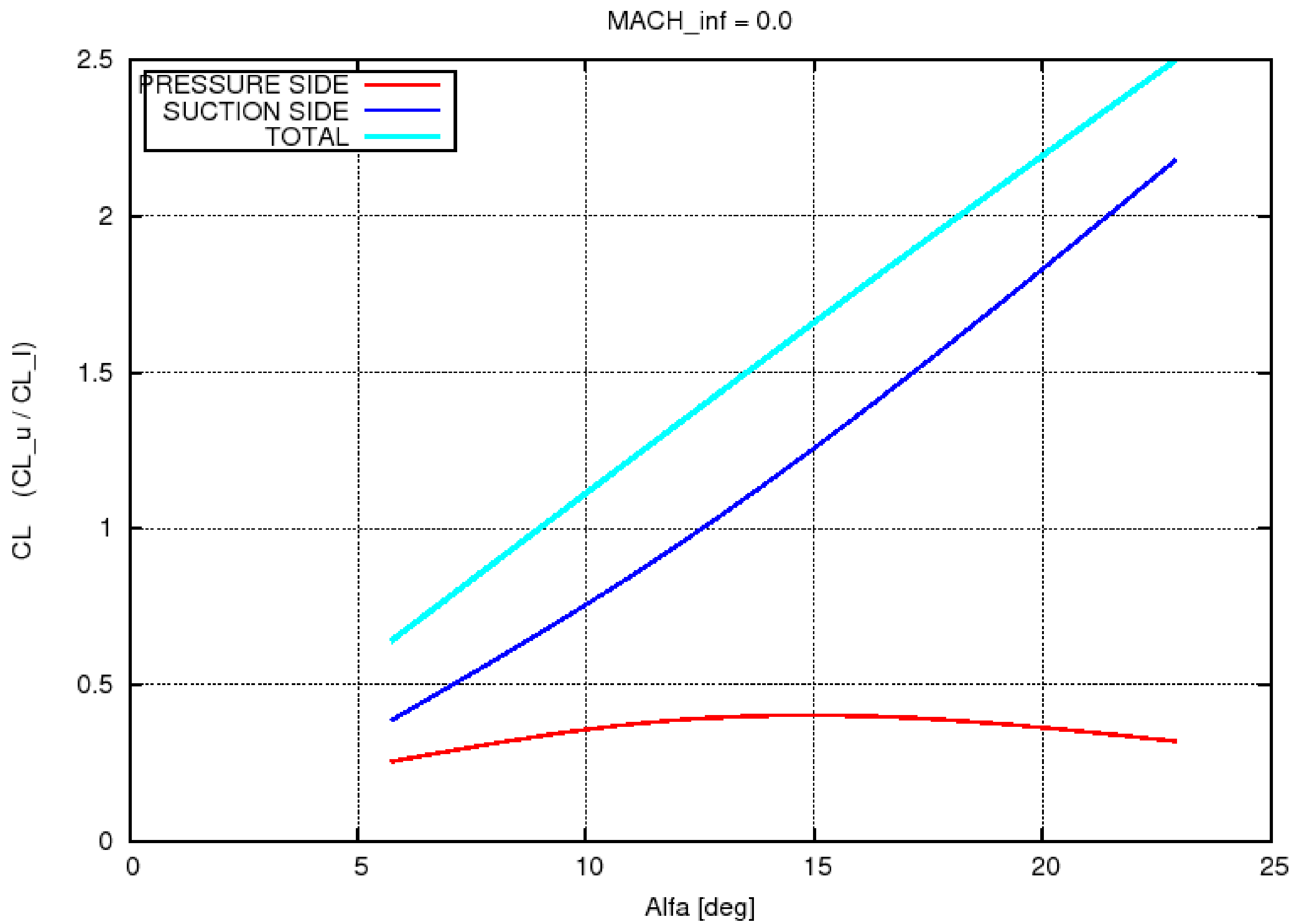
$$C_{L_{MAX}}Ma_\infty^2 = \frac{2}{k}\left(1+\frac{k-1}{2}Ma_\infty^2\right)^{\frac{k}{k-1}} = \left|AIR\right| = \frac{1}{0.7}\left(1+0.2Ma_\infty^2\right)^{3.5}$$

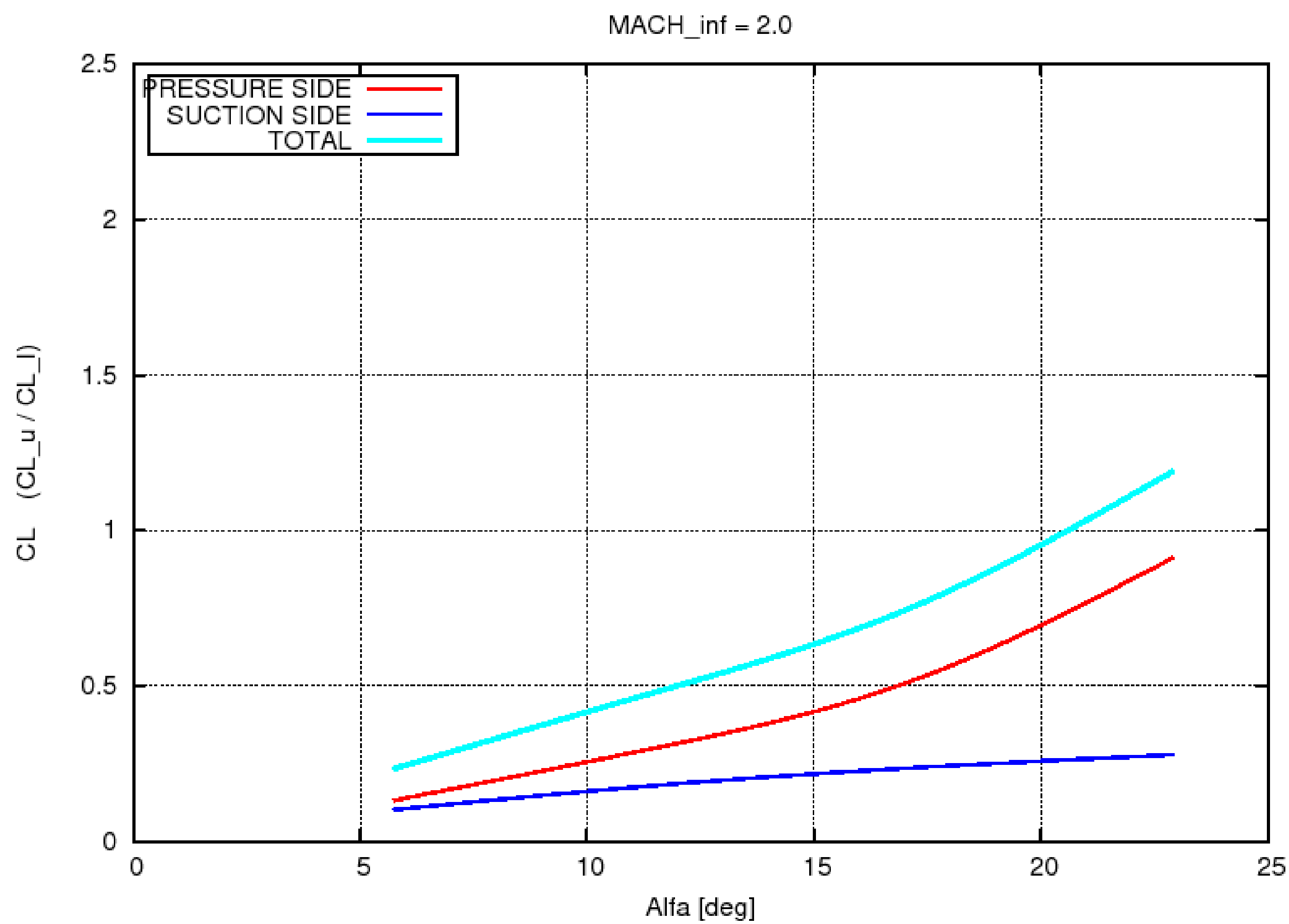
$$\left(\frac{L/S}{p_\infty}\right)_{MAX} = \left(1+0.2Ma_\infty^2\right)^{3.5}/\,0.7$$

