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RESULTS OF LFC EXPERIMENT ON SLOTTED SWEPT SUPERCRITICAL AIRFOIL IN LANGLEY'S 8-FOOT TRANSONIC PRESSURE TUNNEL

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LAMINAR-FLOW CONTROL (LFC) EXPERIMENT OBJECTIVE

A large chord swept supercritical laminar-flow control (LFC) airfoil has been designed, constructed, and tested in the NASA Langley 8-Foot Transonic Pressure Tunnel (8-Ft. TPT). The LFC airfoil experiment was established to provide basic information concerning the design and compatibility of high-performance supercritical airfoils with suction boundary-layer control achieved through discrete fine slots or porous surface concepts. It was aimed at validating prediction techniques and establishing a technology base for future transport designs and drag reduction. Good agreement was obtained between measured and theoretically designed shockless pressure distributions. Suction laminarization was maintained over an extensive supersonic zone up to high Reynolds numbers before transition gradually moved forward. Full-chord laminar flow was maintained on the upper and lower surfaces of the slotted suction wing at the design Mach number of 0.82 up to a chord Reynolds number of 12×10^6. When accounting for both the suction and wake drag, the total drag could be reduced by at least one-half of that for an equivalent turbulent airfoil. Specific objectives for the LFC experiment are given in figure 1.

Conduct basic aerodynamic and fluid dynamics research program on a high-performance, swept supercritical, LFC airfoil to determine:

- Ability to laminarize over extensive supercritical region
- Ability of stability theories to predict transition and suction laminarization requirements
- Relative merit of slotted and perforated suction surfaces for LFC and HLFC
- Effects of surface conditions and boundary layer influences on laminarization

Figure 1

LFC EXPERIMENT DESIGN CONSIDERATIONS

The Langley LFC airfoil experiment was established to provide basic information concerning the design and compatibility of high-performance, swept supercritical airfoils with boundary-layer control. Several methods of laminarization were considered and more than one will be employed during the experiment. The integration of laminar-flow control with swept supercritical wing technology required advanced materials and fabrication processes to construct a suitable airfoil. The tests required modifications to the wind tunnel to provide low levels of free-stream turbulence and acoustic disturbances to ensure that the background disturbance environment would not severely affect the maintenance of laminar flow. In addition, the necessity for a very large chord airfoil posed a problem that required extensive analysis and modification to the 8-Ft. TPT test section to provide a near disturbance-free representation of the flight environment. The test section walls were contoured with a foam liner to conform to the theoretical streamline shape (including wing and wall boundary-layer effects) of the flow about the wing at the design condition. The tunnel stagnation chamber was fitted with suitable honeycomb and screens to reduce turbulence at transonic speeds, and the test section downstream of the model was fitted with an adjustable sonic throat to block diffuser noise (figure 2).

- **Design advanced high-performance, large chord, swept supercritical LFC airfoil**
- **Design and modify 8-FT TPT test section for simulation of free air about infinite-span yawed wing at transonic speeds**
    - **Contoured walls**
- **Design and modify 8-FT TPT for flow quality improvement at transonic speeds**
    - **Honeycomb, screens, sonic choke**

Figure 2

TEST SETUP FOR LFC EXPERIMENT IN THE 8-FT. TPT

A schematic of the overall LFC experiment in the Langley 8-Ft. TPT is shown in figure 3 along with tunnel modifications. The major component was a large chord, 23° swept supercritical LFC airfoil of aspect ratio near one, which spanned the full tunnel height. Laminar-flow control by boundary-layer removal was achieved by suction through closely spaced fine slots extending spanwise on the airfoil surface. After passing through the slots, the air passed through metering holes located in plenums beneath each slot and was collected by spanwise ducts with nozzles located at the ends. From the duct/nozzles, the air passed through airflow system evacuation lines, through airflow control boxes which controlled the amount of suction to each individual duct nozzle, and through sonic nozzles to a 10,000 ft^3/min compressor which supplied the suction. All four walls of the tunnel were contoured in order to produce a transonic wind tunnel flow which simulated unbounded free-air flow about an infinite yawed wing at model design conditions. The contoured liner was shaped to conform to computed streamlines around the wing and corrected for growth of the wall boundary layer. The success of the LFC experiment depended to a large extent on the environmental disturbance levels in the test section. Isolation of the test section from downstream disturbances was achieved by an adjustable two-wall choke (sonic throat). Reduction of upstream disturbances such as pressure and vorticity fluctuations was achieved by the installation of a honeycomb and five screens in the settling chamber.

