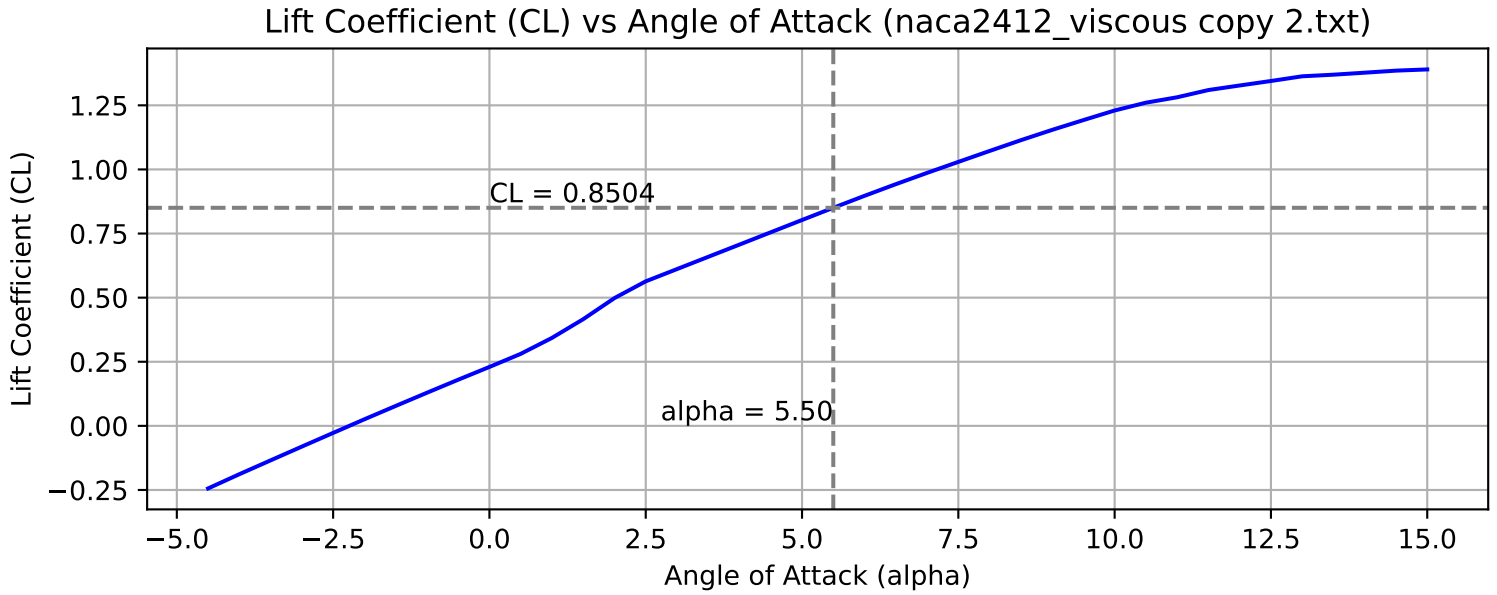
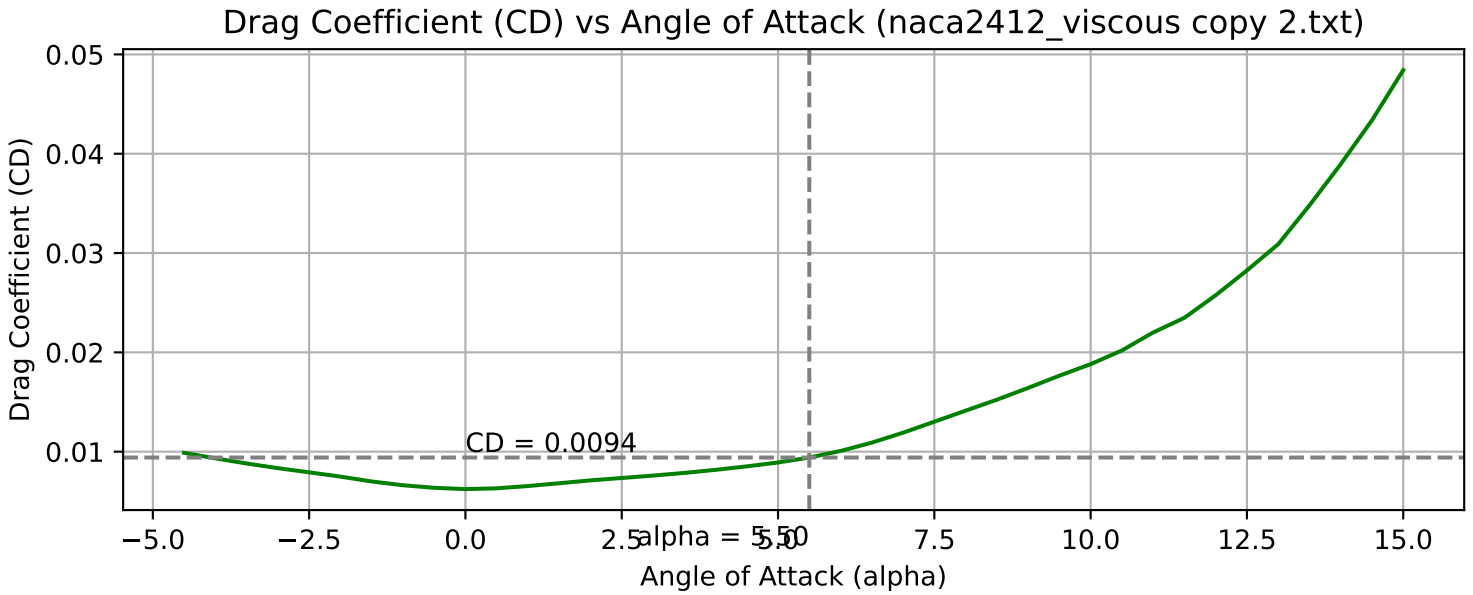
