decrease with increasing freestream Mach number, which is contrary to results for glancing-shock and two-dimensional separation. Also, the three-dimensional separation angle from the incident-shock interaction is approximately half the two-dimensional value at $M_\infty = 5.9$ but nearly equal to the two-dimensional value at $M_\infty = 2.9$. A probable explanation of this is that the large lambda shock formation which occurred in the present experiments did not occur for Reda and Murphy[5] at $M_\infty = 2.9$. (Compare oil flows in Fig. 6 of Ref. 5 with Fig. 2 of this Note). This lambda shock results from flow disturbances on the base plate which extend considerably further upstream of the inviscid glancing-shock location for hypersonic flow than for supersonic flow as illustrated by the oil flows in Fig. 8 of Ref. 5 and Fig. 2 of this Note. From a practical standpoint, this incident-shock interaction result indicates that separation will occur in hypersonic rectangular inlets at smaller flow deflection angles than in supersonic inlets provided that other conditions are similar. This is an unexpected result with important implications.

In summary, it has been shown that separation occurs much earlier for turbulent three-dimensional flow than for two-dimensional flow at hypersonic speeds. Thus, for conditions when both types of separation can occur (such as in hypersonic inlets) three-dimensional rather than two-dimensional separation criteria should be used. A first approximation of three-dimensional incipient separation angles, which are apparently insensitive to turbulent Reynolds numbers, has been given.

References

[1] Wang, K. C., "Separation Patterns of Boundary Layer Over an Inclined Body of Revolution," *AIAA Journal*, Vol. 10, No. 8, Aug. 1972, pp. 1044–1050.

[2] Maskell, E. C., "Flow Separation in Three Dimensions," Rept. Aero. 2565, 1955, Royal Aircraft Establishment, Farnborough, England.

[3] Coleman, G. T., Elfstrom, G. M., and Stollery, J. L., "Turbulent Boundary Layers at Supersonic and Hypersonic Speeds," AGARD Symposium on Turbulent Shear Flows, Paper 31, London, 1971.

[4] Holden, M. S., "Shock Wave-Turbulent Boundary Layer Interaction in Hypersonic Flow," AIAA Paper 72-74, San Diego, Calif., 1972.

[5] Reda, D. C. and Murphy, J. D., "Shock Wave-Turbulent Boundary Layer Interactions in Rectangular Channels," *AIAA Journal*, Vol. 11, No. 2, Feb. 1973, pp. 139–140.

[6] McCabe, A., "The Three-Dimensional Interaction of a Shock Wave with a Turbulent Boundary Layer," *Aeronautical Quarterly*, Vol. 17, Aug. 1966, pp. 231–252.

[7] Stanbrook, A., "An Experimental Study of the Glancing Interaction Between a Shock Wave and a Turbulent Boundary Layer," ARC CP 555, 1961, Aeronautical Research Council, London, England.

An Attempted Improvement of the Method of Integral Relations for a Blunt Body in a Supersonic Flow

B. L. HUNT*

University of Bristol, Bristol, England

THE problem of calculating the flow in the shock layer ahead of a blunt-nosed body in a supersonic stream has been the subject of many investigations. In recent years, two methods have been developed which have received widespread use: these are the time-dependent method, the most effective form for blunt-body work being that due to Moretti,[1] and the Method of Integral Relations, which was developed by Belotserkovskii and co-workers.[2,3]

In the Method of Integral Relations, the partial differential equations of motion are replaced by a set of simultaneous, first-order ordinary differential equations. The number of equations depends on the degree of approximation being attempted. The numerical solution of this set as an initial value problem is not difficult, but a complication arises in that the initial values of some of the dependent variables are not known a priori and must be found by applying regularity conditions at certain singular points of the solution. In the first approximation, one initial value (the initial shock displacement) is not known and one regularity condition must be satisfied. It is found that a high degree of accuracy in the initial shock displacement is needed before an acceptable solution is obtained, but nonetheless, the computing time and programing labor involved are by no means excessive. Higher approximations, involving the application of more regularity conditions, have been used by Belotserkovskii and his associates, but it appears that the computing time and effort required in applying the regularity conditions become large so that, even in the second approximation, the method becomes rather unattractive.

