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RESEARCH MEMORANDUM

APPLICATION OF ONE PART OF VON KÁRMÁN'S TWO-DIMENSIONAL TRANSONIC SIMILARITY LAW TO DRAG DATA OF NACA 65-SERIES WINGS

By

Kenneth B. Amer

Langley Aeronautical Laboratory
Langley Field, Va.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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APPLICATION OF ONE PART OF VON KÁRMÁN'S TWO-DIMENSIONAL TRANSONIC SIMILARITY LAW TO DRAG DATA OF NACA 65-SERIES WINGS

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SUMMARY

In order to determine the usefulness of one part of Von Kármán's two-dimensional transonic similarity law as a means of correlating and extrapolating transonic wing data, the law is applied to drag data of three different thickness wings having NACA 65-series sections and aspect ratios of 7.6 in the Mach number range from 0.85 to 1.15. This portion of the law states that, for two-dimensional airfoils of the same family in the transonic range at zero lift, the parameter $\dfrac{{M_o}^2 C_{D_p}}{(t/c)^{5/3}}$ is only a function of the parameter $\dfrac{M_o - 1}{(t/c)^{2/3}}$. Von Kármán derived this law from the potential-flow equation assuming thin, two-dimensional bodies with local velocities not much different from sonic. The correlation of the data is satisfactory, being somewhat better in the subsonic range than in the supersonic range. The presence of a stagnation point, the probable presence of separation at high subsonic speeds, and a finite aspect ratio of 7.6, all of which were not considered in the derivation of the law, did not appreciably affect the degree of correlation. The presence of a boundary layer, which also was not considered in the derivation, was adequately accounted for by assuming a reasonable value of skin-friction drag coefficient.

INTRODUCTION

A reliable means of correlating and extrapolating transonic wing data would be extremely useful. Dr. Theodore von Kármán has developed theoretically from the potential-flow equation a two-dimensional transonic similarity law for thin bodies with local velocities only slightly different from sonic. One part of this law states that for all thin two-dimensional airfoils of the same family in the transonic range at zero lift, the parameter $\frac{C_{D_p}M_o^2}{(t/c)^{5/3}}$ is only a function of the parameter $\frac{M_o - 1}{(t/c)^{2/3}}$. Derivations of the law are presented in references 1 and 2.

So far as is known, the only data available to determine the usefulness of this part of the law were obtained by the freely-falling-body method and presented in references 3 to 5. Drag data in the Mach number range from 0.85 to 1.15 are presented in these references for wings having aspect ratios of 7.6 and 5.1 and three NACA 65-series airfoil sections, the 65-006, the 65-009, and the 65_1-012. The Reynolds number of these tests, based on the chord, varied from about 1×10^6 to about 5×10^6 for each wing.

The derivation of the similarity law does not take into account all details of the flow about these particular wings. The derivation does not account for the fact that the wings are of finite span having stagnation points, boundary layers, and probably some separation at high subsonic speeds, have local velocities well above sonic at the higher transonic Mach numbers, and (although the airfoil sections are of the same series) their thickness distributions are not exactly proportional. In this paper, the law is applied to the available data to determine the degree of correlation produced by the law and to indicate which of the unaccounted for factors in the flow about the wings affected the degree of correlation appreciably.

SYMBOLS

C_D total drag coefficient based on exposed plan area

C_{D_f} friction drag coefficient

C_{D_p} pressure drag coefficient

M_o free-stream Mach number

t/c airfoil thickness ratio

RESULTS AND DISCUSSION

The pressure drag parameter $\frac{C_{D_p} M_o{}^2}{(t/c)^{5/3}}$ was calculated for the various wings from the continuous curves of C_D against M_o of references 3 to 5 at equal increments of the parameter $\frac{M_o - 1}{(t/c)^{2/3}}$. Because of the lack of any reliable friction drag data in the transonic range, a friction drag coefficient of 0.006 was assumed for all wings. The parameters for the three wings having an aspect ratio of 7.6 are plotted in figure 1. The data are plotted as individual points rather than as continuous curves for clarity.

