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SUPERSONIC LFC - CHALLENGES AND OPPORTUNITIES

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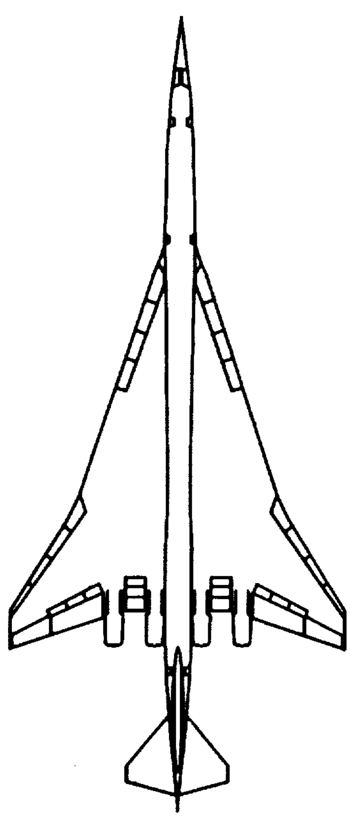
INTRODUCTION

The high fuel fractions required for long range supersonic airplanes give significant leverage to technologies for cruise drag reduction such as Laminar Flow Control (LFC). Fuel burn benefits are further enhanced when sizing effects are considered. These effects may even be powerful enough to reduce airplane production cost over a turbulent baseline. This is an important goal for LFC technology development.

The intent of this paper is to present the results of recent aerodynamics studies on the application of Laminar Flow Control (LFC) technology to the highly swept wings of supersonic airplanes. Important questions of applicability, realistic benefit, and critical application issues were addressed in a NASA-sponsored study conducted by MDC in 1987-88 (ref. 1). Figure 1 outlines the major thrusts of that study, the centerpiece of which was the Mach 2.2, 308 passenger airplane shown. More recent efforts, aimed at establishing the feasibility of demonstrating extensive Laminarization on the F-16XL-2 airplane, are also summarized in this paper.



Feasibility

Realistic Benefit

Critical Application Issues

How to Best Address Issues

Recommendations

Figure 1. Objectives of 1987-88 Supersonic LFC Study

LFC BENEFIT POTENTIAL

The 1987-88 study indicated LFC to be feasible for the Mach 2.2 configuration. The boundary layer instabilities requiring the largest suction flow to subdue were those associated with the highly swept attachment line and leading edge acceleration region. The original wing design featured a gradual acceleration on both upper and lower wing surfaces. An LFC-modified wing, having a steeper acceleration in the leading edge region, showed improvements in drag due-to-lift in addition to reduced suction flow requirements. The drag due-to-lift improvement was not considered fundamental to LFC and was not counted as a benefit.

With both surfaces of the wing and tail laminarized to the flap hinges, a 15% improvement in lift/drag ratio was realized, resulting in a resized fuel burn reduction of 17% and an empty weight reduction of 1.3% relative to a turbulent baseline. This analysis accounted for laminar area lost to bodyside turbulent wedges (ref. 2), the aerodynamic effects of LFC suction, and the weight of the suction system. The wing was assumed to be sized by initial cruise conditions.

Figure 2 shows the sensitivity of LFC benefits to system weight. Empty weight is included since this relates directly to production cost. Note the large payoff for minimizing suction system weight.

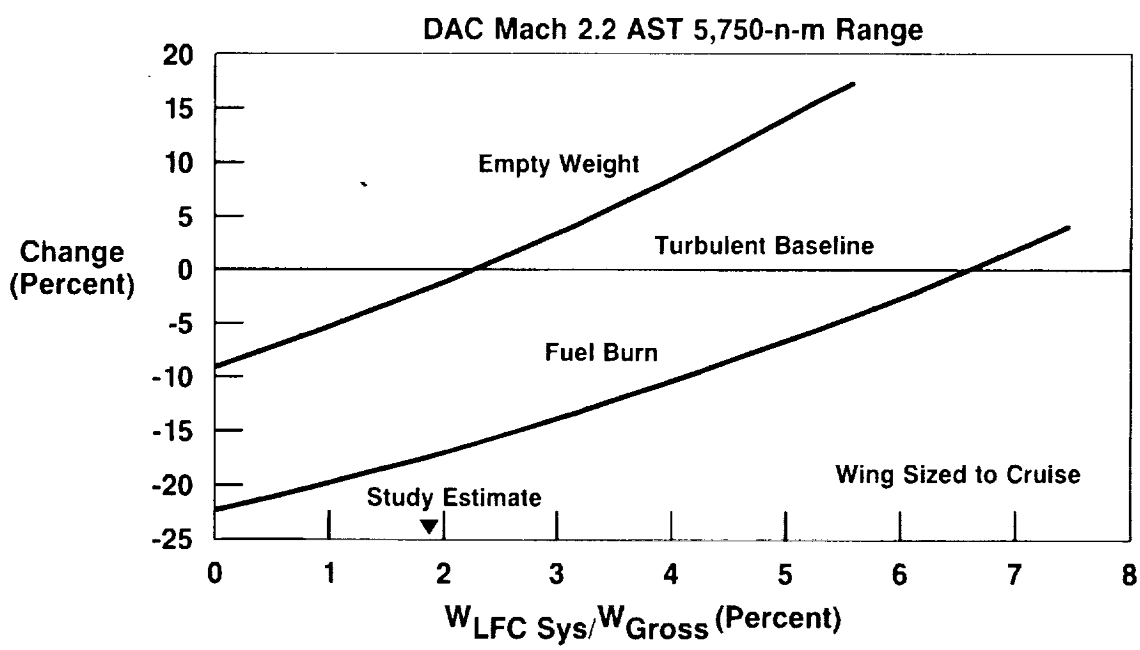


Figure 2. LFC Benefits VS. System Weight

SOME TECHNICAL RESULTS

The 1987-88 study gave several interesting results, summarized in Figure 3 below. In the subsonic case, the upper-surface drag reduction potential for laminarization is roughly twice that of the lower surface. For the Mach 2.2 case roughly 4/7 of the total drag reduction comes from the lower surface, making both surface laminarization more attractive. This is partially due to the lack of a pressure drag benefit due to reduced displacement thickness in the aft region of the wing. No such benefit exists in the supersonic case, where there is essentially no aft recovery. However, this presents an opportunity to laminarize a larger wing area fraction, and to reduce pressure and viscous drag by exhausting the suction air at low speed in a region of closure, thickening the trailing-edge boundary layer. The large chords and high sweeps of typical supersonic wings rule out the use of pressure gradients for stabilization, invalidating the HLFC concept.

