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TECHNICAL NOTE 2720

EFFECTS OF ASPECT RATIO ON AIR FLOW AT HIGH SUBSONIC MACH NUMBERS

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EFFECTS OF ASPECT RATIO ON AIR FLOW AT HIGH SUBSONIC MACH NUMBERS[1]

By W. F. Lindsey and Milton D. Humphreys

SUMMARY

Schlieren photographs were used in an investigation to determine the effects of changing the aspect ratio from infinity to 2 on the air flow past a wing at high subsonic Mach numbers. The results indicated that the decreased effects of compressibility on drag coefficients for the finite wing are produced by a reduction in the compression shock and flow separation.

INTRODUCTION

Previous investigations at high subsonic speeds have shown that the adverse effects of compressibility on the force characteristics of wing sections in two-dimensional flows are greatly alleviated for wings of low aspect ratio. Although reference 1 showed the over-all effect of decreasing the aspect ratio of a wing, it did not provide any concrete information on the changes involved within the flow.

An investigation has been conducted in the Langley rectangular high-speed tunnel to provide information on the differences between the flow past wings of infinite aspect ratio and of aspect ratio 2. The information was obtained in the form of schlieren photographs of the flow at high subsonic Mach numbers. Both wings had the profile of the NACA 0012 airfoil.

SYMBOLS

A aspect ratio

α angle of attack

[1]Supersedes the recently declassified NACA RM L8G23, "Effects of Aspect Ratio on Air Flow at High Subsonic Mach Numbers" by W. F. Lindsey and Milton D. Humphreys, 1948.

C_L wing lift coefficient

C_D wing drag coefficient

M free-stream Mach number

APPARATUS AND TESTS

The tests were conducted in the Langley rectangular high-speed tunnel which is a nonreturn, induction-type tunnel having a 4-inch by 18-inch test section. The models tested were wings of rectangular plan form having NACA 0012 profiles. The 2-inch-chord wings had aspect ratios of infinity and 2.

The wing of infinite aspect ratio completely spanned the 4-inch dimension of the tunnel. The wing was supported at both ends by the glass end plates which formed part of the tunnel walls. The wing of aspect ratio 2 was securely attached to one of the glass end plates and extended one semispan (2 in.) into the air stream. The support system did not interfere with either the air flow or the optical field.

Schlieren photographs taken with an exposure of approximately 2 microseconds were obtained for the models at angles of attack of 0° and 4° at Mach numbers between 0.5 and 0.9. The corresponding Reynolds number range was between 5.3×10^5 and 7.6×10^5.

RESULTS AND PRECISION

The variations in lift and drag coefficient with Mach number for the wings were obtained from reference 1 and are presented in figures 1 and 2. The data are presented to correlate and supplement the schlieren photographs of the flow (figs. 3 to 6) obtained in the present investigation.

The data for the wing of infinite aspect ratio (figs. 1 to 3 and 5) were corrected for tunnel-wall effects by the methods of reference 2. The tunnel-wall corrections of the data for the wing of aspect ratio 2 are of the order of one-half percent and therefore are neglected. Data within a Mach number increment of 0.025 of the choked condition are not included.

Theoretical optics (reference 3) indicates that the contrast obtained in a schlieren photograph as a result of a given density gradient normal to the knife edge is directly proportional to the extent of the gradient traversed by the light. Errors could be expected, therefore, to occur in

a direct comparison of schlieren photographs of the flow past these wings of different spans. The deductions are based upon ideal optical conditions including a fixed sensitivity of the system which, in practice, is difficult to maintain. Further analysis of this problem, including some experimental studies, indicates that comparisons between intensities of shocks in flows past wings of different spans may not be valid; however, the boundary-layer conditions and the presence and extent (normal to stream) of shocks as shown by the photograph should agree with the conditions existing in the field of observation.

