# NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE

No. 1371

## CONSIDERATIONS OF THE TOTAL DRAG OF SUPERSONIC AIRFOIL SECTIONS

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## SUMMARY

The results of calculations of the viscous and pressure drags of some two-dimensional supersonic airfoils at zero lift are presented. The results indicate that inclusion of viscous drag alters many previous results regarding the desirability of certain airfoil shapes for securing low drags at supersonic speeds. At certain Reynolds and Mach numbers, for instance, a circular-arc airfoil may theoretically have less drag than the previously advocated symmetrical wedge-shape profile; although under different conditions, the circular-arc airfoil may have the higher drag.

Drag calculations for 6-percent-thick symmetrical circular-arc and double-wedge airfoils are presented for Mach numbers of 1.35 and 1.6 and Reynolds numbers from $10^5$ to $10^8$. Unseparated flows are considered and approximate corrections for boundary-layer and shock-wave interaction are applied only to the momentum thickness at the trailing-edge shock wave. The theory of viscous supersonic flows will have to be extended before a more exact analysis of the drag is possible.

## INTRODUCTION

Recent experiments indicate that airfoil shapes heretofore considered good for supersonic speeds may in fact be inferior to profiles having higher pressure drags. In order to understand better the behavior of airfoils at supersonic speeds, it is desirable to eliminate and explain apparent contradictions between experiment and theory. The drag of thin airfoils may be considered as the sum of the pressure and viscous drags. Although a great deal of work has been done on the calculation of the pressure drag, little theoretical work has been attempted on the calculation of the viscous drag. The purpose of this paper is to consider the separate effects of viscous and pressure drag on the total drag for two thin airfoil sections.

The pressure drag as determined by the approximate linearized relations is a minimum for a symmetrical wedge-shape airfoil of a given thickness ratio with maximum thickness located at the 50-percent chord (reference 1). The more exact methods of references 2 and 3 show that other locations of the maximum thickness for the symmetrical wedge-shape profile have lower pressure drag. On all these flat-side profiles, however, the velocity gradient is zero over the flat leading and trailing sections of the airfoils and, hence, boundary-layer transition and viscous drag may be somewhat similar to that of a flat plate. Viscous drags lower than those usually obtained for a flat plate can be obtained by using a falling-pressure gradient to increase the extent of the laminar flow. It appears possible then that a curved airfoil, such as a double circular arc, which has a favorable pressure gradient may also have more laminar flow than a flat-side airfoil and therefore less viscous drag over a certain range of Mach and Reynolds numbers. Although the theoretical pressure drag of the circular-arc profile is higher than that of the symmetrical wedge shapes, the total drag (viscous and pressure) of the circular arc might possibly be the lower for certain conditions.

In order to demonstrate the effects of viscous and pressure drag on the total drag, calculations were made for two, 6-percent-thick airfoil shapes - a double circular arc and a symmetrical wedge with maximum thickness at the 50-percent chord. The calculations were made for Mach numbers of 1.35 and 1.6 and covered a range of Reynolds numbers from $10^5$ to $10^8$.

The present paper is intended to serve only as a preliminary study of the total drag of a two-dimensional airfoil and neglects many factors. A more complete analysis would require consideration of the effects of separation, interaction between the boundary layer and the shock waves, and angle of attack. Existing theories need considerable development before all these factors can be included.

## DISCUSSION

The drag of a two-dimensional airfoil at zero lift at supersonic speeds is assumed to be the sum of the pressure drag and the drag due to viscosity. The pressure distribution is calculated by assuming the absence of the boundary layer, and the shock drag is then readily determined from the pressure distribution. The boundary-layer momentum thickness corresponding to this pressure distribution is calculated, and the viscous drag is then determined from the momentum thickness at the trailing edge of the airfoil.

The method of reference 4 was used to determine the shock drag as well as the local values of Mach number, velocity, density, and other conditions along the airfoil. This method requires the existence of attached shock waves and is therefore restricted to sharp-nose airfoils. In the absence of a boundary layer and flow separation the calculations are accurate for the wedge airfoil and are a close approximation for the circular-arc airfoil. In a more complete analysis the pressure distribution should be adjusted for changes in the flow pattern caused by boundary-layer thickness and by sudden changes in slope of the boundary-layer displacement thickness due to transition or separation.

The calculated pressure distribution is used to compute the boundary-layer momentum thickness along the airfoil by the method of reference 5. Reference 5 assumes the following: The skin-friction coefficient is independent of Mach number and pressure gradient; a fixed velocity profile independent of pressure gradient may be used; the Prandtl number is 1; and no heat conduction occurs. Both laminar and turbulent boundary layers may be computed approximately by this method with the use of the appropriate constants given in the reference. For the present calculations, transition from laminar to turbulent flow was considered to occur suddenly. Therefore, in order to compute the momentum thickness along the surface of the airfoil, parameters corresponding to laminar flow were used from the leading edge to the point of transition, and parameters corresponding to turbulent flow were used in the equations from the point of transition back to the trailing edge.

Transition from laminar to turbulent flow is dependent on such factors as Mach number, Reynolds number, pressure gradient, stream turbulence, and surface roughness. Determination of transition for a given body and flow conditions is therefore difficult. Lees (reference 6) and Schlichting (reference 7) have investigated the effect of Mach number and velocity gradient, respectively, on the stability of the laminar boundary layer by assuming vanishingly small disturbances. In the absence of a stability theory which accounts for both velocity gradients and compressibility, it was necessary to combine the work of these references. The boundary-layer thickness for neutral stability was considered to be the value for a flat plate in incompressible flow multiplied by factors to correct for Mach number and pressure gradient. The criterion used to estimate transition in the present investigation was consideration of the neutral stability of the laminar boundary layer. The notation $N = 1$ is used to denote transition occurring when the boundary-layer momentum thickness reaches the value for neutral stability. If the airfoil is well faired and in a stream of low initial turbulence, transition need not occur until the boundary layer is thicker. For example, the transition curve $N = 2$ corresponds to transition at a point where the boundary-layer momentum thickness is twice that for neutral stability.

