Theory of Gases and Liquids (John Wiley and Sons Inc , New York, 1954), pp 230–232

[10] Smith, C E , "Thermodynamic properties of nitrogen," 6-90-62-111, Lockheed Missiles and Space Co , Sunnyvale, Calif (December 1962)

Supersonic Stagnation Point Heat Transfer at Low Reynolds Numbers

Henry Tong*
Boeing Company, Seattle, Wash

and

W H Giedt†
University of California, Berkeley, Calij

Stagnation point heat transfer at low Reynolds numbers in a supersonic air stream was investigated experimentally A transient technique was employed using precooled thin-walled hemisphere-cylinder models Results were obtained for nominal Mach numbers of 2, 4, and 6 and in the Reynolds number (based on conditions behind the bow shock and model diameter) range of 80 to 1500 These results are in good agreement with continuum boundary-layer theory down to a Reynolds number of approximately 300 At values below this an increase approaching 10% is indicated

Nomenclature

c_p	=	specific heat at constant pressure
d	=	cylinder and hemisphere diameter
h	=	heat-transfer coefficient = $q/(T_0 - T_w)$
k	=	thermal conductivity
M	=	Mach number
Nu	=	Nusselt number = hd/k
Pr	=	Prandtl number = uc_p/k
q	=	stagnation point heat rate
Re	=	Reynolds number = $\rho Ud/\mu$
T	=	absolute temperature
$\tilde{u}\,'$	=	$[d(U\,/U_2)/d(x/d)]$
U	=	velocity
x	=	distance from stagnation point
ϵ	=	density ratio across normal shock
μ	=	viscosity
ρ	=	density

Subscripts

0	=	stagnation reservoir
1	=	upstream of shock
2	=	downstream of shock
BL	=	boundary-layer value
e	=	edge of boundary layer

I Introduction

Representative results of recent studies of the heat transfer to the stagnation point of a blunt body at low Reynolds numbers are summarized in Fig 1 Analytical investigations have considered various second-order effects ordinarily neglected in classical boundary-layer theory These include vorticity introduced as the flow passes through the bow shock wave, slip, and temperature jump ‡ To indicate the importance of these second order effects, comparisons are made with continuum boundary-layer results[2] extrapolated to low Reynolds numbers

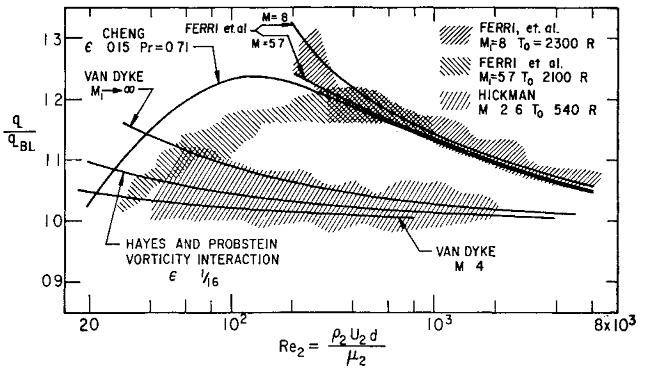

Fig 1 Comparison of theoretical and experimental stagnation point heat transfer

Two trends are apparent in Fig 1 The first is supported by the theories of Ferri, Zakkay, and Ting,[3] Cheng,[4] and the data of Ferri et al [3, 5] This shows an increase in heat transfer above continuum boundary-layer values starting at Re_2 of about 20,000 and increasing to a maximum of 25% at $Re_2 \sim$ 150 The second trend[1, 6–8] indicates the heat transfer begins to rise above the continuum boundary-layer value at magnitudes of $Re_2 \lesssim 1000$ and that the increase is only about one-third to one-half that of the other results

No satisfactory explanation has been advanced for the differences in these two trends It was recognized, however, that the experimental data available were obtained in different test facilities and with different techniques Because of this it was decided that additional tests in the low-pressure wind tunnel at the University of California using a transient technique would be desirable Results obtained are presented herein

II Experimental Equipment and Procedure

Heat-transfer models were thin-walled hemisphere cylinders of electroplated nickel Three sizes were used: 0 250 in , 0 500 in , and 1 00 in in diameter, with 0 004 in nominal wall thicknesses Transient temperatures were measured by means of No 40 gage copper-constantan thermocouples welded into holes drilled at the stagnation points

