# Airfoil Analysis Report

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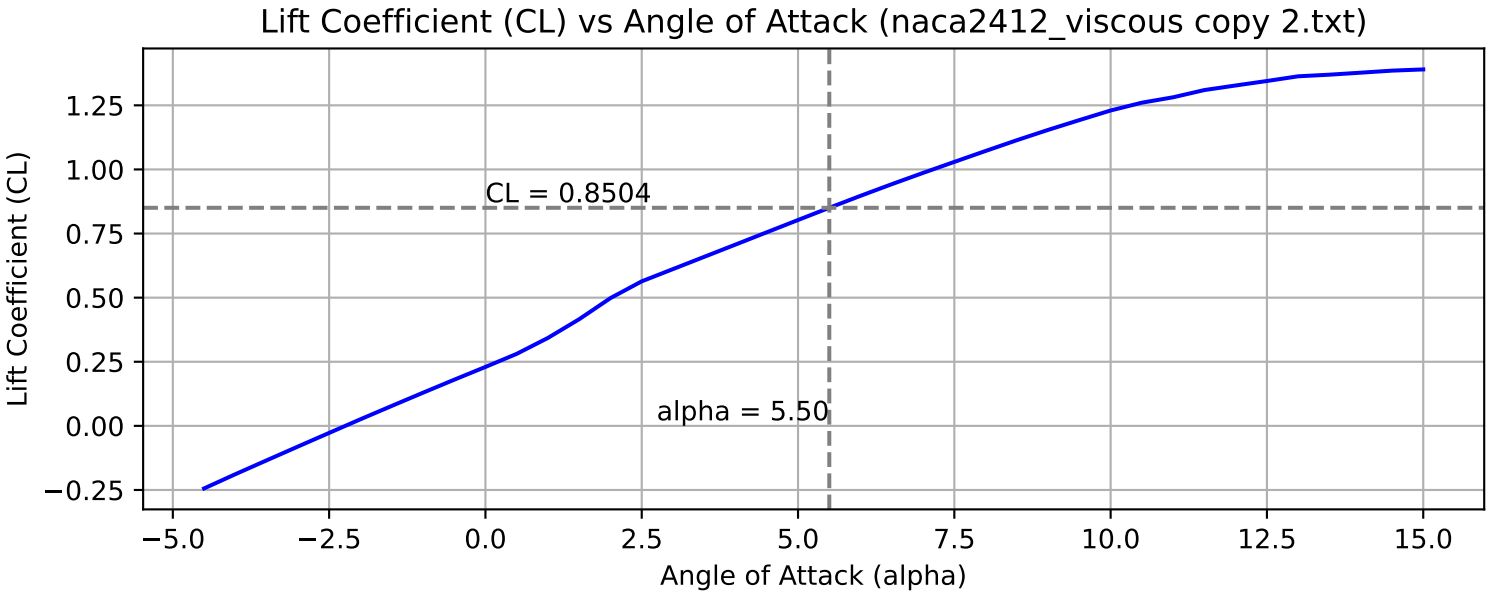