Before a more extended treatment is possible, the stability theory must be developed to include the combined effects of compressibility and pressure gradient. A more complete analysis will require an investigation to determine the existence of transition regions, or shock waves (and hence changes in the pressure distribution) due to transition or separation. In the present paper values of the momentum thickness of the laminar boundary layer were calculated along the surface of the airfoil and compared with the corresponding values of momentum thickness for neutral stability. The intersection of the curves through these points gives a possible point of instability for a given Mach number and Reynolds number. The stability criterions indicate that the critical part of the boundary layer from consideration of possible transition is well forward on the curved airfoil.

The momentum thickness increases in passing through the trailing shock. Approximate corrections for this effect have been supplied to the authors by Mr. Neal Tetervin of the Physical Research Division. The assumptions are that the boundary-layer momentum equation applies and that the length over which the pressure rise takes place on the surface is so short that the skin friction can be neglected. The correction becomes:

$$\frac{\theta_2}{\theta_1} = \left(\frac{V_1}{V_2}\right)^{H_{c_{av}}+2} \frac{\rho_1}{\rho_2}$$

where $\theta$ is the momentum thickness, $V$ is the velocity tangential to the boundary layer, $H_{c_{av}}$ is the average value for compressible flow of the ratio of displacement thickness to momentum thickness across the shock, and $\rho$ is the density at the edge of the boundary layer. The subscripts 1 and 2 refer to conditions before and after the shock, respectively.

In the actual case, the interaction of the shock wave and boundary layer results in an increase in the momentum thickness, an increase in the displacement thickness, and a change in the velocity profile and may cause flow separation. The effect of separation is to reduce the pressure drag and increase the viscous drag. At low Reynolds numbers there may be considerable separation resulting in a total drag less than the theoretical shock drag (reference 8). The method used was chosen for lack of a more exact method.

## PRESENTATION OF FIGURES

Figure 1(a) gives the viscous-drag coefficient of a flat plate as a function of Reynolds number based on chord. The transition curves for a Mach number $M_o$ of 1.35 are numbered according to the ratio of boundary-layer momentum thickness at transition to the boundary-layer thickness for neutral stability (N = 2 to 10). Figures 1(b) and 1(c) give the corresponding curves of viscous drag at $M_o = 1.35$ for a 6-percent-thick circular-arc airfoil and 6-percent-thick double-wedge airfoil, respectively. The transition occurs at a very much higher Reynolds number for the curved airfoil than for the flat plate or wedge airfoil. As the Reynolds number is increased, an abrupt rise in the drag for the circular-arc airfoil is noticed. The reason for the sudden drag rise is that the ratio of local boundary-layer thickness to the thickness for local neutral stability (as given by Schlichting's theory) reaches a maximum well forward on this airfoil so that the critical part of the airfoil (as regards stability or transition) is also well forward. A complete wing would not necessarily experience a sharp drag rise as the Reynolds number is increased since transition may occur at different Reynolds numbers over different sections. Parts of the transition curves for the wedge are dashed to indicate possible theoretical errors introduced by the sudden expansion at the midchord. The actual curves may be somewhat to the left of the dashed curves (closer to flat-plate conditions).

Figures 2(a) and 2(b) give the viscous-drag coefficient at $M_o = 1.6$ for the 6-percent-thick circular-arc and wedge airfoils, respectively. These curves show again that transition occurs at a much higher Reynolds number on the circular arc than on the wedge; however, transition on both types of airfoil occurs at a lower Reynolds number at $M_o = 1.6$ than at $M_o = 1.35$. (See fig. 1.)

Figures 3(a) and 3(b) compare the viscous-drag coefficients of the airfoils at $M_o = 1.35$ and at $M_o = 1.6$, respectively, for a constant transition number N = 5. These figures indicate that the circular-arc airfoil may have a much lower viscous-drag coefficient than the wedge over a certain range of Reynolds numbers.

The pressure-drag coefficients are determined to be as follows (from reference 4):

| Airfoil (6 percent thick) | Pressure-drag coefficient | |
|---|---|---|
| | $M_o = 1.35$ | $M_o = 1.60$ |
| Circular arc | 0.0218 | 0.0158 |
| Double wedge | .0160 | .0116 |

When these pressure-drag coefficients are added to the viscous-drag coefficients of figures 3(a) and 3(b) (or in general to figs. 1 and 2), curves are obtained for the total-drag coefficients. See figures 4(a) and 4(b). For $N = 5$ the circular-arc airfoil has at least slightly more drag than the wedge for all Reynolds numbers investigated at $M_o = 1.35$; however, the analysis shows that the wedge may have the more drag over a certain Reynolds number range at $M_o = 1.6$. The actual comparison obtained depends on the value of $N$ used for the analysis. Since $N$ is a function of many parameters, such as surface finish and stream turbulence, it is difficult to assign $N$ a proper value for a given profile. Some information concerning the effect of stream turbulence on transition at low speeds is given in reference 9.

This paper demonstrates the need for including both Mach number and Reynolds number, as well as such factors as stream turbulence and surface finish where possible, in papers of experimental work since these factors may influence the interpretation of the data for full-scale application.

It must be kept in mind that the present paper has compared the drags of two airfoil shapes of the <u>same thickness ratio</u>. A comparison of airfoils giving the <u>same structural strength</u> or <u>stiffness</u> would be more favorable to the curved airfoil.

## CONCLUDING REMARKS

An analysis has been presented which serves as a preliminary study of the total (viscous and pressure) drag of supersonic airfoil sections at zero lift. Within the limitations of the present paper certain conclusions have been drawn:

The relative pressure drags of airfoils at supersonic speeds is in general different from the relative total drags even at zero lift. The airfoil shape for minimum drag varies with Reynolds number, Mach number, turbulence, surface finish, and other factors, and is not necessarily the shape that would give minimum theoretical pressure drag.

Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va., May 9, 1947

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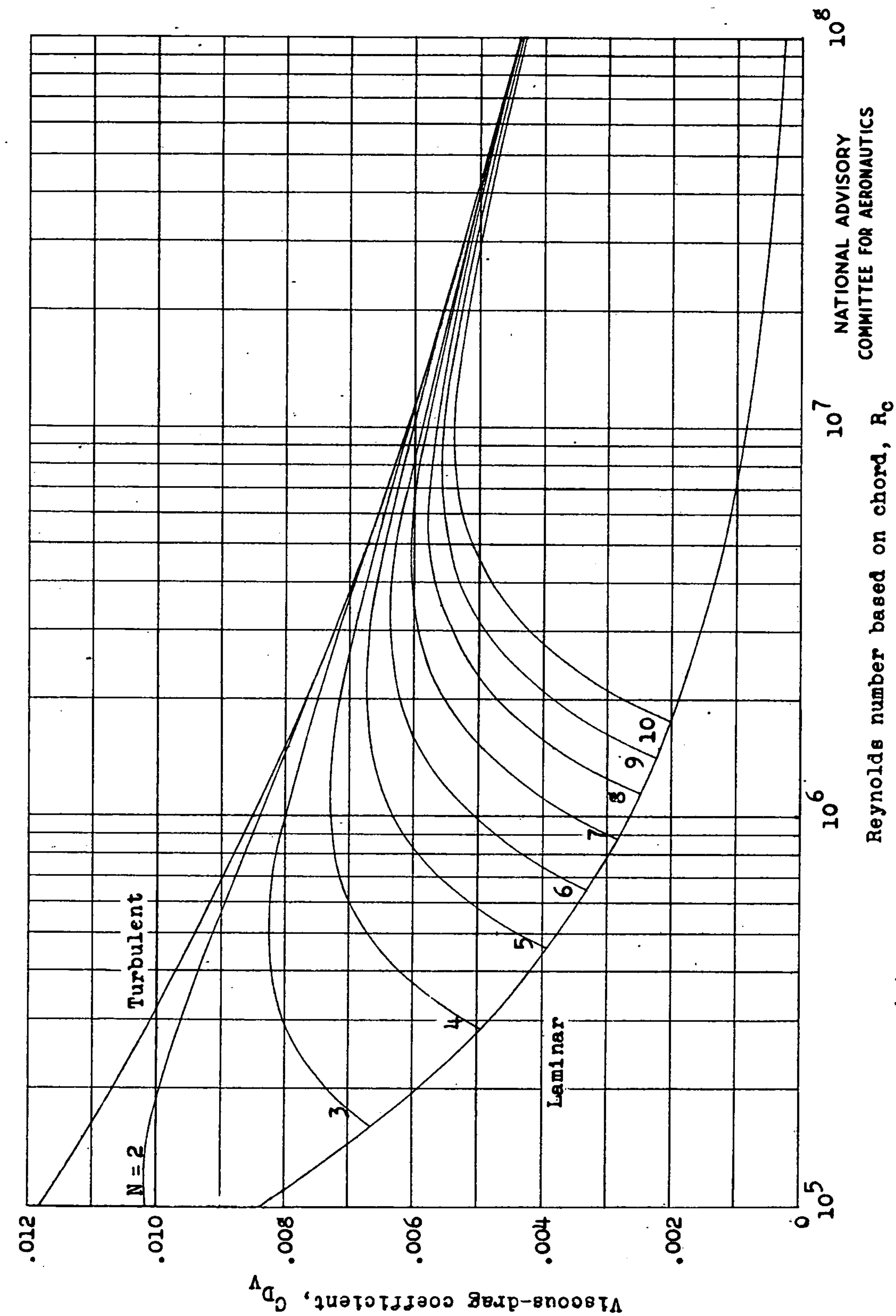
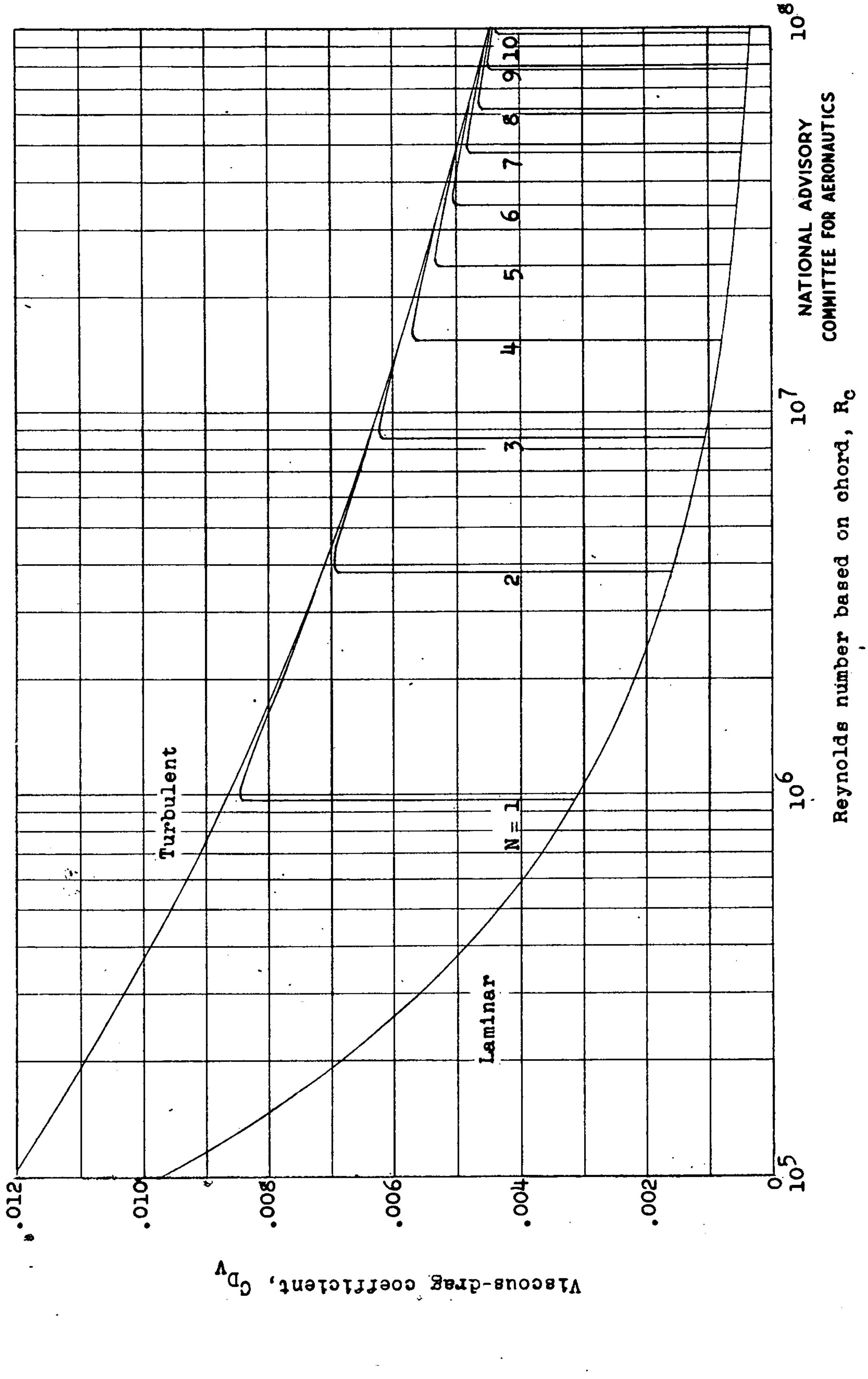