$$\left(\frac{L/S}{p_\infty}\right)_{MAX} \approx 0.5 \quad \rightarrow \quad C_{L_{MAX}}Ma_\infty^2 \approx 0.7 \quad (1.2)$$

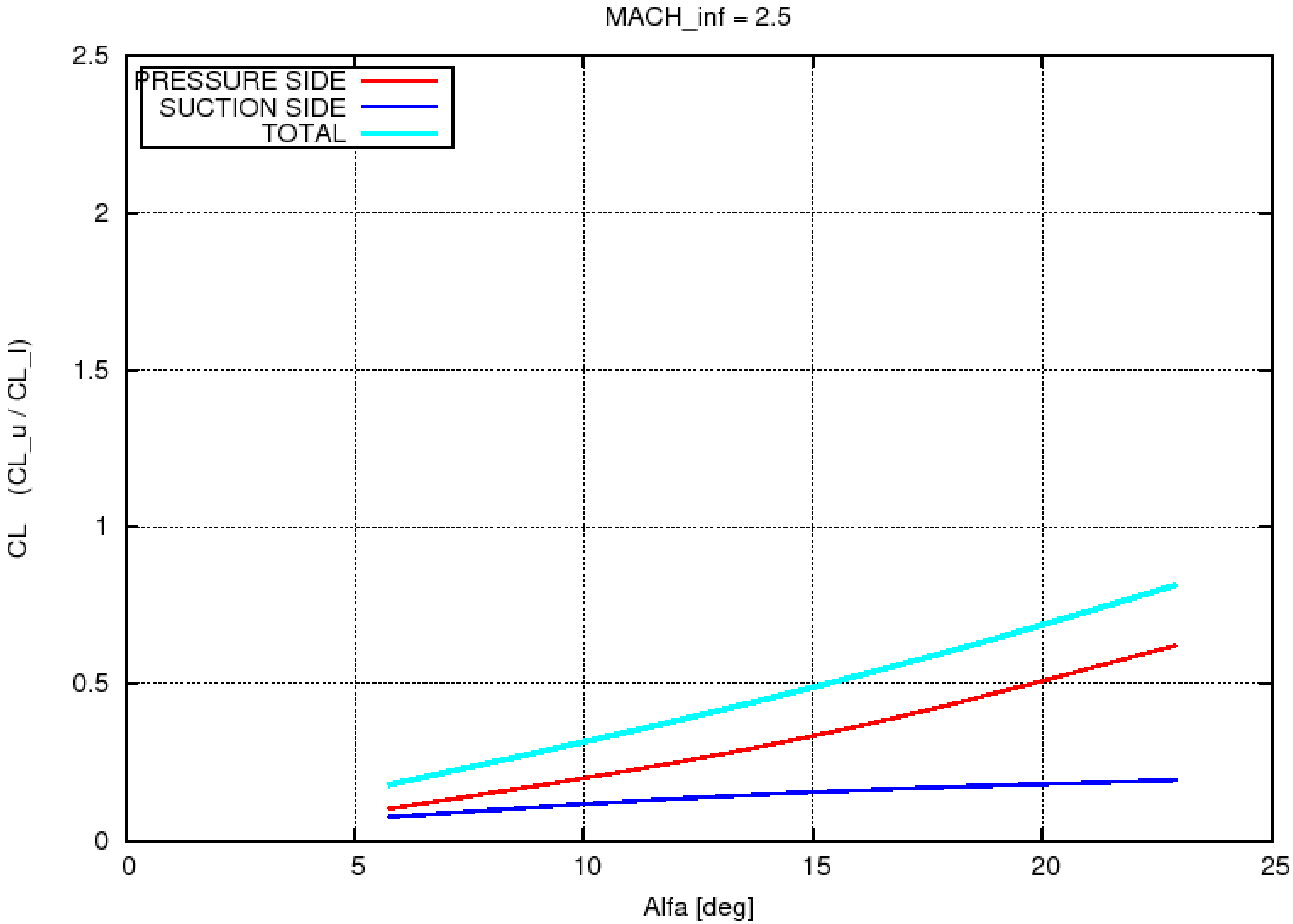

Cp LIMITS
-Cp_vac
Cp_stag
Cp
Ma_inf
3.5
3
2.5
2
1.5
1
0.5
0
0.4
0.6
0.8
1
1.2
1.4
1.6
1.8
2
2.2
2.4


Cp LIMITS
-Cp Ma = 1.2 Ma_inf
Cp Ma = 0.8 Ma_inf
Cp
Ma_inf
0
0.1
0.2
0.3
0.4
0.5
0.4
0.6
0.8
1
1.2
1.4
1.6
1.8
2
2.2
2.4

