

UNIVERSITY OF ILLINOIS UI-1720 AIRFOIL

Max thickness 13.83% at 17.9% of the chord

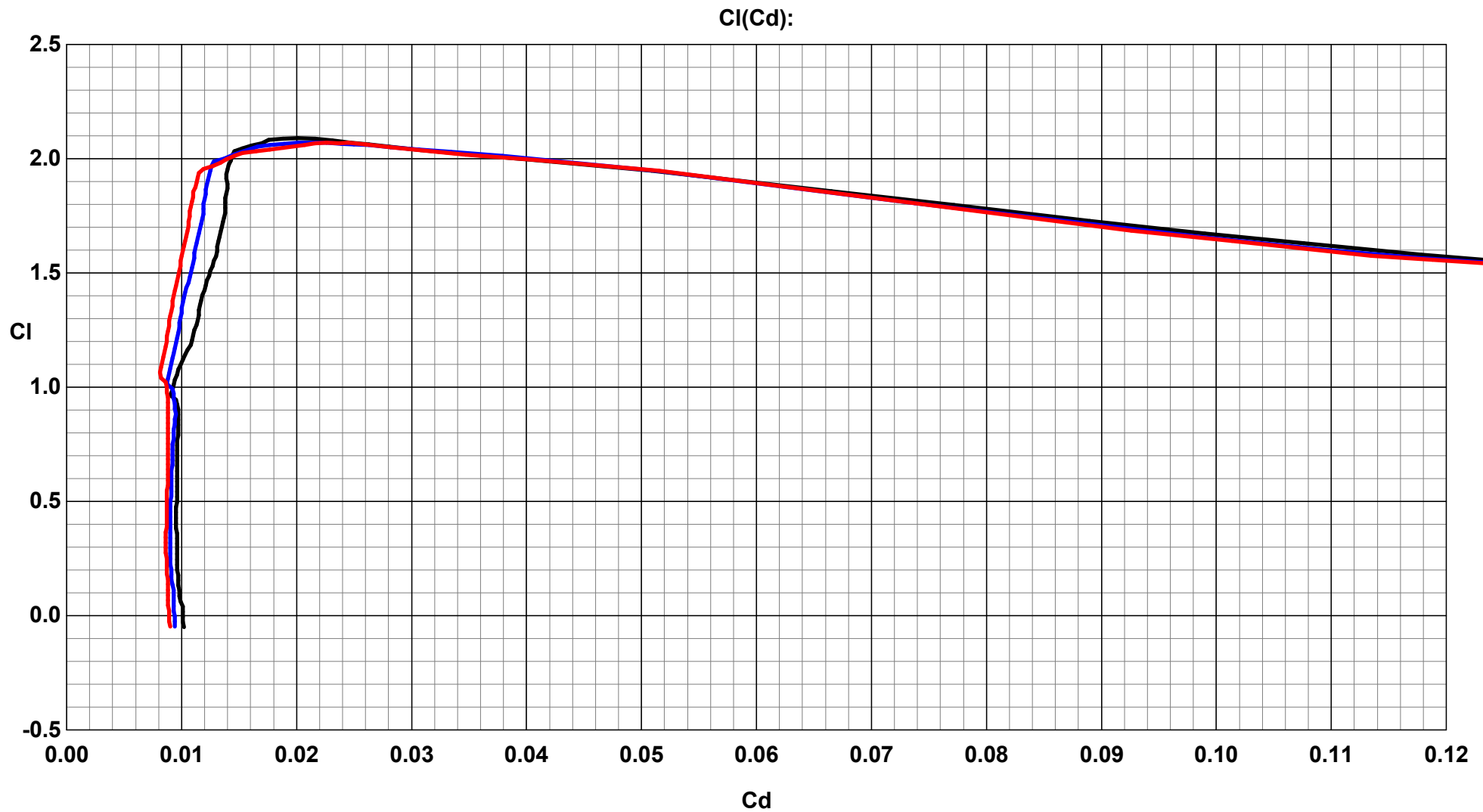
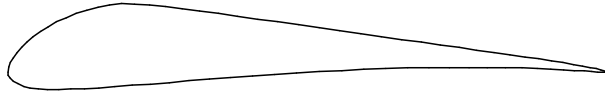
Max camber 4.63% at 23.8% of the chord

