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EFFECTS OF INTERACTION BETWEEN NORMAL SHOCK
AND BOUNDARY LAYER

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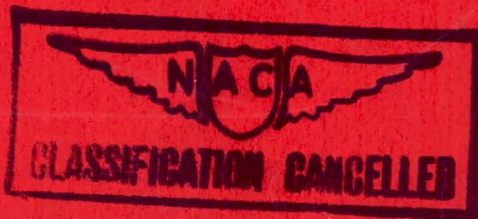
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EFFECTS OF INTERACTION BETWEEN NORMAL SHOCK AND BOUNDARY LAYER

By Coleman duP. Donaldson

SUMMARY

A discussion of the interaction between normal shocks and boundary layers on the basis of experimental evidence obtained in studies of supersonic flows in passages is given. The investigation was made as a result of the inability of the existing normal-shock theory to explain phenomena involving normal shocks that occurred in the presence of boundary layers. Assumptions with regard to the character of the effects of interaction between boundary layer and normal shock are proposed; these assumptions seem to give good agreement with certain experimental results.

INTRODUCTION

For some time, many flows in Laval nozzles and on airfoils at supercritical speeds have been observed that are not explained by the existing normal-shock theory. The shock theory does, however, give good agreement with experimental results in some instances when the shocks being considered are not in contact with a boundary layer. This inconsistency has led to an investigation of the interaction between normal shock and boundary layer.

EFFECTS OF INTERACTION

In order to visualize the interaction, assume that a normal shock occurs in contact with a boundary layer as in figure 1, which might represent one side of a Laval nozzle. The air in the free stream outside the boundary layer is able to get through the shock but the air in the boundary layer of the usually assumed flow, because of its lower momentum, is unable to get through the sharp

adverse pressure gradient that the normal shock presents. This air accumulates and thickens the boundary layer, with the result that the air in the free stream is compressed from its original Mach number to some lower value just in front of the shock. The shock is then less severe and allows the boundary layer to get through more easily. This effect has been confirmed in many instances by experiment. Schlieren photographs at the bases of shock waves show this thickening and compression (see fig. 2), and pressure measurements near shock waves in the presence of a boundary layer have indicated that the shock may produce a smooth rise in pressure rather than a sharp break from one pressure to another (fig. 5 of reference 1).

The discrepancies between experiment and flow calculations neglecting the boundary-layer interaction just discussed led to the following quantitative assumption: The air in the free stream near the boundary layer is compressed to a Mach number close to one in order to permit the boundary layer to negotiate the normal shock. In Laval nozzles, for example, where the flow is nearly one-dimensional, the entire shock seems to be softened to this value; whereas on airfoils, where the flow is more complex, the shock may be so softened only at its base.

In a Laval nozzle, if the Mach number ahead of the normal shock is close to one, then the Mach number immediately behind the shock is also close to one. Any small change of conditions would therefore cause the flow to attain a supersonic velocity again, which would necessitate a further shock. This type of flow may be seen in figure 3, which shows three normal shocks in succession, a phenomenon often observed in the high-speed Laval nozzles that have been investigated at LMAL.

Another result of the softening of the normal shock by the boundary layer is a reduction in the total-head loss experienced by the air in crossing the shock itself. A total-head loss is still present, however, because of mixing of the free stream and boundary layer. Experiments with airfoils at supercritical speeds have indicated that the measured total-head losses through the normal shock near an airfoil may be less than those calculated from normal-shock theory when no softening occurs. Some experimental evidence of this softening effect was indicated by Jacobs in reference 2 and by Stack, Lindsey, and Littell in reference 3, in which it was observed that the

total-head loss behind an airfoil with shock reached a minimum value just outside the region of large total-head loss due to boundary-layer mixing (figs. 18 and 19 of reference 3). The fact that the Mach number behind the base of the normal shock on an airfoil was in the vicinity of one was also noted in reference 3. It appears, however, that no adequate explanation of these departures from normal-shock theory has been offered prior to the present investigation.

