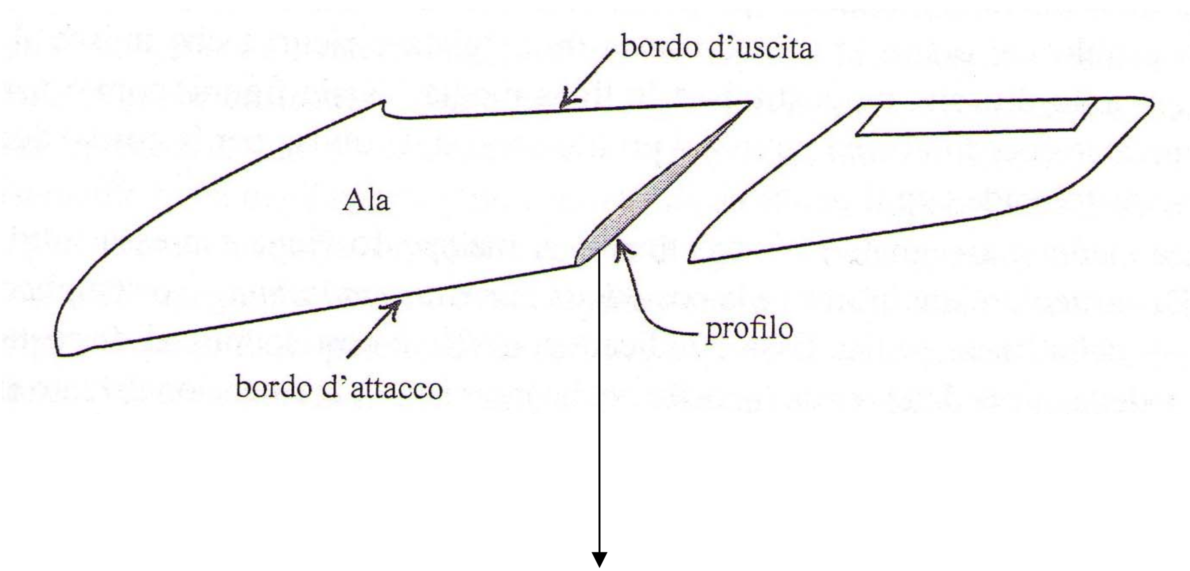
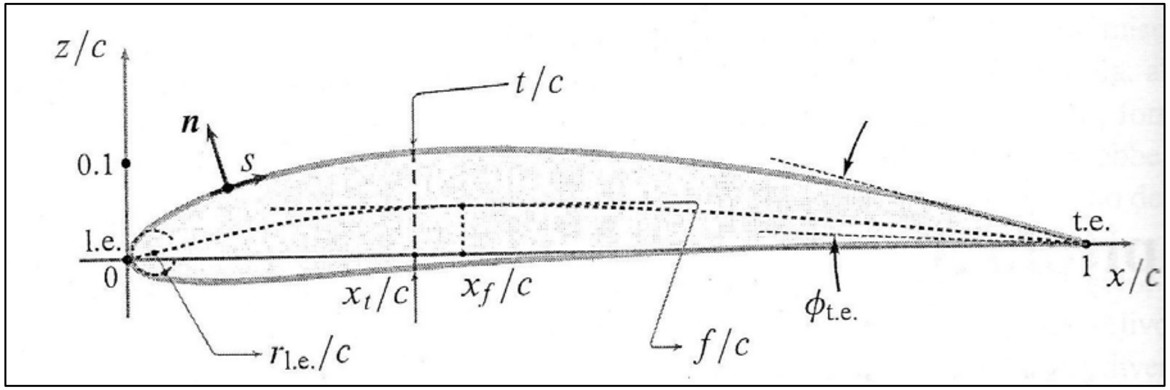
Coefficienti Aerodinamici





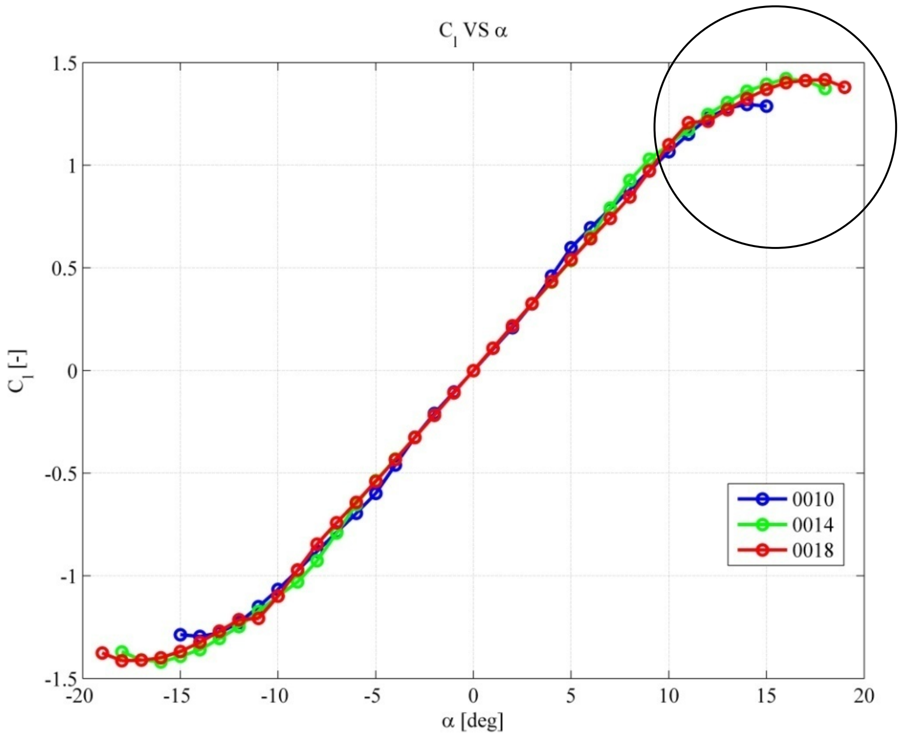
$$Cl = \frac{l}{\frac{1}{2}\cdot\rho\cdot V_\infty^{\;2}\cdot c}$$

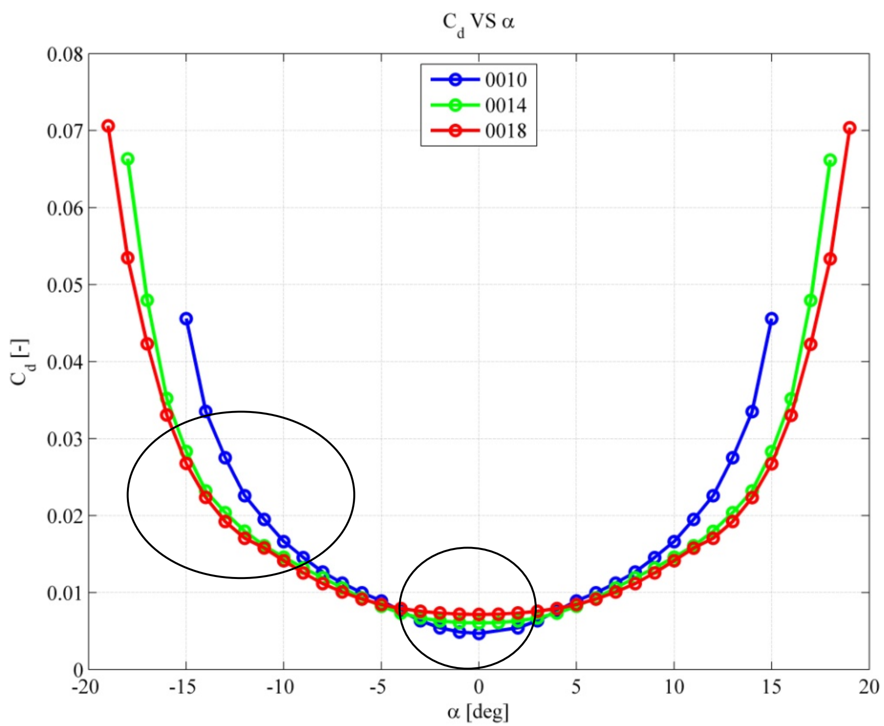
$$Cd = \frac{d}{\frac{1}{2}\cdot\rho\cdot V_\infty^{\;2}\cdot c}$$

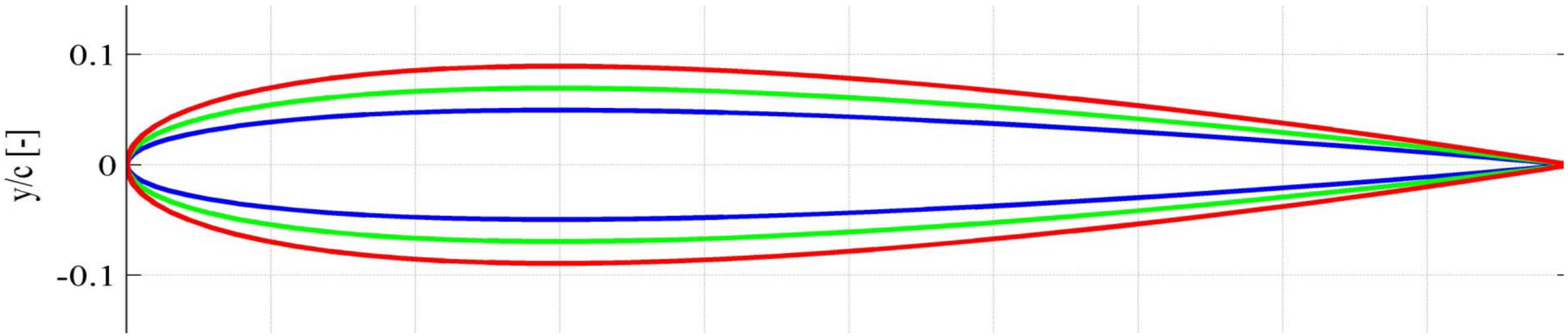
$$Cm_{1/4} = \frac{m_{1/4}}{\frac{1}{2}\cdot\rho\cdot V_\infty^{\;2}\cdot c^2}$$

l [N/m] d [N/m] m [Nm/m]

