ERRATA

NASA Technical Paper 3673

INVARIANCE OF HYPERSONIC NORMAL FORCE COEFFICIENTS WITH REYNOLDS NUMBER AND DETERMINATION OF INVISCID WAVE DRAG FROM LAMINAR EXPERIMENTAL RESULTS

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New pages 4, 30, 31, 41, 52, and 53 are attached. The following changes have been made:

Page 4: The word “coefficient” has been added to the descriptions of figures 11 and 12.

Page 30, figure 3(a), bottom plot: The label for Inviscid theory has been changed to read
Inviscid theory, GASP (ref. 20) and references 23 and 24

Page 31, figure 3(b), bottom plot: The label for Inviscid theory has been changed to read
Inviscid theory, GASP (ref. 20) and reference 23

Page 41, figure 5(e): The curve for TW-PM has been changed from a dashed line to a solid line.

Page 52, figure 11(a): The sublegend has been changed to read
Normal force coefficient.

Page 53, figure 11(b): The sublegend has been changed to read
Axial force coefficient.

The labels on the abscissa were inadvertently omitted from the figure and are now

Plot on left: $1/\sqrt{R_l}$

Plot on right: $1/\sqrt[7]{R_l}$

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Apparatus and Test Conditions

Data measured in four different hypersonic facilities were analyzed and are discussed in this paper. At each of the four installations, the stagnation temperature was set sufficiently high to avoid liquefaction and remain above the supersaturated region, as defined by reference 4 for all tests. All screw, dowel holes, and joints were filled with dental plaster before each test was run.

