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SUPERSONIC LAMINAR FLOW CONTROL ON COMMERCIAL TRANSPORTS

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ABSTRACT

This paper provides an overview of the status of supersonic laminar flow control. Existing research into the aerodynamic problems of subsonic and supersonic laminar flow control is first reviewed to provide a prospective for subsequent discussions of recent studies to evaluate the potential performance benefits of the application of laminar flow control to supersonic transports. A flight research program to provide a realistic assessment of the technical feasibility is then described.

NOMENCLATURE

C, c	Chord
C_D	Drag coefficient
C_p,	Pressure coefficient
C_Q, C_W	Suction coefficient
L/D	Lift to drag ratio
LE	Leading edge
m	Suction massflow parameter
M	Free-stream Mach number
N	Amplification factor
Re, R	Reynolds number
S, s	Surface distance, streamwise
X, x	Distance measured along chord
Greek	
α	Angle of attack
θ	Momentum thickness
η	Wing span station
Subscripts	
C, c	based on chord
CF, cf	cross flow
min	minimum
TS	Tollmien-Schlichting
t	total
y	normal distance from wall

INTRODUCTION

A resurgence of interest in supersonic transports is now occurring in Europe, Asia, and the United States. The expectation is that technology advancements since the introduction of the Concorde in 1969 and termination of the U.S. Supersonic Transport, SST, program in 1971, or technology now within reach, could lead to an economically viable, second generation SST. Many hopes are based upon the realization of projected benefits of laminar flow control, LFC, to such a vehicle. Dramatic reductions in take-off gross weight (TOGW), mission fuel burn, structural temperatures, and sonic boom have been forecast based upon an assumption of laminar, rather than turbulent, boundary-layer flow.(1) The National Aeronautics and Space Administration, NASA, has recently completed in-house and contracted studies to provide a more realistic assessment of the potential benefits of the technology.(2–4) These studies have been so encouraging that the NASA has initiated a supersonic LFC technology development program to explore the technical feasibility and provide validated aerodynamic design methodology. The intent of this paper is to overview the initial study results and describe the much broader technology development program envisioned for the next five years. To place the recent studies and plans in perspective, earlier research into the aerodynamic problems of subsonic and supersonic laminar flow will be discussed as appropriate.

SUBSONIC LFC TECHNOLOGY- THE BACKGROUND

Research on laminar flow dates back to the 1930's, over half a century ago. A review of the early flight research on laminar flow may be found in reference 5. Initial efforts were directed towards achieving extensive laminar flow through wing shaping for favorable pressure distributions on straight wings. Laminar flow was achieved in flight to about 60% chord at chord Reynolds numbers up to 25 million as demonstrated by flight tests of several aircraft. But the technology was not put into practice at that time, because concerns remained that the smooth, wave-free surfaces required were not practical. This was in spite of the British experience with the Armstrong Whitworth AW 52, a prototype with large wings (a surface area comparable to an MD 80 or A320), that met the demanding laminar flow surface requirements but did not achieve laminar flow because of unstable boundary-layer cross flows introduced by wing sweep, a phenomenon unknown at that time. (See reference 5.)

In the 1950's in both the U.S. and the U.K., flight research on suction laminar flow control was carried out on unswept wings to achieve full-chord laminar flow. These efforts were successful with full-chord laminar flow observed at Reynolds numbers up to 36 million. Extension of these methods to swept wings occurred in the 1960's with the Northrop X-21 flight tests of a full-chord, full-span LFC, swept wing.(6) Full-chord laminar flow to Reynolds numbers of 46 million was achieved, but a main goal of the program, to demonstrate operational practicality, was not. Structural flaws in the wing produce surface waves that were faired by an aerodynamic filler and extensive maintenance on the filler obscured the demonstration. The technology was "put on the shelf" as not really practical.

In the mid-1970's the NASA began to explore LFC technology again as part of a program to develop energy efficient technology for future commercial transports.(7) The belief was that progress in aerodynamics and other technologies (materials, structures and fabrication technologies) may have advanced to the point that LFC technology could be made practical. Some of the more significant technology developments that have occurred in the subsequent years include:

(1) Modern computational tools for aerodynamic analyses of laminar flow wings: the inviscid wing flow, the boundary-layer flow, and boundary-layer stability,

(2) Techniques for laser, or electron beam drilling of small perforations in titanium sheet (holes as small as 0.0015-inch in diameter and very closely spaced, 0.010-inch apart); a capability allowing the use of titanium sheet, suction surfaces formed with more or less conventional processes that can meet the demanding surface quality requirements of laminar flow (the reader is referred to reference 8 for more detail),

(3) Current techniques for structural wing panel fabrication that result in production wing-box surfaces that can routinely meet laminar flow standards.(9)

This year, flight tests of a Boeing 757 with a suction, laminar flow system(5) will hopefully bear out the practicality of LFC. A twenty-foot span of the left wing of the 757 has been modified with a hybrid laminar flow control (HLFC) system, a concept developed under the NASA program. With suction applied to the boundary layer up to the front spar, and the wing section tailored to produce a favorable pressure gradient to about 60% chord, extensive laminar flow should be achieved over the wing box surface.

The experience gained with subsonic laminar flow technology encourages extension of the technology to supersonic transports. Much of what has been learned is directly applicable, but the aerodynamic and structural design of supersonic LFC aircraft will present formidable challenges. The supersonic data base is small with only limited wind-tunnel experiments performed in the 1960's. Virtually no flight data exists.

SUPERSONIC LFC TECHNOLOGY-THE CHALLENGE

Conceptually, suction laminarization for SST's with thin and practically unswept wings represents a particularly simple approach. The boundary layer must be stabilized with respect to amplified Tollmien- Schlichting (TS) waves. This is relatively easy at moderate high supersonic speeds (M=2 to 3) due to the strongly stabilizing influence of compressiblity on the TS disturbance growth.(10) At the high chord Reynolds number of such an SST configuration extremely low equivalent wing profile drags could then be possible with full-chord suction LFC. External wing bracing by suction-laminarized struts could allow increased wing span and a correspondingly lower induced vortex drag, while further reducing the average wing thickness ratio to minimize the supersonic wave drag due to wing thickness (for its historical perspective the study by Pfenninger and Bacon(11) is an interesting application of such ideas). The lift of such a practically unswept wing would be carried over a relatively short lengthwise distance resulting in relatively large lift-induced wave drag and modest supersonic L/D and high sonic boom overpressures.

The lift-induced wave drag and sonic boom overpressures could be reduced and L/D raised by distributing the lift over a larger lengthwise distance by wing sweep. One might then choose between a conservative wing design with moderate sweep and supersonic leading and trailing edges, such as the Boeing 1970's SST design,(12) or a bolder approach with subsonic leading edges, such as the NASA Scat 15.(13) Both approaches have merits and disadvantages; compromise may be the best solution.

The supersonic flow over a moderately swept wing with supersonic leading and trailing edges contributes a supersonic wave drag due to wing thickness considerably larger than that of a more strongly swept supersonic wing with a subsonic leading edge. Furthermore, highly swept supersonic wings carry the lift over a larger lengthwise distance to reduce accordingly the lift-induced supersonic wave drag and the sonic boom overpressure, as compared to supersonic leading and trailing edges. The L/D will accordingly be higher for the subsonic leading-edge wing than for moderately swept wings. Even though the latter may be built with larger aerodynamic span to reduce the lift-induced vortex drag, its L/D will be inferior.

The suction-laminarization problems involved with the selection of wing sweep differ fundamentally. A modestly swept supersonic wing with a supersonic sharp leading edge has a very thin attachment-line boundary layer; as a result the momentum thickness Reynolds number, Re_θ, is very small and spanwise turbulence contamination along the attachment line is absent. In contrast, the more strongly swept wing with a subsonic leading edge has a finite nose thickness; as a result, Re_θ is very much larger, and spanwise turbulence contamination becomes critical at supersonic speeds and may have to be controlled by suction at the attachment line. The accelerated supersonic flow on modestly swept supersonic wings with supersonic leading and trailing edges induces a progressively growing cross flow towards the trailing edge. Its stabilization requires suction over a large percentage of the wing chord(14) for full-chord laminarization at the high Reynolds numbers of an SST. In contrast, the boundary-layer cross flow in the leading-edge area of highly swept supersonic wings with subsonic leading edges must be stabilized by relatively strong suction in a narrow spanwise strip located just downstream of the attachment line. If the boundary-layer cross flow contributed by this front acceleration is compensated by a cross flow of opposite sign in a following deceleration zone, boundary-layer cross flow becomes insignificant in the downstream flat rooftop area.(15) The wing boundary layer must then be stabilized primarily against amplified TS-type disturbances by relatively weak suction in the front deceleration zone, and possibly, in addition spanwise suction strips located further downstream. Full-chord laminarization on highly swept supersonic LFC wings with subsonic leading edges and a substantial subsonic type rear pressure recovery requires relatively strong suction in the pressure-rise area.

SUPERSONIC TRANSPORT CONFIGURATIONS

Under contracts and through in-house study, the NASA has initiated evaluations of potential supersonic LFC transport configurations. Recently completed contract studies by Boeing Commercial Airplanes (BCA) and the Douglas Aircraft Company (DAC) have taken conservative approaches wherein baseline turbulent boundary-layer transports are modified to incorporate suction boundary-layer control. These efforts are reported in references 3 and 4. the NASA's in-house studies have explored a more agressive application of LFC starting with a very ambitious supersonic configuration.(2)

BCA SST LFC Concept

An illustration of the baseline turbulent BCA SST is shown in figure 1. The aircraft is designed to cruise at M=2.4 for 5000 nmi. range with 247 passengers and a TOGW of 745,000 lbs. It assumes year 2000 projected technologies in aerodynamics, propulsion, materials (all composite structure), and systems. The aircraft has a double delta-wing planform. This configuration represents a compromise between supersonic cruise and low-speed requirements. The sweep of the inboard part of the wing is 75 degrees with a subsonic leading edge, the Mach number component normal to the wing, M_n, equals 0.62 at cruise. The wing in the inboard region has a modified NACA 65A section with a round leading edge. Outboard, the wing has a sharp supersonic leading edge with 47 degrees of sweep, M_n= 1.64 at cruise; the section outboard is a wedge-slab-wedge (see figure 2).

The basic wing pressure distributions, inboard and outboard, obtained by an Euler code solution are shown in figure 3. Initial boundary-layer analyses indicated that the pressure distribution on the turbulent baseline would have to be modified if a significant run of laminar flow was to be achieved: a milder recovery from the initial pressure peak on the inboard upper surface would be necessary to avoid laminar separation; a steeper initial acceleration to avoid cross-flow development in the leading edge; and flatter pressure distributions where possible over the wing box areas to avoid a cross-flow build up in these areas. The desired modifications are indicated in figure 3. The pressure peaks on the inboard wing follow a suggestion by Pfenninger.(2) The pressure recovery downstream of the peak pressure generates a cross flow in the opposite sense to that generated in the initial flow expansion and hence significantly lowers the cross flow into the wing box area.

To incorporate these changes into the wing geometry an inverse design code is needed; such a code is being developed under the NASA SLFC program (see the later section entitled "NASA Supersonic LFC Research Program"). These changes were only approximated in the BCA study.(3) Clearly in the design of future supersonic laminar flow wings, to achieve pressure distributions compatible with laminar boundary-layer control will require incorporation of LFC concerns at the onset.

