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LAMINAR-FLOW FLIGHT EXPERIMENTS

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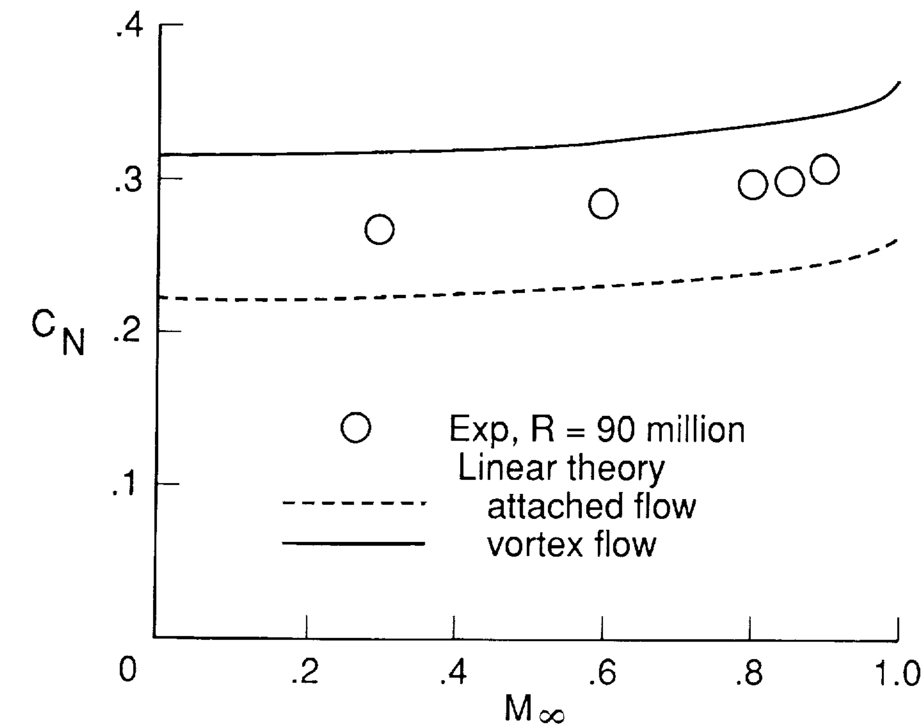
INTRODUCTION

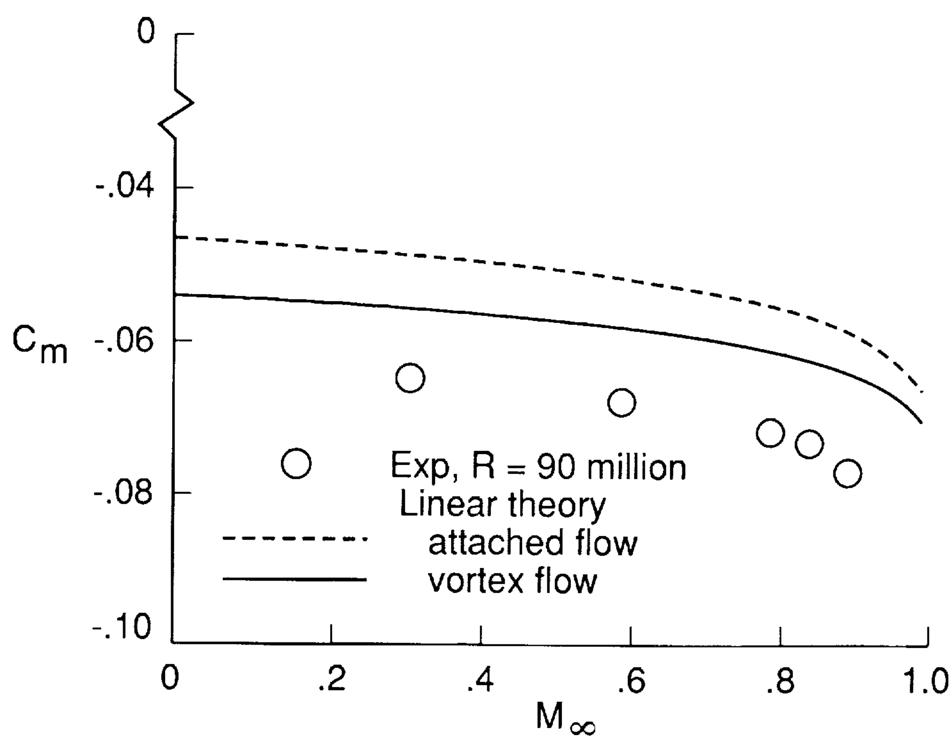
On December 17, 1937, B. Melvill Jones presented the first Wright Brothers' lecture at Columbia University in New York (ref. 1). His lecture, which was entitled "Flight Experiments on the Boundary Layer," dealt specifically with the first British flight observations of transition of the boundary layer from laminar to turbulent flow. These data, Jones concluded, showed that it is possible to retain a laminar layer over at least one-third of the whole wing surface even when the chord Reynolds number is as high as 8 millions. In the 50 years since this presentation, much flight research has been performed to explore the potential of laminar-flow control for drag reduction. Both passive control and active control by suction (designated as natural laminar flow and laminar-flow control, respectively) have been researched and impressive results achieved. The successes of the early natural laminar-flow (NLF) flight testing were remarkable, with the achievement of an extent of laminar flow and transition Reynolds numbers which were not to be exceeded in flight for over 40 years. Nevertheless, mid-century manufacturing capabilities were such that insufficiently smooth or wave-free wing surfaces led to failure of attempts to transfer this technology to practice. The experience with laminar-flow control (LFC) nearly paralleled that of NLF. LFC was recognized as a potentially more powerful means for achieving extensive laminar boundary layer flow, although admittedly more complex from the systems standpoint. LFC flight research began in the 1940's and peaked in the 1960's with the USAF/Northrop X-21 program, the most ambitious LFC flight test to date, which attempted to achieve full chord and full span laminar flow on a swept wing (ref. 2). Two WB-66 aircraft were fitted with new, full chord suction controlled laminar-flow wings and flight tested over 3 years. The main result of the program was that laminar-flow control was observed to be aerodynamically achievable, but surface quality and structural complexity still appeared formidable barriers to LFC applications.

For a period of almost 10 years, research in NLF and LFC was dormant. The energy crisis of the early 1970's revived interest in the technology and flight testing resumed. Today, the prospects for a practical technology are brighter than ever. We have a greater understanding of the phenomena involved and new, less-complex systems concepts are evolving. Critically important to this new outlook is the fact that our manufacturing capabilities have dramatically advanced since the 1960's and the needed wing surface quality appears within our reach.

In this paper, the flight testing conducted over the past 10 years in the NASA LFC program (ref. 3) will be reviewed. The LFC program has been directed towards the most challenging technology application, the high subsonic speed transport. To place these recent experiences in perspective, earlier important flight tests will first be reviewed to recall the lessons learned at that time.



(a) Normal force, $\alpha = 10°$.



(b) Pitching moment, $C_L = 0.3$.

Figure 13.- Theoretical estimates for compressibility effects.