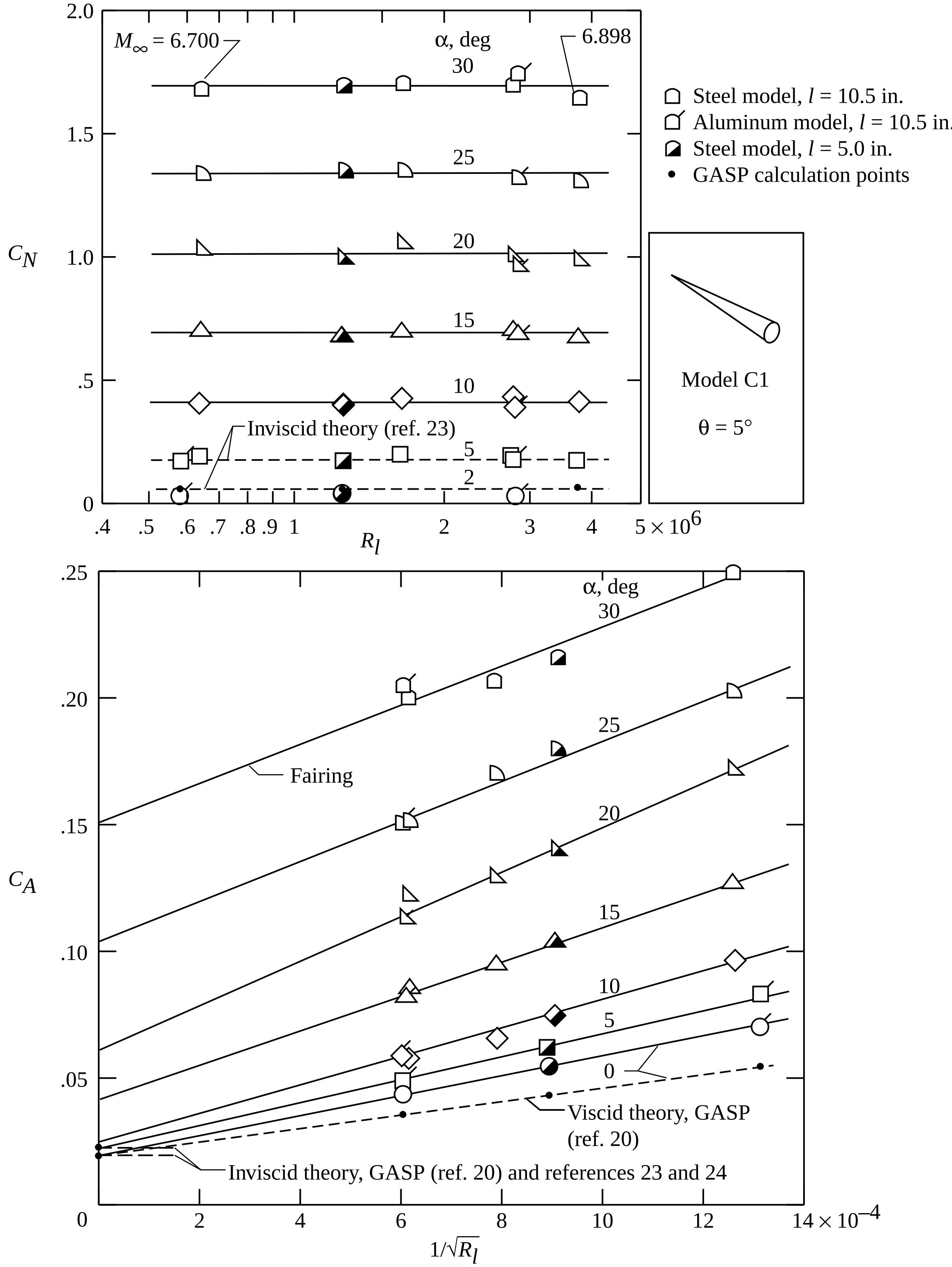
Langley 11-Inch Hypersonic Tunnel

Most of the tests presented in this paper on right circular cones, rectangular wings, delta wings, and a caret wing were conducted in the Mach number 6.86 test section of the Langley 11-Inch Hypersonic Tunnel (now decommissioned). The design of this facility may be found in references 5 and 6. The contours of the two-dimensional nozzle constructed of invar were calculated by Ivan E. Beckwith and are presented in figure 13. Invar was used to construct this nozzle to alleviate the deflection of the first minimum that occurred in the steel nozzle of reference 6 because of differential heating of the nozzle blocks. The tunnel-wall boundary-layer thickness and, therefore, the free-stream Mach number of this test section were dependent upon the stagnation pressure.

For these tests, the stagnation pressure was varied from about 74 to 515 psia, and the stagnation temperature varied from 1040°R to 1150°R. These conditions resulted in an average free-stream Mach number from 6.70 to 6.90 and a unit Reynolds number per foot from 0.617×10^6 to 4.29×10^6, as well as an average Reynolds number based on model length from 0.58×10^6 to 5.35×10^6. The absolute humidity was kept to less than 1.9×10^{-5} lb of water/lb of dry air for all tests. The 11-Inch Hypersonic Tunnel had predominantly laminar flow conditions at all operating pressures; this was substantiated by tests where the transition on a sharp-edged hollow cylinder was experimentally measured in this tunnel at Reynolds numbers as high as 5.7×10^6 (ref. 7). Similar tests on a sharp-edged flat plate showed transition began at a Reynolds number of about 2×10^6 in this tunnel (ref. 8). A private communication from Pierce L. Lawing of the Langley Research Center, who retested the flat plate of reference 8, indicates, however, that his tests showed that, by meticulously cleaning the tunnel walls and throat of dust particles and other debris before each test, he could increase the transition Reynolds number to about 5×10^6, and conversely, by intentionally adding roughness to the tunnel walls in the form of minute glass beads, he could reduce the transition Reynolds number to values approaching the 2×10^6 shown in reference 8. All models were tested on two-, three-, or six-component strain-gauge balances. The size of models for the 11-Inch Tunnel was determined by the method described in appendix A.