DISCUSSION

The data of reference 1 indicated for an angle of attack of 0° that approximately zero-lift coefficients were obtained for the symmetrical wings of aspect ratios 2 and infinity over the Mach number range (fig. 1). For the zero-lift condition ($\alpha = 0^{\circ}$), the wing of low aspect ratio was shown to have a higher Mach number for drag rise and lower drags at Mach numbers above 0.79 than had the infinite wing or section (fig. 2). The schlieren photographs of the flow past the two wings at zero lift show strong disturbances on the infinite wing at a Mach number of 0.78 (fig. 3(c)), whereas comparable disturbances are not obtained on the wing of low aspect ratio until a Mach number of 0.83 has been reached or exceeded (fig. 4(d)). The occurrence of the strong disturbances or shocks in figures 3(c) and 4(d) corresponds to the Mach number regions of drag rise. The figures thus illustrate that the reduction in aspect ratio produced a delay in the Mach number for drag rise as a result of a delay in shock formation. The delay in shock formation on the wing of low aspect ratio can be attributed to the known effect of the inflow at the tip of a finite wing in producing a decrease in the induced velocities over the wing.

At zero lift and Mach numbers of 0.83 and 0.86 the drag coefficients of the wing of low aspect ratio are approximately 30 percent of the values for the wing of infinite aspect ratio (fig. 2). The corresponding schlieren photographs (figs. 3(e) and 3(f), 4(d) and 4(e)) show that the decrease in drag coefficient for the wing of low aspect ratio is a result of the occurrence of less extensive compression shocks and less severe flow separation than was encountered by the infinite wing.

The variations in shock with Mach number at an angle of attack of 4° for both wings are shown in figures 5 and 6. The lift coefficients for the two wings generally differed over the Mach number range (fig. 1). At a Mach number of 0.83, lift coefficients of approximately 0.15 were obtained for both wings ($\alpha = 4^{\circ}$) and the drag for the wing of low aspect ratio is approximately one-third of the drag for the infinite wing

(fig. 2). The flow photographs for this lifting condition (figs. 5(e) and 6(d)) corroborate the results obtained for the wing at zero lift and indicate that the decrease in drag coefficient is a result of reductions in shock and separation losses.

Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va., July 22, 1948

REFERENCES

1. Stack, John, and Lindsey, W. F.: Characteristics of Low-Aspect-Ratio Wings at Supercritical Mach Numbers. NACA Rep. 922, 1949. (Formerly NACA TN 1665.)

2. Allen, H. Julian, and Vincenti, Walter G.: Wall Interference in a Two-Dimensional-Flow Wind Tunnel, with Consideration of the Effect of Compressibility. NACA Rep. 782, 1944. (Formerly NACA ARR 4K03.)

3. Schardin, Hubert: Toepler's Striation Method. Principles for Its Application and Quantitative Evaluation. British Air Ministry Translation No. 249, May 30, 1935.