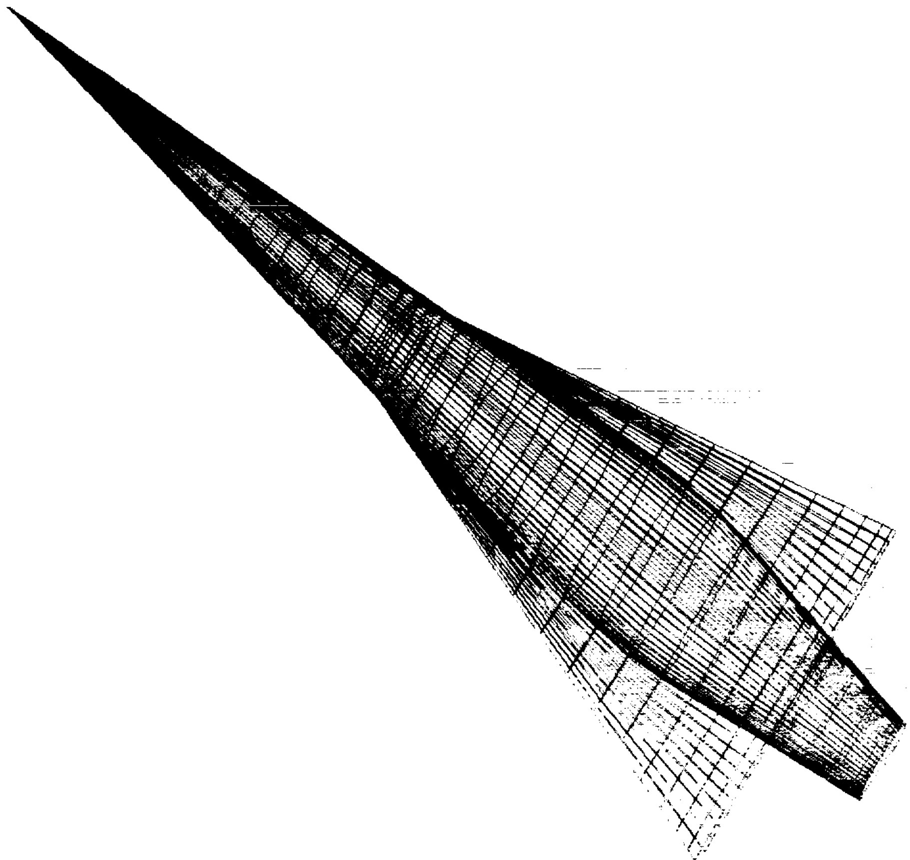


Figure 14.- Surface grid representation.

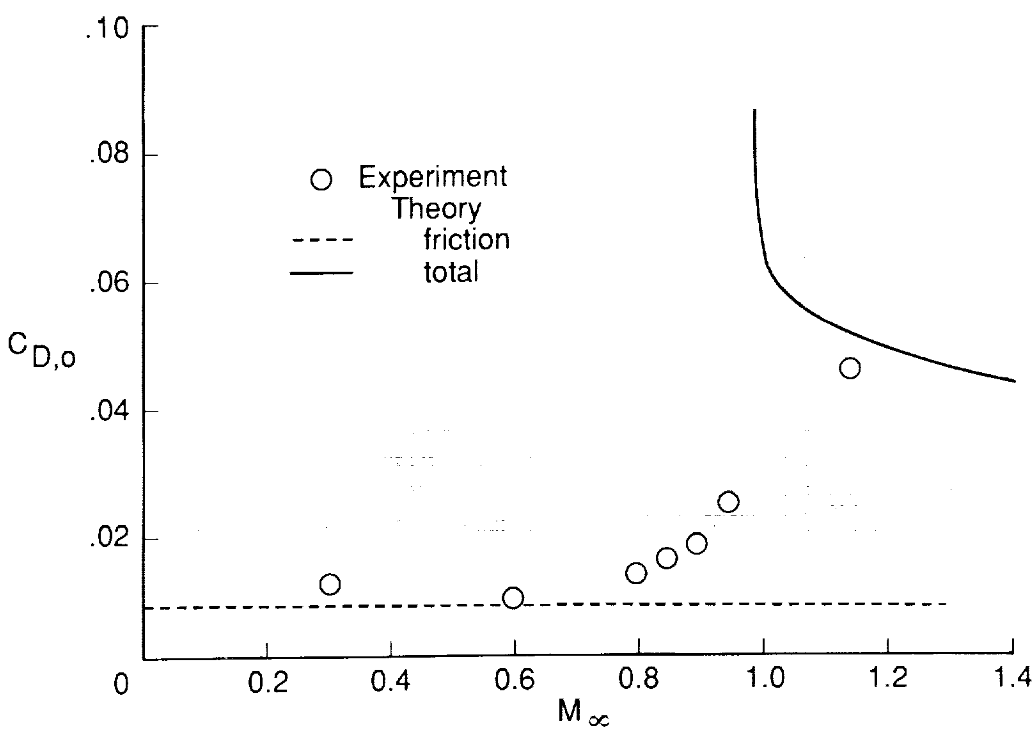


Figure 15.- Zero lift drag rise. R = 90 million.

ACRONYMS

AFWAL	Air Force Wright Aeronautical Laboratory
TACT	Transonic aircraft technology
NACA	National Advisory Committee on Aeronautics
NPL	National Physics Laboratory
LFC	Laminar-flow control
mac	Mean aerodynamic chord
LTPT	Low-Turbulence Pressure Tunnel
Max	Maximum
PW	Pratt and Whitney
NLF	Natural laminar flow
LE	Leading edge
OASPL	Overall sound pressure level
rpm	Revolutions per minute
dB	Decibel
N-OASPL	Normalized overall sound pressure level
Hz	Hertz
HLFC	Hybrid laminar-flow control

SYMBOLS

a	Speed of sound
$C_{\bar{p}}$	Pressure coefficient
$\bar{c}$	Mean chord
c	Chord
R_c	Reynolds number based on chord
R_T	Transition Reynolds number
R_x	Reynolds number based on x
U	Free-stream velocity
M	Mach number

x	Distance along chord in streamwise direction
ℓ	Liter
C_L	Lift coefficient
L	Lift
A	Area
M_{FAN}	Fan Mach number
N	Engine rpm
N_{TS}	N-factor for Tollmien-Schlichting (TS) calculation
N_{CF}	N-factor for crossflow (CF) calculation
P	Pressure
S	Surface distance along normal chord
t	Time

GREEK

ρ	Density
μ	Laminar coefficient of viscosity, or microns
Λ	Sweep angle
η	Span station
β	Sideslip

EARLY LAMINAR-FLOW FLIGHT RESEARCH

A brief review of some of the most significant past efforts on laminar flow is beneficial in understanding the needs for further research. Examples of some of the most pertinent flight tests will be presented to highlight the knowledge gained. In such a review, wind tunnel tests cannot be ignored because of the often elucidative impact they had.

The earliest known attempts to attain extensive regions of laminar flow in flight were made in the late 1930's and early 1940's. Both NLF with favorable chordwise pressure gradients and active LFC with boundary layer suction were investigated.

The B-18 flight test by the NACA in 1939 (ref. 4) was a major milestone in the development of NLF. Therein, an attempt was made to prolong the run of laminar flow to higher Reynolds number than had previously been achieved by flight testing a 17-foot chord, 10-foot span wooden NLF glove on the wing of the test aircraft (Figure 1). An exceptional effort was made to evaluate surface quality effects by working the wing to previously unattained smoothness and fairness. The flight test clearly displayed the importance of surface discontinuities and finish. In fact, the adverse effect of surface disturbances (surface waves, two-dimensional type steps, and three-dimensional type roughness) was the most pervasive factor observed in the early tests and continued to be the principal cause of limited laminar flow in most future flight and wind-tunnel tests. Although the severity of the surface disturbances was always shown to be aggravated by increased unit Reynolds number, it was not until considerable research (made possible by the development of the Langley Low-Turbulence Pressure Tunnel , LTPT) had been completed in the late 1940's and 1950's that an understanding of this phenomenon was developed, resulting in a quantitative ability to predict the magnitude of permissible three-dimensional disturbances (ref. 5). It was not until the 1960's that a quantitative ability to predict permissible waviness and two-dimensional type disturbances was developed.

The maximum transition Reynolds number, based on free-stream conditions and distance to the position of transition, attained in the B-18 flight tests was about 11.3 million with laminar flow to 42.5-percent chord on an NACA 35-215 section with only a 3-percent chord loss of laminar flow due to engine and propeller noise. This NLF transition Reynolds number was not to be exceeded in flight for over 40 years, until the NASA F-111/TACT NLF glove flight tests to be discussed later in this paper. The B-18 flight test was very encouraging in its time, because it indicated that the flight environment was possibly more benign for laminar flow than wind tunnels which until then had achieved laminar flow only at lower Reynolds number. The wind-tunnel tests were highly compromised because the higher unit Reynolds numbers of the tunnels exacerbated the roughness problem. Later wind-tunnel tests, in the quiet LTPT (ref. 6), suggested that the B-18 maximum length of NLF was constrained by the glove dimensions or achievable aircraft unit Reynolds number, and even higher transition Reynolds numbers might be obtained in flight. Indeed, the wind-tunnel experiments performed by Braslow (refs. 6 and 7) showed natural transition Reynolds numbers of 14 to 16 million for 6-series airfoils.

