Influence of Moderate Wall Cooling on Cone Transition at $M_e = 13.7$ in Helium

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Nomenclature

M = Mach number
P = pressure, N/m^2
$\dot{q}$ = convective heat-transfer rate, w/m^2
R/m = unit Reynolds number per meter
t = time, sec
T = temperature, °K
x = distance along cone surface from tip, m
u = velocity, m/sec
δ = boundary-layer thickness, cm
θ_c = cone half angle, deg

Subscripts

aw = adiabatic wall
e = boundary-layer edge
t = stagnation conditions
tr = start of transition
w = wall conditions
∞ = freestream conditions

Superscripts

$\sim$ = root mean square

OF the many factors influencing hypersonic boundary-layer transition, one of the most controversial is the effect of wall cooling (or heating). Some investigations[1–3] indicate that heat transfer has a moderate to strong effect on hypersonic boundary-layer transition. Conversely, other studies [4–6] indicate that heat transfer has no measurable effect. In addition, transition "reversal" has been detected at extreme wall cooling for hypersonic,[2,7] as well as supersonic[8] conditions.

This Note presents additional hypersonic transition measurements on a moderately cooled slender cone at a local Mach number considerably higher than previously reported in the literature. The present results, combined with lower hypersonic Mach number data, allow speculation as to the influence of wall cooling on transition Reynolds number for the noise dominated hypersonic case.[9] The tests were conducted in the Mach 20 leg of the Langley High Reynolds Number Helium Facility. This tunnel has an axisymmetric contoured nozzle with a 1.525-m-diam test section. Heat-transfer measurements were obtained at zero angle of attack on a 1.525-m-long, 2.87°-half-angle cone with a nose radius of 0.010 cm. The model was instrumented with thermocouples for determining heating rates (from which the transition location was determined). Local unit Reynolds number for the tests was essentially constant at about $R_e/m \approx 37 \times 10^6$. The helium freestream flow was unheated with $T_{t,\infty} \approx 300$°K. Previous transition data for the same model and facility combination, for the uncooled case, are available in Ref. 9.

Cooling of the model surface was accomplished by venting cold helium vapor into the interior of the cone. The cold helium vapor entered through a tube which ran from the helium supply through the base of the model, and terminated internally about midway along the cone length. The helium vapor exited at this point and distributed itself inside the cone, eventually passing into the test section (maintained at a low vacuum) through bleed holes in the cone base. Prior to testing, the coolant flow was shut off.

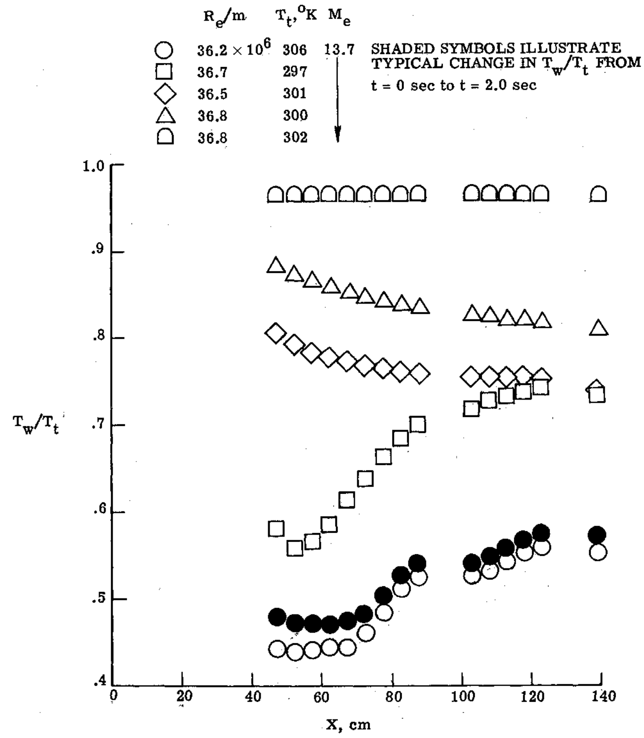

Fig. 1 Model wall temperature distributions prior to testing (t = 0 sec).

Model wall temperature distributions for the test case at prerun conditions are illustrated in Fig. 1. For the lowest wall-to-total temperature ratio conditions, there were steep wall temperature gradients along the cone surface just prior to testing. Comparison of the shaded and unshaded symbols for the lowest wall-to-total temperature ratio case indicate the small increase in wall temperature from prerun conditions to the time when the heat transfer (and, thus transition) data were obtained. The influence of these wall temperature gradients (as opposed to a uniform wall temperature distribution) on the transition location is unknown.

Although attempts were made to eliminate the formation of frost, light to heavy frost coatings were present. However, the presence of frost was not considered detrimental to the transition measurements for two reasons. First, a previous study[1] at $M_e = 6$ verified that the transition location was little affected by the formation of heavy frost as compared to no frost. Secondly, the higher Mach number boundary layer of the present case ($M_e = 13.7$) should be even less susceptible to surface roughness effects caused by frost on boundary-layer transition.