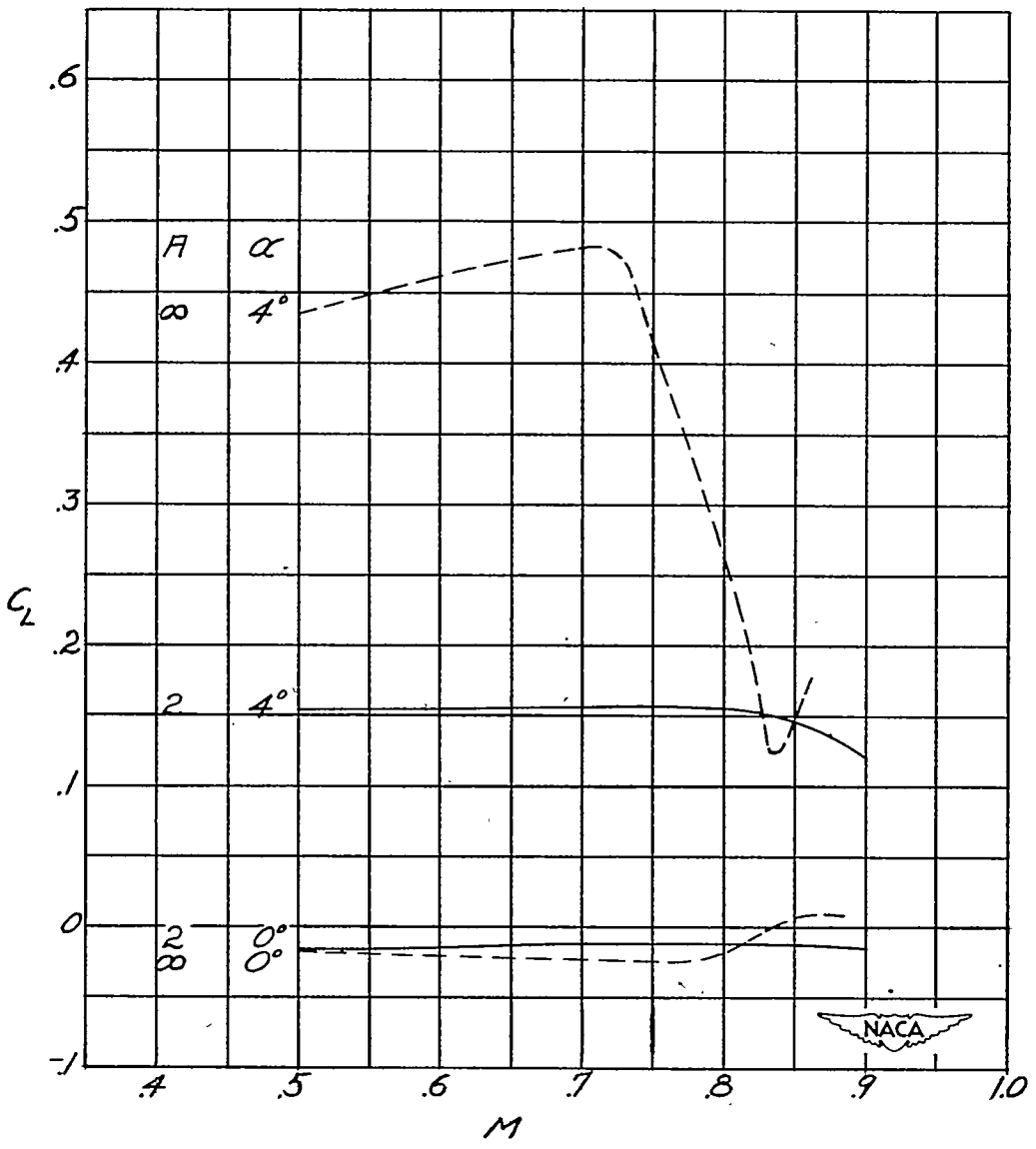


Figure 1.- Effect of aspect ratio on the variation of lift coefficient with Mach number for wings having the NACA 0012 profile. (Data from reference 1.)