During World War II, several military aircraft were built with NACA 6-series airfoils, which were designed to achieve extensive natural laminar flow. Perhaps the most notable of these airplanes was the P-51 Mustang. But it is doubtful that much laminar flow was achieved on these aircraft because attention was not given to the surface quality that was required to maintain laminar flow. These aircraft flew in a harsh environment for obtaining laminar flow (i.e., at high speeds and low altitudes such that the unit Reynolds number was high) which placed stringent demands on surface smoothness and fairness. But after the war, attempts were made to see if NLF technology could be reduced to practice. The flight tests of King Cobra and Hurricane aircraft reported in references 8 through 10 are examples of such efforts (see figures 2 and 3). The King Cobra used production wing surfaces that were highly polished and filled to reduce waviness. The Hurricane employed an NLF section in a special, "low-drag construction" wing thought to be suitable for the maintenance of laminar flow. With highly polished surfaces good NLF performance was achieved on these aircraft, but underlying concerns with the practicality of the wing surface tolerances and maintenance defeated these efforts. Now, some 40 years later, the general aviation industry is just beginning to explore the use of NLF on aircraft for which the Reynolds number capability was more than demonstrated by the early NLF flight testing. Many general aviation aircraft now fly at higher altitudes, where unit Reynolds numbers are lower, and recent advancements in wing fabrication techniques now offer the possibility of routinely producing small aircraft with sufficient surface smoothness and fairness.

Active laminar-flow control with boundary-layer suction has been used in attempts to extend the laminar flow into the region of adverse chordwise pressure gradient, which is not possible to any appreciable degree with NLF. Suction through porous materials, multiple slots, and perforations were tried with various degrees of success (e.g. refs. 11-15). Three Vampire aircraft (figure 4 and refs. 12 through 14) with a number of suction surface configurations (continuous porous, perforated, and porous strips) and an F-94 aircraft (figure 5 and ref. 15) with suction through multiple slots were flight tested in the mid-1950's. The F-94 tests were very encouraging. With 69 slots between 41-to 95-percent chord, full-chord laminar flow to length Reynolds numbers of 36.4 million was obtained on the F-94. The addition of slots and suction in the favorable gradient (x/c less than 41 percent) was found to significantly broaden the lift-coefficient range for low drag with laminar-flow achievement. Laminar flow was lost behind shock waves on the F-94 when the aircraft speed was increased to the point where the local Mach number on the airfoil surface exceeded about 1.09. An important observation of the F-94 flight program was that the remains from insect impacts at low altitudes became subcritical at high speeds and altitudes above 20,000 feet for the boundary-layer flow of the unswept, F-94 wing. Unfortunately, this experience did not prevail in later flight test of swept wings, for which smaller critical roughness height has been observed in the regions of boundary-layer crossflow (ref. 16). The Vampire aircraft tests experienced unusual surface roughness difficulties. Continuous suction (from 6 to 98-percent chord) through a porous panel cloth (covered with nylon) or through 0.007 inch diameter perforations proved nearly as successful as the slotted F-94 surface, but each of these surfaces was thought to be impractical to manufacture and maintain. Two, "more practical" surfaces were tested, but with poor results. One incorporated porous strips of suction with sintered metal inserts; the other had perforations, 0.020 inch in diameter (the smallest holes then thought to be practical for manufacturing). The metal inserts caused surface discontinuities under flight loads, and the larger perforations caused transition by introducing unstable secondary flow in the boundary layer.

In the early 1960's, the most ambitious LFC flight program to date was undertaken by the Northrop Company. Under U.S. Air Force sponsorship, two WB-66 aircraft were modified with slotted suction wings and designated X-21 experimental aircraft (figure 6 and ref. 2). At the end of the program, full-chord laminar flow with suction was routinely obtained at Reynolds number of about 20 to 25 million. This was only after a long and difficult effort to improve performance through the systematic isolation and solution of problems, many due to wing sweep.

The most troublesome phenomenon encountered with the X-21 involved leading-edge turbulence contamination, a problem unique to swept wings. On the X-21, and at about the same time on a swept slotted-suction wing mounted vertically on the fuselage of a Lancaster bomber (figure 7 and ref. 14 and 17), the significance of this problem became apparent. Although previous small-scale wind tunnel and flight experimentation by the British (refs. 18 and 19) had indicated the existence of the spanwise turbulence contamination problem; its significance had gone unrecognized until the large scale flight tests. Subsequent flight and wind-tunnel tests indicated that leading-edge scale was a predominant factor and that proper treatment of the inboard wing leading edge could prevent turbulence contamination of the swept wing from disturbances that propagate down the wing leading edge along the attachment line (e.g. ref. 16). Although this phenomenon is now understood, it requires careful attention in the design of large LFC aircraft.

Another adverse effect of wing sweep on the ability to attain laminar flow had been found earlier during flights by the British with an AW52 airplane in 1951 (ref. 18) with a natural laminar-flow airfoil. A series of tests were performed where transition was shown to occur very close to the leading edge as a result of the formation of streamwise vortices in the laminar boundary layer. Later, an inability to obtain laminar flow in the last 20-percent chord of a Vampire trainer aircraft (ref. 14) was attributed to the forward sweep of the trailing edge. This sweep-induced boundary-layer instability was caused by the large crossflows resulting from strong, favorable or adverse chordwise pressure gradients on swept wings. Research prior to the X-21 program (ref. 20) showed that the proper application of suction is effective in controlling this crossflow instability; a result borne out in the X-21 flight tests.

Although structural flaws in the X-21 wing design produced surface waves and discontinuities that required liberal use of filler for smoothness, extensive laminar flow was routinely obtained at cruise altitudes of 40,000 ft. at Mach 0.75. A composite of the best wing surface laminar-flow performance is shown in figure 8 with remarkably good results obtained on the upper and, to a lesser degree, the lower wing surfaces. Flexing of the wing in flight continually deteriorated the surface quality due to the filler loss, but a series of 12 flights showed good repeatability even with major surface discrepancies on the last flight, figure 9. Nonetheless, the X-21 wing structure was just not good enough to provide the surface quality needed for a convincing demonstration that LFC was ready for application. The poor laminar-flow performance at lower altitudes and higher chord Reynolds number was undoubtedly due to the aggravated effects of poor surface quality at higher unit Reynolds numbers (figure 10). Still, maximum-length laminar-flow Reynolds numbers up to 45.7 million were observed in some areas.

Another phenomenon adverse to achieving laminar flow was also realized and investigated during the X-21 program. It was noted that flight through visible cirrus clouds, and sometimes very light haze, caused loss of laminar flow. At cruise altitudes, cirrus clouds are composed mainly of ice crystals; entrainment of the crystals in the boundary layer produced local turbulence leading to the loss of laminar flow (figure 11). Turbulent vortices shed by ice particles in the boundary layer were thought to trigger transition for certain combinations of particle size, concentration, and residence time in the boundary layer. At the termination of the X-21 program, concerns about this phenomenon and other unanswered issues on the operation of LFC aircraft were high.

In summary, when interest in laminar-flow technology was rekindled by the energy crisis in the early 1970's, the fundamental aerodynamic concepts of both passive and active laminar-flow control had been well established, verified in wind-tunnel tests and demonstrated in various flight tests. The aerodynamics of the technology appeared to be well in hand. Laminar flow to Reynolds numbers up to 16 million had been observed on two-dimensional NLF sections, and it was not clear that an upper bound on the transition Reynolds number had been reached. Suction control had been demonstrated for boundary layers in adverse pressure gradients and on swept wings at Reynolds numbers well above 16 million; specifically, full chord laminar flow to about 36 million chord Reynolds for the former and 46 million for the latter. Yet, no practical application had been made with any suction method. The ability to manufacture and maintain aircraft surfaces with admissible tolerances, considerably smaller than required for turbulent aircraft, and at acceptable cost was still viewed as a formidable challenge. Neither suction slots nor perforations could be manufactured economically within required tolerances, and the latter were believed to generate disturbances that adversely affected the ability to attain large length Reynolds numbers. Criteria for the proper design of slots were greatly improved during the X-21 flight program. With respect to perforated surfaces, early research indicated the need for hole diameters smaller than could be practically fabricated at that time. Porous surfaces with the required structural characteristics and aerodynamic smoothness were not available.

Over the past decade, NASA and the aircraft industry have launched programs to continue the development of this technology and to provide the information needed for objective decisions on its application to new aircraft. The flight tests reviewed in this paper have been an integral part of those efforts.