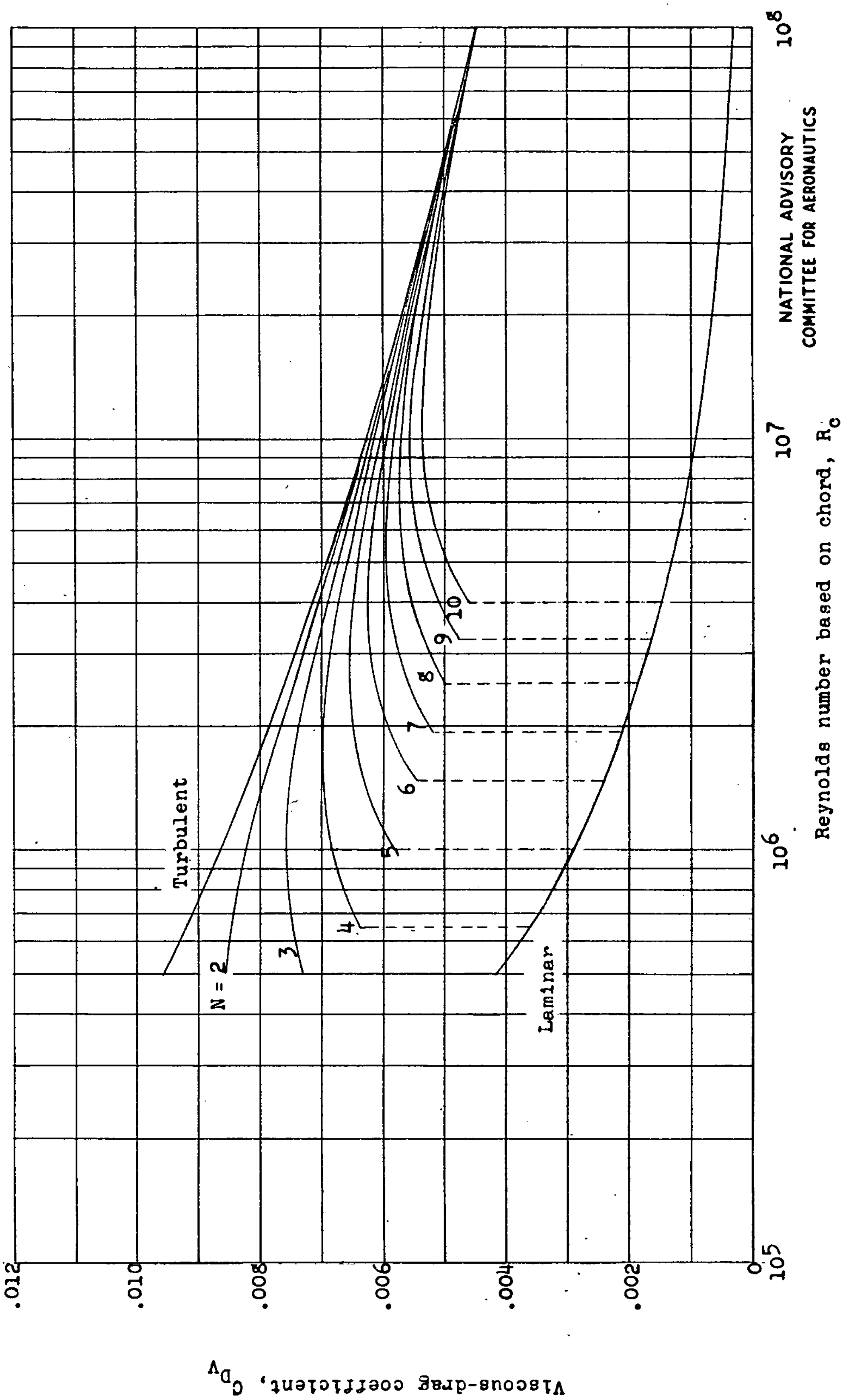

(a) Flat plate.

Figure 1.- Effect of Reynolds number on viscous drag. $M_o = 1.35$.



