SCALE EFFECTS ON SUPERCRITICAL AIRFOILS

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Abstract

A survey of research relative to scale effects on supercritical airfoils has been conducted. The results of this survey indicated that Reynolds number scale effects have a significant impact on airfoil design and performance. Further, this impact is greater for supercritical airfoils than for conventional airfoils. It was found that low Reynolds number drag data could be extrapolated to high Reynolds number conditions provided the flow was attached and the pressure distribution shape did not change appreciably. Airfoil lift and pitching-moment data obtained at low Reynolds numbers cannot be extrapolated to full-scale values. Viscous theoretical transonic analysis methods currently under development will significantly improve the ability of the designer to account for scale effects. Boundary-layer manipulation in low Reynolds number facilities using natural transition or aft located transition strips to simulate high Reynolds number conditions was shown to be an uncertain test procedure and reliance should be made on high Reynolds number facilities if available.

Introduction

Over the years, aerodynamicists have accounted for Reynolds number scale effects* by simply extrapolating scale model wind-tunnel data to flight conditions assuming the dominant Reynolds number effect to be that on skin friction drag. This was shown to be inadequate during the flight testing of the Lockheed C-141 aircraft where large excursions in wing shock travel and variations in force characteristics were experienced relative to wind-tunnel results. This effect is illustrated in Figure 1.

New transports currently under development are incorporating supercritical airfoils in their wing designs. To date, very little data has been published in the open literature quantifying in detail the scale effects on supercritical airfoils. Furthermore, supercritical airfoils appear to be subject to the same type of scale effects as the C-141. Hence, the ability of the designer to extrapolate low Reynolds number data to obtain the full-scale characteristics of the supercritical airfoil is quite uncertain.

A detailed knowledge of the scale effects that might occur on a new aircraft design utilizing supercritical airfoils is essential. In today's competitive environment, scale effect suprises such as those in Figure 1 can be ill afforded. Furthermore, scale effects must be taken into account in the airfoil design process such that the most aerodynamically efficient aircraft possible is achieved.

To quantify and understand these scale effects, Lockheed has designed and tested a large matrix of supercritical airfoils, as well as, several conventional airfoils over a wide range of Reynolds numbers spanning both conventional wind-tunnel and full-scale flight values. This work was accomplished in the Lockheed-Georgia

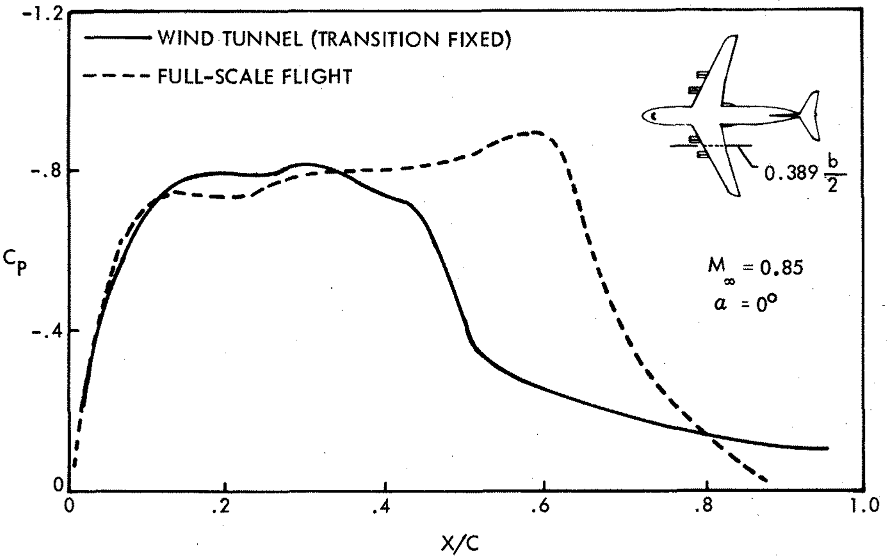

Figure 1. Lockheed C-141 Pressure Distribution

* The term "scale effects" will be used to indicate those changes in aerodynamic characteristics that occur due to large variations in Reynolds number.

Company Compressible Flow Wind-Tunnel (CFWT), which is a high Reynolds number, transonic blow-down tunnel.

The objectives of this paper in dealing with the subject of "scale effects on supercritical airfoils" are:

- o Review the aerodynamic phenomena that are sensitive to scale effects in supercritical airfoil design.
- o Quantify the magnitude of the various types of scale effects for selected supercritical airfoils.
- o Indicate the differences in supercritical and conventional* airfoil scale effects.
- o Assess the feasibility of extrapolating low Reynolds number wind-tunnel data to full-scale flight values.
- o Determine the status of available theoretical and experimental approaches for evaluating scale effects on supercritical airfoils.

Discussion

Scale Effect Sensitive Flow Phenomena

A review of the changes in airfoil aerodynamic characteristics due to large variations in Reynolds number indicates the following viscous flow phenomena are sensitive to scale effects.

- o Boundary-layer thickness
- o Skin friction
- o Pressure-gradient induced boundary-layer separation
- o Shock induced boundary-layer separation

In the following paragraphs, the reaction of each of these viscous flow phenomena to scale effects as they relate to supercritical airfoils will be summarized.

Boundary-Layer Thickness. The flow around an airfoil can generally be separated into an inviscid region and a viscous boundary-layer region. The airfoil geometry combined with the boundary-layer is known as the fluid airfoil. It is the shape of the fluid airfoil that determines the airfoil aerodynamic characteristics. Since scale effects alter the boundary-layer thickness, the fluid airfoil also changes shape with the attendant changes in the airfoil aerodynamic characteristics. This effect is summarized for a supercritical airfoil at subsonic and transonic speeds in Figures 2 and 3, respectively.

At subsonic speeds (Figure 2), the primary scale effect is to change the airfoil camber and angle of attack. As the Reynolds number is decreased from flight values to conventional wind tunnel values, there is a loss in airfoil lift resulting in the zero-lift angle of attack shifting to a more positive value and the pitching-moment coefficients becoming more positive. The loss in lift for supercritical airfoils as Reynolds number is decreased from high to low values is generally on the order of 5 to 10%, but can be as much as 50% as will be demonstrated subsequently.

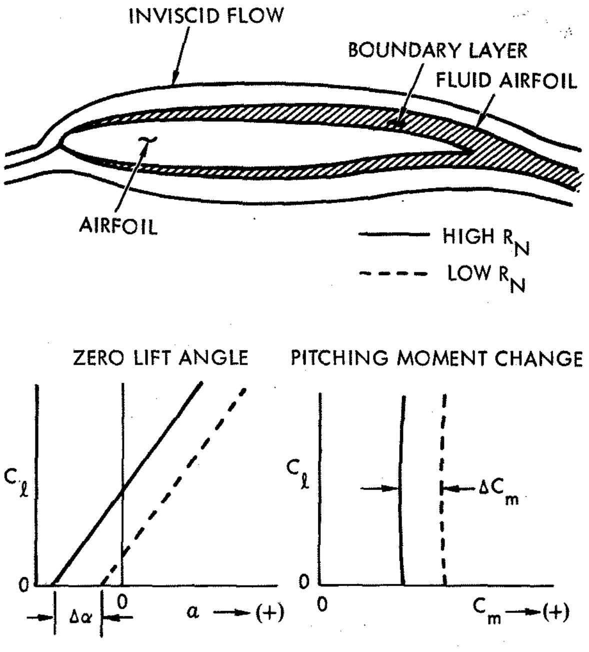

Figure 2. Scale Effect on Boundary-Layer Thickness (Subsonic)

At transonic speeds, a shock wave is added to the system (Figure 3). Assume that the high Reynolds number pressure distribution (curve 1) shown in Figure 3 is the desired supercritical airfoil pressure distribution for cruise conditions. As Reynolds number is lowered (Curve 2), there is a loss in camber and effective angle of attack similar to that shown in Figure 2. Also, for a constant lift coefficient, a stronger shock wave occurs (relative to curve 1) and an earlier drag-rise results.

A second high Reynolds number pressure distribution designated as curve 3 is sketched in Figure 3 to illustrate another possible scale effect problem on supercritical airfoils at transonic speeds. The pressure distribution indicated as curve 3 is characterized by a second shock on the airfoil upper surface. This situation can occur if an airfoil shape is designed at low Reynolds numbers with too much geometric curvature incorporated over the aft portion of the airfoil. The second shock cannot be detected at low Reynolds numbers since the thick boundary layer masks the curvature effect. As the Reynolds number is increased from low to high values, the flow becomes more sensitive to the geometric curvature due to the thinning of the boundary layer, and if sufficient curvature is present a second shock appears. If a second shock does occur, the anticipated improvement in airfoil

* Standard NACA airfoil sections.