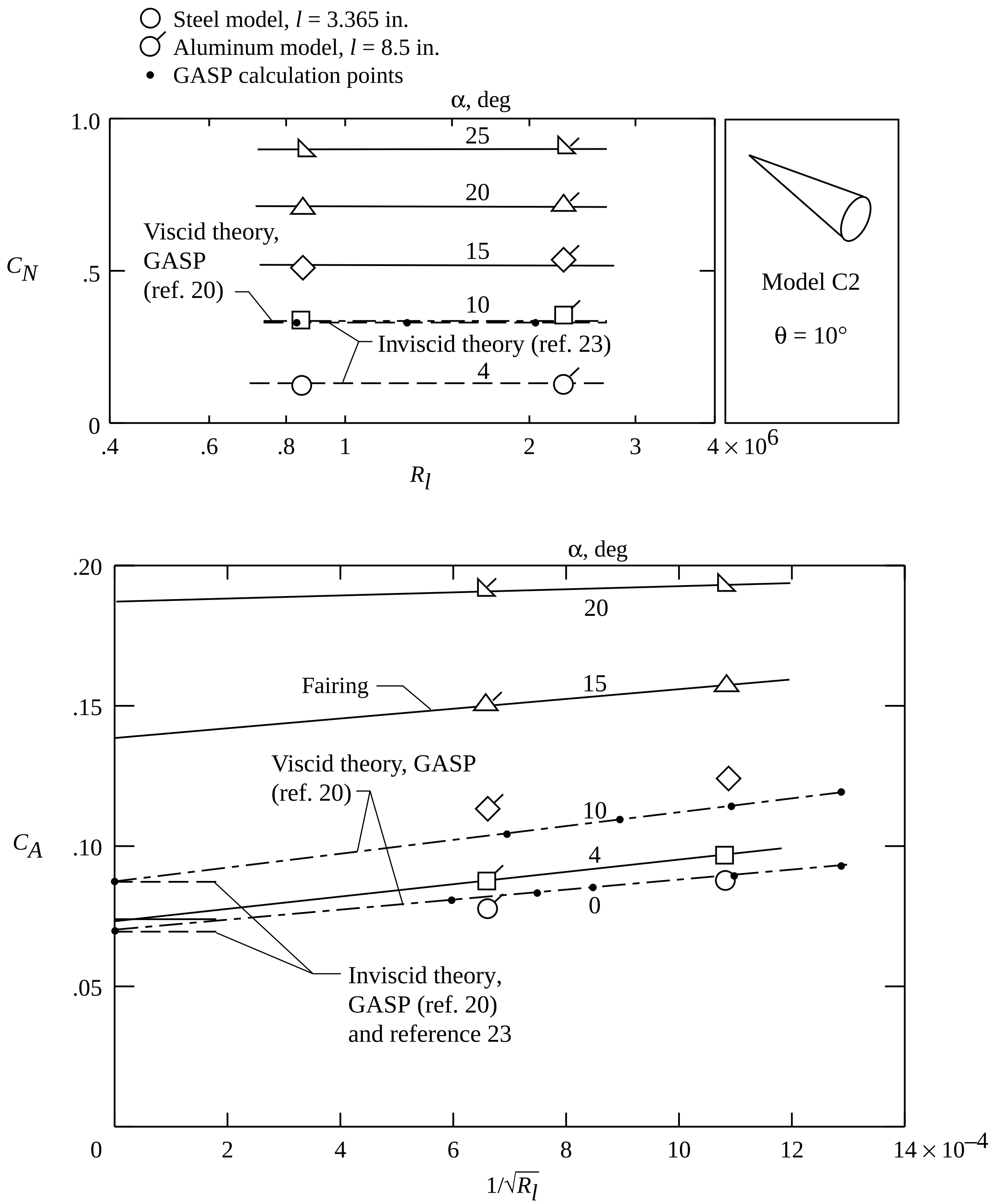
Langley Mach 8 Variable-Density Tunnel

The Langley Mach 8 Variable-Density Tunnel (VDT) (now decommissioned) consisted of an axially symmetric nozzle with contoured walls, had an 18-in-diameter test section, and operated on a blowdown cycle. The tunnel-wall boundary-layer thickness, and therefore the free-stream Mach number, were dependent upon the stagnation pressure. For these tests, the stagnation pressure was varied from about 128 to 2835 psia and the stagnation temperature was varied from about 1135°R to 1480°R. These conditions resulted in an average free-stream Mach number from 7.74 to 8.07 and a Reynolds number based on fuselage length from 1.371×10^6 to 27.084×10^6 (0.636×10^6 to 12.539×10^6/ft). Dry air was used for all tests to avoid any condensation effects. The calibration of this tunnel for the present tests is discussed in reference 1. The model of an advanced blended body-wing hypersonic cruise concept (fig. 2(g)), was