Mach = 0.0000 - NCrit = 6.00

Re 1500000 =

Re 2000000 =

Re 2500000 =



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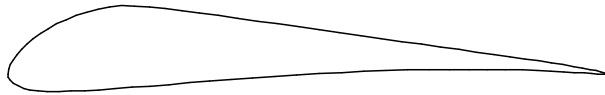
Max camber 4.63% at 23.8% of the chord

Mach = 0.0000 - NCrit = 6.00

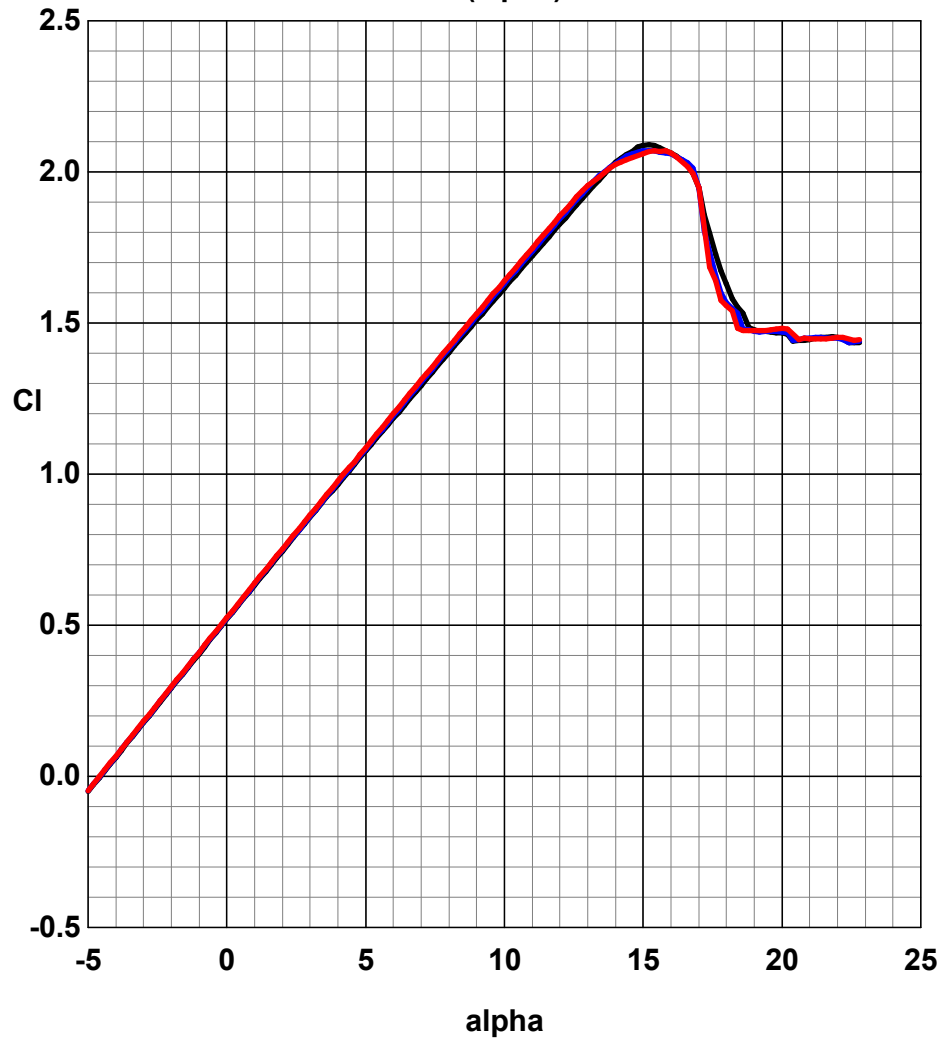