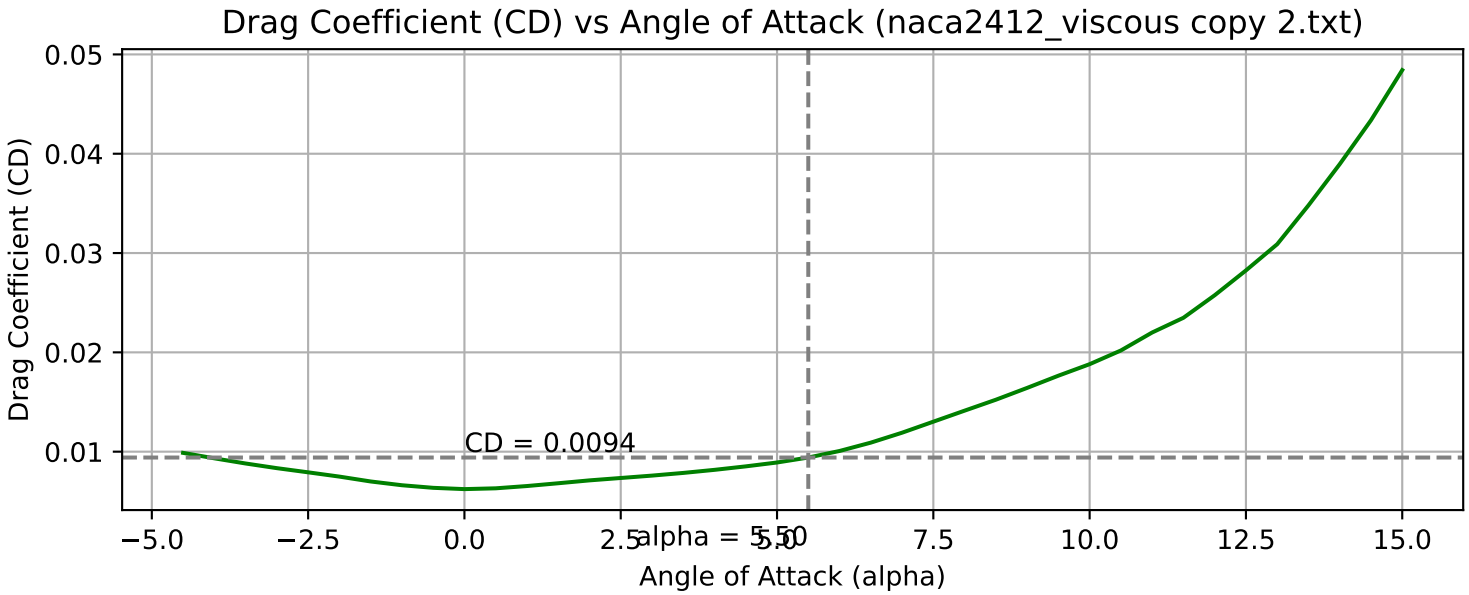
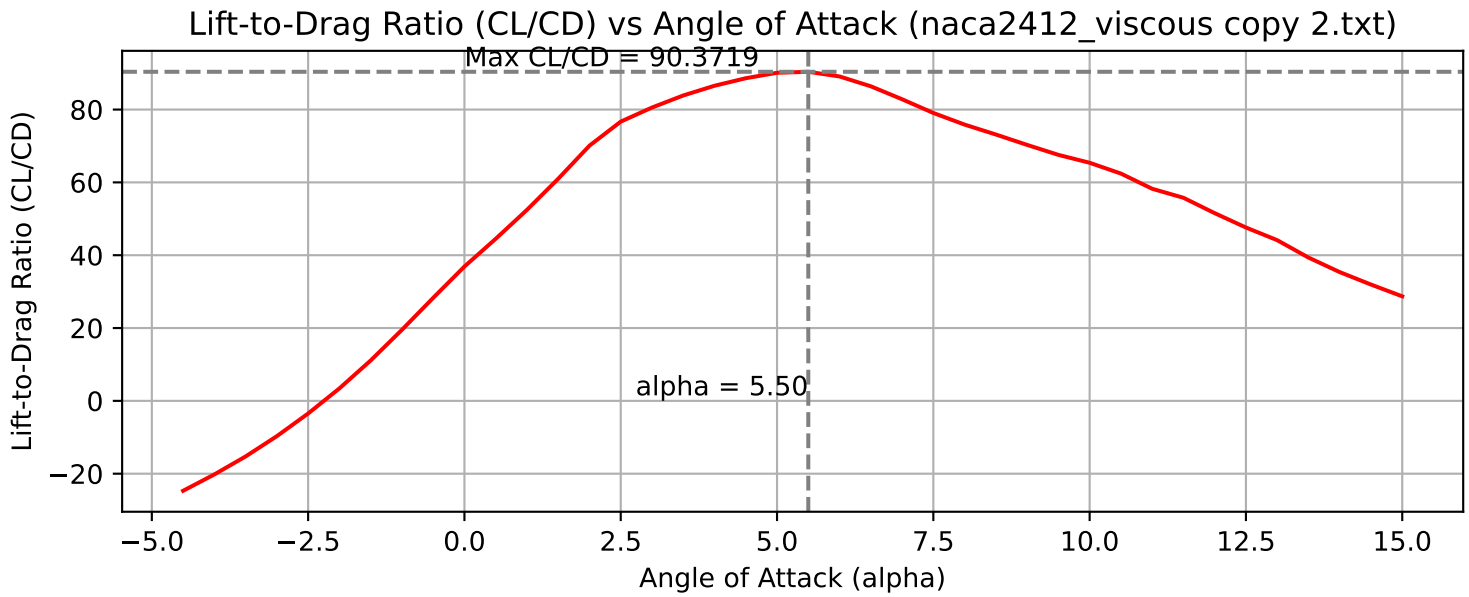
# NACA 2412, Re = 5.220e+05