F-111/TACT NLF GLOVE FLIGHT EXPERIMENT

The NACA 6-series airfoils were originally developed for low-drag, NLF applications. In actuality, these airfoils were used on many of the early jet aircraft because they had very good performance as turbulent airfoils. However, modern supercritical airfoil technology has since led to improved airfoils with greatly enhanced turbulent performance (i.e., drag divergence Mach number, thickness ratio and lift coefficient capability). For this reason, in the late 1970's, the Boeing Company designed a new supercritical, NLF airfoil in a NASA contract study to evaluate NLF for transport aircraft applications (ref. 21). With the Boeing airfoil as a starting point, a new supercritical NLF airfoil was designed at the Langley Research Center and flight tested at the Ames-Dryden Flight Research Facility on the F-111/TACT aircraft (refs. 22-24). The objective of the flight test program was to investigate natural laminar flow at transonic speeds.

A supercritical, natural laminar-flow airfoil glove was installed on the right wing panel of the F-111/TACT aircraft (figure 12). The glove was made of fiberglass skins with an inner core of polyurethane foam and bonded to the metal wing skin. For symmetry, an uninstrumented glove was also installed on the left wing panel. The glove had a 6-foot span, a 10-foot chord, and was finished to "sail-plane" quality. The glove airfoil design pressure distribution (figure 13) had a favorable gradient that extended to about 70-percent chord on the upper surface ($dCp/dx/c = -0.4$) and to about 50-percent chord on the lower surface ($dCp/dx/c = -0.8$). The airfoil design lift coefficient was 0.5 at a Mach number of 0.77 and a Reynolds number of 25 million. On the upper surface at this condition, supersonic flow extended from about 20-percent chord to 70-percent chord where the favorable gradient terminated in a weak shock. The glove was installed on the airplane to achieve the design pressure distribution at 10 degrees of leading-edge sweep (figure 12); however, wind-tunnel tests had indicated that the pressure distributions at the higher sweep angles (up to 26 degrees) were acceptable for obtaining transition data at these conditions (i.e., no leading-edge peaks or premature adverse gradients). In hindsight, the low design sweep angle of the glove was very conservative, but at that time, some studies had been very pessimistic regarding the amount of laminar flow that could be obtained at even moderate sweep angles and Reynolds numbers approaching 30 million (ref. 21).

Results from the flight-test program (ref. 24) indicate that the maximum extent of laminar flow was about 55-percent chord on the upper surface at 10 degrees of sweep for a chord Reynolds number of 28 million. However, as the wing sweep was increased to 26 degrees, the transition location moved forward to the 10 to 20-percent chord range (figure 12). On the lower surface at 28 million chord Reynolds number, the maximum extent of laminar flow was about 50-percent chord (the start of the adverse gradient) and this was achieved to sweep angles as high as 15 degrees.

The wind-tunnel pressure distributions on the glove upper surface were much smoother than those obtained in flight (figure 13), particularly at the higher sweep angles. Although the majority of the wind-tunnel results have not been published, stability analyses are presented for five cases in reference 25. Based upon the wind-tunnel pressure distributions, these analyses predicted transition locations significantly further aft than those measured in flight on the upper surface. The irregularities in the flight upper-surface pressure distributions, which led to premature transition, were apparently caused by shocks propagating onto the glove from the inboard wing and not by surface waves in the glove skin. In retrospect, the 6-ft. span of the glove was too small to isolate the glove from the flow on the remainder of the basic F-111/TACT wing over a broad range of conditions. Even for the design point at 10 degrees of sweep, there was a weak shock wave on the glove near 55-percent chord that limited the extent of laminar flow to this point instead of further aft near the pressure minimum (figure 13). Since the lower-surface flow was subcritical, the lower-surface flight pressures were much smoother than those obtained on the upper surface, and in several cases laminar flow was obtained to the pressure minimum (approximately 50-percent chord). However, the steeper favorable gradient on the lower surface (figure 13) was not suitable for achieving large runs of laminar flow at the higher sweep angles because of increased crossflow instability.

The F-111/TACT NLF experiment was brief, and consequently the transition data were very limited. However, the results were very encouraging. The maximum transition Reynolds numbers of about 15 million on the upper surface for 10 degrees of sweep, and 14 million on the lower surface at 15 degrees of sweep were significantly higher than values obtained in previous NLF flight tests. The closest comparable flight test had been conducted over 40 years earlier on the B-18 bomber previously discussed. During that test, a maximum transition Reynolds number of about 11.3 million was obtained.

F-14 VARIABLE SWEEP TRANSITION FLIGHT EXPERIMENT

Since the F-111/TACT NLF glove pressure distributions had not been designed to minimize crossflow at the higher sweep angles, and since the maximum extent of laminar flow on both the upper and lower surface was determined by adverse pressure gradients, even larger transition Reynolds numbers at moderate sweep angles seemed possible. In addition, the techniques for fabricating and bonding large and very smooth foam and fiberglass test surfaces or gloves to metal wings had been developed and proven acceptable for flight testing. Consequently, the F-111/TACT NLF experiment paved the way for a follow-on program that could provide a much broader transition data base.

The F-14 Variable Sweep Transition Flight Experiment was initiated in 1984 (refs. 26 and 27)) with flight tests being completed in 1987. These tests were conducted with an F-14 (figure 14) on loan to NASA from the Navy. Obtaining transition data was the primary objective of the program - not airfoil design verification. Therefore, only the upper surface of the wing was gloved in order to provide a laminar-flow test surface. The gloves extended from about 10-percent chord on the lower surface to about 60-percent chord on the upper surface (spoiler hinge line) and covered the majority of the variable-sweep outer panel (figure 15). Four rows of flush static pressure orifices and three arrays of hot-films were distributed along the span for determination of the local wing pressure distributions and transition locations. These data and the other associated flight parameters were monitored in real time on the ground during all the testing.

Two gloves were flight-tested during the program: one was a "clean-up" or smoothing of the basic F-14 wing (modified NACA 6-series airfoils), while the second involved significant contour modifications to the basic F-14 wing. The second glove, designed at NASA Langley (ref. 27), provided more moderate favorable pressure gradients than the "clean-up" glove, and achieved more of a two-dimensional type flow (straighter isobars) over a larger part of the span. Both gloves were constructed of fiberglass skins with an inner core of polyurethane foam (ref. 28). Measurements taken on the gloves with a mechanical deflection gauge having support feet two inches apart indicated wave amplitudes no larger than 0.002 in. Representative pressure distributions at several Mach numbers are presented in figures 16 and 17 for both gloves. The Langley glove design provided a wide variety of pressure distributions with different favorable gradients to about 50-percent chord over a broad Mach number range (0.6 to 0.8).

The transition location at 0.7 Mach number on the Langley-designed glove is presented in figure 18 as a function of wing sweep for altitudes of 20,000 ft. and 30,000 ft. Transition locations for the "clean-up" glove are presented in reference 27 and for the F-111/TACT NLF glove in figure 12 of the present paper. However, to compare various transition or laminar-flow experiments, transition Reynolds number is a more appropriate parameter for comparison than just transition location. Therefore, the maximum transition Reynolds number observed in several of the more significant flight and wind-tunnel experiments are presented in figure 19. For this figure, transition Reynolds number is based on free-stream conditions, rather than local conditions. In addition to the F-111 and F-14 experiments, included in figure 19 are results from several natural laminar flow tests: the B-18 flight test (ref. 4); the King Cobra flight test (refs. 8 and 9); a T-33 flight test (unpublished data from the Boeing Company); the 757 NLF glove flight test conducted by the Boeing Company (refs. 29 through 31); and low-speed wind-tunnel tests conducted in the 12-Foot Tunnel at NASA-Ames Research Center (ref. 32) and the LTPT at the Langley Research Center (refs. 6 and 7). Prior to the F-111 and F-14 flight tests, the highest NLF transition Reynolds numbers for airfoils or wings had been obtained in the LTPT at Langley and the 12-Foot Tunnel at Ames. These are very quiet tunnels and only until recently have airplanes (i.e., jet-powered aircraft) been able to match the Reynolds number capability of these facilities. More importantly, very few large aircraft have had the capability of providing large runs of laminar flow.