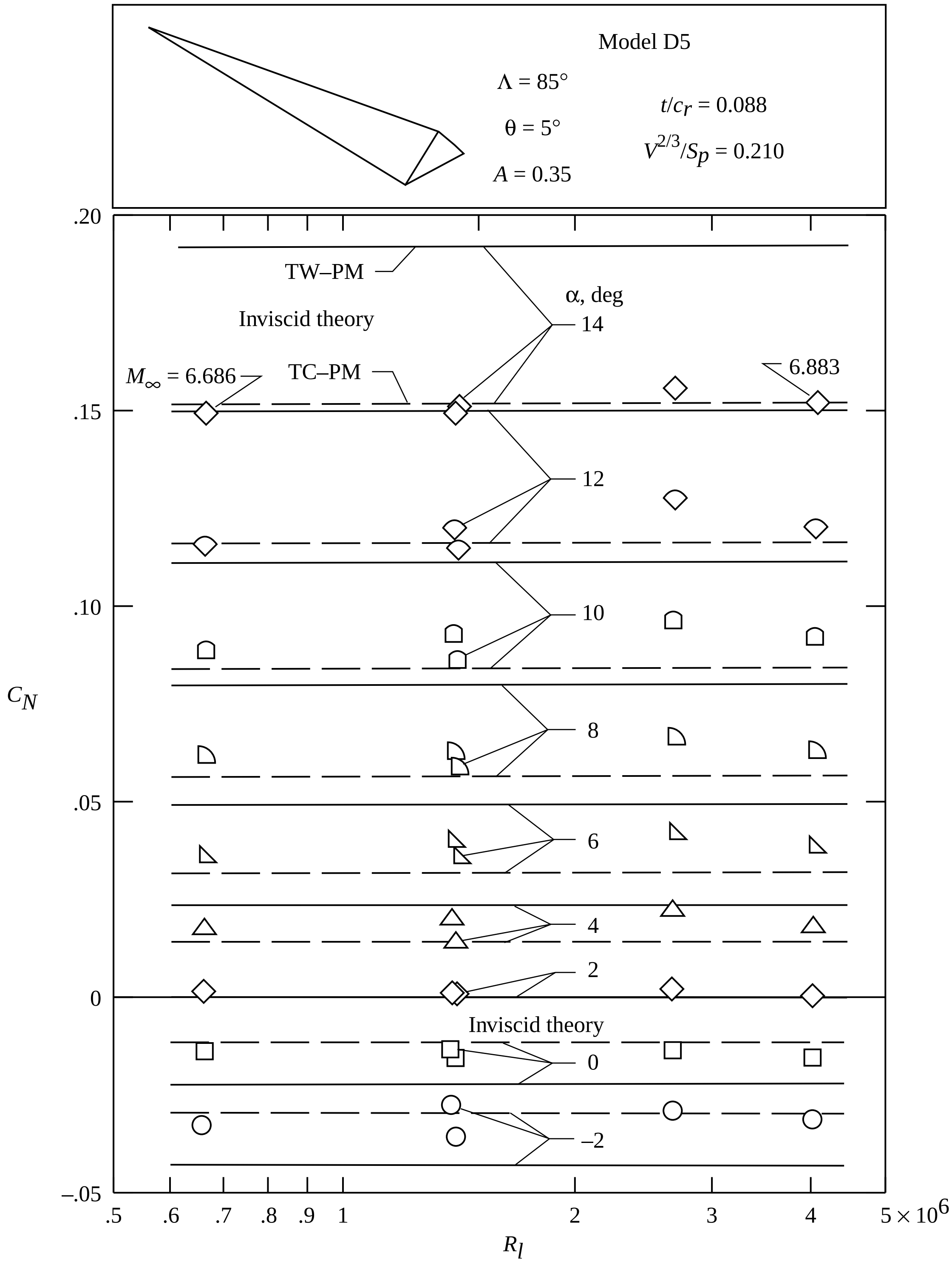
(a) Model C1; θ = 5°.

Figure 3. Variation of normal force and axial force coefficients with Reynolds number at $M_\infty \approx 6.86$ for cones.