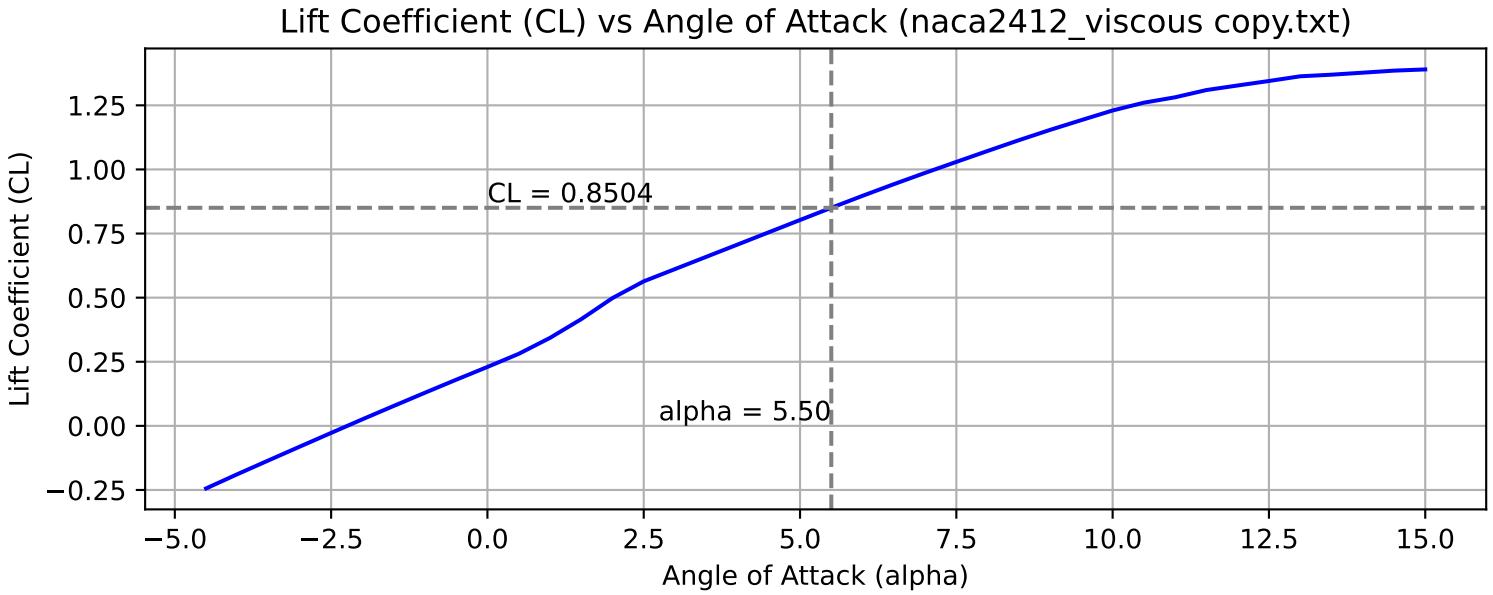


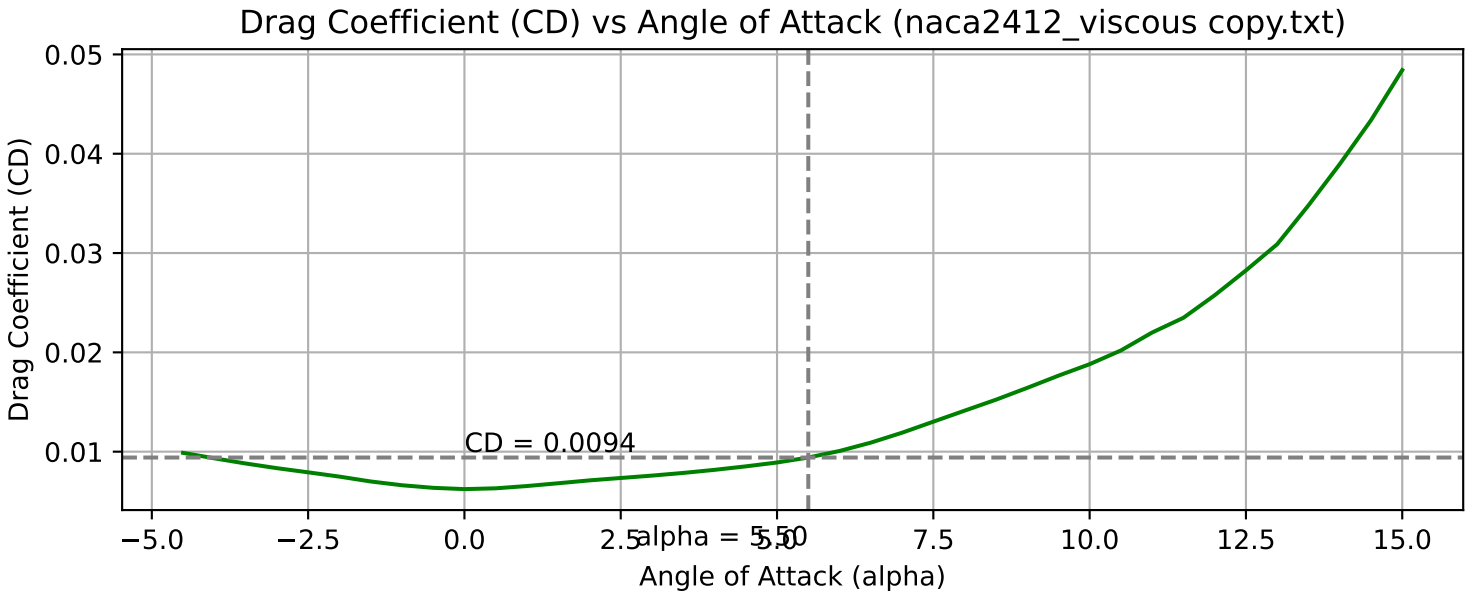



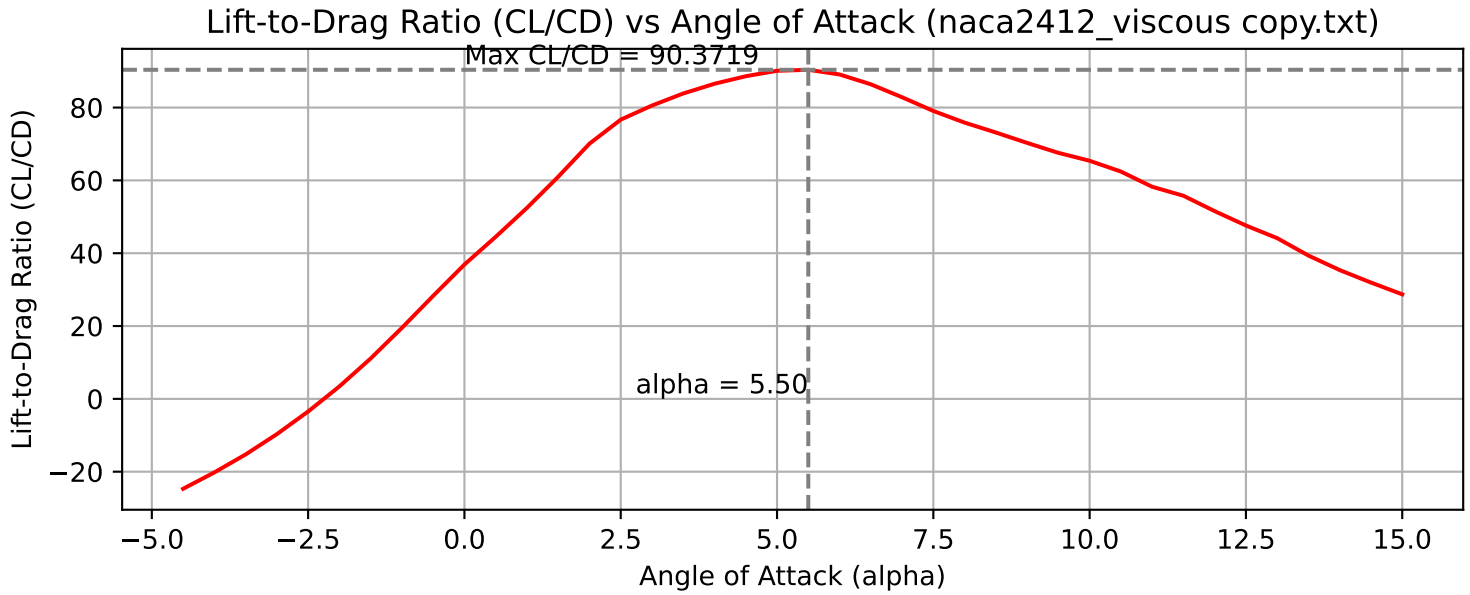