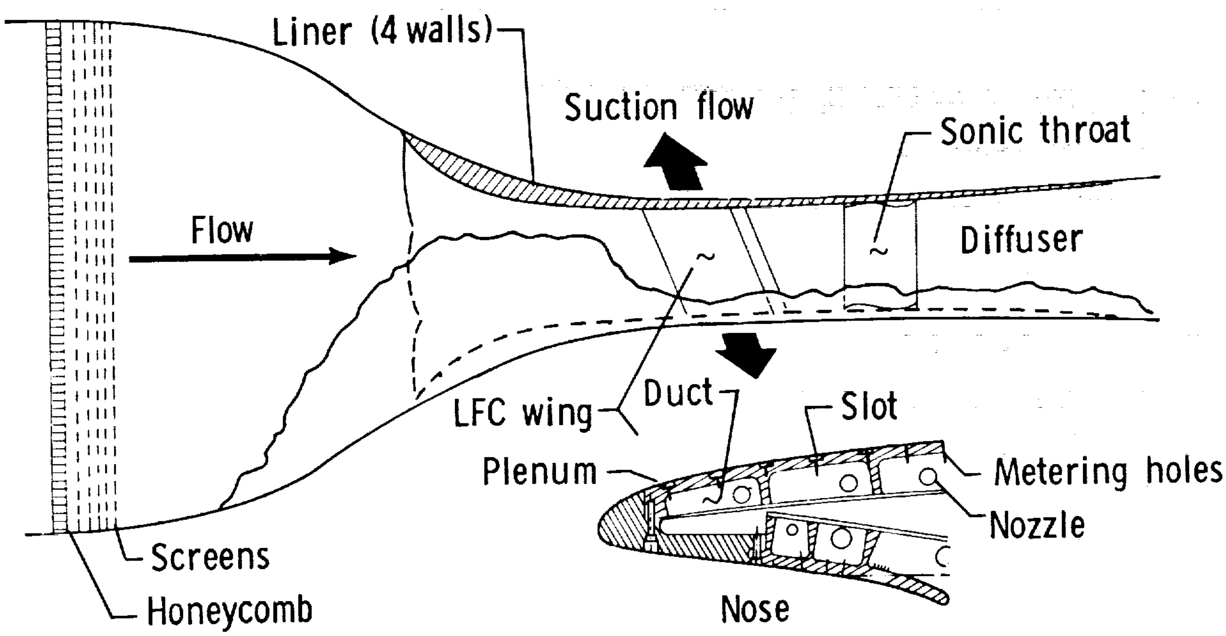

Figure 3

Figure 4 is a downstream view of the upper surface of the model taken immediately upstream of the model. This figure illustrates the smooth streamline contour of the liner and how it blended with the model. The dark areas at the top and bottom liner model juncture regions are suction panels in the "collar" around the ends of the model to control the growth of the boundary layer in these regions. The dark area on the left vertical wall and downstream of the model is one of the flexible two-wall chokes (sonic throat). The choke plate on the opposite wall is hidden behind the model. The dark area immediately in back of the model is the tunnel test section access door followed by the downstream high-speed diffuser.

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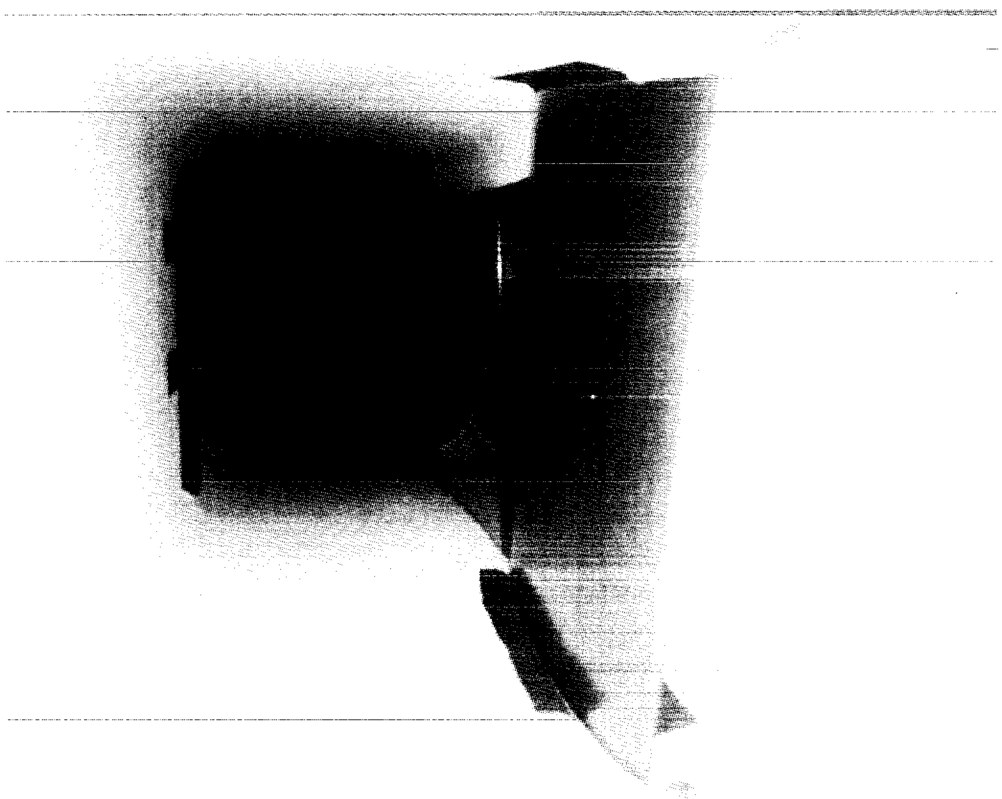

Figure 4

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MODEL INSTALLED IN TUNNEL - UPSTREAM VIEW

Figure 5 is an upstream view of the finished liner and wing trailing edge as seen from the test section diffuser entrance where the liner faired into the original tunnel lines. The LFC model extended from floor to ceiling and blended with the liner. The offset of the wing mean plane from the tunnel centerline may be seen as well as the development of the liner floor and ceiling step which resulted from the differential spanwise flow displacement in the tunnel channels "above" and "below" the wing surfaces. The dark vertical area on the left of the photograph and downstream of model trailing edge is the edge of the test section access door. The dark rectangular area ahead of the model is the tunnel contraction region.