As previously discussed, results from the F-111/TACT NLF Glove Experiment had exceeded the prior maximum values for natural laminar-flow transition Reynolds numbers that had been obtained in flight on the B-18 and King Cobra. Results obtained during the F-14 VSTFE indicate maximum transition Reynolds number values exceeding F-111/TACT and wind-tunnel values up to 30 degrees of sweep. For the F-14 VSTFE, a maximum transition Reynolds number of about 17.6 million was obtained at 15 degrees of sweep, 13.5 million at 20 degrees, and 12 million at 25 degrees. Beyond 25 degrees of sweep, maximum transition Reynolds number decreased rapidly to about 5 million at 35 degrees of sweep. It should be pointed out that for all the maximum transition Reynolds number cases on both the F-111 and F-14, the amount of laminar flow was limited by either adverse pressure gradient or shock wave location. This suggests that even higher transition Reynolds numbers are possible in flight.

In comparison to the NLF tests, as would be expected, maximum transition Reynolds numbers for most suction or laminar-flow control experiments are much higher. As previously discussed, transition Reynolds numbers of about 30 to 36 million were obtained in flight on the Vampire and F-94, and a value of about 46 million was obtained in a small area of the X-21 wing. However, with suction only in the leading-edge region of swept wings, the transition Reynolds number for natural laminar-flow designs can be significantly increased — possibly doubled. This concept, called hybrid laminar-flow control (HLFC), is discussed later in the paper.

757 WING NOISE SURVEY AND NLF GLOVE FLIGHT TEST

In 1985, under a NASA contract, the Boeing Company performed a flight test to measure the acoustic environment in cruise on the wing of a 757 aircraft with a view towards the determination of the potential effects of the acoustic environment on boundary-layer transition (refs. 29, 30, and 31). Prior to this flight test, there were no extensive measurements of the noise environment on the wing of a commercial transport with wing-mounted, high-bypass-ratio turbofan engines. Engine noise concerns had led to conservatism in LFC aircraft design studies, with designs restricted to aft engine placement with a potentially severe adverse impact on performance and a degradation of LFC fuel savings potential. A major part of the 757 flight test was an attempt to achieve a limited amount of laminar flow over the wing and measure the impact of the engine noise intensity on the extent of laminar flow. Although the primary goals differed, an interesting parallel exists between the 757 and the B-18 tested some 45 years earlier. As with the B-18, the 757 experiments yielded encouraging results with regard to engine noise effects on laminar flow.

Boeing removed a leading-edge slat on the 757 wing just outboard of the starboard engine and installed a 10-foot span, NLF glove constructed of dense foam with a fiberglass epoxy overlay to produce a smooth, nearly wave-free surface (figure 20). The glove was designed to achieve laminar flow on both the upper and lower surface, with 20-to 30-percent chord laminar flow expected without adverse engine noise effects. This anticipated result was made possible in part by unsweeping the wing to 21 degrees at the glove location and by the favorable pressure distribution over the wing. A single microphone was installed on the glove leading edge and eight others were installed on each of the wing surfaces (upper and lower) — three on the glove and five distributed over the remaining wing surface (figure 20). Hot films were used to detect the transition front on the upper and lower surfaces. Measurements were made over cruise altitudes of 25 to 41 thousand feet at Mach numbers of 0.63 to 0.83. The starboard engine was throttled from maximum continuous thrust to idle at cruise speeds and altitudes.

Typical measurements of the overall sound pressure level (OASPL) at the design cruise condition are shown in figure 21. Generally, the OASPL is lower on the upper surface of the wing with measurements ranging from 111-to 131-decibels. On the lower surface, the measured levels ranged from 121-to 136-dB. Collectively, the acoustic data presents a rather confusing picture, undoubtedly due to the broad range of flight conditions and changing phenonmena that influence the OASPL's on the wing. Additional uncertainty is due to the acoustic instrumentation because the magnitude of probe interference and self noise generation are difficult to assess. However, some important observations that have strong implications in laminar-flow wing designs were made.

Some order in the acoustic data is achieved by normalization of the OASPL's with the ambient pressure over the altitude range of the test conditions. These data are shown for two microphone locations in figure 22. Normalized OASPL's are presented for a microphone on the glove and one aft of the glove for various flight Mach numbers and engine power settings; the latter reflected by fan exhaust Mach number variations. Flight Mach number and engine power setting effects are measureably different on the upper and lower wing surface. The lower surface normalized OASPL's show a strong dependence on engine power setting with about a 20 dB. increase occurring from engine idle (fan Mach number equal to 0.7) up to maximum continuous thrust (fan Mach number equal to 1.28) when the lower surface acoustic characteristics seem dominated by engine noise. Engine power setting has little influence on the OASPL's on the upper surface, but significant variations occur with flight Mach number. The wing appears to effectively shield the upper surface from radiated engine noise and the dominant noise sources are presumed to be of aerodynamic origin. The data present strong evidence that the wing upper-surface flow field has a major influence on the radiated acoustic field, particularly at higher cruise speeds when shock waves occur on the wing. The supercritical flow over the upper surface inhibits forward radiation of sound from downstream sources, aerodynamic or engine related.

Attempts have been made to analyze the 757 acoustic data and make comparisons with theoretical predictions (ref. 31). A procedure developed by the Lockheed Georgia Company under NASA contract has been used (ref. 33). To the authors' knowledge this is the only code available to make near field noise predictions that include all the potentially relevant noise sources at flight cruise conditions. However, the theory lacks inclusion of the important effects of scattering, refraction, and reflection of sound fields due to the airframe or flow fields about it. For this reason, predictions for only the lower surface OASPL's and spectra have been made. We will not discuss these results in any depth herein. Generally, the results indicate that our ability to predict the acoustic environment at high cruise speeds and altitudes is poor. Theory suggests that the lower wing surface noise should be dominated by the fan exhaust broad band shock noise at cruise thrust conditions, which is consistent with the observed correlation with the fan Mach number; but the predicted levels of OASPL are 10 to 40 dB. too high. Trailing-edge noise is predicted to be an important aerodynamic noise source, particularly at aft wing locations; the data doesn't confirm this. Convective and dynamic amplification effects have large impacts upon the predictions. These effects or the methodology for their implementation are made suspect by the data. Clearly, more analyses of these acoustic data are needed to unravel the confused picture presented by the data and theory. With further analyses, the broad range of conditions for the 757 data could possibly permit useful calibration of the Lockheed code.

The amount of laminar flow obtained on the NLF glove was very encouraging. This result indicates that the acoustic environment may be benign enough to achieve extensive laminar flow on wings with wing-mounted engines. The results are summarized in figure 23 wherein the design condition and conditions of maximum extent of laminar flow are shown. A maximum of nearly 30-percent chord laminar flow was obtained on both surfaces. At the design condition, best results were obtained on the upper surface, athough laminar flow was not uniform across the gloved span and was most extensive inboard. The upper surface pressures on the glove showed peaks in the outboard region which presumably led to earlier boundary-layer transition. Transition was more uniform across the lower surface with 26-percent chord laminar flow achieved when the aircraft was sideslipped to reduce the leading-edge sweep by 6.8 degrees.

On the upper wing surface, the extent of laminar flow was essentially unaffected by engine power setting. Since the power setting had no effect on the upper-surface noise levels, the unchanging extent of laminar flow is not surprising. On the lower surface, however, the noise levels varied over 20 dB., but almost imperceptibly small (2 to 3-percent chord decreases at most) changes in the extent of laminar flow were observed. Over the range of flight conditions, boundary-layer stability analysis (ref. 31) identified stationary crossflow vortices in the boundary layer to be highly unstable and possibly the dominant disturbances leading to transition. The results for the design flight condition are typical (figure 24). The crossflow disturbance amplification is two to three orders of magnitude greater than the amplification of Tollmien-Schlichting disturbances in both the upper and lower surface boundary layers. The small observed effect of variations in engine noise level on the transition location on the lower surface may indicate that engine noise does not have a significant effect on crossflow disturbances. If crossflow disturbance growth in the leading edge is controlled by suction, laminar flow much more extensive than achieved in this flight test could be possible even in the presence of engine noise. However, in an HLFC application, the Tollmien-Schlichting wave growth may be comparable or greater than the crossflow disturbance growth; engine noise might then be expected to limit the extent of laminar flow.