In order to gain a quantitative idea of what the deviations shown in figure 1 mean in terms of drag coefficient and Mach number, the faired curve of figure 1 was used to recompute a curve of C_{D_p} against M_o for each wing. In figure 2 these curves are compared with curves of C_{D_p} against M_o computed from the reference data. The experimental uncertainty in Mach number is about ± 0.01. At the highest Mach numbers tested, the experimental uncertainty in C_D is about ± 0.0025 for the 12-percent-thick wing and about ± 0.0005 for the 6-percent- and 9-percent-thick wings. The uncertainties in drag coefficient increase somewhat as M_o is reduced. The uncertainty in the assumed value of C_{D_f} is probably within about ± 0.001. Thus, the two curves for the 6-percent- and 9-percent-thick wings differ in the subsonic range by less than the maximum possible uncertainties in the experimental data and skin-friction drag coefficient. The difference between the two curves for the 12-percent-thick wing exceeds this uncertainty slightly at about $M_o = 0.98$ because of the hump in the experimental curve. The difference between the two curves for each wing in the supersonic range is less than twice the sum of the experimental uncertainty and the uncertainty in the assumed value of C_{D_f}.

Apparently, in the supersonic range, the similarity law does not entirely correlate the effect of variations in thickness ratio, and there is a consistent variation in the drag parameter with thickness ratio. In figure 3 this trend is compared with the calculated variations in drag parameter for circular-arc airfoils of the same thickness ratios at higher Mach numbers where the supersonic type of flow pattern is wholly established. The circular-arc-airfoil drag coefficients were calculated from the formula (formula 9.23, reference 6)

$$C_{D_p} = \frac{16}{3\sqrt{M_o{}^2 - 1}} \left(\frac{t}{c}\right)^2$$

For the 12-percent-thick airfoil at a low supersonic Mach number, this formula checked within a few percent a more accurate calculation based on exact shock-expansion relations. The plots for these circular-arc airfoils start at approximately the Mach numbers at which the shock waves become attached to the leading edges. These Mach numbers are about 1.3 for the 6-percent-thick airfoil, 1.4 for the 9-percent-thick airfoil, and 1.6 for the 12-percent-thick airfoil.

Figure 3 shows that there is a spread of about 25 percent in the supersonic drag parameters for the circular-arc airfoils, which is due to the fact that the similarity law does not fit the actual flow conditions since the local velocities are not near-sonic. This spread is in the same direction as the spread of the experimental drag parameters on the supersonic side of the transonic region. Therefore, the trend of the drag parameters with the thickness shown in the experimental data indicates that, because the local velocities are no longer near-sonic, the similarity law correlates less well as the Mach number increases above 1.0, and there is a gradual transition to the theoretical variation of the drag parameter with thickness ratio in the wholly supersonic regime of flow. It appears that an empirical correction for this trend could be developed.

The satisfactory correlation obtained by the use of the similarity law indicates that the factors in the flow field not accounted for by the theory (other than the presence of local velocities well above sonic at the higher Mach numbers) did not affect appreciably the degree of correlation; that is, the presence of a stagnation point, the probable presence of separation at high subsonic speeds, the finite aspect ratio of 7.6, and the slight nonproportionality of the thickness distributions did not appreciably affect the degree of correlation. The presence of a boundary layer was adequately accounted for by assuming a reasonable value of skin-friction coefficient.

Limited data for wings having aspect ratios of 5.1 are presented in figure 4. The data for the 9-percent- and 12-percent-thick airfoils show a greater spread in the supersonic range than the corresponding data of aspect ratio 7.6. These data indicate the desirability of further investigation of the degree of correlation of low-aspect-ratio data obtainable with Von Kármán's similarity law.

CONCLUDING REMARKS

Application of one part of Von Kármán's two-dimensional transonic similarity law to drag data of wings having NACA 65-006, 65-009, and 65_1-012 profiles and aspect ratios of 7.6 in the Mach number range from 0.85 to 1.15 shows satisfactory correlation which is somewhat better in the subsonic range than in the supersonic range. The presence of a stagnation point, the probable presence of separation at high subsonic speed, and a finite aspect ratio of 7.6, all of which were not considered in the derivation of the law, did not appreciably affect the degree of correlation. The presence of a boundary layer, which also was not considered in the derivation, was adequately accounted for by assuming a reasonable value of skin-friction drag coefficient.