The suction-laminarization schemes proposed for the modified pressure distributions on the upper wing surface are shown in figures 4a and 4b. Therein, the pressure distributions, cross-flow Reynolds number (Re_{CF}), and TS and cross-flow disturbance growth N-factors (N_{TS} and N_{CF}) are shown. Inboard on the upper surface (figure 4a), strong suction is required to control attachment-line turbulence contamination and the initial cross-flow instabilities in the leading-edge flow acceleration. Aft of the pressure minimum, the pressure recovery results in a suppression of further cross-flow development and a reduced suction level is adequate control for the TS-disturbance growth until more moderate growth occurs in the constant pressure area. Laminar flow to about 30% chord appears possible without suction further downstream. In the pressure recovery aft of 40 percent chord, substantial additional suction would be required to suppress the cross-flow disturbance growth and the possibility of extending the laminar flow beyond 30 percent chord was not explored. Outboard on the wing (figure 4b), two suction strips are used with moderate suction levels to produce laminar flow to about 60% chord. The suction is needed to control both the TS-disturbances and cross-flow disturbances; the latter grow particularly rapid in the expansion centered about x/c=0.25. A further improvement in the run of laminar flow might be obtained by a more rapid pressure expansion in this area.(3) Similar suction distributions were used on the lower wing surface inboard and outboard.

Several such schemes for achieving laminar flow over the wing were considered, and while these approaches would not be judged to be aggressive applications of LFC, the impact upon aircraft performance of even a modest application is quite impressive. The drag benefits for one suction scheme are indicated in figure 5. With 41% of the wing surface area laminarized, the friction drag is reduced 34%. If one includes the reduction in configuration skin friction drag due to an assumed, twenty percent chord, natural laminar flow on the nacelles and empennage, the total cruise drag reduction is 8.2 percent. The impact of these friction drag reductions on a resized configuration are indicated in figure 5. Gross weight and empty weight reductions of 8.5 and 6.2 percent with 12% reduction in fuel burn are possible for an aircraft designed for a 5000 nmi. range. These benefits would have a dramatic impact upon the economics of such a vehicle and are achieved with a relatively modest LFC system. The total suction air required for laminarization is appoximately 10 lbs./sec, which is less than 2 percent of the engine airflow at cruise, and the suction power expended would be about 1220 hp., extracted as shaft power from the main engines. The suction system weight was estimated at about 8600 lbs., about 1 percent of the gross take off weight. The fuel volume displaced by the suction ducts was estimated to be significantly less than the block fuel saved by the laminarization. With the aircraft resized for a 6500 nmi. range the performance benefits are even more impressive: 12.6 and 9.8 percent reductions in take off gross weight and operating empty weight, respectively, and a 15.9 percent reduction in block fuel.

Douglas SST LFC Concept

The turbulent baseline for the Douglas Aircraft Company contract study for the NASA(4) was the Douglas Advanced Supersonic Transport configuration shown in figure 6. This configuration was developed under a previous NASA contract in the 1970's.(16) The aircraft is designed to carry 308 passengers at M=2.2 for a range of 5750 nm. with a TOGW=750,000 lbs. The aircraft would use variable-cycle engines and super-plastic-formed and diffusion-bonded (SPF/DB) titanium wing structure. It has a cranked arrow wing with the majority of the wing swept behind the free-stream Mach line at 71 degrees of sweep; outboard of the 70% wing span station the leading edge is supersonic at cruise with 61.5 degrees of sweep. The lower sweep in the outer wing panels is a compromise to achieve better low-speed performance and handling, but this planform still leads to an aerodynamically more efficient turbulent wing at cruise than the BCA baseline. The cruise L/D is 10.25, compared to the BCA baseline aircraft L/D of 9.2.

An attempt was made to achieve near full-chord suction laminar flow control on the basic turbulent wing (i.e.- laminar flow to the hinge line of the control surfaces), and while this was judged to be possible, an inordinate amount of suction was predicted and some concern about the duct volumes arose. The inboard, subsonic part of the wing presented the difficult laminarization problems and some modification of the wing sections was necessary. The section changes that were made are indicated in figure 7. As shown in figure 8, these changes resulted in inboard wing pressure distributions that were more conducive to laminar flow. The blunter sections produced more rapid accelerations from the attachment line and more uniform pressures over the wing box area. The wing redesign was accomplished with an Euler code in an iterative approach relying upon transonic analyses of normal wing sections to guide the redesign.(4) The blunter leading edges resulted in a near doubling of attachment-line Re_θ, but the increased suction on the attachment line to reduce Re_θ to tolerable levels was judged to be a good trade with the reduced suction for cross-flow control. The wing planform was also changed to improve the prospects for laminar flow outboard on the supersonic portion of the wing where the sweep was reduced to 50 degrees. It is noteworthy that Görtler vortices on the concave regions of the lower, unmodified wing surface were judged not to be a potential problem as they would be curtailed by the development of the cross-flow vortices.(4)

Representative pressure and suction distributions for near full-chord suction laminar flow control are shown in figures 9a and 9b. Intense suction is required only in the attachment-line and leading-edge area to control turbulence contamination and cross-flow growth in the initial acceleration. Thereafter, a very low level of suction is necessary to maintain laminar flow all the way to the control surface hinge lines; this is made possible by the supersonic trailing edge and absence of a pressure rise in the aft part of the wing. The total suction mass flow was 22.3 lbm./sec at the initial cruise condition of Mach 2.2 and 58,000 feet. With laminar flow on the vertical and horizontal tails an L/D improvement of 15.1%, to L/D=11.8, is predicted. A fuel burn reduction of 17.2% would occur. These potential benefits are so impressive that laminar flow control is now being incorporated in the Douglas baseline for their High Speed Civil Transport (HSCT) Study for the NASA.

SLFC Concept- A Bold Approach

The requirement of a particularily low sonic boom overpressure, necessary to enable supersonic flight over land, may well turn out to be more decisive than pure performance considerations alone in favoring highly swept supersonic LFC wings with a particularily low lift-induced supersonic wave drag and high L/D's during cruise. For this reason such design approaches have been emphasized in in-house studies directed by Dr. Werner Pfenninger at the Langley Research Center.(2) To minimize the lift-induced wave drag and sonic boom over-pressure and maximize the supersonic cruise L/D, supersonic LFC wings, swept behind the free-stream Mach cone, have been laid out such that the lift is distributed over a particularily large lengthwise extent. Furthermore, the flow Mach number component normal to the upper surface isobars in the lifting zone of the wing is chosen to be a high subsonic or transonic value about equal to the design values of advanced supercritical LFC airfoils.(17) An example of these ideas is illustrated in figure 10. A supersonic airplane is envisioned that would have a highly swept supersonic LFC wing of very high structural aspect ratio and large wing span. Such relatively large wing spans and aspect ratios with a correspondingly low lift-induced vortex and supersonic wave drag appear possible by bracing the wing externally with highly swept suction-laminarized wide chord struts of low parasite drag. These struts take out both bending and torsional moments and deflections, as discussed in ref. 15 for high subsonic speed LFC transports, and must be carefully contoured according to the undisturbed, and effectively subsonic (or transonic) flow around an infintely long highly swept strut. Indeed, such contouring is possible even in the wing-fuselage juncture area as evidenced by the experiments of Hilton.(18) With strut-braced wings, the vertical distance between the wing attachment on the fuselage and the strut attachment on the fuselage can be used as a large basis for the wing rotation around the vertical axis to alleviate accordingly the structural problems and minimize weight penalties involved with variable sweep to improve the low-speed performance.

The aeroelastic problems involved with such highly swept supersonic wings have been considered excessively severe, and industry has accordingly abandoned such an approach. With materials of high strength and stiffness, active controls, and other advanced technologies becoming available, these problems may be less critical now. Questions arise as to how best to handle the structural, aeroelastic, flutter, stability and control problems involved; how to ensure satisfactory low-speed characteristics and a sufficiently high airplane L/D with a reasonably low induced vortex drag during take off, climb, loitering and flight over land at lower flight speeds; and how to minimize sonic boom with minimum performance penalities. If it should prove possible to drastically reduce the lift and volume induced supersonic cruise wave drag by a suitable design, displace the shocks from various airplane components lengthwise, avoid or minimize N-waves and maintain the near field pressure pattern over large distances to the ground, the supersonic boom overpressures may eventually be reduced such that supersonic cruise over land may be acceptable.

The questions regarding the suction laminarization of such highly swept supersonic LFC wings have been addressed.(2) As compared to the previously described transport designs, the wing chord Reynolds numbers of the highly swept, strut-braced high aspect ratio supersonic LFC wing are subtantially lower, especially based on maximum wing chord, alleviating substantially the suction-laminarization problems involved. In particular, with the smaller wing chords inboard and lower unit Reynolds numbers (due to higher optimum cruise altitudes with larger aspect ratio), the attachment-line Re_θ is substantially lower which eases the attachment-line boundary-layer stability problems and possibly avoids the need for attachment-line suction. Advanced supercritical LFC sections of the Pfenninger design(15) greatly simplify the problems of maintaining laminar flow (possibly to the trailing edge). An illustration of these ideas is shown in figure 11 which gives a typical inboard section, pressure distribution, and suction distribution that should permit full-chord laminar flow on the wing upper surface at the cruise condition, chord Reynolds number of 80 million. Figure 11 shows the corresponding plots of C_p, C_Q, and N-factor versus the x/c across the chord. Suction to stabilize the attachment line would not be needed as the attachment-line Re_θ would be less than 240. One might provide at discrete stations along the attachment line local suction patches which would reduce the Re_θ below the value for turbulence contamination to reestablish an undisturbed laminar attachment line, if necessary. Relatively strong local suction in a relatively narrow chordwise suction strip is needed in the front acceleration zone of the upper surface for optimum boundary-layer cross-flow control in this area. The overall suction mass flow and power needed in this zone are surprisingly low for such a highly swept wing. This is explainable by the relatively sharp leading edge and the optimum type of boundary-layer cross-flow control used; namely the rapid acceleration to limit cross-flow development, the accompanying suction, and the pressure peak to establish an opposed cross flow downstream of the initial acceleration. With the latter pressure peak, the boundary layer in the extensive, nearly flat, rooftop zone must be stabilized primarily against amplified oblique TS-disturbances. This is possible with relatively weak suction in one or several spanwise suction strips, starting shortly downstream of the pressure minimum from 0.05c to 0.30c. In the aft pressure-rise area, the combination of a severe deceleration and relatively thick laminar boundary layer generates a particularly severe boundary-layer cross flow. The cross-flow vortex strength is minimized by decelerating the flow over a short chordwise distance. Strong suction is needed in this zone, but suction rates and powers can be held to modest levels leading to low equivalent wing profile drags. Similar suction schemes can be ultilized for the lower wing surface and remarkably high L/Ds (L/D = 19 at cruise Mach number of 2) appear feasible with reasonably extensive laminar flow over the airplane exposed wing surfaces.

Clearly, the design of an advanced LFC supersonic transport presents a difficult compromise with many different conflicting requirements. To satisfy them, the designer may have to accept unconventional design approaches and variable geometry in one form or another. But bold, new ideas, such as those explored by Pfenninger,(2) could lead to a dramatic reversal of the economics of passenger-carrying supersonic flight.