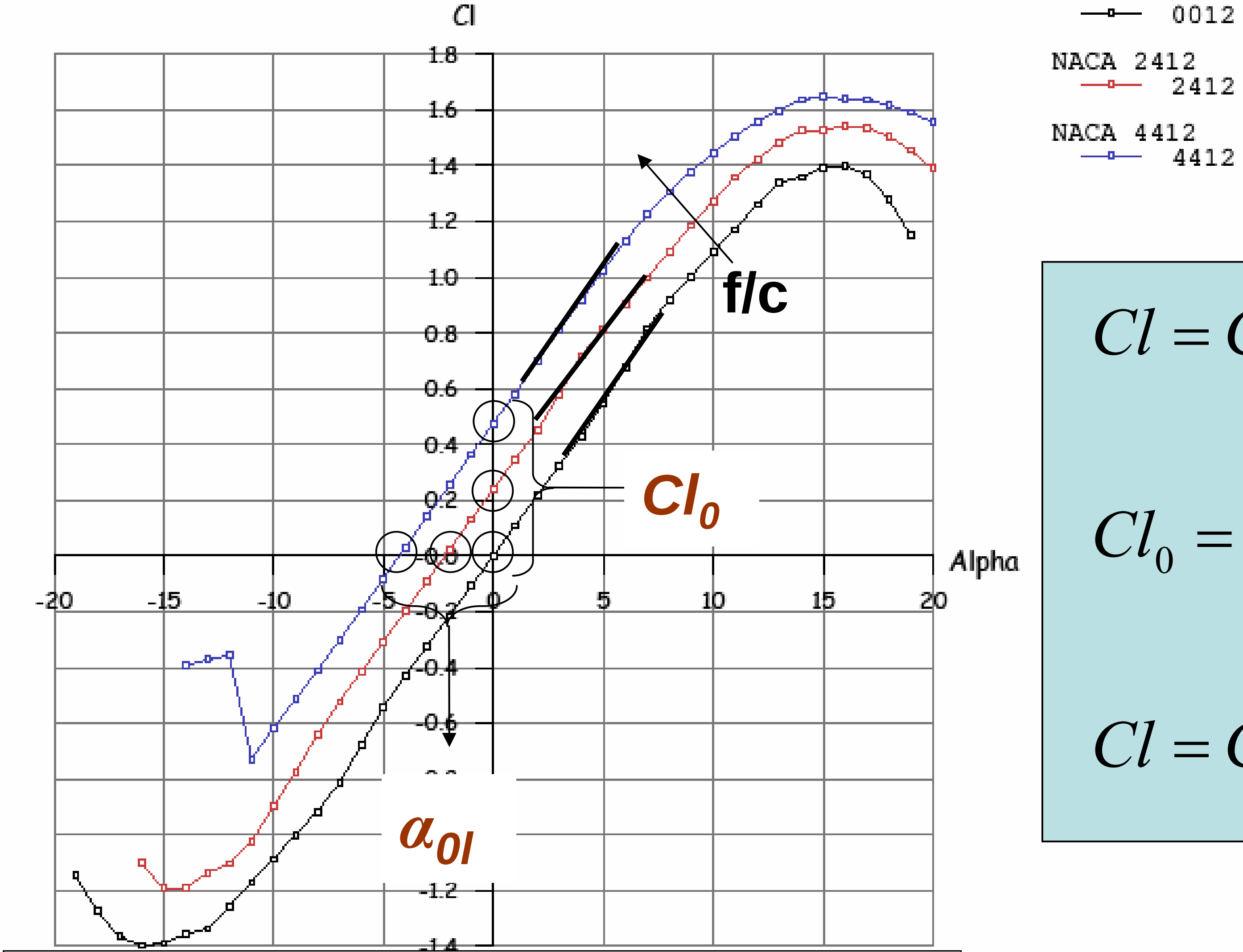
Effetti dello spessore







Effetti della curvatura

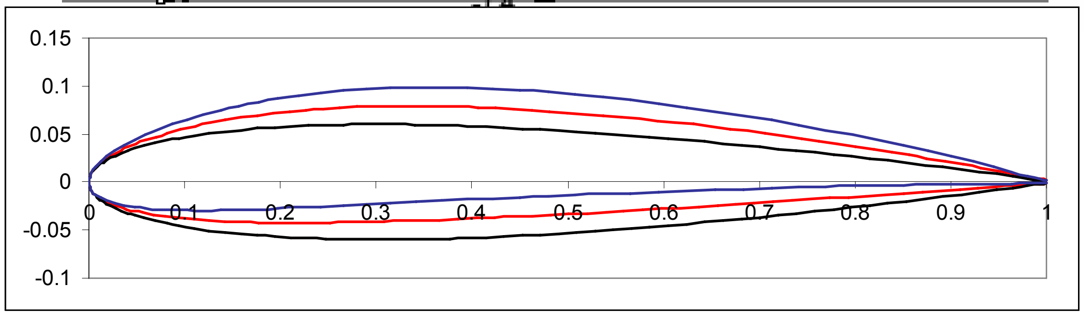


$$Cl = Cl_0 + Cl_\alpha \cdot \alpha$$

$$Cl_0 = -Cl_\alpha \cdot \alpha_{0l}$$

$$\Downarrow$$

$$Cl = Cl_\alpha \cdot (\alpha - \alpha_{ol})$$



Esercizio 1

Profilo Naca 0012

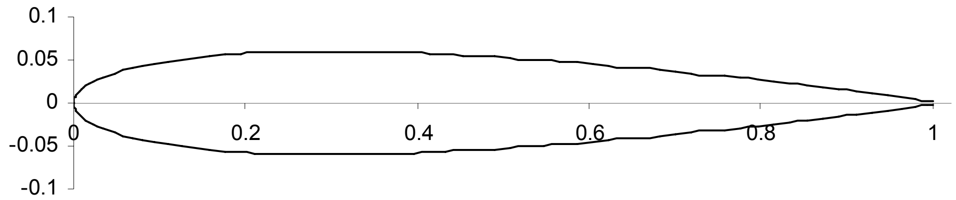
$$Cl_{\alpha} = 2 \cdot \pi \cdot rad^{-1}$$