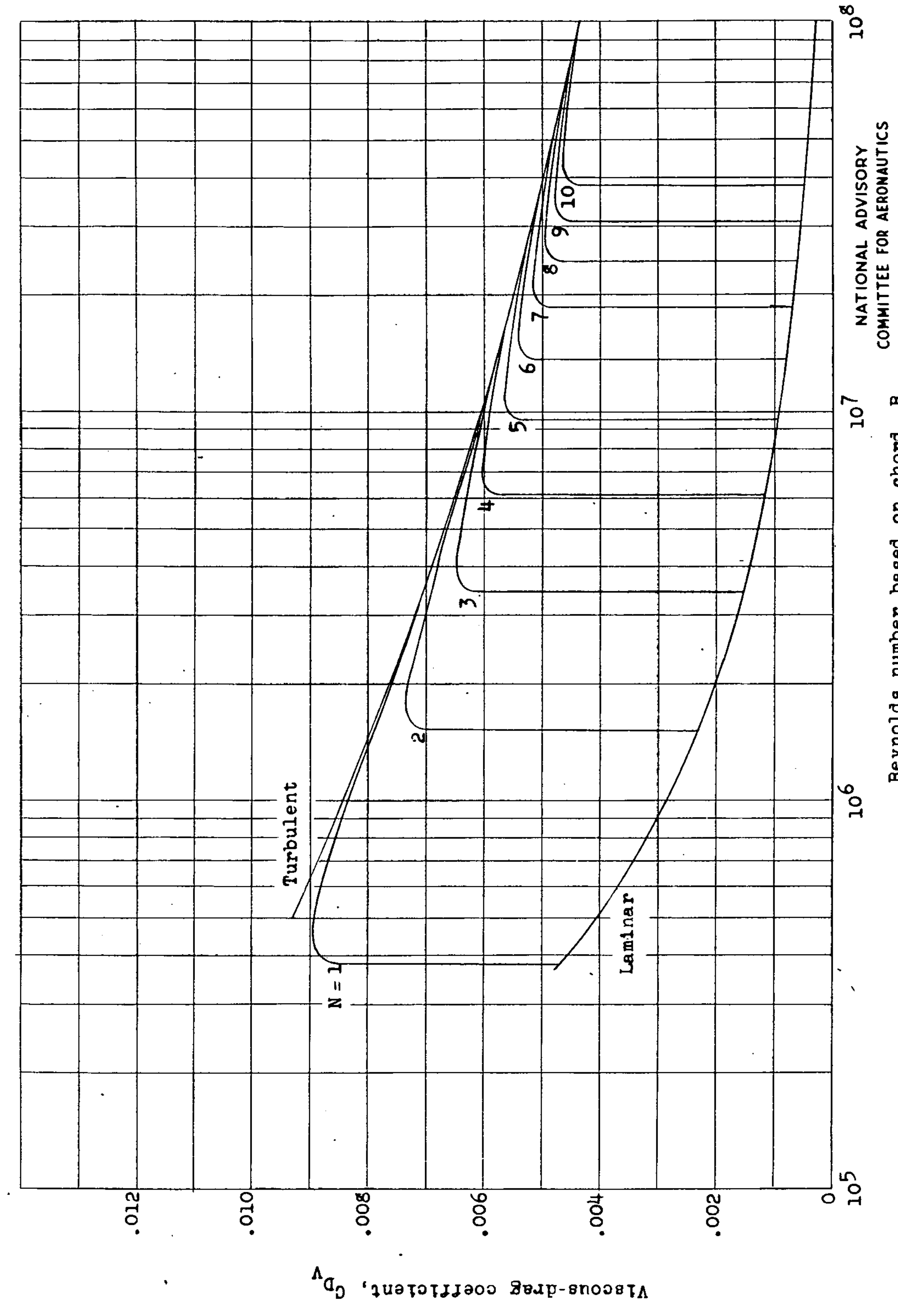

(b) Circular-arc airfoil.