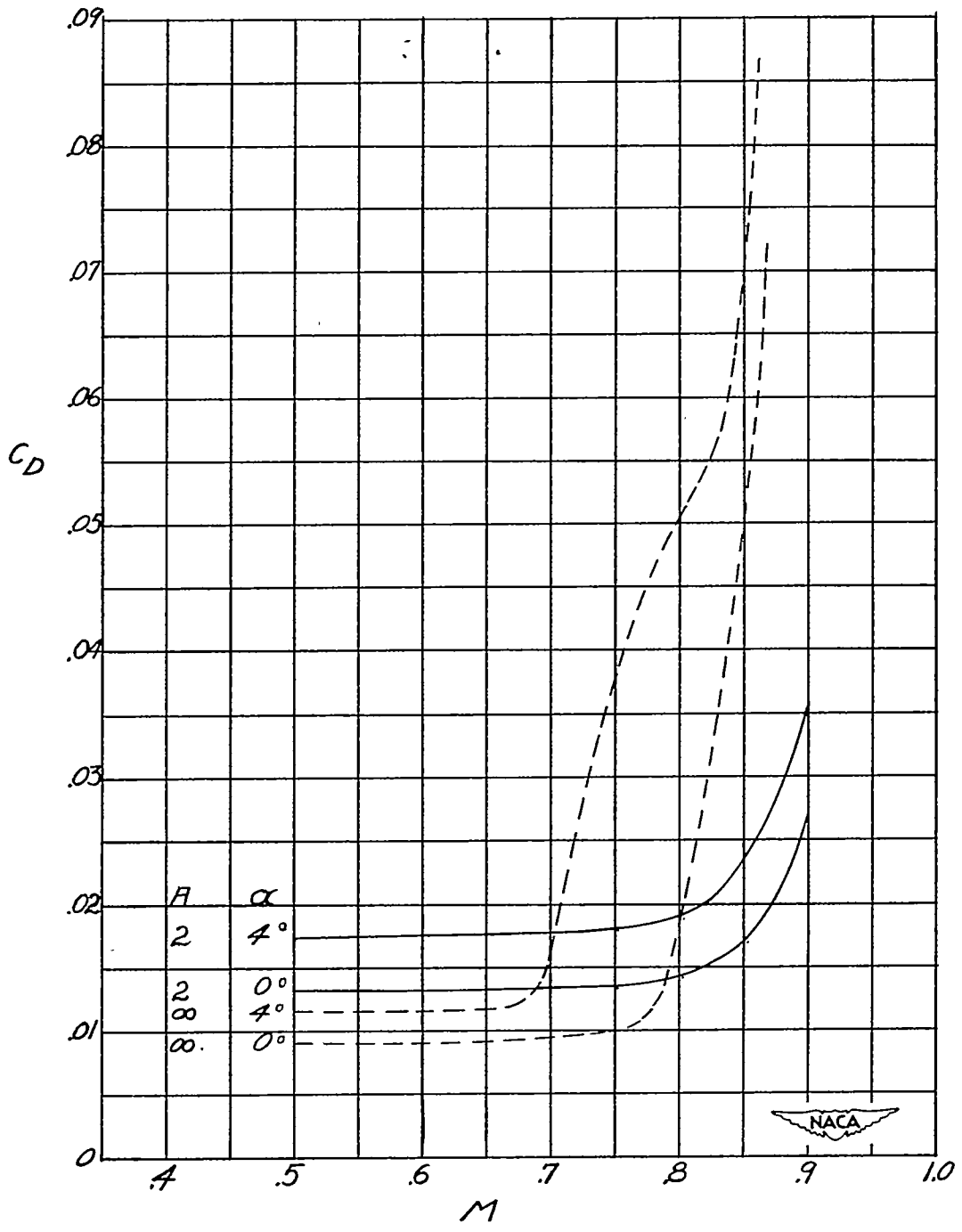
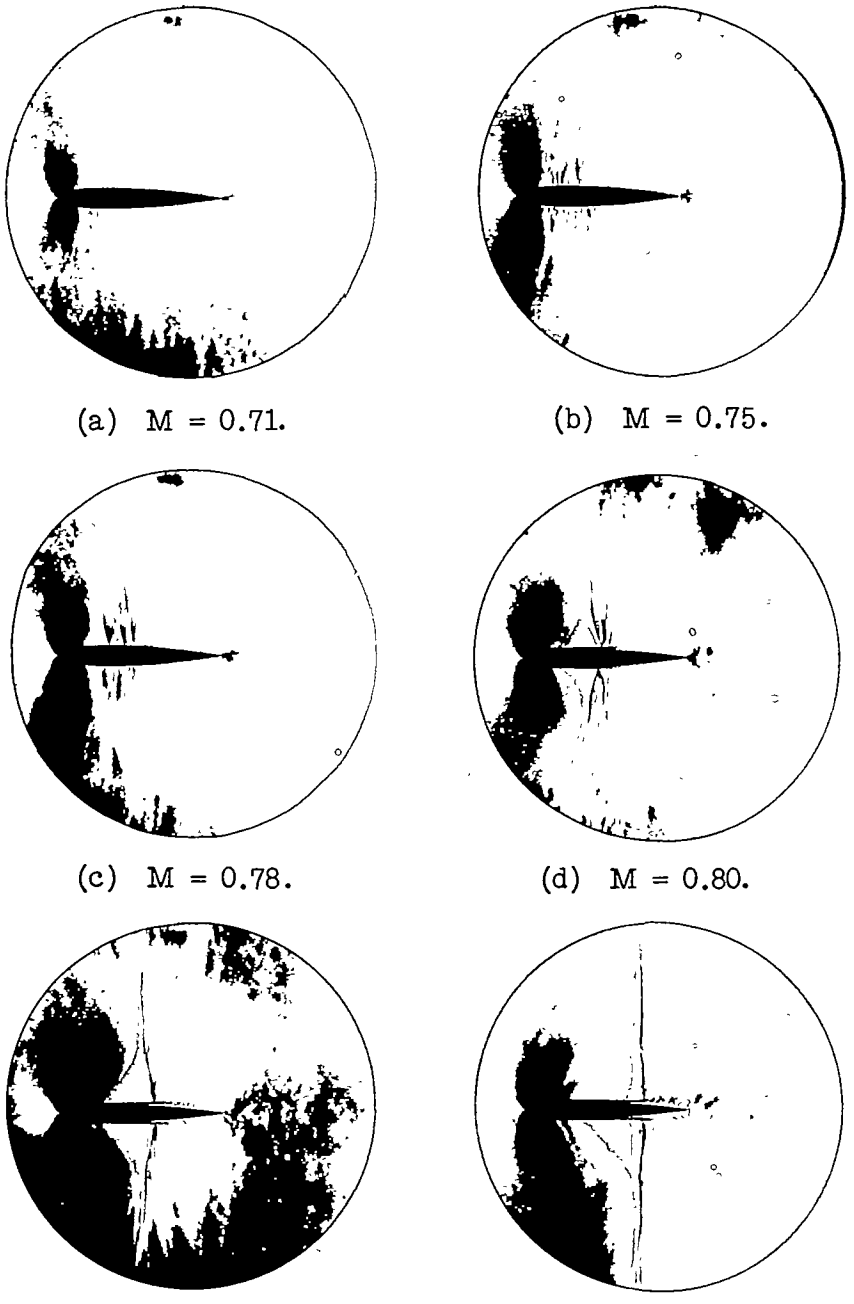


