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A CRITICAL ASSESSMENT OF WIND TUNNEL RESULTS FOR THE NACA 0012 AIRFOIL*

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ABSTRACT

A large body of experimental results, which were obtained in more than 40 wind tunnels on a single, well-known two-dimensional configuration, has been critically examined and correlated. An assessment of some of the possible sources of error has been made for each facility, and data which are suspect have been identified. It was found that no single experiment provided a complete set of reliable data, although one investigation stands out as superior in many respects. However, from the aggregate of data the representative properties of the NACA 0012 airfoil can be identified with reasonable confidence over wide ranges of Mach number, Reynolds number, and angles of attack. This synthesized information can now be used to assess and validate existing or future wind tunnel results and to evaluate advanced Computational Fluid Dynamics codes.

I. INTRODUCTION

Reliable determination and assessment of the accuracy of aerodynamic data generated in wind tunnels remains one of the most vexing problems in aeronautics. Aerodynamic results are seldom duplicated in different facilities to the level of accuracy that is required either for risk-free engineering development or for the true verification of theoretical and numerical methods. This shortcoming is particularly acute with regard to today's rapid proliferation of new Computational Fluid Dynamic (CFD) codes that lack adequate validation [1].

On the other hand, the NACA 0012 profile is one of the oldest and certainly the most tested of all airfoils; and it has been studied in dozens of separate wind tunnels over a period of more than 50 years. Although no single high-quality experiment spans the complete subsonic and transonic range of flow conditions, the combined results of this extensive testing should allow some conclusions to be drawn about wind-tunnel data accuracy and reliability, at least for two-dimensional (2-D) testing. This paper attempts to extract as much useful, quantitative information as possible from critical examinations and correlations of existing data from this single, well-known configuration, obtained in over 40 wind tunnels and over wide ranges of Mach number, Reynolds number, and angles of attack.

A preliminary comparison by the author [2] in 1982 of results from about a dozen widely-quoted investigations for the NACA 0012 airfoil revealed significant and unacceptable differences between wind tunnels, and subsequent examinations of more data sets merely compounded the confusion, as indicated in Figs. 1 and 2. Therefore, a major part of the present investigation was the development of a filtering process for screening the available data and classifying the experimental sources into broad categories of estimated reliability. This process is described in the next section. Detailed comparisons, correlations, and uncertainty estimates are discussed in subsequent sections, where the the following results are considered:

1. Lift-curve slope versus Mach and Reynolds number
2. Minimum drag versus Mach and Reynolds number
3. Maximum lift-to-drag ratio versus Mach and Reynolds number
4. Maximum lift versus Mach and Reynolds number
5. Shock-wave position versus Reynolds number at $M = 0.8$

As this list indicates, the present study deals mostly with the integral quantities, lift and drag. Despite the large number of references available on this most popular of all airfoils, it was found that there is insufficient overlap in the experiments to make many meaningful, direct comparisons of more detailed quantities, such as pressure distributions, in the transonic regime. It is acknowledged that pitching moment is also a sensitive integral parameter that displays interesting transonic behavior, but C_m is not considered in this paper.

II. THE FILTERING AND ANALYIS PROCESS

The main objective of this section is to combine the critical, relevant information that is available on airfoil testing and on airfoil aerodynamic behavior into a systematic screening, or "filtering," process that can be used to assess the quality of individual experimental sources of data. This process will then be used to classify each data set and to weigh the accuracy of those data against the quantitative or qualitative information that they can provide about the aerodynamic characteristics of the NACA 0012 airfoil.

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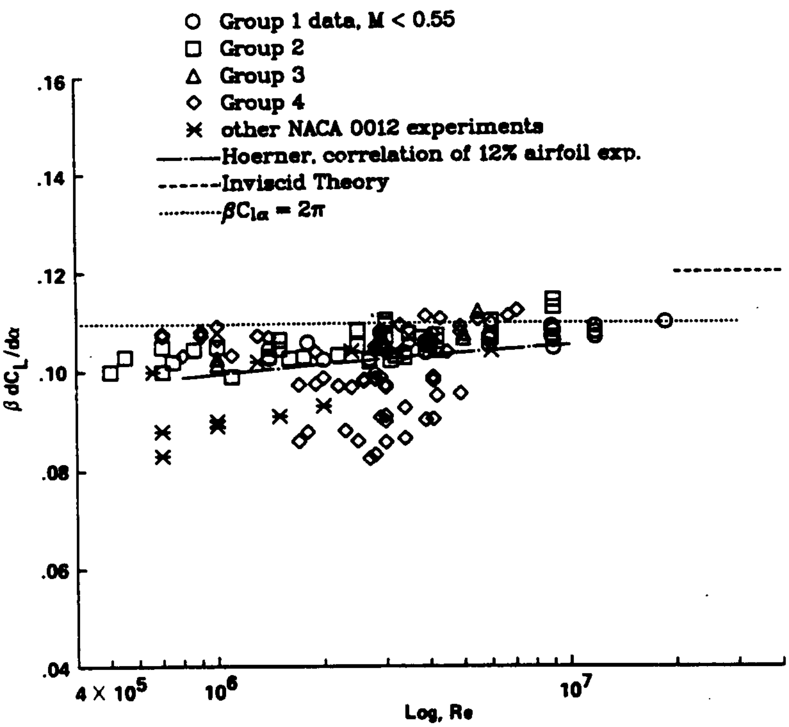

Fig. 1. Lift-curve slope at zero lift vs. Reynolds number; all data, M < 0.55. Legend explained in Tables 1-4, $\beta = \sqrt{1 - M^2}$

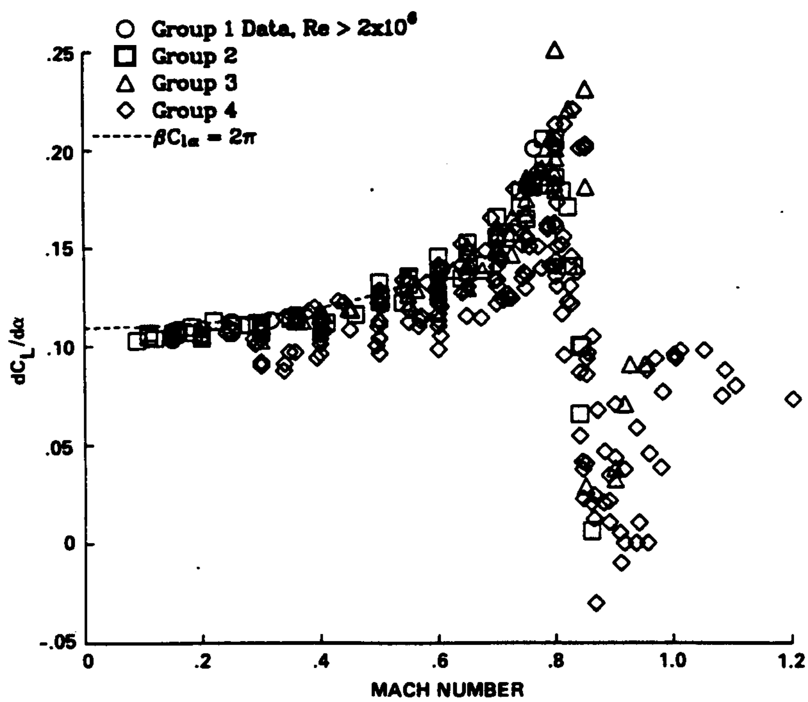

Fig. 2. Lift-curve slope vs. Mach number; all data. Legend explained in Tables 1-4.

A. Development of the Process

The critical information used in the development of the process is derived from four broad categories, as follows:

1. A very large collection of wind-tunnel data for the NACA 0012 which varies widely for many possible reasons.

2. A modest collection of "facts," i.e.,

   a. well-established theories and similarity laws
   b. generally-accepted empirical laws
   c. recent advances in identifying, analyzing, and correcting for wind-tunnel wall effects.

3. A fuzzy collection of "folklore" about airfoil behavior, test techniques, and wind-tunnel characteristics.

4. Recent CFD results for a few standard airfoil cases in both simulated free-air conditions and combined airfoil/wind-tunnel installations.

This aggregate of information firmly establishes some important sources of wind-tunnel errors and certain properties of airfoils such as the NACA 0012. This knowledge can be summarized as follows: first, all four wind-tunnel walls generally interfere with the flow around the airfoil, and this phenomenon is generally more acute than for three-dimensional (3-D) bodies. The top and bottom walls particularly affect the effective angle of attack, the shape of the pressure distribution (and hence pitching-moment coefficient), and the shock-wave location, and to a lesser extent, lift, drag, and effective Mach number. Solid walls increase the effective α and Mach number, but these effects are considered to be easily correctable, at least in subsonic and mildly transonic flows. Slotted or porous walls lower the effective α; attemps are often made to correct for this, but it is difficult.