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

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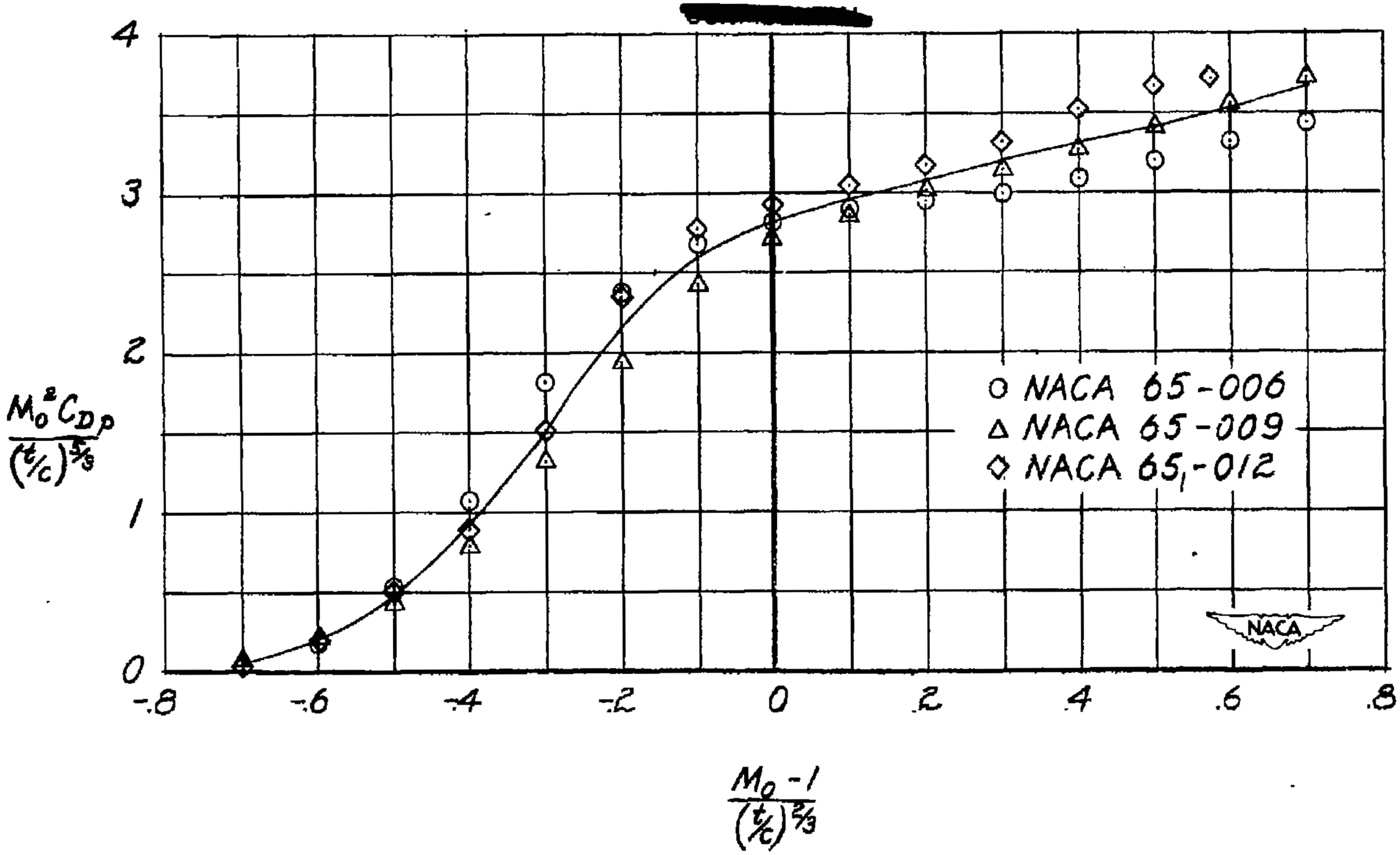

Figure 1.- Transonic drag data (with C_{D_f} assumed = 0.006) for three wings having an NACA 65-series section and an aspect ratio of 7.6 plotted according to Von Kármán's two-dimensional transonic similarity law.