The Tollmien-Schlichting mechanism of laminar boundary layer instability is known to be significantly weakened at supersonic speeds (ref. 3), while the attachment line and crossflow mechanisms are strengthened by the high leading edge sweep. These latter mechanisms were found to dominate, accounting for nearly all of the suction required. With careful aerodynamic design, particularly in the leading edge region of the wing upper surface, suction flows much lower than those of the study are possible. On the wing lower surface, careful aerodynamic design can allow wall cooling using fuel to partially supplant suction for boundary layer stabilization. Maximum LFC benefit requires suction minimization through aerodynamic design.

Both-Surface Active Stabilization Is Required

Attachment Line and Crossflow Effects Dominate

Sensitivities:

Benefits → **Suction Flow** → **Aerodynamic Design**

Figure 3. 1987-88 Supersonic LFC Study Technical Findings

CRITICAL APPLICATION ISSUES

As part of the 1987-88 study, a prioritized list of technical issues for supersonic LFC application was formulated. This list is shown in Figure 4 below. Heading the list is contamination protection, which is more difficult for cases where lower-surface laminarization is required, since the Kreuger-shield cannot be used. If liquids are to be used, their distribution over the wing is critical, and must match accretion patterns.

Attachment line criteria, well developed for the subsonic case (ref. 4) need to be extended into the supersonic regime. This impacts leading edge radius and suction. Step and gap criteria, also developed for the subsonic case (ref. 5,6), need extension to higher Mach numbers. This is important in integrating LFC and high lift systems. The supersonic excrescence criterion relates to environmental contamination, especially insect remains, the majority of which are supercritical subsonically. A supersonic transition database, taken in the actual flight environment, will be useful in the further development and calibration of transition prediction methods. Other potential issues exist, but are considered to have lesser impact or to be better understood.

Contamination Protection

Attachment Line Criteria

Step, Gap, and Excrescence Criteria

Supersonic Transition Database

Others

Figure 4. Technical Issues - 1987-88 Supersonic LFC Study

F-16XL-2 TEST ARTICLE

The 1987-88 study identified the F-16XL-2 as the best available testbed for supersonic LFC flight research. NASA LFC program personnel have reached the same conclusion independently. Both prototype F-16XL aircraft have been acquired for this and other HSR-related testing purposes. The LFC test program will be directed by the LFC Program Office at Langley Research Center, with the flight testing done at the Dryden Flight Research Facility.

Douglas Aircraft has been asked by the NASA LFC Program Office to help determine the feasibility of conducting meaningful supersonic LFC testing on the F-16XL-2 airplane. Part of the intent of this study was to uncover specific technical issues peculiar to using this vehicle for this type of testing. A possible LFC test article configuration is shown below in Figure 5. The left wing is gloved from the bodyside to the leading edge sweep break. The glove extends from forward of the original leading edge aft to the elevon hingeline. The crosshatched area is the laminar test region. This layout makes possible a laminar run of 21 feet. LFC suction air would pass through ducts imbedded in the external glove to an engine-bleed driven turbocompressor located in the gun bay area. The selection of a suitable turbocompressor unit will depend critically on the suction airflow, collection conditions, projected ducting and mixing losses, and local static pressure at exhaust.