Because of these difficulties, Traugott[4] suggested some years ago a modification which would improve the degree of approximation without introducing any further regularity conditions. Traugott's suggestion was as follows. In the first approximation of the Method of Integral Relations, the governing equations are integrated across the shock layer and the variations of certain flow functions across the layer are approximated linearly. In the nth approximation, the shock layer is divided into n strips and integrations are performed to each strip boundary; the flow function variations are now approximated by an nth order polynomial and, it turns out, there are n regularity conditions to be applied. Traugott pointed out that even if the shock layer is not subdivided, the flow variations can be approximated by cubics if use is made of expressions for the vorticity at the body and at the shock. This improved approximation is achieved without introducing further regularity conditions. The initial formulation of the equations is a little involved but, once accomplished, the computation would be expected to be only slightly longer than the conventional, linear calculation. In his original paper, Traugott used cubics to find corrections to the linear solution, not altogether successfully. Surprisingly, no further work on this approach appears to have been reported.

Some improvement over the first approximation is particularly desirable for the cases of low Mach number flows over bodies with small curvature noses[4] and of low Mach number supersonic jets impinging on flat surfaces.[5] It was therefore decided to formulate and program Traugott's suggestion for the case of a flat-faced, axisymmetric body. Numerical difficulties were encountered on running the program. Very considerable effort was put into checking the formulation and coding: these were found to be correct. The source of the difficulty was finally traced to an inherent instability[6] in the governing differential equations. No way around this difficulty can be seen and it is concluded that Traugott's suggestion, while physically attractive, does not lead to a viable method of calculation, at least in this case. Even in retrospect, there does not seem to be any way in which the instability could have been foreseen.

The purpose of this very brief Note is to prevent others from embarking on the considerable labor of following up Traugott's suggestion, only to arrive at the disappointing result reported here. A report giving a complete formulation of the equations and a discussion of the instability is available from the author.[7]

Received April 20, 1973; revision received June 29, 1973. Helpful comments on the numerical aspects of the problems were made by J. F. Baldwin and J. H. Sims Williams and are gratefully acknowledged.

Index category: Supersonic and Hypersonic Flow.

* Lecturer, Department of Aeronautical Engineering.

References

[1] Moretti, G. and Abbett, M., "A Time-Dependent Computational Method for Blunt Body Flows," *AIAA Journal*, Vol. 4, No. 1, Jan. 1966, pp. 2136–2141.

[2] Belotserkovskii, O. M., "Flow with a Detached Shock Wave about a Symmetrical Profile," *Journal of Applied Mathematics and Mechanics (PMM)*, Vol. 22, 1958, pp. 279–296.
[3] Belotserkovskii, O. M., "Supersonic Gas Flow around Blunt Bodies," TT F-453, 1967, NASA.
[4] Traugott, S. C., "An Approximate Solution of the Direct Supersonic Blunt-Body Problem for Arbitrary Axisymmetric Shapes," *Journal of the Aerospace Sciences*, Vol. 27, 1960, pp. 361–370.
[5] Gummer, J. H. and Hunt, B. L., "The Impingement of a Uniform, Axisymmetric Supersonic Jet on a Perpendicular Flat Plate," *The Aeronautical Quarterly*, Vol. 22, 1971, pp. 403–420.
[6] Fox, L., *Numerical Solution of Ordinary and Partial Differential Equations*, Pergamon Press, Oxford, England, 1962, Chap. 4.
[7] Hunt, B. L., "An Attempted Improvement of the Method of Integral Relations for a Blunt Body in a Supersonic Flow," BLH/7301, 1973, Dept. of Aeronautical Engineering, Bristol Univ., Bristol, England.

Heat Flux Probe as a Flowfield Diagnostic

IRWIN E. VAS*
Princeton University, Princeton, N.J.

IN recent years, several new developments have been made in the area of flowfield diagnostics. A few years ago, a new technique was developed which appeared ideally suited for measurements in a hot gas stream.[1] The "heat flux probe," as it is called, is small in size and can be used with good spatial resolution. The purpose of the present Note is to indicate that some published results obtained with this probe are quite inaccurate even though the probe technique and calibration appear very reasonable.

The heat flux probe operates on the same principle as a constant temperature hot wire. The sensor is a glass tube 0.006 in. o.d. with a wall thickness of 0.001 in. A thin platinum film is deposited on the tube to a length of 0.05 in. In a hot gas stream a balance is made between the convective heating of the sensor by the gas, cooling of the sensor by passing cold gas (or liquid) through it, and resistive heating of the sensor. The electronic circuitry is the same as required for a constant temperature hot wire. The probe and electronic circuitry was purchased from Thermo-Systems.

The initial concept, design and use of the heat flux probe was published in a thesis by Fingerson[1] in 1961. Since that time several other investigators have used this technique.[2–5] It is the results from McCroskey, Horstman, and Vas using this probing technique which will be discussed, and compared with results using other techniques.