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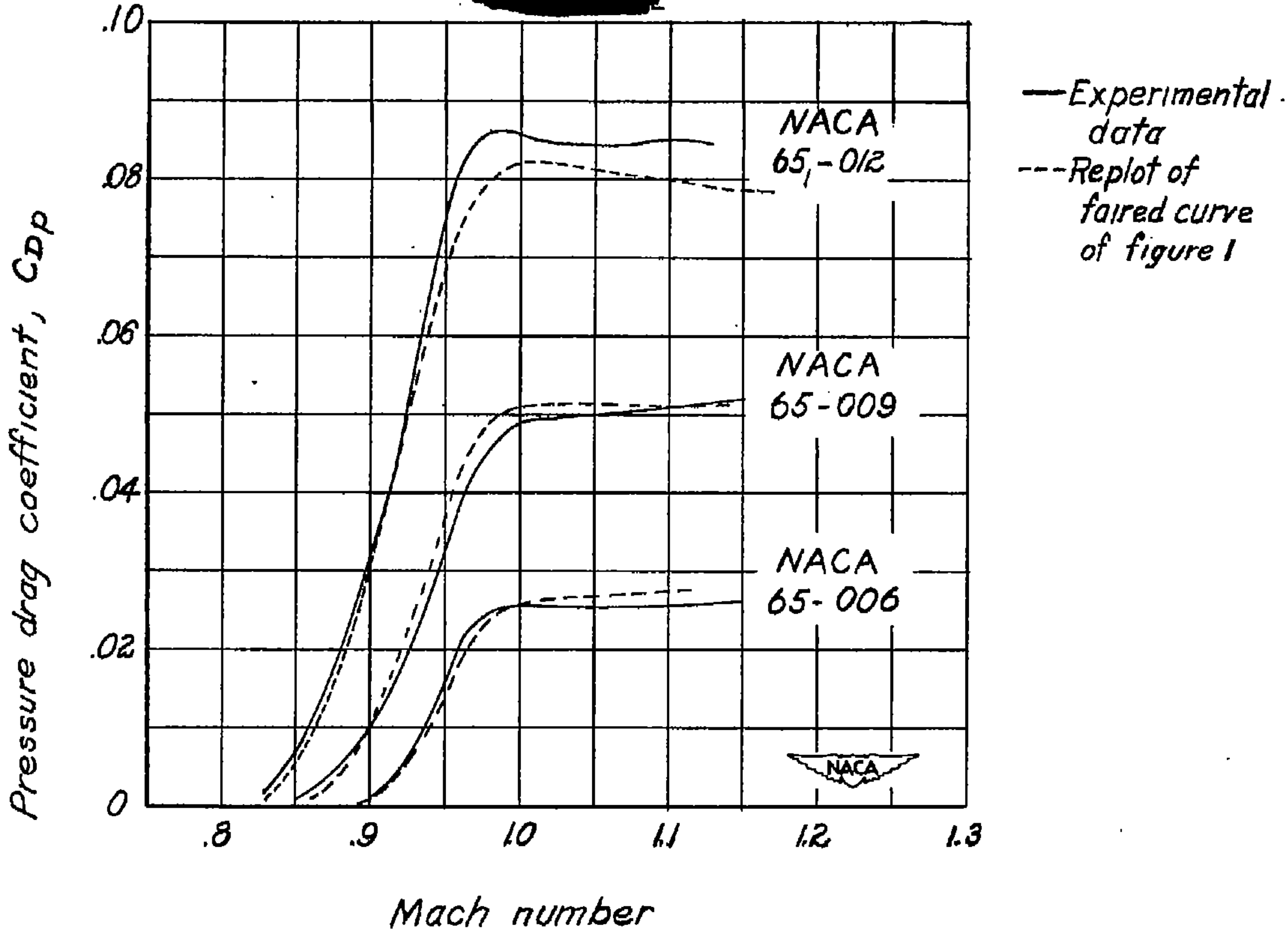

Figure 2.- Comparison between measured drag coefficients (with C_{D_f} assumed = 0.006) and pressure drag coefficients obtained from faired curve of figure 1. Aspect ratio of wings is 7.6.