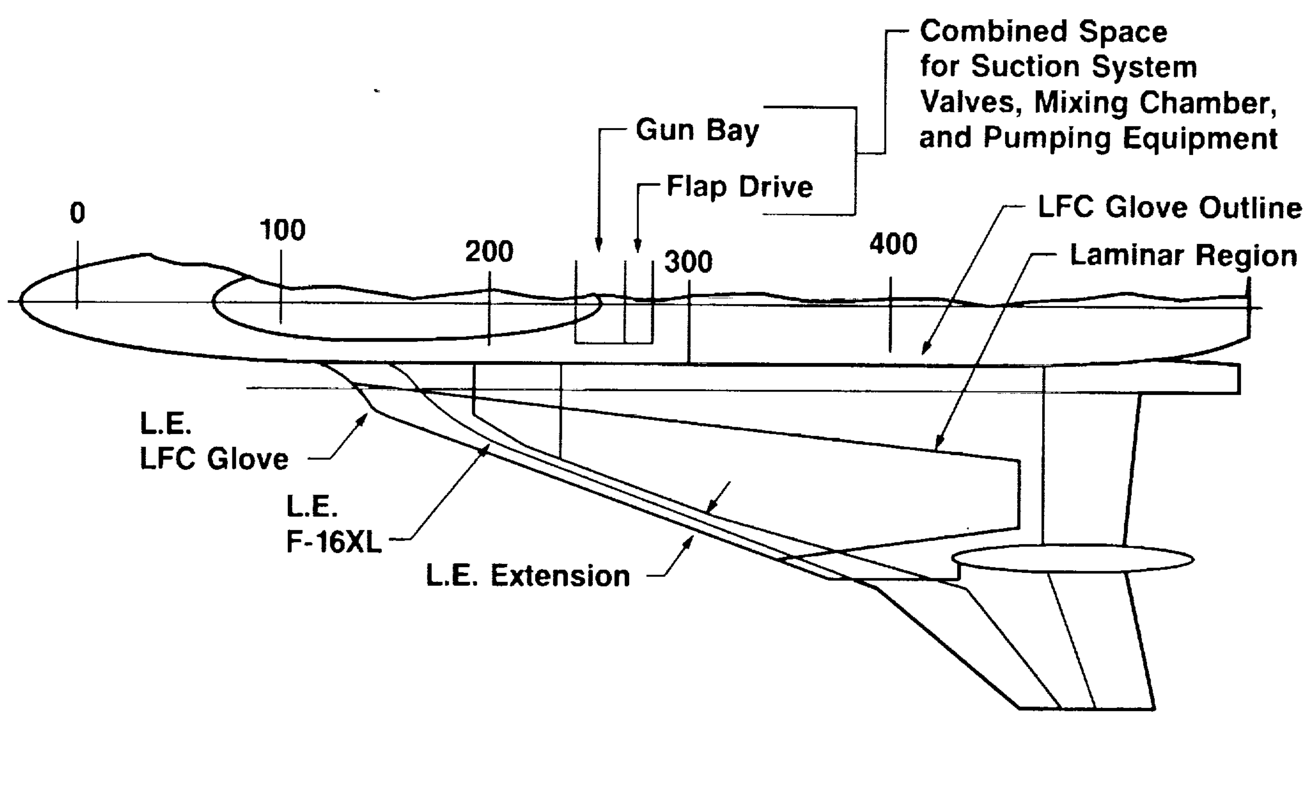


Figure 5. F-16XL-2 Study LFC Glove Planform

ESTIMATED TEST ENVELOPE

Figure 6 shows an estimated supersonic test envelope for the clean F-16XL-2 with an F110-GE-129 engine. Dashed lines of constant unit Reynolds number are shown. A study design point was selected at 1.90 Mach and 44 kft. The tropopause is indicated at 36,089 feet. In the stratosphere,where the ambient temperature is invariant with altitude, the additional pressure drag of the test article can be compensated for by taking data in descending flight without spurious thermal effects. This allows the potential of realizing the full envelope. In the troposphere, where the temperature lapse rate is nonzero, all data must be taken in level flight. Test article drag will likely limit maximum Mach numbers to something inside the envelope. The additional test article drag is not fundamental to design for LFC; it stems from large differences in design objectives between the original wing and the glove, and the necessity of providing room inside the glove for ducting.

Note the extremely wide range of unit Reynolds number available with this fighter airplane. The test article design should reflect this capability in terms of aerodynamics, temperature capability, and structural strength and stiffness in order to maximize its experimental value. Properly designed, a test article on this airplane could demonstrate laminar runs in excess of 120 million.

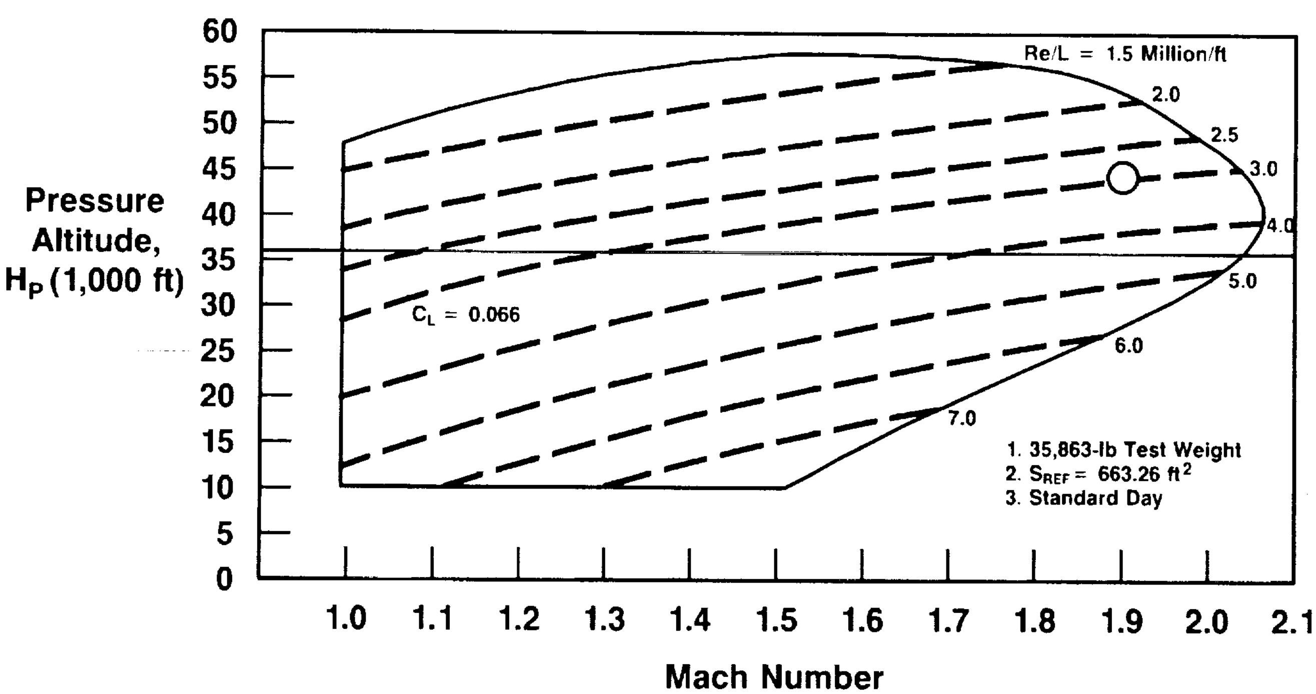


Figure 6. F-16XL-2 Estimated Supersonic Envelope

CRITICAL EXCRESCENCE HEIGHT

Figure 7 is an estimate of the effect of Mach number on critical excrescence height along a 70 degree attachment line, such as that of the F-16XL-2. Calculations were done for two values of laminar attachment line momentum-thickness Reynolds number, 100 and 240. This Reynolds number is based on attachment line external velocity and temperature. These two values have significance in the case of the incompressible, laminar attachment line. Below 100 a turbulent attachment line will relaminarize downstream. Above 240 a laminar attachment line will spontaneously transition to turbulence, due to amplification of Tollmien-Schlichting waves.