Figure 2.- Effect of aspect ratio on the variation of drag coefficient with Mach number for wings having the NACA 0012 profile. (Data from reference 1.)



(a) M = 0.71.

(b) M = 0.75.

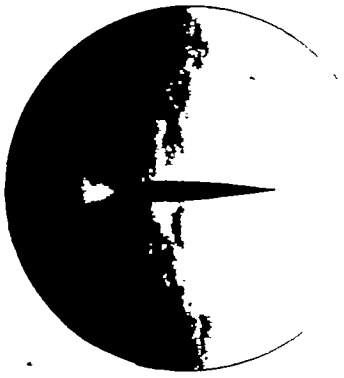
(c) M = 0.78.

(d) M = 0.80.

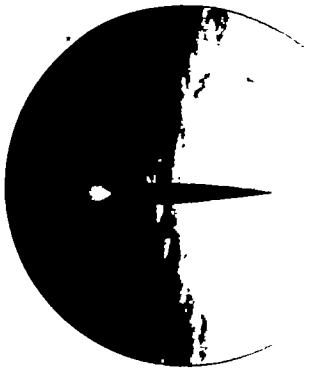
(e) M = 0.83.

(f) M = 0.86.

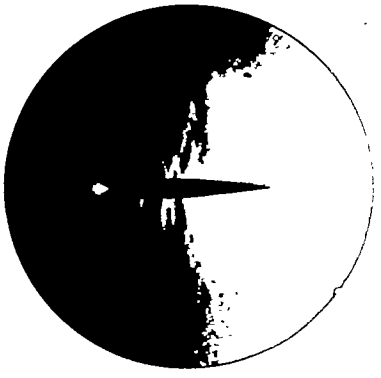
Figure 3.- Schlieren photographs of the flow on the NACA 0012 wing of infinite aspect ratio at 0° angle of attack.