FLIGHT DATA/BOUNDARY-LAYER STABILITY THEORY CORRELATIONS

Under NASA contract, the Boeing Company has attempted to correlate the F-111/TACT and 757 data using linear boundary-layer stability theory (refs. 24 and 31). Their approach attempts to account for an interaction of crossflow disturbances and the Tollmien-Schlichting disturbances as predicted by linear stability theory by cross plotting the amplification N factors for these disturbances at transition. Stationary crossflow vortices and oblique Tollmien-Schlichting waves (inclined to the streamline at the angle of greatest amplification) are considered. A total of 21 flight-test cases were analyzed for the 757 NLF glove and are shown in figure 25. Included are points for the F-111/TACT NLF glove flight test that were reported in reference 24. One point from a 20 degree swept NLF glove on a T-33 aircraft (unpublished data by the Boeing Company) is also included. The F-111/TACT and 757 data complement one another, since the former is mostly for conditions where Tollmien-Schlichting waves were dominant, and in the latter, crossflow disturbances were dominant. The band enclosing the data is the Boeing recommended transition criteria.

This approach has been critized because only stationary crossflow vortices and Tollmien-Schlichting waves are examined and their interaction is presumed. Stationary crossflow vortices have been shown to not always be the most highly amplified crossflow like disturbances in the boundary layer (ref. 34). But whether or not a particular type of disturbance is present in the boundary layer would seem to be the major question. The existence of stationary crossflow vortices is well established with observations in many experiments. Some authors have reported observations of non-stationary disturbances in the crossflow field (refs. 34 and 35); Kohama (ref. 36) suggests that these observances are, in fact, evidence of secondary instabilities produced by growth of the primary instabilities, stationary crossflow vortices. The existence of an interaction of Tollmien-Schlichting and crossflow disturbances is likewise a controversial issue, and the authors of references 37 and 38 have proposed that in transition predictions one need only consider the most amplified disturbance, with transition occurring when an N factor of around 7 to 11 first occurs. More analyses of data are required to determine the best approach.

In stability analyses underway at Langley, the F-14, JetStar, and 757 transition data are being examined with the maximum amplification option of reference 39, but these analyses are also including surface and in-plane streamline curvature effects (refs. 34 and 40). The initial efforts have concentrated on transition data for conditions where Tollmien-Schlichting wave growth is small and crossflow-like disturbances dominate the transition process. Stability analyses are performed for both stationary and nonstationary crossflow disturbances.

Illustrated in figures 26 through 29 are results for two typical flight conditions on the "clean-up" glove of the F-14 aircraft. These flight conditions produce strong, favorable pressure gradients (figures 26 and 28) that lead to little or no Tollmien-Schlichting wave growth and dominance of crossflow-like disturbances. The crossflow Reynolds number development (figures 26 and 28) is indicative of strong crossflow vortices, and transition occurs for both conditions when this parameter exceeds 400, a value somewhat higher than the 175 to 300 range observed at low speeds. Previous analyses (ref. 41) indicate that the effects of compressibility on crossflow-like disturbances are small; comparison of the N factors for stationary and nonstationary crossflow vortices, with and without compressiblilty effects, confirms this, as shown in figures 27 and 29. Also, stationary crossflow vortices are not the most highly amplified disturbances for these conditions, but indeed, nonstationary vortices with frequencies of about 2000-to 3000-Hertz are more highly amplified. In the absence of significant compressibility effects, the incompressible code developed by Malik and Poll (ref. 34) has been used to examine surface and inplane curvature effects on the disturbance development. These effects, shown in figures 27 and 29, are quite significant; the N factors of the most highly amplified waves are reduced from about 15 to around 10 at the measured transition location. Similar calculations are shown for two conditions on the 757 NLF glove (figures 30 and 31). Curvature effects are again large, and with these effects included, the N factors at transition are around 10. Crossflow Reynolds numbers at transition fall in the range of 300 to 500.

The magnitude of the curvature effects in all the flight data analyzed to date gives rise to concerns over any previous attempts to correlate transition data with stability codes ignoring curvature effects and questions the generality of those correlations. Immediate plans are to begin examination of data for flight conditions with stronger Tollmien-Schlichting wave growth to explore the possibility of interactions of Tollmien-Schlichting waves and crossflow vortices.

THE NASA LEADING-EDGE FLIGHT TEST

Earlier in this paper, some of the key laminar-flow flight programs that laid the foundation for today's knowledge were briefly reviewed. These flight tests removed any doubt that extensive laminar flow could be achieved in flight. They did not, however, resolve concerns relative to the practicality of producing surfaces sufficiently smooth and wavefree, and of maintaining the required surface quality during normal service operations. In the late 1970's, with the recent significant progress made in the development of new materials, fabrication techniques, analysis methods, and design concepts, a reexamination of these issues appeared warranted.

Previous experience had shown that the leading-edge region of the swept wing presented the most difficult aerodynamic problems associated with attainment of laminar flow. In addition, the leading edge is subject to foreign object damage, insect impingement, rain erosion, icing, and other contaminants. Also, an anti-icing system, an anti-contaminant system, and a suction and perhaps purge system must all be packaged into a relatively small leading-edge box volume. Most of these problems are common to all the concepts under consideration for the achievement of extensive laminar flow, and solutions are needed to establish the practicality of laminar flow for various types of aircraft.

In 1980, the NASA Leading-Edge Flight Test (LEFT) program was initiated as a flight validation of two leading-edge systems then under development in NASA contract efforts with industry. The flight program objectives were to (1) demonstrate that required leading-edge systems can be packaged into a wing leading-edge section of a size representative of a commercial transport aircraft, and (2) demonstrate systems performance under operational conditions representative of subsonic commercial transport aircraft. Complete LFC leading-edge systems were installed in the leading-edge box of a JetStar airplane (figure 32). Descriptions of the systems illustrated in figures 33 and 34 are provided in references 42 and 43. Two leading-edge test articles were built and flown using a perforated and a slotted suction concept. Each spanned about 6-foot of the wing and had the same external contour, dimensionally about equivalent to the leading-edge box of a DC-9-30 at the mean aerodynamic chord. The wing leading-edge sweep was 30 degrees. Different systems were used in each test article. One used suction through approximately 1 million, 0.0025-inch diameter, 0.035-inch spaced, electron-beam perforated holes in a 0.025-inch thick titanium skin to maintain laminar flow on the test article upper surface. A Krueger-type flap served as a protective shield against insect impact on this leading edge. In future applications, the Krueger shield could also serve as a high-lift leading-edge device. A freezing-point depressant liquid, Propylene Glycol Methyl Ether (PGME) was sprayed on the perforated, wing upper surface from nozzles mounted underneath the shield to augment the insect shield protection and to provide an anti-icing capability. To prevent clogging of the perforations by the wetting fluid, a purging system was included to clear the LFC passages by pressurizing the subsurface and thus remove PGME fluid from the LFC ducts and surface. The second test article used suction through 27 narrow spanwise slots (about 0.004-inch wide) on both upper and lower titanium surfaces. This test article contained anti-contaminant and anti-icing systems consisting of PGME fluid dispensed through dual purpose slots in the leading edge. Purge was also provided for this leading edge.

After an initial flight test program to optimize the system's performance, the LEFT systems were flight tested in a simulated airline service in different geographical areas, seasons, and weather conditions in the United States (figure 35). During the simulated service, one-to four-flights per day were made from three "home base" airports (Atlanta, Pittsburgh, and Cleveland). A total of 62 flights to 33 airports were made. Flights were made from Atlanta in July 1985, Pittsburgh in September 1985, and Cleveland in February 1986. The weather experienced thus varied from severe summer to severe winter conditions. To realistically simulate typical transport operations, an on-off operation of all systems was imposed; no adjustments were made prior to or during flights. Transport cruise flight conditions were emphasized, but investigations were also made of the ability to attain laminar flow at other than cruise conditions. References 44 through 47 provide a summary description of the program results.

The emergence of electron-beam perforated titanium as a practical manufacturing surface which meets laminar-flow waviness specifications with practical aircraft fabrication methods is considered a major development of the LEFT program. The perforated titanium leading-edge presented no difficult fabrication problems. This test article yielded clearly superior performance (relative to the slotted configuration) and was in virtually the same condition when flights ended in October 1987, as when flights began in November 1983. Four years of flying resulted in no degradation of laminar-flow performance as a result of service, and no evidence of any deterioration in surface quality was observed. Essentially,complete laminar flow on the test article was consistently obtained from 10,000 to 38,000 feet altitude with no need for any special maintenance.