The location of boundary-layer transition for the cold wall conditions was taken as the point where the heat transfer was a minimum. For the heated wall conditions ($-\dot{q}$), the decrease in $\dot{q}$ from the laminar $\frac{1}{2}$-power slope (as $dT_w/dx \approx 0$) marked the start of transition (Ref. 9). The test conditions and transition results of the present investigation are given in Table 1. For convenience, the wall-to-total temperature ratio used hereafter to define each test condition is the value at the transition location.

Cooled wall hypersonic transition data of previous investigations,[1–6] along with the results of the present study, are presented

Table 1 Summary of cooled and heated wall transition data

$T_{t,\infty}$ °K	R_∞/m × 10^6	R_e/m × 10^6	x_{tr} m	$R_{e,tr}$ × 10^6	M_∞	M_e	$T_w/T_{t,\infty}$	T_w/T_{aw}
302	26.0	36.8	0.889	32.8	18.0	13.7	0.96	1.17
300	26.0	36.8	0.838	30.9	18.0	13.7	0.84	1.03
301	25.8	36.5	0.825	29.9	18.0	13.7	0.77	0.94
297	26.0	36.7	0.813	29.8	18.0	13.7	0.69	0.84
306	25.6	36.2	0.825	29.9	18.0	13.7	0.53	0.65

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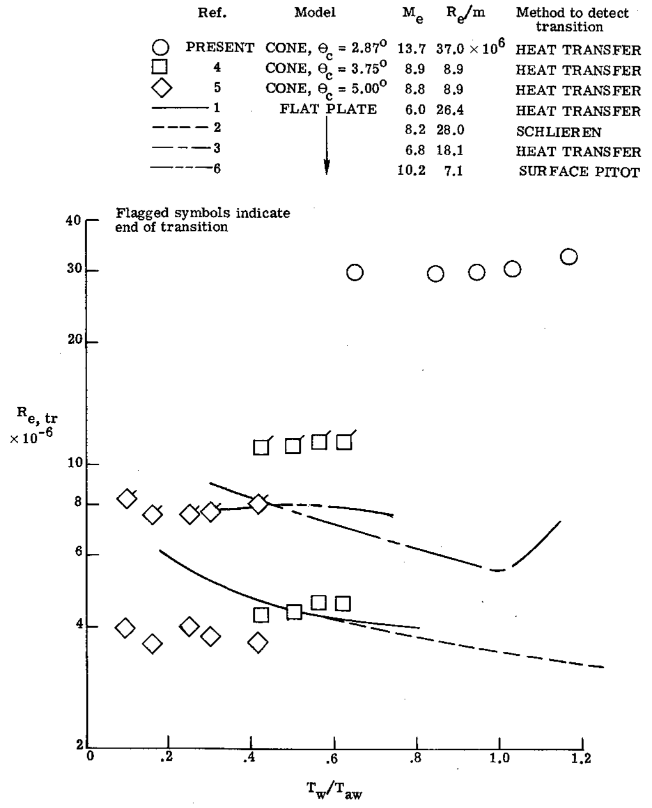

Fig. 2 Influence of wall temperature ratio on hypersonic boundary-layer transition.

in Fig. 2. The data are presented in terms of the adiabatic wall temperature, T_w/T_{aw}, since the majority of previous data are given in this form. The laminar recovery factor used to compute T_{aw} for the present data was 0.816, which is appropriate for the high Mach number helium conditions of the present study.[10] Some of the results[4–6] are represented by the actual symbols, while results from other investigations[1–3] are represented by a fairing through numerous data points. Because of the dominating influence of the freestream disturbance level on hypersonic transition,[9] the absolute level of transition Reynolds number shown in Fig. 2 is dependent on the disturbance environment in the particular facility. Since unit Reynolds number was held constant during each investigation, the cold wall transition results presented in Fig. 2 were each obtained at one (unknown) disturbance level. For the present results, however, this disturbance level was measured and reported[9] as $\tilde{P}_\infty/\bar{P}_\infty \approx 4\%$.

The transition results in Fig. 2 indicate a trend. The lowest Mach number data presented,[1–3] $M_e = 6.0$–8.2, suggest an increasing transition Reynolds number (stabilizing effect) with decreasing wall-to-adiabatic temperature ratio, whereas the higher Mach number data,[4–6] $M_e = 8.8$–13.7, suggest that moderate wall cooling has no appreciable influence on transition Reynolds number. A possible explanation for this observed Mach number influence on cooled boundary-layer transition results may be found in previous experimental work,[11,12] and in the stability analysis of Mack.[13] Spark schlierens of the present cone model[12] revealed a wavy disturbance structure at the boundary-layer edge ahead of the surface transition location. The frequency of these wavy disturbances was approximately 90 kHz ($u_e = 1830$ m/sec, wave length $\approx 2\delta$). Using the results of Mack,[13] the frequency of the most unstable two-dimensional disturbance for the present test conditions was approximately 80 kHz for the second mode and 46 kHz for the first mode. These calculations suggest that Tollmien-Schlicting-like wave forms characteristic of the second mode could be present in the cone boundary layer for the sound-dominated disturbance environment of the present study. If this is the case, then increased wall cooling should not increase transition Reynolds number for the present conditions since the second mode is not stabilized by wall cooling[13] (unlike the first mode, where wall cooling is stabilizing).

