



To get this plot (plot.ps):  
`$ xfoil`  
`XFOIL c> naca 0012`  
`XFOIL c> oper`  
`.OPERi c> re 4.18E8`  
`.OPERi c> mach 0.289`  
`.OPERi c> visc`  
`.OPERi c> alfa 6`  
`.OPERi c> hard`

XFOIL v6.97 plot of Cp vs. x Explanation of terms.  
 \*\*\* This is a study aid I have developed for myself using a cruising B-17 as an example. Please send any corrections/comments to: kord@koberco.com

B-17 cruising assumptions:  
 Wing root airfoil NACA 0012  
 Wing span (l) = 36.1m (103'9")  
 Wing Area (S) = 132m(sq) (1402 ft(sq))  
 Velocity (V) = 325kph (220mph)  
 Altitude = 7625m (25,000')  
 Weight = 22,475 kg (49,500lb)

XFOIL  
 V 6.94

-2.0  
-1.5  
Cp  
-1.0  
-0.5  
0.0  
0.5  
1.0

Mach number ( $V/a$  = velocity/speed of sound):  $a_0 = 760$ mph, B747 cruise 0.83 = 560mph = 900 kph = 258 fps at sea level

Reynolds Number (viscosity/(density\*V\*l)): ratio of internal & viscous forces. Higher RE closer to inviscous flow. B747 cruise 1E8

Lift Coefficient (Lift/(1/2\*density\*velocity(sq)\*wing area): 747 cruise 0.5, Cessna 150 cruise 0.35

Moment Coefficient (moment/(1/2\*density\*velocity(sq)\*wing area): longitudinal stability, trim

Drag Coefficient (Drag/(1/2\*density\*velocity(sq)\*wing area): 747 cruise 0.03, Cessna 150 cruise 0.03

Glide Ratio ( == Fitness == Glide Number = Lift/Drag): budgeriar 4.5, albatross 20, 747 cruise 15-18, sailplane 40

NACA 0012

$Ma = 0.2890$

$Re = 418.0 \times 10^6$

$\alpha = 6.0000^\circ$

$C_L = 0.7447$

$C_M = -0.0028$

$C_D = 0.00502$

$L/D = 148.50$

$N_{cr} = 9.00$

Angle of attack (alpha): angle between chord line and freestream velocity 6 degrees good cruise est.

for airfoil upper surface

Pressure Coefficient Cp  
 $(dp/(1/2*density*velocity^2)) = dp/q$ , where q=freestream dynamic pressure):  
 (-)Cp for upper surface adds to lift  
 (+) Cp for lower surface adds to lift  
 $Cp(lower)-Cp(upper) \sim lift$

for airfoil lower surface

Transition criteria ( $N_{cr}$  = free/forced): sailplane 12-14, clean/dirty wind tunnel 12-14/4-8

References: Fundamentals of Flight, 2nd ed. 1989. R. S. Shevell.; Introduction to Flight, 4th ed. 1999. J.D. Anderson; The Simple Science of Flight, 2000. H. Tennekes; XFOIL v.6.94 <http://raphael.mit.edu/xfoil>

----- Cp inviscous distribution  
 ——— Cp viscous distribution

$Cp(max) = 1.0$ .  
 Stagnation Point. Pt at  $V = 0$  (inviscous fluid)

Boundary layer: raises effective trailing edge of airfoil reducing effective alpha (reducing lift?)