Further experimental support of the proposed assumptions was obtained from studies of the flow in a converging-diverging nozzle. The position of the normal shock in the nozzle reported in reference 1 was calculated first by means of the normal-shock theory and then by assuming that the shock had been so softened that the Mach number at the shock was 1. These methods of calculation are explained in the appendix. The results presented in figure 4 show that the normal-shock theory alone is greatly in error but that the proposed treatment of the effects of interaction between boundary layer and normal shock gives good agreement with experimental results.

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APPENDIX

CALCULATION OF POSITION OF NORMAL SHOCK IN A NOZZLE

Normal-shock theory.- The position of the normal shock in the nozzle is assumed; thus the pressure ratio across the shock is known. The pressure in the chamber that will make the pressure at the exit of the jet equal to atmospheric pressure is then calculated. Adiabatic expansion up to the shock and adiabatic compression after the shock are assumed.

Proposed treatment.- Approximation of softening is given by the assumption that the Mach number at the position of the shock is 1 and that the pressure at this point is 0.528 times the chamber pressure (ratio of the heat capacities taken as 1.4). For example, assume

$$p_o = 1.5p_a$$

and

$$p_s = 0.528p_o$$

where

p_o pressure of air at rest, assumed herein as chamber pressure

p_a atmospheric or exit pressure

p_s pressure at shock

Then

$$p_a/p_s = 1.263$$

The relation between the area at the shock and the area at the exit is 0.957 obtained from figure 5 for a value of

$$\begin{aligned} p/p_0 &= 0.528 \times 1.263 \\ &= 0.667 \end{aligned}$$

This condition is found from figure 6 to exist at a point 0.690 inch from the end of the nozzle. This point is plotted in figure 4.

REFERENCES

1. Kantrowitz, Arthur, Street, Robert E., and Erwin, John R.: Study of the Two-Dimensional Flow through a Converging-Diverging Nozzle. NACA CB No. 3D24. April 1943.
2. Jacobs, Eastman N.: Methods Employed in America for the Experimental Investigation of Aerodynamic Phenomena at High Speeds. Misc. Paper No. 42, NACA, 1936.
3. Stack, John, Lindsey, W. F., and Littell, Robert E.: The Compressibility Burble and the Effect of Compressibility on Pressures and Forces Acting on an Airfoil. Rep. No. 646, NACA, 1938.