The models were cooled below the wind tunnel stagnation temperature (~530°R) by enclosing them in a coil of copper tubing through which liquid nitrogen was pumped The coil was then suddenly removed and the model rotated rapidly into the tunnel air stream Further details of the construction techniques, instrumentation, and evaluation of the test results are given in Ref 9

III Results

The stagnation-point heat-transfer results determined are shown graphically in Fig. 2 in terms of the Nusselt number divided by the square root of the dimensionless velocity gradient at the edge of the boundary layer, $Nu/\tilde{u}_e'^{1/2}$, as a function of the Reynolds number downstream of the shock, Re_2 Evaluation of all temperature records yielded duplicate results for each test condition Since the variation of $Nu/\tilde{u}\,'^{1/2}$ obtained at any value of Re_2 was indicative of the experimental accuracy, all data points were included in Fig 2 As can be noted, only a few points fell outside the lines deviating ±10% from the continuum boundary-layer reference line

An average $Nu/\tilde{u}\,'^{1/2}$ was computed at each value of Re_2 for which two or more experimental points were available

Received November 8, 1963 Based on research supported by the Aeronautical Research Laboratory of the U S Air Force Aeronautical Systems Division under Contract No AF33(657)-8607

* Research Engineer, Aero-Space Division

† Professor of Aeronautical Sciences Member AIAA

‡ The various effects considered by different investigators are reviewed and discussed in Ref 1

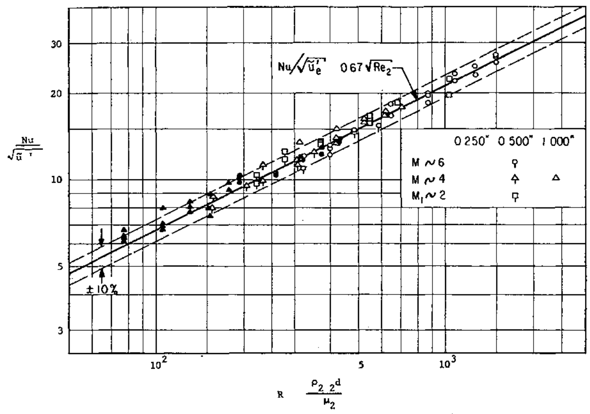

Fig 2 Comparison of measured stagnation point heat transfer with continuum boundary-layer theory

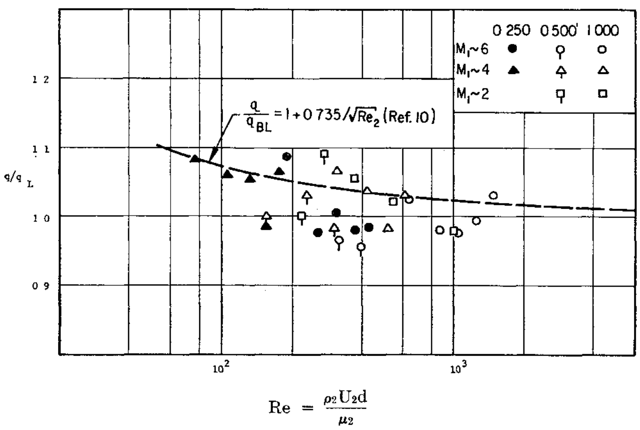

Fig 3 Ratio of average values of measured stagnation point heat transfer to continuum boundary-layer theory

The ratios of these average values to continuum boundary-layer theory have been plotted as a function of Re_2 in Fig 3 The curve representing a least-squares fit of the data reported in Ref 1 has also been included in this figure Although the results from this investigation are in agreement with this curve, it should be noted that the data at the lower Reynolds numbers are primarily from the 0 250 in model tested at $M_1 \sim$ 4 0 In view of this and the experimental scatter, a firm conclusion regarding an increase over boundary-layer theory is not considered justified

The results are therefore considered to be in agreement with boundary layer theory over the range of Reynolds numbers investigated However, since the trends predicted by the analyses of Refs 6–8 lie within the experimental scatter it is possible that small second-order effects are present Reference to Fig 1 shows that the experimental results of Ferri et al [3 5] as well as the theories of Ferri[3] and Cheng[4] are significantly higher and also show a deviation from continuum boundary-layer theory at much higher Reynolds numbers It is noted that the data of Ferri et al were obtained at higher stagnation temperatures (~2300° compared to 530°R) This has been advanced as the reason for the differences in the theoretical predictions and experimental results This explanation is, however, questioned by Van Dyke [8]