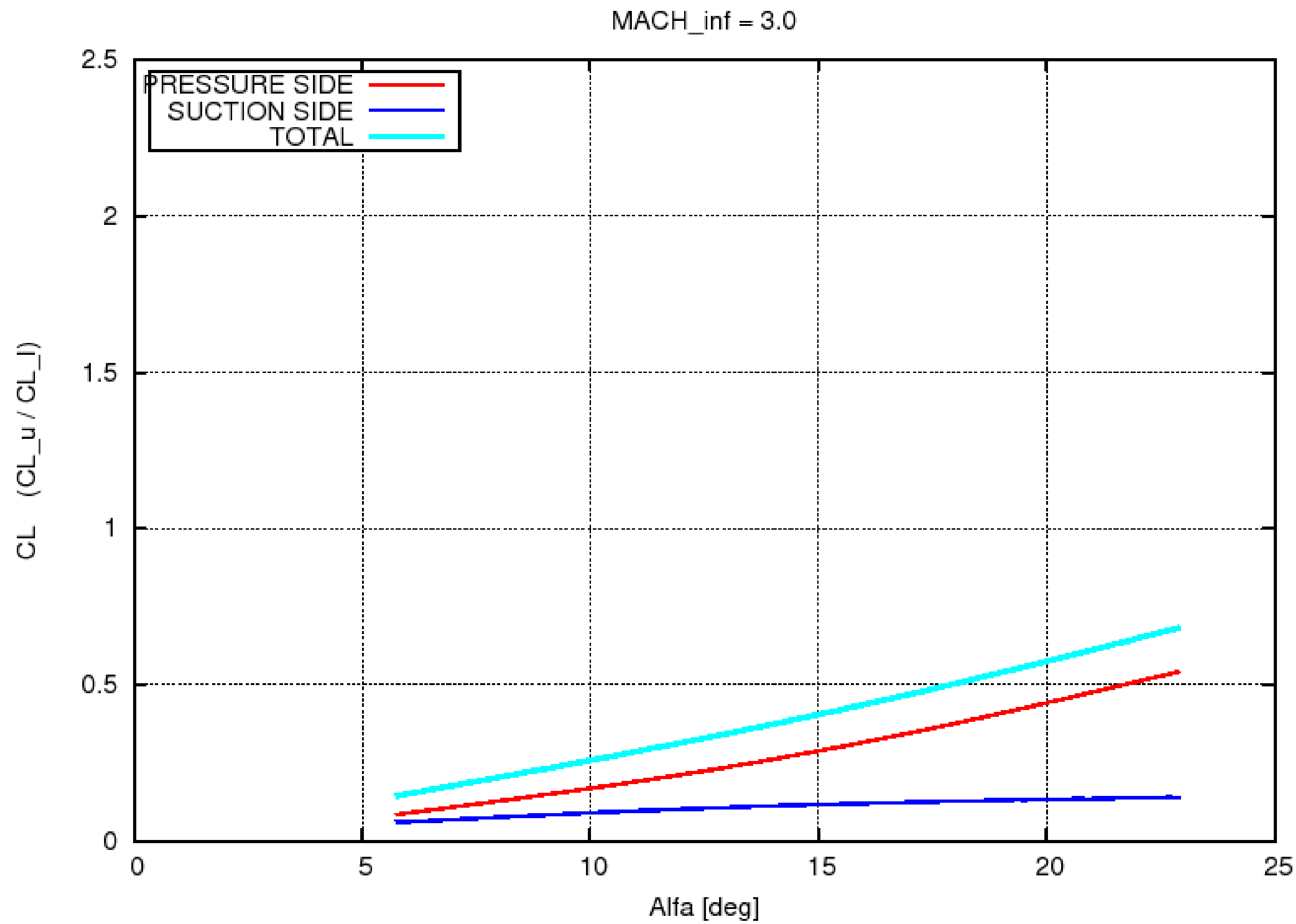

MACH_inf = 0.0
PRESSURE SIDE
SUCTION SIDE
TOTAL
CL (CL_u / CL_l)
Alfa [deg]
0
0.5
1
1.5
2
2.5
5
10
15
20
25