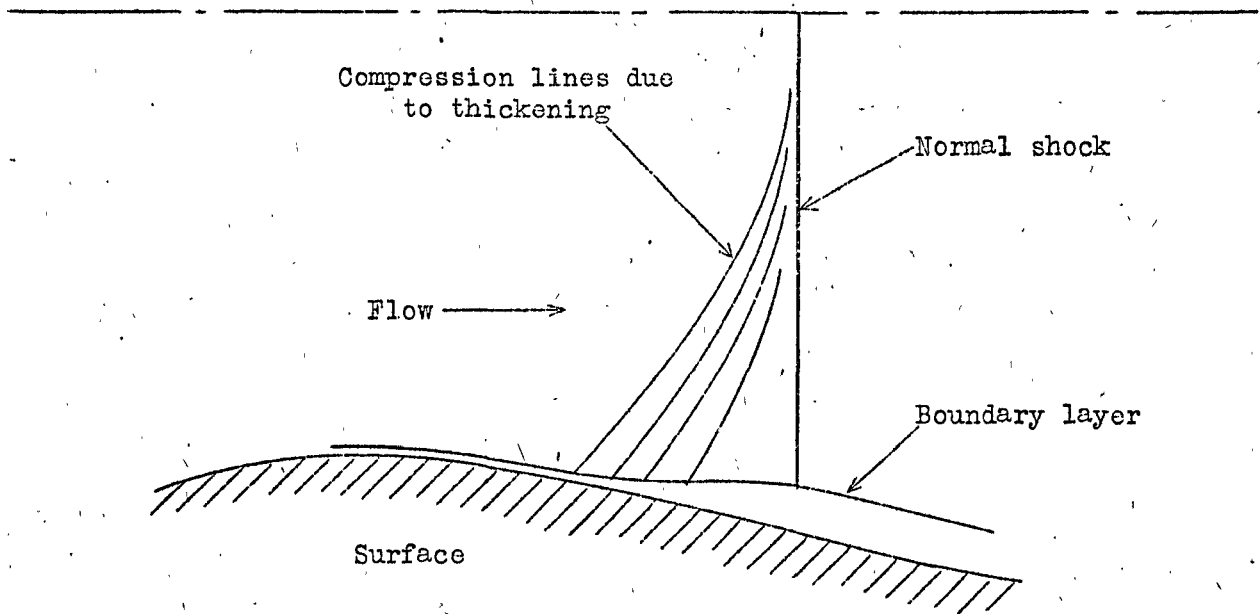


Figure 1.- Thickening of boundary layer before the shock and resulting compression of free stream.

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Figs. 2,3

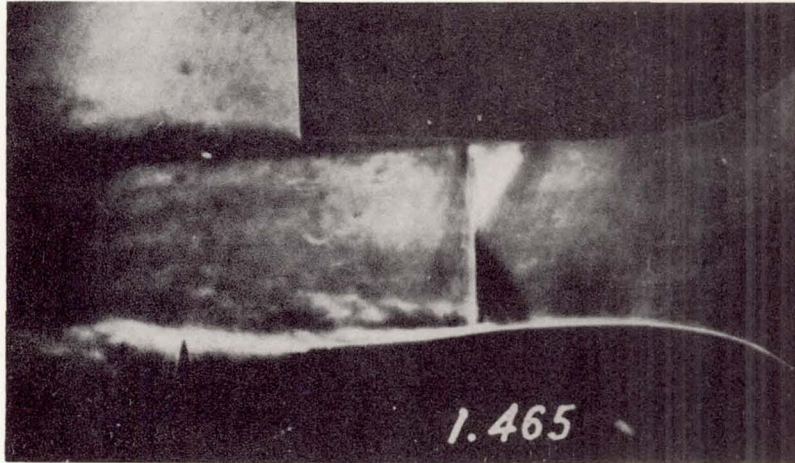


Figure 2.- Schlieren photograph of flow in a converging-diverging nozzle showing compression due to thickening of the boundary layer before a normal shock. Knife edge horizontal. (From fig. 2 of reference 1.)

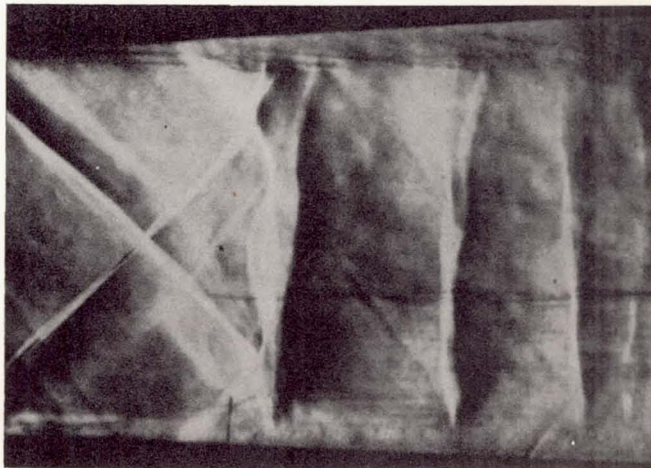


Figure 3.- Schlieren photograph of flow in a diverging passage showing three normal shocks in succession. Knife edge vertical.