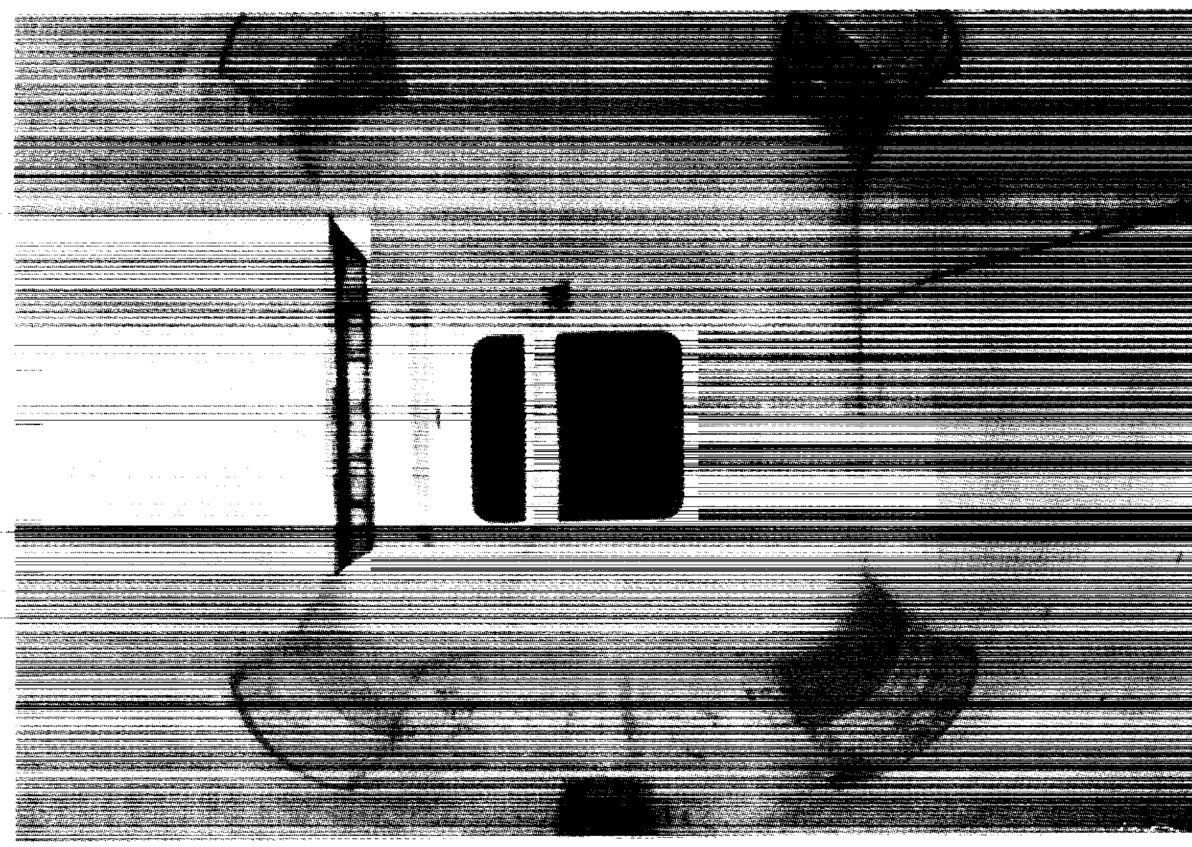

Figure 5

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MEASURED AND DESIGN PRESSURE DISTRIBUTIONS

Measured and design chordwise pressure distributions on the upper and lower surfaces of the LFC model are shown in figure 6 for two chord Reynolds numbers at the design Mach number of 0.82. In general, these representative results indicate measured pressure distributions very close to design. Shockfree flow is shown for a 10 million Reynolds number and nearly shockfree flow for 20 million. The slightly overall higher velocities on the upper surface and the chordwise deviation from the design pressure distribution were attributed to classical problems associated with wind tunnel testing, wall interference, and model deformation under design air loads. The velocity field between the upper surface and tunnel wall (supersonic bubble zone) was slightly higher than predicted due to the liner contour and inability to completely account for boundary-layer displacement effects in the design analysis. The irregularity in the upper surface pressure distributions appears to be associated with local surface contour deviations, caused by deformation of the model under load, and perhaps amplified by the proximity of the contoured wall. As chord Reynolds number increased above 10 million, transition moved forward rapidly on the lower surface and the flow could no longer sustain the adverse pressure gradient leading into the trailing-edge cusp, and separation occurred at about 80-percent chord. This separated flow changed the local effective area distribution of the test section resulting in a slightly higher free-stream Mach number and increased upper surface shock strength at a chord Reynolds number of 20 million.

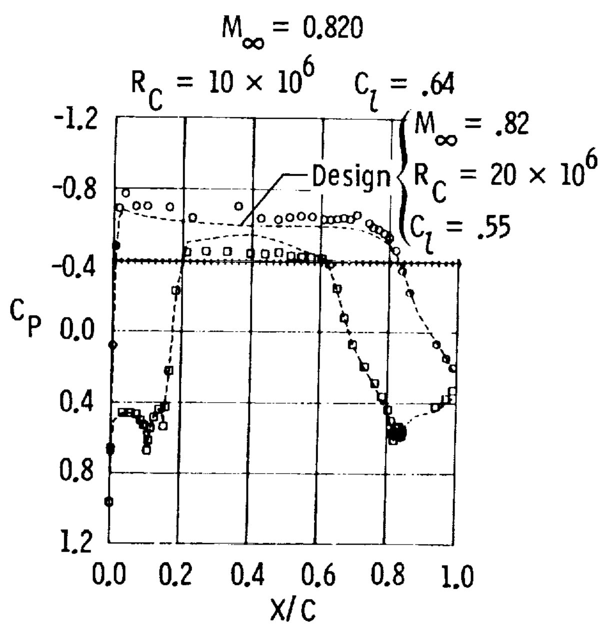

Figure 6

MEASURED AND THEORETICAL SUCTION DISTRIBUTIONS

The measured chordwise suction coefficient (C_Q) distribution required to maintain full-chord laminar flow over both surfaces at 0.82 Mach number and 10-million chord Reynolds number is shown in figure 7 compared to the theoretical suction distribution. The required suction level was higher than the theory over most of the upper and lower surfaces. Only about one-third of the total suction requirement is attributable to the upper surface and the remaining two-thirds to the lower surface. The higher suction requirements were due to the upper surface overvelocities and the surface pressure irregularities, as well as to higher suction required to overcome the cross-flow instabilities associated with the steep pressure gradients on the upper and lower surfaces and the minimization of centrifugal Taylor-Görtler-type boundary-layer instabilities and interactions in the concave regions of the lower surface. The overall higher suction levels are also influenced by tunnel disturbance levels which are inherently higher than free-air turbulence levels expected in flight.