Re 1500000 =

Re 2000000 =

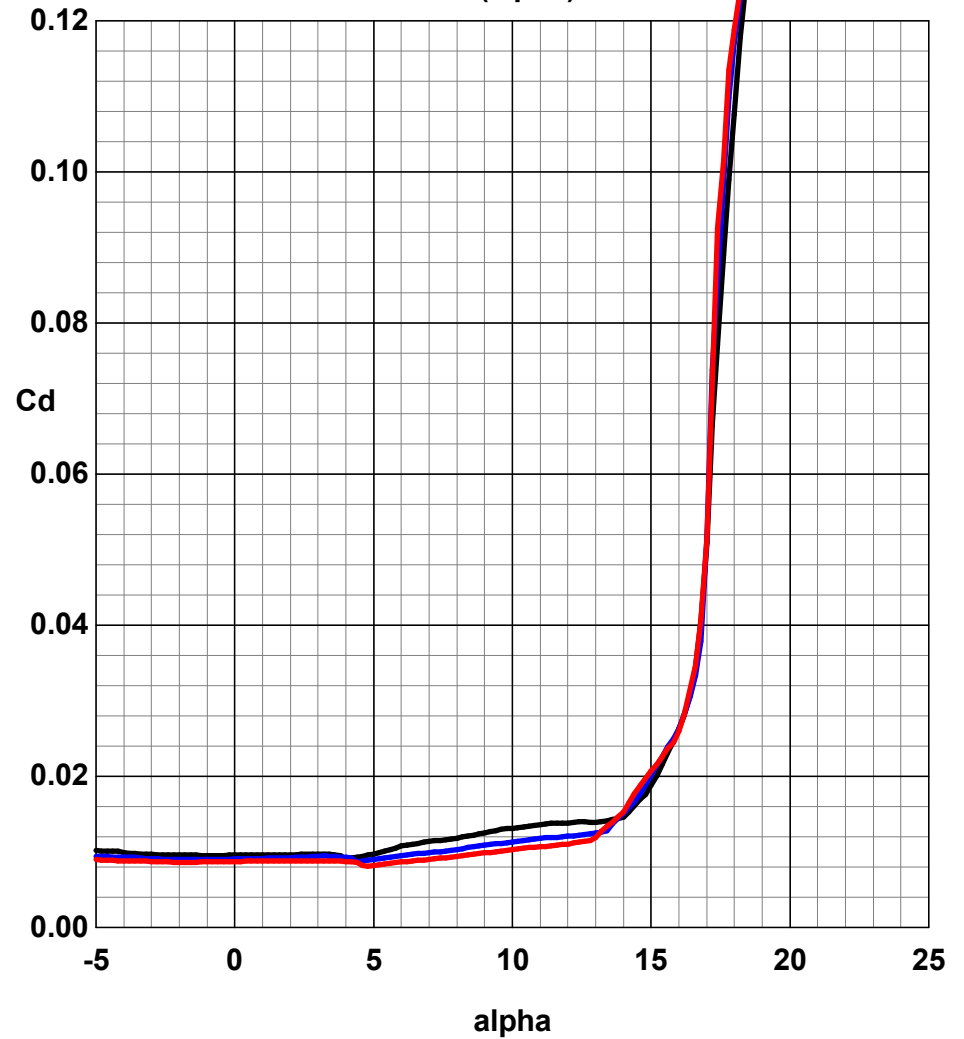
Re 2500000 =



Cl(alpha):



Cd(alpha):



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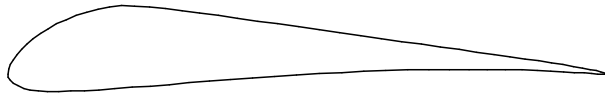
Max camber 4.63% at 23.8% of the chord