Also shown are sonic height limits: a shock will be created by any particle taller than the limit, presumably causing transition. Little relief is seen as Mach number is increased. The insect on the plot is indicative of the average height of insects deliberately collected on the JetStar Leading Edge Test Article during one flight (ref. 7). Subsonic and supersonic transports typically fly at unit Reynolds numbers between 1.5 to 2.0 million/foot, so insect impingement still must be protected against.

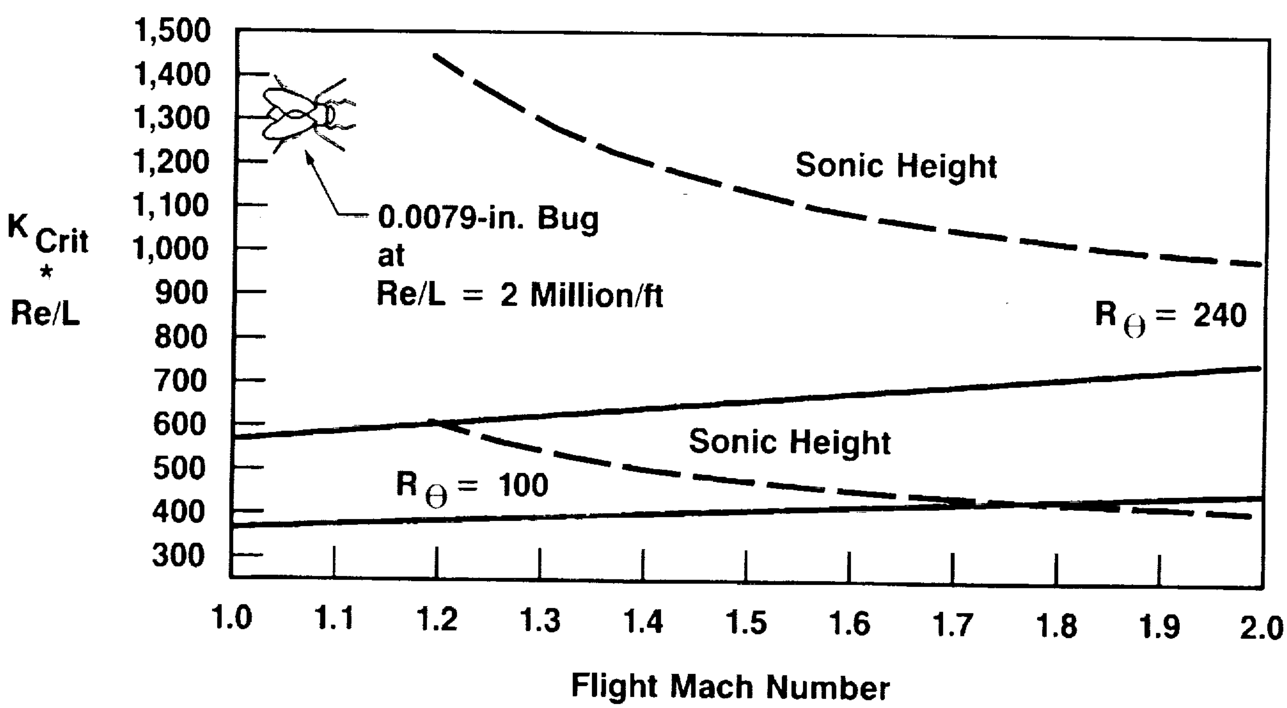


Figure 7. Estimated Critical Excrescence Height on 70 Degree Swept Attachment line

STEPS AND GAPS

Figure 8, below, is an estimate of the beneficial effect of compressibility on laminarization criteria for steps and gaps. The incompressible values were taken from the final X-21 report (ref. 5). These types of disturbances do not project upward into the boundary layer, but affect the boundary layer at the wall. The higher temperatures and viscosities at the wall create increased damping of disturbances as Mach number is increased. A single curve represents this estimated benefit. Sweeping steps and gaps beyond the local Mach angle avoids shock waves, the effect of which on transition is not known a priori. The improvement with Mach number is important if the supersonic airplane is to have leading-edge high lift devices.

Verification testing is needed. It would be valuable to know the effect of supersonic flow normal to a step or gap. The correct noise and freestream disturbance environment is critical in developing an experimental database for step and gap laminarization criteria; meaningful testing can only be done in flight. Data control calculations prior to testing are very important, so that expensive test time and fuel are not wasted.