Second, side-wall boundary layers have been shown to lower C_ℓ, C_d, and the effective M, and to move the shock forward. Flow separation at the airfoil-wall juncture affects the shock location and reduces $C_{\ell_{max}}$. The effects can be reduced substantially by the application of suction on the side walls, and corrections can be applied if there is no separation in the corners.

Third, free-stream turbulence and boundary-layer trips increase C_d and often affect C_ℓ, C_m, and shock location. Many airfoils, including the NACA 0012, may be particularly sensitive to Reynolds number variations if no trip is used; however, extreme care must be exercised in tripping the boundary layer to avoid causing excessive drag increments and erroneous changes in C_ℓ and shock position. The effects of both trips and turbulence are difficult to quantify.

Concerning airfoil behavior, two important "facts" have been established about the behavior of lift and drag in subsonic flow at small angles of attack. At high Reynolds numbers, both C_d at zero lift and the quantity $\sqrt{1 - M^2}C_{\ell_\alpha}$ are independent of M and are only weakly dependent upon Re. Unfortunately, most other aspects of airfoil characteristics are not as firmly established, and even these two quantities are not well defined in transonic flow. However, measurements of general trends and qualitative behavior are generally accepted, even if the absolute values of C_ℓ, C_d, and C_m, for example, are uncertain.

To improve on this situation, the following filtering or screening process is proposed. First, an attempt will be made to identify the highest-quality experiments in which the aforementioned wind-tunnel problems were carefully controlled, corrected for, or otherwise ameliorated. Second, the results of these tests will be used to establish the quantitative, "factual," behavior of the critical parameters C_{d_o} and βC_{ℓ_α}, where $\beta = \sqrt{1 - M^2}$, as functions of Re in the subsonic regime where they are essentially independent of M. This information comprises the filters that are <u>necessary</u>, although not <u>sufficient</u>, screening criteria for judging the credibility of the remaining data. Third, these filters will be used to help identify obviously erroneous aspects of all the data sets and to classify each experiment accordingly. Fourth, all the data will be critically examined <u>outside</u> the range of Mach and Reynolds numbers for which the filters were developed. Finally, a subjective extension of the fourth step will be made. The "folklore" correlations and other information referred to above, and established transonic similarity laws, will be used to combine selected NACA 0012 and other airfoil data in order to estimate the transonic properties of the NACA 0012 over a range of Mach numbers, $0.85 < M < 1.1$, for which virtually no reliable data exist.

B. Application of the Process

Table 1 lists and summarizes the experiments which clearly stand out as having been conducted with the utmost care and/or as most nearly eliminating the important sources of wind-tunnel errors. These sources are referred to throughout this paper as Group 1. It will be noted from Table 1 that, unfortunately, only one of the experiments extends slightly into the transonic regime, and that the turbulence level in that test was relatively high. Also, for the present purposes, it is unfortunate that the only data reported from that experiment were obtained with a boundary-layer trip, although some unpublished data were also obtained without a trip.

The results for βC_{ℓ_α} from Group 1 are plotted versus Re in Fig. 3. It is clear that the results shown in this figure represent a major improvement over the large scatter in Fig. 1. A good fit of the lift-curve slope data in the limited range $2 \times 10^6 < Re < 2 \times 10^7$ is given by

$$\beta C_{\ell_\alpha} = 0.1025 + 0.00485 \,\text{Log}(Re/10^6) \quad \text{per degree} \tag{1}$$

with an rms standard error of 0.00024 and a maximum error of 0.0029 for the 30 points shown.

Similarly, the results for C_{d_0} are plotted in Fig. 4. The meaning of the various groups is explained below. The drag data from Group 1 without a boundary-layer trip, i.e. the open circles, can be approximated well by

$$C_{d_0} = 0.0044 + 0.018\, Re^{-0.15} \tag{2}$$

with an rms standard error of 0.00005 and a maximum error of 0.0007 for the 36 points from Group 1. The data with a boundary layer trip show a greater sensitivity to Reynolds number. In accord with the approximate variation of fully turbulent skin friction with Reynolds number [3], a good fit to the Group 1 tripped data is given by

$$C_{d_0} = 0.0017 + 0.91/(\text{Log}\, Re)^{2.58} \tag{3}$$

where the constant 0.0017 was chosen to optimize the curve fit shown in Fig. 4.

For reference, it is estimated that the individual values of βC_{ℓ_α} and C_{d_0} can be determined or calculated from the individual Group 1 data points to an overall precision of about ±0.0005 and ±0.0002, respectively. It may be mentioned that Ref. 4 lists the desired accuracy of C_d from wind tunnels as 0.0005 for the assessment of configuration changes and 0.0001 for the validation of CFD codes.

The information in Eqns. 1-3 can now be used to assess the accuracy of the data from the remaining sources and to group the data into separate categories. After much deliberation, it was decided to define Group 2 as comprising those data which generally agree with <u>both</u> the lift and drag criteria expressed in Eqns. 1-3, to within ±0.0040 for βC_{ℓ_α} and to within ±0.0010 for C_{d_0}. These experiments are listed in Table 2. Foremost in this group is the experiment of C. D. Harris [5]. Although this experiment was carefully conducted and offered the advantage of a large aspect ratio, lift-interference corrections on the order of 15% are required for the angles of attack. These were a major concern initially, but in the subsequent discussions and figures it will become evident that these results are comparable in accuracy to those of Group 1.

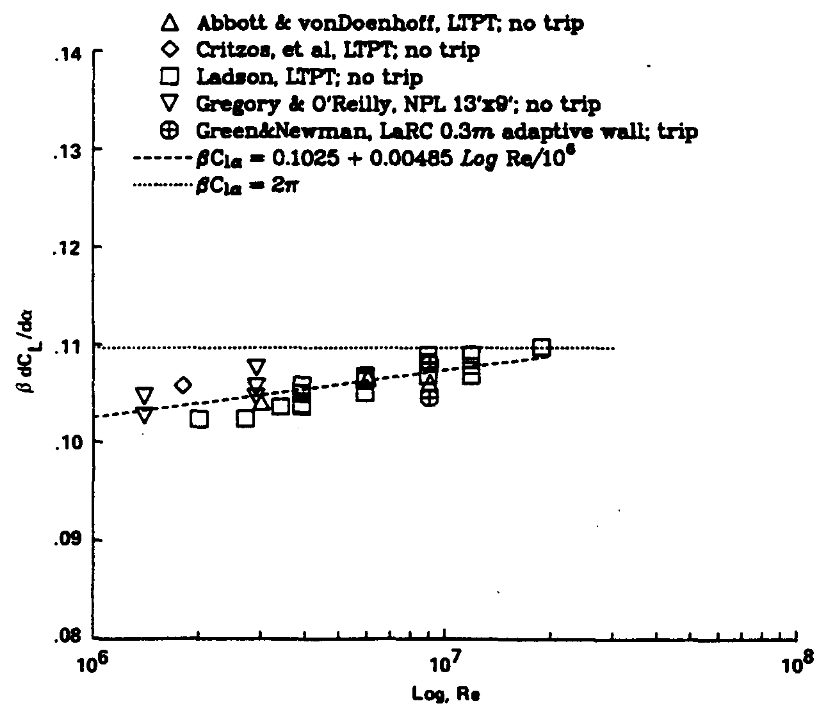

Fig. 3. Lift-curve slope at zero lift vs. Reynolds number; Group 1 data, M < 0.55. Expanded vertical scale.

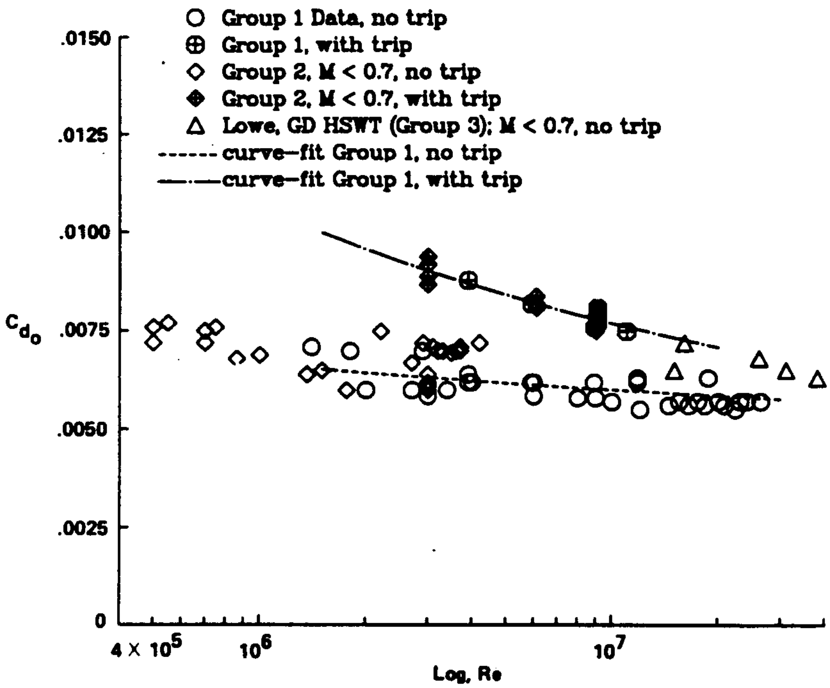

Fig. 4. Drag coefficient at zero lift vs. Reynolds number.