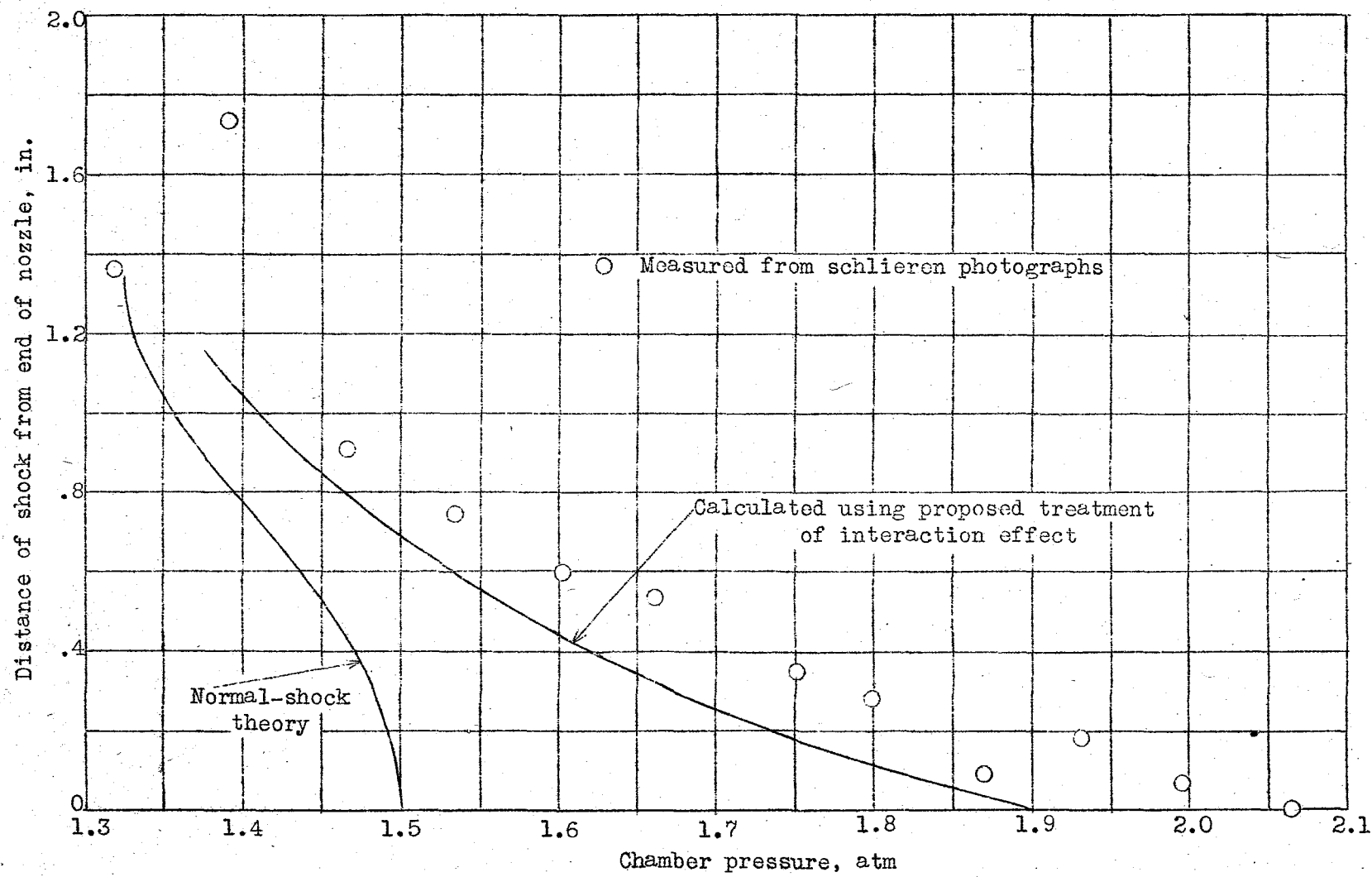


Figure 4.- Position of normal shock in the converging-diverging nozzle of reference 1 calculated by two methods and measured experimentally.