STATE OF THE ART FOR SUPERSONIC LAMINAR FLOW CONTROL

The data base for supersonic laminar boundary-layer control by suction is sparse compared to that existing for subsonic applications. Some remarkable wind-tunnel experiments were performed in the 1960's, and although some flight results on laminar boundary-layer transition are available, flight data on suction laminar boundary-layer control at supersonic speeds does not exist in the open literature. What is available will be discussed in the following section.

LFC Experiments at Supersonic Speeds

As compared to compressible laminar boundary layers, the larger kinematic viscosity and correspondingly stronger dissipation in the hot, low density inner zone of adiabatic supersonic laminar boundary layers raise the Tollmien-Schlichting (TS) stability limit Reynolds number and lower the TS-disturbance growth at supersonic speeds up to M=4. This theoretical result has been confirmed by supersonic low drag suction experiments of E. Groth in the Tullahoma Arnold Engineering and Development Center (AEDC) 1x1 meter, supersonic tunnel for adiabatic wall conditions at M=2 to 3.5 on a slotted low drag suction plate and on a slotted suction body of revolution.(19–20)

Figures 12 and 13 show drag results for the 42-inch chord supersonic flat plate suction model with 76 spanwise suction slots. One-hundred percent laminar flow was obtained at length Reynolds numbers of 26 million at M=3 and 22 million at M=3.5, corresponding to the upper tunnel limit. The minimum equivalent plate drag coefficients (including the equivalent suction drag but excluding the shock wave drag) were 28 percent and 39 percent, respectively, of the turbulent plate skin friction at the same M and length Reynolds numbers. To maintain full length laminar flow in the presence of the severe acoustic disturbances generated by the quadrupole-type noise of the turbulent tunnel-wall boundary layer, the suction mass flow coefficients had to be increased to $2\text{x}10^{-4}$ to $3\text{x}10^{-4}$.

Figure 14 shows the minimum equivalent drag (including the equivalent suction drag but excluding the shock wave drag) and the required suction mass flow rates of the supersonic low drag suction body of revolution. Full-length laminar flow was obtained by means of suction through 150 closely spaced slots up to a Reynolds number of 42 million at M=2.0 and a Reynolds number of 51.5 million at M=3, again corresponding to the upper tunnel limit. In spite of the severe acoustic disturbances generated by the quadrupole noise of the turbulent tunnel-wall boundary layer, the equivalent drag of the supersonic low drag suction body of revolution at M=3 and 51 million Reynolds numbner was only 23 percent of the turbulent flat plate skin friction at the same M and Reynolds number (figure 14).

In view of the possibility of weak incident shocks, generated by components of a supersonic LFC airplane, and their impingement on the laminarized surfaces of such an airplane, the question arises as to how to maintain laminar flow by means of suction in, and downstream of the boundary-layer shock-interaction zone. To answer this question, laminar boundary-layer shock-interaction experiments were conducted jointly by I. Greber and the Northrop LFC research group on a flat suction plate at M=2 and relatively low length Reynolds number in the 8-inch supersonic tunnel of the Massachusetts Institute of Technology (MIT) gas turbine laboratory.(21) Incident shocks were generated by an inclined flat plate and impinged on the test plate. Suction was applied in the shock impingement area of the plate using closely-spaced spanwise slots to maintain laminar flow in and downstream of the interaction zone up to the plate trailing edge.

Figures 15a through 15c present the experimental setup, the chordwise variation of the surface skin friction, pressure distribution and boundary-layer development over the plate with and without suction. As shown by the surface skin-friction and boundary-layer measurements, without suction the laminar boundary layer separated in the area downstream of the front oblique leg of the λ-shock to become turbulent at the downstream end of the laminar separation bubble. The corresponding chordwise pressure distribution without suction shows a weak initial pressure rise close to the front leg of the λ-shock, followed by a flat plateau in the area of the laminar separation bubble and a steep pressure rise with a turbulent boundary layer in the area of the main shock.

In contrast, surface-friction and boundary-layer measurements with suction (figures 15b and 15c) reveal that the formation of λ-shocks and the laminar separation bubble in the shock region could be eliminated by suction in the shock-interaction zone. The presence of the boundary layer smears the surface pressure rise in the shock-interaction zone over a finite distance similar to a rapid subsonic pressure rise to render suction laminarization in boundary-layer shock-interaction zones possible at all. Laminar boundary layers were, indeed, obtained in and downstream of the shock region along the entire plate by means of sufficiently strong suction in the region of the incident shock (15c).

Subsequently, the supersonic boundary-layer shock-interaction experiments with suction through closely spaced slots at MIT were extended by E. Groth on the previous 42-inch chord flat supersonic suction plate in the 1 by 1 meter supersonic Tullahoma tunnel to higher Reynolds numbers.(22) An incident shock was generated by an inclined flat plate (figure 16). Additional closely-spaced spanwise suction slots, connected to separate suction chambers, were provided in the boundary-layer shock-interaction area of the plate. At M=3 full-length laminar plate flow was obtained up to a Reynolds number of 26 million with a 1.6 pressure ratio through the shock-interaction zone, using relatively strong local suction in the shock-interaction zone and somewhat increased suction rates over the remaining plate areas (as compared to the plate without incident shocks). As in the MIT-experiments, the absence of a plateau in the pressure distribution prior to the strong pressure rise across the shock with suction indicates that the relatively strong local suction on the boundary-layer shock-interaction zone had prevented the formation of a laminar separation bubble in the shock-interaction region.

In the previously described supersonic LFC experiments boundary-layer cross flow had been absent. The required boundary-layer stabilization against amplified TS-type disturbances had been relatively easy due to the stabilizing influence of compressibility on TS-instability. Considering the severe initial disturbances introduced into the boundary layer by the quadrupole noise of the turbulent tunnel-wall boundary layer the laminar flow length Reynolds numbers obtained with low suction are remarkably high.

In view of the fact that aerodynamically efficient supersonic airplanes need swept wings, the question arises concerning the suction laminarization of supersonic LFC wings swept ahead as well as behind of the free-stream Mach angle. On one hand, due to the higher temperatures and the correspondingly lower air density in the vicinity of the wall of compressible supersonic boundary layers, the kinetic energy of the boundary layer in the wall zone is smaller than in the incompressible flow to withstand the same cross-flow pressure gradients, raising accordingly the boundary-layer cross-flow velocity and cross-flow Reynolds number in supersonic flow over the incompressible values. These higher cross-flow Reynolds number values are partially compensated by the alleviation of the boundary-layer cross-flow instability by compressibility at supersonic speeds, though not nearly to the same degree as with the TS-instability, since the critical cross-flow disturbance layer is located much further away from the surface than the critical TS-layer. Therefore, the increase in temperature and thus of viscosity and dissipation in the critical cross-flow disturbance layer is correspondingly smaller than in the critical TS-layer. As a result, compressibility alleviates boundary-layer cross-flow instability substantially less than TS-instability. Thus, boundary-layer cross-flow instability decisively influences the boundary layer and design of supersonic LFC wings swept ahead and behind of the free-stream Mach angle, especially at the high chord Reynolds numbers of supersonic LFC airplanes.

In order to verify suction laminarization of swept supersonic LFC wings, supersonic low drag suction experiments were conducted in the Tullahoma 1x1 meter supersonic tunnel on a 36-degree yawed, 2.5 percent thick supersonic biconvex LFC wing by E. Groth at M=2.5, 3, and 3.5.(14) Figures 17a through 17c show the equivalent profile drag versus chord Reynolds number (including the equivalent suction drag but excluding shock wave drag) and suction mass flow for the upper test surface of the 36 degree biconvex supersonic LFC wing of 37.8 inch chord with 100 percent laminar flow at M=2.5, 3, and 3.5. Full-chord laminar flow and low drag were obtained at M=2.5, 3, and 3.5 up to chord Reynolds numbers of 17, 25, and 20 million, respectively. These laminar flow chord Reynolds numbers compare with those obtained with the 42-inch chord flat suction plate. The drag coefficient, though, is somewhat higher and partially erratic, presumably caused by three-dimensional disturbances originating from the intersection of the model with the tunnel walls and propagating along characteristics into the test area.

In view of the low lift-induced wave drag of highly swept supersonic LFC wings, swept sufficiently behind the free-stream Mach angle such that the flow in the direction normal to the isobars is subsonic or transonic with embedded, relatively shallow supersonic bubbles, the question arises concerning their suction laminarization. To answer this question low-drag suction experiments had been conducted in the Tullahoma 1x1 meter supersonic tunnel by J. Goldsmith on a 72-degree yawed supersonic LFC wing model at M=2 and 2.2 with subsonic type flow in the direction normal to the wing.(23) To enable two-dimensional flow along the entire test span and, thereby, maintain subsonic type flow in the test region the intersection of the wing model with the wind-tunnel wall was shaped according to the undisturbed streamlines around an infinitely long yawed wing of the same crossection working in infinite flow (figure 18a).

The strong rear pressure rise on the upper surface generates a severe boundary-layer cross flow on the 72 degree swept wing model at M=2 to require relatively strong suction for full-chord laminarization. Figure 18b shows $C_{D,suction}$, $C_{D,wake}$, and $C_{D,total}$ versus the total suction massflow coefficient for the upper wing surface at M=1.99, Re_c= 7.3 million and α= 0.15 degrees. Full-chord laminar flow with $C_{D,total,min.} = 0.00135$ was obtained for the upper surface. At chord Reynolds numbers greater than 9 million $C_{D,total}$ increased rapidly (figure 18c). An analysis of the attachment-line Re_θ indicates that spanwise turbulence contamination along the front wing attachment line, triggered by the turbulent tunnel-wall boundary layer at the upstream end of the wing leading edge may have been responsible for the drag increase.

NASA SUPERSONIC LFC RESEARCH PROGRAM

The NASA has initiated a technology development program to provide a base for determination of the feasibility of supersonic laminar flow. The centerpiece of this program is the F-16XL Supersonic Laminar Flow Control (SLFC) Experiment. The F-16XL test aircraft has a highly swept cranked wing planform that closely resembles subsonic leading-edge configurations proposed by industry (figure 19). The experiment has the following overall objectives:

1. Achieve 50-60% chord laminar flow on a highly swept wing at supersonic speeds,
2. Deliver validated computer codes and design methodology to industry for the design of supersonic laminar flow wings,
3. Establish initial LFC suction system design criteria to allow industry to more accurately integrate the concept into the HSCT and determine benefits.

In support of the F-16XL flight experiment, there are several experimental and computational Research and Technology activities underway at the NASA Langley Research Center and the NASA Ames Research Center. A highly swept (no suction) leading-edge model is being designed for tests in the Langley pilot M=3.5 Quiet Tunnel. This model will be used to study attachment-line turbulence contamination mechanisms and other leading-edge flow phenomena. A suction model will be designed and tested at a later date to demonstrate suction control of amplified cross-flow disturbances and evaluate suction hole-induced disturbances. Also, a fully three-dimensional laminar boundary-layer code is being coupled with an Euler code and an advanced boundary-layer stability code to provide higher accuracy modeling of the supersonic flow for these highly swept wings. An inverse Euler design code is being developed for the design of test surfaces to be evaluated in the three flight-test phases described below. The inverse code will identify the aerodynamic contour which produces a prescribed surface pressure at flight design conditions. Full Navier-Stokes calculations of the complete F-16XL configuration are being performed to provide flow field and wing leading-edge boundary-layer details. At Ames, advanced transition detection methods such as pressure-temperature luminescent paint are being evaluated in wind tunnels, and a flight experiment on an F-104 is scheduled to determine their suitability for supersonic application on the F-16XL.