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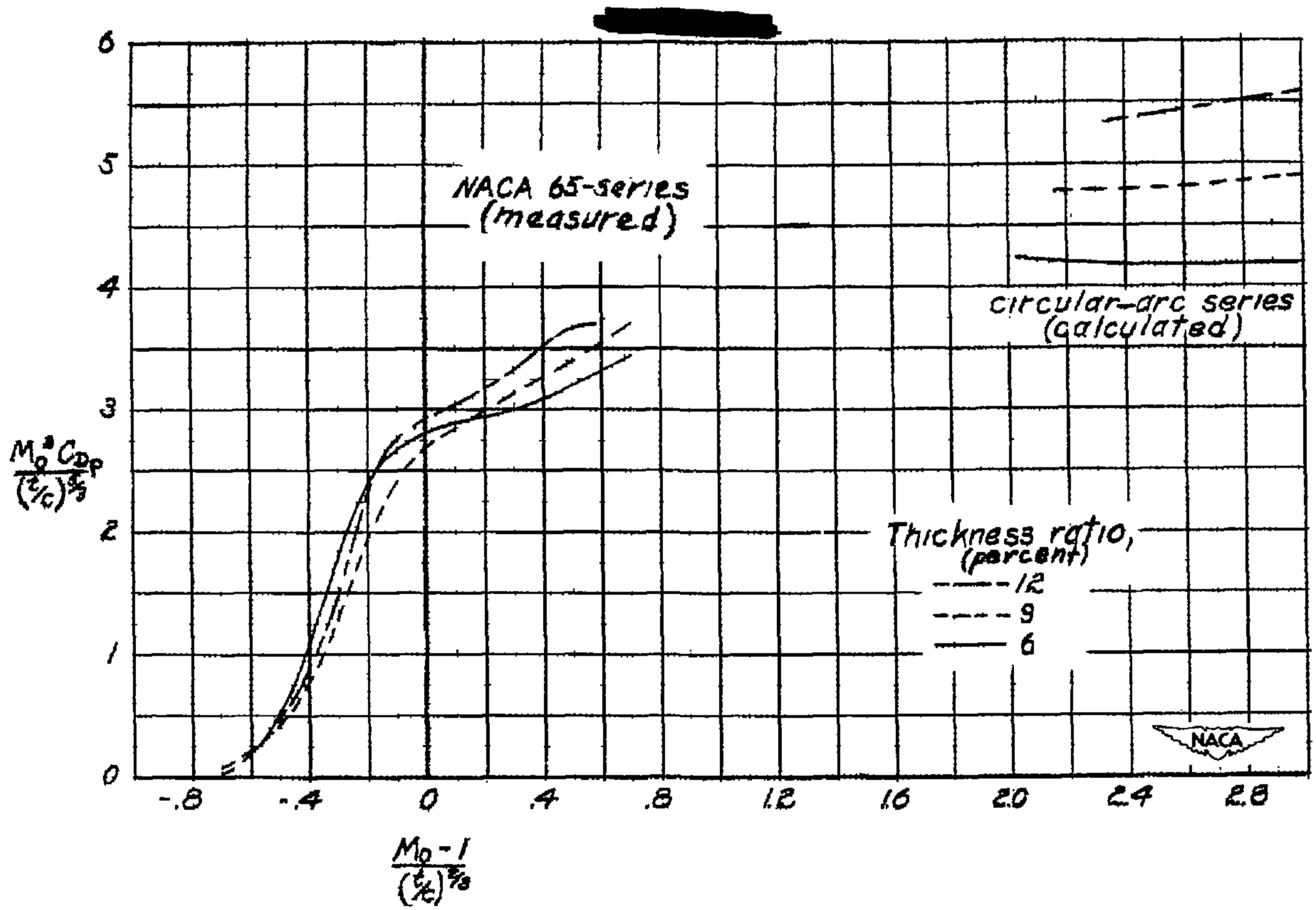

Figure 3.- Comparison between the measured effect of thickness ratio on wings having NACA 65-series sections and aspect ratios of 7.6 in the transonic range and its calculated effect on circular-arc airfoil sections in the supersonic range. [redacted]