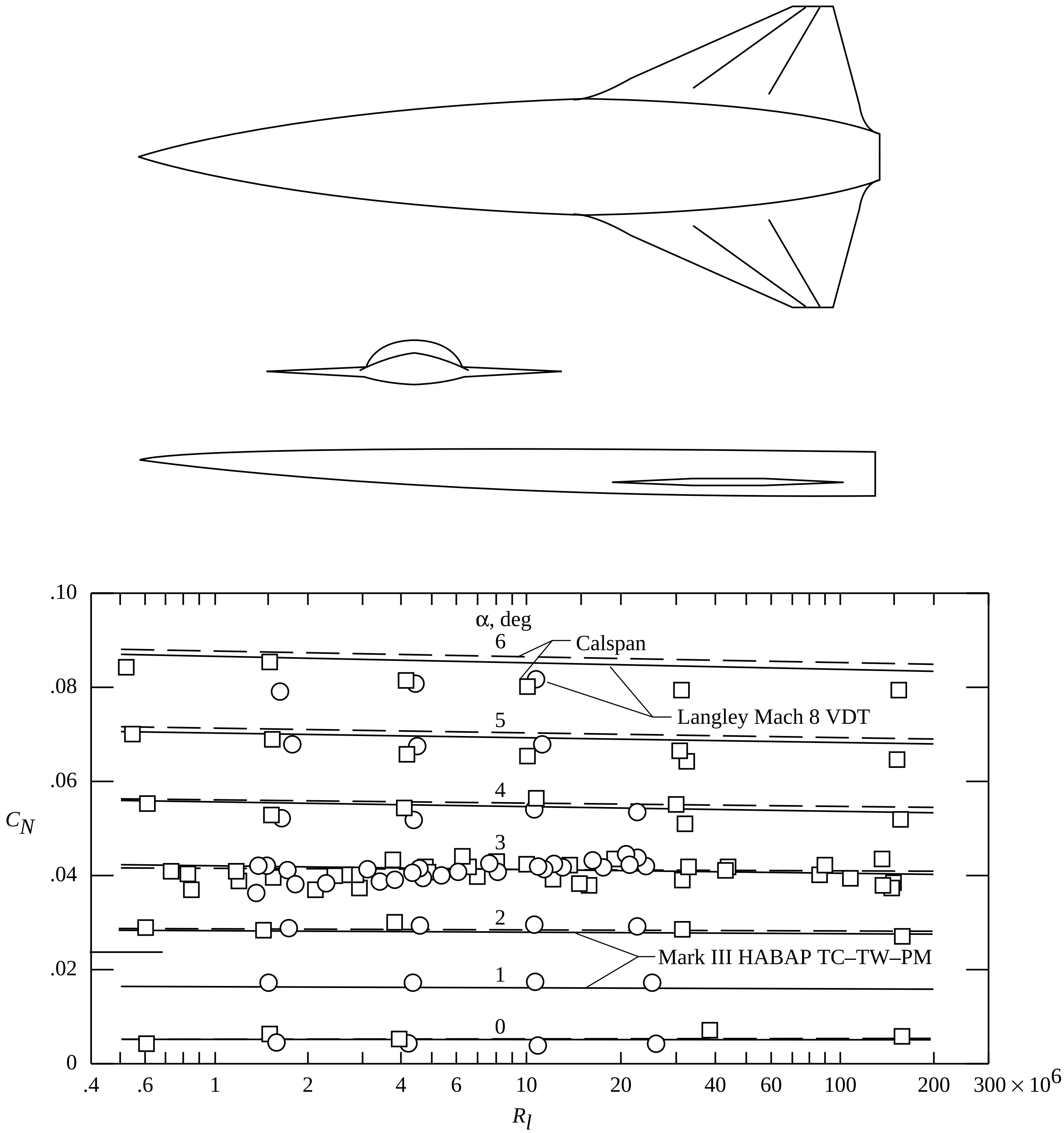
(b) Model C2; θ = 10°.

Figure 3. Continued.