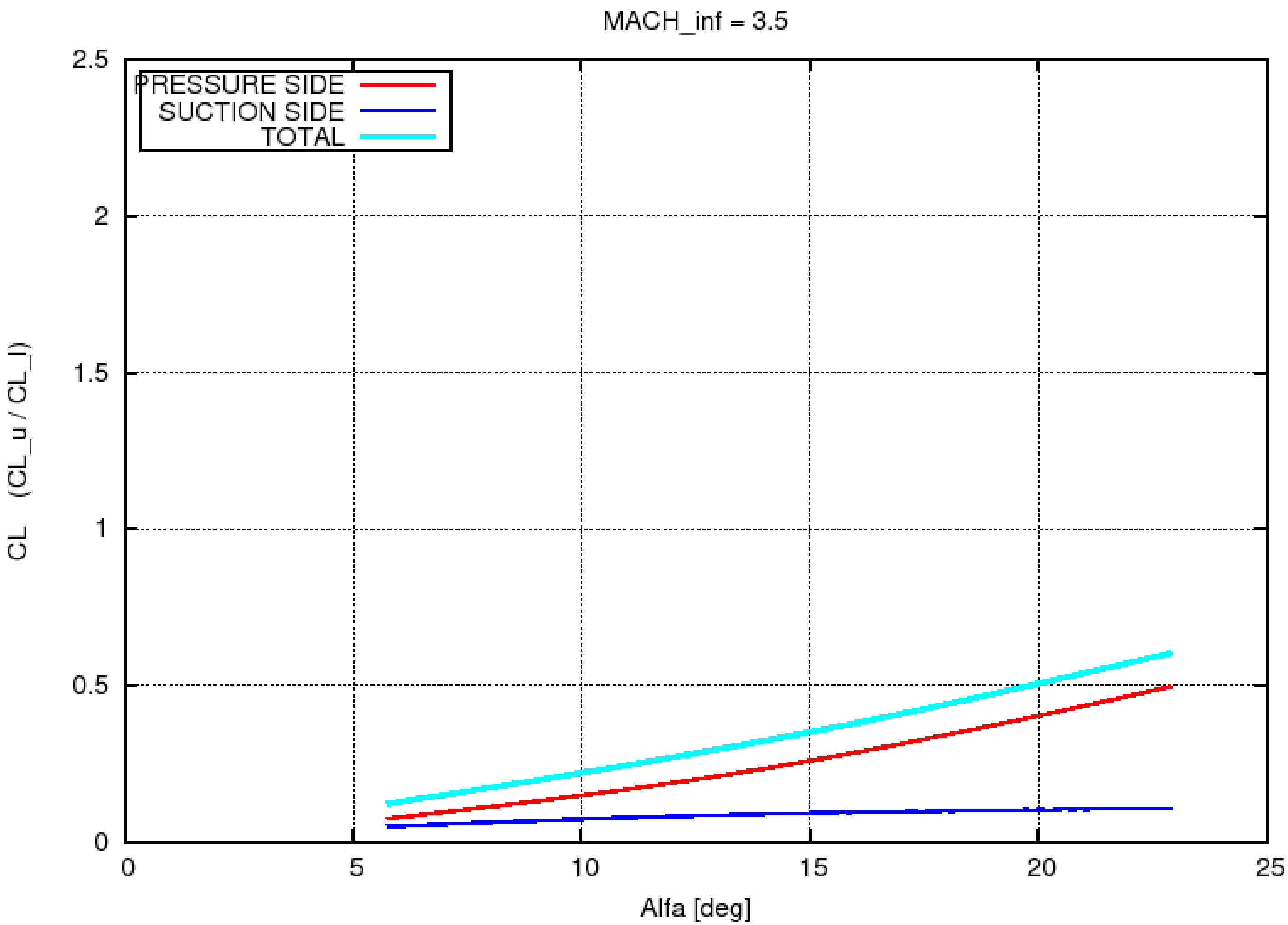


MACH_inf = 2.0
PRESSURE SIDE
SUCTION SIDE
TOTAL
CL (CL_u / CL_l)
Alfa [deg]
0
0.5
1
1.5
2
2.5
0
5
10
15
20
25



MACH_inf = 2.5
PRESSURE SIDE
SUCTION SIDE
TOTAL
CL (CL_u / CL_l)
Alfa [deg]
2.5
2
1.5
1
0.5
0
0
5
10
15
20
25


MACH_inf = 3.0
PRESSURE SIDE
SUCTION SIDE
TOTAL
CL (CL_u / CL_l)
Alfa [deg]
0
0.5
1
1.5
2
2.5
0
5
10
15
20
25



MACH_inf = 3.5
PRESSURE SIDE
SUCTION SIDE
TOTAL
CL (CL_u / CL_l)
Alfa [deg]
0
0.5
1
1.5
2
2.5
0
5
10
15
20
25

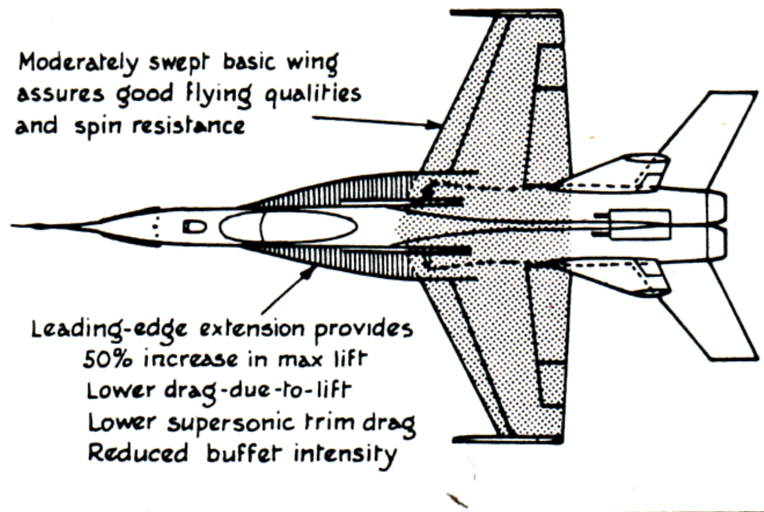
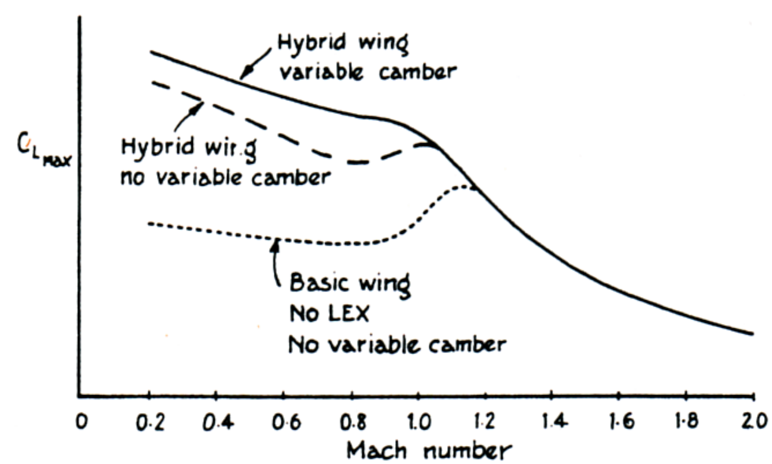


Fig 84 Effect of hybrid wing and variable camber on maximum lift.[27]