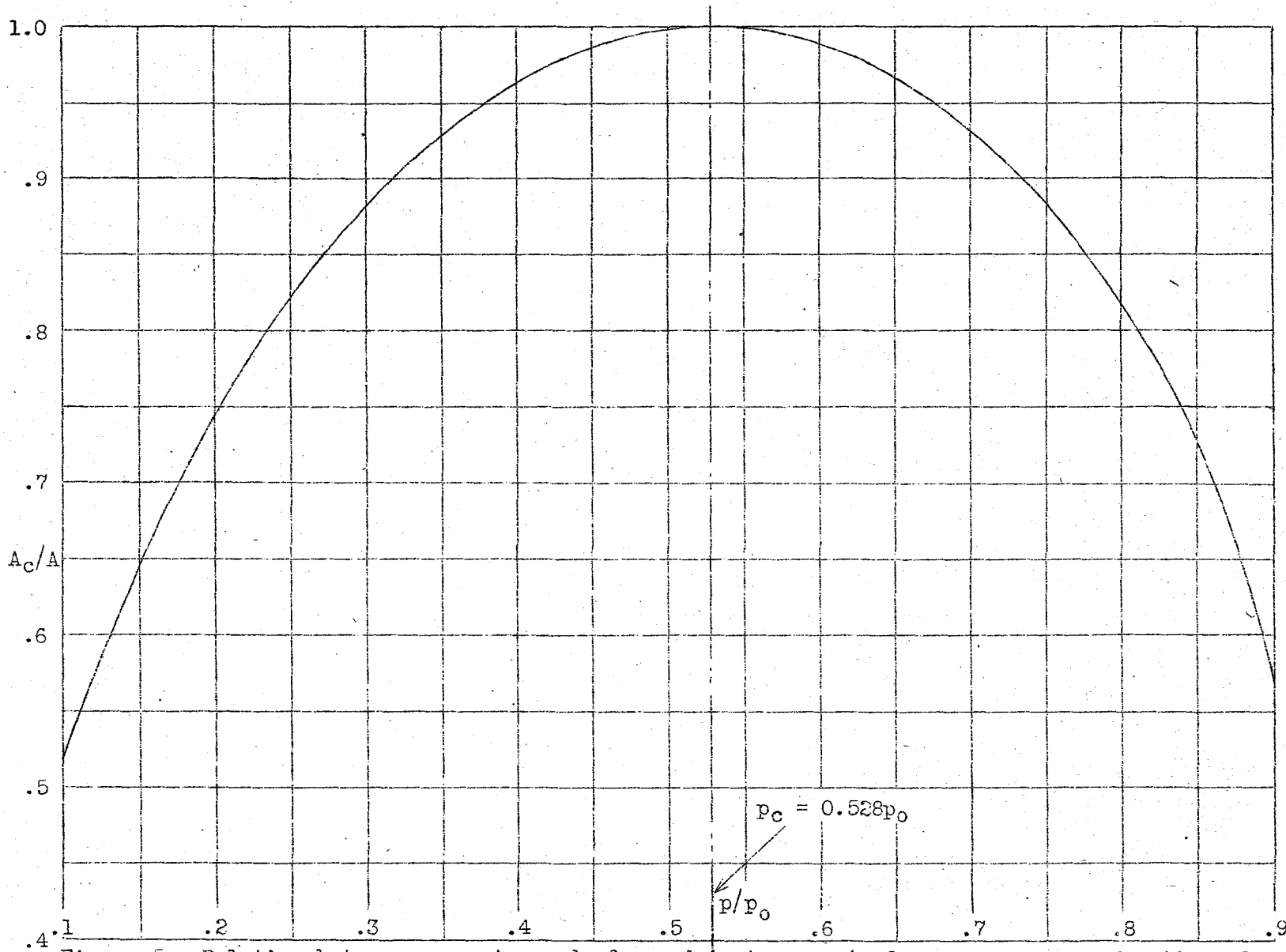


Figure 5.- Relation between area at speed of sound A_c to area A of pressure p as a function of ratio of p to the pressure of the air at rest p_0 .