**Figure 1.- Continued.**



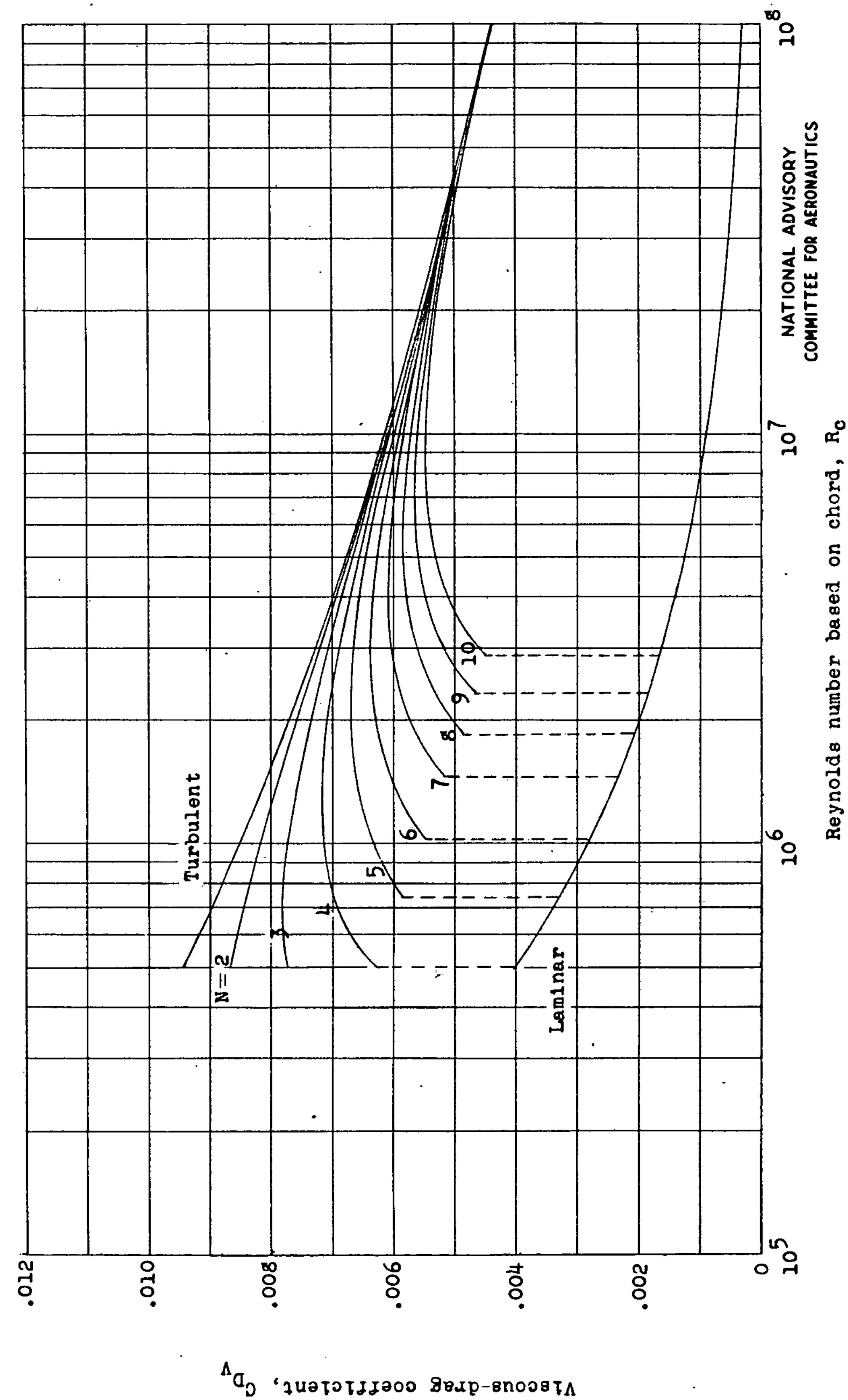

(c) Wedge airfoil.

Figure 1.- Concluded.



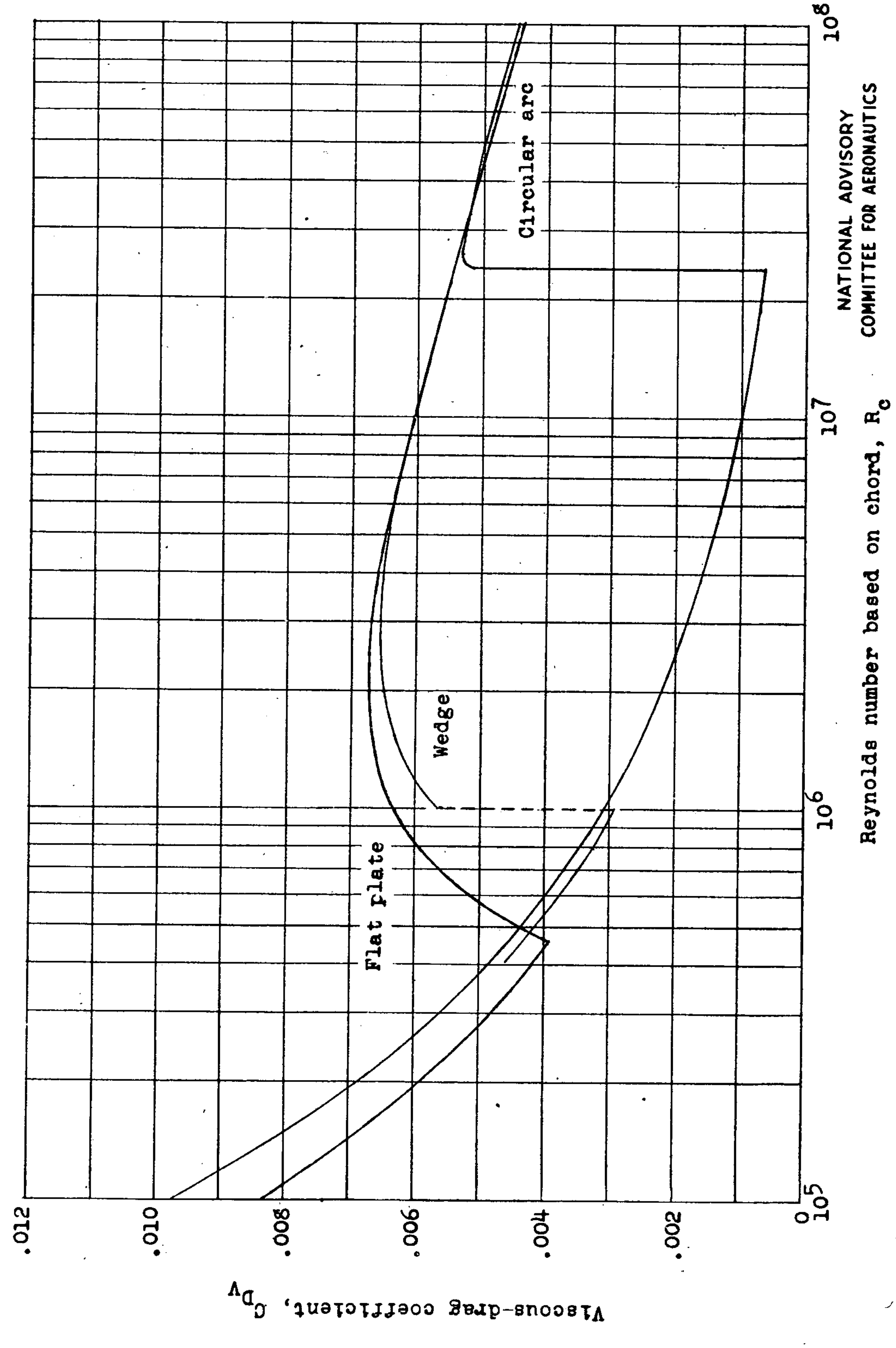

(a) Circular-arc airfoil.

Figure 2.- Effect of Reynolds number on viscous drag. $M_o = 1.6$.




(b) Wedge airfoil.