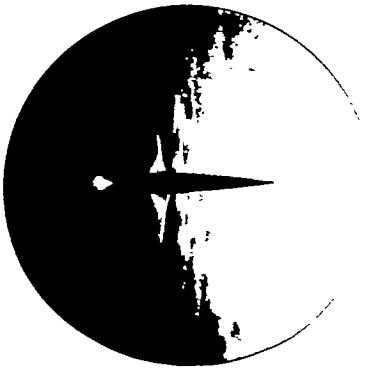
(a) M = 0.63.



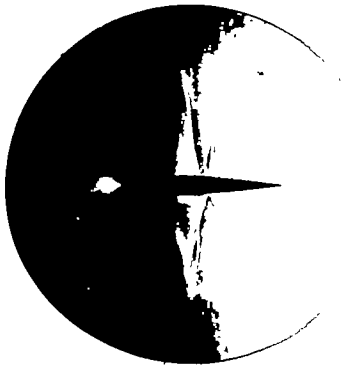
(b) M = 0.73.



(c) M = 0.78.



(d) M = 0.83.



(e) M = 0.86.

Figure 4.- Schlieren photographs of the flow on the NACA 0012 wing of aspect ratio 2 at 0° angle of attack.

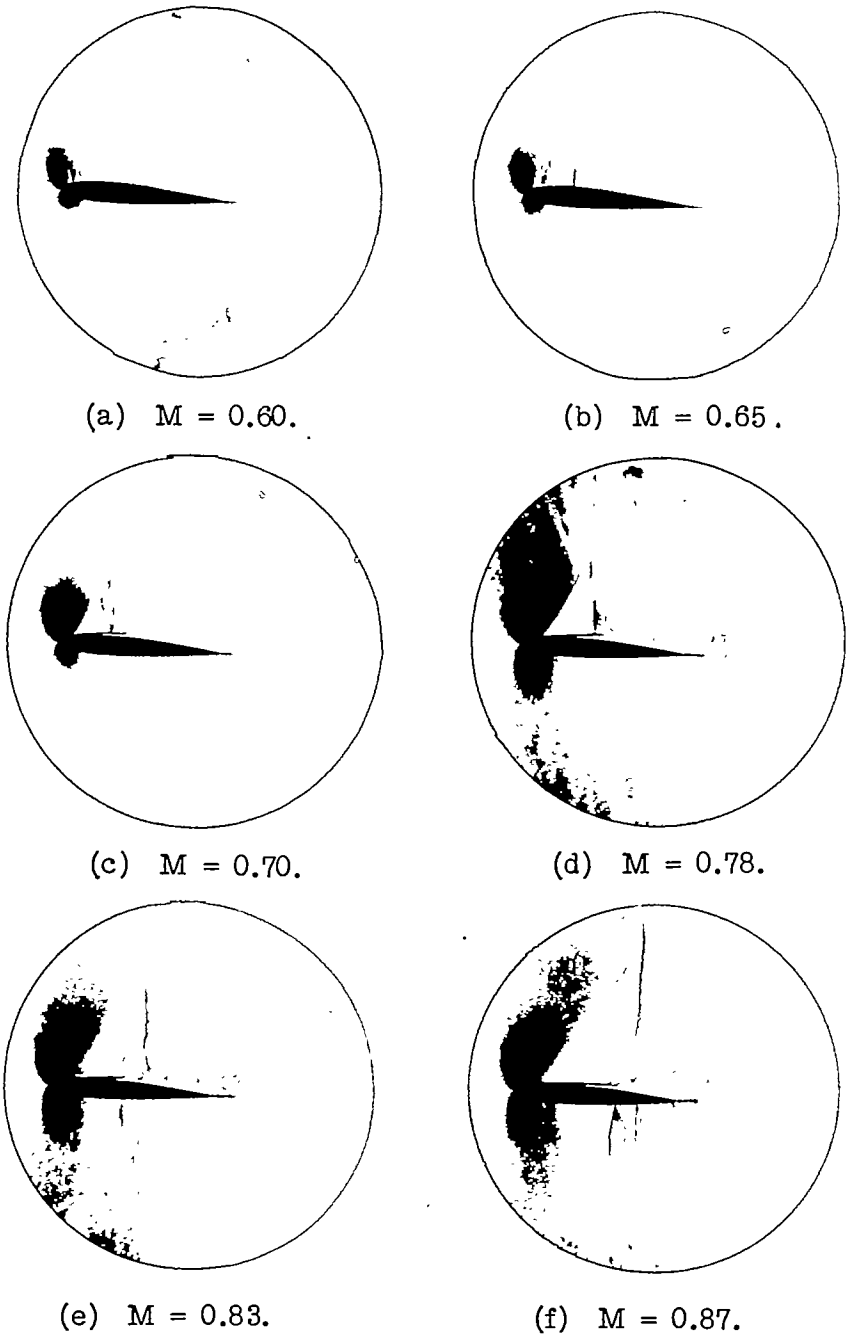
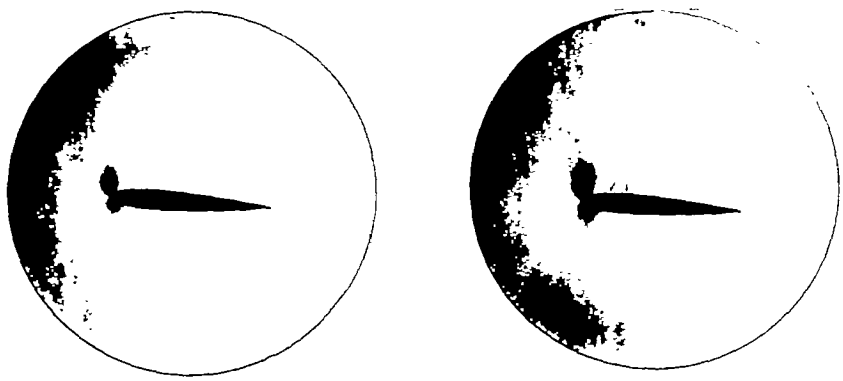
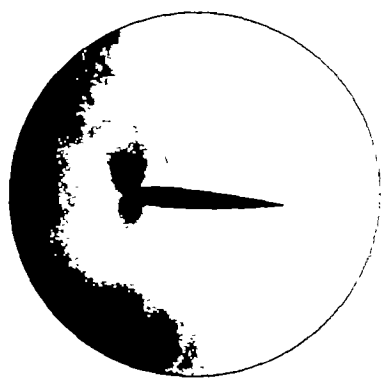