Mach = 0.0000 - NCrit = 6.00

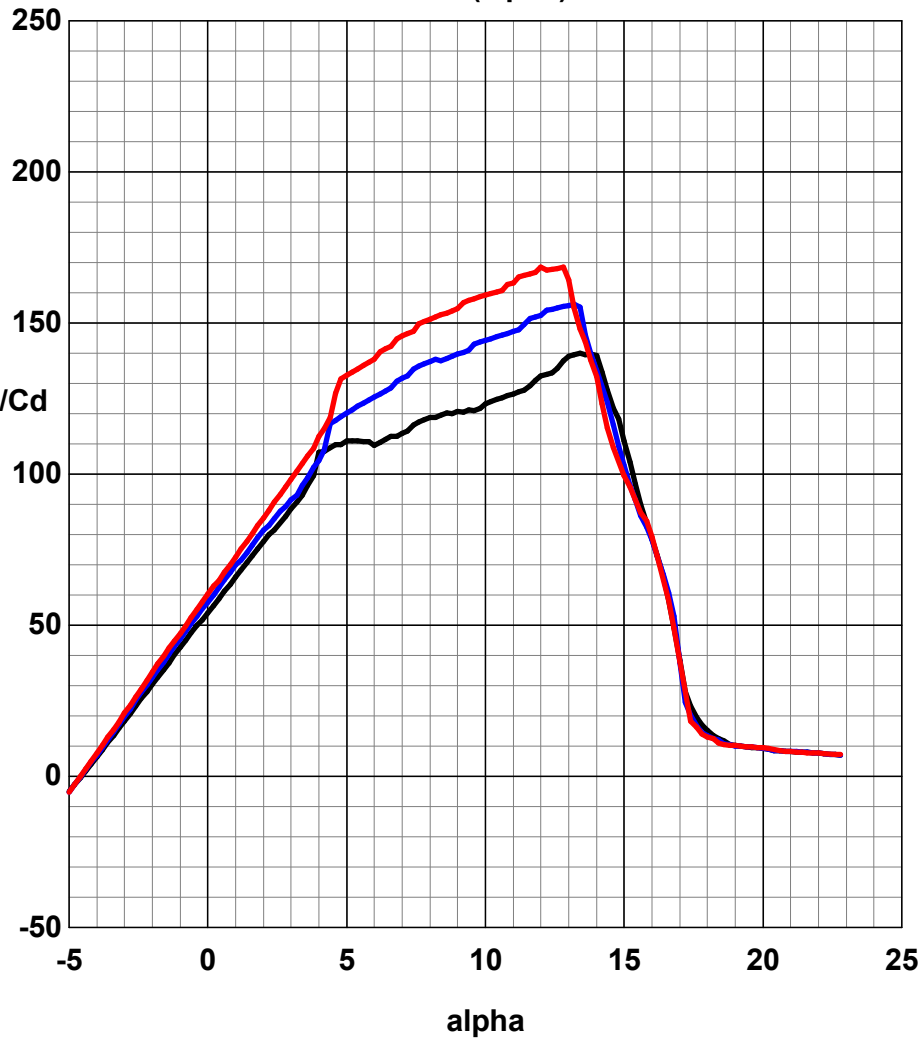
Re 1500000 =

Re 2000000 =

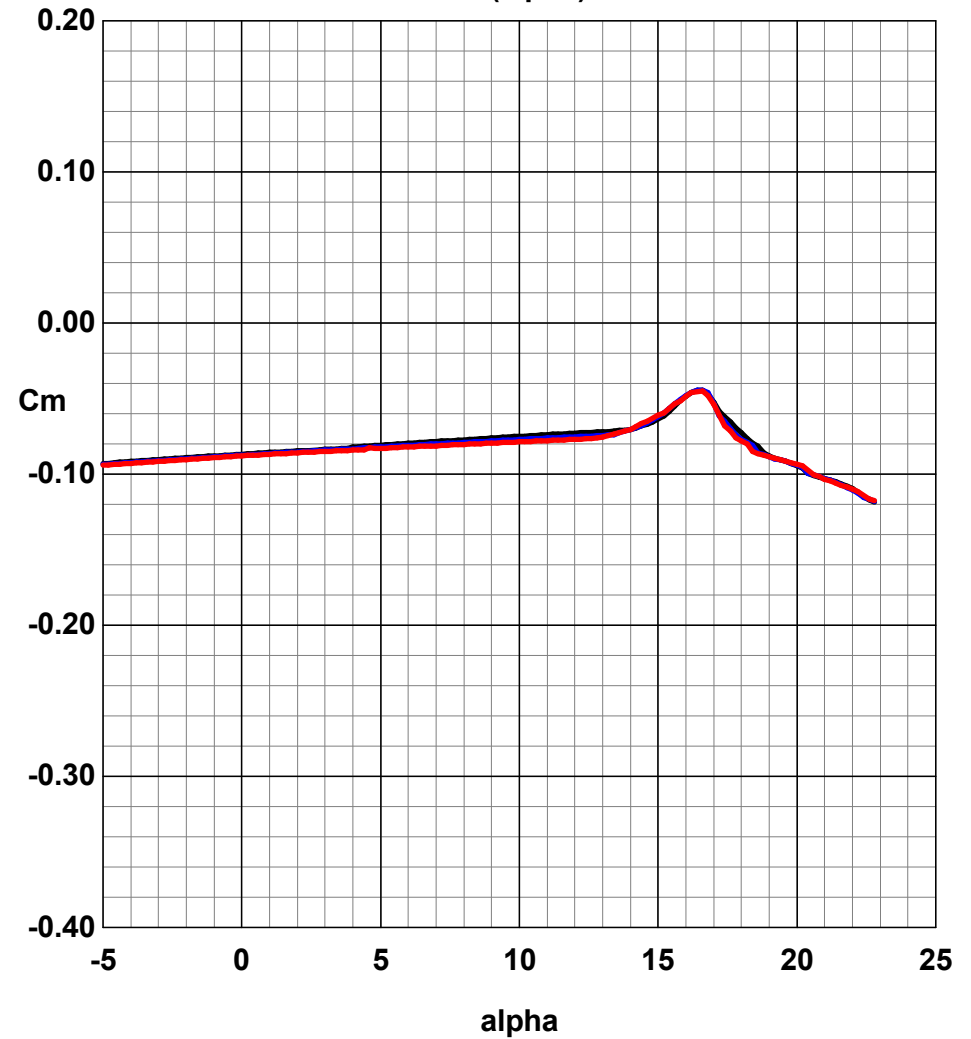
Re 2500000 =



Cl/Cd(alpha):



Cm(alpha):



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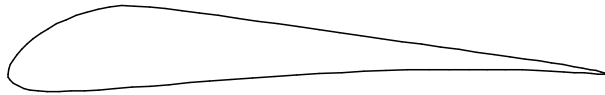
Max camber 4.63% at 23.8% of the chord