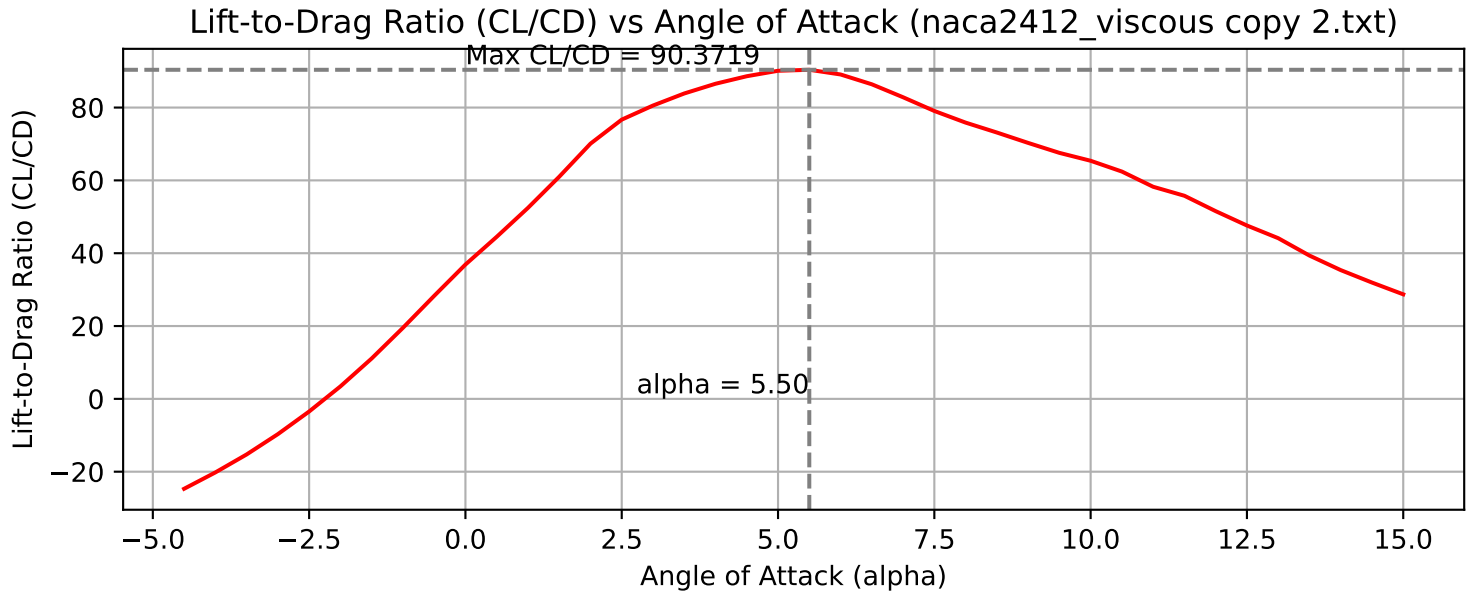
Airfoil Analysis Report

NACA 2412

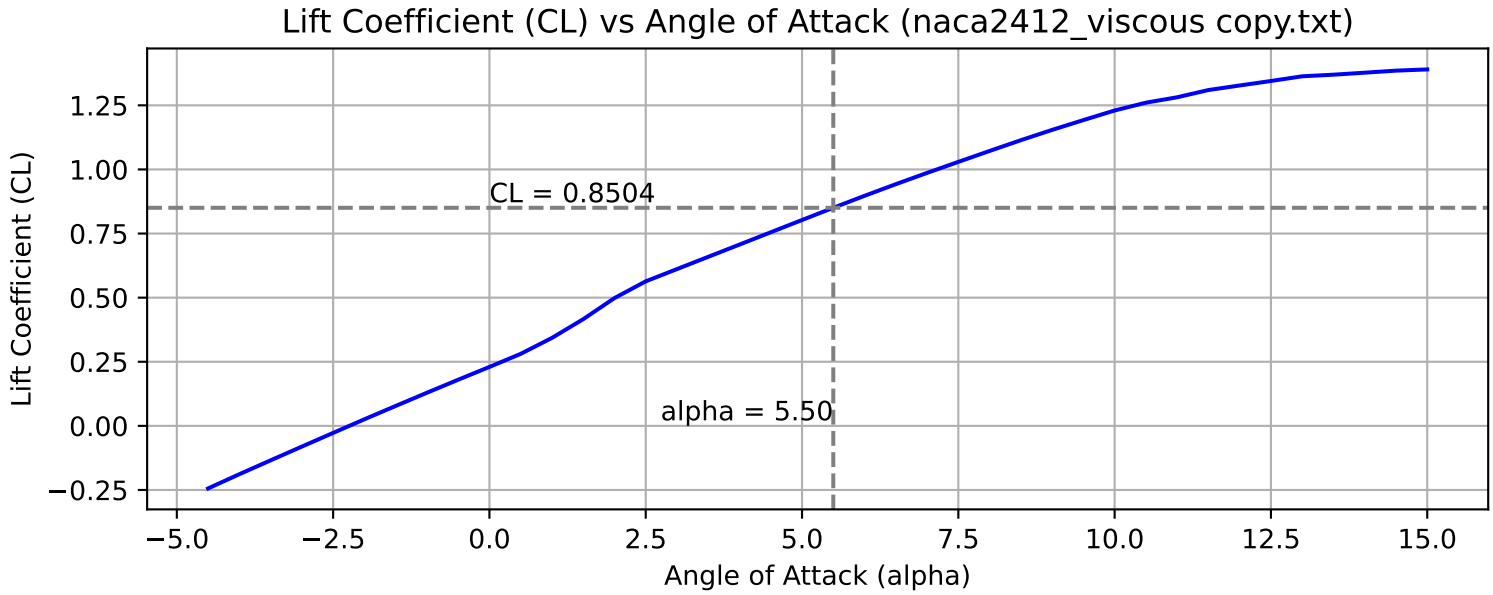