Several sources provide data that agree well with the Group 1 results for <u>either</u> βC_{ℓ_α} <u>or</u> C_{d_0}, but not for both. In some cases, only one of these key quantities was measured. These are classified as Group 3 and are listed in Table 3. An example of this group is the essentially interference-free experiment of Vidal et al. [6], which provides good lift data, but which used a large trip that evidently produced excess drag.

A few sources provided data that generally satisfy the basic lift and/or drag criteria outlined above, but for which other major problems have been identified. In addition, a significant number of tests fail to satisfy <u>either</u> of these two criteria, but they do cover ranges of Mach number where even qualitative information is helpful. These sources are referred to as Group 4 and are briefly summarized in Table 4. Finally, still other sources were examined that failed to satisfy the criteria, and which did not appear to offer any significant additional information relevant to the present investigation. For information purposes these are listed in Table 5, but their results are not used in this paper.

III. RESULTS AND DISCUSSION

In this section, the results from Groups 1-4 and from the other sources alluded to Section II.A are used collectively to establish the primary characteristics of the NACA 0012 airfoil over a wide range of Mach number, Reynolds number, and angle of attack.

A. <u>Lift-Curve Slope, $dC_\ell/d\alpha$</u>

Figure 5 shows the data from Groups 1-3 for βC_{ℓ_α} as a function of Reynolds number, for $M < 0.55$. Harris' results [5], at $Re = 3$ and 9×10^6, are highlighted by solid symbols, and this convention will be followed in most of the remaining figures. The scatter in the Group 2 data is slightly greater than that of the Group 1 results, but the quantitative behavior of βC_{ℓ_α} seems to be established now over the range of most wind-tunnel tests for aeronautical purposes.

The complex transonic behavior of C_{ℓ_α} is illustrated in Fig. 6, where the relevant Group 3 data have been added. This figure clearly represents a major improvement over Fig. 2. For these conditions, the good agreement between Harris' results [5] and those of Green and Newman [7] constitute further validation of the former. The largest discrepancies that remain occur with the data from Vidal et al. [6] below $M = 0.8$, which seems to be mostly a Reynolds-number effect, and Sawyer [8], who reported large values at $M = 0.8$. It is unclear whether this is due to side-wall interference, or something else. But in all cases, the peak in C_{ℓ_α} occurs at $M = 0.80 \pm 0.01$.

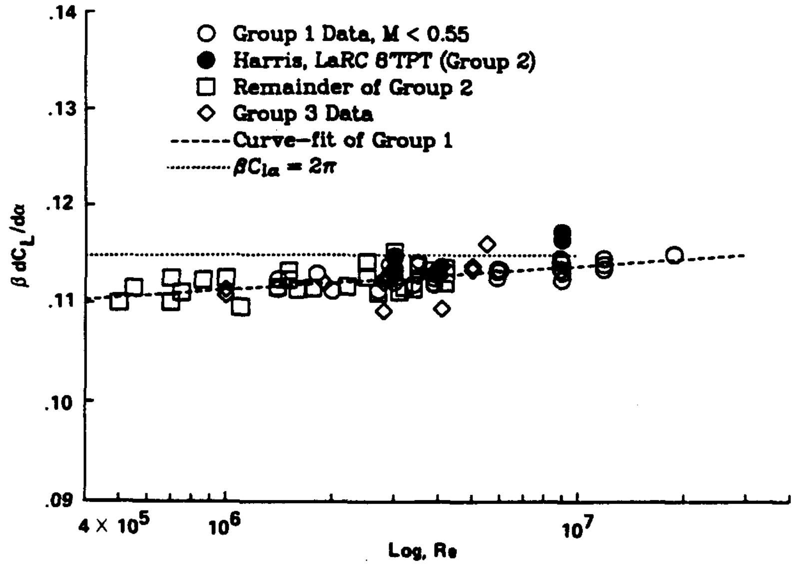

Fig. 5. Lift-curve slope vs. Reynolds number. Same scales as Fig. 1.

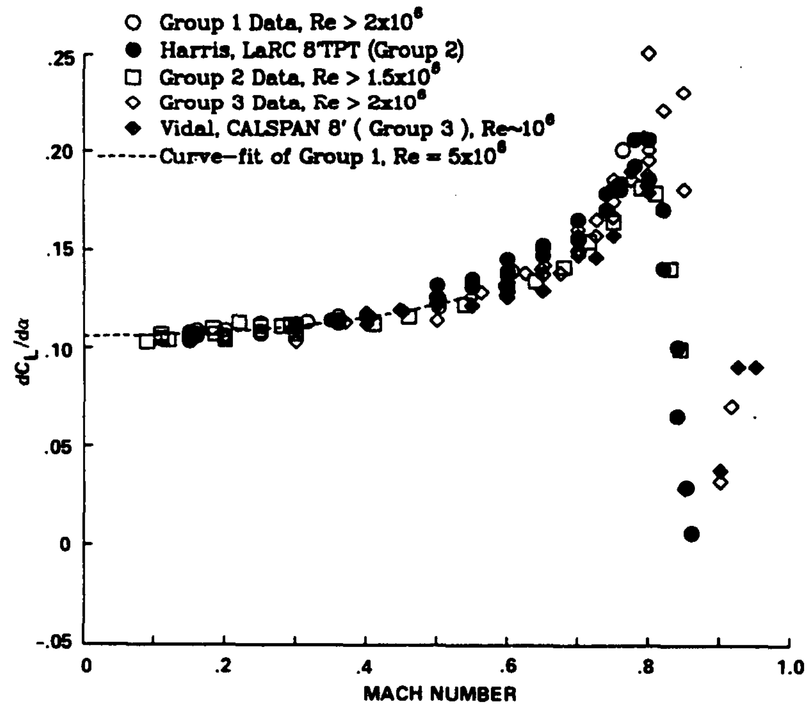

Fig. 6. Lift-curve slope vs. Mach number.

The data in Fig. 6 indicate rapid variations with Mach number in the narrow range $0.8 < M < 0.9$. Unfortunately, the Group 2 and 3 data are very sparse in this region, and are nonexistant above $M = 0.95$. Therefore, an attempt was made to extract selected additional information from the Group 4 data and from other sources, as discussed above. Three points are relevant here. First, in the transonic portion of Fig. 2, the results of Scheitel & Wagner [9] can be argued to be the most reliable of the Group 4 measurements, because side-wall suction was used and because their results are more nearly consistent with the Group 2 and 3 data where there is some overlap. Second, all of the supersonic data points of Group 4 are in good agreement with one another and with the similarity correlation given below which encompasses other symmetrical airfoils [10,11],

$$C_{\ell_\alpha} = 0.055[(\gamma + 1)M^2 t/c]^{-1/3} \quad \pm 10\% \tag{4}$$

It must be noted that this simple relation is only valid in the low supersonic range, $0.1 < \tilde{M} < 1$, where $\tilde{M} = (M^2 - 1)[(\gamma + 1)M^2 t/c]^{-1/3}$, and although it is based on transonic similarity, the thickness correlation breaks down for $\tilde{M} < 1$ [10].

A third important aspect of Figs. 2 and 6 is the behavior around $M \sim 0.9$. There is a wide variation in the minimum value of C_{ℓ_α} and in the Mach number at which this occurs; and Refs. 9 and 12 of Group 4, and Ref. 13 of Group 5 reported negative values of C_{ℓ_α}. This phenomenon was investigated briefly in Ref. 14, wherein Navier-Stokes calculations at $M = 0.88$ and $\alpha = 0.5°$ produced a marginally-stable solution with $C_\ell = 0$. These calculations were repeated recently with a time-accurate code, and this time they produced an unsteady solution with periodic oscillations with an amplitude of $\Delta C_\ell = 0.1$ around a mean value of approximately zero. This behavior appears to be qualitatively the same as the transonic self-induced oscillations reported on a biconvex airfoil by Levy [15] and in several subsequent investigations. On the other hand, only "steady" results have been reported in the NACA 0012 experiments, and this unsteady behavior may have been overlooked. Furthermore, it is not known what effect the wind-tunnel walls may have. Considering these factors, it is the author's subjective opinion that the correct behavior for the <u>mean</u> value of C_{ℓ_α} is a minimum value somewhere between 0 and -0.05, occurring at $M = 0.88 \pm 0.02$. This area needs further investigation.