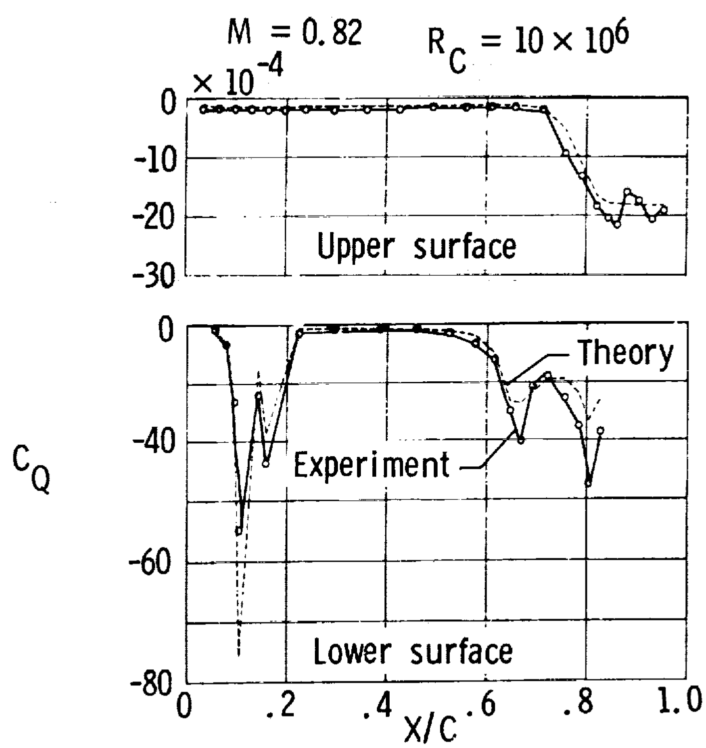

Figure 7

EFFECT OF REYNOLD'S NUMBER ON UPPER-SURFACE TRANSITION PATTERN

A planform view of the laminar region of the LFC wing upper surface is shown in figure 8 with flush-mounted surface thin-film sensors for transition detection indicated. The thin-film sensor oscillograph output trace for fully laminar flow is essentially flat, while that for fully turbulent flow resembles "white noise" and is easily distinguishable. Intermediate cases are assigned a measure based on the percentage of time in a local sample that is taken up with turbulent bursts. On these diagrams, lines have been drawn to indicate the estimated boundaries of the laminar, transitional, and turbulent regions. For a Reynolds number of 10 million (at design Mach number), the flow is laminar to the trailing edge over most of the span. As the Reynolds number increases to 20 million, the transition zone sweeps forward. The bottom end of the wing has more upstream transition at all Reynolds numbers, and this is thought to be an endplate effect.

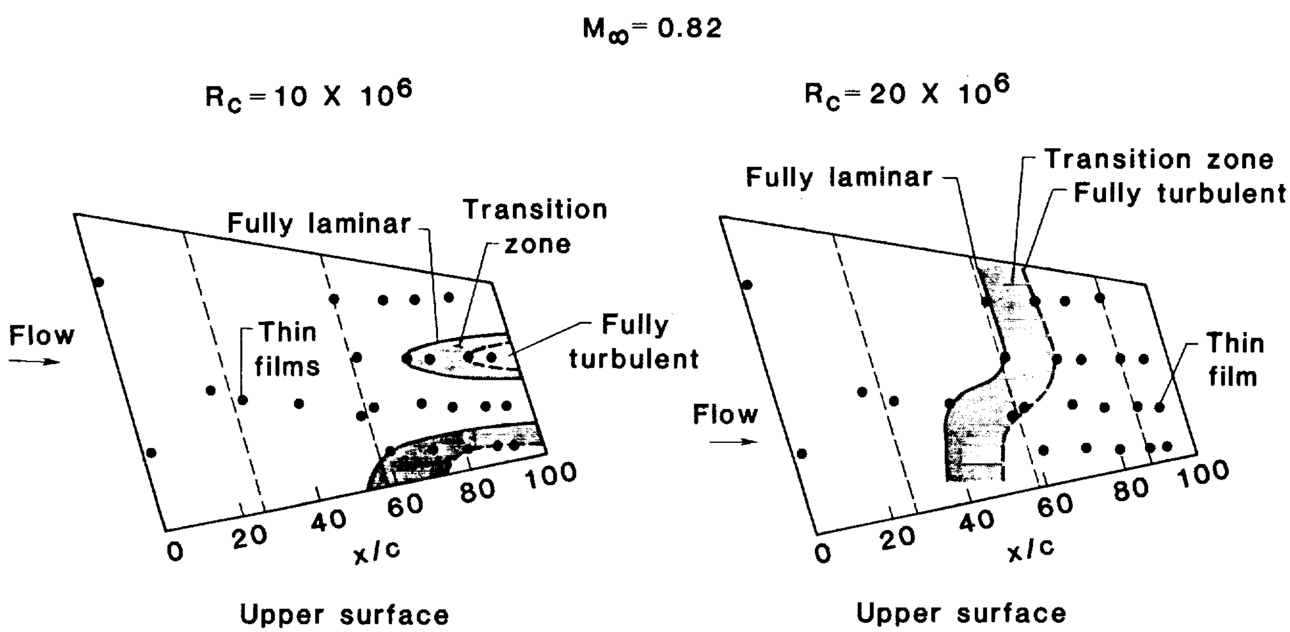

Figure 8

TRANSITION VARIATION WITH REYNOLDS NUMBER

The data presented in figure 9 show the chordwise extent of laminar flow achieved on the upper surface for several Mach numbers up to the design Mach number of 0.82, as determined by a grid of flush-mounted surface thin film gages (fig. 8). At R_C = 10 million, full-chord laminar flow could be maintained over the upper and lower surfaces for all Mach numbers. As Reynolds number was increased for constant Mach number, transition moved gradually forward on the upper surface. The Reynolds number at which this forward movement began was dependent on Mach number and occurred at progressively lower Reynolds numbers as Mach number increased. For the design Mach number of 0.82, the forward movement began between 11 and 12 million and reached about 65-percent chord at R_C = 20 million. Transition on the lower surface moved more rapidly than on the upper surface and occurred near the leading edge for M = 0.82 and R_C = 20 million. It was concluded that suction laminarization over a large supersonic zone is feasible to high-chord Reynolds numbers even under non-ideal surface conditions on a swept LFC airfoil at high lift.