$$\alpha = 3 \text{ deg}$$

$$c = 2\,m$$

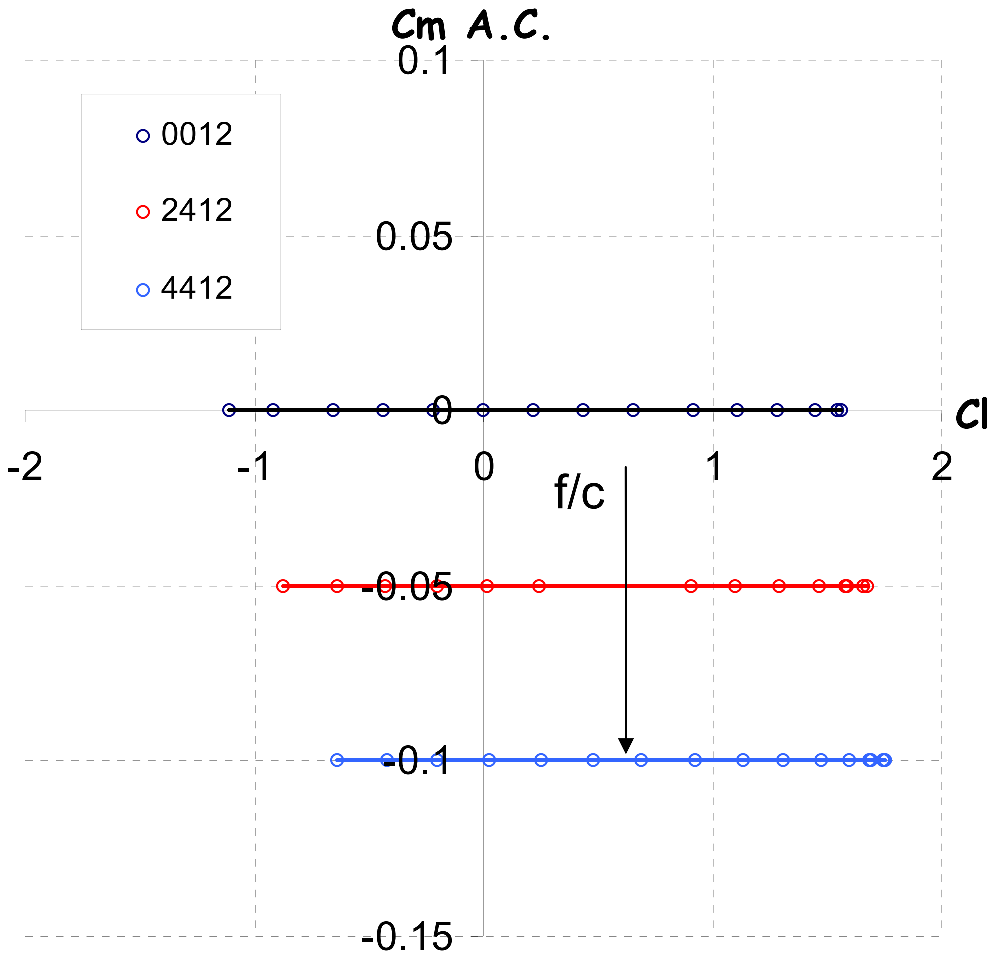
$$V_{\infty} = 30\,m/s$$

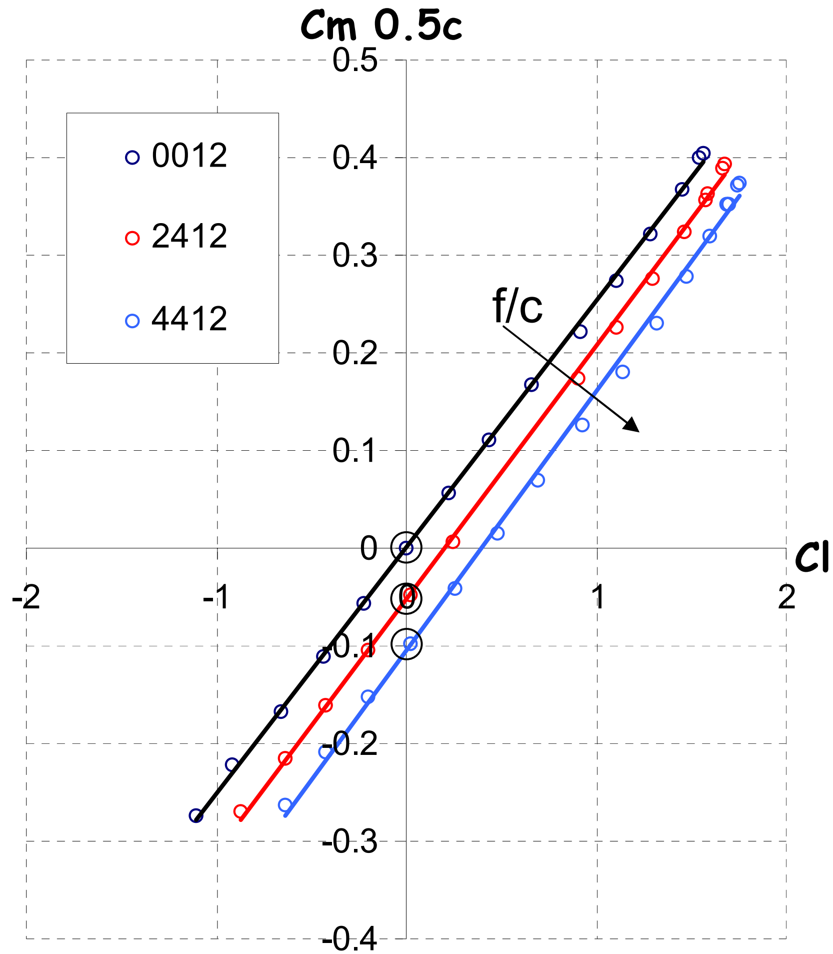
Calcolare *l* in kgf/m

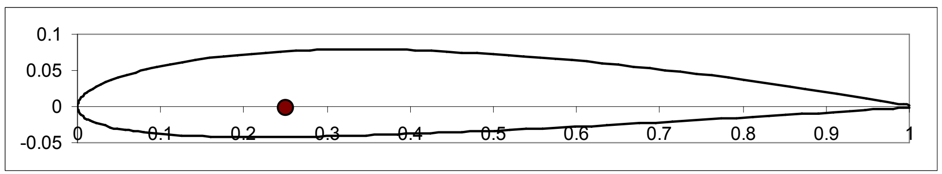


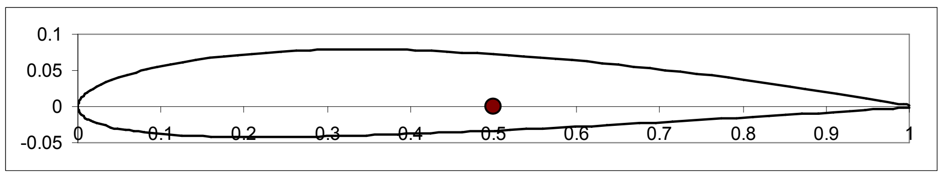
l =364 N/m = 37 kgf/m

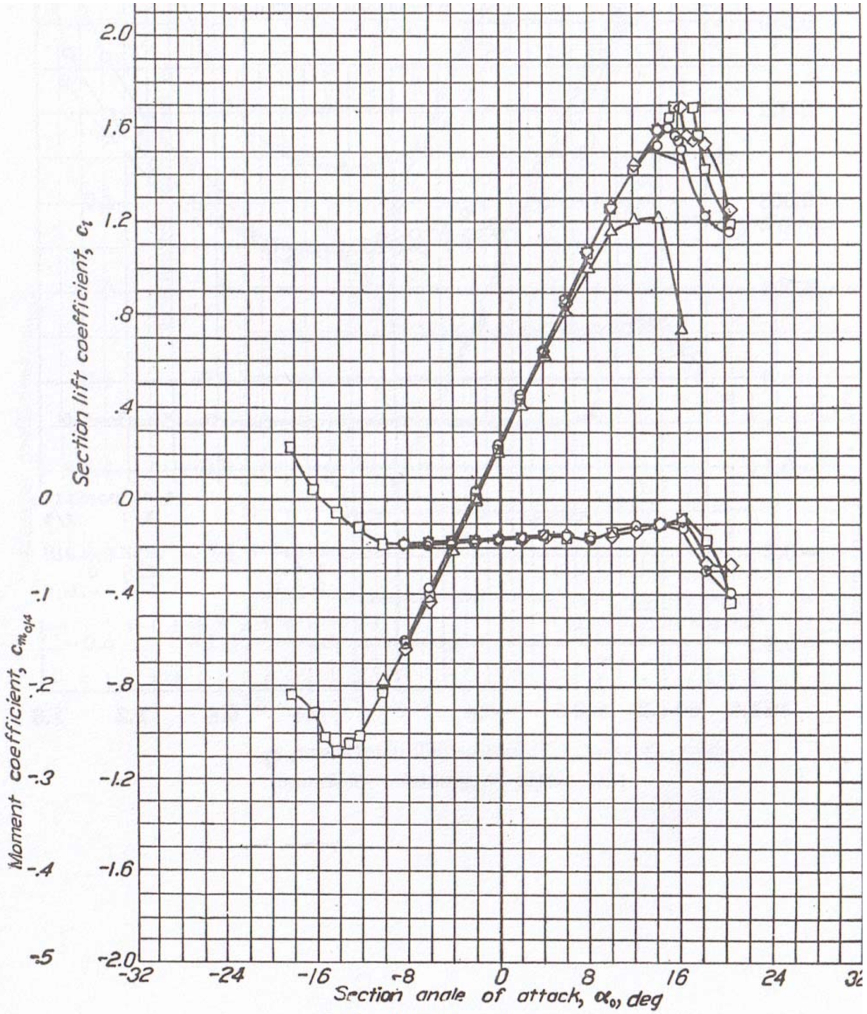
Effetti della curvatura



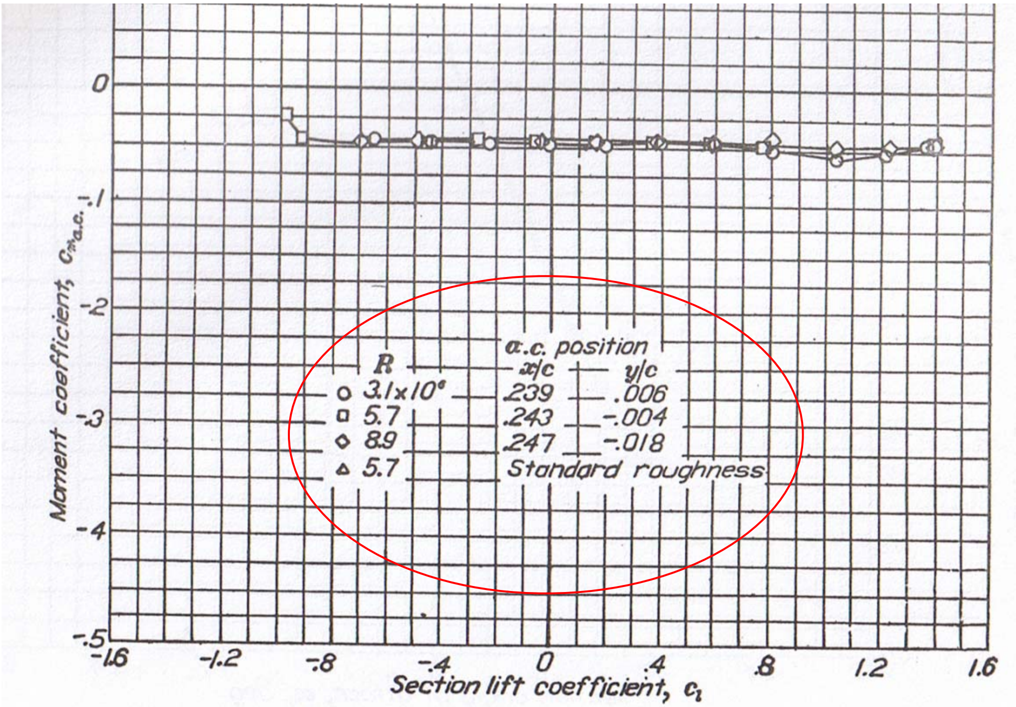








NACA 2412 Wing Section



NACA 2412 Wing Section (*Continued*)

Esercizio 2

- *Profilo Naca 2412*

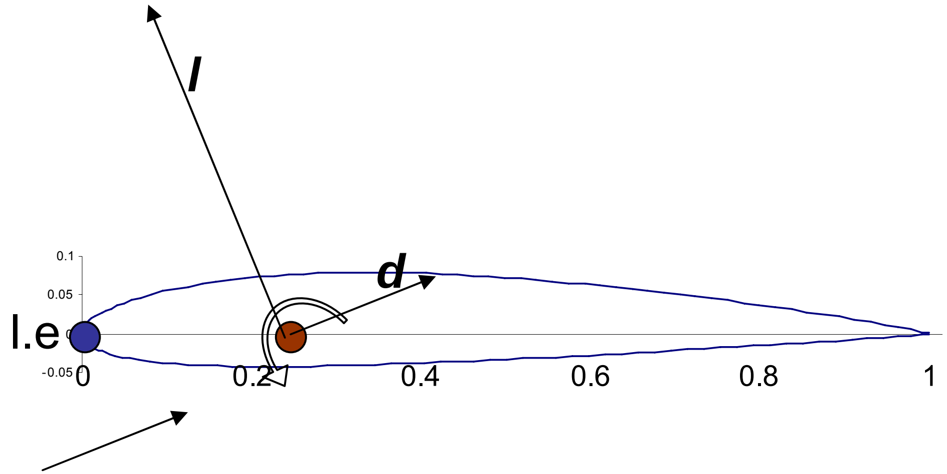
$$Cl = 0.71$$

$$Cd = 0.0064$$

$$\alpha = 4 \deg$$

$$Cm_{1/4} = -0.05$$

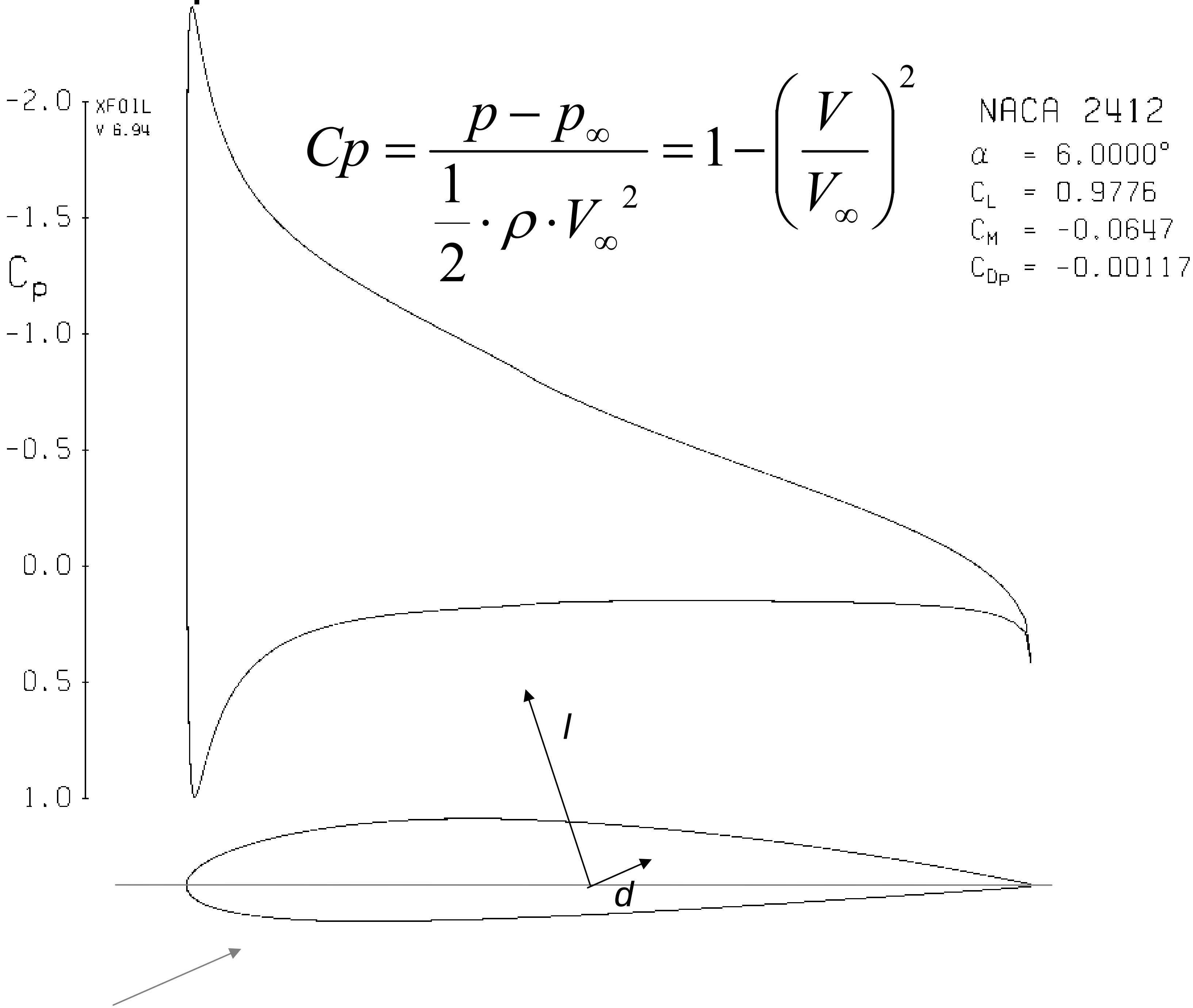
Calcolare il valore del Cm scegliendo come polo dei momenti il bordo di attacco del profilo