Figure 5.- Schlieren photographs of the flow on the NACA 0012 wing of infinite aspect ratio at 4° angle of attack.

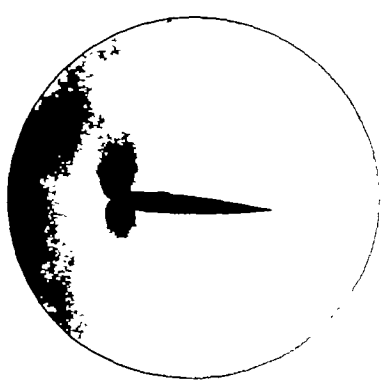


(a) M = 0.63.

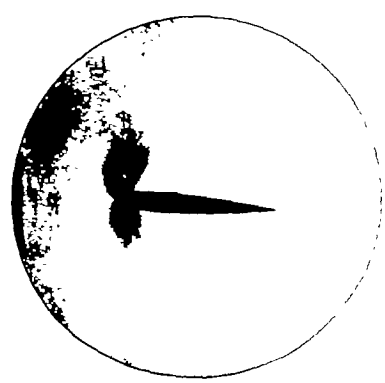
(b) M = 0.73.



(c) M = 0.78.



(d) M = 0.83.



(e) M = 0.86.

Figure 6.- Schlieren photographs of the flow on the NACA 0012 wing of aspect ratio 2 at 4° angle of attack.

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