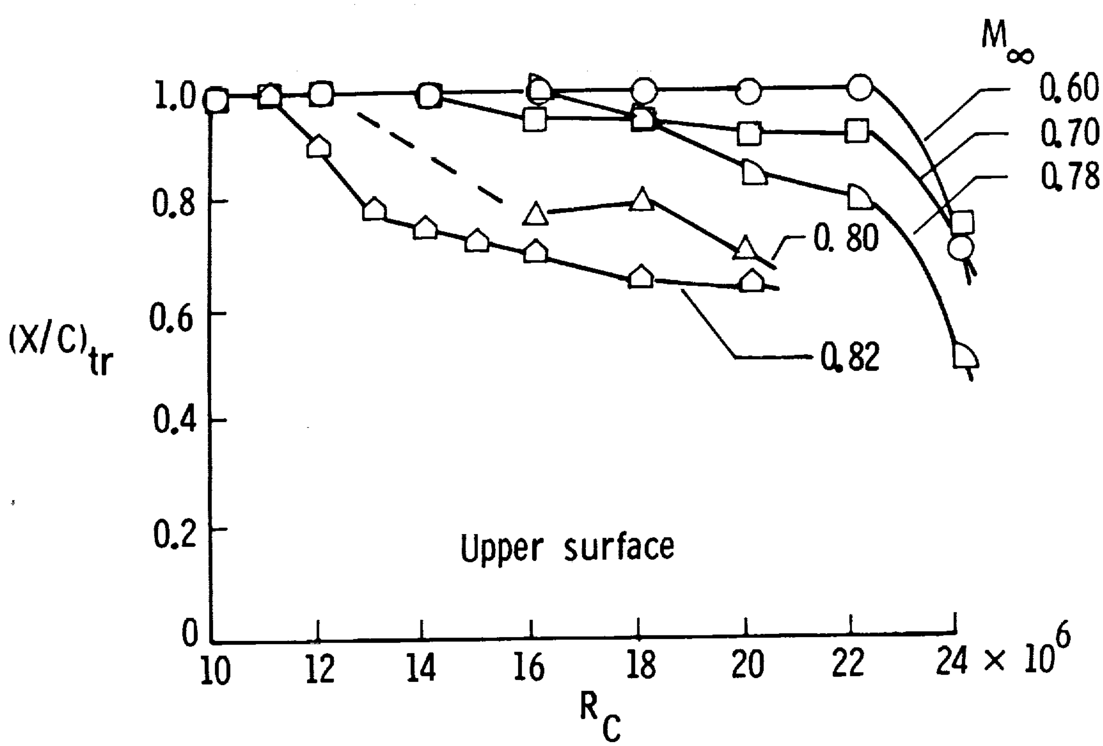

Figure 9

UPPER- AND LOWER-SURFACE TRANSITION VARIATION WITH R_C

At a chord Reynolds number of 10 million, the airfoil is fully laminar on both upper and lower surfaces (fig. 10). For chord Reynolds numbers above 11 million, transition moves forward on both the upper and lower surfaces. Between 11 and 13 million, the upper surface transition moves just upstream of 80 percent, and then more gradually to about 65 percent at 20 million. On the lower surface, the transition line moves from the trailing edge at a Reynolds number of 10 million to about 75 percent at 13 million. At a Reynolds number of 15 million, the lower-surface transition has moved to 30-percent chord (just beyond the end of the favorable pressure gradient) and the flow in the aft cusp is separated.