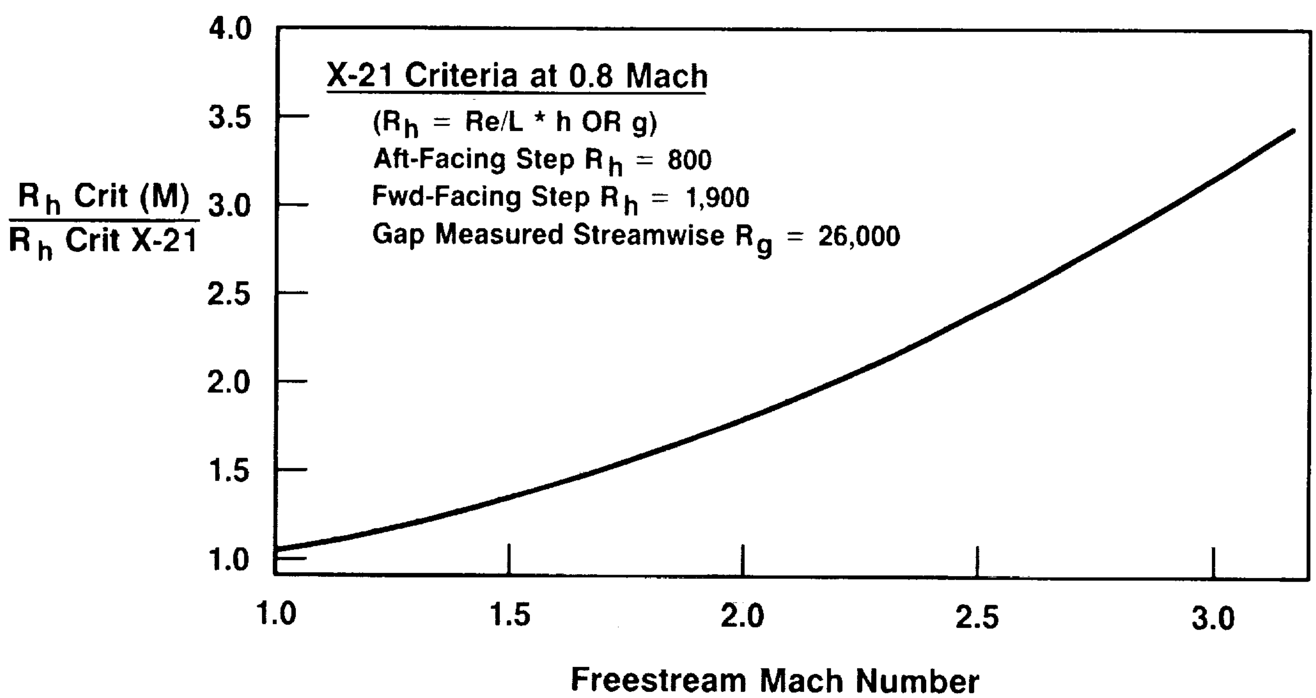


Figure 8. Estimated Mach Number Effect on Criteria for Steps and Gaps Swept Beyond Local Mach Angle

SUCTION AND HOLE SIZE LIMITS

As Mach number is increased, the increase in skin temperature causes a lowering of density and an increase of viscosity for the air entering the suction holes. Since the flow through the suction holes is laminar, these effects tend to reduce the per-hole massflux at any given pressure drop. This can be countered by reducing hole spacing and/or increasing hole size. The latter is advantageous as it also increases the hole Reynolds number, allowing more massflux through the hole. However there exists a criterion for maximum hole flow, beyond which the boundary layer is tripped (ref. 8).

A study was conducted to determine if, under likely test conditions, there would be a problem getting sufficient suction flow through the skin at the attachment line without tripping the boundary layer. The results are shown in Figure 9. For a given hole pitch-to-diameter ratio, the limiting hole diameter and corresponding largest suction coefficient was found. A large amount of latitude clearly exists. This is important since careful suction surface design will be necessary in order to allow testing at high unit Reynolds numbers.

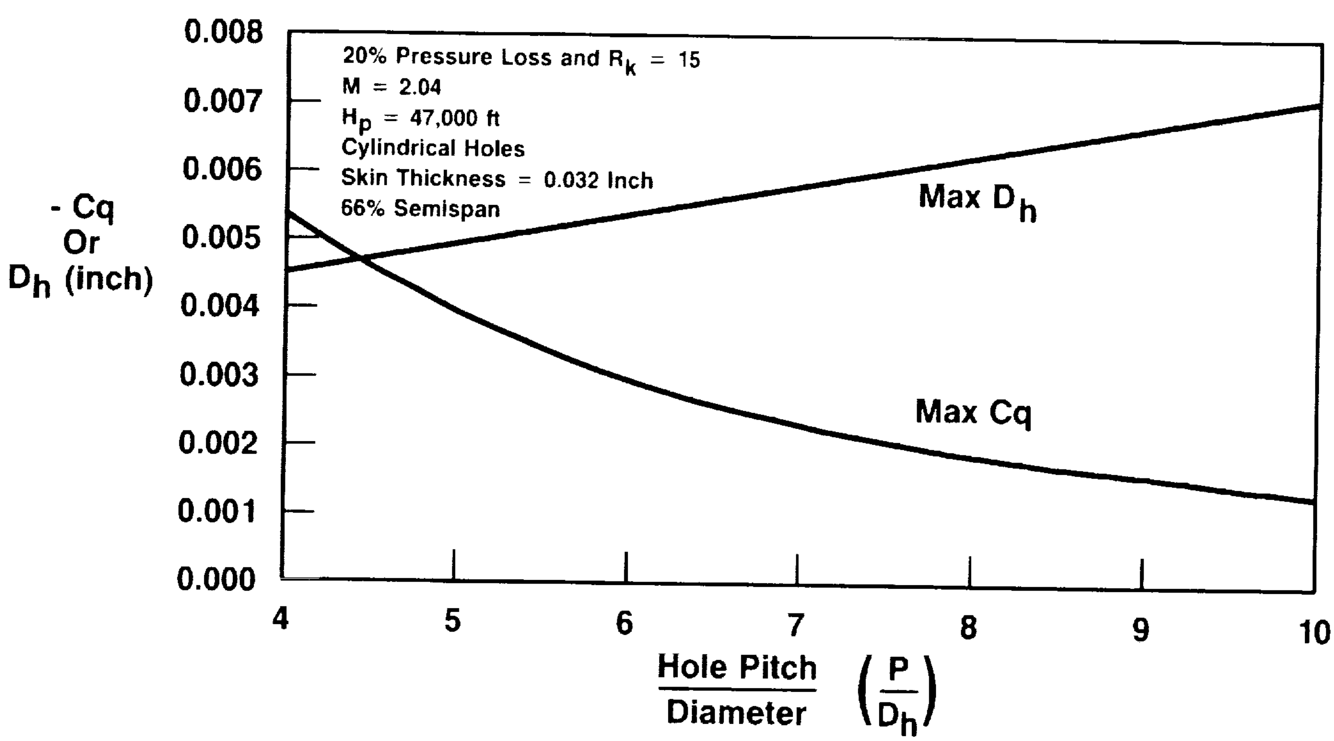


Figure 9. Estimated Maximum Suction and Perforation Size, 70 Degree Swept Leading Edge

LEADING EDGE RADIUS

The selection of leading edge radius for the test article is strongly driven by attachment line and suction criteria, and attachment line travel under off-design conditions. Laminarization considerations will set leading-edge radius and shape on a laminar flow supersonic transport as well. At the present time, attachment line criteria are only known for the subsonic case: essentially zero attachment line tangential Mach number (ref. 4). Indications are that these may not vary too much with Mach number, but sufficient experimental latitude must be allowed for in the design of the test article. Computational work at NASA Langley is underway to estimate attachment line laminarization criteria under conditions typical of the F-16XL-2 test.