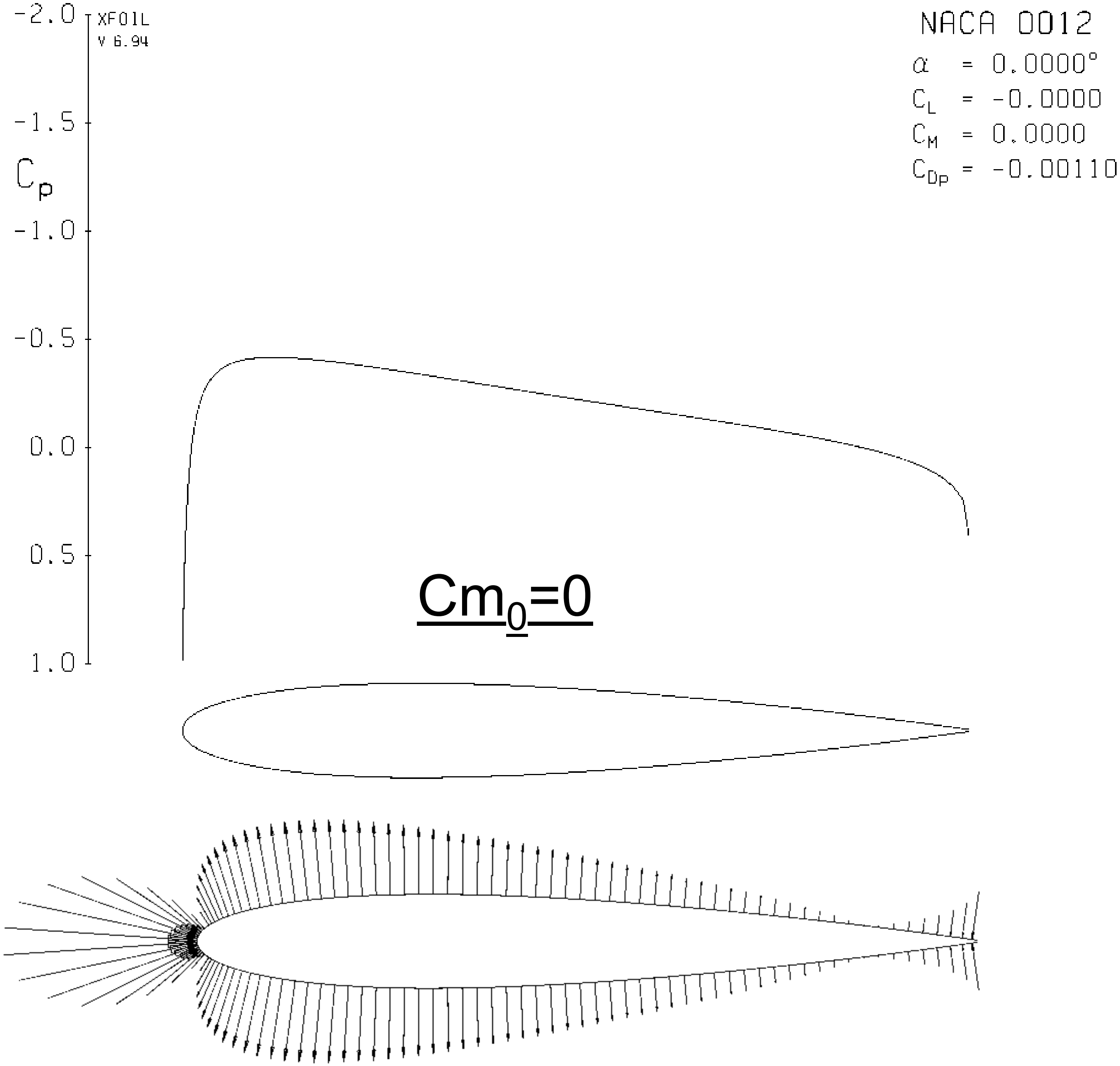


$Cm_{l.e.}$=-0.23

Coefficiente di pressione

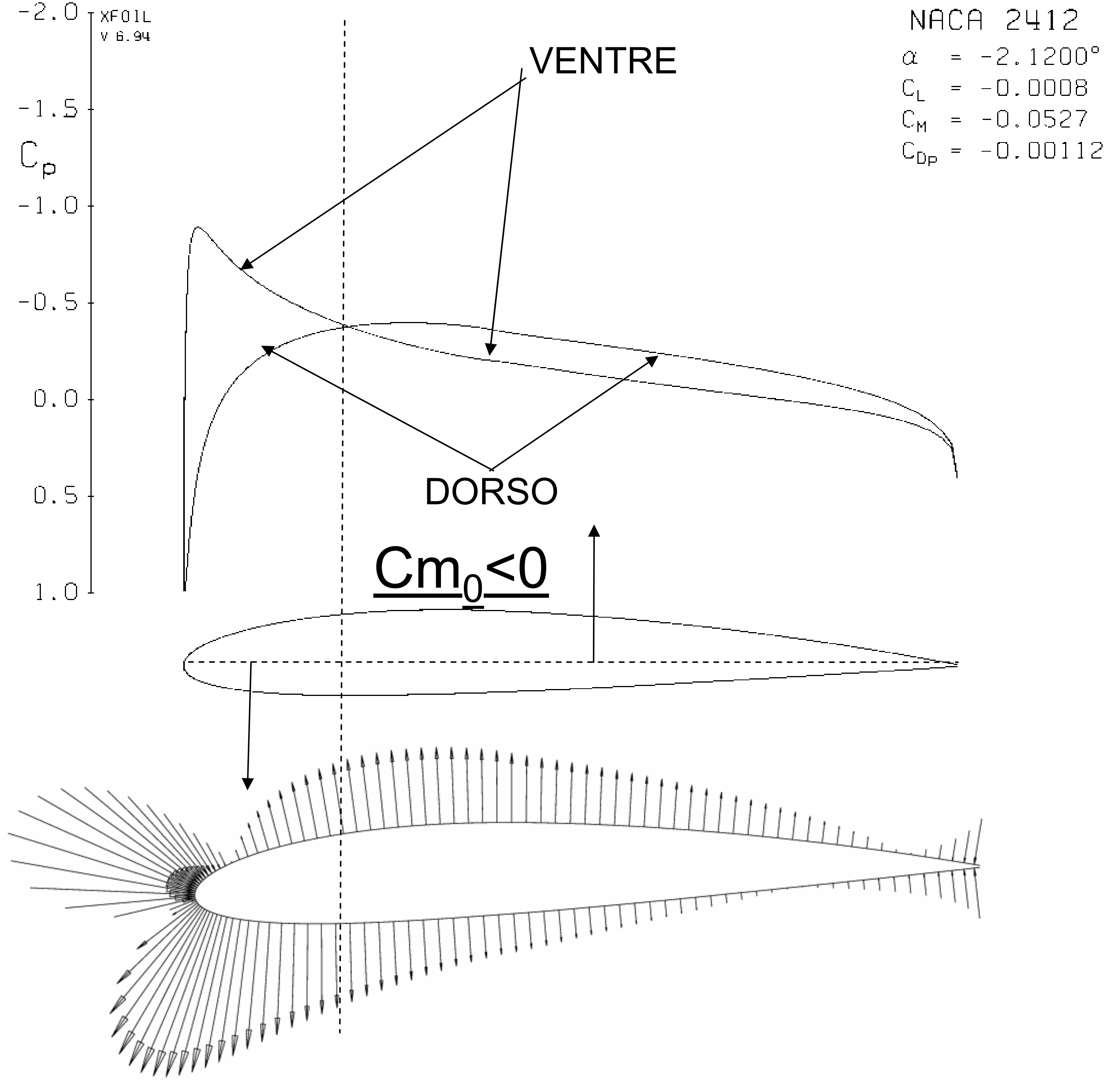
$$Cp = \frac{p - p_\infty}{\frac{1}{2} \cdot \rho \cdot {V_\infty}^2} = 1 - \left(\frac{V}{V_\infty} \right)^2$$





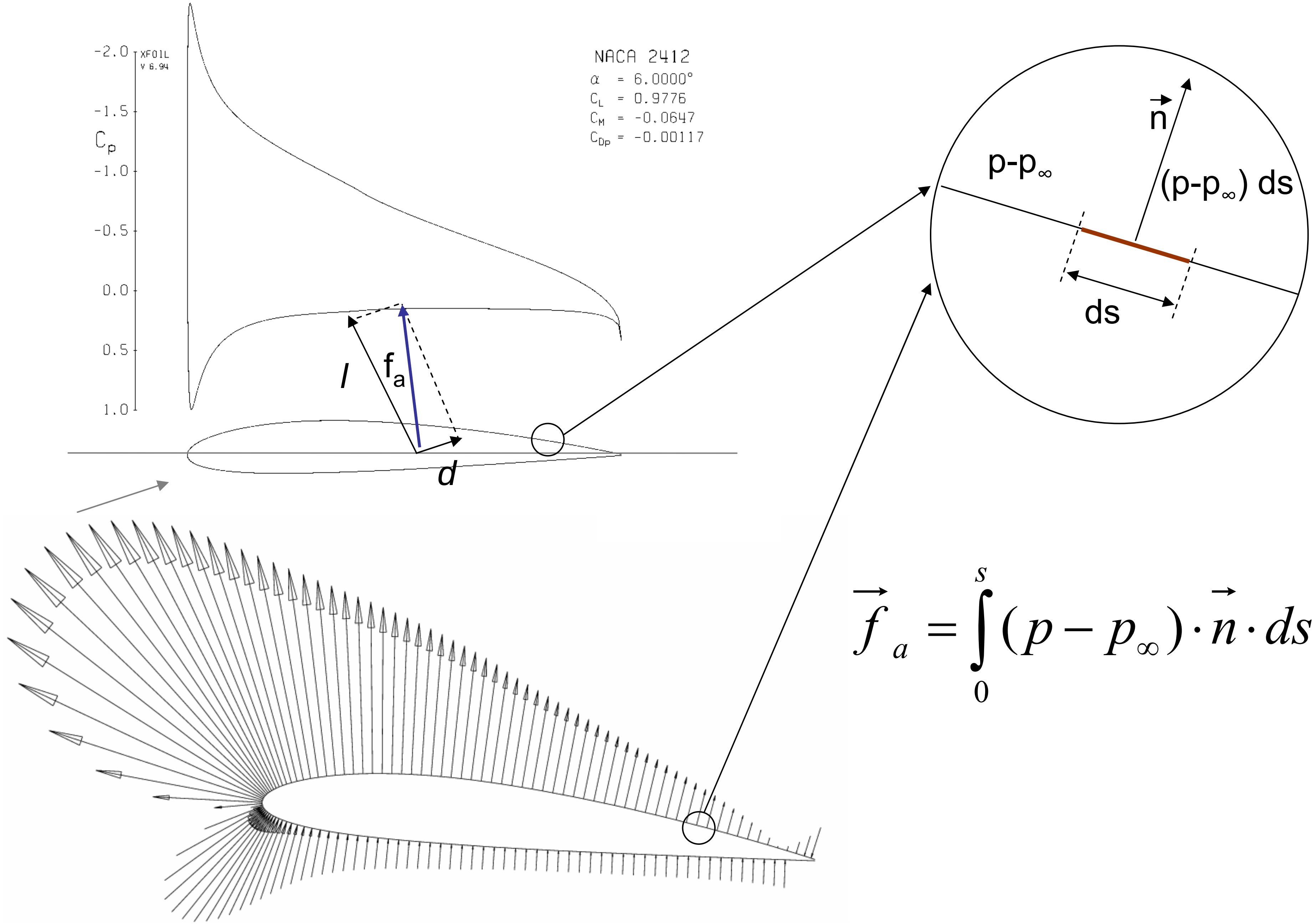
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
XFOIL
V 6.94
NACA 0012
α = 0.0000°
CL = -0.0000
CM = 0.0000
CDp = -0.00110