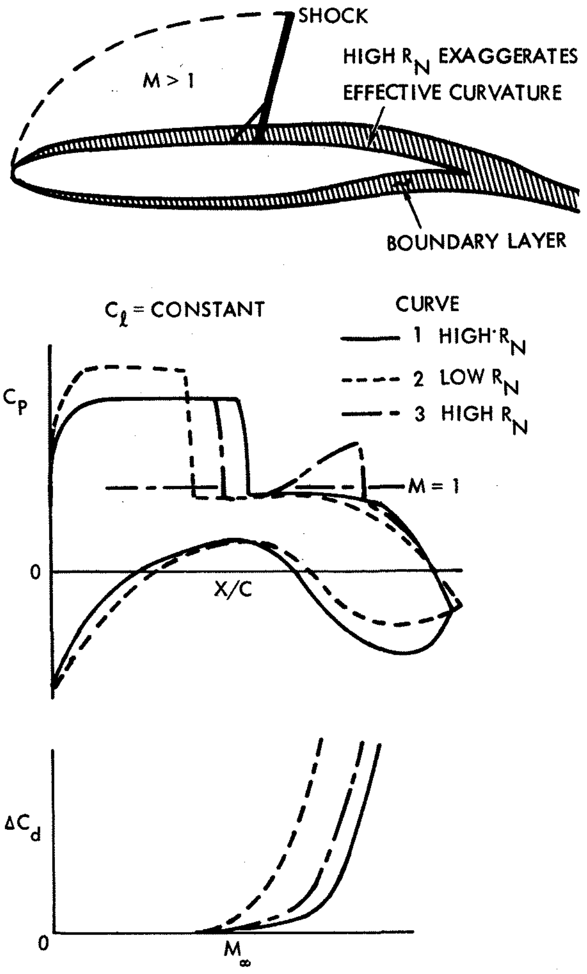

Figure 3. Scale Effect on Boundary-Layer Thickness (Transonic)

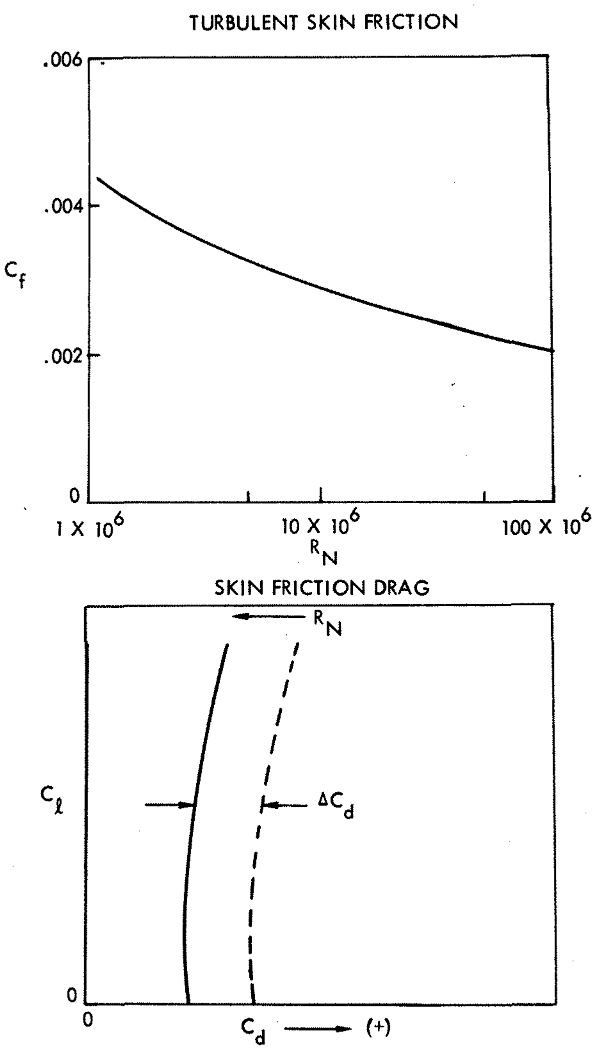

Figure 4. Scale Effect on Skin Friction Drag

drag performance with increasing Reynolds number will be reduced.

Skin Friction. The scale effect on turbulent skin-friction drag is illustrated in Figure 4. The variation of skin friction drag on a flat plate for zero pressure gradient with increasing Reynolds number is well know. As long as the flow is attached, subcritical, and not close to separation, the airfoil profile drag will generally scale according to flat plate skin-friction curves similar to that in Figure 4.

Pressure-Gradient Induced Separation. In Figure 5, the problems resulting from scale effect on pressure-gradient induced boundary-layer separation are indicated. This particular scale-effect phenomena is more prevalent today than in the past due to the advent of supercritical airfoils. In the design of an airfoil which has critical adverse pressure gradients such as near the airfoil trailing-edge, the amount of adverse pressure gradient allowed is highly dependent on Reynolds number. This can pose several significant problems to the airfoil designer. For instance, if a gradient is optimized for high Reynolds number conditions, the flow will separate when the boundary-layer encounters this pressure gradient at low Reynolds numbers. This precludes the use of conventional low Reynolds number facilities for design verification. If the airfoil design is accomplished at low Reynolds numbers such that separation does not occur due to the adverse pressure gradients, then the design is too conservative for high Reynolds number operation and does not represent an optimum configuration.

Shock-Induced Boundary-Layer Separation. A model of a transonic shock-wave/boundary-layer interaction typical of that occurring on supercritical airfoils is sketched in Figure 6. Of particular note are the two types of shock-induced separation shown: separation bubble at the foot of the shock wave and a rear separation at the airfoil trailing-edge. The rear separation is the most common type of shock-induced separation on supercritical airfoils. This is due to the upper-surface shock wave being sufficiently weak near cruise conditions so as not to precipitate a separation bubble at the shock. For non-optimized supercritical airfoils which have strong upper-surface shock waves (local Mach number ahead of shock >1.25) bubble separations tend to form but

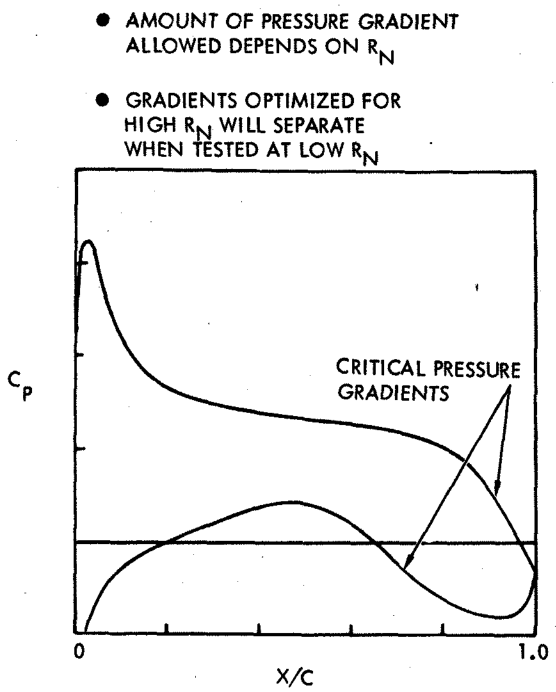

Figure 5. Scale Effect on Pressure Gradient Induced Boundary-Layer Separation (Subsonic)

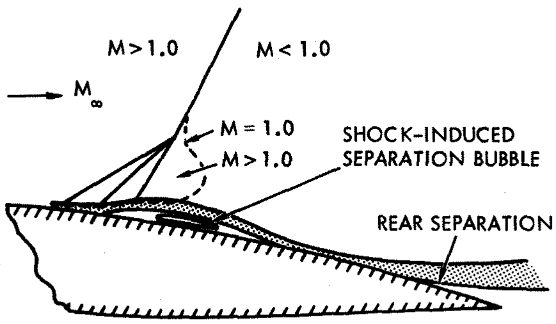

Figure 6. Model of Transonic Shock/Boundary-Layer Interaction

do not grow rapidly. This is probably due to the near zero pressure gradient that follows the shock (see Figure 3, curve 1) which stabilizes the boundary-layer. A similar type bubble separation can occur on supercritical airfoils at lift coefficients substantially higher than cruise conditions where strong shock waves are encountered.

Scale effects on the two types of separation have been shown to be very important in Reference 1. However, the sensitivity of each type of separation to variations in Reynolds number is different with the rear separation being the most sensitive.