Figure 10 shows the effect of suction coefficient on the leading-edge radius required to maintain attachment-line momentum-thickness Reynolds number at 100 and 240, respectively, at the study design point of 1.90 Mach, 44 kft. The compressible curves were computed using the formulation of Poll (ref. 9). A normal leading-edge radius of 0.800 inch was selected for the study.

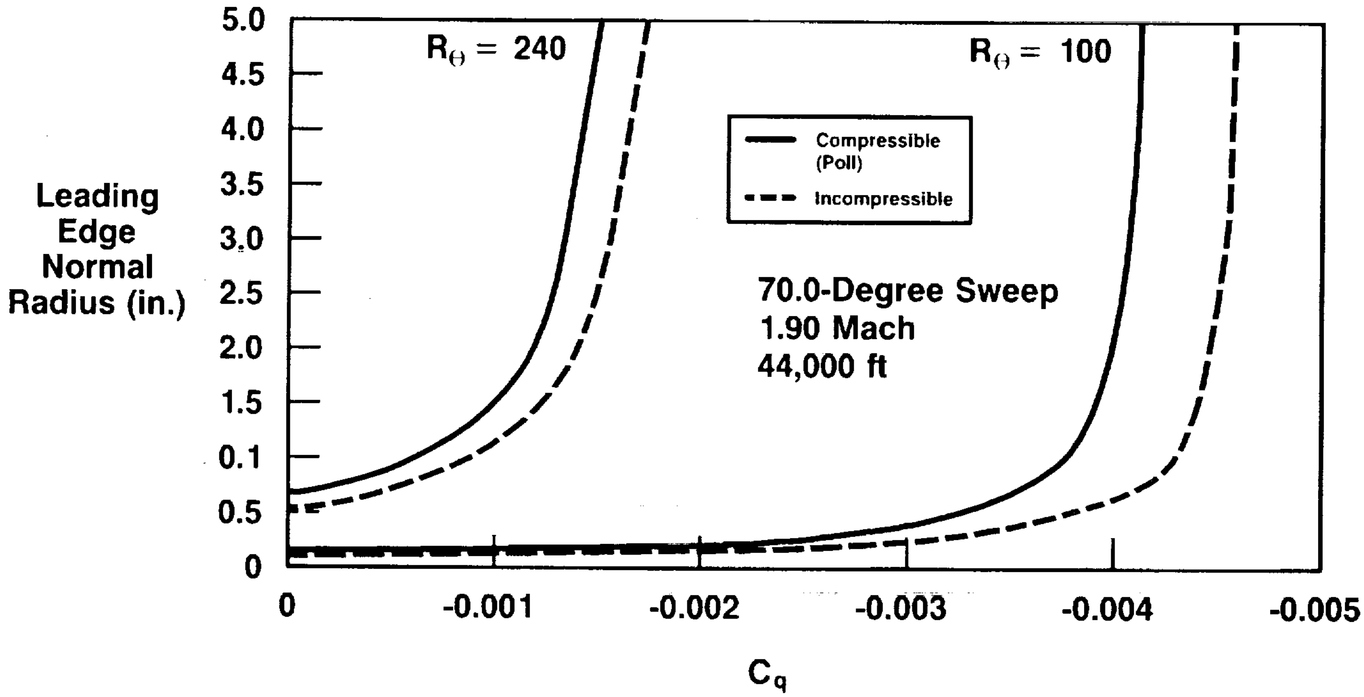


Figure 10. Attachment Line Normal Radius VS. Suction for Given R_θ, F-16XL-2 LFC Glove

STUDY GLOVE GEOMETRY

Figure 11 shows a candidate geometry for an LFC test article on the F-16XL-2. The glove extends forward of the original leading edge a nominal 4.00 inches in the normal direction, and has a minimum vertical clearance of 1.00 inches. The leading edge sweep of 70 degrees is retained. In order to create the kind of pressure distribution required for suction flow minimization at the design point it was necessary to extend the glove inboard to the bodyside, especially in the leading edge region. In the bodyside region the glove leading edge sweep is decreased to 30 degrees and the radius decreased to near zero to act as a turbulence diverter. This inboard part of the glove nullifies geometrical features of the original wing which were found to contribute substantially to the extended region of favorable gradient found in the leading edge region. The glove extends aft to the elevon hingeline. The convex region leading to glove aft termination causes an accelerating pressure field in this area, but this was intentionally located underneath the canopy closure shock at the design point, so its effect is minimized. At lower Mach numbers the canopy closure shock unsweeps, moving forward and potentially limiting achievable laminar run. A fuselage fairing designed to remove or block the canopy closure shock would be useful in allowing a wider range of useful test conditions. Lower Mach numbers are important since high unit Reynolds number conditions are only achievable at lower altitudes, where maximum speeds are lower.