$Cm_0=0$



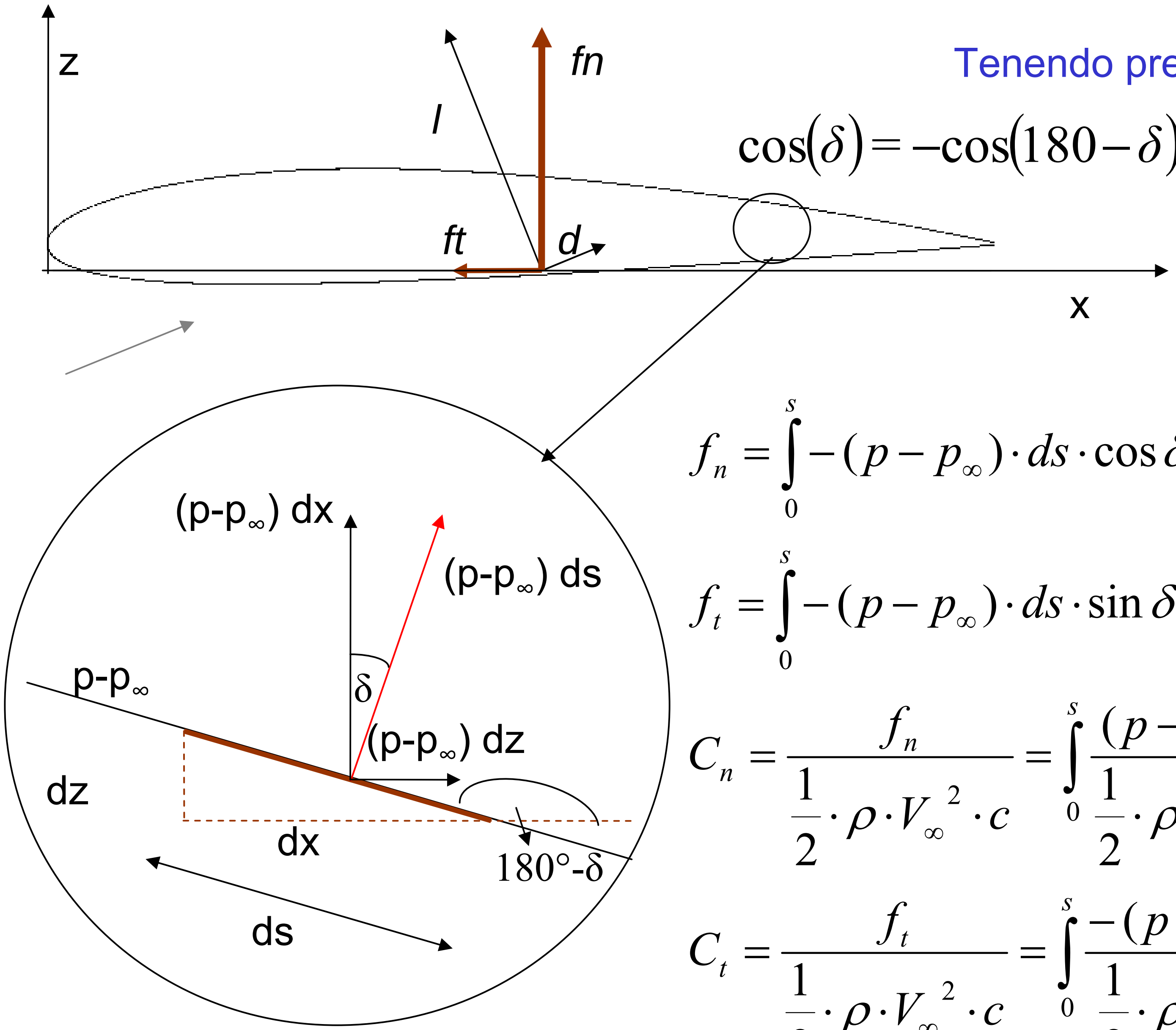
XFOIL
V 6.94
NACA 2412
α = -2.1200°
C_L = -0.0008
C_M = -0.0527
C_Dp = -0.00112
VENTRE
DORSO
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cm0<0

Integrazione della distribuzione di pressione



$$\vec{f}_a = \int_0^s (p - p_\infty) \cdot \vec{n} \cdot ds$$

Integrazione della distribuzione di pressione



Tenendo presente che:

$$\cos(\delta) = -\cos(180-\delta); \sin(\delta) = \sin(180-\delta)$$

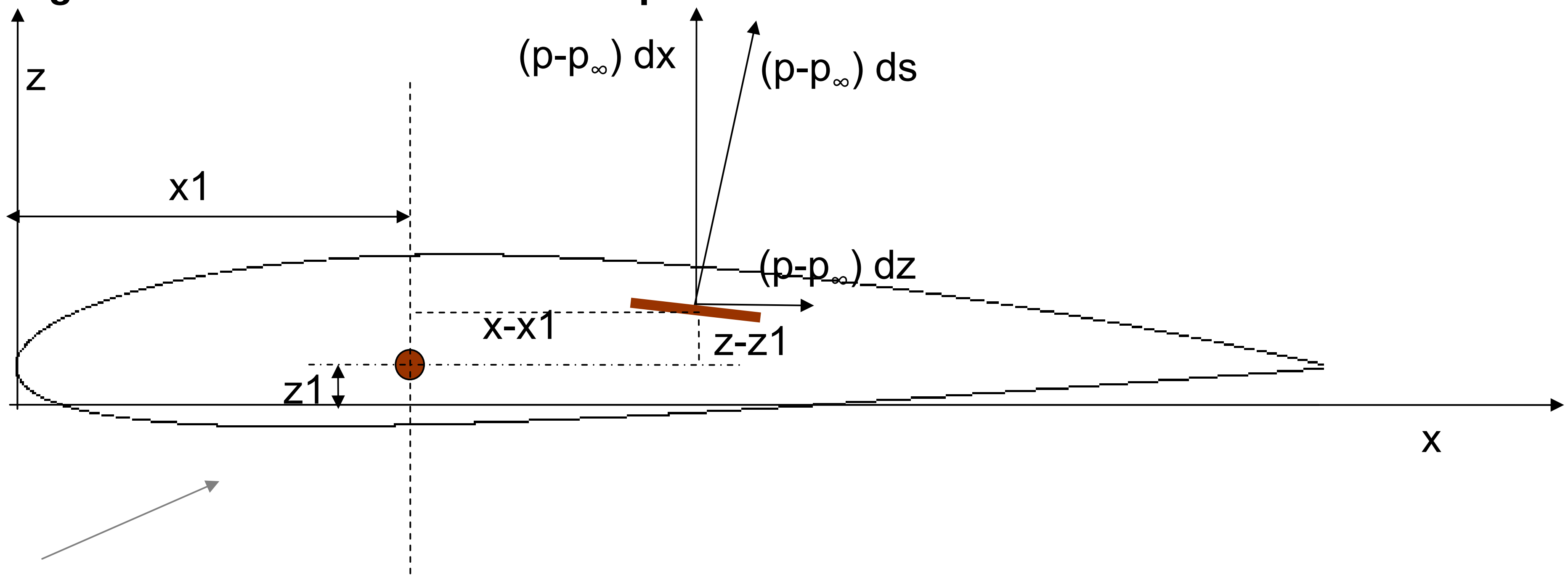
$$f_n = \int_0^s -(p-p_\infty)\cdot ds\cdot\cos\delta = \int_0^s (p-p_\infty)\cdot dx$$

$$f_t = \int_0^s -(p-p_\infty)\cdot ds\cdot\sin\delta = \int_0^s -(p-p_\infty)\cdot dz$$

$$C_n = \frac{f_n}{\frac{1}{2}\cdot\rho\cdot V_\infty^2\cdot c} = \int_0^s \frac{(p-p_\infty)}{\frac{1}{2}\cdot\rho\cdot V_\infty^2}\cdot\frac{dx}{c} = \int_0^s Cp\cdot d\overline{x}$$

$$C_t = \frac{f_t}{\frac{1}{2}\cdot\rho\cdot V_\infty^2\cdot c} = \int_0^s \frac{-(p-p_\infty)}{\frac{1}{2}\cdot\rho\cdot V_\infty^2}\cdot\frac{dz}{c} = \int_0^s -Cp\cdot d\overline{z}$$