In a previous investigation[11] with a 10° half-angle cone ($M_e = 7.6$) tested in the same helium tunnel as the present study, a wavy structure ahead of transition was also visually observed. A crude wave form frequency was calculated to be approximately 250 kHz ($u_e = 1700$ m/sec, wave length $= 3\delta$). Characteristic frequencies for the first and second modes, from Mack,[13] were approximately 225 kHz and 370 kHz, respectively. These results suggest that for this lower Mach number ($M_e = 7.6$), Tollmien-Schlichting-like waves characteristic of the first mode could be present.

The implication from the above crude analysis is that the higher Mach number transition data in Fig. 2 ($M_e \geqslant 8.8$) displayed no apparent effect of wall cooling because Tollmien-Schlichting-like waves characteristic of the second mode (not stabilized by cooling[13]) were present. Conversely, it is conjectured that the lower Mach number transition data in Fig. 2 ($M_e \leqslant 8.2$) was influenced by wall cooling because of the presence of wave forms characteristic of the first mode, which is stabilized by cooling.[13]

An alternate explanation for the insensitivity of the higher hypersonic Mach number data to wall cooling could be found in the work of Refs. 12 and 14. That is, since at high Mach number transition originates at the outer edge of the boundary layer (in the region of the critical layer), transition Reynolds number should be less influenced by conditions at or near the wall (surface roughness and wall cooling, for example).

The above hypotheses have some experimental and theoretical backing; however, the conclusions are limited to sound-dominated wind-tunnel transition results, and may not be applicable to flight transition results where a different mode of disturbance environment (at much lower amplitude) exists.

References

[1] Cary, A. M., Jr., "Turbulent-Boundary-Layer Heat-Transfer and Transition Measurements With Surface Cooling at Mach 6," TN D-5863, June 1970, NASA.

[2] Richards, B. E. and Stollery, J. L., "Transition Reversal on a Flat Plate at Hypersonic Speeds," *Recent Developments in Boundary Layer Research*, Pt. I, AGARDograph 97, May 1965, pp. 477–501.

[3] Maddalon, D. V., "Effect of Varying Wall Temperature and Total Temperature on Transition Reynolds Number at Mach 6.8," *AIAA Journal*, Vol. 7, No. 12, Dec. 1969, pp. 2355–2357.

[4] Everhart, P. E. and Hamilton, H. H., "Experimental Investigation of Boundary-Layer Transition on a Cooled 7.5° Total-Angle Cone at Mach 10," TN D-4188, 1967, NASA.

[5] Sanator, R. J., DeCarlo, J. P., and Torrillo, D. T., "Hypersonic Boundary-Layer Transition Data for a Cold-Wall Slender Cone," *AIAA Journal*, Vol. 3, No. 4, April 1965, pp. 758–760.

[6] Deem, R. E. and Murphy, J. S., "Flat Plate Boundary Layer Transition at Hypersonic Speeds," AIAA Paper 65-128, New York, 1965.

[7] Richards, B. E. and Stollery, J. L., "Further Experiments on Transition Reversal at Hypersonic Speeds," *AIAA Journal*, Vol. 4, No. 12, Dec. 1966, pp. 2224–2226.

[8] Wisniewski, R. J. and Jack, R. J., "Recent Studies on the Effect of Cooling on Boundary-Layer Transition at Mach 4," *Journal of the Aerospace Sciences*, Readers' Forum, Vol. 28, No. 3, March 1961, pp. 250–251.

[9] Fischer, M. C. and Wagner, R. D., "Transition and Hot-Wire Measurements in Hypersonic Helium Flow," *AIAA Journal*, Vol. 10, No. 10, Oct. 1972, pp. 1326–1332.

[10] Rudy, D. H. and Weinstein, L. M., "Investigation of Turbulent Recovery Factor in Hypersonic Helium Flow," *AIAA Journal*, Vol. 8, No. 12, Dec. 1970, pp. 2286–2287.

[11] Fischer, M. C., "Turbulent Bursts and Rings on a Cone in Helium at $M_e = 7.6$," *AIAA Journal*, Vol. 10, No. 10, Oct. 1972, pp. 1387–1389.

[12] Fischer, M. C. and Weinstein, L. M., "Cone Transitional Boundary-Layer Structure at $M_e = 14$," *AIAA Journal*, Vol. 10, No. 5, May 1972, pp. 699–701.

[13] Mack, L. M, "Boundary-Layer Stability Theory," JPL 900-277, Rev. A, Nov. 1969, Jet Propulsion Lab., Pasadena, Calif.

[14] Fischer, M. C., "Spreading of a Turbulent Disturbance," *AIAA Journal*, Vol. 10, No. 7, July 1972, pp. 957–959.