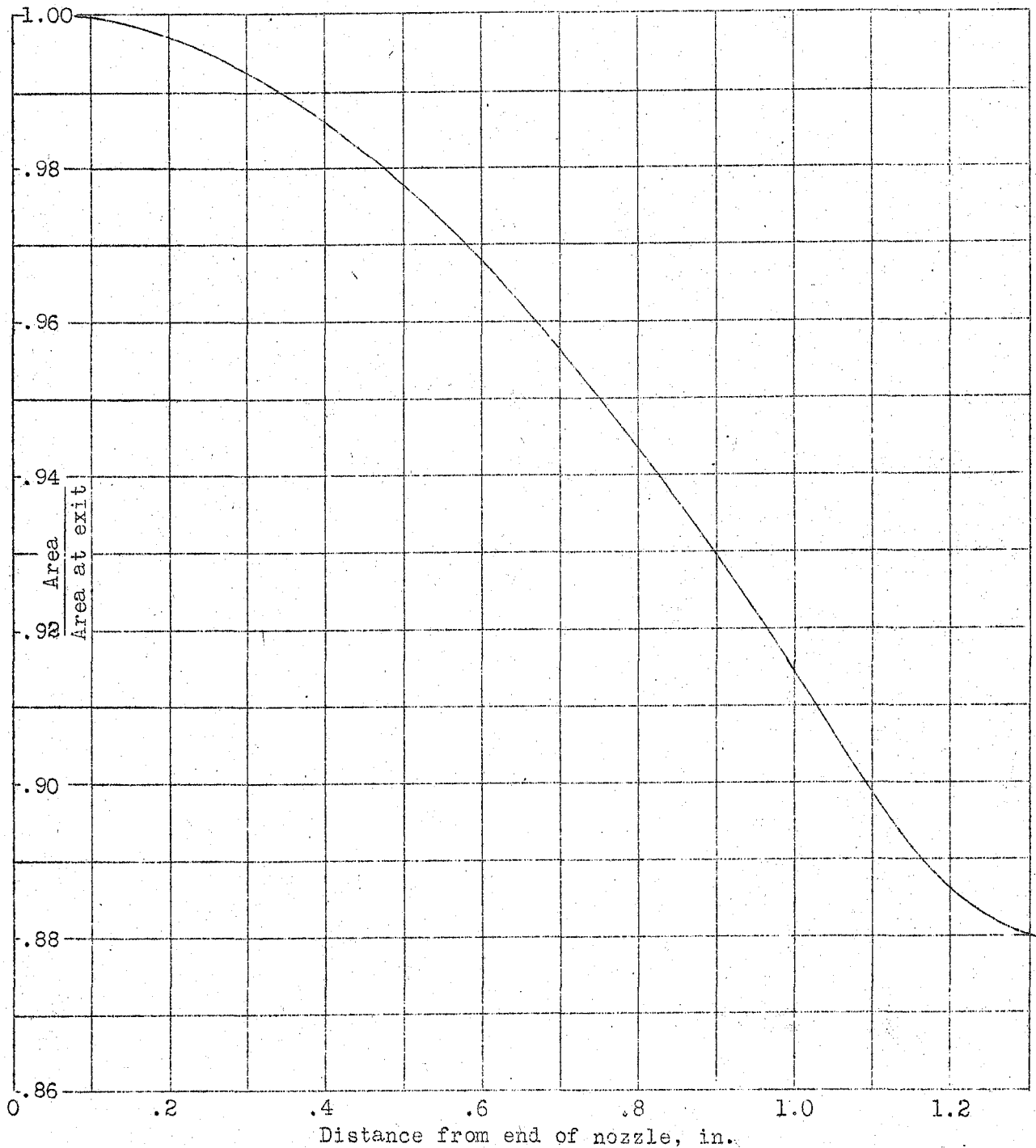


Figure 6.- Properties of converging-diverging nozzle of reference 1 for determining position of shock.

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