At alpha = 5.50: CL = 0.8504, CD = 0.0094, Max CL/CD = 90.3719

# NACA 2412, Re = 5.220e+05

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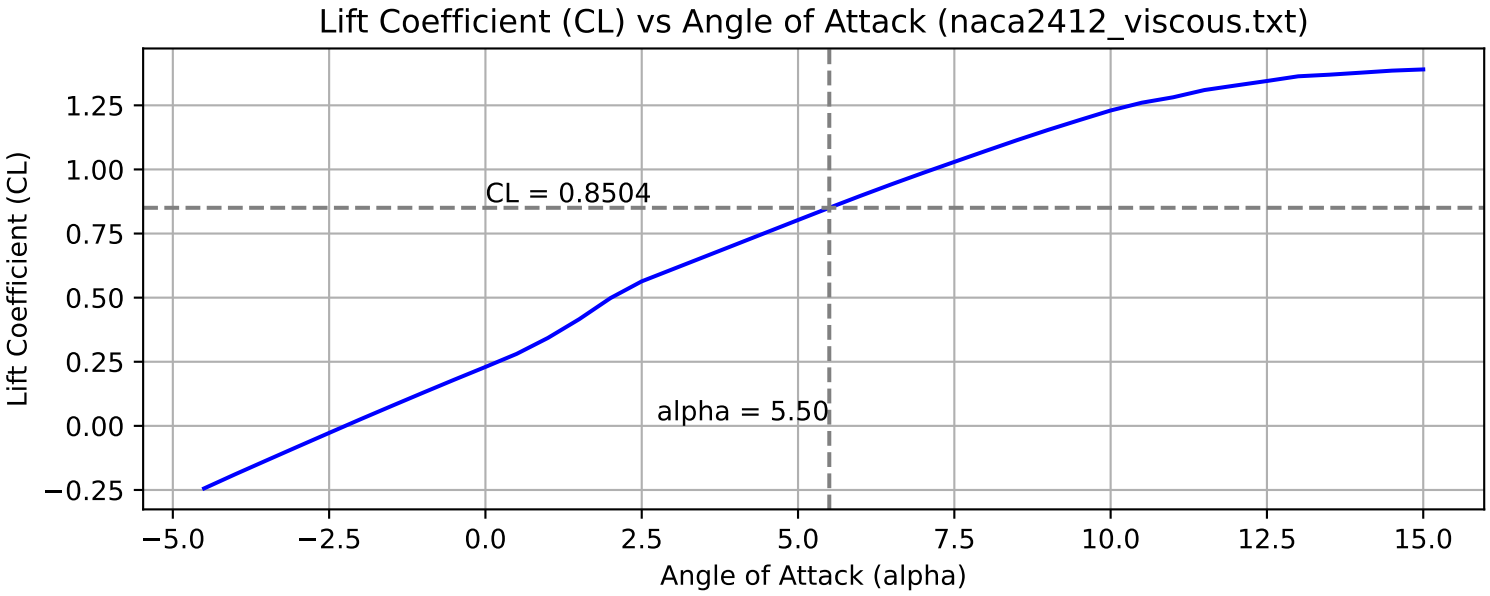


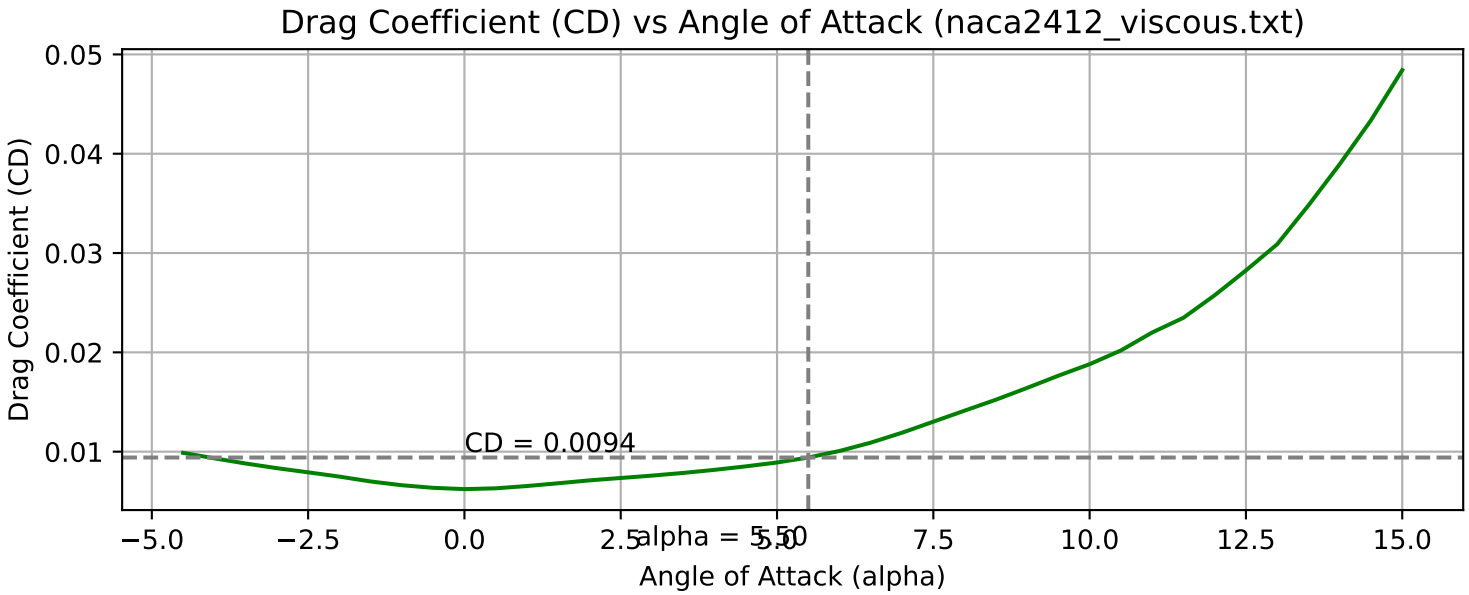