The effects of varying Reynolds number on the airfoil performance for conditions where shock/boundary-layer separation is present is shown in Figure 7. As the Reynolds number is increased, the shock wave on the airfoil generally moves rearward. If rear separation is present, the extent of the separation decreases. Also, as the

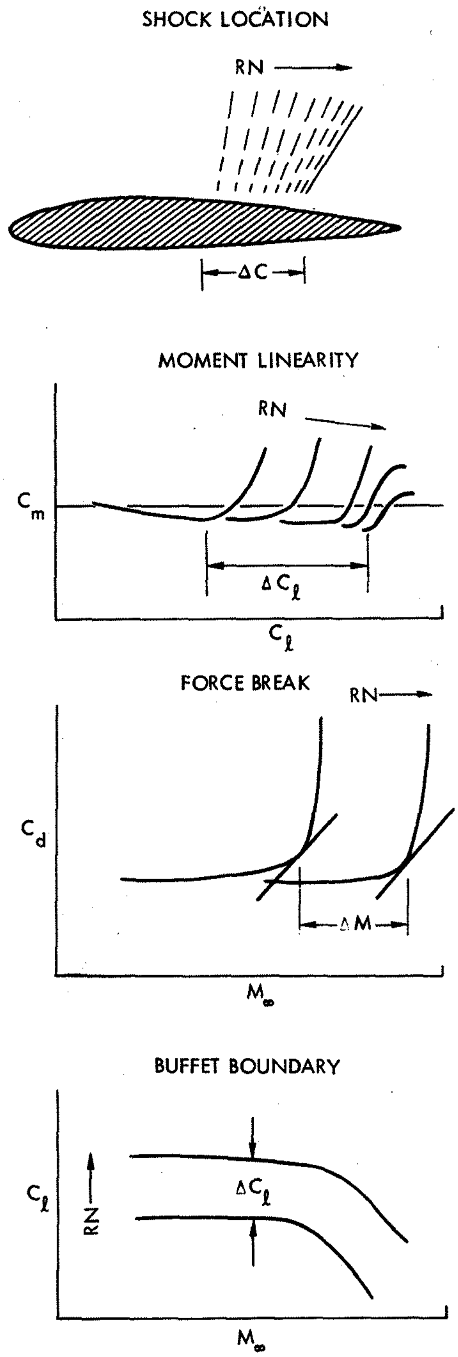

Figure 7. Scale Effect on Shock-Induced Boundary Layer Separation

Reynolds number is increased the drag-rise Mach number increases, the pitching-moment linearity is extended, and improvements in the airfoil trailing-edge pressure divergence occur (which is indicative of improvements in the airfoil buffet characteristics.)

Experimental Illustrations of Scale Effect Magnitudes

In order to illustrate in a simple way the magnitude of the scale effects that could be expected for the various types of phenomena discussed in the previous section, experimental data for a variety of airfoils* will be presented over a Reynolds number range indicative of conventional wind-tunnel and flight Reynolds numbers. All data presented in this section were obtained in the Lockheed-Georgia Compressible Flow Wind-Tunnel (CFWT).

The CFWT is a variable porosity transonic wind tunnel capable of a Mach number range of 0.2 to 1.2 and a Reynolds number range from 3 million to 32 million based on model chord. The test section is 50.8 cm (20 in.) wide by 71.2 cm (28 in.) high by 183 cm (72 in.) long. The airfoil chords were 17.8 cm (7 in.) in length. The wall porosity for all tests was 4%. A typical installation in the CFWT is shown in Figure 8.

All low Reynolds number data were obtained with transition fixed at 5% chord unless otherwise noted.

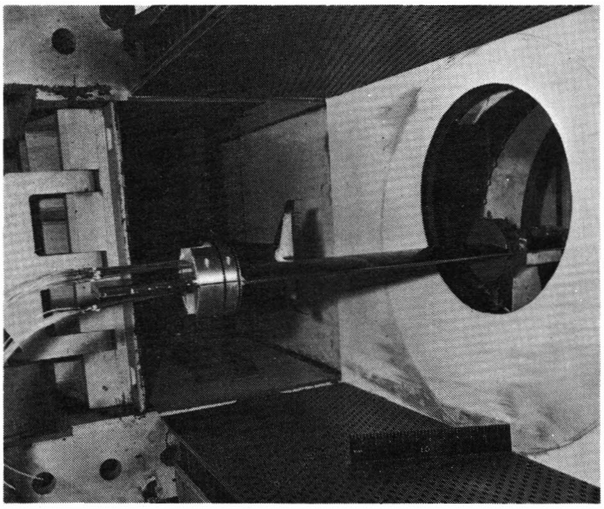

Figure 8. Model of NACA 65_1-213 Airfoil Installed in Lockheed-Georgia Compressible Flow Wind Tunnel (One Wall Removed)

<u>NACA 65_1-213 Airfoil.</u> Experimental results will first be presented on a conventional airfoil to provide a basis of comparison for the scale effects on supercritical airfoils. The NACA 65_1-213 airfoil (Figure 9) was chosen since it is representative of conventional airfoils and it has been the subject of several previous scale-effect investigations[2,3].

Figure 9. NACA 65_1-213 Airfoil

The scale effect on the NACA 65_1-213 airfoil pressure distribution is shown in Figure 10. The three million Reynolds number data is indicative of scale-model tests in conventional facilities. The 32 million data is representative of flight conditions. Looking first at conditions representatative of cruise for this particular airfoil (α = 0, M_∞ = 0.75), it can be seen that a strong shock is present; however, the flow is still attached. The scale effects at this condition are very small. The scale effects are more pronounced as the angle of attack is increased to three degrees (Figure 10(b)). For the low Reynolds number condition, the flow separates at the trailing-edge. As Reynolds number is increased, the shock moves rearward approximately 3% chord and the extent of flow separation decreases.

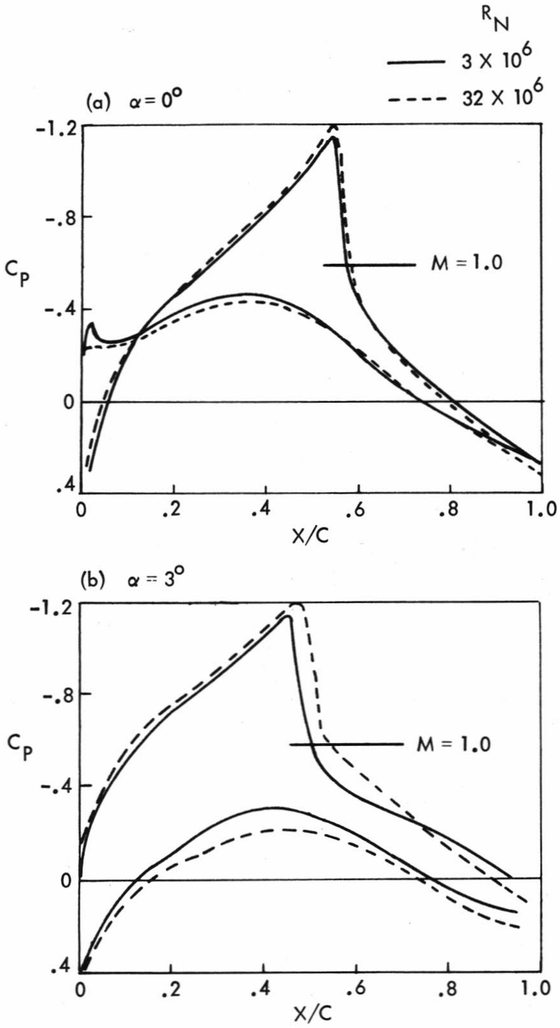

Figure 10. Scale Effect on Pressure Distributions for NACA 65_1-213 Airfoil, M_∞ = .75

The scale effects on drag for the NACA 65_1-213 airfoil are presented in Figure 11. For the condition shown, the flow is subsonic and attached. The purpose of this data is to indicate

* The airfoils were selected to illustrate certain phenomena and therefore do not necessarily represent optimum designs.

whether or not low Reynolds number drag data can be extrapolated to full-scale values. The estimated curve in Figure 11 was obtained by correcting the low Reynolds number data for skin friction variations (Figure 4) due to Reynolds number. Comparing the estimated and experimental curves shows that it is possible to extrapolate low Reynolds number data to high Reynolds numbers provided the flow is attached and the shape of the pressure distribution does not change appreciably.

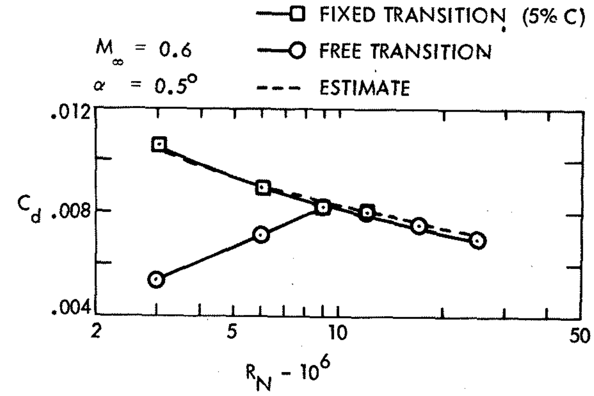

Figure 11. Scale Effect on Drag for NACA 65_1-213 Airfoil

The effect of Reynolds number on the drag-rise characteristics of a conventional airfoil are shown in Figure 12. For the cruise condition ($\alpha = 0^o$) there is virtually no effect. This is consistent with the results of Figure 10 where hardly any shock movement was noted. At an angle of attack of 3 degrees, some scale effects on drag creep are noted, but little effect on drag-rise Mach number is evident. Although not shown in Figure 12, large-scale effects on drag level were obtained for conditions where the flow was well separated (Figure 10, M = 0.75, $\alpha = 3^o$).

<u>10% Thick Supercritical Airfoil.</u> The 10% thick supercritical airfoil is representative of airfoil sections that would be used on future long-range subsonic transports. A sketch of a NASA 10% supercritical airfoil[4] is shown in Figure 13.

The scale effect on force and pressure data at subcritical speeds is indicated in Figure 14 with a Reynolds number of 4 million used as a datum. As Reynolds number is increased from 4 to 25 million, the principal effect is the viscous uncambering (Figure 2) that occurs over the aft portion of the airfoil.

At conditions indicative of cruise (M_∞ = 0.80, $\alpha = 1.5^o$), the scale effects on the 10% supercritical airfoil are much larger (Figure 15) than those shown at subsonic speeds. As can be seen, the viscous uncambering effects have increased and a 4% shift in shock-wave location is noted. The flows for the conditions in Figure 15 are attached.

Also of interest in Figure 15 is the re-expansion of the flow behind the upper-surface shock wave at a Reynolds number of 32 million. It is this tendency to re-expand that leads to the double shock condition noted in Figure 3.

Comparing the cruise results for the 10% supercritical airfoil in Figure 15 to similar results for the NACA 65_1-213 airfoil in Figure 10 ($\alpha = 0^o$) indicates the supercritical airfoil to be considerably more sensitive to scale effects.

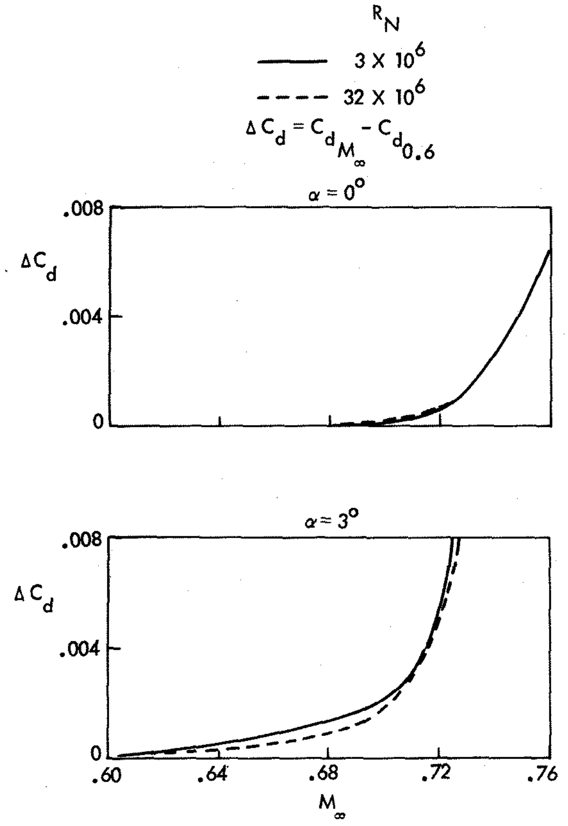

Figure 12. Scale Effect on Drag-Rise Characteristics for NACA 65_1-213 Airfoil

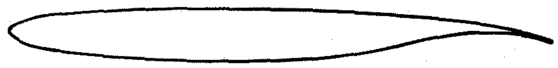

Figure 13. NASA 10% Thick Supercritical Airfoil

In Figure 16 the scale effect on the 10% supercritical airfoil is presented for a condition well into the drag-rise. Large effects are evident. As Reynolds number is increased the shock moves aft approximately 7% and the flow behind the shock goes from separated to attached. Substantial uncambering effects are also noted.

In Figure 17 the scale effect on drag is presented for a subsonic attached flow condition. As with the NACA 65_1-213 airfoil (Figure 11), the results indicate the low Reynolds number data can generally be extrapolated to high Reynolds number

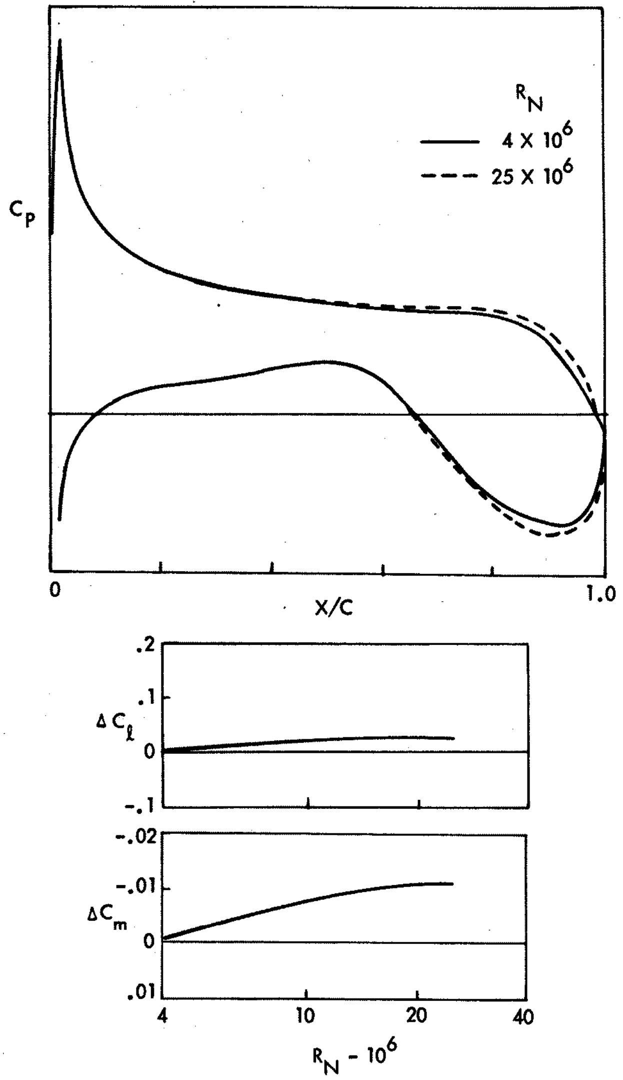

Figure 14. Scale Effect on Force and Pressure Data for NASA 10% Thick Supercritical Airfoil, $M_\infty = 0.60$, $\alpha = 1.5^\circ$

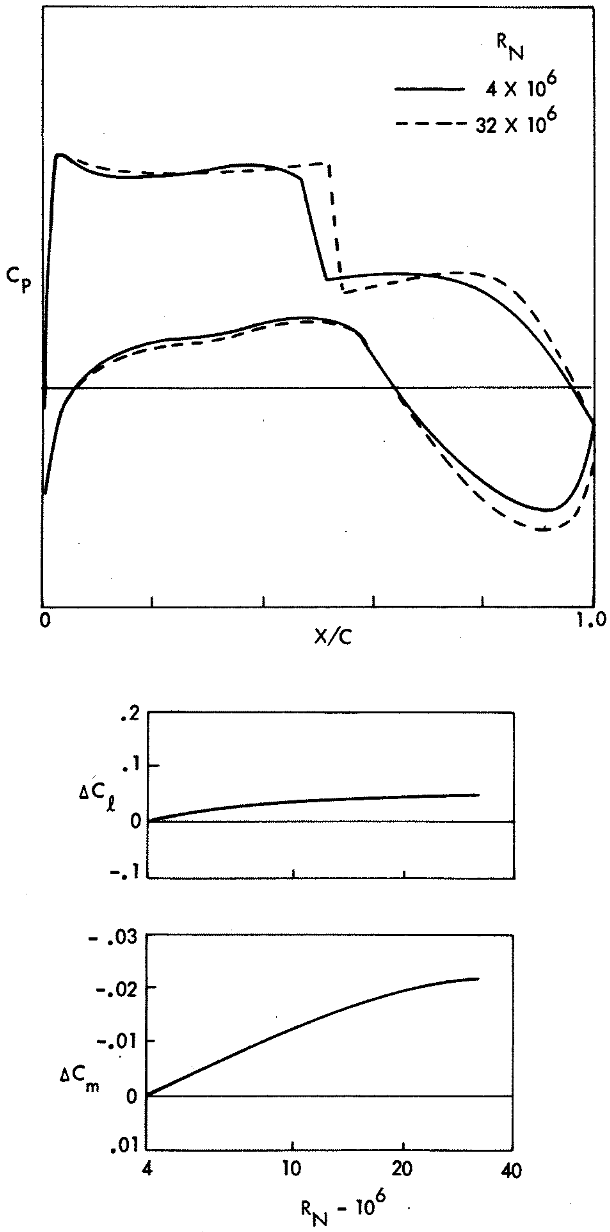

Figure 15. Scale Effect on Force and Pressure Data for NASA 10% Thick Supercritical Airfoil, $M_\infty = 0.80$, $\alpha = 1.5^\circ$

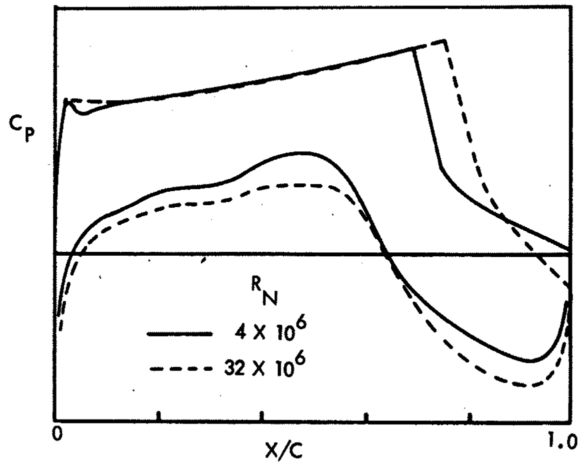

Figure 16. Scale Effect on Pressure Distributions for NASA 10% Thick Supercritical Airfoil, $M_\infty = 0.84$, $\alpha = 1.5^\circ$

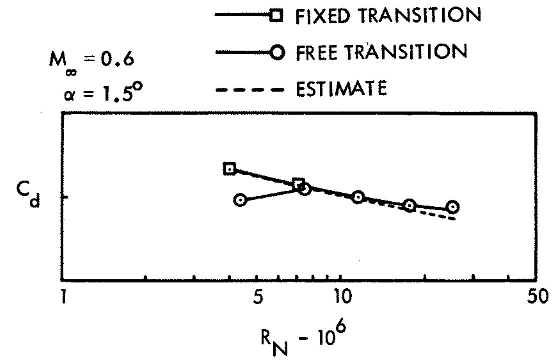

Figure 17. Scale Effect on Drag for NASA 10% Thick Supercritical Airfoil

conditions provided the flow is attached and the pressure distribution shape does not change appreciably.

The effect of large variations in Reynolds number on the drag rise characteristics of the 10% thick supercritical airfoil at constant angle of attack is presented in Figure 18. Near the cruise angle of attack (α = 1.5°), increasing Reynolds number reduces the airfoil drag creep. Little scale effect on drag rise Mach number is noted, however, for either angle of attack shown. Drag-rise results are also presented in Figure 19 where the section lift coefficient is held constant and is equal to the design lift. Significant scale effects are seen. The airfoil at low Reynolds number must operate at a higher angle of attack to achieve the design lift coefficient than it does at high Reynolds numbers. This results in a stronger upper surface shock and precipitates an earlier drag rise Mach number.

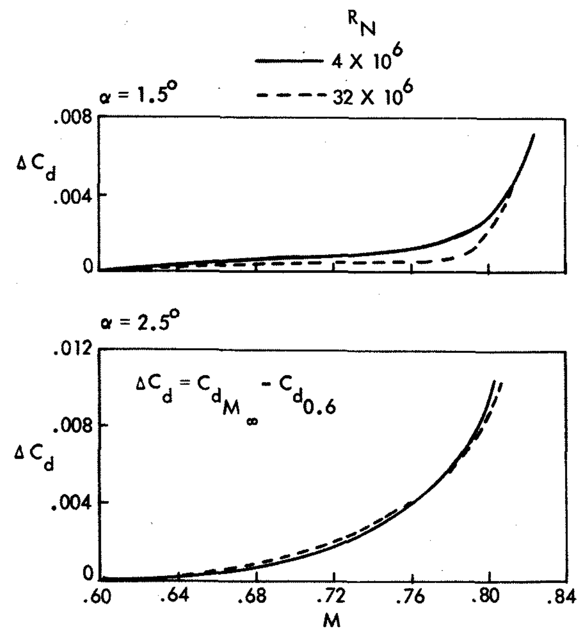

Figure 18. Scale Effect on Airfoil Drag Rise for NASA 10% Thick Supercritical Airfoil

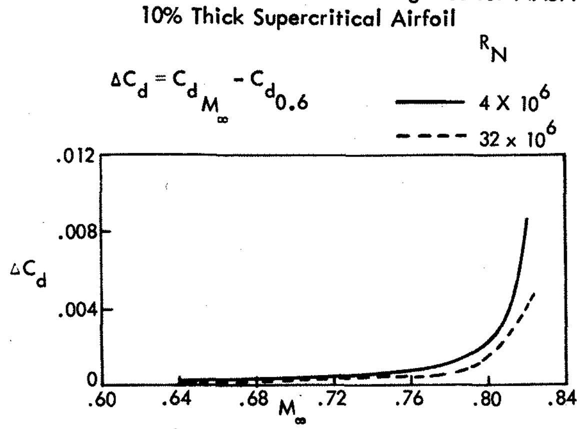

Figure 19. Scale Effect on Airfoil Drag Rise for NASA 10% Thick Supercritical Airfoil

In Figures 18 and 19, it was shown that significant variations in airfoil creep drag and wave drag occur for changes in Reynolds number spanning conventional wind-tunnel and flight values. These drag variations preclude the use of any low to high Reynolds number extrapolation procedure based on skin friction considerations.

Inspection of Figures 14 and 15 indicate the lift and pitching-moment variations with Reynolds number are non-linear. Thus, it can be concluded that for a fixed angle of attack the lift and pitching-moment coefficients obtained at low Reynolds number cannot be extrapolated to full-scale values.

<u>16% Thick Supercritical Airfoil.</u> In Figure 20, a sketch of a Lockheed 16% thick supercritical airfoil is shown. This airfoil was designed for STOL wing application.

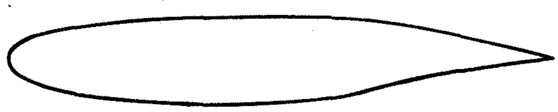

Figure 20. Lockheed 16% Thick Supercritical Airfoil

The scale effect on force and pressure data for the 16% thick supercritical airfoil is presented in Figure 21 at subcritical conditions. As can be seen, the viscous uncambering effects are substantial. As Reynolds number is increased, the section lift coefficient is increased approximately 27% over the range investigated.

At transonic speeds, several scale-effect phenomena are evident in Figure 22 for the 16% thick supercritical airfoil at conditions representative of cruise. As Reynolds number is increased, the shock moves rearward approximately 5% and the lift coefficient increases approximately 56%. At low Reynolds numbers, there is a greater tendency to form a lower surface shock near the airfoil crest than there is at higher Reynolds numbers.

Much of the sensitivity of this airfoil to scale effects is due to the severe adverse pressure gradients over the aft portion of the airfoil. Wind tunnel flow observations indicated that at low Reynolds numbers the boundary-layer is extremely thick and approaching separation. The closer the flow is to separation the more sensitive it is to scale effects.

<u>21% Thick Supercritical Airfoil.</u> A sketch of a Lockheed 21% thick supercritical airfoil design for a span-loader type aircraft (cargo-in-the-wing) is presented in Figure 23. This particular airfoil was designed with too high a subsonic leading-edge pressure peak (Figure 24) that resulted in too strong a shock wave at the transonic cruise conditions (Figure 25). Although not an optimum design, this airfoil demonstrates rather dramatically several types of scale effect phenomena.*

* An optimized 20% thick supercritical airfoil for span-loader application, developed by Lockheed, is reported in Reference 5.