The flight research program is divided into three distinct phases which span a period of six years (figure 20). Phase I began in October 1989.

Phase I- Passive Glove

This effort will involve the aerodynamic design and testing of a passive (no suction) glove which will be fabricated from foam and fiberglass and installed over the existing F-16XL wing. A primary goal in Phase I is to obtain detailed surface pressure data, particularly in the leading-edge region, to calibrate the Euler analysis codes. Existing supersonic analysis codes are not able to accurately capture the pressures in the leading-edge region which is a crucial area for a supersonic laminar flow control wing. Another key concern is the extremely high sweep of proposed HSCTs which aggrevates attachment-line turbulence contamination, a condition which can require high local suction levels or prevent attainment of laminar flow entirely. Solutions to the turbulence-contamination problem will be explored in Phase I and promising concepts will be evaluated. One concept will involve the installation of a suction patch on the inboard leading edge with the suction surface is vented to a low pressure area on the upper surface to establish a natural "vented" flow. A limited amount of laminar flow (1-2 percent chord) may be obtained on the upper surface (if the turbulence-contamination problem is resolved) which will provide some data for stability code calibration. The remaining goal in Phase I is to measure and characterize the acoustic disturbance environment incident on the aircraft wing. It is expected that the radiated pressure field from the fuselage turbulent boundary layer will be a major source of disturbances which could limit the extent of laminar flow achievable in Phases II and III, or require higher suction levels to ensure laminar flow is achieved.

Phase II- Leading-Edge Suction Panel

In this phase, the leading edge of the aircraft will be replaced by a fully active suction test article designed to maintain laminar flow to the front spar and beyond, depending on the extent of suction applied and the chordwise pressure distributions achieved at various test conditions. The leading-edge aerodynamic contour will be designed to produce a steep acceleration zone on the upper surface so that the minimum pressure is reached within a few percent chord, followed by a small deceleration zone downstream of the pressure minimum to generate a cross flow of opposite direction to that produced in the steep pressure acceleration zone. This reverse cross-flow generation concept was conceived by Pfenninger(2) to cancel out the remaining strong cross flow from the steep acceleration zone so that downstream only a modest suction level is required to control oblique TS type disturbance wave growth. A non-suction foam fiberglass glove fairing will continue the desired contour behind the front spar to ensure the proper pressure distribution is achieved to sustain some laminar flow beyond the leading edge. Achieving a significant amount of laminar flow (15-20% chord) will demonstrate control of the key flow phenomena in the leading edge, i.e., turbulence contamination and stabilization of strong cross-flow disturbance growth by suction. Surface pressure and laminar flow data obtained in this phase will be used to further calibrate the aerodynamic analysis and boundary-layer stability codes. These codes will then be utilized to design the extended suction panel to be tested in the next phase.

Phase III- Extended-Suction Panel

The same leading-edge suction panel used in Phase II will be joined with an extended active suction panel for Phase III. The combination of these two suction panels will enable achievement of laminar flow to 50-60% chord. The aerodynamic contour of the two panels will produce a steep acceleration zone on the upper surface leading edge (as discussed in Phase II above) followed by a flat rooftop pressure distribution over the remaining surface to be laminarized. Only Tollmien-Schlichting type disturbances will be present in this region so that a relatively low level of suction should be sufficient to maintain extensive laminar flow over the wing box. Operational sensitivities of laminar flow to suction levels and distributions, external surface pressure distributions, and manufacturing tolerances will be determined. Data obtained will permit validation of aerodynamic design and analysis codes and advanced stability codes, as well as refinement of design methodologies. The benefits accrued at the end of Phase III will include experience in the successful design and integration of a suction system concept into a supersonic vehicle, which will enable the industry to assess in greater depth the benefits of the application of laminar flow control to their HSCT concepts.

CONCLUDING REMARKS

After decades of research, laminar boundary-layer control technology has progressed to the point that applications may be seen in future subsonic transport aircraft. This technology has been quite elusive; the technical problems, both imagined and real, have been difficult to overcome. The potential benefits to commercial supersonic transports are compelling reasons to attempt to extend this technology to supersonic cruise speeds. The new NASA program will undoubtably unveil new phenomena or concerns, especially in the F-16XL flight research. But, this research should lay the ground work for the evaluation of the feasibility and practicality of suction laminar flow control for supersonic transport aircraft.

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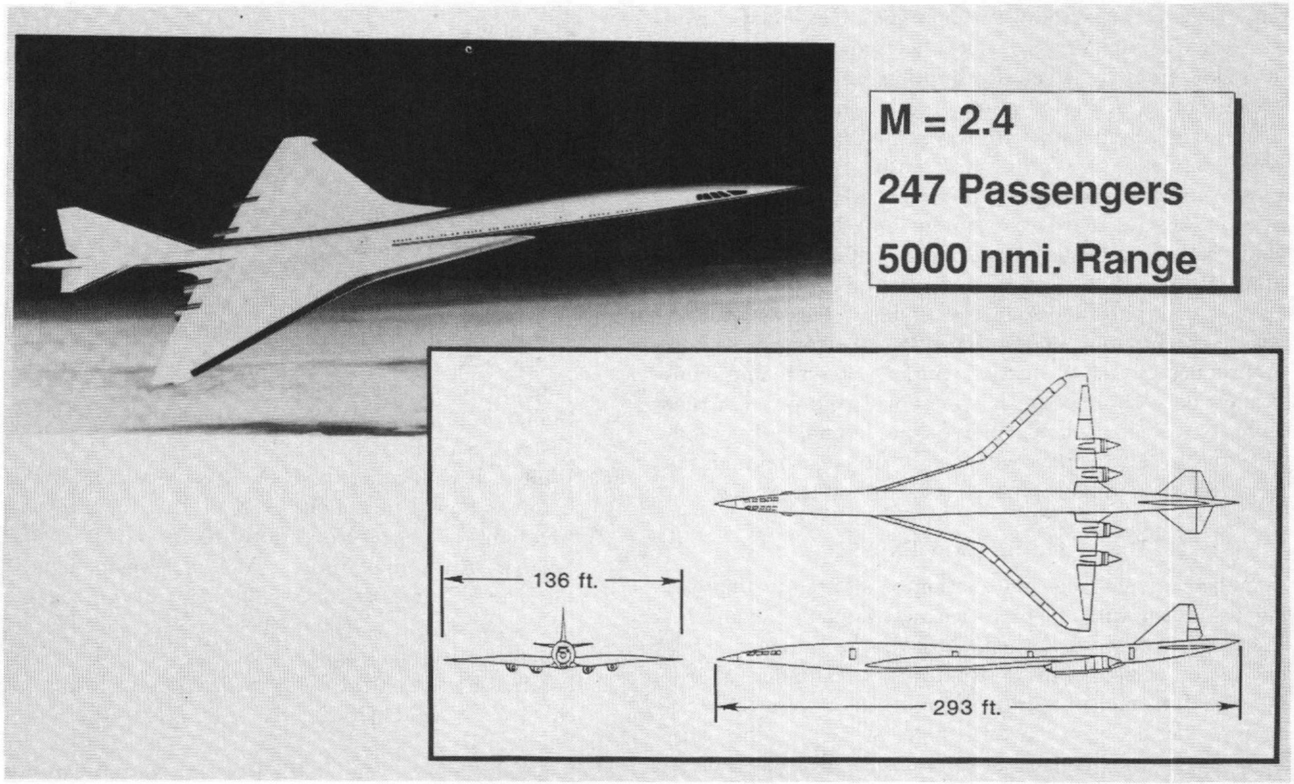


Fig. 1 BCA Baseline Supersonic Transport.

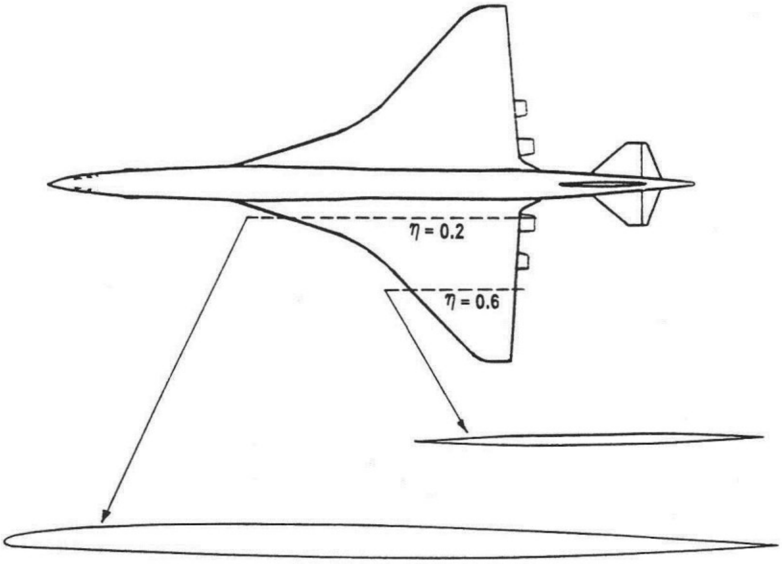


Fig. 2 Typical wing sections- BCA Baseline.

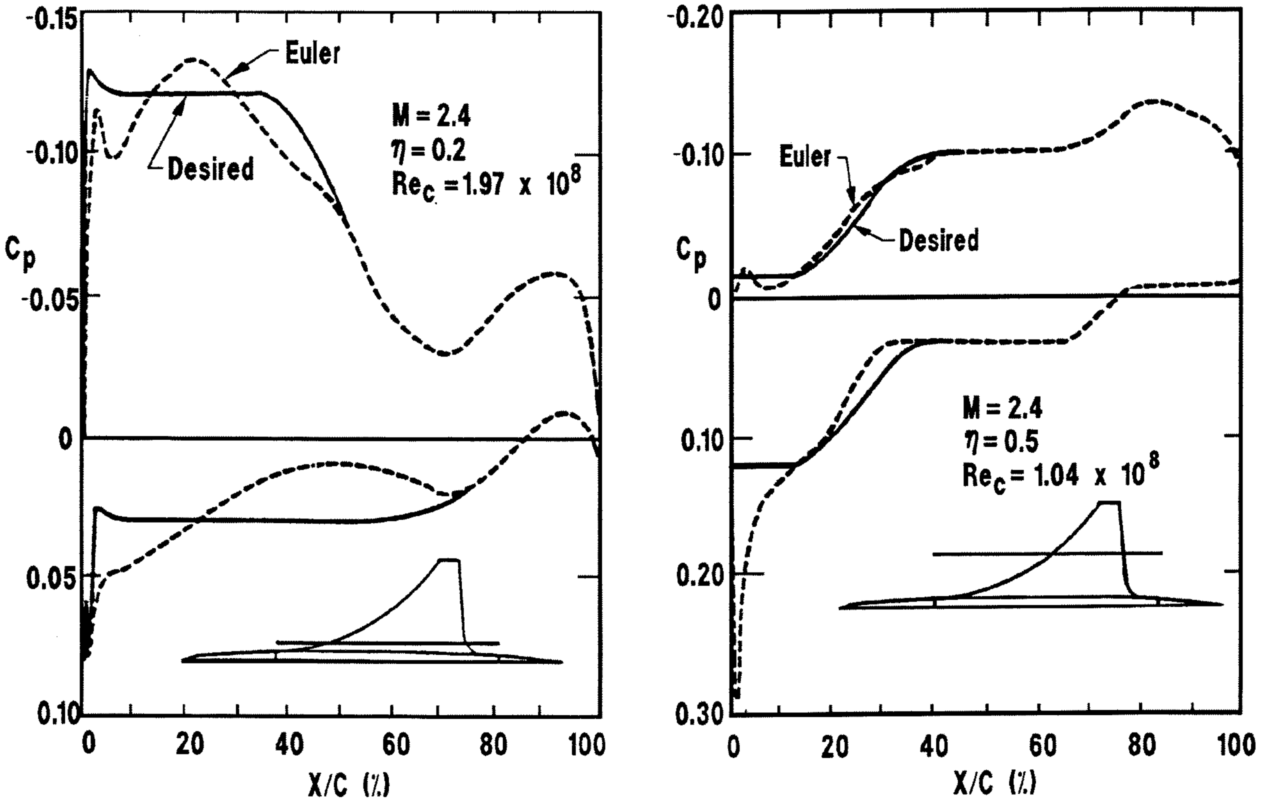


Fig. 3 Baseline versus desired LFC pressure distributions.