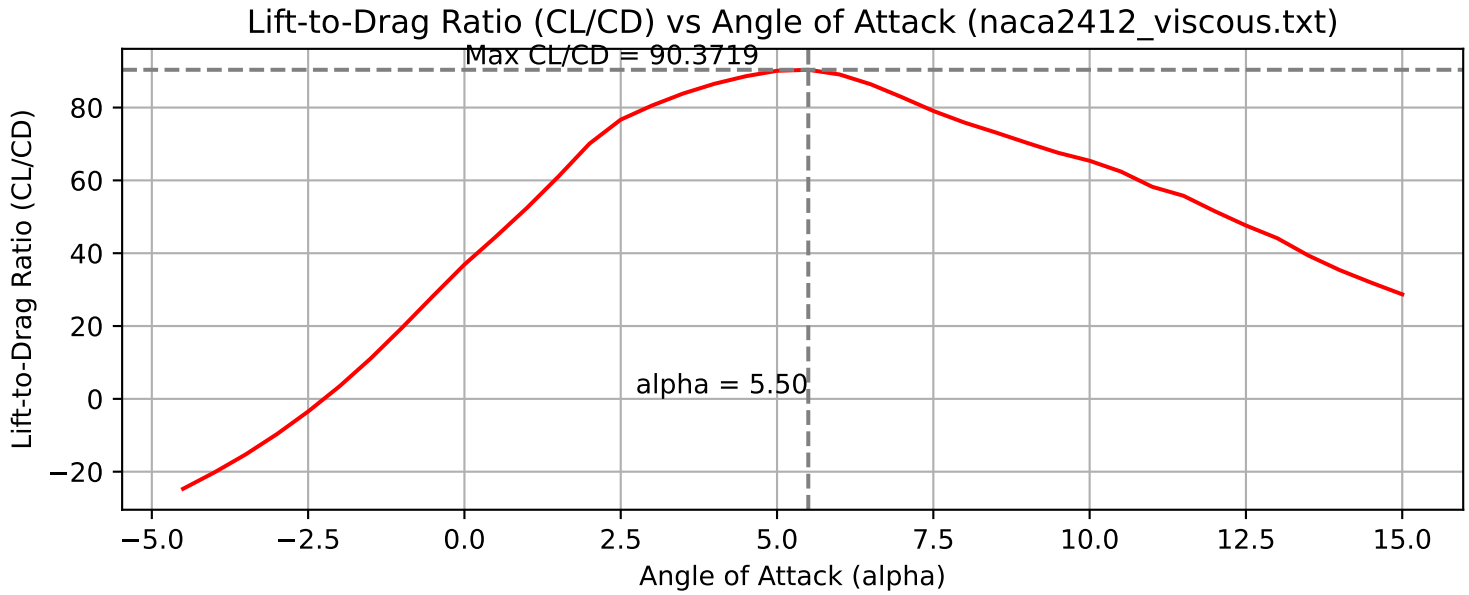


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