Hemisphere-Cylinder in Low Supersonic Flow

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Theme

THE application of a hemisphere-cylinder as a flow probe or sensing element is well-known; therefore the flowfield about a hemisphere-cylinder is of major interest. As a first step in understanding the flow phenomena about a hemisphere-cylinder in the transonic and low supersonic flow ranges, a systematic study, both theoretical and experimental, has been conducted about a hemisphere-cylinder at zero incidence for Mach number M_∞ from 0.7 to 2. For $M_\infty > 1$, a direct method of a time-dependent finite difference solution has been used for the hemispherical nose portion from $M_\infty = 1.05$ to 2; theoretical results for $1.05 < M_\infty < 1.2$ have not been previously reported. The method of characteristics is used to compute the cylindrical portion. Agreement between theory and experiment is satisfactory. Comparison with other theoretical work is also presented. The hemisphere solution is also useful in extending to three-dimensional flowfield calculations for a hemisphere-cylinder at incidence.[1]

A relaxation method is also used for M_∞ from 0.7 to 1.3 for comparison. This method satisfactorily predicts the surface pressure for M_∞ from 0.95 to 1.3. Experimental results indicate a strong interaction of shock and boundary-layer separation at $M_\infty \sim 0.8$; therefore, inviscid theory fails to give good prediction of surface pressure.

Contents

The details of this work are contained in Ref. 2. The experimental results for M_∞ from 0.7 to 1.0 will be published separately. Herein concentration will be on the theoretical results and comparisons for $M_\infty \geq 1.05$.

The direct method of a time-dependent finite difference solution to the unsteady Euler's equation as reported in Refs. 3 and 4 is employed in this study. It is assumed that the sonic line is located ahead of the tangent point, and the downstream body shape has no influence on the flow upstream of the tangent point (these assumptions hold approximately for $M_\infty \geq 1.05$). As a result, the hemisphere nose portion may be calculated independently of the cylindrical afterbody portion. The scheme of Ref. 3 is referred to as Code A and Ref. 4 is referred to as Code B. It must be pointed out that even though Code A and Code B have shown to be valid methods for supersonic and hypersonic flows, their applicability for $M_\infty < 1.5$ and over a complete hemisphere has not been investigated. It was found that Code B does not give satisfactory results for $M_\infty \leq 1.2$, whereas Code A can compute to a Mach number of 1.05 but with very slow convergence in time steps. With the solution for the initial portion of the cylinder obtained from the method just described, the GASL method of characteristics code[5] was used to calculate the flow around the cylinder. These results were compared with the Lockheed code[6] for axially symmetric internal flow for a number of cases. The relaxation solution to the steady, full potential equation as reported in Ref. 7, referred to as Code C, was also used in the present study for M_∞ up to 1.3.

The bow shock stand-off distance, Δ/R, where R is the radius of the cylinder, is presented in Fig. 1 for comparison between presently obtained results and available theoretical and experimental data. The fluid considered is air ($\gamma = 1.4$) for all available data in Fig. 1. The agreement for different theoretical predictions is excellent except for that given by the inverse method of Van Dyke and Gordon and the results obtained by Code C (It should be noted that Code C does not use the conservative differencing with a "shock point" operator, therefore the shock is predicted farther forward.) The comparison of the present theoretical results with experiment is also satisfactory except in the very low Mach number range, $1.05 < M < 1.3$, where the experimental data of Ref. 8 are slightly less.

In Fig. 2, the bow shock position as calculated by Codes A and B is compared with other prediction methods for air only. The agreement is satisfactory except for the results of Van Dyke and Gordon, which predict a smaller shock layer. A comparison between the present calculation with the available experimental data is shown in Fig. 3. Again the agreement is satisfactory. It should be mentioned that the sonic lines predicted by different theoretical methods are in good agreement for $M_\infty \geq 1.3$; for $M_\infty = 1.2$, however, the predictions of the sonic lines deviate among the different prediction methods.

The comparison of surface pressure from experiment and theory is show in Fig. 4. It is shown that satisfactorily agreement is obtained for all cases. Therefore, it is concluded that the flowfield over a hemisphere-cylinder can be satisfactorily predicted by inviscid theory for $M_\infty > 1.05$.