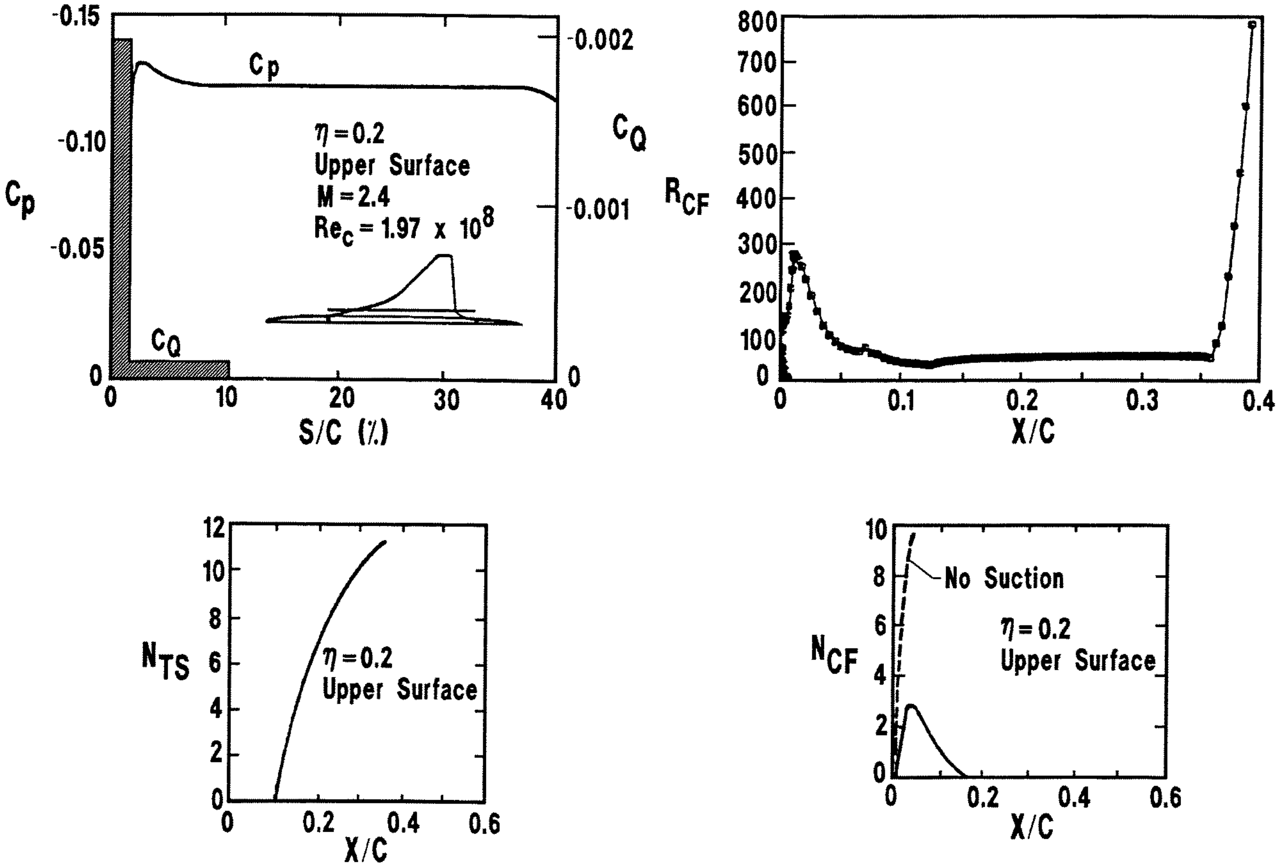


Fig. 4a Inboard wing analysis.

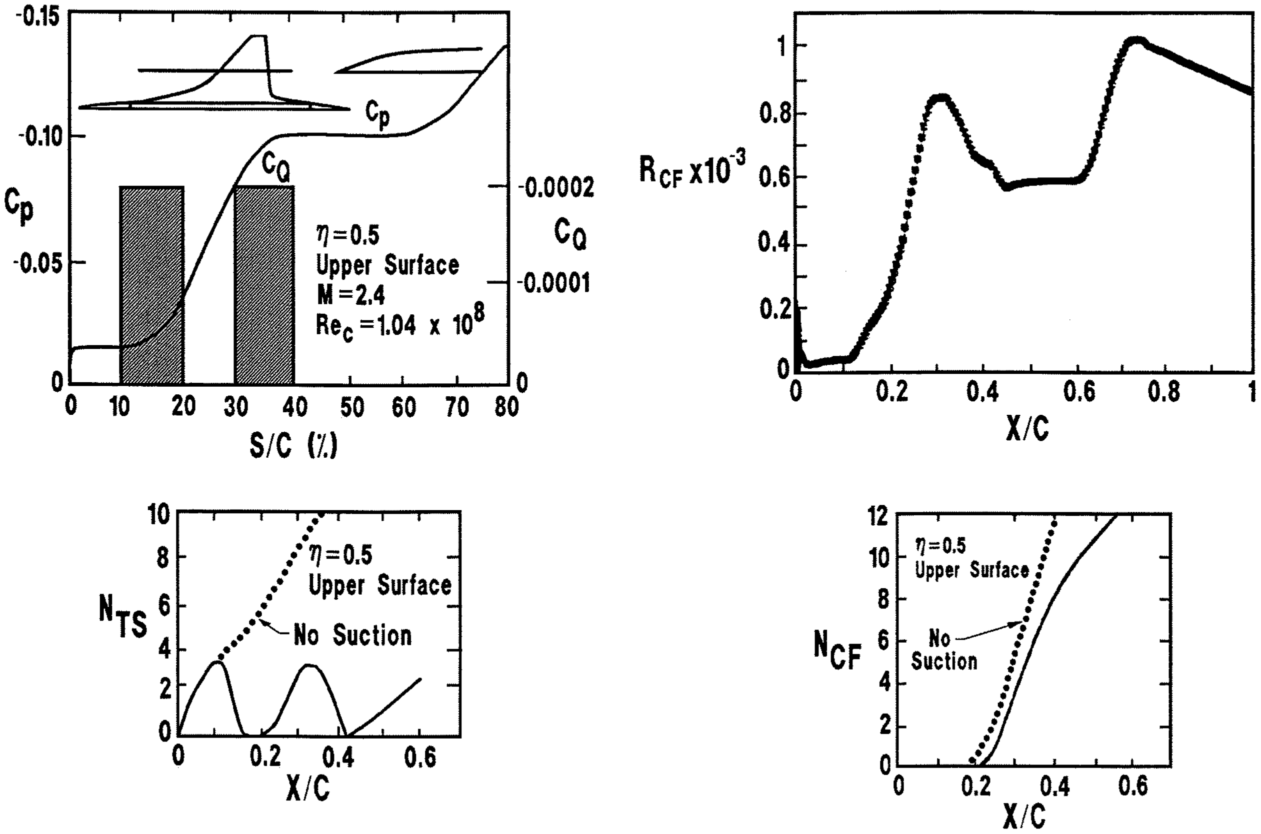
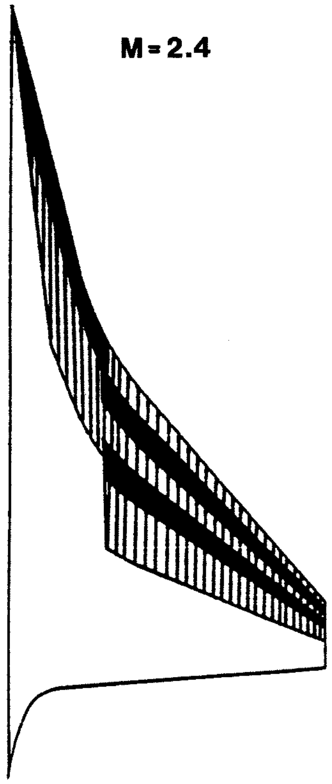


Fig. 4b Outboard wing analysis.

■ LAMINAR FLOW WITH SUCTION ▨ NATURAL LAMINAR FLOW



TURBULENT BASELINE
745,000 LB TOGW
323,000 LB OEW (ALL COMPOSITE)
5000 NMI RANGE
PROJECTED YEAR 2000 TECHNOLOGIES

HLFC RESULTS:

CRUISE DRAG BENEFIT: 8.2% REDUCTION IN TOTAL A/C DRAG

PENALTIES:
- 8600 LB SUCTION SYSTEM WEIGHT (2.7% OF BASELINE OEW)
- 1220 HP EXTRACTION FROM ENGINES (0.2% SFC PENALTY)
- 0.3% INCREASE IN C_D DUE TO SUCTION AIR MOMENTUM DRAG

NET BENEFITS:
- 8.5% REDUCTION IN TOGW (63,000 LB)
- 39,000 LB REDUCTION IN FUEL BURN (12.0% IMPROVEMENT IN FUEL EFFICIENCY)
- 6.2% REDUCTION IN OEW (20,000 LB)
- 25% REDUCTION IN FUEL HEATING RATE

Fig. 5 Results of design benefits studies.