Figure 7 shows the collective, "filtered" information described above in the Mach number range from 0.6 to 1.2, including the author's judgement of the upper and lower bounds of the correct transonic lift characteristics of the NACA 0012 airfoil at moderate Reynolds numbers and small angles of attack. In summary, the most important points are the following:

1. In the subsonic range $M < 0.5$, C_{ℓ_α} is given by Eqn. 1 to within ±2%.
2. The maximum value of C_{ℓ_α} is 0.21 ±5% and it occurs at $M = 0.80 \pm 0.01$.
3. The minimum value of C_{ℓ_α} is -0.025 ±0.025 and it occurs at $M = 0.88 \pm 0.02$.
4. A secondary maximum in C_{ℓ_α} occurs near $M = 1$, with a value of 0.09 ±10%.
5. In the low supersonic range $1.05 < M < 1.2$, C_{ℓ_α} is given by Eqn. 4 to within ±10%.

These estimates represent the maximum precision that can be extracted from the existing information, and they represent what is probably the best absolute accuracy to which interference-free lift can be measured on airfoils in wind tunnels today for an arbitrary angle of attack.

B. <u>Minimum Drag, C_{d_o}</u>

The baseline information for this fundamental quantity in subsonic flow was discussed earlier in connection with Fig. 4. Although the data from Groups 1 and 2 are self-consistent, the scatter in the results from Groups 3 and 4 (not shown), owing to free-stream turbulence, surface roughness and/or boundary layer trips, wall interference, and measurement errors, would almost totally mask the variation of drag with Reynolds number. Numerical results compiled by Holst [16] in his recent validation exercise for transonic viscous airfoil analyses, suggest that fully-turbulent C_{d_o} lies between the values given by Eqns. 2 and 3, but this has not been validated adequately.

Another interesting situation is the transonic drag rise, Fig. 8, for which only a limited number of high-quality sources are available. Here the scatter is excessive, but below $M = 0.7$, each individual data set seems to be essentially independent of Mach number. This suggests subtracting out an average of the subsonic values for any given data set, as follows:

$$\Delta C_{d_o} = C_{d_o}(M) - \bar{C}_{d_o}(M) \tag{5}$$

where $\bar{C}_{d_o}$ is the average of the measurements for $M < 0.7$.

The results of applying this procedure are shown in Fig. 9, which is an obvious improvement over Fig. 8. Remarkably, even the Group 3 data are in good agreement for ΔC_{d_o}. The drag-divergent Mach number can now be estimated at $M_{dd} = 0.77 \pm 0.01$, with a small amount of drag creep for $M > 0.72$.

The behavior at higher transonic Mach numbers is much more difficult to establish. All of the data from Groups 1-4 are plotted in Fig. 10, along with estimates based on transonic similarity correlations of data from many other symmetrical airfoils [10,11,14,17-20]. These latter sources indicate that airfoil behavior in the low superonic region is given by

$$C_{d_o} = \bar{C}_{d_o} + a(t/c)^{5/3}[(\gamma + 1)M^2]^{-1/3} \tag{6}$$

where a is a "constant" that varies from source to source, but which is bounded by about 4.0 and 5.6. The dashed line in Fig. 10 is for $a = 4.8$.

Data from Groups 1-4 do not extend beyond $M = 0.95$. Between $M = 0.8$ and 0.9, where C_{d_o} is rising rapidly, there is a large amount of scatter, and the uncertainty in the measurements is virtually impossible to assess. The solid lines represent the author's subjective judgement of the probable upper and

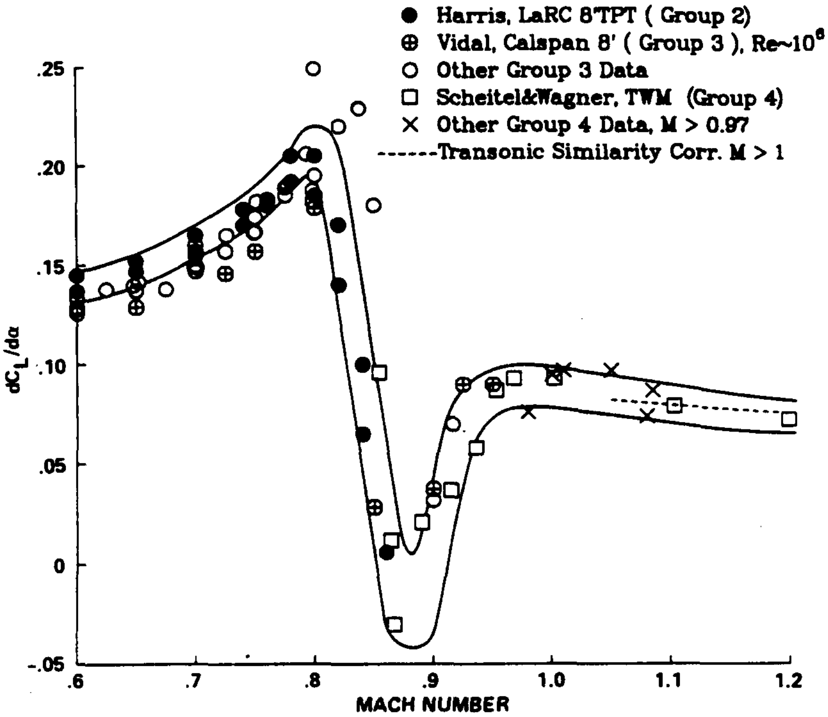

Fig. 7. Lift-curve slope vs. Mach number, including estimated upper and lower bounds.

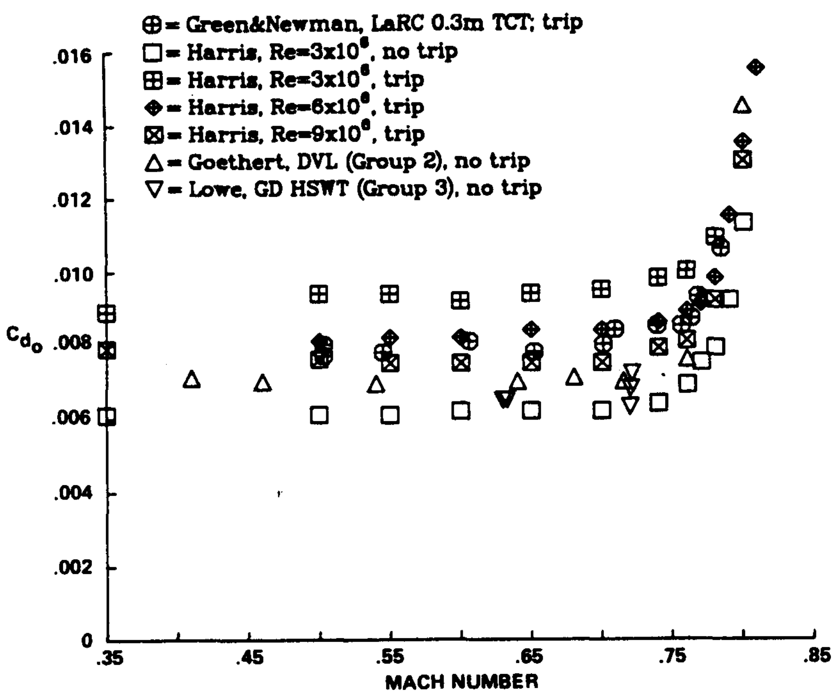

Fig. 8. Minimum drag vs. Mach number; $2 \times 10^6 < Re < 4 \times 10^7$.

lower bounds of the correct transonic drag characteristics for this airfoil. In brief, the most important points concerning mimimum drag may be summarized as follows:

1. The subsonic behavior without a boundary layer trip is given by Eqn. 2 to within about ±0.0003 in the range $10^6 < Re < 3 \times 10^7$.

2. The subsonic behavior with a fully-developed turbulent boundary layer over the entire airfoil is given approximately by Eqn. 3. The uncertainty is difficult to estimate from the available data, but the value ±0.0005 is proposed.

3. The drag-divergence Mach number is between 0.76 and 0.78. Above M_{dd}, C_{d_0} rises rapidly to a maximum value of 0.11 ±10%, which occurs between $M = 0.92$ and 0.98.

4. In the low supersonic range $1.05 < M < 1.2$, C_{d_0} is given by Eqn. 6 to within ±10%. In this regime, both C_{d_0} and C_{ℓ_α} vary as $M^{-2/3}$.