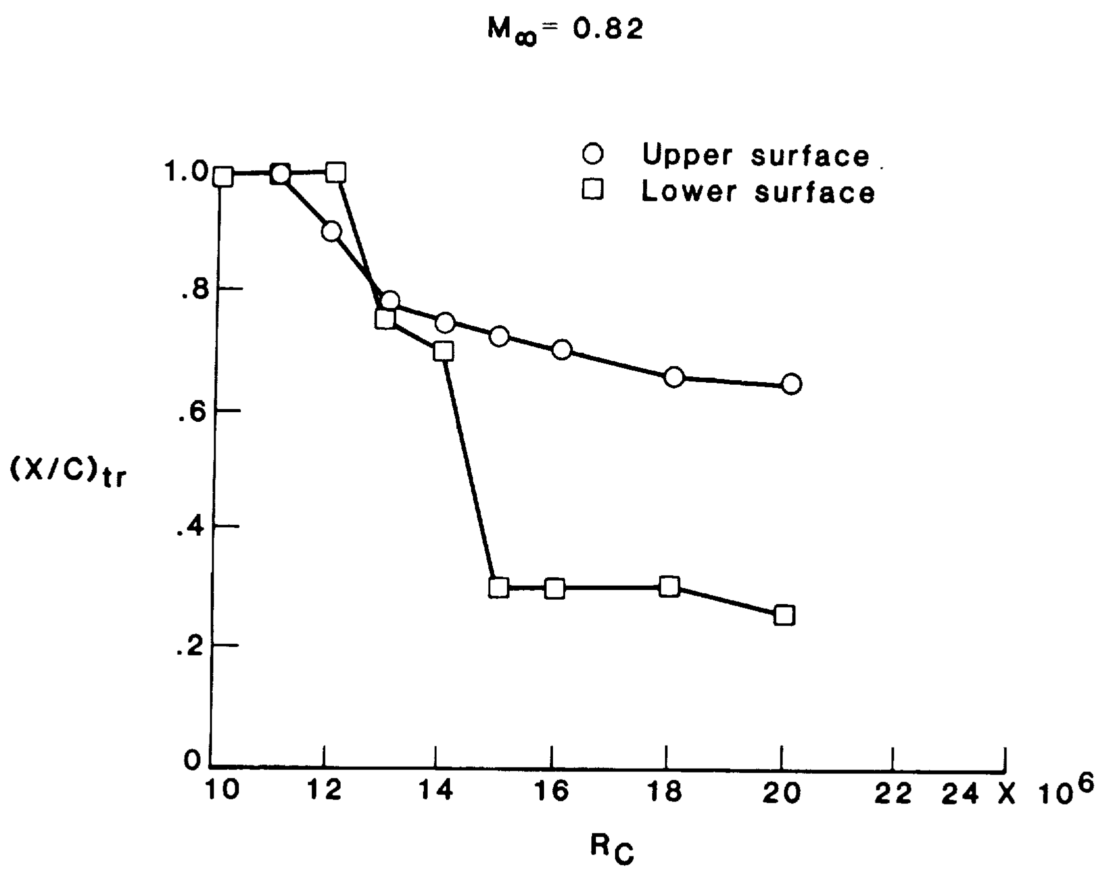

Figure 10

EFFECT OF REYNOLDS NUMBER ON DRAG AT DESIGN MACH NUMBER

The contributions to total section drag coefficient are shown in figure 11 for the Mach number of 0.82 over the chord Reynolds number range of 10 to 20 million. The assignment of suction drag contributions to the upper or lower surface is accountable since the suction drag is computed duct-by-duct. The wake drag is separated into upper and lower surface components on the basis of the assumption that the wake can be divided between the upper- and lower-surface at the point on the wake rake where the stagnation pressure loss is the greatest. The data indicate that the larger contribution to the total drag is from the lower surface. The sharp rise in wake drag on the lower surface between 14 and 15 million Reynolds number is associated with rapid forward movement of transition (fig. 10) and separation of the boundary layer in the pressure recovery region of the lower aft cusp.

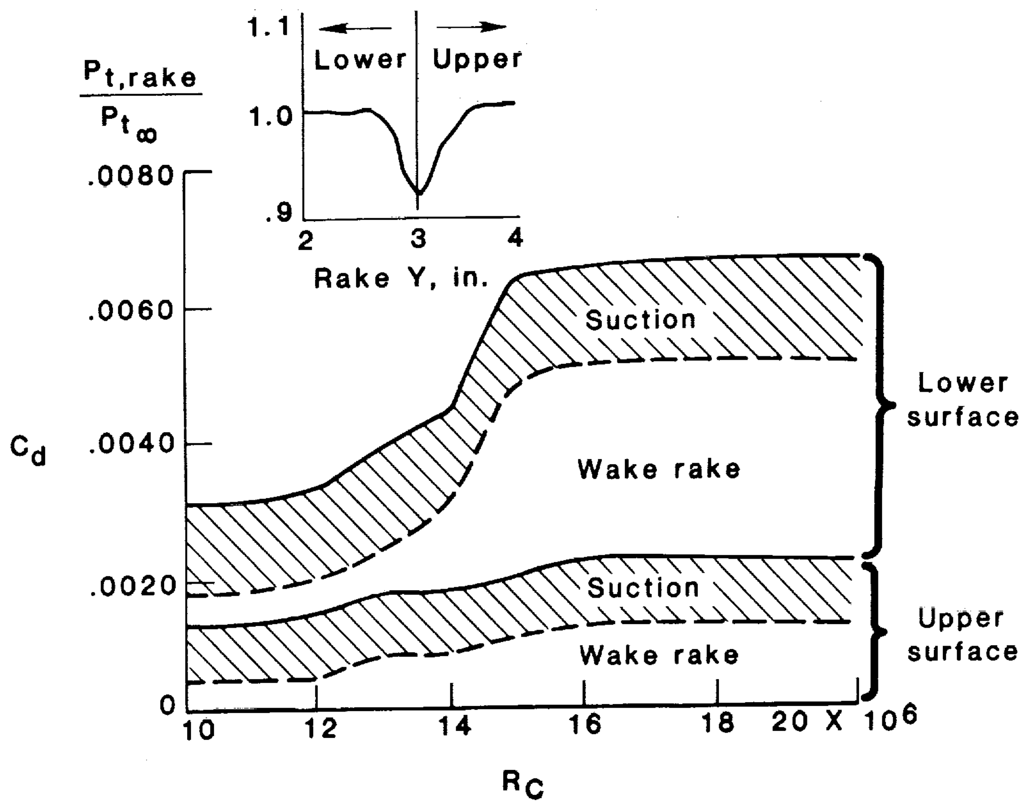

Figure 11

MEASURED TOTAL DRAG ON AIRFOILS WITH/WITHOUT SUCTION CONTROL

A comparison of the measured drag of the present LFC supercritical airfoil (identified as SCLFC(1)-0513F) with an equivalent turbulent supercritical airfoil is shown in figure 12. The total drag of the swept supercritical LFC airfoil with suction slots (solid symbols) includes the suction drag penalty required to maintain full-chord laminar flow and thus represents drag levels obtained with the maximum extent of laminar flow at the design lift conditions. The open symbols indicate drag levels obtained with an unswept supercritical airfoil with fully turbulent (tripped) attached flow in the same Langley 8-Ft. TPT. In general, the results indicate that about 60-percent drag reduction may be achieved with boundary-layer control over the speed range when compared with a turbulent drag level of about 80 counts.