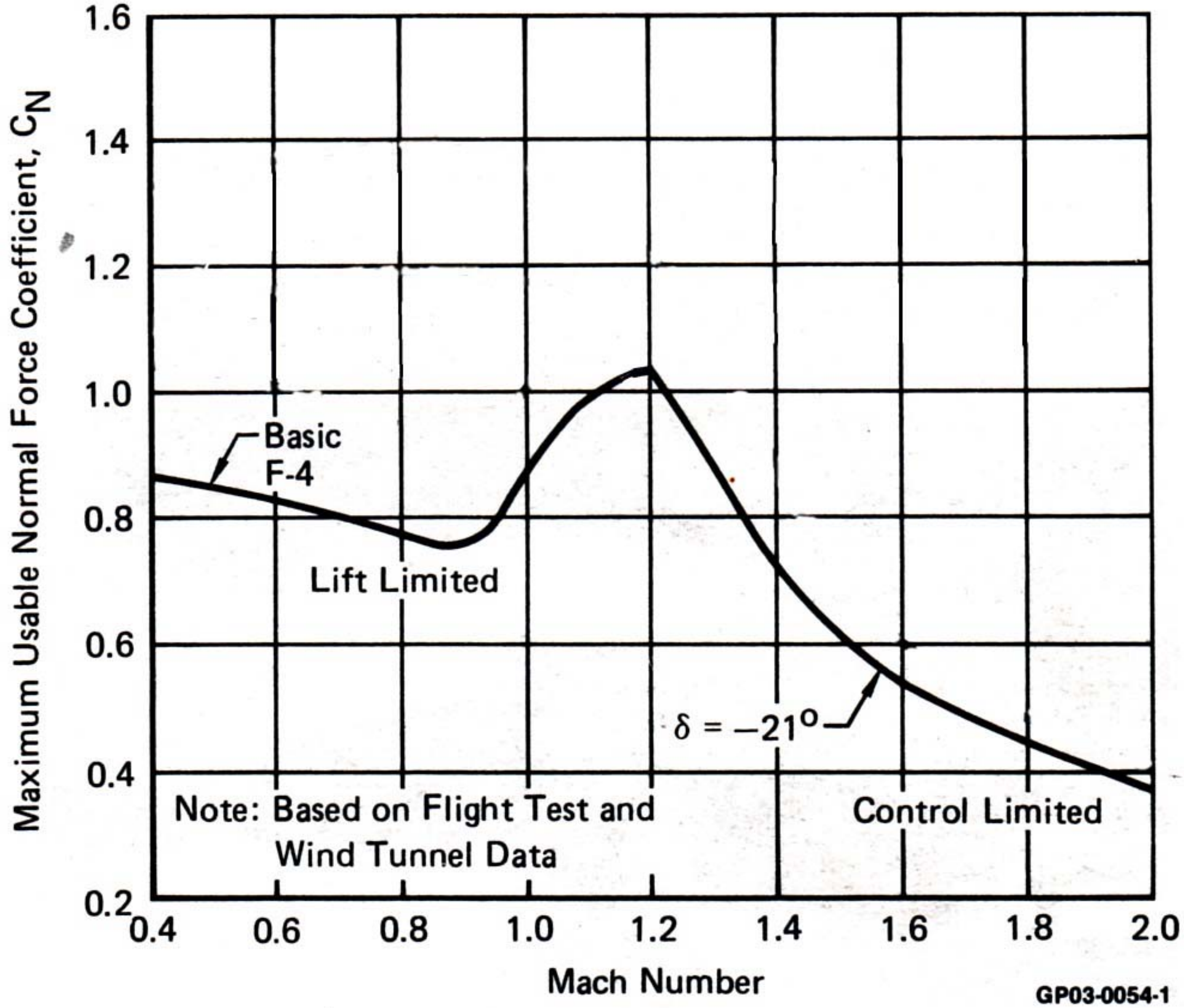
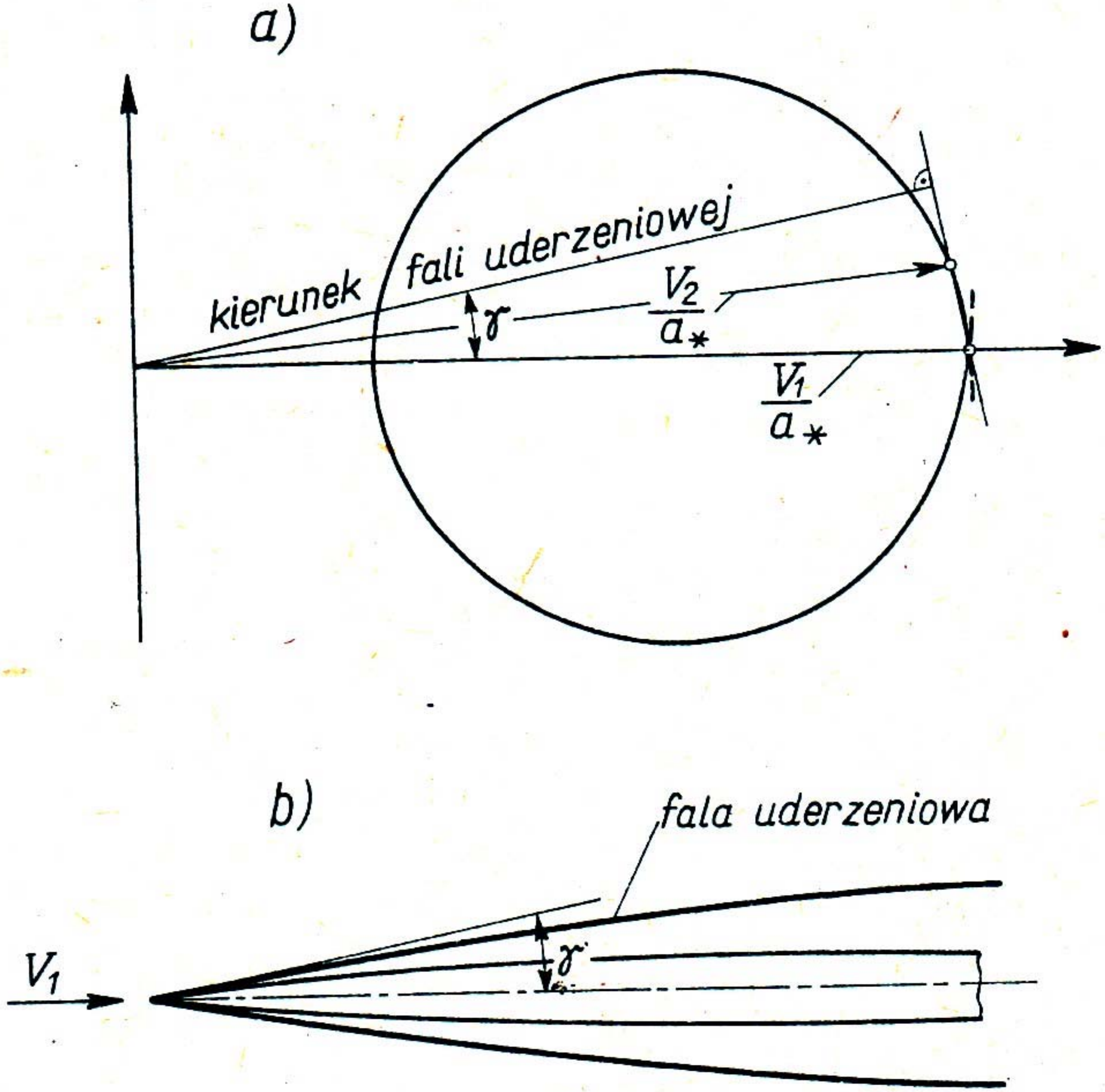
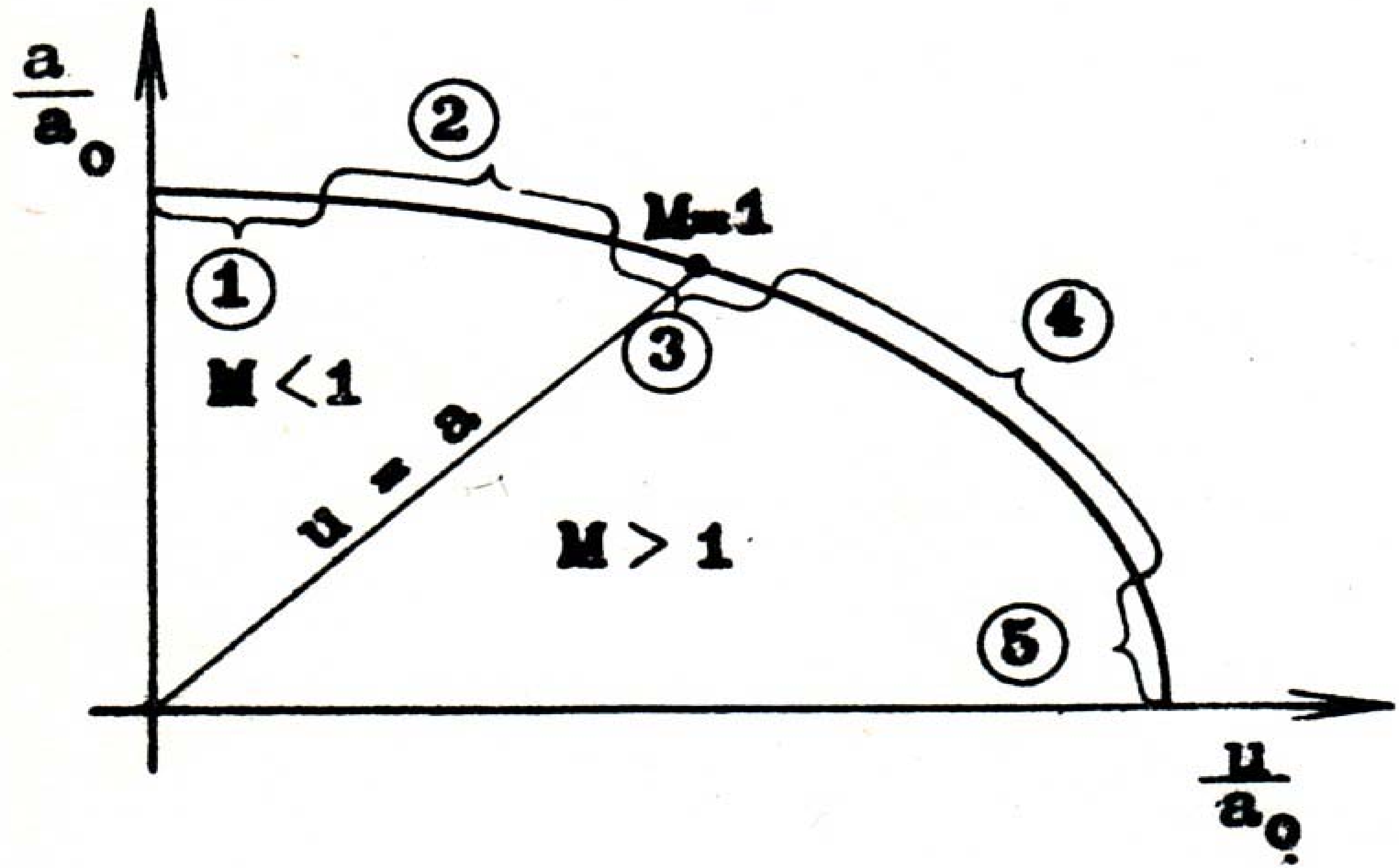