A brief review of the three particular studies will initially be made. McCroskey examined the flow about a sharp flat plate in nitrogen at a freestream Mach number (M_∞) of 25, Reynolds number (Re_∞) of 10^4/in. and stagnation temperature (T_{t_∞}) of 2000°K. Flowfield measurements of pitot pressure (p_t) and heat flux ($\dot{q}$) were made. Horstman determined the flowfield about a 3° half-angle cone at $M_\infty \sim 41$. Re_∞/in. $\sim 5.6 \times 10^4$ and $T_{t_\infty} \sim$ 900°K using helium as the test gas. The measurements in the flowfield included pitot pressure and heat flux. Detailed flowfield measurements were made by Vas on a 10° half-angle cone at two conditions: a) $M_\infty \sim 19$, Re_∞/in. $\sim 4.0 \times 10^4$ and $T_{t_\infty} =$ 1700°K; b) $M_\infty \sim 25$, Re_∞/in. $\sim 10^4$ and $T_{t_\infty} = 1850$°K. The traverses were made using four measuring techniques: pitot pressure, heat flux, total temperature (T_t) and density (ρ).

Received April 26, 1973; revision received June 12, 1973. This work was supported jointly by the Aerospace Research Laboratory, Office of Aerospace Research under contract AF33615-70-C-1244 and the U.S. Air Force Office of Scientific Research under contract AF44620-71-C-0032.
Index category: Research Facilities and Instrumentation.
* Senior Research Engineer and Lecturer, Department of Aerospace and Mechanical Sciences, Gas Dynamics Laboratory. Associate Fellow AIAA.

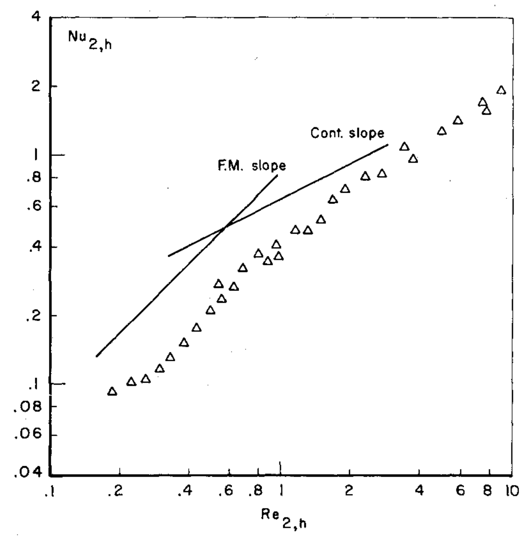

Fig. 1 Heat flux probe calibration.

In addition to the previous measurements on a flat plate in the hypersonic nitrogen gas flow, direct density measurements were performed by Harbour and Lewis[6] at conditions similar to Ref. 2. Recent flowfield measurements were made by Petraites[7] on a flat plate at these same conditions using a pitot probe and total temperature probe. Density and velocity (u) profiles were derived from the measurements of McCroskey and Petraites. The density profiles can be compared with the direct measurements of Harbour and Lewis.

In using the heat flux probe, a calibration was performed by each of the research workers. McCroskey carried out a calibration in the hypersonic helium and nitrogen facilities at Reynolds numbers $Re_{2,h}$ (based on conditions behind the shock and probe diameter) from 1 to 10. To obtain a larger range of Reynolds number a calibration was performed by Vas at $M_\infty \sim 2$ and 6 and stagnation temperatures between 550°K and 1150°K to give Reynolds numbers from 0.15 to 10. This was in addition to calibrations carried out in the hypersonic nitrogen facility. Over a range of Reynolds number the heat transfer to the sensor was measured ($Nu_{2,h}$) and is shown in Fig. 1. The slope of the measurements at the low and high Reynolds number ends approach the free molecule and continuum values, respectively. These measurements give the same trend as a hot wire.[8]

The flowfield characteristics were determined in Ref. 2 using measured pitot pressure and heat flux profiles and the wall static pressure. From these measurements, density and velocity profiles were calculated and are shown in dimensionless form ($\bar{\rho} \equiv \rho/\rho_\infty$, $\bar{u} \equiv u/u_\infty$) in Figs. 2 and 3 for the value of the rarefaction parameter, $[\bar{V} \equiv M_\infty (C)^{1/2}/(Re_{\infty,x})^{1/2}]$, of about 0.12. These profiles are not all made at identical distances x from the leading edge. The height above the surface y is nondimensionalized by the shock layer thickness y_s. Included in Fig. 2 is a direct measurement of density by the electron beam technique.[6] Behind the shock the density obtained by the electron beam technique is higher than that obtained by the other two methods. The profiles obtained by these methods are in agreement in the outer portion of the shock layer, but the results using the heat flux method deviate considerably from those obtained by the total temperature method, as well as the direct measurement, in the inner portion of the shock layer. In the shear layer, the