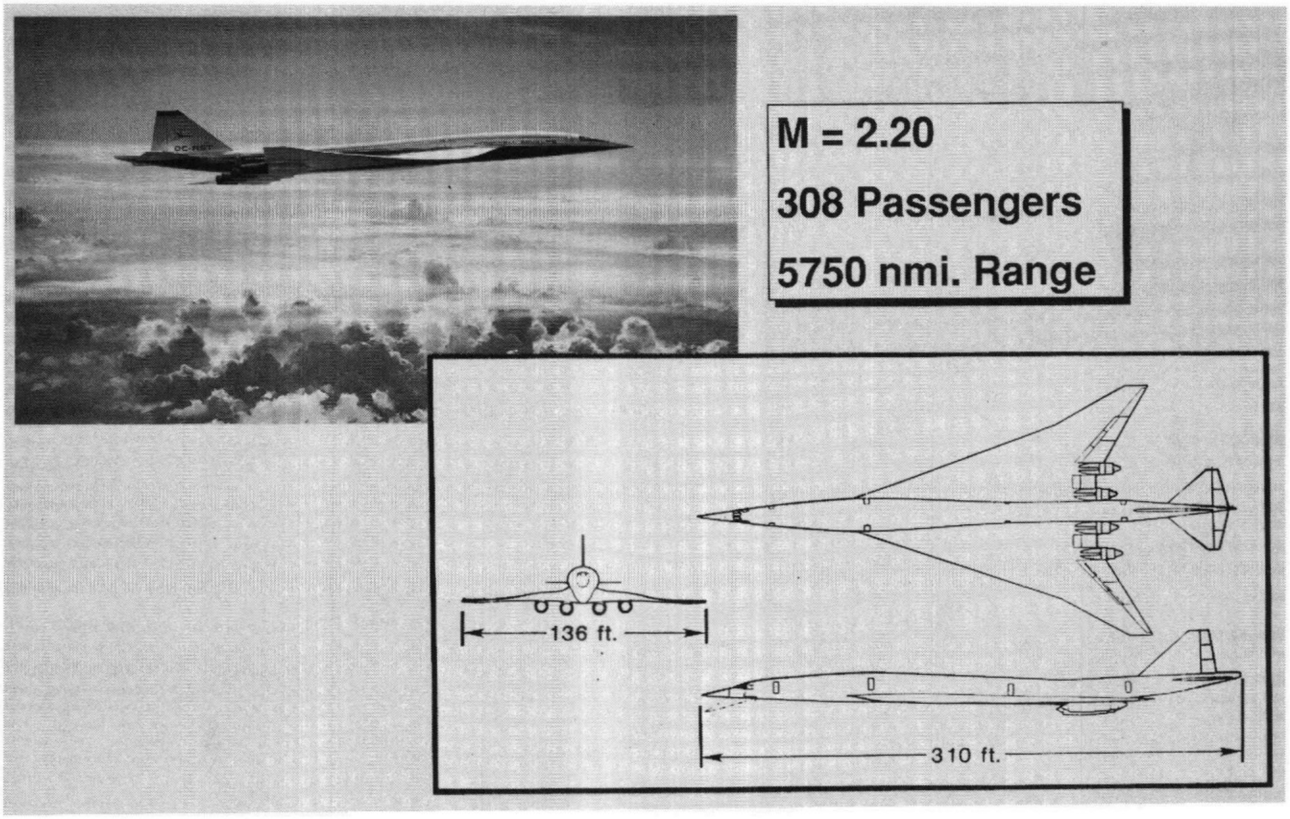


Fig. 6 DAC Baseline Supersonic Transport.

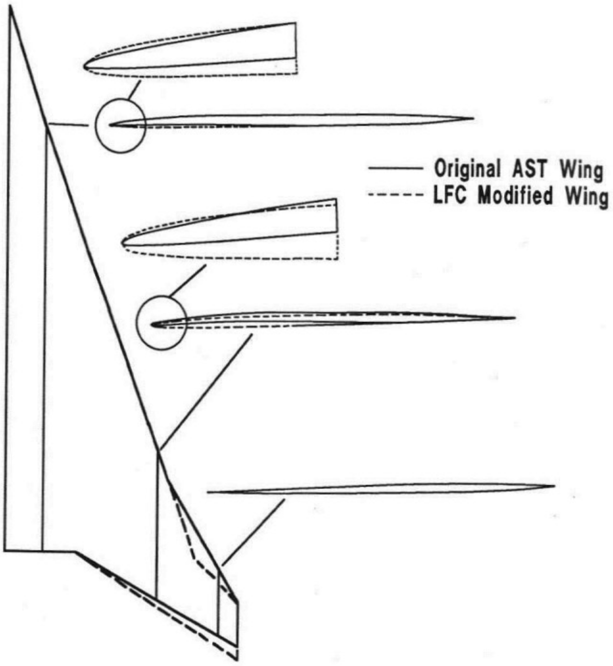


Fig. 7 Baseline versus LFC-modified sections.

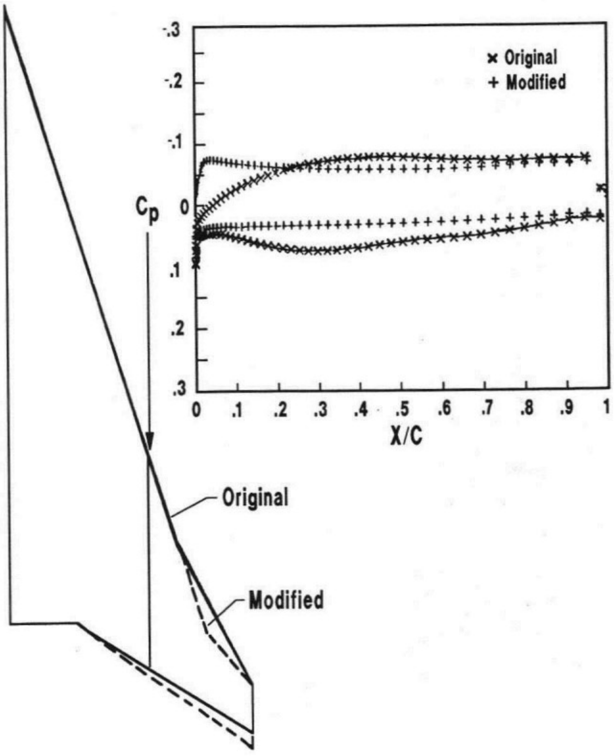


Fig. 8 Typical baseline and LFC-modified wing pressure distributions.

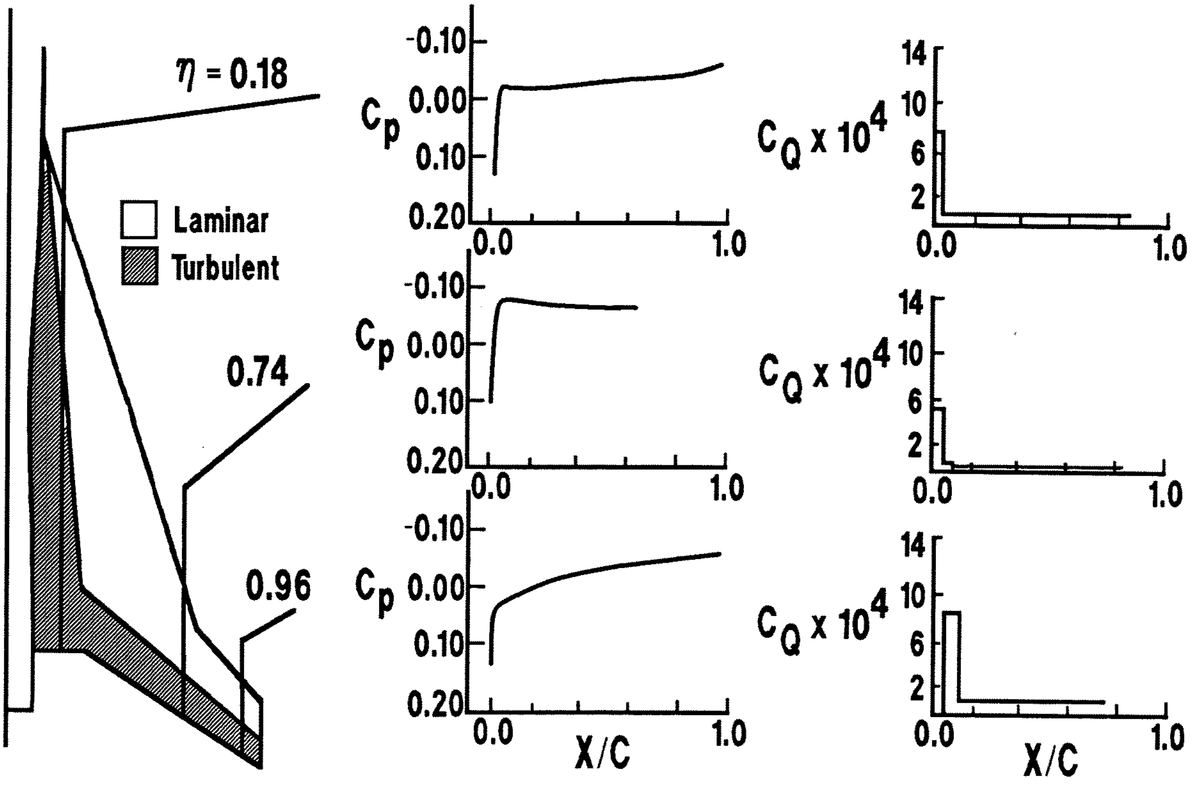


Fig. 9a Modified wing upper-surface pressure and suction distributions.

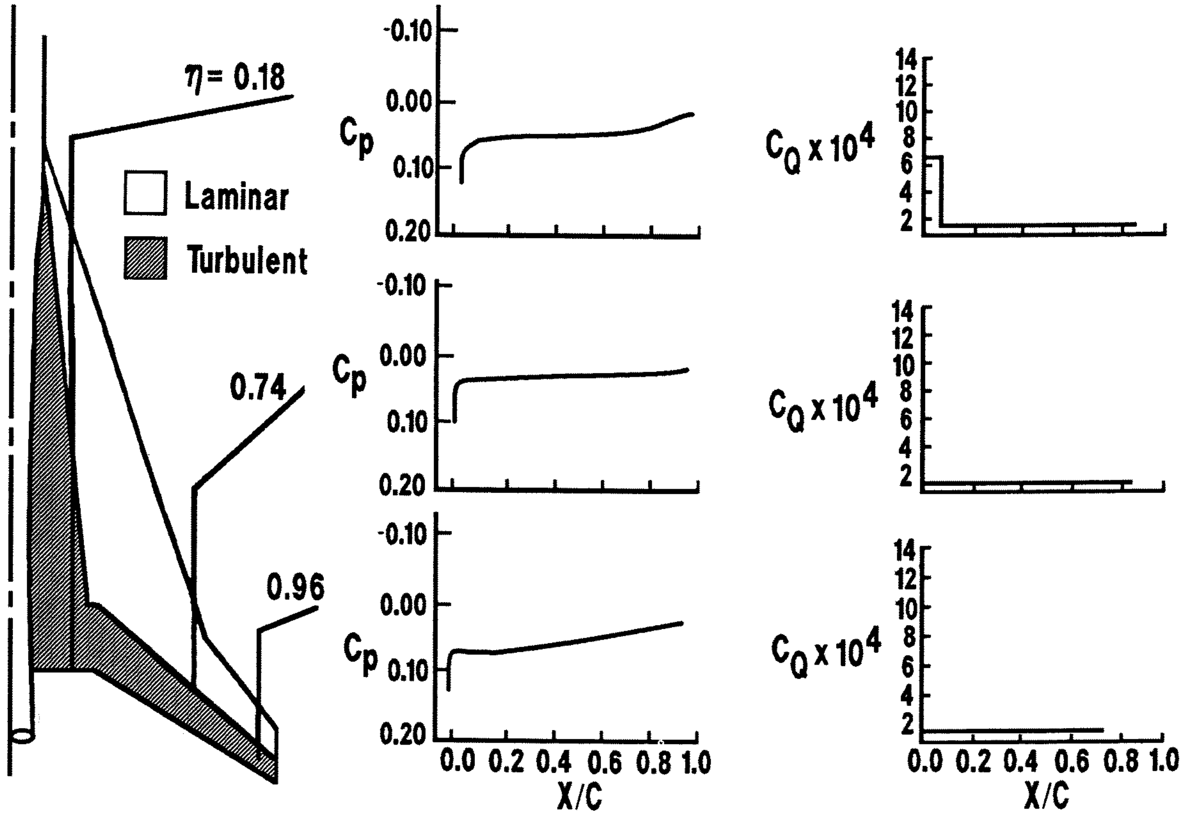


Fig. 9b Modified wing lower-surface pressure and suction distributions.