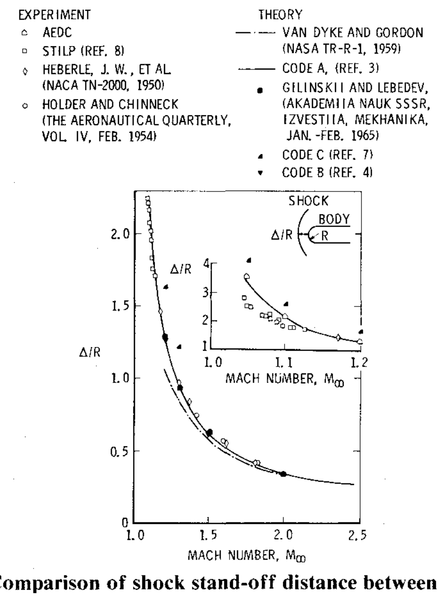

Fig. 1 Comparison of shock stand-off distance between calculation and experiment.

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Index categories: Subsonic and Transonic Flow; Supersonic and Hypersonic Flow.

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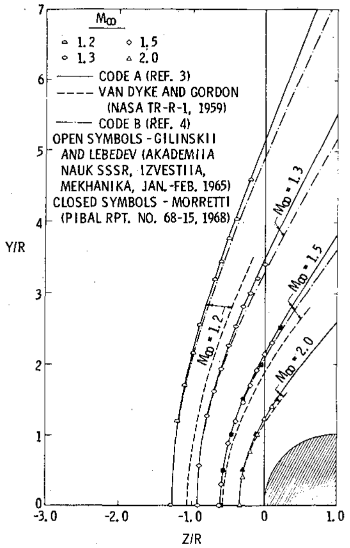

Fig. 2 Comparison of theoretical shock position.

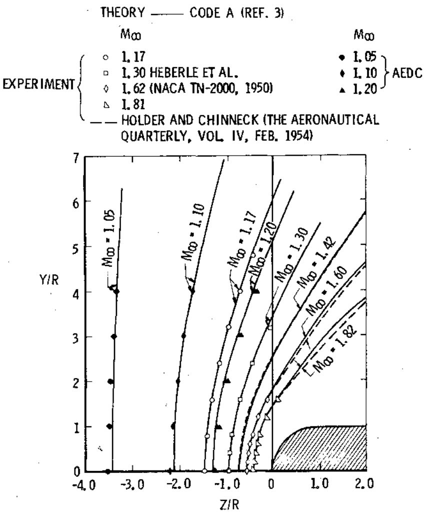

Fig. 3 Comparison of theoretical shock position with experiment, $\gamma = 1.4$.

All computations were carried out by an IBM-370-155 computer. For an indication of the computer time required in performing the calculation, a case of $M_\infty = 1.2$ is taken as an example. Code A requires a mesh of 24×24 and 12,000 time steps which requires about 2.1×10^4 sec. The method of characteristics part of the solution (GASL code) requires about 10 min. Code C with a mesh of 97×97 requires about 60 min for the complete hemisphere-cylinder. It should be noted that when utilizing Code A, the pressure reaches its steady value with many fewer time steps than the shock standoff distance requires. For example, at $M_\infty = 1.1$ the pressure distribution can reach the steady-state value to within 1% at the nosetip and 6% in the tangent point in 2,000 time steps, whereas obtaining the steady shock stand-off distance

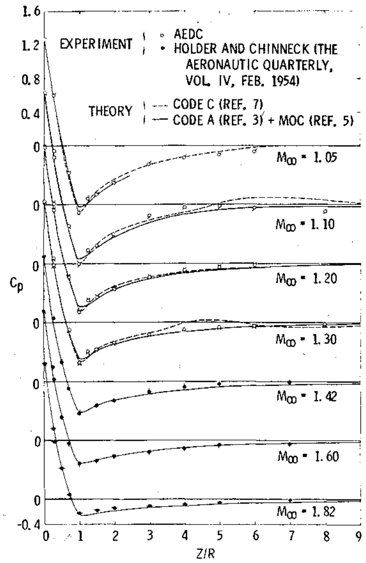

Fig. 4 Comparison of the theoretical pressure distribution with experiments.

requires 22,000 time steps. From the above study, it is recommended that for $1.0 \leq M_\infty \leq 1.3$, Code C be used for predicting the surface pressure, and Code A or B should be used if accurate flowfield in the shock layer is required. For $M_\infty > 1.3$, Code A or B should be used for the blunt nose calculation.

References

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