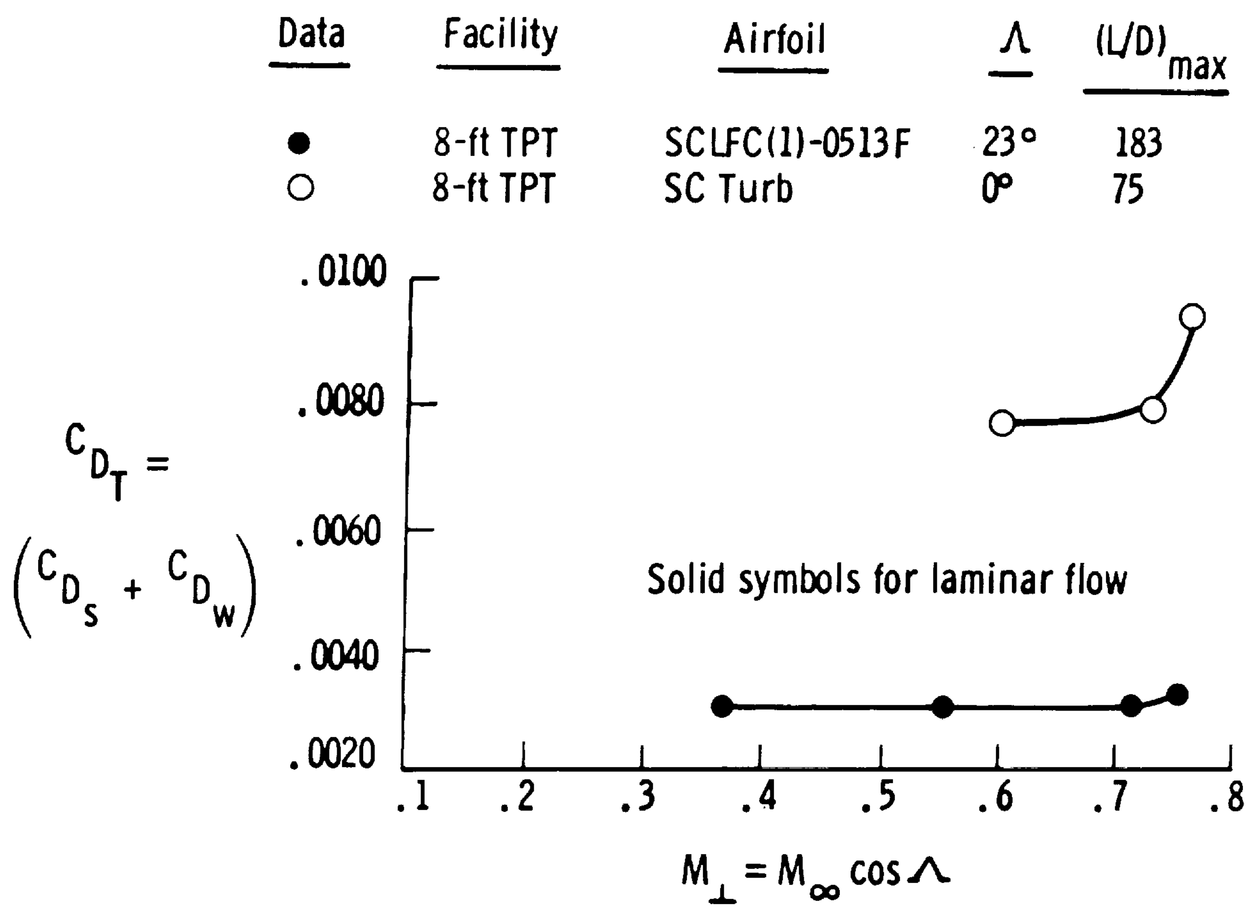

Figure 12

PRESSURE DISTRIBUTIONS ON SWEPT LAMINAR SUCTION MODELS

Figure 13 shows the upper-surface pressure distributions at the design lift condition ($C_L \approx .3$) on two earlier low-speed LFC airfoils of different sweep as compared to the supercritical LFC design pressure distribution. These low-speed LFC designs, based on standard NACA airfoil profiles, have favorable pressure gradients over the first 50 percent or more of the chord, and much less severe adverse pressure gradients aft than the supercritical design.

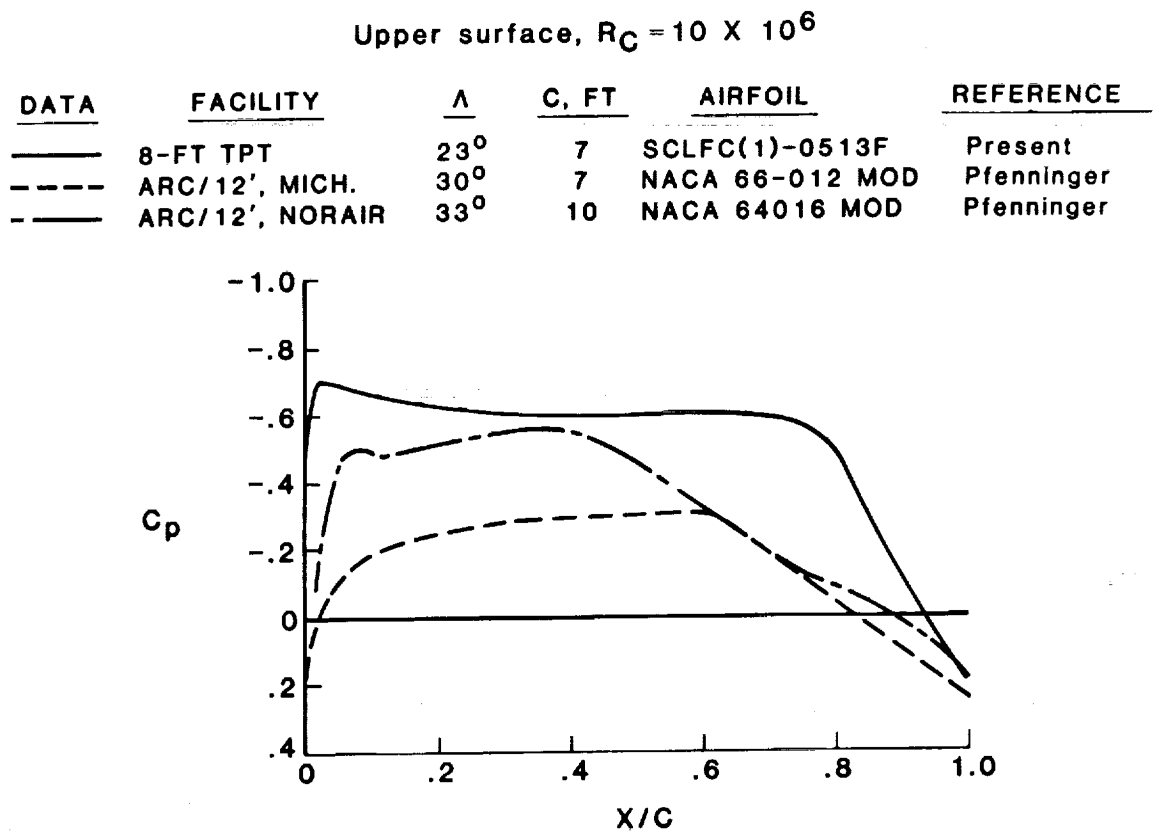

Figure 13

MEASURED MINIMUM PROFILE DRAG ON SWEPT LAMINAR SUCTION MODELS

Figure 14 shows only the upper-surface measured minimum profile drag coefficients corresponding to the upper-surface pressure distributions of figure 13. The breakdown shows that, as might be expected, the supercritical design has a somewhat larger suction drag penalty than the NORAIR model, which has the greatest extent of favorable pressure gradient (fig. 13), and only a slightly larger suction drag penalty than the University of Michigan 5- by 7-Foot Tunnel model while the wake drag contribution is about the same in all cases.