Integrazione della distribuzione di pressione

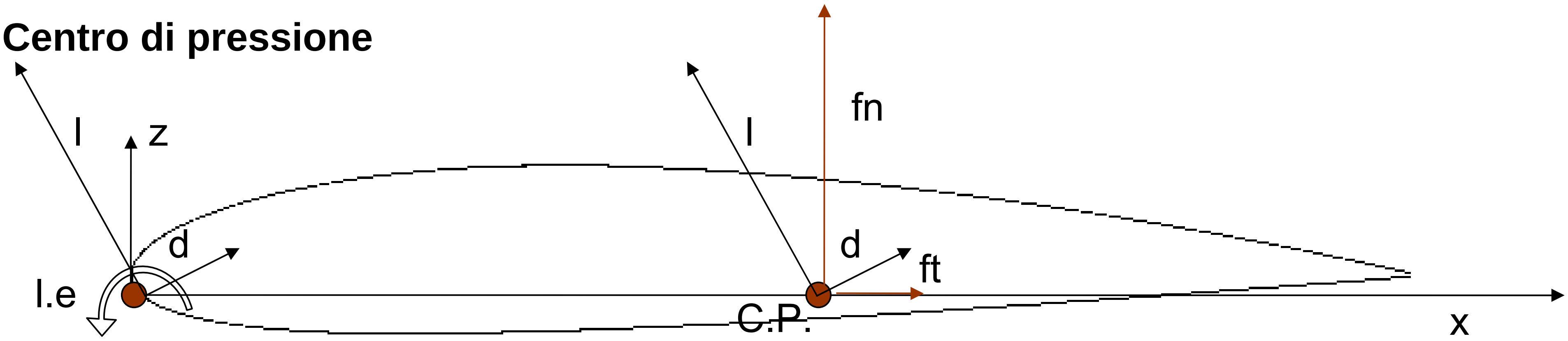


$$m = \int_0^s -(p-p_\infty)\cdot(x-x1)\cdot dx + \int_0^s -(p-p_\infty)\cdot(z-z1)dz$$

$$C_m = \frac{m}{\frac{1}{2}\cdot\rho\cdot V_\infty^{\ 2}\cdot c^2} = \int_0^s -\frac{(p-p_\infty)}{\frac{1}{2}\cdot\rho\cdot V_\infty^{\ 2}}\cdot\frac{(x-x1)}{c}\cdot\frac{dx}{c} + \int_0^s -\frac{(p-p_\infty)}{\frac{1}{2}\cdot\rho\cdot V_\infty^{\ 2}}\cdot\frac{(z-z1)}{c}\cdot\frac{dz}{c} =$$

$$= \boxed{\int_0^s -Cp(\overline{x}-\overline{x1})\cdot d\overline{x} + \int_0^s -Cp(\overline{z}-\overline{z1})\cdot d\overline{z}}$$

Centro di pressione



$$Cm_{CP}=\int_0^s -Cp\cdot(\bar{x}-\bar{x}_{CP})\cdot d\bar{x}+\int_0^s -Cp\cdot\bar{z}\cdot d\bar{z}=0$$

$$\underbrace{\int_0^s -Cp\cdot\bar{x}\cdot d\bar{x}+\int_0^s -Cp\cdot\bar{z}\cdot d\bar{z}}_{Cm_{l.e}}+\bar{x}_{CP}\cdot\underbrace{\int_0^s Cp\cdot d\bar{x}}_{C_n}=0$$

$$Cm_{l.e}=-Cn\cdot\bar{x}_{CP}\Rightarrow\bar{x}_{CP}=-\frac{Cm_{l.e}}{Cn}$$

Esercizio 3

- *Profilo Naca 2412*

$$Cl = 0.71$$

$$Cd = 0.0064$$

$$\alpha = 4 \deg$$

$$Cm_{l.e.} = -0.234$$

Calcolare la posizione del centro di pressione

$X_{C.P.} = 0.33$

- *Profilo Naca 2412*

$$Cl = 0.24$$

$$Cd = 0.0055$$

$$\alpha = 0 \deg$$

$$Cm_{l.e.} = -0.112$$

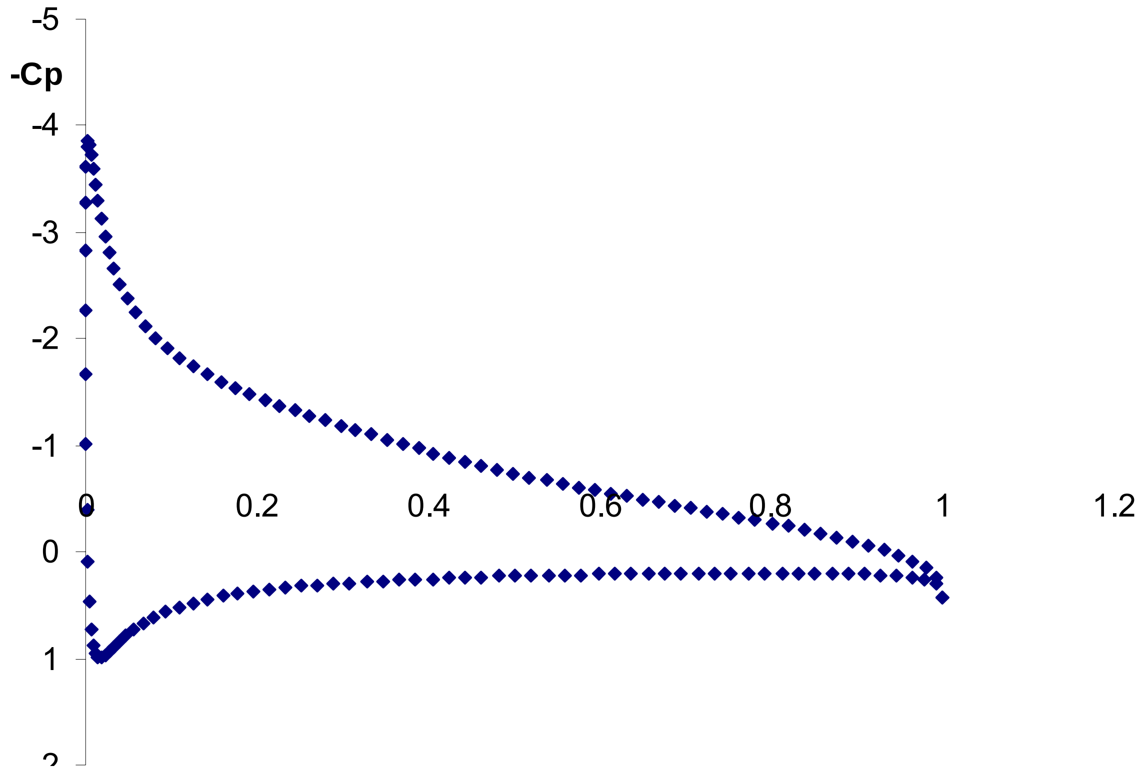
Calcolare la posizione del centro di pressione

$X_{C.P.} = 0.47$

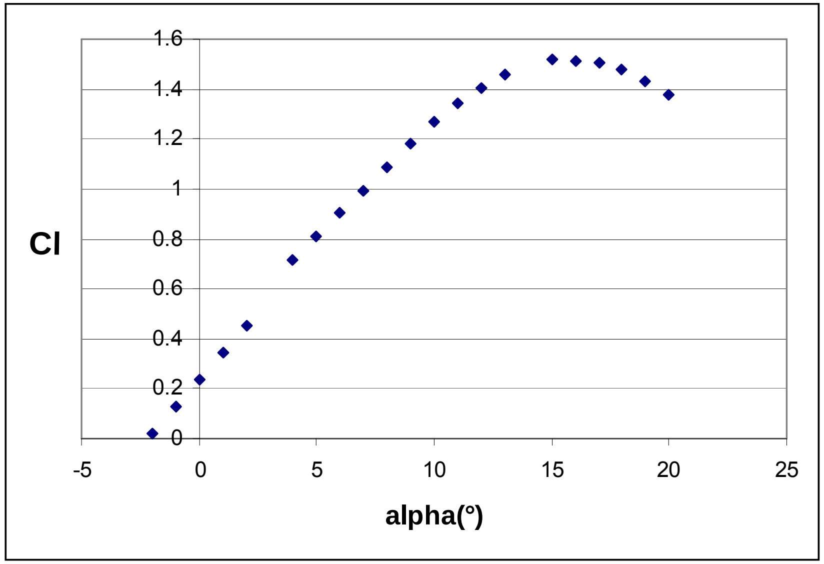
Integrazione numerica – Esempio Cp numerici o sperimentali

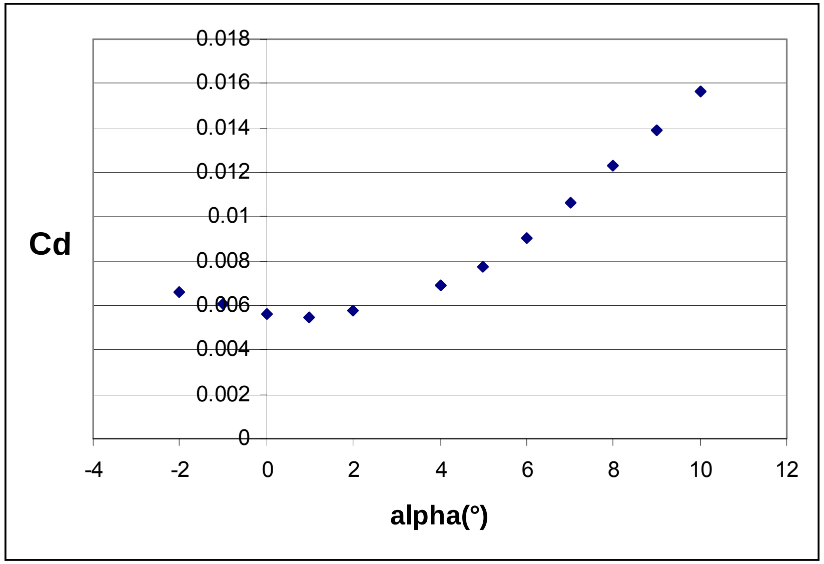
NACA 2412

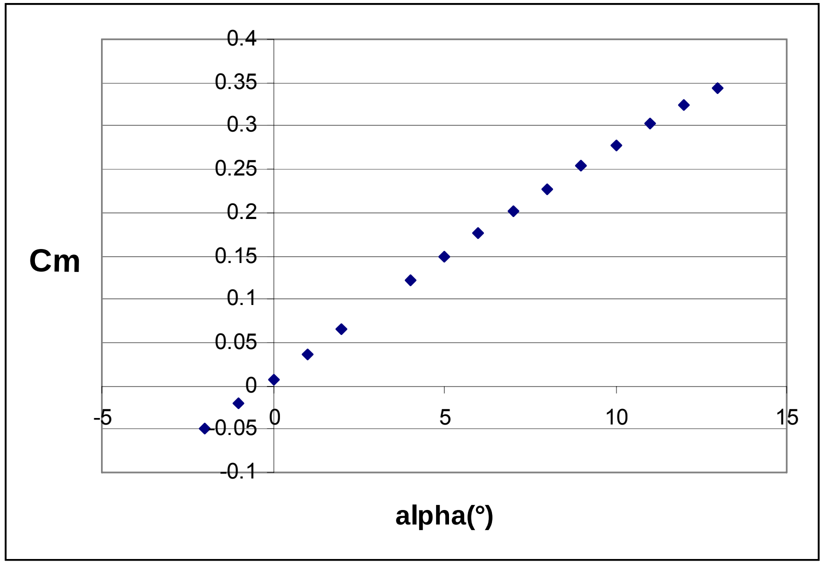
x	z	Cp
1	0.00126	0.43059
0.992863	0.00273	0.23665
0.980187	0.005305	0.15376
0.965251	0.008283	0.08598
0.948591	0.011532	0.03049
0.930885	0.014904	-0.01854
0.912638	0.018292	-0.06238
0.894131	0.021641	-0.10197
0.875499	0.024924	-0.13828
0.856802	0.028131	-0.17316
0.838069	0.031256	-0.20565
0.819311	0.0343	-0.23678
0.800534	0.037259	-0.26729
0.946892	-0.00509	0.22416
0.96423	-0.00386	0.23861
0.979694	-0.00274	0.26151
0.992716	-0.0018	0.30065
1	-0.00126	0.43059