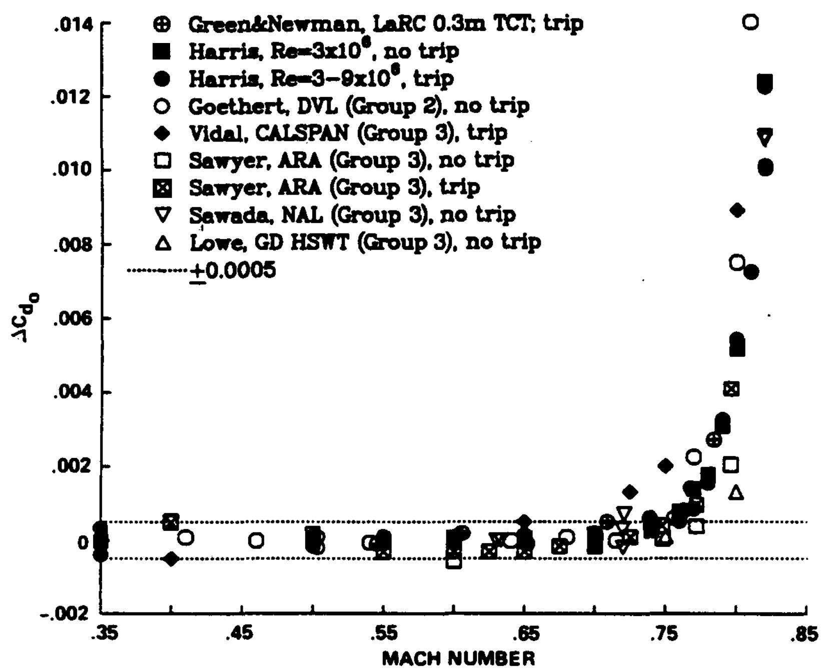

Fig. 9. Incremental drag vs. Mach number; Groups 1-3.

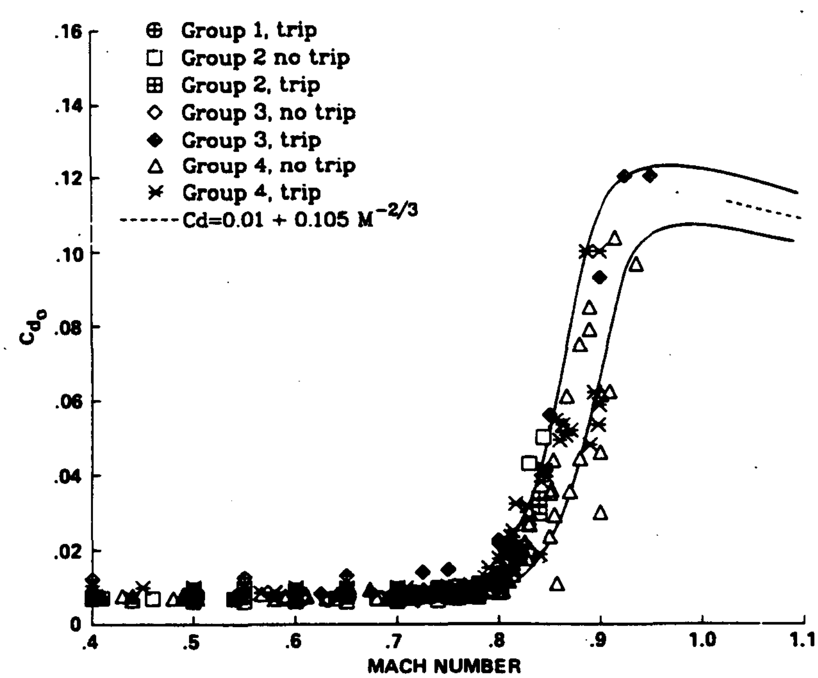

Fig. 10. Minimum drag vs. Mach number; all data, including estimated upper and lower bounds.

C. Maximum L/D Ratio

This quantity has important practical consequences for both fixed-wing aircraft and rotorcraft, and it also represents a rather different and sensitive check on wind-tunnel accuracy and flow quality. On the one hand, it compounds the uncertainty in both lift and drag, but does so under test conditions that are less severe than $C_{\ell_{max}}$, for example. On the other hand, errors in angle of attack or uncertainties in the α-corrections are not at issue here. Therefore, some experiments in which C_{ℓ_α} is suspect may still provide useful information on $(L/D)_{max}$.

Reynolds-number effects on $(L/D)_{max}$ can be isolated for examination if the Mach number is less than about 0.5. This is illustrated in Fig. 11, which shows an increase in $(L/D)_{max}$ by about a factor of two between $Re = 10^6$ and 10^7. In Fig. 11, the Group 1 results generally show the highest values of $(L/D)_{max}$, consistent with the overall high quality of these investigations. Several of the Group 2 experiments extend the Reynolds number range to lower values than those of Group 1. In addition, the Group 3 results and three sets of data from Group 4 are in fair agreement. Unfortunately, Harris [5] did not provide lift and drag polars for untripped conditions, but it is interesting to note that his results with a boundary-layer trip are in fair agreement with the other data shown. This was not the case for any other tripped data.

At higher Mach numbers the variations in $(L/D)_{max}$ with Mach and Reynolds number are almost impossible to separate from one another. As a compromise between the limitations of so few data available at a given Reynolds number and the large changes in $(L/D)_{max}$ with Re, Fig. 12 shows the available results for the narrow range $4 \times 10^6 < Re < 9 \times 10^6$. The data from Groups 3 and 4 are of interest here, because they are the only available results without a trip that extend into the transonic regime. However, they are suspicious because they lie significantly below the tripped data of Harris [5]. Additional transonic data would be particularly valuable to clarify the quantitative behavior of (L/D).

D. Maximum Lift

Conventional wisdom holds that three-dimensional separated boundary-layer effects are almost impossible to control at the stall conditions, and there is some question as to whether true two-dimensional stall exists, even for extremely high aspect ratios. Parenthetically, the accurate prediction of $C_{\ell_{max}}$ for the NACA 0012 airfoil also remains one of the greatest challenges to CFD. Therefore, this quantity needs to be established experimentally.

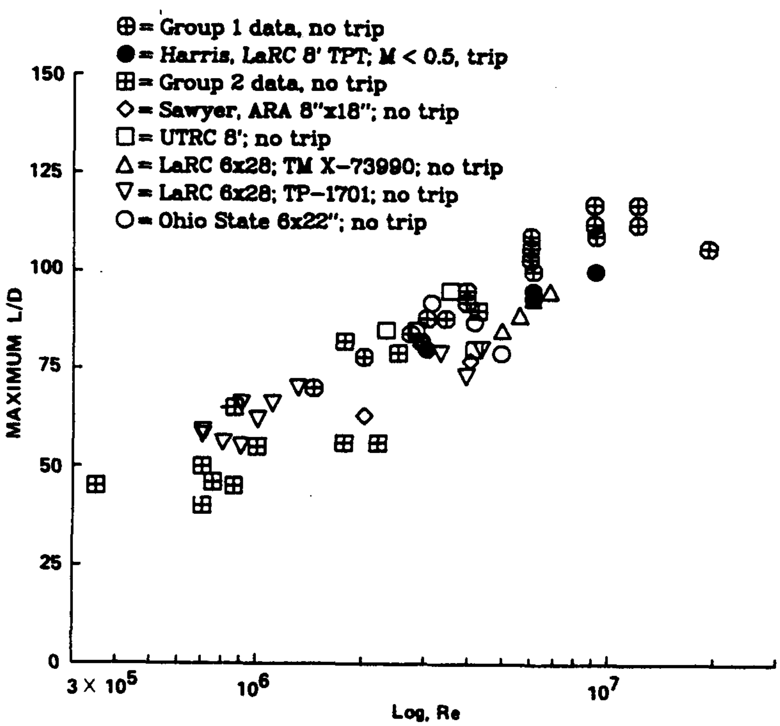

Fig. 11. Maximum lift-to-drag ratio vs. Reynolds number; M < 0.5.

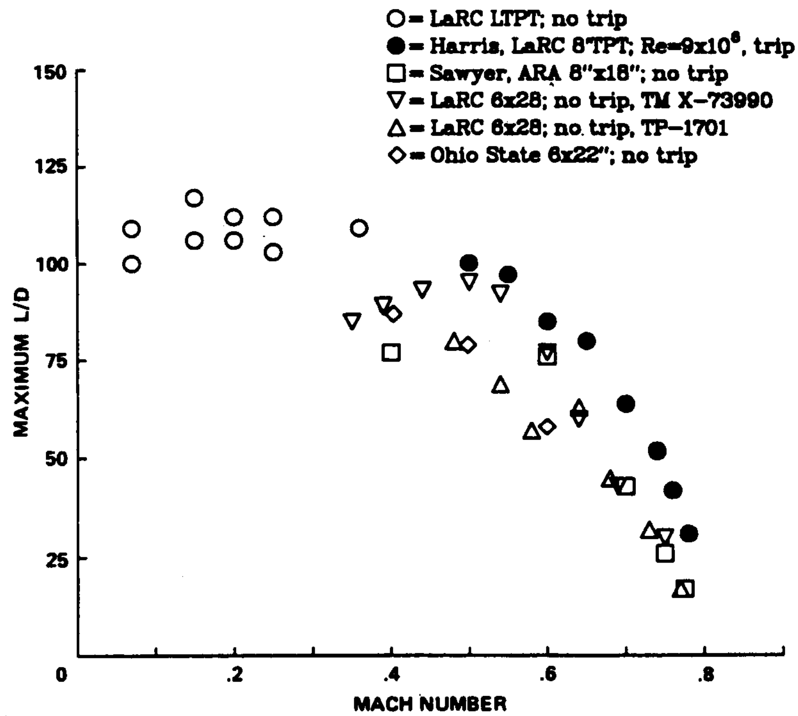

Fig. 12. Maximum lift-to-drag ratio vs. Mach number: $4 \times 10^6 < Re < 9 \times 10^6$.