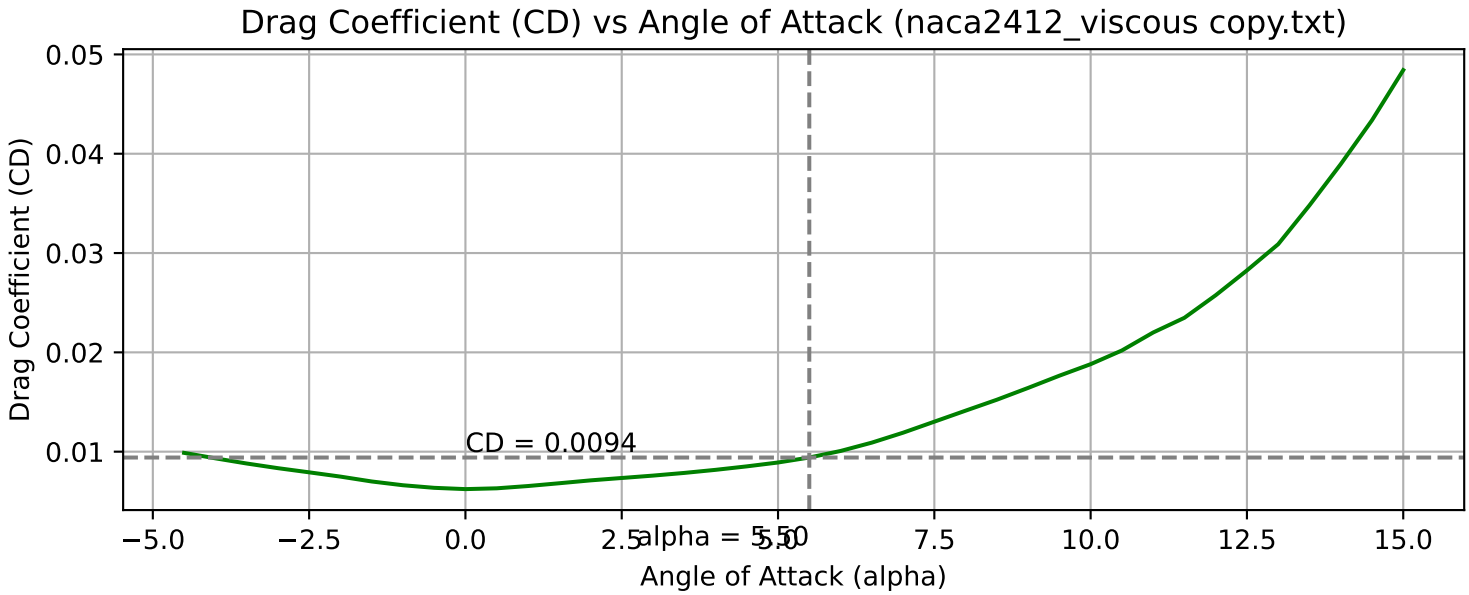


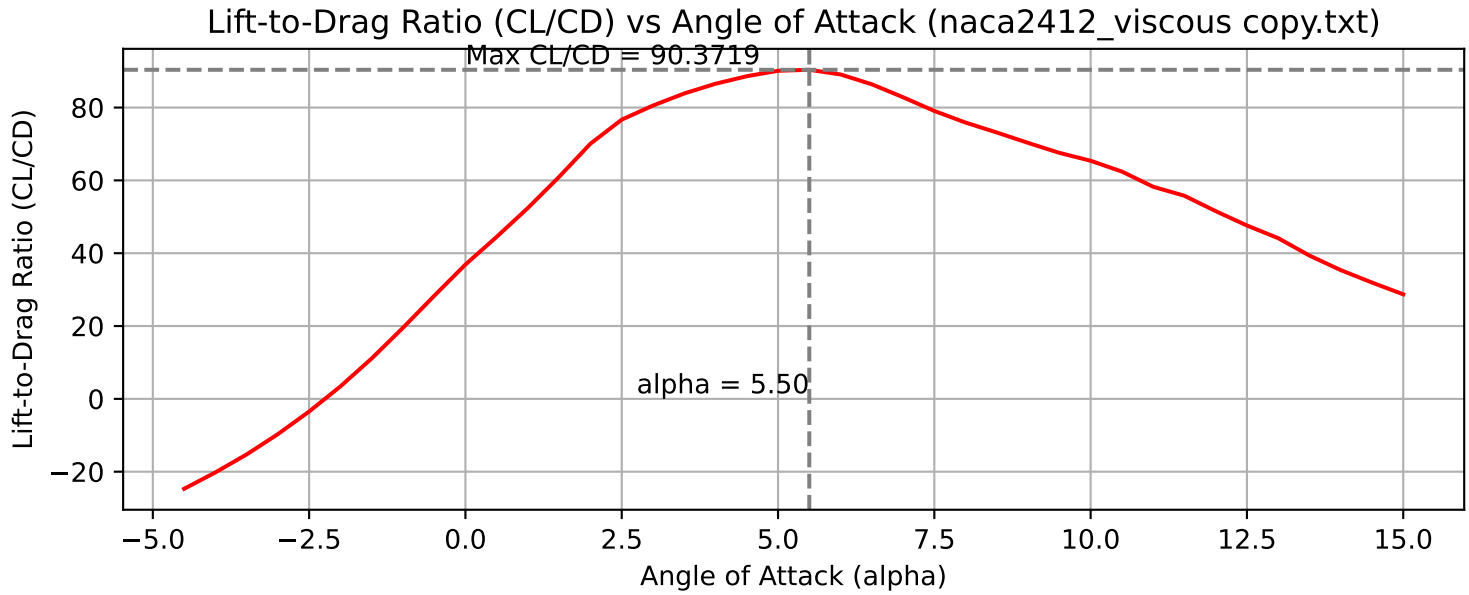