Mach = 0.0000 - NCrit = 6.00

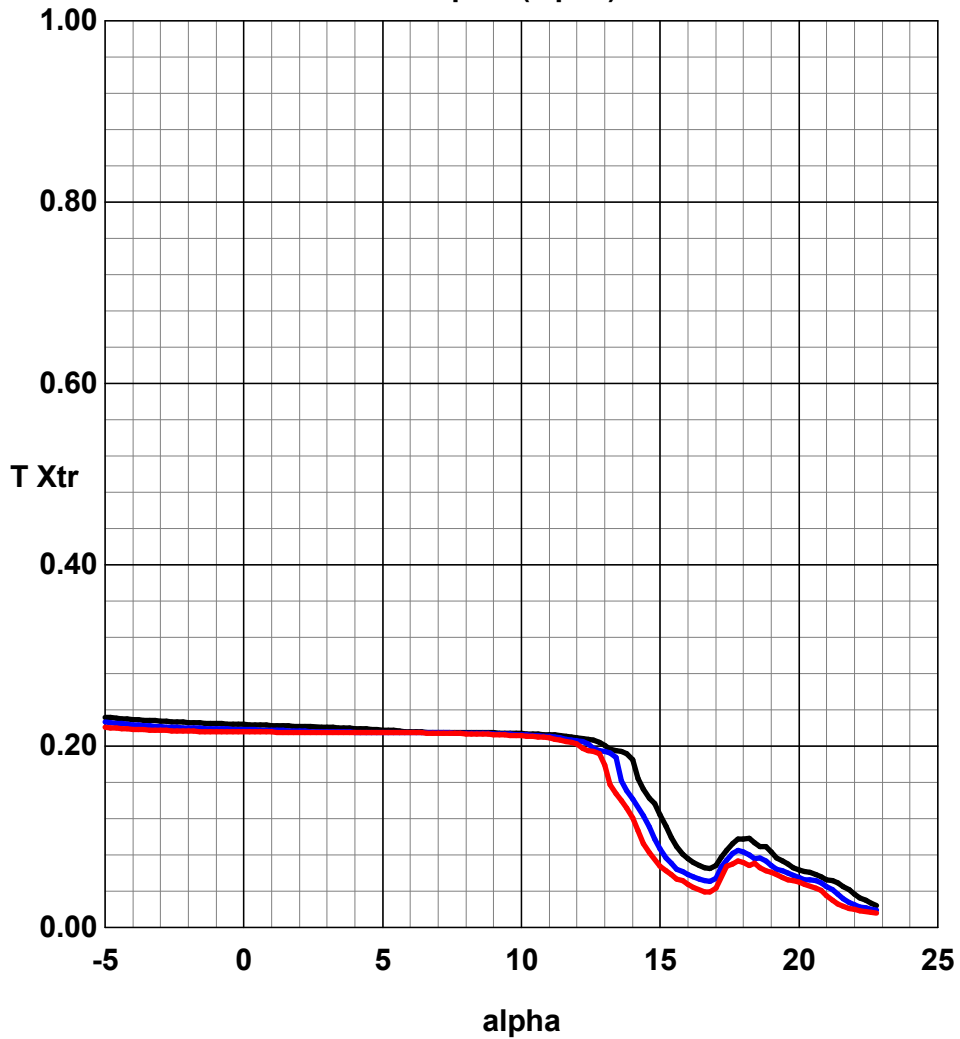
Re 1500000 =

Re 2000000 =

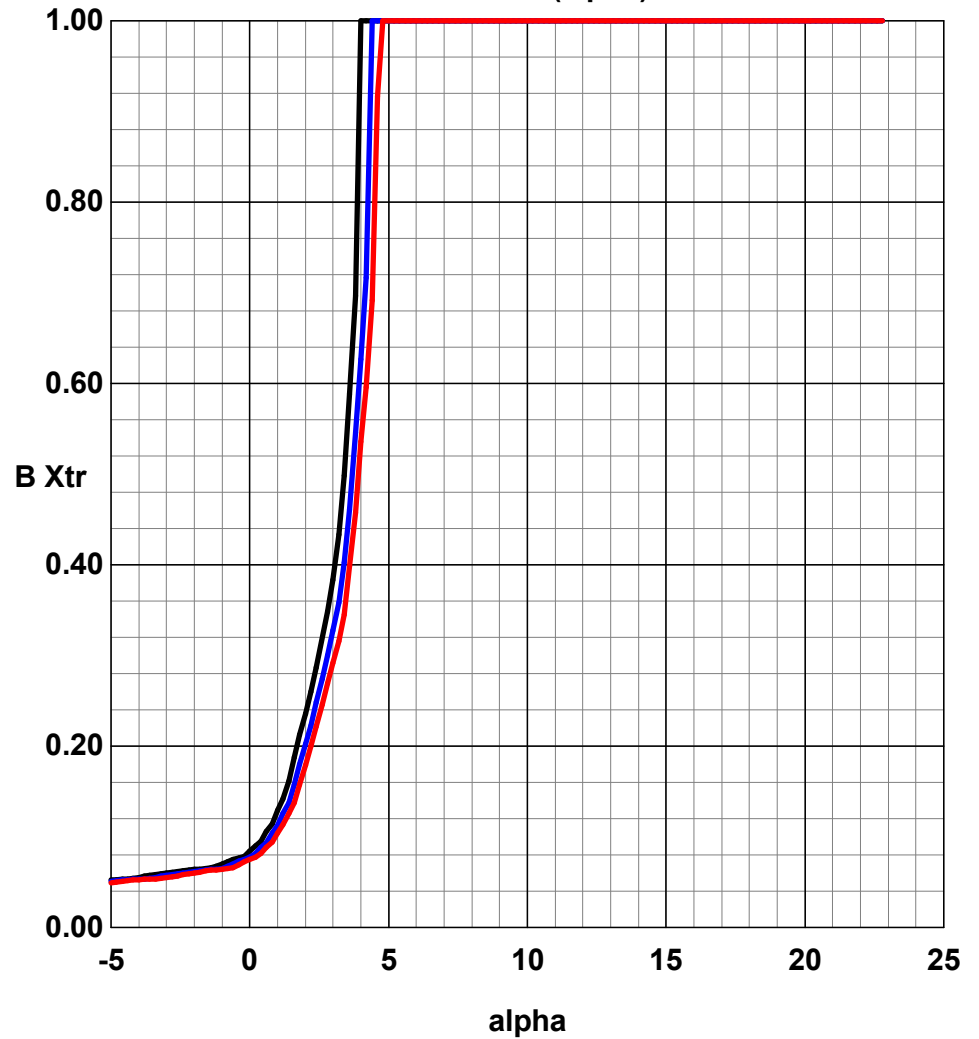
Re 2500000 =



Top Xtr(alpha):



Bottom Xtr(alpha):



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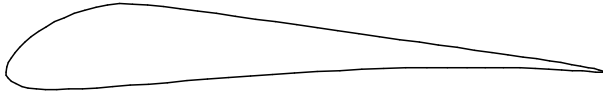
Max camber 4.63% at 23.8% of the chord

Mach = 0.0000 - NCrit = 6.00

Re 1500000 =

Re 2000000 =

Re 2500000 =



Power Factor(alpha) = $Cl^{1.5}/Cd(alpha)$:

