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WIND-TUNNEL–FLIGHT CORRELATION OF SHOCK-INDUCED SEPARATED FLOW

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WIND-TUNNEL—FLIGHT CORRELATION OF SHOCK-INDUCED SEPARATED FLOW[1]

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SUMMARY

A preliminary study is made of the discrepancies between wind-tunnel predictions and actual flight results for conditions of supercritical separated flow. The limited results obtained for two combinations of Mach number and lift, both involving supercritical-flow separation, suggest that the problem is related to Reynolds number and that an improvement in the correlation might be obtained by fixing the transition on a model so as to produce a relative boundary-layer thickness at the trailing edge comparable to that calculated to exist in flight. The need for continued research is indicated.

INTRODUCTION

The purpose of this discussion is to caution experimenters concerning the use of wind-tunnel results in predicting flight loads and moments when supercritical separated flow is present. Whenever separated flow has been observed on wind-tunnel models, the extrapolation of these results to flight conditions has always been subject to question. The discrepancies between aerodynamic results from flight and wind-tunnel investigations disclosed herein should not come as a surprise. They are merely additional evidence of the problem associated with separated flows.

Two combinations of Mach number and lift, both involving supercritical flow separation, are examined. One is for Mach numbers above cruise at lifting conditions near cruise, and the other is for Mach numbers near cruise at lifting conditions higher than cruise.

An example of the difficulty that might be encountered was observed during recent flight tests of a cargo-transport airplane. At supercritical Mach numbers the wing pressures and pitching moments of the airplane were considerably different from those predicted in wind-tunnel tests. No general procedure has been developed for resolving such discrepancies. Investigations are being conducted, however, to provide a better understanding of the factors involved, and the results herein are presented to report on the progress of these efforts.

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[1]Presented at the classified "Conference on Aircraft Aerodynamics," Langley Research Center, May 23-25, 1966, and published in NASA SP-124.

SYMBOLS

C_D	drag coefficient, $\text{Drag}/q_\infty S$
C_L	lift coefficient, $\text{Lift}/q_\infty S$
C_m	pitching-moment coefficient, $\text{Pitching moment}/q_\infty S\bar{c}$
C_p	local pressure coefficient, $(p_l - p_\infty)/q_\infty$
b	span of wing, meters
c	chord of wing, meters
$\bar{c}$	mean aerodynamic chord of wing, meters
M	free-stream Mach number
p_l	local static pressure, newtons/meter^2
p_∞	free-stream static pressure, newtons/meter^2
q_∞	free-stream dynamic pressure, newtons/meter^2
S	total area of wing, meters^2
x	longitudinal distance, measured from wing leading edge, meters
α_f	angle of attack of fuselage, degrees

DISCUSSION

An indication of the differences between wind-tunnel and flight data is shown by the pressure distributions in figures 1 and 2. In figure 1 is shown a comparison of the chordwise pressure distributions on the upper surface of a cargo-transport wing at a Mach number of 0.75, for a fuselage angle of attack of -0.6°, where the lift coefficients for the complete configurations are slightly less than 0.3 and the wing pressures are all subcritical. Transition was fixed near the leading edge of the wind-tunnel model by the method discussed in reference 1. The data are for the approximate 40-percent-semispan station. The chordwise trend of the pressures shows good agreement in shape between the wind-tunnel and flight results, although a small increase in the negative pressure-coefficient level is noted for the flight results.

In figure 2 is shown the same type of comparison at a higher-than-cruise Mach number of 0.85 for an angle of attack of approximately 0°. The lift coefficients associated with these wind-tunnel and flight conditions are 0.24 and

0.34, respectively. The flow over the wing for these conditions is supercritical. For example, a local supercritical Mach number of about 1.32 is associated with the flight peak pressure. As may be seen, the pressure distributions obtained in the wind tunnel and in flight are markedly dissimilar in shape. The adverse pressure gradients in this plot indicate that the location of the flight shock and attendant separation is rearward of that in the wind tunnel by about 20 percent chord. Associated with this shift in shock, of course, is a rearward shift of the center of pressure and therefore more negative pitching moments relative to values predicted from the wind-tunnel tests.

Because of the possible impact of this discrepancy on the satisfactory prediction of loads, stability, and performance of aircraft of this type, an investigation has been undertaken to resolve this difficulty.

A wind-tunnel investigation of several twist distributions indicated that differences in wing flexibility did not greatly contribute to the differences between wind-tunnel and flight results. It then was assumed that the problem might be associated with Reynolds number or scaling effects. Consideration of various factors suggested that the difficulty might be caused by differences in the boundary-layer conditions that affect shock-induced flow separation. At a given free-stream Mach number, the parameter that has the largest effect on shock-boundary-layer interaction is the boundary-layer thickness. For the particular problem of supercritical-flow separation the "relative thickness" was presumed to be a major factor. Relative thickness is defined as the ratio of the absolute thickness at any station to chord length. A study of these effects was thus considered a reasonable approach.

Figure 3 illustrates, in an exaggerated manner, the relative thickness effect. Since the relative thickness of the turbulent boundary layer varies inversely with a power (1/5) of the Reynolds number, the relative thickness at any given percent chord station would be greater on a small-scale wind-tunnel model with transition fixed near the leading edge than on a similar full-scale wing with natural transition in flight. When the local flow becomes sonic for both of these configurations, the displacement of the separated flow would tend to push the shock and accompanying separation farther forward on the wind-tunnel model.

It appeared, therefore, that the solution to the problem might be to develop a method that would provide a turbulent boundary layer near the trailing edge of the wind-tunnel model with the same relative thickness as would be encountered in actual flight.

With this objective as a goal, a transition strip was moved progressively rearward on a model during tests conducted in the Langley 8-foot transonic pressure tunnel. Theoretically, as the strip is moved downstream the extent of laminar flow ahead of the strip will increase, and the distance over which the turbulent layer can build up will be reduced. As a result, at any given position downstream of the strip, the turbulent layer will become thinner and tend to approach the relative thickness of the boundary layer in flight. It then was reasoned that when supercritical-flow conditions were reached on the two configurations, the shock position on the model would tend to approach the same position as on the airplane. The thinnest turbulent layer would be attained, of course, with the strip removed.

Figure 4 shows the effect on the chordwise supercritical-pressure distribution of moving the transition strip on the model. The test conditions and the wing are the same as those used to obtain the previously presented pressure data (fig. 2). However, for these results, the model was tested with the tail off, and the fuselage was somewhat different. The changes should not have any effect on the basic phenomena under discussion. As the strip was moved from 7.5 percent chord to 50 percent chord, the shock position moved rearward. Visual observations of the flow pattern, obtained by the fluorescent-oil film method (ref. 2), indicated that a number of isolated roughness particles present on the surface of the wing produced wedges of turbulent flow in the predominantly laminar flow ahead of the strip, so that the average location of transition for the strip at 50 percent chord was actually near 40 percent chord. When the strip was removed, visual observations of the flow patterns for this natural-transition condition indicated that the average location of transition was near 50 percent chord, which means that, along with turbulent wedges in the laminar boundary layer over the forward portion of the wing, some laminar flow extended behind the 50 percent chord; and the shock moved farther rearward to the downstream position shown in this figure. When the flight data points from figure 2 are compared with these natural transition model results, the shock positions appear to be, for all practical purposes, the same. For this particular natural transition location, calculations were made and indicated that the relative thickness of the boundary layer at the trailing edge of the model was the same as that of the full-scale airplane in flight.

These recent results appear to give evidence that the relative boundary-layer thickness at the trailing edge may be a primary parameter in determining the shock location and resultant pressure distribution. Additional experimentation is necessary, of course, to validate this tentative conclusion. The results thus far obtained, however, do indicate that the discrepancies between wind-tunnel and flight data are a relative boundary-layer thickness effect; that is, a scale effect.

The changes in aerodynamic forces that occurred as the transition strip was moved are presented in figure 5 for a near-cruise angle of attack of 2° and a Mach number of 0.85. Plotted in this figure as solid lines are the variations of lift, drag, and pitching-moment coefficients as a function of the transition-strip location. The short-dash lines indicate the level of the forces and moment with the transition strip removed. The difference between the lift and drag for the usual forward position of a transition strip and the values obtained with natural transition is indicative of an increase in lift-drag ratio of about 20 percent. Of even more importance for the same test conditions, the variation of pitching moment is representative of a rearward shift in the center of pressure of 11 percent.

The results of this wind-tunnel investigation on a high-aspect-ratio subsonic wing at above-cruise Mach numbers, near cruise lift, provide evidence that the discrepancy between wind-tunnel and flight pressure and force data apparently results from a relative boundary-layer-thickness effect on supercritical-flow separation. It would be expected that the same phenomena also would exist near the cruise Mach number, but at higher-than-cruise lift, since shock-induced separation also occurs for these conditions. In figure 6 are plotted the wind-tunnel pitching-moment coefficients as a function of lift for the same model just discussed with two extreme boundary-layer test conditions at a Mach number of 0.75. For the configuration with transition fixed near the leading edge ($x/c = 0.075$) a reduction in stability occurs at lift coefficients slightly above cruise. When the strip is removed, not only are the pitching-moment coefficients more negative, but the trend toward instability is delayed to a higher lift coefficient. An examination of the wind-tunnel pressure data (which are not presented) indicated that this difference is associated with the same separation phenomena just described for the subsonic wing operating beyond its cruise Mach number; with the transition strip removed, shock-induced separation occurred farther rearward along the chord. As was indicated in the previous discussion, it is probable that the natural-transition configuration more nearly simulates flight conditions than the fixed-transition configuration. Available flight data do not go up to the point of divergence, so they have not been included in the figure.

CONCLUDING REMARKS

Because, at supercritical speeds, pressure distributions obtained from model and full-scale flight tests may be different, a study has been made for the purpose of improving this correlation.

On the basis of this study, a reasonable assumption appears to be that the problem is one of a Reynolds number effect on shock-induced boundary-layer separation. This effect appears associated with differences between the relative thickness of the boundary layer on models and full-scale airplanes.

At the present time no conclusive means are established for exactly simulating the supercritical-flow phenomena on models as they exist in flight. On the basis of present knowledge, however, it does appear that full-scale characteristics may be obtained, at least, on subsonic wings by locating transition on a model so as to produce the same relative boundary-layer thickness at the trailing edge as has been calculated to exist in flight.

Until this or other methods can be more definitely established, it is suggested as an interim recommendation that wind-tunnel studies be made with transition occurring at various locations. In this manner, at least, the sensitivity of shock-induced separation to modification of the boundary-layer conditions can be determined.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., May 23, 1966,
126-13-03-22-23.

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2. Loving, Donald L.; and Katzoff, S.: The Fluorescent-Oil Film Method and Other Techniques for Boundary-Layer Flow Visualization. NASA MEMO 3-17-59L, 1959.

SUBCRITICAL PRESSURE DISTRIBUTION

$M = 0.75;\ \alpha_f = -0.6°$

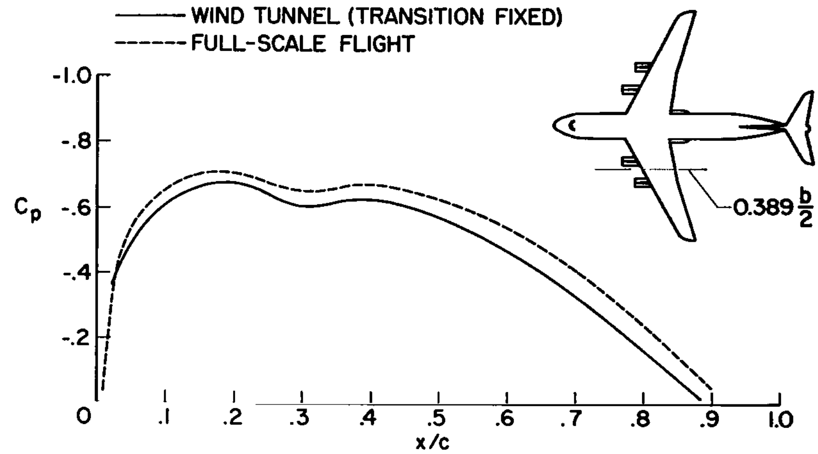

Figure 1

SUPERCRITICAL PRESSURE DISTRIBUTION

$M = 0.85;\ \alpha_f \approx 0°$

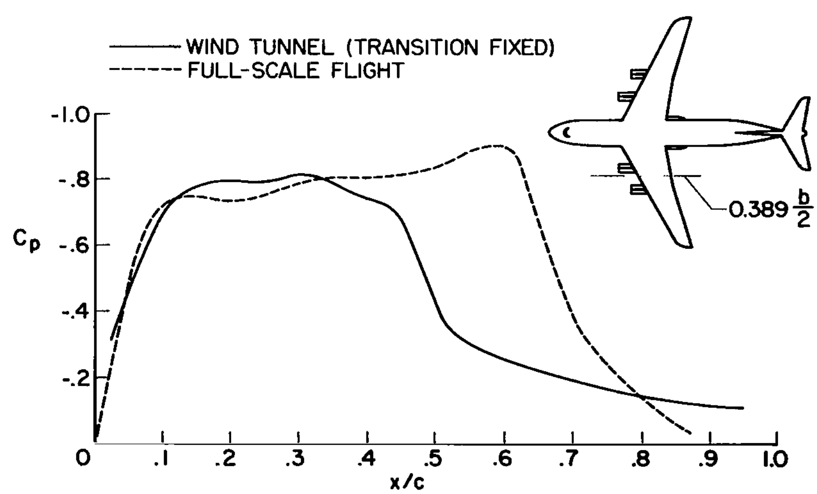

Figure 2

EFFECT OF BOUNDARY LAYER ON SHOCK-INDUCED SEPARATION

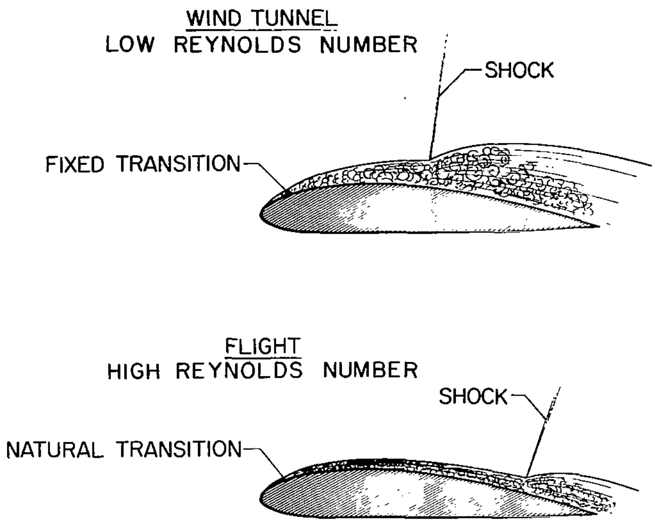

Figure 3

EFFECT OF TRANSITION LOCATION ON PRESSURE DISTRIBUTION

M = 0.85; $\alpha_f = 0°$

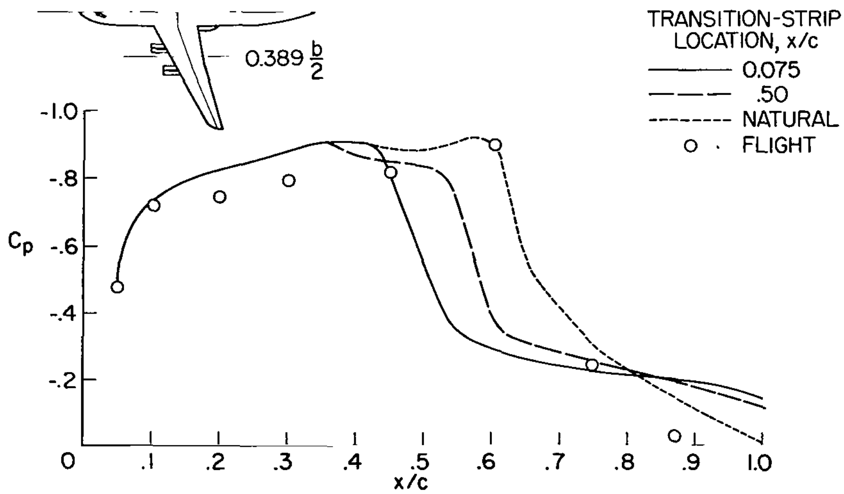

Figure 4

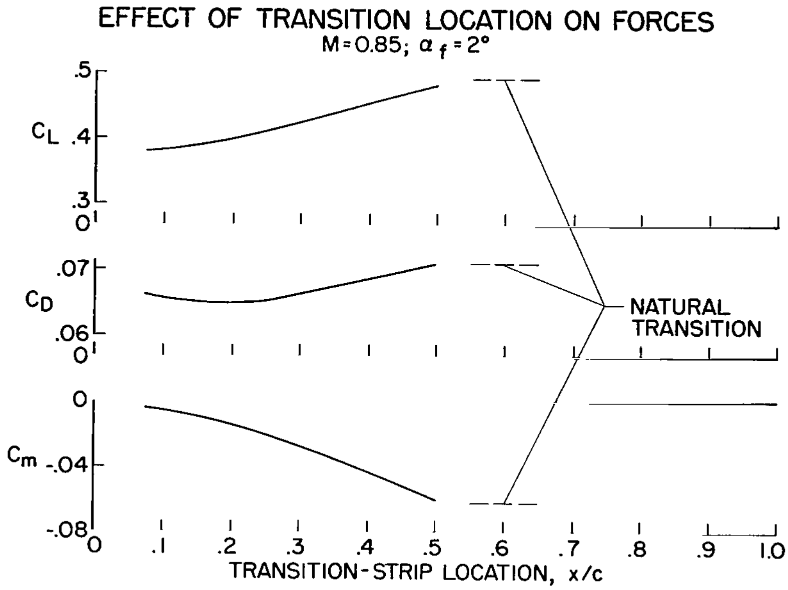

Figure 5

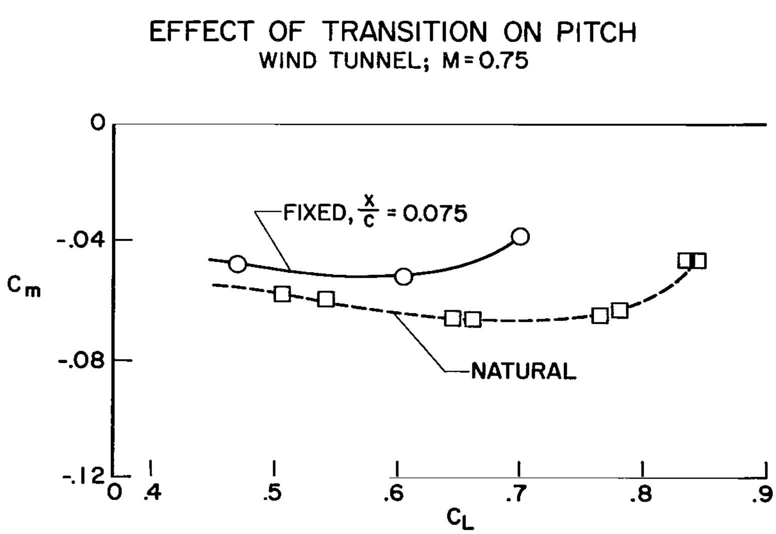

Figure 6

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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