At alpha = 5.50: CL = 0.8504, CD = 0.0094, Max CL/CD = 90.3719

NACA 2412







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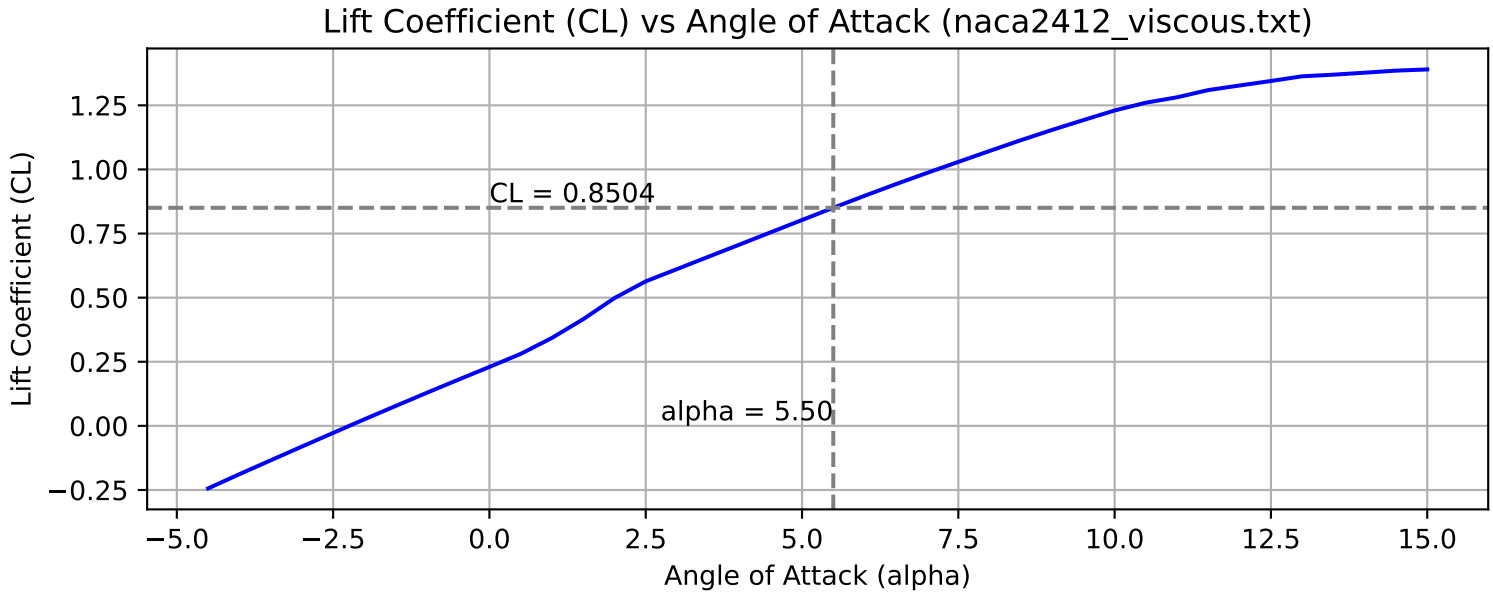

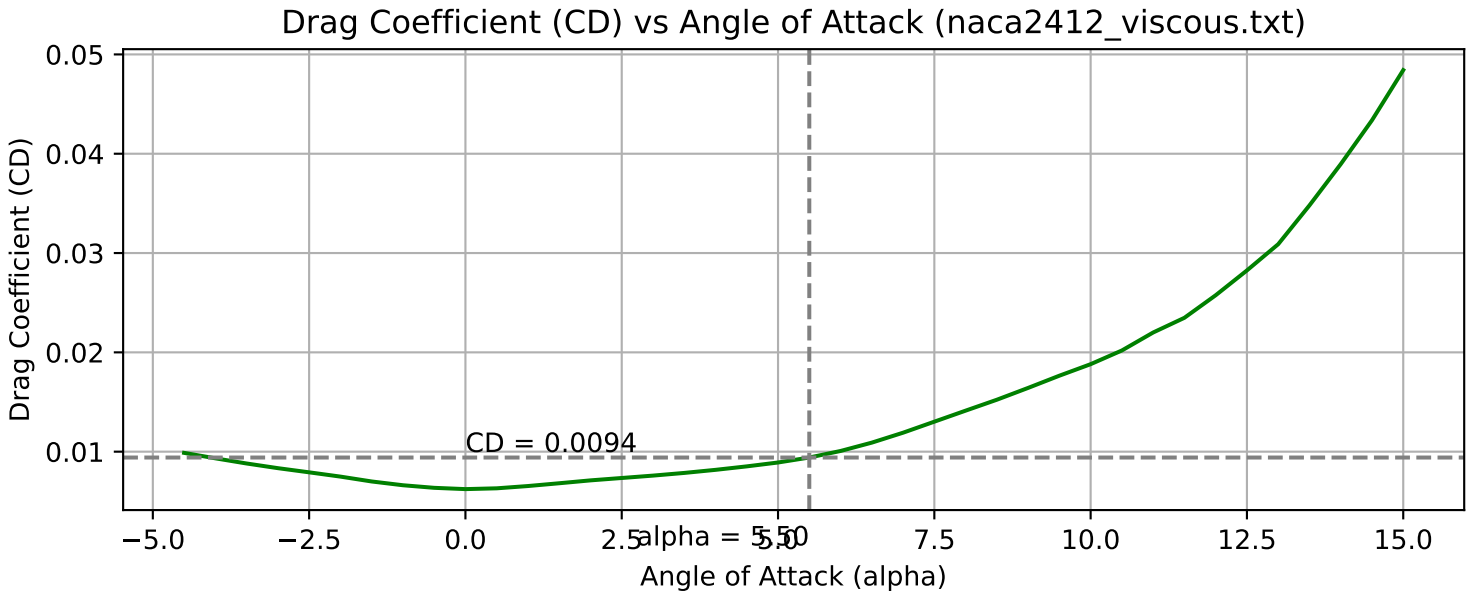