Figure 13 shows the variation of $C_{\ell_{max}}$ vs Re for the available data from Groups 1 and 2, at Mach numbers less than 0.25. A monotonic increase in maximum lift with Reynolds number is evident. These particular results are surprisingly consistent, whereas the values from Groups 3 and 4 (not shown) were found to be significantly lower, in general. Also, it should be mentioned that the data shown at Re < 10^6 are somewhat higher than the values often quoted (e.g., Ref. 3), based on older sources.

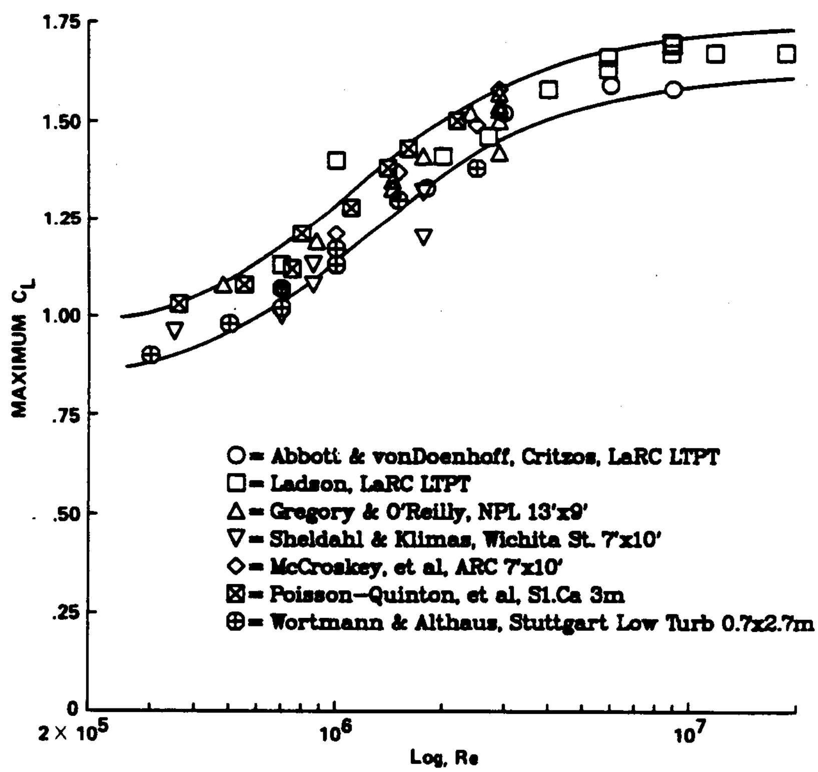

Fig. 13. Maximum lift vs. Reynolds number; Groups 1-2, no trip; M < 0.25.

The effect of Mach number on $C_{\ell_{max}}$ is shown in Fig. 14, for $Re > 2 \times 10^6$. The scatter below $M = 0.25$ seems to be partly due to Reynolds number and partly due to wind-tunnel wall effects. However, local transonic effects in the leading-edge region evidently play an increasingly dominant role in the stall process at $M = 0.25$ and above, where the maximum lift starts to monotonically decrease with increasing M. It is interesting to note that most of the Group 4 data are only slightly below the data from Groups 1-3 at $M > 0.4$, and the scatter in this regime is surprisingly small.

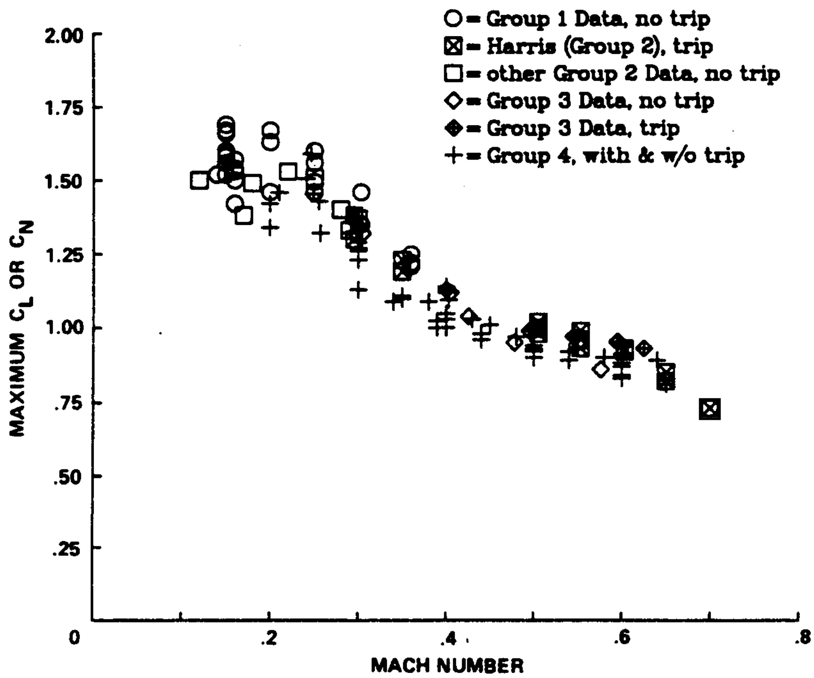

Fig. 14. Maximum lift vs. Mach number; all data, $2 \times 10^6 < Re < 10^7$.

E. Shock-Wave Position

As noted in the Introduction, there is so little overlap in the specific transonic test conditions of the myriad experiments, that most comparisons are necessarily limited to force and moment data. However, some interesting comparisons can be made of the measured shock-wave positions, as this quantity appears to be particularly sensitive to wall-interference effects and to errors in Mach number.

Data from 17 experiments at $M = 0.80$ and $\alpha = 0$ are plotted in Fig. 15, where X_s is defined as the approximate midpoint of the pressure rise across the shock wave. In this figure, the open diamond symbols represent data obtained at sufficiently-large aspect ratios that side-wall boundary layer effects should be minimal, and the solid diamond is a data point corrected by W. G. Sewall in a private communication using his theoretical analysis of side-wall effects [21]. (The principal effect is to increase the effective Mach number by about 0.01). The squares denote experiments in which the side-wall boundary layer was either removed or its effect corrected for. The circles represent the remaining sources, for which no particular attention appeared to be given to side-wall effects.

The grouping of the data in Fig. 15 is inspired by recent numerical analyses [22,23], which showed the tendency of three-dimensional viscous effects on airfoils in wind tunnels to move the shock wave forward of its two-dimensional position. This explanation is tempting for some of the data with unreasonably small values of X_s, but data from several other sources without side-wall treatment appear "normal." Neither does there seem to be any systematic effect of other factors, such as boundary-layer trips or the amount of tunnel slot or perforation openness. Although the majority of the results seem to lie between $X_s = 0.44$ and 0.48, the overall scatter is disturbing, and the actual reason for it remains a mystery. Therefore, this is yet another area where the key experimental information that would be valuable for CFD code validation is not satisfactory.