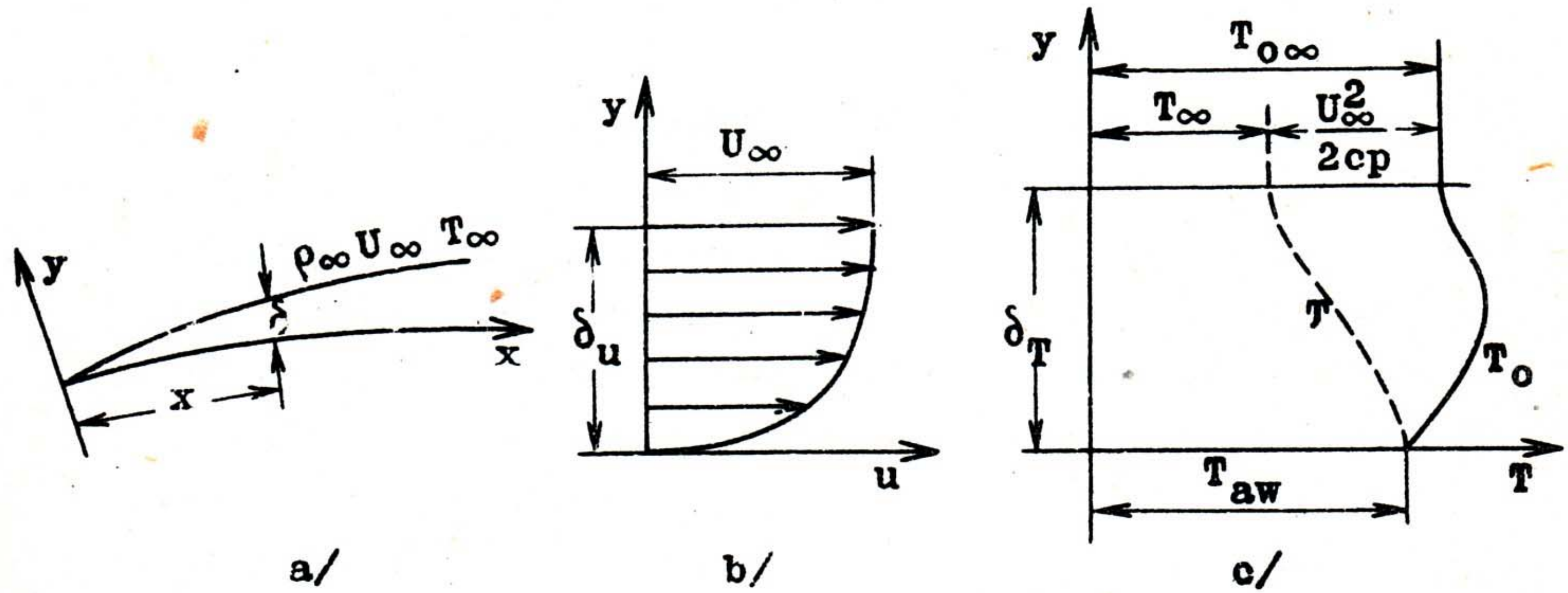
Fig. 24 Model F-4 Maximum Usable Normal Force
C.G. = 31% c



Rys. 10.48. Hipersoniczny opływ cienkiej bryły



U, a, T ?



$$r = \frac{T_{aw} - T_{\infty}}{T_{0\infty} - T_{\infty}} = \frac{T_{aw} - T_{\infty}}{U_{\infty}^{2} / \left(2C_{p}\right)}$$

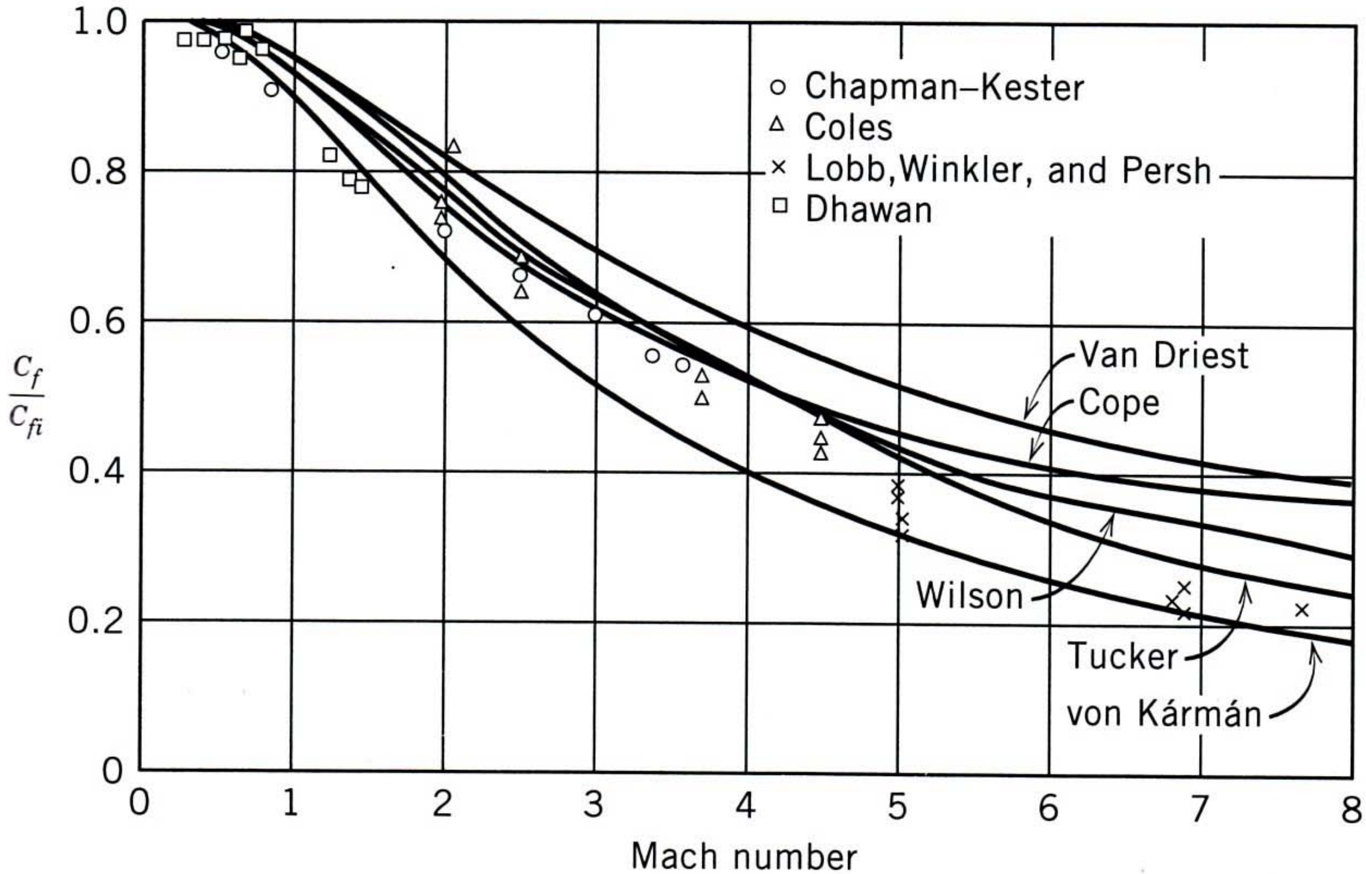


Fig. 18.24. Ratio of compressible to incompressible friction coefficient for turbulent boundary layer on a flat plate as a function of free-stream Mach number.