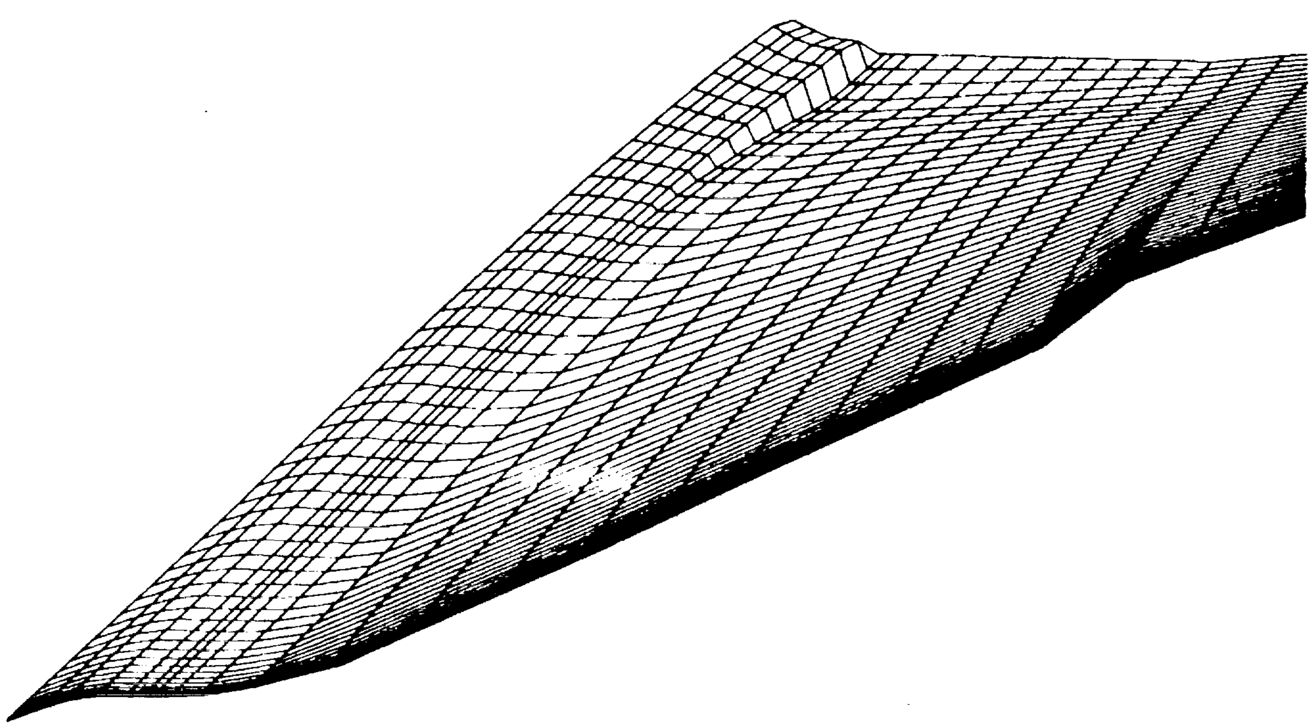


Figure 11. F-16XL-2 Study LFC Suction Test Article

COMPUTED PRESSURE DISTRIBUTION

Figure 12 compares FLO-58 - computed pressure distributions of the original wing and the study glove at the study design point of 1.90 Mach, 44 kft. The values of Cp are much smaller than one is accustomed to seeing transonically. Note the extensive region of accelerating pressure gradient on the original wing. This is very unfavorable for laminar flow, since the resulting cross-stream pressure gradients give rise to crossflow instabilty, which takes considerable suction to suppress. Note the considerable improvement achieved by the glove. Further improvements are possible through design refinement. The canopy closure shock is visible as a region of compression in the original pressure distribution. Although the shock is relatively weak, its static pressure rise is of the same order of the wing upper surface Cp. This is due to the low lift coefficient at the glove design point. The degree to which it is spread out chordwise in the Euler solution is probably a creature of the grid density, which is locally low so that computational points could be bunched in the leading edge region. Eliminating the shock or moving it aft via a fuselage fairing would enable demonstation of very high Reynolds number laminar runs at the lower Mach, high unit Reynolds number test points.

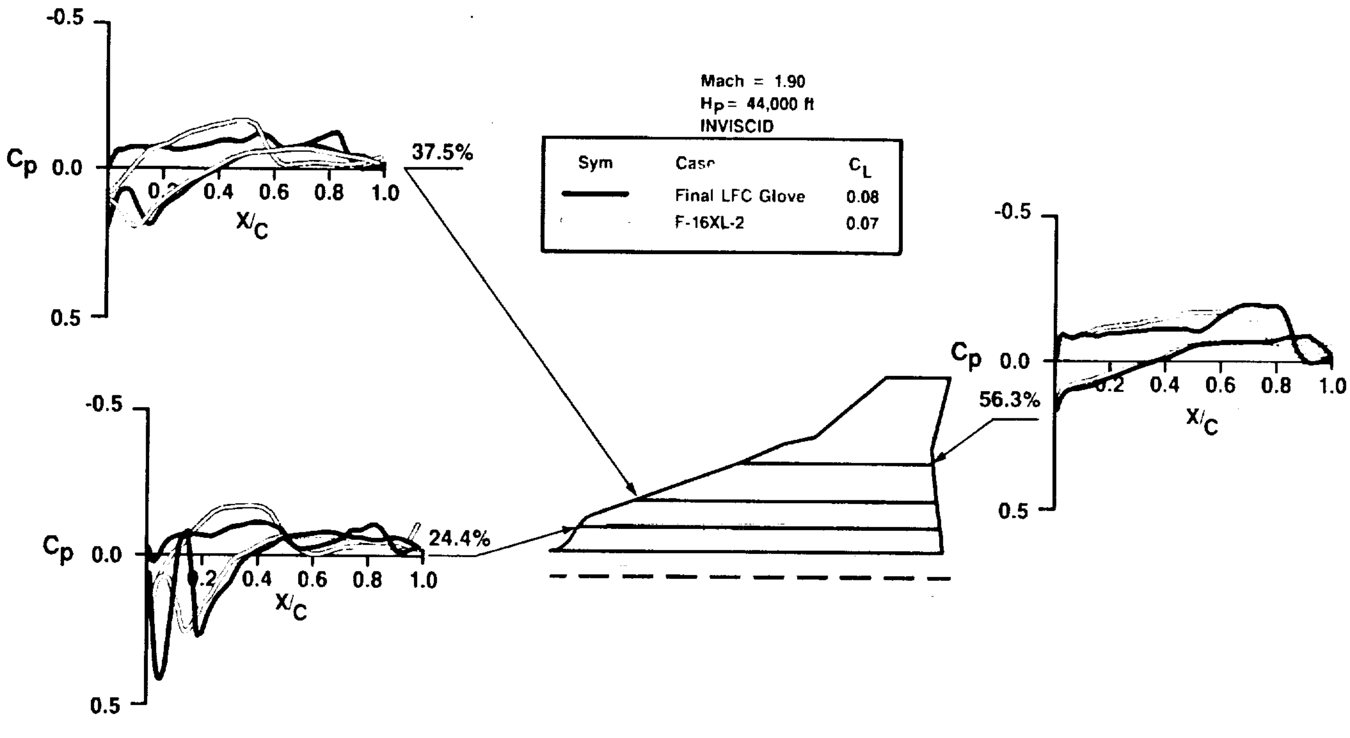


Figure 12. FLO-58 - Computed LFC Glove Chordwise C_P Distributions in Fuselage Presence