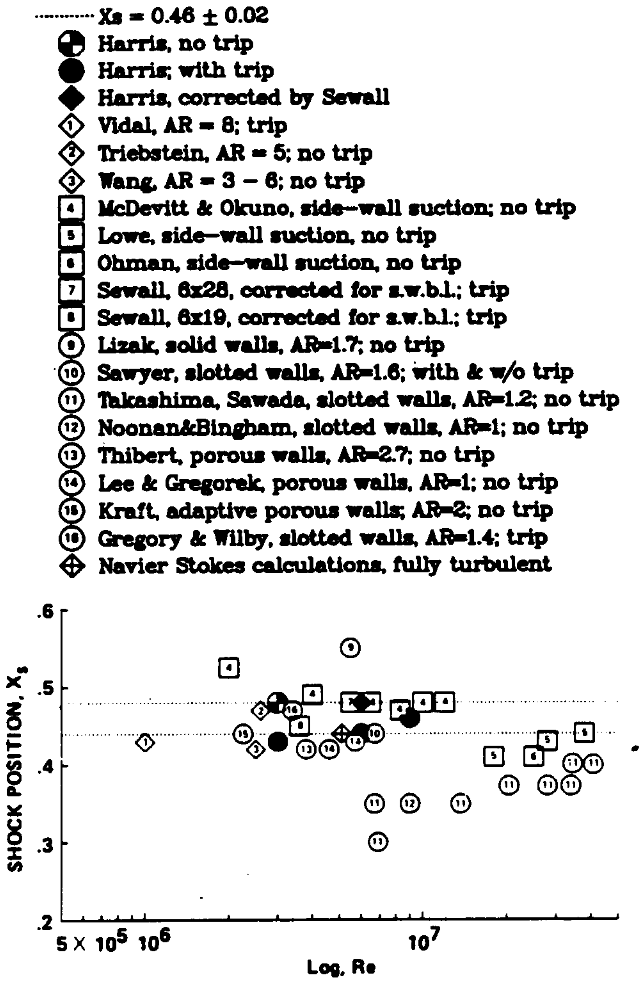

Fig. 15. Shock-wave position vs. Reynolds number at M = 0.80 and α = 0; all data.

IV. SUMMARY AND CONCLUSIONS

Results from more than 40 two-dimensional wind-tunnel experiments have been critically examined and analyzed.* Sadly, the scatter in the total ensemble of data is unacceptable in the author's view, and it is not readily apparent which of these results are correct. It is clear, however, that the requirements for flow quality and data accuracy set forth in AGARD Advisory Report 184 [4] are seldom met in airfoil testing.

The results of this investigation also suggest that no single existing experiment is adequate either for defining the complete aerodynamic characteristics of the NACA 0012 airfoil, or for validating CFD codes.

Nevertheless, the aggregate of available data is extremely useful. A systematic screening process has been used to help define the relative merits of the various experiments and to filter considerable useful, quantitative information from the confusion. Correlations of key parameters with Mach and Reynolds number have also narrowed the uncertainty in the airfoil section characteristics to acceptable levels, and the judicious use of airfoil theory and numerical calculations permits extrapolations to be made into regimes where hard evidence is sparse. This combined information serves three important functions. First, it allows individual experiments to be critiqued with more confidence than heretofore; second, it allows the complete NACA 0012 airfoil characteristics to be estimated more precisely. Third, the synthesized results presented in the figures and equations can be used to establish the credibility of individual airfoil facilities.

On the basis of both completeness and accuracy, the experiment of Harris [5], chosen by Holst [16] in his recent validation exercise for viscous transonic airfoil analyses, emerges as the most satisfactory

*Tabulations of the data presented in this paper are available from the author upon written request.

single investigation of the conventional NACA airfoils to date. Harris' range of flow conditions is not nearly as complete as desired, and the accuracy of the data was not evident a priori, as lift-interference corrections on the order of 15% were proposed for the angles of attack. However, the present study indicates that Harris' estimates of this phenomenon are, in fact, adequate, at least for low angles of attack, and that most other major sources of errors were minimized. On the other hand, the author is persuaded by the arguments of Mr. W. G. Sewall [21] that some side-wall boundary-layer interference existed. Therefore, it is strongly recommended that this be corrected for before using Harris' data for CFD code validation.

As discussed in Section III, the values of lift-curve slope and minimum drag in subsonic flow can now be established with high confidence in the Reynolds number range $10^6 < Re < 3 \times 10^7$. The behavior of these key quantities can also be estimated throughout the transonic regime and up to low supersonic Mach numbers, but with rapidly-deteriorating confidence above $M = 0.8$. The issue of self-induced oscillations and the possibility of negative values of C_{ℓ_α} in the range $0.85 < M < 0.90$ need further investigation. A better definition of the behavior at and above $M = 1$ would be useful for CFD code validation.

The variations of $C_{\ell_{max}}$ with M and Re can now be specified with a moderate degree of confidence, and the data from most of the available sources are surprisingly consistent above $M = 0.4$. This conclusion appears to contradict folklore, conventional wisdom, and recent numerical studies of wall interference.

On the other hand, the behavior of the maximum lift-to-drag ratio and shock-wave position is not nearly as well defined, and both these quantities appear to be particularly sensitive to wind-tunnel wall effects and turbulence. Therefore, additional studies under carefully-controlled conditions are strongly recommended. It is also suggested that both of these quantities would be especially important criteria for CFD code validation, if they could be reliably established by well-documented experiments.

Finally, the results of this investigation indicate that measurements, corrections, and/or treatments for all four walls of the test section are essential for any reasonably-sized model under transonic flow conditions. Although results from some facilities appeared to suffer more than others from wall-interference effects, no facility that failed to address the potential problems on all four walls provided data that could be judged entirely satisfactory.

V. ACKNOWLEDGEMENTS

The author is extremely grateful to the many people who generously shared stimulating ideas and insights, background information, reference sources, and unpublished results during the course of this investigation. The manifold contributions of Mssrs. Charles Ladson and William Sewall of NASA-Langley, including extensive unpublished data, were truly invaluable. Grateful acknowledgement is also extended to Dr. Terry Holst of NASA-Ames, Mr. Frank Harris of Bell Helicopter Textron, and Mr. Ray Prouty of McDonnell-Douglas Helicopters, for their helpful comments, suggestions, and unpublished information. Mr. Lars Ohman of the National Aeronautical Establishment and Mssrs. B.F.L. Hammond and T.E.B. Bateman of the Aircraft Research Association, Ltd. provided Mach-number corrections and other useful information concerning their respective facilities. Also, Mssrs. Lawrence Green, Clyde Gumbert, and Perry Newman of NASA-Langley, Herr D. Althaus of the Universität Stuttgart, Prof. Siegfried Wagner of Universität der Bundeswehr München, Mr. Kazuaki Takashima of the National Aerospace Laboratory, and Ms. Mary Berchak of Ohio State University kindly provided explanations and tabulations of unpublished data, and their generous assistance is deeply appreciated.

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Table 1. NACA 0012 - Summary of Experiments -- Group 1

SOURCE	MACH range	Re (10^6) range	TRIP ? Xt	TUNNEL CHAR.	REMARKS
1. Abbott et al.; Langley LTPT	0.07-0.15	0.7-26	yes & no "Std. R"	solid walls AR = 0.75-6 h/c= 1.9-15	linear wall corrections; very low turbulence; excessively thick trip; possible minor side-wall boundary-layer effects
data available: C_ℓ, C_m, C_d, $(L/D)_{max}$, $C_{\ell_{max}}$					
2. Ladson; Langley LTPT	0.07-0.36	0.7-19	yes & no Xt=0.05	solid walls AR = 1.5 h/c = 3.8	linear wall corrections; very low turb. at low M; possible minor side-wall boundary-layer effects
data available: C_ℓ, C_m, C_d, $(L/D)_{max}$, $C_{\ell_{max}}$					
3. Gregory and O'Reilly; NPL 13'x9'	0.08-0.16	1.4-3	yes & no varying	solid walls AR = 3.6 h/c = 5.2	linear wall corrections; with & w/o side-wall boundary-layer control
data available: C_ℓ, C_m, C_d, l.e. C_p, $(L/D)_{max}$, $C_{\ell_{max}}$					
4. Green & Newman; Langley 0.3m TCT	0.5 - 0.8	9	yes Xt = 0.05	adaptive walls AR = 2 h/c = 2	four-wall corrections; moderate turb. level
data available: C_{ℓ_α}, C_{d_0} (low α only)					

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Table 2 - Summary of Experiments -- Group 2

SOURCE	MACH range	Re (10^6) range	TRIP ? Xt	TUNNEL CHAR.	REMARKS
5. Harris; Langley 8' TPT	0.3 - 0.86	3 - 9	yes & no Xt=0.05	slotted walls AR = 3.4 h/c = 3.4	large α corrections; possible side-wall boundary effects on X_s & C_d
data available: C_n, C_m, C_d, C_p, $(L/D)_{max}$, X_s, limited $C_{\ell_{max}}$					
6. Goethert; DVL 2.7m W.T.	0.3 - 0.85	2 - 6	no	solid walls AR = 2.6 h/c = 5.4	wall and end-plate corrections; turbulence level ≈1%; some flow asymmetry
data available: C_ℓ, C_m, C_d, C_p					
7. Sheldahl & Klimas Wichita St. 7'x10'	0.1-0.2	0.35-1.8	no	solid walls AR = 2.4-6 h/c= 5.6-15	linear wall corrections; some flow asymmetry; $0 < \alpha < 180$
data available: C_p, C_d, $(L/D)_{max}$, $C_{\ell_{max}}$					
8. McCroskey, et al Ames 7'x10' No.2	0.1-0.3	1 - 4	yes & no Xt = 0.01	solid walls AR = 3.5 h/c = 5	linear wall corrections; continuous dynamic data
data available: C_ℓ, C_m, limited C_d, C_p, $(L/D)_{max}$					
9. Bevert; Poisson Quinton & de Sievers; S1.Ca 3m	0.06-0.11	1.1-2.2	no	solid walls AR = 1.3 h/c = 4	linear wall corrections; Tu < 0.2%
data available: C_ℓ, C_m, C_d, C_p, $(L/D)_{max}$, $C_{\ell_{max}}$					
10. Wortmann & Althaus; Techn. Hochs. Stuttgart Lam. W.T.	0.07-0.17	0.3-2.5	no	solid walls AR = 1.5-3 h/c= 5.5-11	side-wall suction; very low turbulence early C_{ℓ_α} suspect
data available: C_ℓ, C_d, $(L/D)_{max}$, $C_{\ell_{max}}$					