The results obtained with the slotted-surface test article, however, were not as favorable. Fabrication of this configuration involved some extremely difficult problems that led to a suction surface that was only marginally acceptable with respect to surface smoothness and waviness. This was reflected in consistently poorer laminar-flow flight performance than for the test article with the perforated surface. Still, as much as 80 percent of the slotted upper surface suction area was observed to be laminar in routine flight service.

Since no attempt was made to obtain laminar flow beyond the front spar, the LEFT tests should not be interpreted as showing that perforations are aerodynamically better than slots. Indeed, the perforated approach should be pursued with caution because additional flight testing is required to larger values of length Reynolds number. At higher Reynolds number, the experience of the early flight tests with larger holes (i.e., progressive performance deterioration with increased Reynolds number) could be repeated. Slots may, therefore, be preferred at higher Reynolds number. Accordingly, it is clear that more development of fabrication techniques for slotted suction surface configurations is required; some initial work in this direction has been undertaken (ref. 48).

The LEFT program relaxed concerns about the operational loss of laminar flow when entering clouds or haze. It provided some confirmation of an extensive analysis of world-wide cloud-cover (based on 6250 flight hours of specially instrumented commercial aircraft) which resulted in an estimate of 6 percent for the amount of flight time spent in clouds and haze (ref. 49). During the simulated service flights, measurements were also taken of the time spent in clouds and haze. These LEFT results, based on 6 hours and 52 minutes of data taken during 13 flights within the United States, showed that clouds and haze were encountered about 7 percent of the time (ref. 47). No effort was made to avoid cloud encounters, and a sample of one flight including a cloud penetration is shown in figure 36. As expected, laminar flow was lost during cloud penetrations, but was regained afterwards. The small percentage of time that clouds are encountered indicates that laminar flow loss during cloud penetrations in cruise will not appreciably decrease the large economic and fuel gains predicted for laminar-flow transport aircraft. However, potential cloud encounters en route and flight management to avoid clouds could be operational considerations for future aircraft.

To summarize the LFC systems performance during the simulated service, all operational experience was positive. No dispatch delays were encountered due to the LFC systems. There was no need to adjust suction system controls throughout the test range of cruise altitude, Mach number, and lift coefficient. Laminar flow was obtained after exposure to heat, cold, humidity, insects, rain, freezing rain, snow, ice, and moderate turbulence. The insect alleviation systems were required during descent as well as ascent and were effective when used. Perforated test article results indicated that the supplemental spray system is not necessary for LFC transport airplanes equipped with a properly designed insect shield/high-lift device, although the spray system may be necessary for anti-icing purposes. Ground deicing of the LFC test articles was no more difficult than normal deicing of commercial transports, and snow and ice accumulation was easily eliminated using hand-held deicing equipment. The NASA LEFT simulated airline service flights demonstrated that effective practical solutions for the problems of suction laminar-flow aircraft leading edges are available for commercial transport aircraft.

The LEFT program has been the only laminar-flow flight test with suction since the X-21 ended in 1965. The original intent of the LEFT program was to examine systems suitable for future laminar-flow control aircraft, but these systems would be equally applicable to hybrid laminar-flow control aircraft that use suction only in the leading-edge box.

HLFC FLIGHT EXPERIMENT

One of the most significant developments of the NASA research on laminar flow in the last few years has been the recognition of a hybrid laminar-flow control concept that integrates LFC and NLF and avoids the objectionable characteristics of each. The leading-edge sweep limitation of NLF is overcome by applying suction in the leading-edge box to control crossflow instabilities. Wing shaping for favorable pressure gradients to allow NLF over the wing box removes the need for inspar LFC suction and greatly reduces the system complexity. The possibility of achieving extensive laminar flow on commercial or military transport aircraft is offered with a system no more complex than that already proven in the Leading-Edge Flight Test Program on the NASA JetStar. To explore this possibility, the NASA Langley Research Center, the Air Force Wright Aeronautical Laboratory, and the Boeing Commercial Airplane Company have initiated a cooperative flight program. A high Reynolds number HLFC Flight Experiment will be performed on a 757 aircraft equipped with a partial-span HLFC system for the upper surface of the left wing.

The test aircraft and test region are illustrated in figure 37. A 20-foot span of the wing just outboard of the left engine pylon will be modified. A new leading-edge box will be installed with suction achieved through a perforated titanium surface. The structural concept will be similar to that used on the JetStar for the Leading-Edge Flight Test and will include a leading-edge Krueger integrated into the full wing high-lift system and designed to also be an insect shield for the wing (figure 38). The leading-edge box will be contoured to achieve the desired pressure distribution over the test surface (figure 39). Analyses indicate that this can be accomplished without changing the inspar contour of the 757. Indeed, measurements of the 757 production wing surface have shown that only minor shaving or filling of some rivets will be necessary to meet laminar-flow smoothness and fairness criteria. The inspar production wing surface will thus serve as the test surface downstream of the new leading-edge box.

The HLFC concept, untried to date in flight or in the wind tunnel, will be evaluated to chord Reynolds numbers over 30 million at the cruise conditions of modern transport aircraft. An extended flight test program is planned for calendar year 1990 to achieve operational experience with HLFC and to fully evaluate the potential for future applications. Success could lead to the long-awaited transfer of this technology to the drawing board and ultimately to practice.

CONCLUDING REMARKS

The potential benefits of laminar-flow technology have been so enticing that possibly no other technology has received such persistent attention in flight research over so long a time. The misgivings of the critics are fading with the accomplishments of this research. The aerodynamic issues seem nearly resolved, and the manufacturing capabilities of the airframe industry appear to have advanced to the point that the aerodynamic criteria for smooth, wave-free wing surfaces is a practical production goal. The current NASA, AFWAL, and

Boeing HLFC Flight Experiment could provide the verification needed to place this technology in practice. Initial applications may provide only modest improvements, but with the confidence of success, bolder steps could revolutionize aircraft design.

REFERENCES

1. Jones, B. M.: Flight Experiments on the Boundary Layer. JAS, Vol. 5, No. 3, pp. 81-94. January 1938.

2. Whites, R. C.; Sudderth, R. W.; and Wheldon, W. G.: Laminar Flow Control on the X-21. Astronautics and Aeronautics, Vol. 4, No. 7, July 1966.

3. Wagner, R. D.; and Fischer, M. C.: Developments in the NASA Transport Aircraft Laminar Flow Program. AIAA-83-0090, January 1983.

4. Wetmore, J. W.; Zalovcik, J. A.; and Platt, R. C.: A Flight Investigation of the Boundary-Layer Characteristics and Profile Drag of the NACA 35-215 Laminar-Flow Airfoil at High Reynolds Numbers. NACA L-532, 1941.

5. von Doenhoff, A. E.; and Braslow, A. L.: Effect of Distributed Surface Roughness on Laminar Flow. Chapter of Boundary Layer and Flow Control-Its Principles and Applications, Volume 2, edited by G. V. Lachmann, Pergamon Press, New York, 1961.

6. Braslow, A. L.: Investigation of Effects of Various Camouflage Paints and Painting Procedures on the Drag Characteristics of an NACA 65(421)-420, a = 1.0 Airfoil Section. NACA CB No. L4G17, July 1944.

7. Braslow, A. L.; and Visconti, F.: Investigation of Boundary-Layer Reynolds Number for Transition on an NACA 65(215)-114 Airfoil in the Langley Two-Dimensional Low-Turbulence Pressure Tunnel. NACA TN 1704, October 1948.

8. Gray, W. E.; and Fullam, P. W. J.: Comparison of Flight and Tunnel Measurements of Transition on a Highly Finished Wing (King Cobra). RAE Report Aero 2383, 1945.

9. Smith, F.; and Higton, D.: Flight Tests on King Cobra FZ.440 to Investigate the Practical Requirements for the Achievement of Low Profile Drag Coefficients on a "Low Drag" Aerofoil. ARC Report, R&M 2375, 1950.

10. Plascoff, R. H.: Profile Drag Measurements on Hurricane II z. 3687 Fitted with Low-Drag Section Wings. RAE Report No. Aero 2153, 1946.