**Figure 2.- Concluded.**

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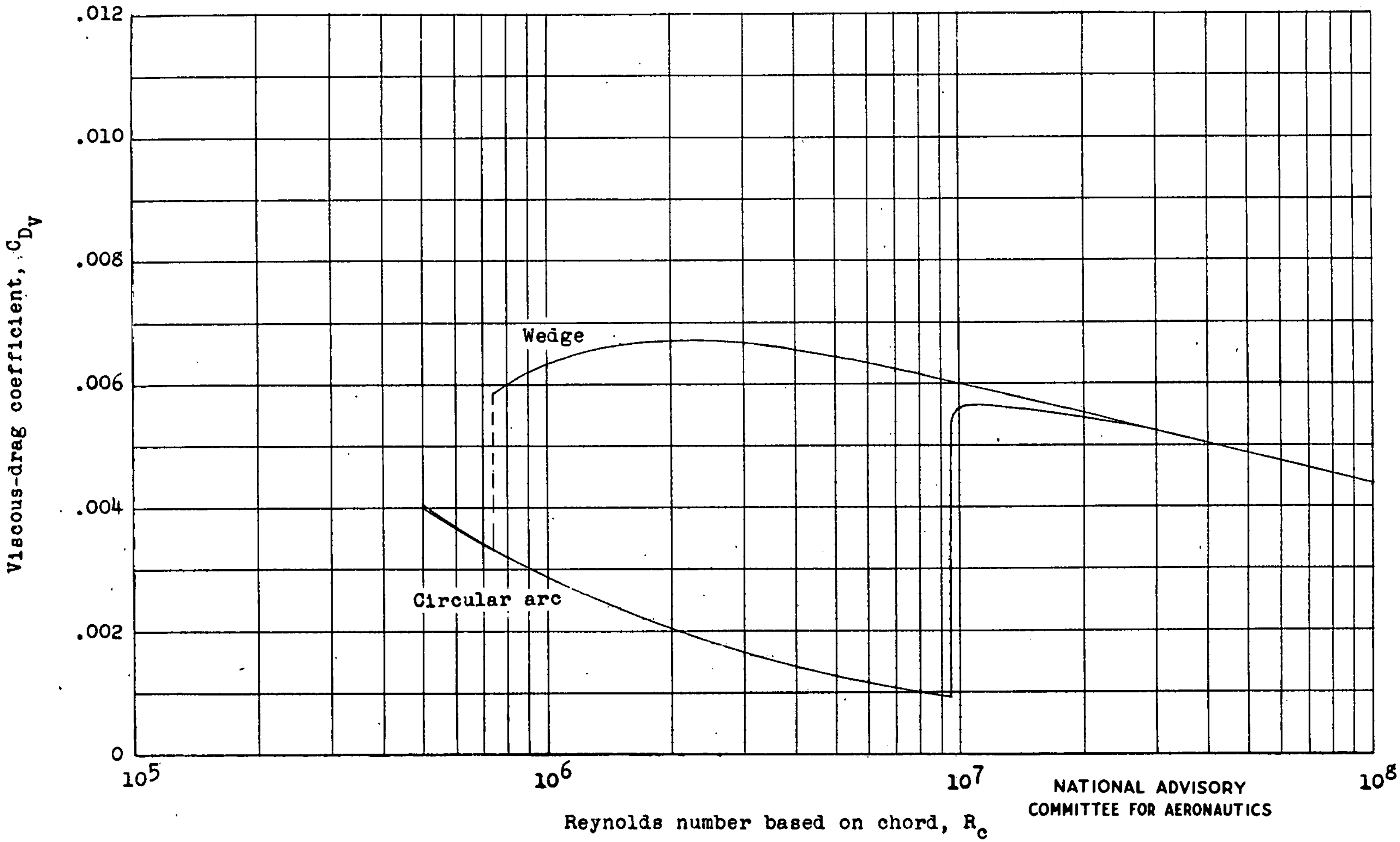
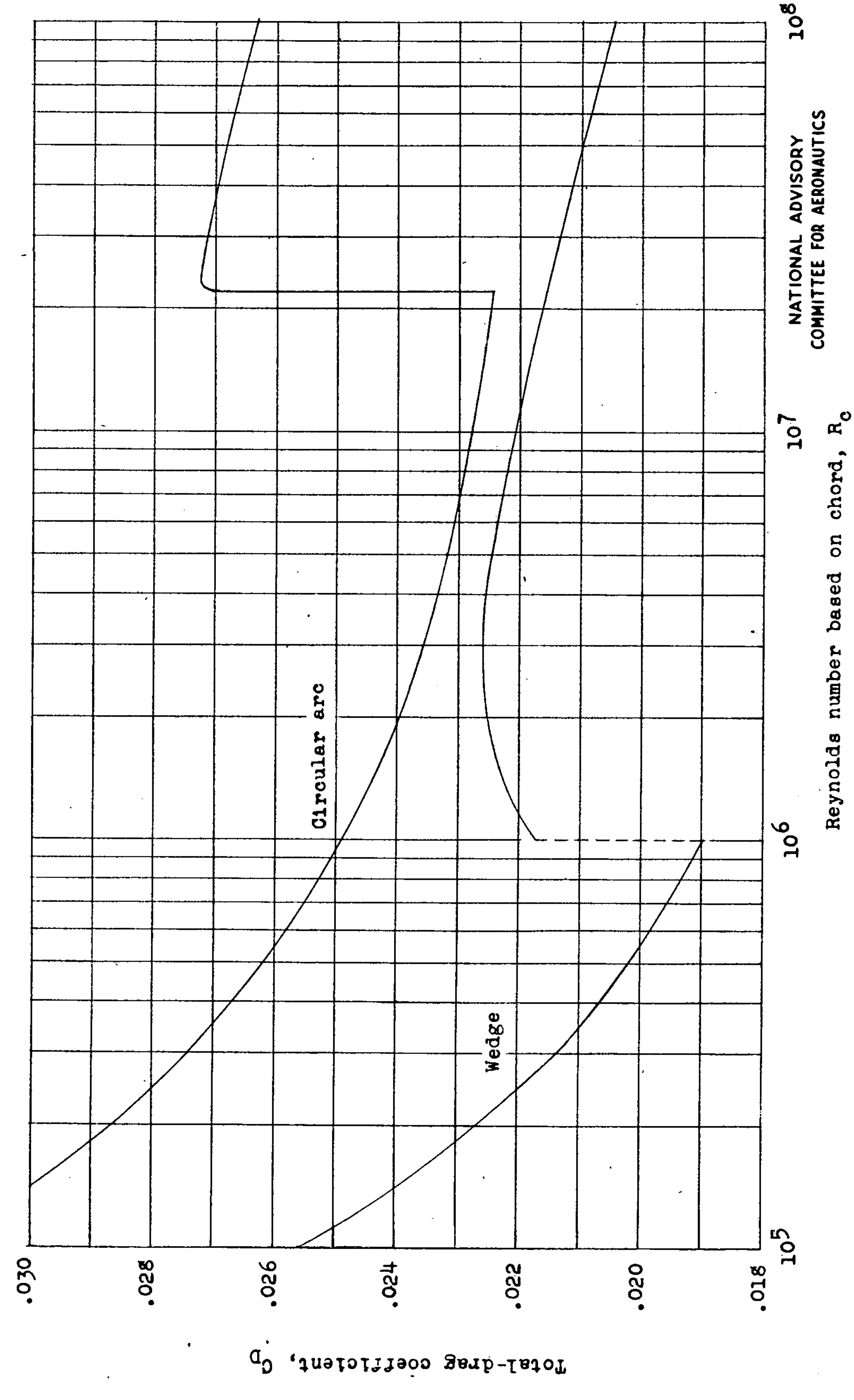
(a) Comparison of flat plate, circular-arc airfoil, and wedge airfoil. $M_o = 1.35$.