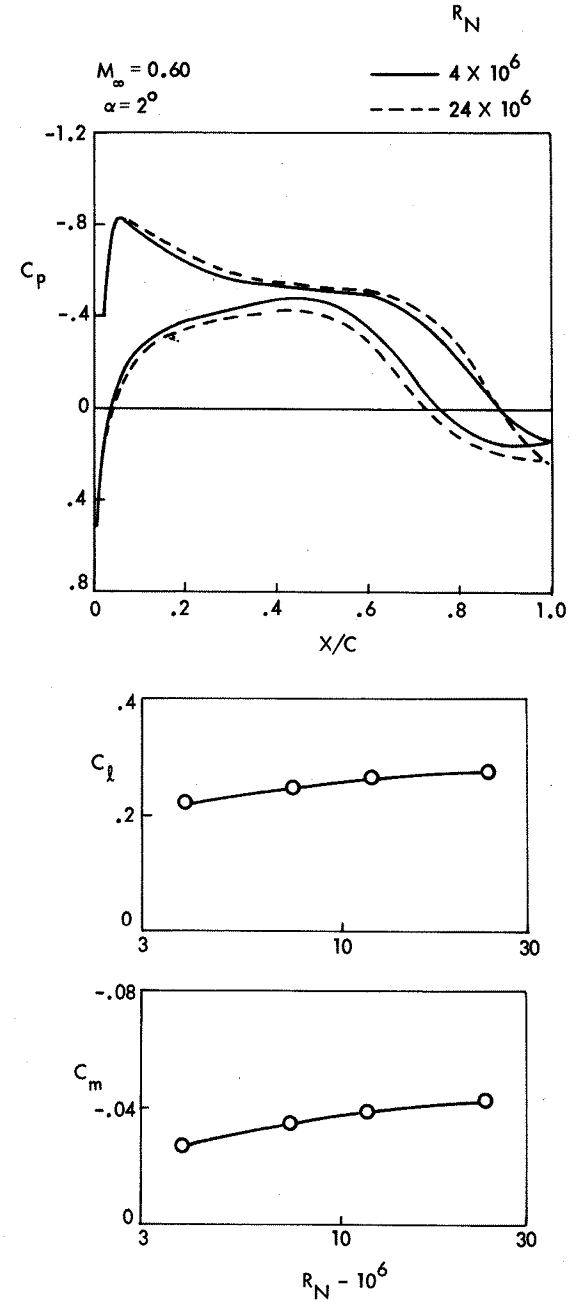

Figure 21. Scale Effect on Force and Pressure Data for Lockheed 16% Thick Supercritical Airfoil

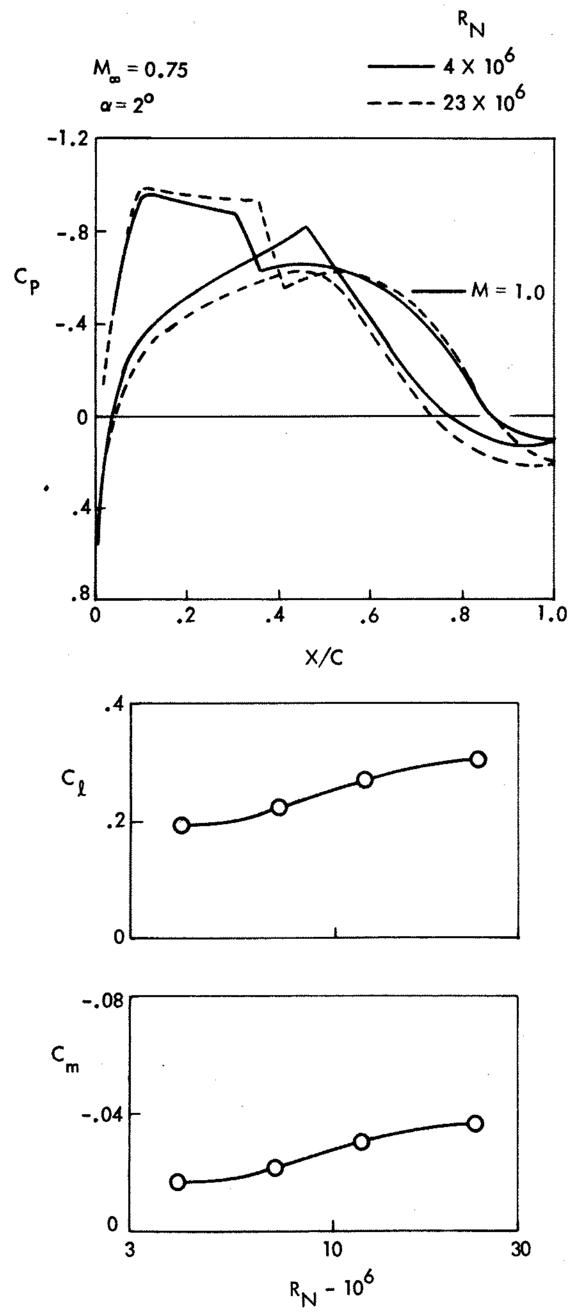

Figure 22. Scale Effect on Force and Pressure Data for Lockheed 16% Thick Supercritical Airfoil

In Figure 24, the sensitivity of pressure-gradient induced separation to scale effects (Figure 5) is shown. At high Reynolds numbers, the boundary layer flow through the aft upper-surface pressure gradients remains attached. However, at low Reynolds numbers, the upper-surface boundary-layer near the trailing-edge separates. The flow over the aft portion of this airfoil is particularly sensitive since the large airfoil thickness (21%) required a substantial adverse pressure gradient to recover the flow which in turn drove the boundary-layer toward separation.

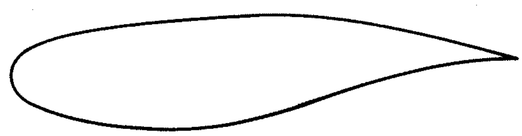

Figure 23. Lockheed 21% Thick Supercritical Airfoil

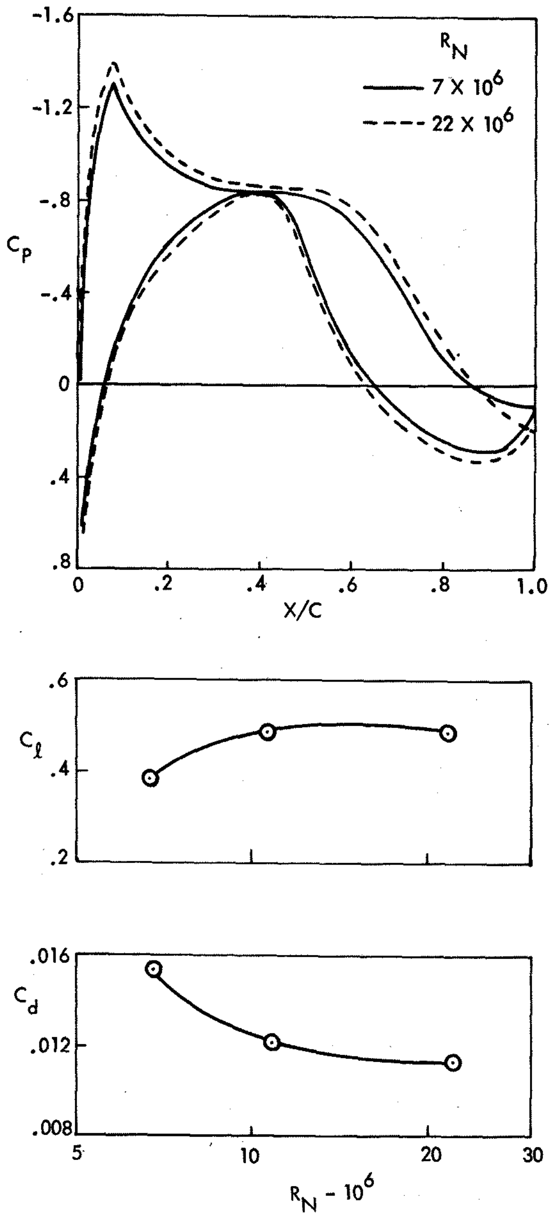

Figure 24. Scale Effect on Force and Pressure Data for Lockheed 21% Thick Supercritical Airfoil, $M_\infty = 0.60$, $\alpha = 3^o$

The scale effect on section lift and drag at subsonic speeds is shown in Figure 24. The change in lift coefficient with Reynolds number is similar to that shown for the other supercritical airfoils. Of special interest is the drag. Due to the low Reynolds number separation, it is obvious that the low Reynolds number subsonic data would not extrapolate correctly (Figure 4) to high Reynolds numbers in the manner shown in Figures 11 and 17. Thus, it can be stated that if separation is present at low Reynolds numbers, extrapolations of drag to high Reynolds numbers cannot be made with confidence.