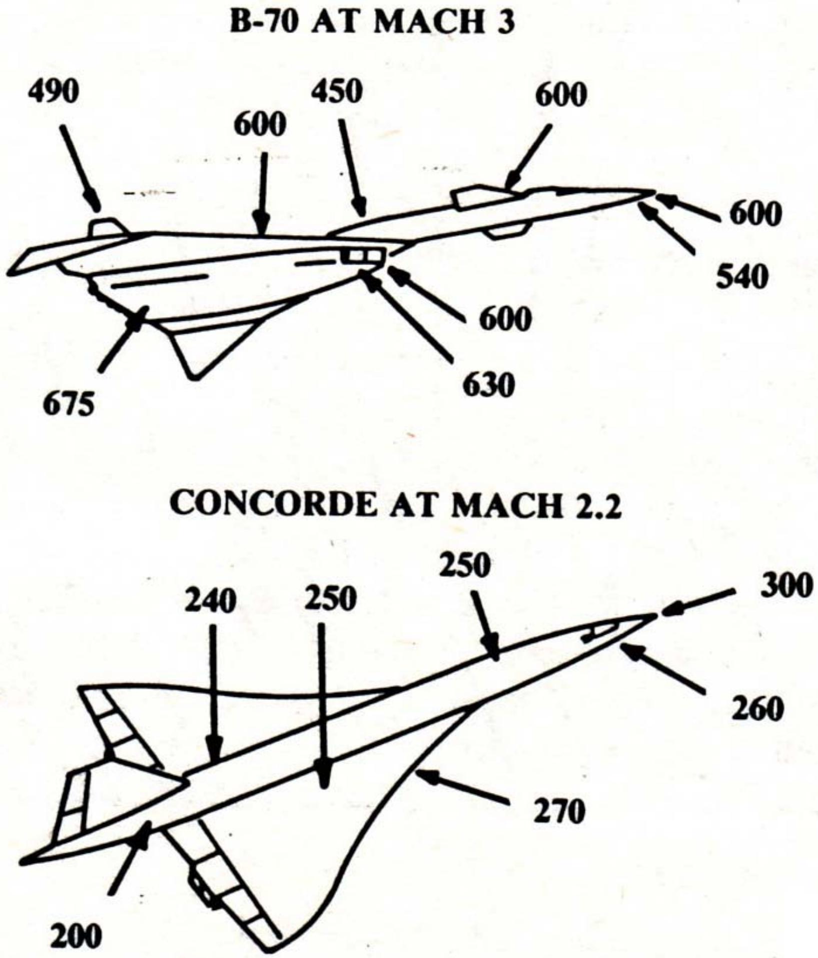
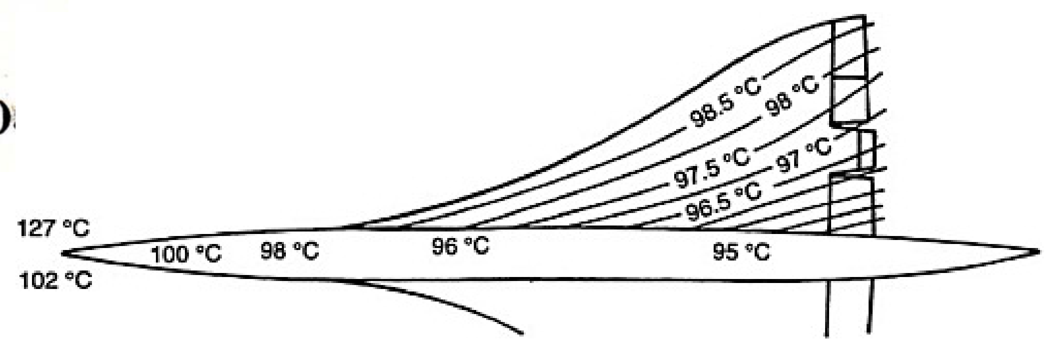


Fig. 14.21 Supersonic skin temperatures (°F)



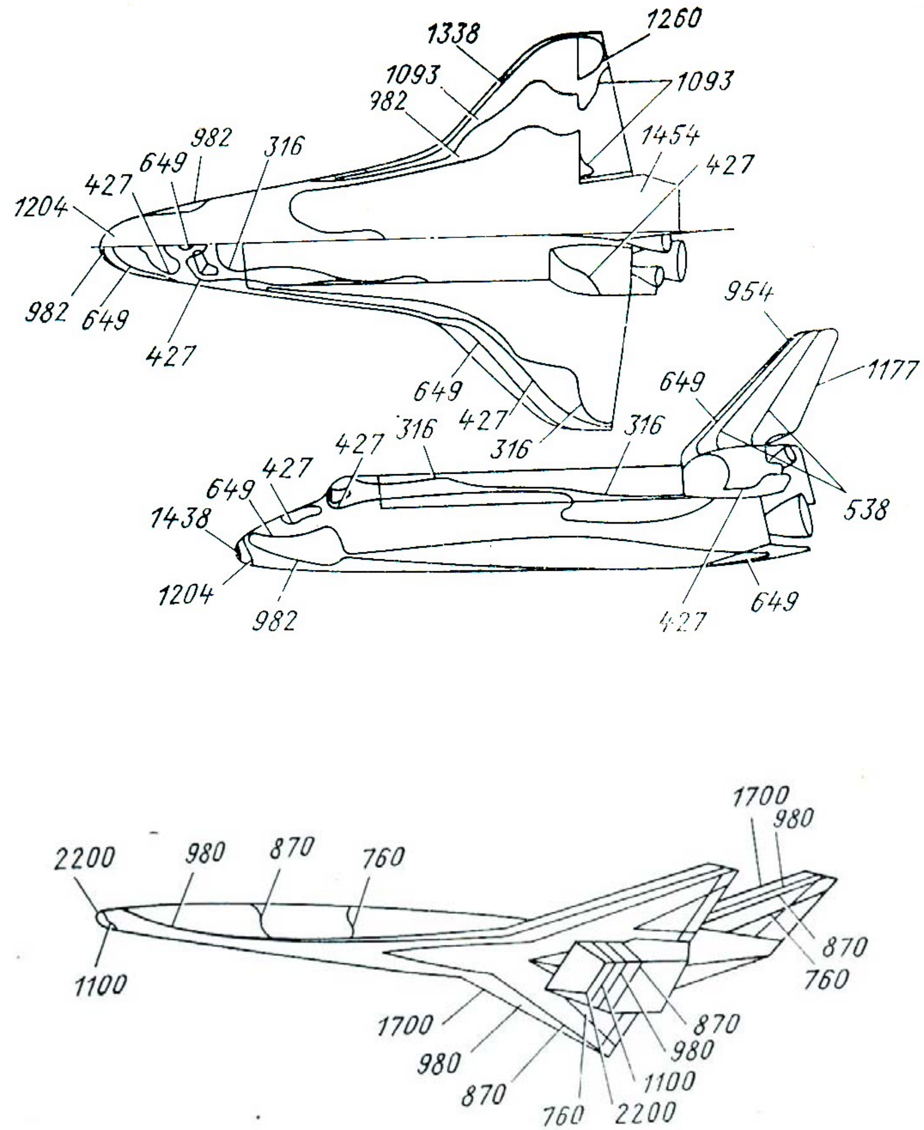


Рис. 12.12. Установившаяся температура на поверхности самолета в градусах Цельсия при длительном полете ($V = 2400$ м/с; $H = 34$ км)