Integrazione numerica – Esempio di polari numeriche o sperimentali







$$Cl_{\alpha}$$

$$Cd_{\alpha}$$

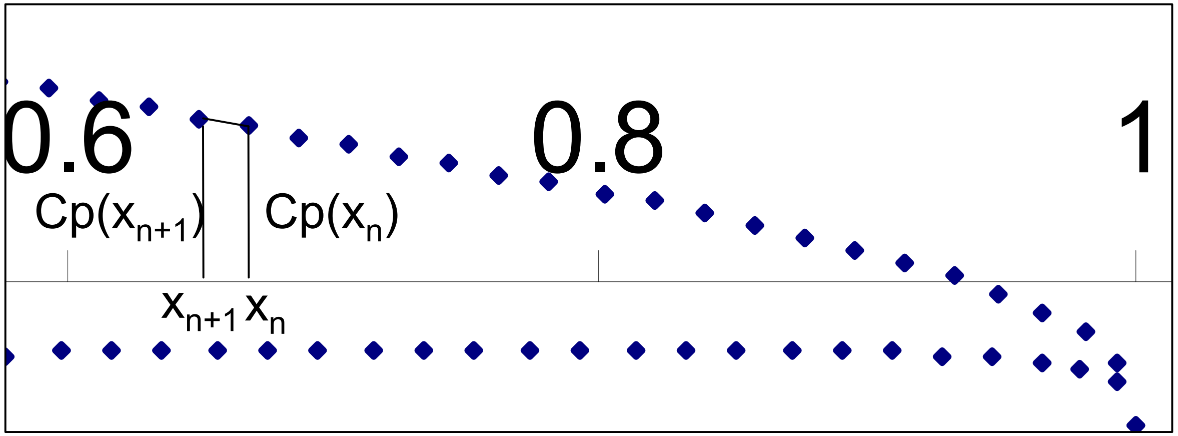
$$Cm_{\alpha}$$

Integrazione numerica – Derivate dei coefficienti aerodinamici

alpha	CL	CD	CM
0	0.2394	0.00551	-0.0524
2	0.4531	0.00528	-0.0495
4	0.7104	0.00638	-0.0566

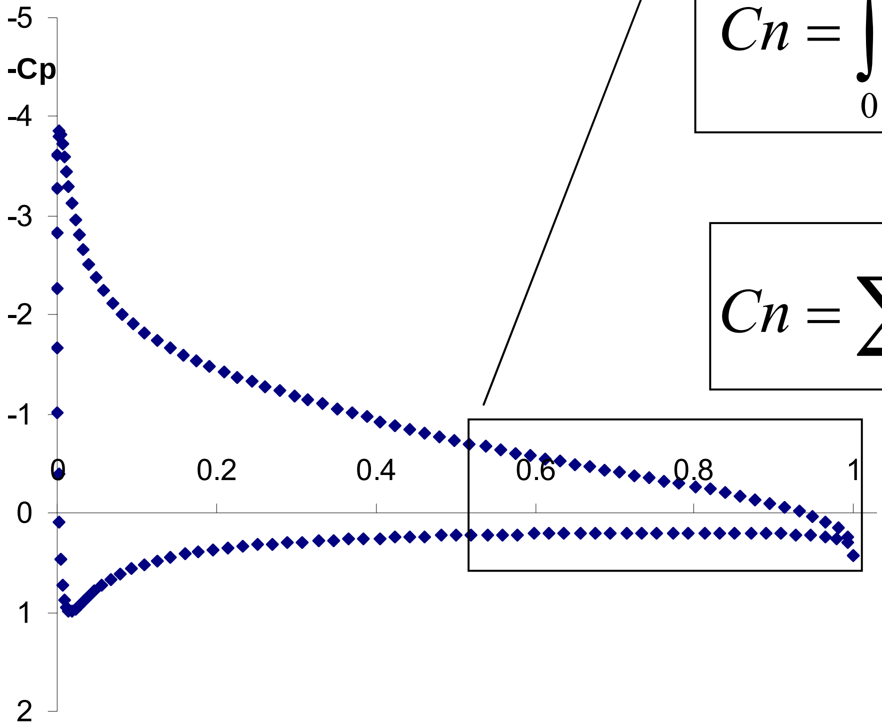
Integrazione numerica - Cn

x	z	Cp
0.687652	0.053233	-0.43788
0.668821	0.055589	-0.46589
0.64999	0.057854	-0.49401
0.631162	0.060026	-0.52243
0.612339	0.062103	-0.55114
0.593522	0.064082	-0.58029



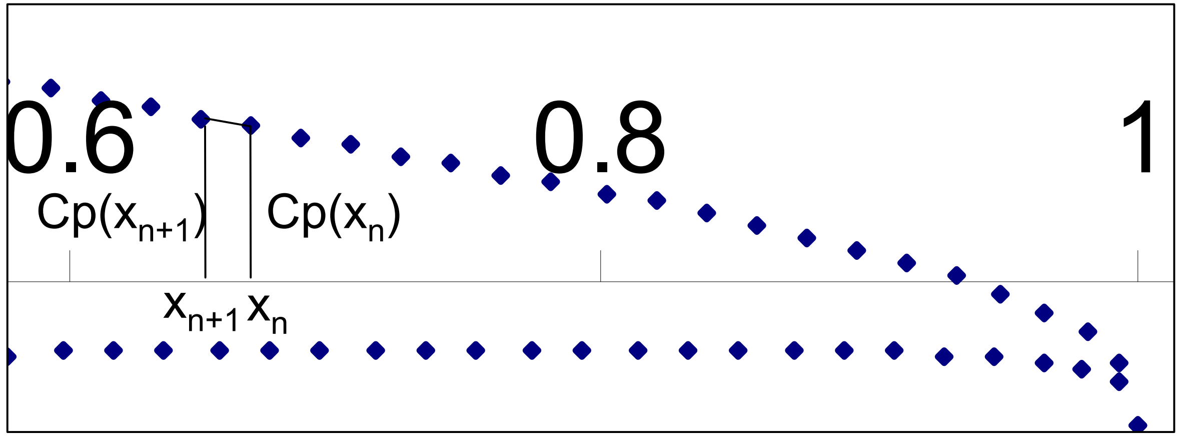
$$Cn = \int_0^s Cp \cdot d\bar{x}$$

$$Cn = \sum_1^{n-1} \frac{Cp(x_{n+1}) + Cp(x_n)}{2} \cdot (x(n+1) - x(n))$$



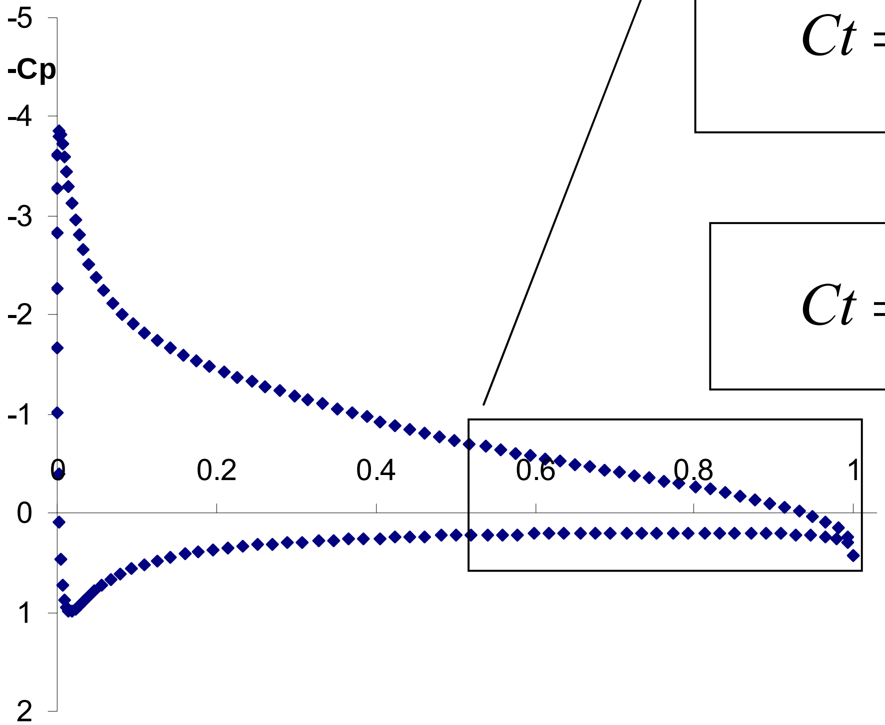
Integrazione numerica - Ct

x	z	Cp
0.687652	0.053233	-0.43788
0.668821	0.055589	-0.46589
0.64999	0.057854	-0.49401
0.631162	0.060026	-0.52243
0.612339	0.062103	-0.55114
0.593522	0.064082	-0.58029