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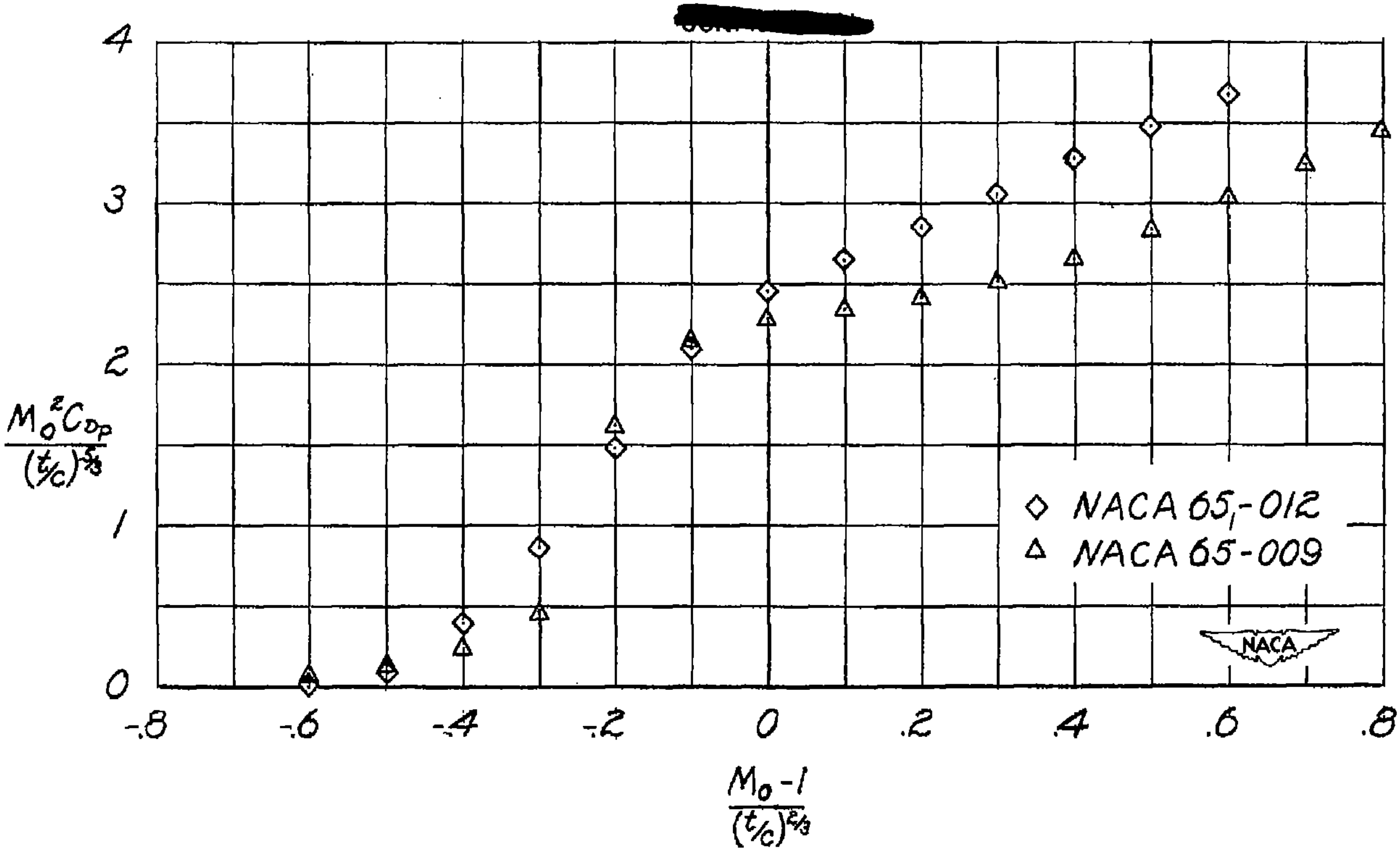

Figure 4.- Transonic drag data (with C_{D_f} assumed = 0.006) for two wings having an NACA 65-series section and an aspect ratio of 5.1 plotted according to Von Kármán's two-dimensional transonic similarity law. [redacted]