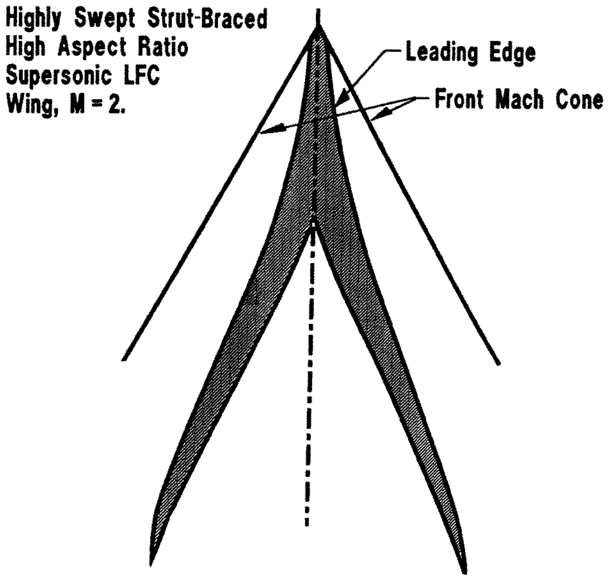


Fig. 10 Pfenninger SST wing configuration.

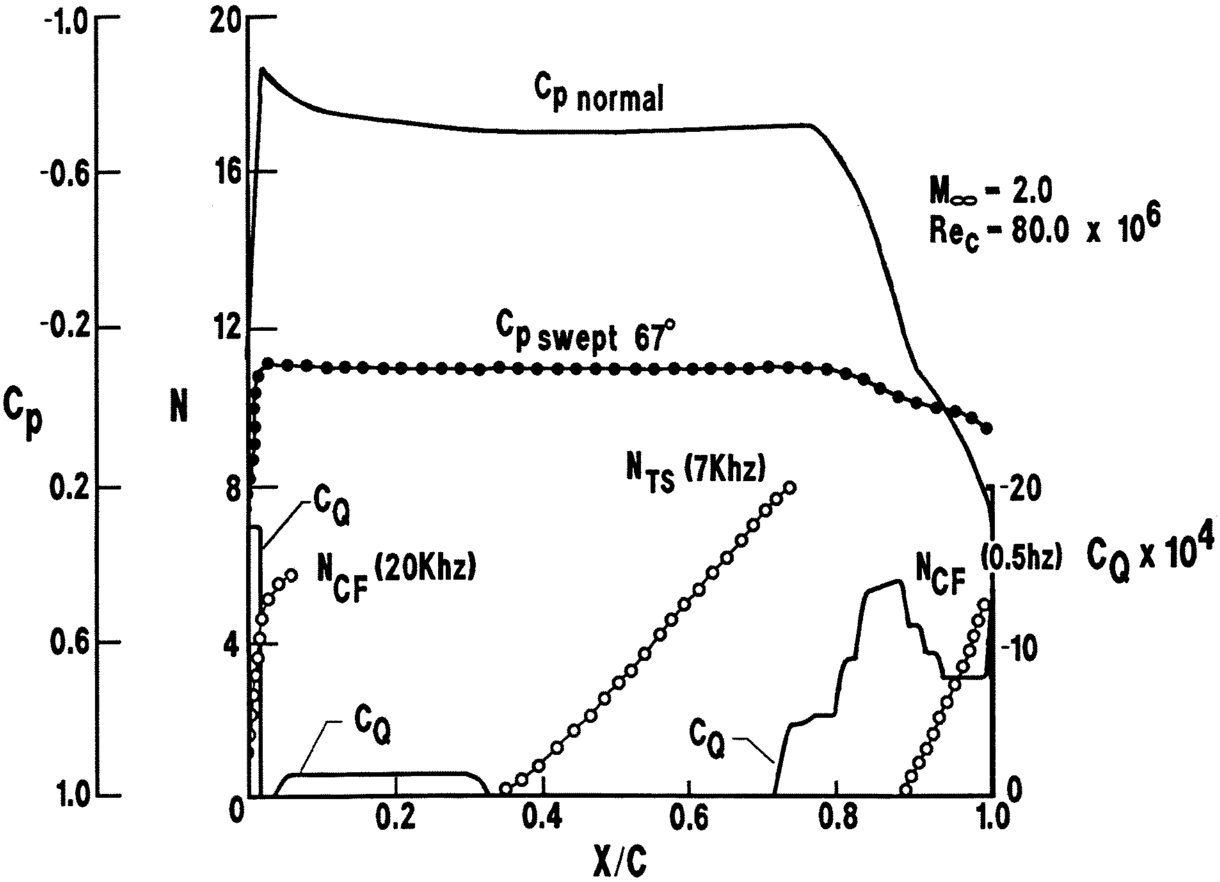


Fig. 11 Pfenninger wing- 67° yawed X-66 airfoil analysis.

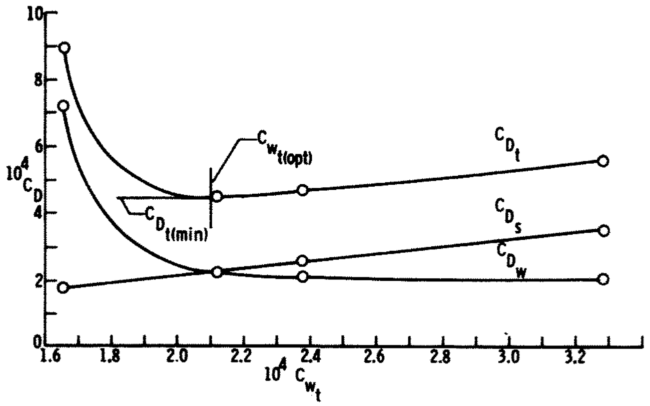


Fig. 12 Supersonic LFC plate- $C_{D,suction}$, $C_{D,wake}$, $C_{D,total}$ versus suction weight flow coefficient $C_{W,t}$ at $M=3$, $Re_c = 25.7 x 10^6$.

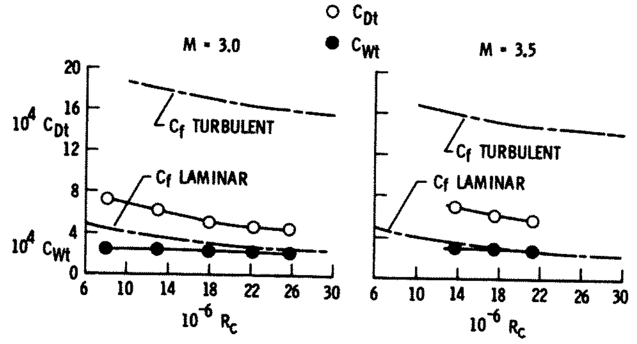


Fig. 13 Supersonic LFC plate- minimum total drag $C_{D,tot,min.}$, and optimum total suction weight flows $C_{W,t}$ at $M=3$ and 3.5 (c=40.2 inches).

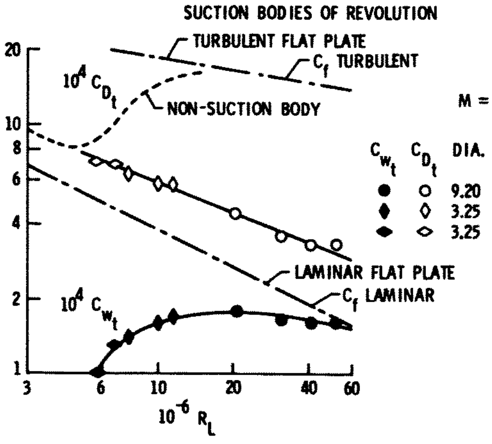


Fig. 14 Suction bodies of revolution- Minimum total drag and optimum total suction coefficients.

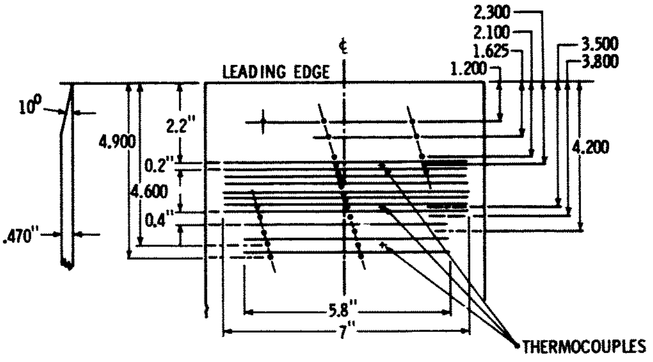


Fig. 15a Boundary-layer incident-shock-interaction experiments on a flat plate with suction slots.

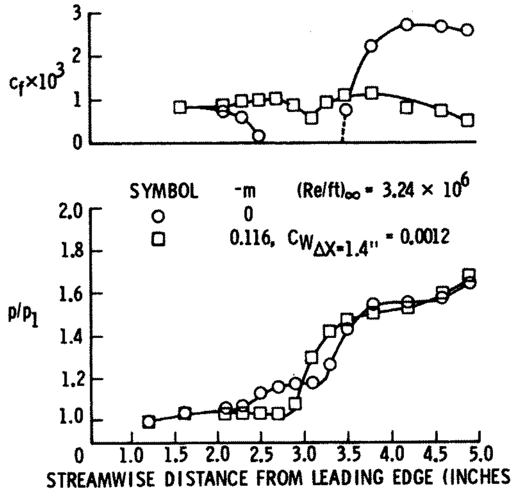


Fig. 15b Chordwise pressure and wall shear-stress distribution with and without slot suction on a flat plate with incident shocks.

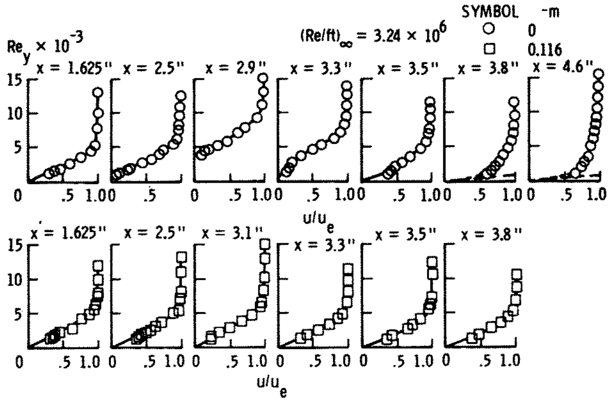


Fig. 15c Boundary-layer profiles on a flat plate with incident shock with and without suction.