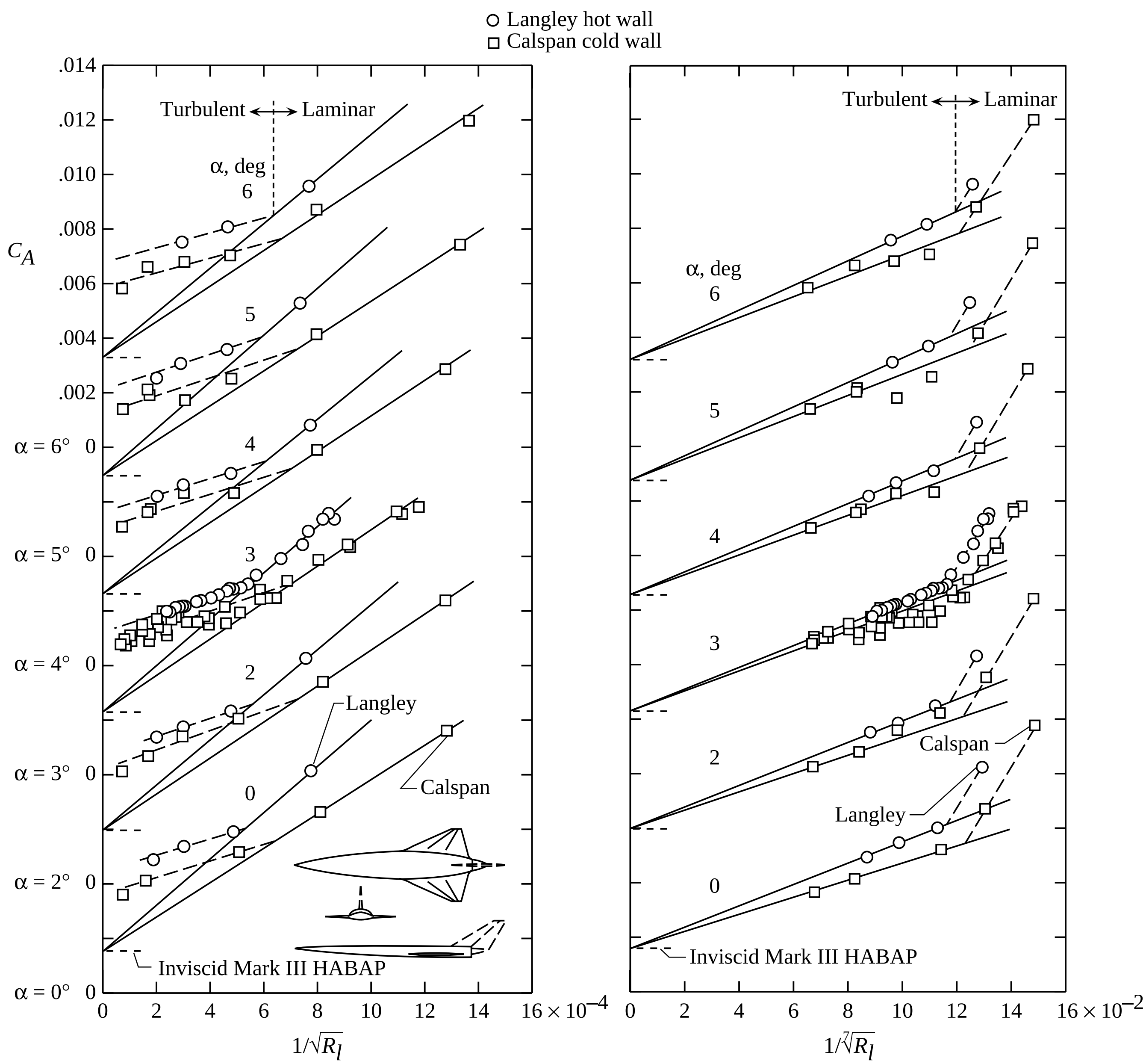
(e) Model D5; $\Lambda = 85°$; $\theta = 5°$; $t/c_r = 0.088$; $V^{2/3}/S_p = 0.210$; $A = 0.35$.

Figure 5. Concluded.



(a) Normal force coefficient.

Figure 11. Variation of force coefficients with Reynolds number at $M_\infty \approx 8.00$ for advanced blended body–wing hypersonic cruise configuration.



(b) Axial force coefficient.

Figure 11. Concluded.