References

[1] Hickman, R S and Giedt, W H, "Heat transfer to a hemisphere-cylinder at low Reynolds numbers," AIAA J 1, 665–672 (1963)

[2] Fay, J A and Ridell, F R, "Theory of stagnation point heat transfer in dissociated air," J Aeronaut Sci 25, 73–85 (1958)

[3] Ferri, A, Zakkay V and Ting, L, "Blunt body heat transfer at hypersonic speed and low Reynolds numbers" J Aerospace Sci 28, 962–971 (1961)

[4] Cheng, H K, "Hypersonic shock-layer theory of the stagnation region at low Reynolds number," *Proceedings of the 1961 Heat Transfer and Fluid Mechanics Institute* (Stanford University Press, Stanford, Calif, 1962), pp 161–175

[5] Ferri, A and Zakkay, V, 'Measurements of stagnation point heat transfer at low Reynolds numbers," J Aerospace Sci 29, 847–850 (1962)

[6] Hayes, W D and Probstein R F, *Hypersonic Flow Theory* (Academic Press Inc, New York, 1959), p 372

[7] Van Dyke, M, 'Second-order compressible boundary-layer theory with application to blunt bodies in hypersonic flow," *Hypersonic Flow Research* (Academic Press Inc, New York and London, 1962) p 37

[8] Van Dyke, M, 'A review and extension of second-order hypersonic boundary-layer theory," *Rarefied Gas Dynamics* (Academic Press Inc, New York, 1963), Vol II, pp 212–227

[9] Tong, H and Giedt, W H, "Supersonic stagnation point heat transfer to hemisphere cylinders at low Reynolds numbers," Air Force Aeronaut Res Labs ARL 63-25 (1963)

Measurements of Test Time in the GALCIT 17-Inch Shock Tube

ANATOL ROSHKO* AND JEROME A SMITH†

California Institute of Technology, Pasadena, Calif

EXPERIMENTAL measurements of test time were obtained in the GALCIT 17-in shock tube[1] using both air and argon for driven gases One series of tests was conducted using a constant driver pressure (pure helium) for various initial pressures of the driven gases Another series was conducted using air for the driven gas at various initial pressures holding the shock Mach number constant The data are presented and compared to theoretical predictions computed from the theory in two recent papers by Mirels for the case of a laminar[2] and turbulent[3] wall boundary layer

Test times were obtained at the centerline of the shock tube using two different contact surface probes to detect arrival of the contact surface (in a manner similar to that described in Ref 4); these were a stagnation-point heat-transfer gage and a cold-wire gage The stagnation-point heat-transfer gage consisted of a thin platinum film deposited on a $\frac{1}{8}$-in-diam quartz rod The cold-wire probe consisted of a 0 0005-in-diam platinum wire; because of its low resistance, it was useful at higher Mach numbers for avoiding shorting by the slightly ionized gas (particularly argon) Initially, it was felt that the lifetime of the stagnation-point heat-transfer gage would be longer than that of the cold wire, but this was found not to be the case, and so the cold-wire probe was used to obtain all the data for the series of constant Mach number tests For the very low initial pressures of the driven gas ($p_1 \leq 100\ \mu$ Hg), it was possible to measure the shock-wave contact surface separation distance (and thus test time) from a station a few centimeters from the end wall ($x = 20\ 332$ m from the diaphragm) Test times for the higher initial pressures were obtained at a station farther from the end wall ($x_s = 16\ 668$ m)

In order to determine the time between shock passage and transition (if any) to a turbulent boundary layer, the response of a thin film resistance gage on the side wall was recorded along with the oscillograph recording of the voltage change of the contact surface probe during each test Transition Reynolds numbers as defined in Refs 5 and 6 were found to be between 2×10^6 and 4×10^6

Received November 5, 1963 This work was carried out under the sponsorship of NASA

* Professor of Aeronautics, Graduate Aeronautical Laboratories Member AIAA

† Graduate Student, Graduate Aeronautical Laboratories Student Member AIAA