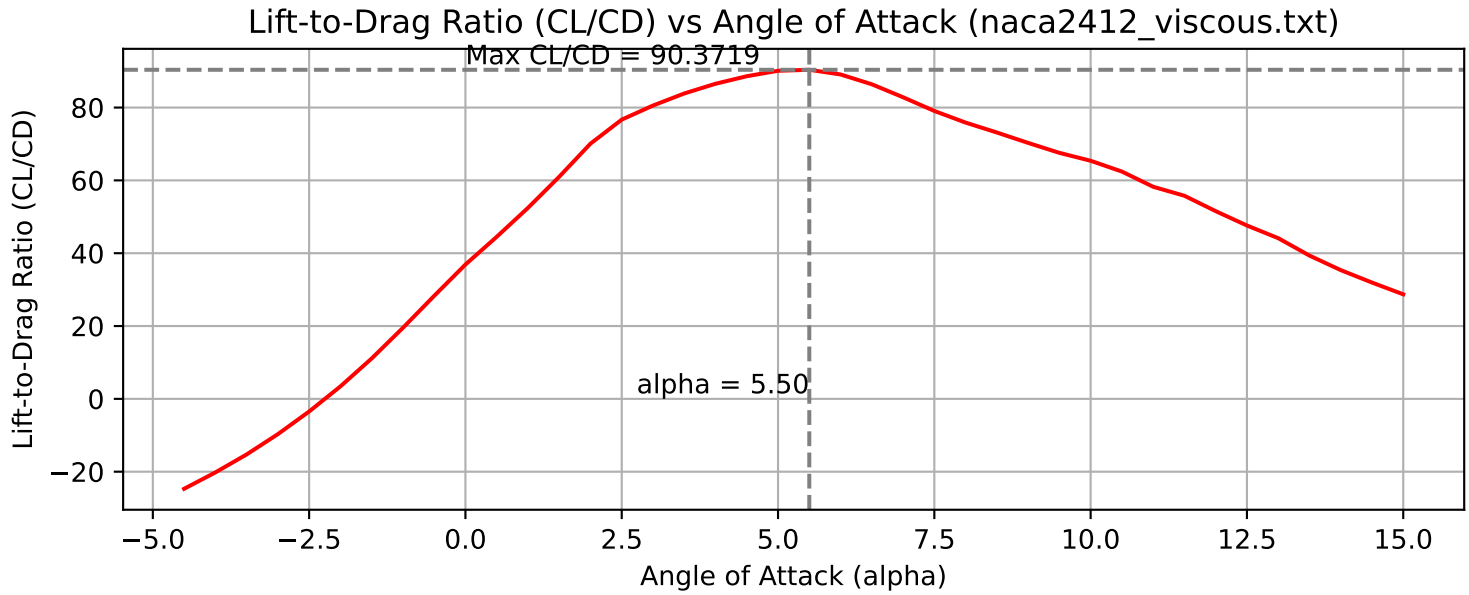

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