Figure 3.- Effect of Reynolds number on viscous drag. $N = 5$.



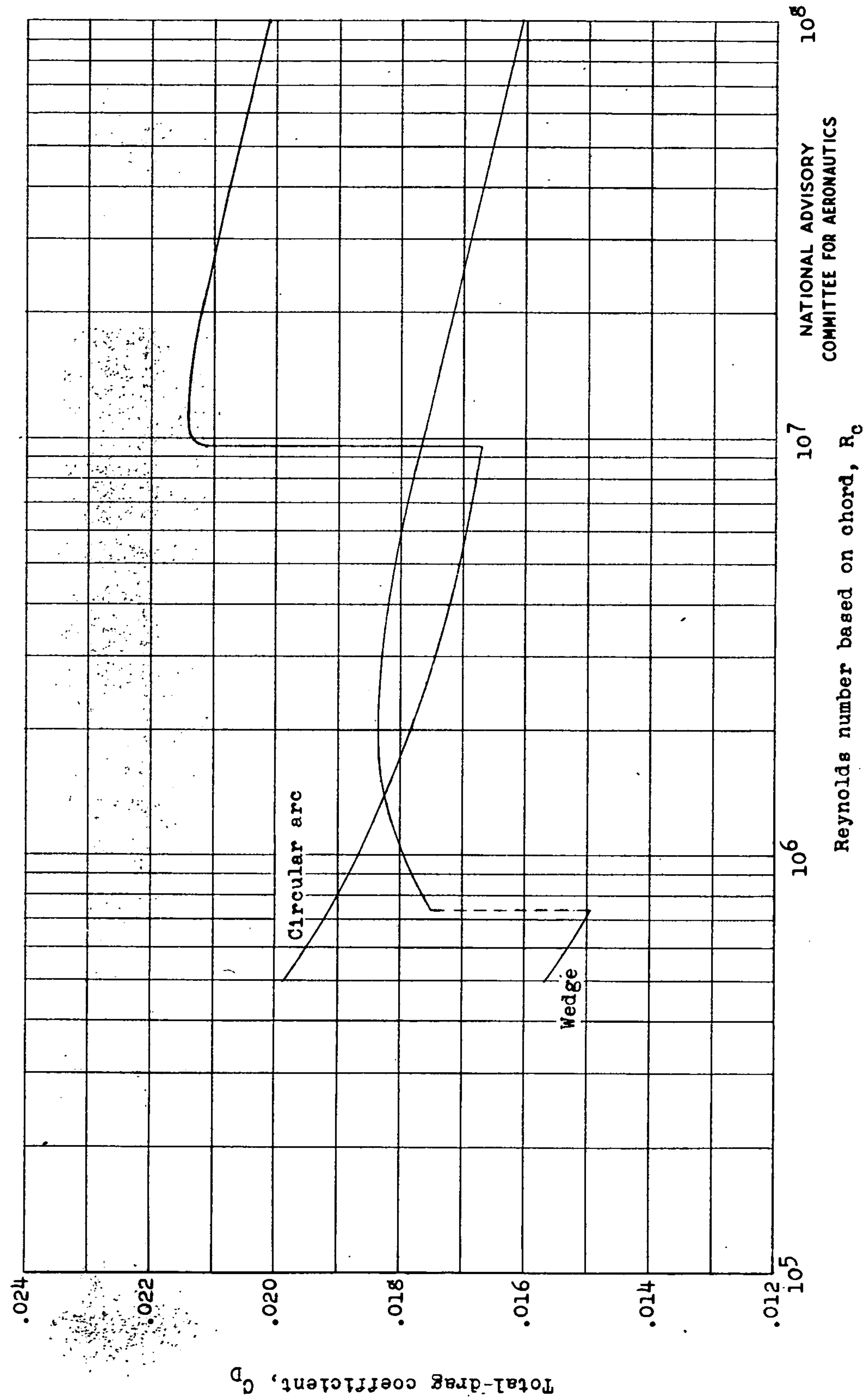

(b) Comparison of circular-arc airfoil and wedge airfoil. $M_o = 1.6$.

Figure 3.- Concluded.




(a) $M_o = 1.35$.

**Figure 4.- Effect of Reynolds number on total drag for circular-arc airfoil and wedge airfoil. N = 5.**




(b) $M_o = 1.6$.

Figure 4.- Concluded.