Extrapolations of low Reynolds number drag to high Reynolds number values is also not possible for the 21% supercritical airfoil at transonic speeds (Figure 25). This is true not only because the separation characteristics change, but also because Reynolds number causes the shape of the pressure distribution to change. In particular, the shock strength changes with Reynolds number which negates any simple skin friction extrapolation. The overall effect of Reynolds number for a wide range of Mach numbers is shown in Figure 26. The effects of Reynolds number on separation (change in drag creep) and on wave drag can be noted.

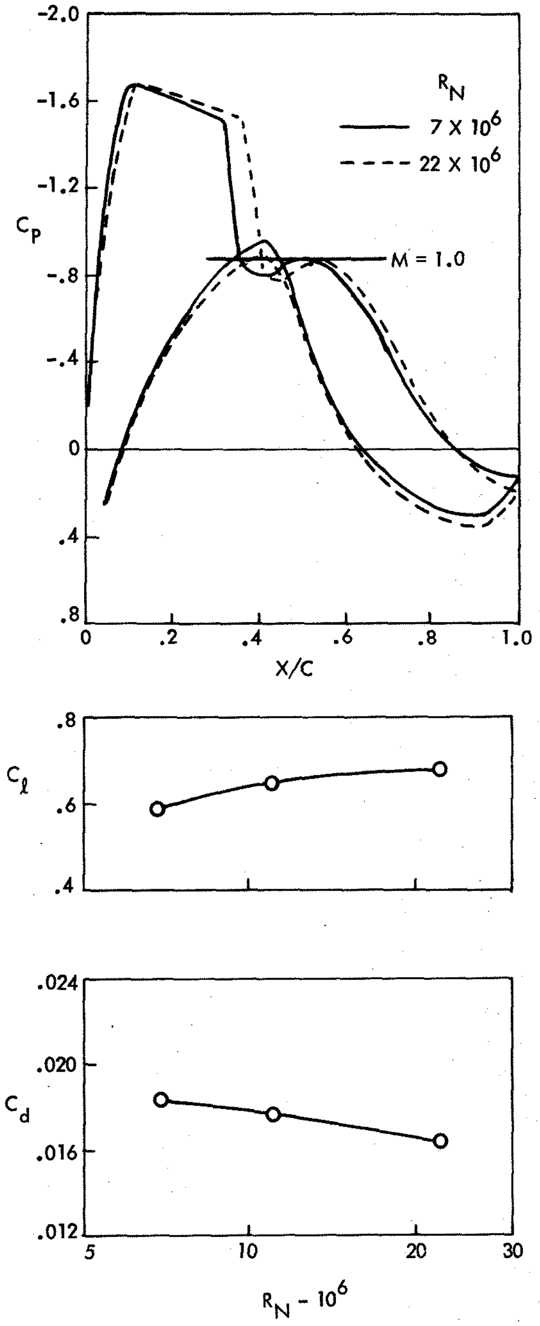

Figure 25. Scale Effect on Force and Pressure Data for Lockheed 21% Thick Supercritical Airfoil, $M_\infty = 0.68$, $\alpha = 4^o$

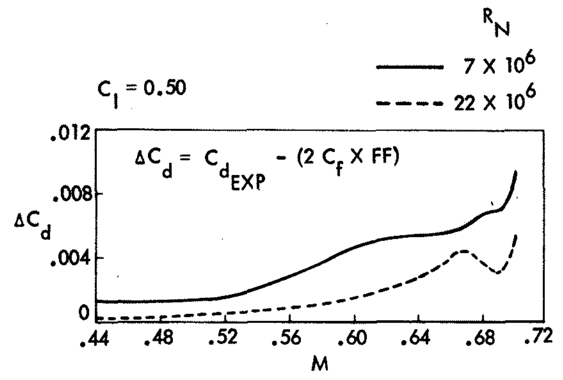

Figure 26. Scale Effect on Drag-Rise Characteristics of Lockheed 21% Thick Supercritical Airfoil

Status of Theoretical Methods for Scale Effect Analysis on Airfoils

To effectively evaluate scale effects on supercritical airfoils requires a theoretical program that is capable of accurately modeling:

- o Transonic flows
- o Boundary layer
- o Shock-wave/boundary-layer interactions

Unfortunately, a program that has these capabilities does not currently exist. However, progress is rapidly being made in these areas. In this section, the status of this progress on theoretical method development will be briefly reviewed.

To emphasize the importance of the viscous effects (which lead to scale effects) in airfoil analysis, inviscid and viscous transonic calculations were made for the Lockheed 21% thick supercritical airfoil (Figure 23) using the Bauer et al. program[6]. These results are shown in Figure 27 and 28. Inviscid calculations are compared to experimental data in Figure 27 for both constant angle of attack* and constant lift coefficient. As can be seen, the viscous effects are so dominant that neither solution is close to the experimental values. In Figure 28, the viscous calculations are presented. The correlation can be seen to be much improved over that in Figure 27. Several discrepancies still exist, however. Areas indicated in Figure 28 that still require research are: shock/boundary-layer interaction modeling, thick boundary layer modeling for flow approaching the trailing-edge region, and trailing-edge separation modeling.

The Bauer code generally represents the state-of-the-art in viscous analysis programs. This code is currently being extended by Melnick et. al.[7] to account for static pressure variations across the boundary layer in the trailing-edge region and to include a consistent treatment of airfoil wake effects. When this code becomes available, it will greatly assist in evaluating the thick boundary-layers experienced over the aft portions of the Lockheed 16% and 21% supercritical airfoils.

Research has also been underway by Deiwert[8] at NASA-Ames and others to compute transonic flow over airfoils with separation using the full Navier-Stokes equations. This work holds considerable promise for the future. The primary problems with this approach currently center around developing an adequate description of the flow turbulence and developing efficient algorithms that have reasonable run times.

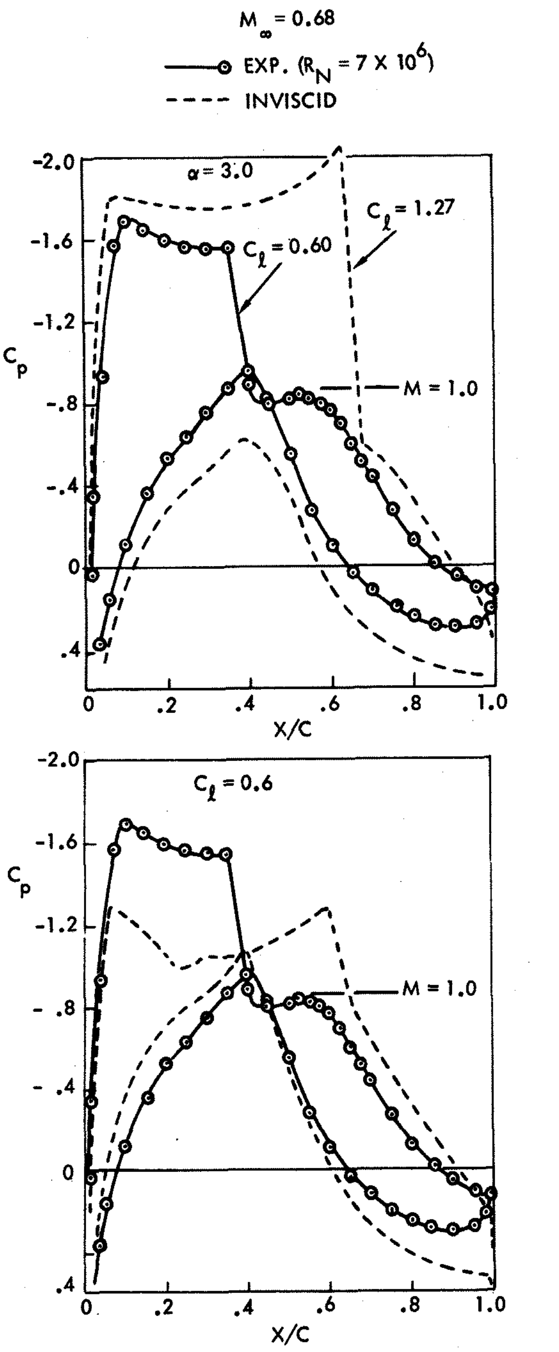

Figure 27. Viscous Effects on Lockheed 21% Thick Supercritical Airfoil

* Wind-tunnel angle of attack corrected for wind-tunnel wall interference using AGARD corrections.