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Table 3 - Summary of Experiments -- Group 3

SOURCE	MACH range	Re (10^6) range	TRIP ? Xt	TUNNEL CHAR.	REMARKS
11. Bernard-Guelle; ONERA R1.Ch	0.325	3.5	no(?)	solid walls AR = 0.67 h/c = 3.3	side-wall suction, careful study of side-wall effects
data available: limited C_ℓ, C_m, C_d					
12. Sawyer; ARA 8"x18" Trans. W.T.	0.3 - 0.85	3 - 6	yes & no Xt=0.07	slotted walls AR = 1.6 h/c = 3.6	α, M, and curvature corrections; poss. side-wall boundary layer effects
data available: C_ℓ, C_d, C_p, $C_{\ell_{max}}$, $(L/D)_{max}$, X_s					
13. Vidal et al. CALSPAN 8'	0.4 - 0.95	1	yes xt=0.1	porous walls AR = 8 h/c = 16	thick transition strips; slight flow angularity; minimum interference
data available: C_ℓ, C_m, C_d, C_p, $(L/D)_{max}$, limited $C_{\ell_{max}}$, X_s					
14. McDevitt & Okuno; Ames Hi-Re Channel	0.72 - 0.8	2 - 12	no	solid walls AR = 2 h/c = 3	contoured walls, wall pressure meas.; side-wall suction; unsteady measurements
data available: C_{ℓ_α}, C_p, X_s (low α only)					
15. Gumbert & Newman; Langley 0.3m TCT	0.7 - 0.8	3 - 9	yes & no Xt=0.05	slotted walls AR = 1.3 h/c = 4	α corrected; side-wall boundary-layer corrections
data available: C_{ℓ_α}, C_{d_0} (low α only)					
16. Takashima, Sawada et al. NAL Transonic W.T.	0.6 - 0.8	4 - 39	no	slotted walls AR = 1.2 - 2 h/c = 4 - 6.7	wall pressure-rail meas.; poss. side-wall b.l. effect on shock position;
data available: C_ℓ, C_d, C_p, X_s (low α only)					
17. Sewall; Langley 6" x 28" (revised)	0.3 - 0.83	4 - 9	yes & no Xt=0.08	slotted walls AR = 1 - 2 h/c= 4.7-9.3	α and side-wall b.l. corrections
data available: C_ℓ, C_m, C_d, $C_{\ell_{max}}$, X_s					
18. Lowe General Dyn. Hi-Re 2D Test Sect, HSWT	0.63-0.82	15-38	no	perfor. walls AR = 1 h/c= 4	22% perforation, side-wall suction; uncertain α corr.
data available: C_ℓ, C_d, C_p, X_s					
19. Jepson; Lizak; Carta; UTRC 8'	0.3 - 0.9	2 - 6	no	solid walls AR= 1.7-5.8 h/c=4.7-5.8	linear wall corrections; multiple entries; various models and end plates
data available: C_ℓ, C_m, C_d, C_p, $(L/D)_{max}$, $C_{\ell_{max}}$, X_s					
20. Wang et al. Chinese Aero. Inst. Transonic W.T.	0.7 - 0.9	~3(?)	yes Xt=0.06	perfor. walls AR= 3.2-6.4 h/c=2.6-5.2	porosity adjusted for min. interference
data available: limited C_ℓ, C_p, X_s					

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Table 4 - Summary of Experiments -- Group 4

SOURCE	MACH range	Re (10^6) range	TRIP ? Xt	TUNNEL CHAR.	REMARKS
21. Sewall; LaRC 6"x19"	0.58 - 0.92	3 - 4	yes Xt=0.08	slotted walls AR = 1 h/c = 3.2	data corrected for thick side-wall boundary interference but not lift interference
data available: C_{n_α}, C_{d_o}, X_s					
22. Noonan & Bingham; Ladson; LaRC 6"x28"	0.35 - 1.0	1-10	yes & no Xt = 0.1	slotted walls AR = 1.0 h/c = 4.7	α corrected; side-wall b.l. effects on shock position and $C_{\ell_{max}}$
data available: C_n, C_m, C_d, C_p, $(L/D)_{max}$, $C_{\ell_{max}}$, X_s					
23. Ohman, et al; NAE 5' x 5' with 2D insert	0.5 - 0.93	17-43	no	porous walls AR = 1.3 h/c = 5	20% porosity; side-wall suction; data slightly asymmetric; Mach No. corrected herein
data available: C_{d_o}, C_p, X_s at $\alpha = 0$					
24. Thibert, et al; ONERA S3.Ma	0.3 - 0.83	1.9 - 4	no	porous walls AR = 2.7 h/c= 3.7	large wall corrections, but wall press. measured; thick side-wall b.l.
data available: C_ℓ, C_d, C_p, X_s					
25. Scheitle & Wagner; TWT München Univ. Bundeswehr	0.36 - 1.6	3 - 10	no	slotted walls AR = 1.5 h/c = 3.4	suction on all four walls, variable with M to match other facilities; moderate turb. level
data available: C_{ℓ_α}, $C_{d_{min}}$, $(L/D)_{max}$, $C_{\ell_{max}}$					
26. Jepson; NSRDC 7'x10'	0.3 - 1.08	2 - 5	no	slotted walls AR = 7.5 h/c = 5.3	large lift interference
data available: C_ℓ, C_m, C_d, $(L/D)_{max}$, $C_{\ell_{max}}$					
27. Lee, et al; Ohio State 6"x22" Trans. Airf. Facil.	0.2 - 1.06	2 - 12	no	porous walls AR = 0.5 - 2 h/c= 0.9-7.1	independent plenums for top and bottom walls
data available: C_ℓ, C_m, C_d, $(L/D)_{max}$, $C_{\ell_{max}}$, X_s, limited C_p					
28. Prouty; LAC 15"x48"	0.34-0.96	3 - 7	no	slotted walls AR = 1.5 h/c = 4.6	large lift interference; poss. side-wall boundary layer effects; some flow asymmetry
data available: C_ℓ, C_m, C_d, $(L/D)_{max}$, $C_{\ell_{max}}$					
29. Gregory & Wilby; NPL 36"x14"	0.3-0.85	1.7-3.8	yes Xt=0.02	slotted walls AR = 1.4 h/c = 3.6	probable wall effects on all data fairly large roughness
data available: C_ℓ, C_m, C_d, C_p, $(L/D)_{max}$, $C_{\ell_{max}}$, X_s					

Table 4 - Concluded.

30. Kraft & Parker; AEDC 1-T	0.8 - 0.9	2.2	no	adaptive walls AR = 2 h/c = 2	variable porosity and hole angle; no side-wall treatment
data available: C_p, X_s					
31. Triebstein; DFVLR 1m TWT	0.5 - 1.0	1 - 3	no	porous walls AR = 5 h/c = 5	no corrections applied; unsteady measurements
data available: X_s, C_p					
32. Ladson; LaRC 6"x19"	0.5 - 1.1	1.5 - 3	no	slotted walls AR = 1.5 h/c = 4.8	α corrected for lift interference but not side-wall boundary layer
data available: C_n, C_m, C_p, surface oil flow, schlieren					
33. Ladson; LaRC ATA 4"x19"	0.8 - 1.25	2.7	no	slotted walls AR = 1.0 h/c = 4.8	no corrections applied
data available: C_n					

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Table 5 - Experiments examined but not used -- Group 5

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16. Abstract

A large body of experimental results, which were obtained in more than 40 wind tunnels on a single, well-known two-dimensional configuration, has been critically examined and correlated. An assessment of some of the possible sources of error has been made for each facility, and data which are suspect have been identified. It was found that no single experiment provided a complete set of reliable data, although one investigation stands out as superior in many respects. However, from the aggregate of data the representative properties of the NACA 0012 airfoil can be identified with reasonable confidence over wide ranges of Mach number, Reynolds number, and angles of attack. This synthesized information can now be used to assess and validate existing or future wind tunnel results and to evaluate advanced Computational Fluid Dynamics codes.

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