STATIONARY CROSSFLOW

A cursory analysis of stationary crossflow stability was conducted at the design point using the MARIA code (ref. 10). This code computes and integrates the growth of stationary crossflow vortices only, utilizing an approximate method involving table lookups. Experience has shown this code to be conservative in supersonic cases, but does a good job of identifying the wavelengths of the most amplified waves and giving trends. One question of interest in the design of the test article is whether or not it will be possible to distinguish between attachment line and crossflow effects. Figure 13 indicates that even with no suction, transition by crossflow is not predicted until 2 percent chord or later on the study glove. This strongly suggests that the effects will be separable experimentally if transition instrumentation is properly located.

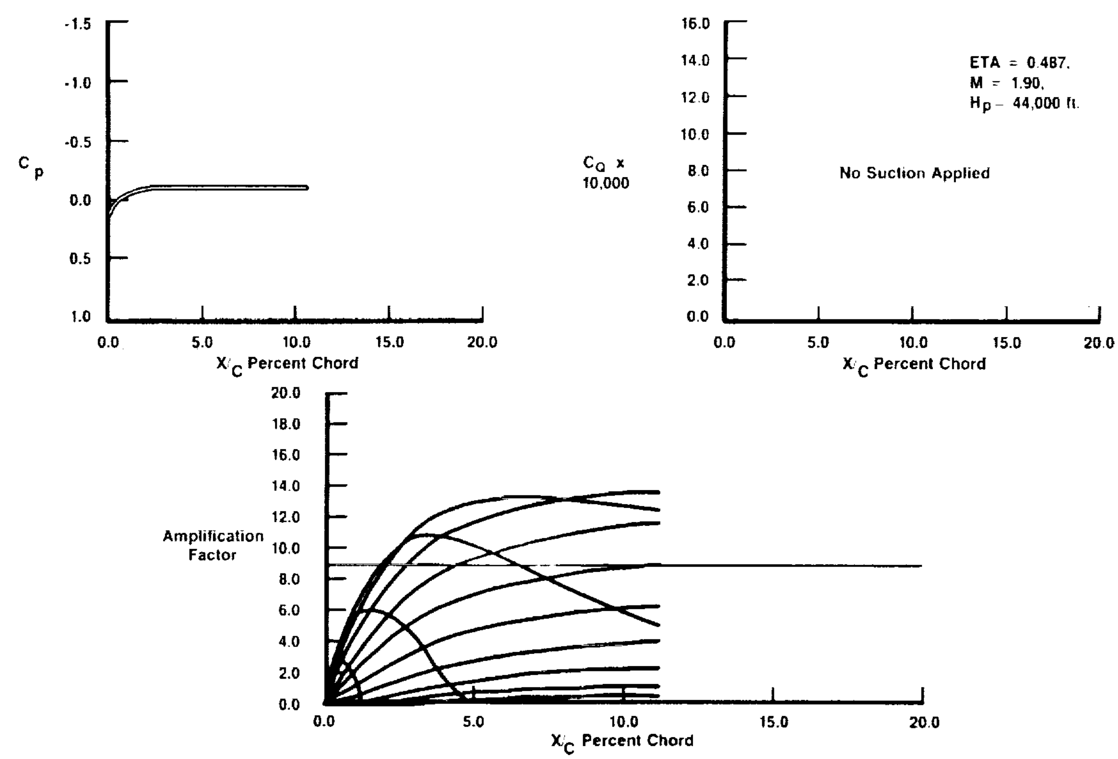


Figure 13. Stationary Crossflow Stability Analysis F-16XL-2 Study Glove

CONCLUSIONS

Figure 14 presents the major conclusions of the F-16XL-2 LFC Test Article Study. The study has identified no major roadblocks to a successful experimental program. A carefully designed test article, used in a well designed test program keyed to agreed upon major experimental objectives could provide a wealth of information directly applicable to HSCT laminarization at overall minimum program cost. It is important that the test article design reflect technological as well as demonstration goals.

Analysis Indicates Feasibility

Very Large Re/L Range Possible

Attachment Line and Crossflow Effects Are Separable

Meaningful Test Program Will Require Careful Design

- **Glove Shape**
- **Perforated Surface**
- **Structure**
- **Flying Qualities**
- **Instrumentation**
- **Test Program**

Figure 14. F-16XL-2 LFC Study Conclusions

TECHNOLOGY INTEGRATION

In order for LFC technology to earn its way onto the HSCT, it must be demonstrated to be feasible, to reliably produce the expected benefit, and integrate well with other technologies, a list of which is given in Figure 15, below. The F-16XL-2 Flight Test program is expected to establish feasibility and demonstrate the low suction levels required. Follow-on activities should focus on technology integration issues. Attention should be paid to technology combinations having possible synergisms. For example, incorporation of nonlinear effects into the aerodynamic design process is expected to result in optimized wings having lower sweep, blunter leading edges, and upper-surface pressure distributions essentially compatible with LFC requirements (ref. 12). Consistent with this design direction, alternative approaches to achieving high levels of leading-edge thrust at low speeds have been demonstrated which do not require a movable leading edge, and do not rely on suction for boundary layer separation control (ref. 13).

The contamination avoidance issue must be given serious attention. Although it is always possible in principle to design a liquid system that will work, various alternatives (ref. 14) should be investigated. The F-16XL-2 flight test should be used to document accretion patterns for future studies.

After design studies and testing have defined the best integration of technologies, bringing technical risk to acceptable levels may require in-flight demonstration.

Laminar Flow Control

Contamination Avoidance

Nonlinear High-Speed Design

Low-Speed System

Structures and Materials

Sonic Boom

Figure 15. HSCT Wing Technologies

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