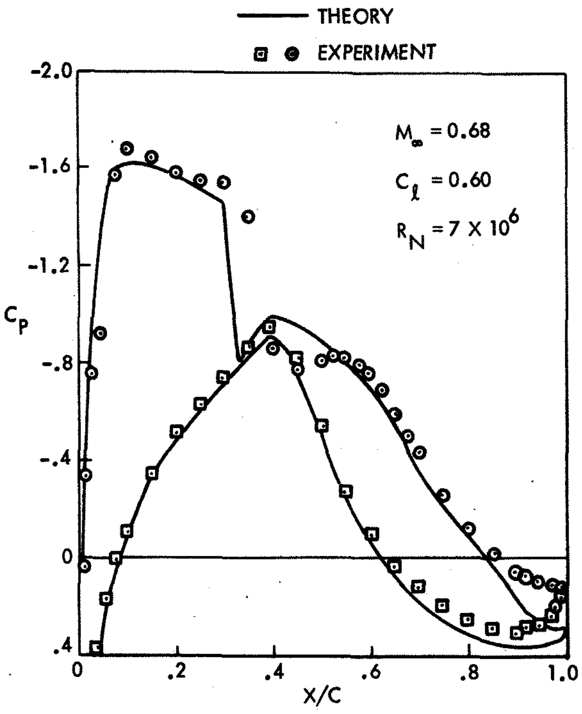

Figure 28. Correlation of Theory and Experiment for Lockheed 21% Thick Supercritical Airfoil

Status of Experimental Means for Evaluating Scale Effects on Airfoils

Although considerable confidence can be gained in the performance of a new airfoil design using available theoretical programs, current codes, as noted above, are as yet unable to reliably predict scale effects when complex flow phenomena are involved. For critical design situations, an aerodynamicist must then resort to experimental means to evaluate the magnitude of the scale effects on his airfoil. Currently, only a few production transonic wind tunnels exist with low and high Reynolds number capability that are available for airfoil investigations. Therefore, many designers have had to resort to other experimental means to simulate the proper scale effects on supercritical airfoils.

Over the last decade several means of simulating scale effects experimentally have been proposed. Those currently in use on supercritical airfoils are:

- Natural transition experiments in conventional wind-tunnels.
- Aft located transition strips in conventional wind-tunnels.

The objective of the "natural transition" approach is to approximate the high Reynolds number boundary-layer characteristics at low Reynolds numbers by allowing a longer extent of laminar boundary-layer before it transitions naturally to turbulent flow. This has the overall effect of thinning the low Reynolds number boundary-layer and more closely simulating the flight boundary-layer characteristics in regions sensitive to pressure-gradient induced separation (Figure 5) and in the shock/boundary-layer interaction region (Figure 6). Furthermore, the low Reynolds number viscous uncambering effects are reduced (Figures 2 and 3).

Unfortunately, there are several drawbacks to this approach. First, considerable chordwise variation in transition location is possible when the shape of the pressure distribution changes with Mach number or angle of attack. This introduces considerable problems in data analysis. Furthermore, since the transition location on the airfoil is not generally known for the test conditions, difficulties arise in correcting the low Reynolds number drag level (skin friction effects) to full-scale values. Another problem is the possibility of a laminar shock/boundary-layer interaction which is considerably different from a turbulent shock/boundary-layer interaction experienced in flight both in character and in sensitivity to scale effects.

The "aft transition location" scale effect simulation concept was proposed by Blackwell[2] to alleviate some of the problems associated with the natural transition approach previously discussed. The objective of this approach was to determine a fixed transition location on the airfoil at low Reynolds numbers that would result in boundary-layer conditions at the airfoil trailing-edge similar to that experienced in flight. This is known as the "trailing-edge criteria". The rearward movement of the transition location was constrained such that the boundary layer was tripped prior to the shock and hence a turbulent shock/boundary-layer interaction occurred. An example of this low RN and high RN trailing-edge boundary-layer matching procedure is illustrated in Figure 29. The method was developed for supercritical airfoils that generally experience trailing-edge separation due to "Class B" shock/boundary-layer interactions[1].

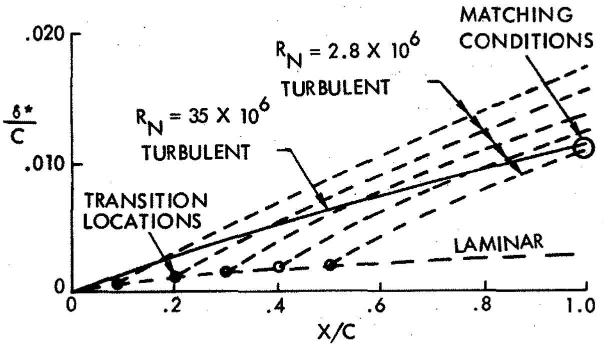

Figure 29. Scale Effects Simulation using Variable Location Artificial Transition

Use of this approach to simulate high Reynolds number data on a NACA 65_1-213 airfoil is shown in Figure 30. Very good correlation is obtained for the low Reynolds number data with transition rearward and the high Reynolds number data. In particular, the boundary layer profiles at the trailing-edge are identical.

For general use in simulating scale effects, the aft transition fixing approach has several

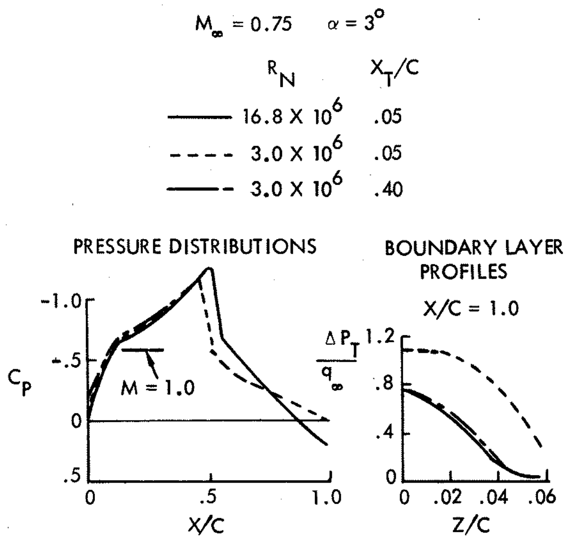

Figure 30. Transition Simulation of Scale Effects NACA 65_1-213 Airfoil

limitations. To be effective, the shape of the pressure distribution must be condusive to promoting laminar flow back to the transition strip. This is not always possible, in particular for supercritical airfoils at low Mach numbers or at high angles of attack where a pressure peak near the airfoil leading-edge causes boundary-layer transition. If natural transition occurs ahead of the fixed transition location, uncertainty in the data analysis is introduced. This is particularly true in the case of drag since the transition movement that occurs on the airfoil with increasing Mach number at a fixed angle of attack or lift coefficient gives the data an "appearance" of a lower drag creep.

The aft transition location approach still appears to be the best available procedure (as opposed to natural transition) to determine the sensitivity of an airfoil to scale effects when the aerodynamicist is forced to evaluate his design in a low Reynolds number wind tunnel. However, only selective tests should be conducted for critical conditions such as near drag-rise or when the boundary-layer is approaching separation and only then when there is assurance that the wing pressure distribution will promote laminar flow back to the transition strip.

Concluding Remarks

The preceding discussion has led to the following concluding remarks:

1. Reynolds number scale effects have a significant impact on supercritical airfoil performance.

2. The aerodynamic performance of supercritical airfoils is considerably more sensitive to scale effects than conventional airfoils.

3. Low Reynolds number drag data can be extrapolated to high Reynolds number conditions provided the flow is attached and the pressure distribution shape does not change appreciably as Reynolds number is increased. The latter requirement generally eliminates the extrapolation possibility for supercritical airfoil drag at transonic speeds. It does not appear that low Reynolds number lift and pitching-moment data can be extrapolated to flight conditions.

4. Viscous theoretical analysis methods currently under development will significantly improve the ability to account for scale effects in airfoil design.

5. Additional high Reynolds number airfoil test facilities are needed to evaluate scale effects. If high Reynolds number facilities are unavailable, extreme care should be taken when resort is made to boundary-layer manipulation in a low Reynolds number wind tunnel to simulate high Reynolds number conditions.

Acknowledgement

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Symbols

b	wing span, cm (in.)
c	chord, cm (in.)
c_f	skin-friction coefficient
c_d	section drag coefficient
c_ℓ	section lift coefficient
c_m	section pitching-moment coefficient
C_p	pressure coefficient
FF	airfoil form factor
M	local Mach number
M_∞	freestream Mach number
ΔP_T	incremental total pressure, N/m^2 (lb/ft^2)
q_∞	freestream dynamic pressure, N/m^2 (lb/ft^2)
RN	Reynolds number based on airfoil chord
X,Z	coordinate directions, cm (in.)
α	angle of attack, deg.
δ^*	boundary-layer displacement thickness, cm (in.)

Subscripts:

d	design
EXP	experimental
T	transition location

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