Upper surface, $R_c = 10 \times 10^6$

DATA	FACILITY	Λ	C, FT	AIRFOIL	REFERENCE
□ ○	8-FT TPT	23°	7	SCLFC(1)-0513F	Present
⬒ ◓	ARC/12', MICH.	30°	7	NACA 66-012 MOD	Pfenninger
◧ ◐	ARC/12', NORAIR	33°	10	NACA 64016 MOD	Pfenninger

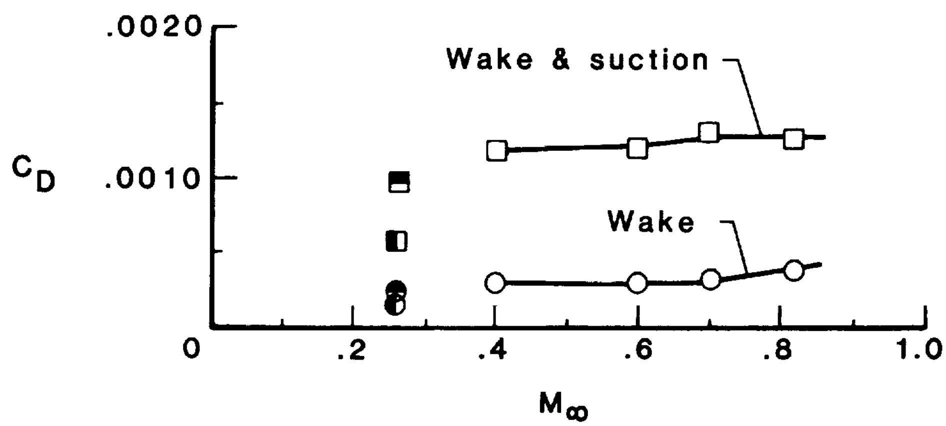

Figure 14

VARIATION OF TRANSITION FOR SIMULATED HLFC

A natural follow-on to full chord active laminar-flow control will be a hybrid configuration which will combine suction over forward regions of the upper surface with natural laminar-flow concepts over rearward regions. An attempt was made to simulate such hybrid laminar-flow conditions by turning off suction over rear portions of the LFC model while maintaining suction for $0 < x/c < .25$ on the lower surface. The simulation involved first establishing laminar flow to the most rearward thin film location (about 95 percent chord), then progressively turning off upper surface suction starting at the most rearward suction slots and measuring the extent of laminar flow downstream of the turn-off point. The variation of transition location on the upper surface with chordwise extent of suction is shown in figure 15 for M = 0.82 and two chord Reynolds numbers. The data indicate that extensive laminar flow could be maintained beyond the point where suction stopped. Clearly, this simulated HLFC approach must be considered somewhat "non-idealistic" and the results conservative since no effort was made to seal or smooth the unsucked regions of the airfoil. Furthermore, the pressure distribution (fig. 6) is not like that which would be designed for a hybrid configuration.

Accomplishments of the swept LFC airfoil experiment in the 8-foot TPT are listed in figure 16, and current and future tests are outlined in figure 17.

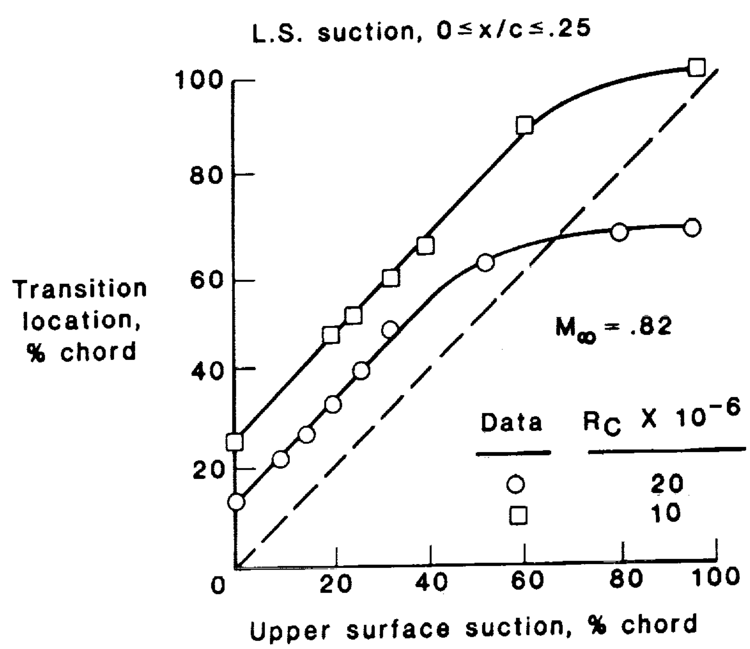

Figure 15

SWEPT LFC AIRFOIL EXPERIMENT IN 8-FT TPT

Accomplishments

- Achieved shockless pressure distribution
- Achieved full chord laminar flow on upper and lower surfaces for $0.4 \leq M_\infty \leq 0.82$
- Maintained full chord laminar flow at subcritical speeds and over large supercritical zone at transonic speeds for $R_C \leq 22 \times 10^6$
- Achieved about 80% drag reduction for upper surface only and 60% for both surfaces for $0.4 \leq M_\infty \leq 0.82$
- Demonstrated feasibility of combined suction laminarization and supercritical airfoil technology

Figure 16

SWEPT LFC AIRFOIL EXPERIMENT IN 8-FT TPT

Current and Future Tests

- Complete evaluation of porous-suction upper surface model for comparison with geometrically identical slotted model
- Evaluate hybrid LFC design with reshaped non-suction aft upper surface

Figure 17