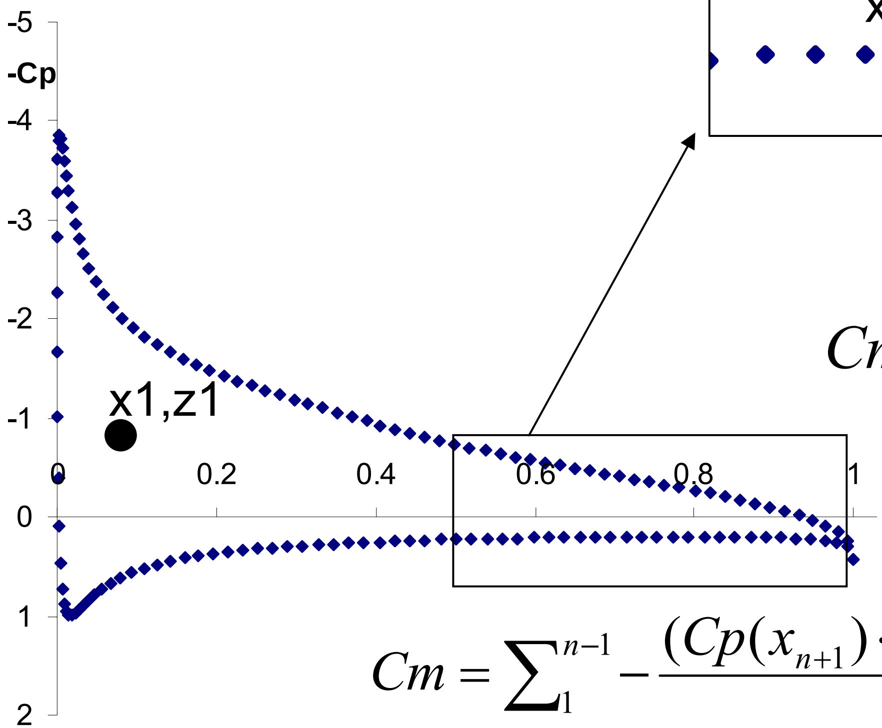


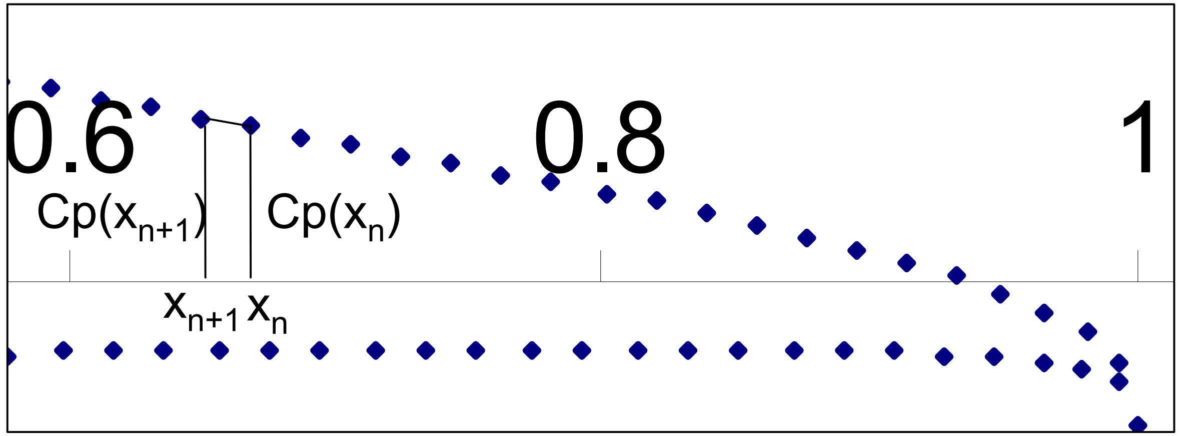
$$Ct = \int_0^s -Cp \cdot d\bar{z}$$

$$Ct = \sum_1^{n-1} -\frac{Cp(z_{n+1}) + Cp(z_n)}{2} \cdot (z_{n+1} - z_n)$$



Integrazione numerica - Cm



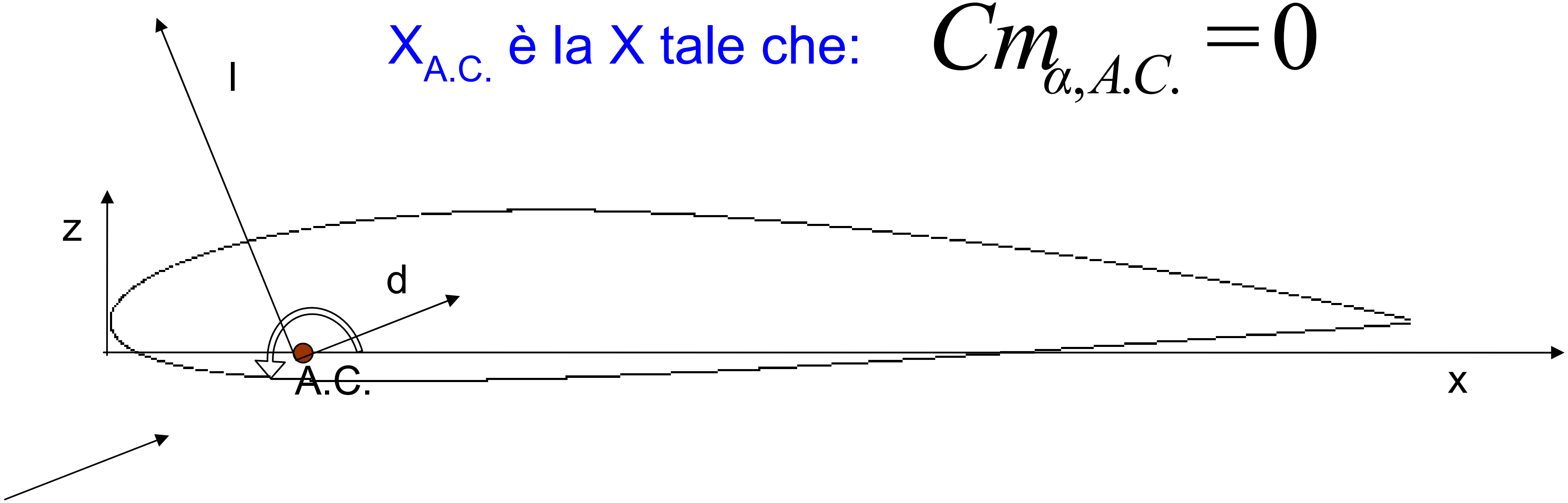


$$Cm = \int_0^s Cp(\overline{x} - \overline{x1}) \cdot d\,\overline{x} + \int_0^s -Cp(\overline{z} - \overline{z1}) \cdot d\,\overline{z}$$

$$Cm = \sum_1^{n-1} - \frac{(Cp(x_{n+1}) \cdot (x_{n+1} - x1) + Cp(x_n) \cdot (x_n - x1))}{2} \cdot (x_{n+1} - x_n) -$$

$$+ \frac{(Cp(z_{n+1}) \cdot (z_{n+1} - z1) + Cp(z_n) \cdot (z_n - z1))}{2} \cdot (z_{n+1} - z_n)$$

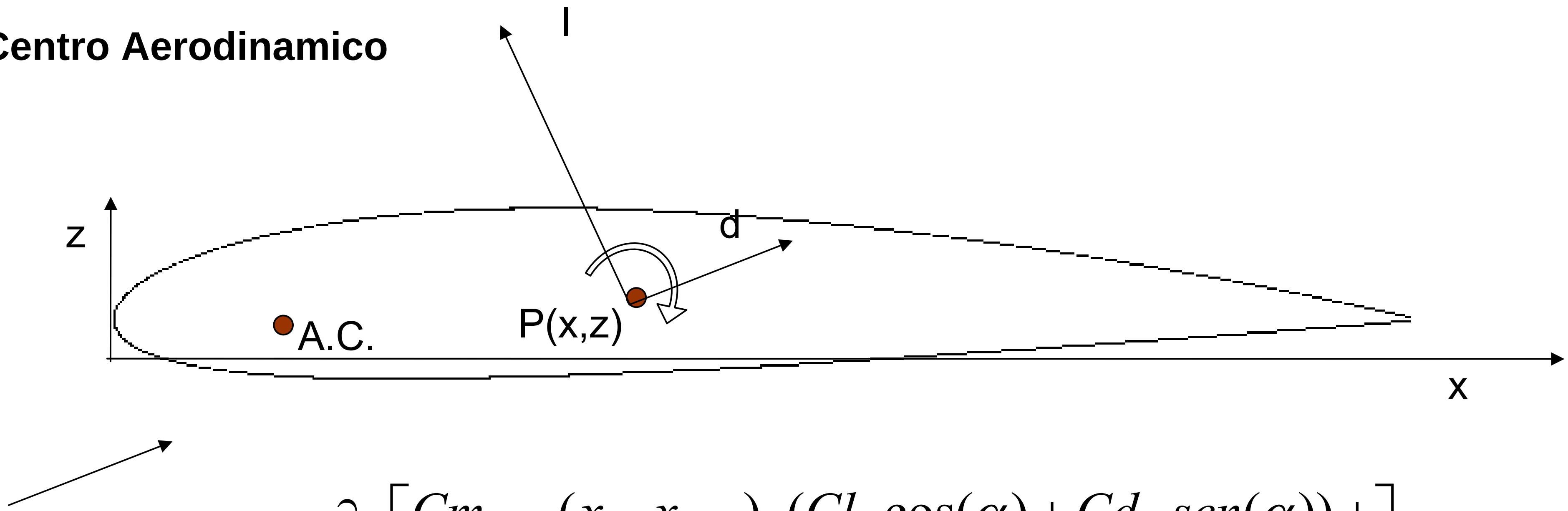
Centro Aerodinamico - Definizione



$X_{A.C.}$ è la X tale che: $$Cm_{\alpha,A.C.} = 0$$

Solitamente per profili con curvatura positiva: $$Cm_0 \leq 0$$

Centro Aerodinamico



$$Cm_{\alpha\,A.C.} = \frac{\partial}{\partial \alpha}\left[\begin{array}{l} Cm_P - (x - x_{A.C.})\cdot(Cl\cdot\cos(\alpha) + Cd\cdot sen(\alpha)) + \\ + (z - z_{A.C.})\cdot(-Cl\cdot sen(\alpha) + Cd\cdot\cos(\alpha)) \end{array}\right] = 0$$

$$\left[\begin{array}{l} -(x - x_{A.C.})\cdot(Cl_\alpha\cdot\cos(\alpha) + Cd_\alpha\cdot sen(\alpha)) + \\ + (z - z_{A.C.})\cdot(-Cl_\alpha\cdot sen(\alpha) + Cd_\alpha\cdot\cos(\alpha)) + \\ + (x - x_{A.C.})\cdot(Cl\cdot sen(\alpha) - Cd\cdot\cos(\alpha)) + \\ -(z - z_{A.C.})\cdot(Cl\cdot\cos(\alpha) + Cd\cdot sen(\alpha)) \end{array}\right] = -Cm_{\alpha\,P}$$

Centro Aerodinamico

$$\left.\begin{array}{l} Z_{A.C.} \\ X_{A.C.} \end{array}\right\}$$ Due incognite ⟶ Due equazioni ⟶ Due **α (α1, α2)**

$$\begin{cases} \begin{bmatrix} -(x - x_{A.C.}) \cdot (Cl_\alpha(\alpha 1) \cdot \cos(\alpha 1) + Cd_\alpha(\alpha 1) \cdot sen(\alpha 1)) + \\ +(z - z_{A.C.}) \cdot (-Cl_\alpha(\alpha 1) \cdot sen(\alpha 1) + Cd_\alpha(\alpha 1) \cdot \cos(\alpha 1)) + \\ +(x - x_{A.C.}) \cdot (Cl(\alpha 1) \cdot sen(\alpha 1) - Cd(\alpha 1) \cdot \cos(\alpha 1)) + \\ -(z - z_{A.C.}) \cdot (Cl(\alpha 1) \cdot \cos(\alpha 1) + Cd(\alpha 1) \cdot sen(\alpha 1)) \end{bmatrix} = -Cm_\alpha(\alpha 1)_P \\ \begin{bmatrix} -(x - x_{A.C.}) \cdot (Cl_\alpha(\alpha 2) \cdot \cos(\alpha 2) + Cd_\alpha(\alpha 2) \cdot sen(\alpha 2)) + \\ +(z - z_{A.C.}) \cdot (-Cl_\alpha(\alpha 2) \cdot sen(\alpha 2) + Cd_\alpha(\alpha 2) \cdot \cos(\alpha 2)) + \\ +(x - x_{A.C.}) \cdot (Cl(\alpha 2) \cdot sen(\alpha 2) - Cd(\alpha 2) \cdot \cos(\alpha 2)) + \\ -(z - z_{A.C.}) \cdot (Cl(\alpha 2) \cdot \cos(\alpha 2) + Cd(\alpha 2) \cdot sen(\alpha 2)) \end{bmatrix} = -Cm_\alpha(\alpha 2)_P \end{cases}$$