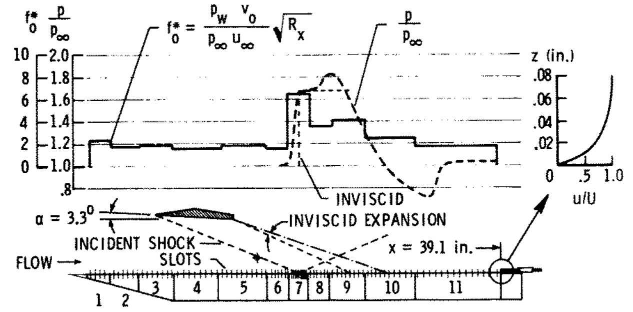


Fig. 16 Boundary-layer incident-shock-interaction experiments- Pressure and suction distribution and boundary-layer velocity profiles at X=39.1 inches, M=3, $R_x = 25.9 \times 10^6$, $C_{W,t} = 4.3 \times 10^{-4}$.

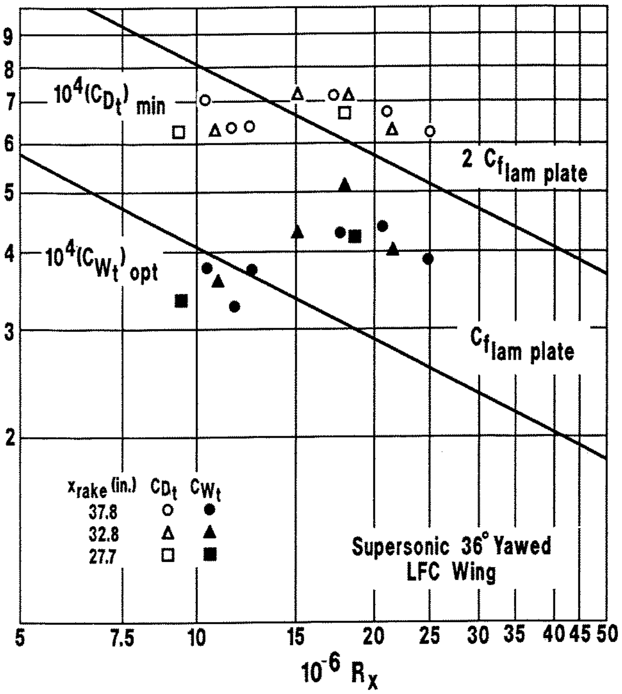


Fig. 17b Minimum total drag and optimum total suction coefficients versus length Reynolds numbers at M=3.0

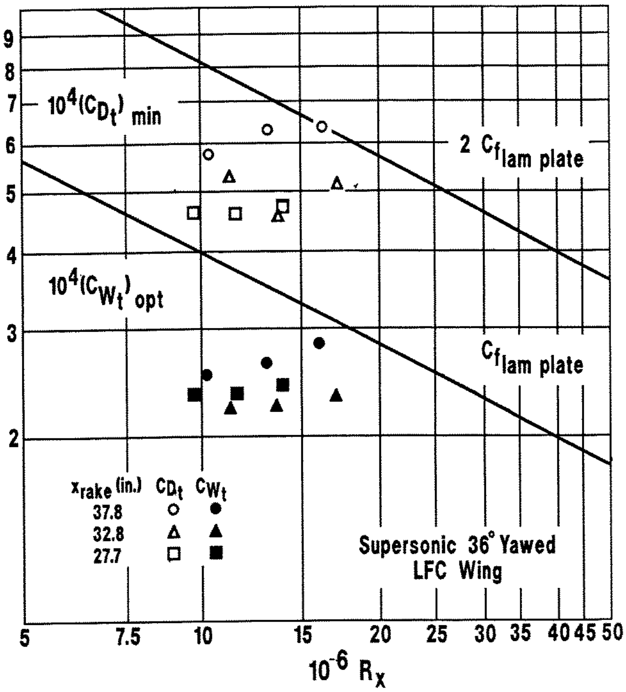


Fig. 17a Minimum total drag and optimum total suction coefficients versus length Reynolds numbers at M=2.5.

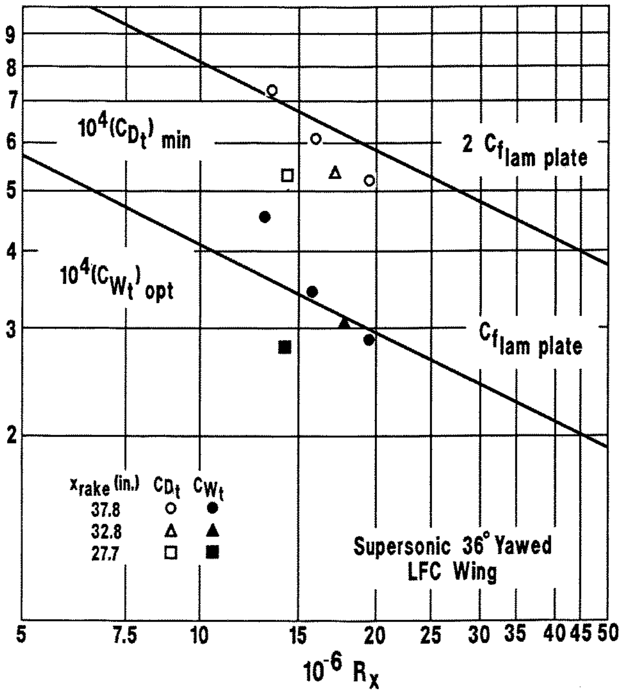


Fig. 17c Minimum total drag and optimum total suction coefficients versus length Reynolds numbers at M=3.5.

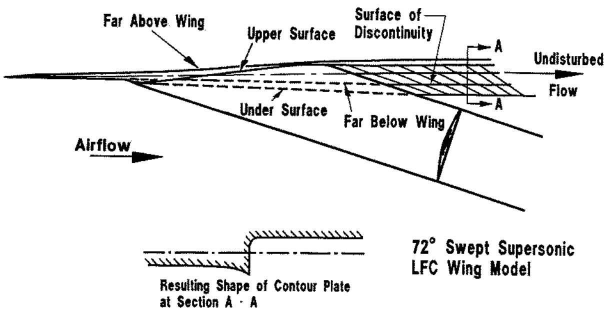


Fig. 18a Streamline paths on 72° swept supersonic LFC wing model.

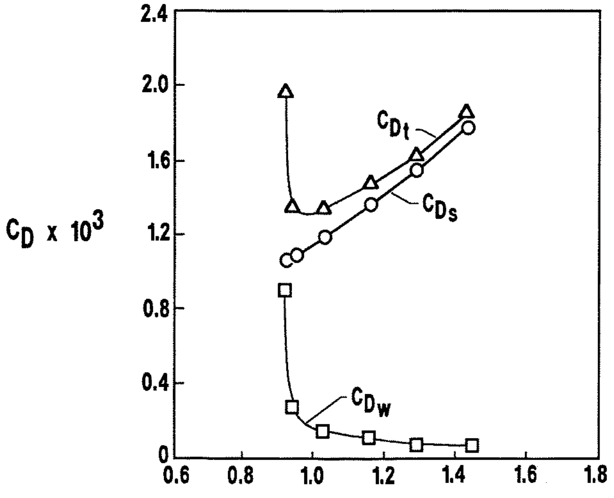


Fig. 18b Variation of drag components with suction coefficient M=1.99, $R_c = 7.3 \times 10^6$, $\alpha = 0.15°$.

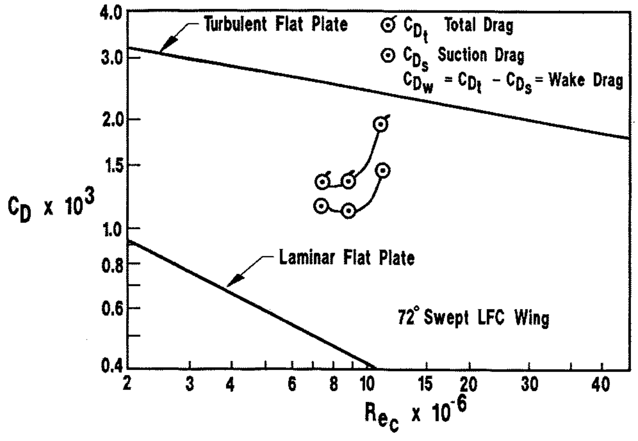


Fig. 18c Variation of corrected optimum drag coefficients for the upper surface with Reynolds number M=1.99, $\alpha = 0.15°$.

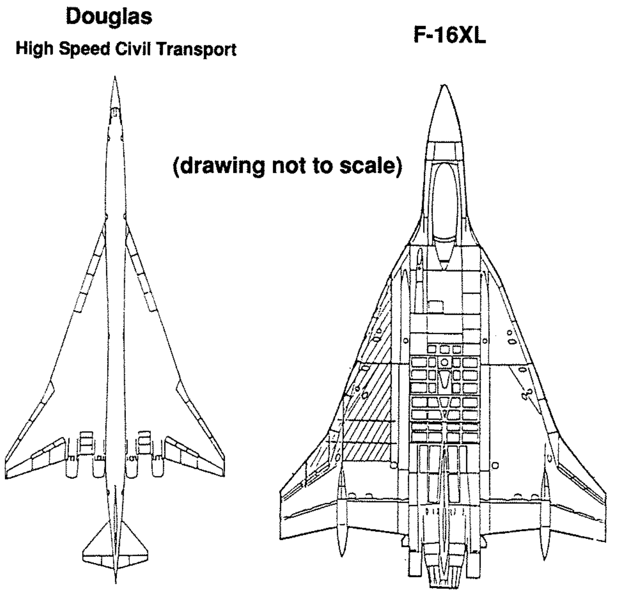
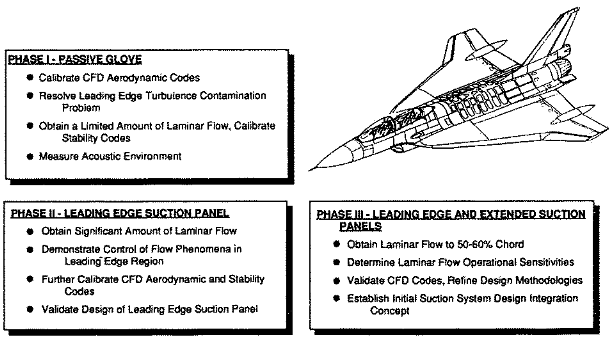


Fig. 19 Planform comparison of representative high-speed civil transport and the F-16XL.



PHASE I - PASSIVE GLOVE

- Calibrate CFD Aerodynamic Codes
- Resolve Leading Edge Turbulence Contamination Problem
- Obtain a Limited Amount of Laminar Flow, Calibrate Stability Codes
- Measure Acoustic Environment

PHASE II - LEADING EDGE SUCTION PANEL

- Obtain Significant Amount of Laminar Flow
- Demonstrate Control of Flow Phenomena in Leading Edge Region
- Further Calibrate CFD Aerodynamic and Stability Codes
- Validate Design of Leading Edge Suction Panel

PHASE III - LEADING EDGE AND EXTENDED SUCTION PANELS

- Obtain Laminar Flow to 50-60% Chord
- Determine Laminar Flow Operational Sensitivities
- Validate CFD Codes, Refine Design Methodologies
- Establish Initial Suction System Design Integration Concept

Fig. 20 F-16XL supersonic LFC experiment.