11. Zalovcik, J. A.; Wetmore, J. W.; and von Doenhoff, A. E.: Flight Investigation of Boundary-Layer Control by Suction Slots on an NACA 35-215 Low-Drag Airfoil at High Reynolds Numbers. NACA WR L-521, 1944.

12. Head, M. R.: The Boundary Layer With Distributed Suction. R&M No. 2783, British A. R. C., 1955.

13. Head, M. R.; Johnson, D.; and Coxon, M.: Flight Experiments on Boundary Layer Control for Low Drag. R&M No. 3025, British A. R. C., March 1955.

14. Edwards, B.: Laminar Flow Control - Concepts, Experiences, Speculations. AGARD Report No. 654. Special Course on Concepts for Drag Reduction, 1977.

15. Pfenninger, W.; Groth, E. E.; Whites, R. C.; Carmichael, B. H.; and Atkinson, J. M.: Note About Low Drag Boundary Layer Suction Experiments in Flight on a Wing Glove of a F94-A Airplane. Reference No. NAI-54-849 (BLC-69), Northrop Aircraft, Inc., December 1954.

16. Pfenninger, W.: Laminar Flow Control Laminarization. AGARD-R-654, June 1977.

17. Landeryou, R. R.; and Porter, P. G.: Further Tests of a Laminar Flow Swept Wing with Boundary Layer Control by Suction. Report Aero. No. 192, College of Aeronautics, Cranfield (England), May 1966.

18. Gray, W. E.: The Effect of Wing Sweep on Laminar Flow, RAE TM Aero. 255, 1952.

19. Anscombe, A.; and Illingworth, L. N.: Wind Tunnel Observations of Boundary Layer Transition on a Wing at Various Angles of Sweepback. R and M No. 2968, British A. R. C., 1956.

20. Pfenninger, W.; Gross, L.; and Bacon, J. W., Jr.: Experiments on a 30 Degree Swept, 12 Percent Thick, Symmetrical, Laminar Suction Wing in the 5-Foot by 7-Foot Michigan Tunnel. Northrop Corp., Noair Division Report. NAI-57-317 (BLC-93), February 1957.

21. Boeing Commercial Airplane Company: Natural Laminar Flow Airfoil Analysis and Trade Studies, NASA CR-159029. May 1979.

22. Anom.: Advanced Aerodynamics and Active Controls - Selected NASA Research, NASA CP-2172, October 1980.

23. Anom: Advanced Aerodynamics - Selected NASA Research, NASA CP-2208, September 1981.

24. Boeing Commercial Airplane Company: F-111 Natural Laminar Flow Glove Flight Test Data Analysis and Boundary-Layer Stability Analysis. NASA CR-166051. January 1984.

25. Runyan, L. J.; and Steers, L. L.: Boundary-Layer Stability Analysis of a Natural Laminar Flow Glove on the F-111 TACT Airplane, Viscous Flow Drag Reduction, Vol. 72, Progress in Astronautics and Aeronautics, pp. 17-32, 1980.

26. Rozendaal, R. A.: Variable Sweep Transition Flight Experiment (VSTFE) - Parametric Pressure Distribution, Boundary-Layer Stability Study and Wing Glove Design Task. NASA CR-3992, 1986.

27. Anom: Research in Natural Laminar Flow and Laminar Flow Control. NASA CP-2487, Part 3, March 1987.

28. Bohn-Meyer, M.: Constructing Gloved Wings for Aerodynamic Studies, AIAA 88-2109, May 1988.

29. Boeing Commercial Airplane Company: Flight Survey of the 757 Wing Noise Field and Its Effect on Laminar Boundary Layer Transition. NASA CR-178216, Vol. I - Program Description and Data Analysis. March 1987.

30. Boeing Commercial Airplane Company: Flight Survey of the 757 Wing Noise Field and Its Effect on Laminar Boundary Layer Transition. NASA CR-178217, Vol. 2 - Data Compilation. March 1987.

31. Boeing Commercial Airplane Company: Flight Survey of the 757 Wing Noise Field and Its Effect on Laminar Boundary Layer Transition. NASA CR-178419, Vol. 3 - Extended Data Analysis. May 1988.

32. Boltz, F. W.; Kenyon, G. C.; and Allen, C. Q.: Effects of Sweep Angle on the Boundary-Layer Stability Characteristics of an Untapered Wing at Low Speeds. NASA TN D-338, 1960.

33. Swift, G; and Mangur P.: A Study of the Prediction of Cruise Noise and Laminar Flow Control Noise Criteria for Subsonic Air Transports. NASA CR-159104, August 1979.

34. Malik, M. R.; and Poll, D. I. A.: Effect of Curvature on Three-Dimensional Boundary Layer Stability. AIAA Journal, Vol. 23, No. 9, pp. 1362-1369. September 1985.

35. Poll, D. I. A.: Some Observations of the Transition Process on the Windward Face of a Long Yawed Cylinder. J. Fluid Mech., Vol. 150, pp. 329-356. January 1985.

36. Kohama, Y.: Some Expectation on the Mechanism of Cross-Flow Instability in a Swept Wing Flow, Acta Mechanica 66, pp. 21-38, 1987.

37. Hefner, J. N.; and Bushnell, D. M.: Status of Linear Boundary Layer Stability Theory and the e^N Method, With Emphasis on Swept Wing Applications. NASA TP-1645, 1980.

38. Anom: Research in Natural Laminar Flow and Laminar-Flow Control. NASA CP-2487, Part I, pp. 219-244, March 16-19, 1987.

39. Malik, M. R.: Cosal-4 A Black Box Compressible Stability Analysis Code for Transition Prediction in Three-Dimensional Boundary Layers. NASA CR-165925, 1982.

40. Collier, F. S., Jr.; and Malik, M. R.: Stationary Disturbances in Three-Dimensional Boundary Layers. AIAA Paper 87-1412, June 1987.

41. Mack, L. M.: Boundary Layer Stability Theory. AGARD Report No. 709, Special Course on Stability and Transition of Laminar Flows, VKI, Brussels, 1984.

42. Etchberger, F. R.: Laminar Flow Control Leading Edge Glove Flight Aircraft Modification Design, Test Article Development, and Systems Integration. NASA CR-172136, 1983.

43. Douglas Aircraft Company Staff: Laminar Flow Control Leading Edge Glove Flight Test Article Development. NASA CR-172137, 1984.

44. Fisher, D. F.; and Fischer, M. C.: The Development Flight Tests of the JetStar LFC Leading-Edge Flight Experiment. NASA CP-2487, pp. 117-140, 1987.

45. Maddalon, D. V.; Fisher, D. F.; Jennett, L. A.; and Fischer, M. C.: Simulated Airline Service Experience with Laminar Flow Control Leading-Edge Systems. NASA CP-2487, pp. 195-218, 1987.

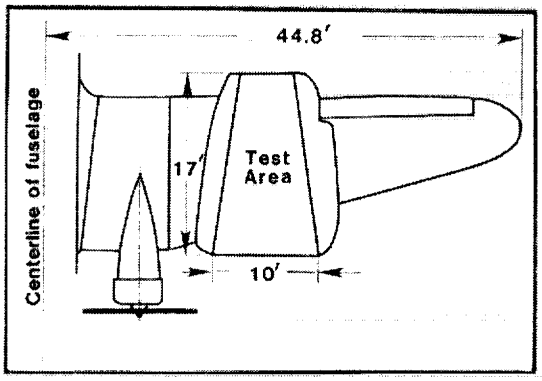
46. Powell, A. G.; and Varner, L. D.: The Right Wing of the LEFT Airplane. NASA CP-2487, pp. 141-161, 1987.

47. Davis, R. E.; Maddalon, D. V.; and Wagner, R. D.: Performance of Laminar Flow Leading Edge Test Articles in Cloud Encounters. NASA CP-2487, pp. 163-193, 1987.

48. Goodyear, M. D.: Application of Superplastically Formed and Diffusion Bonded Aluminum to a Laminar Flow Control Leading Edge. NASA CR-178316, 1987.

49. Jasperson, W. H.; Nastrom. G. D.; Davis, R. E.; and Holdeman, J. D: GASP Cloud - and Particle Encounter Statistics, and Their Application to LFC Aircraft Studies (2 Vols.). NASA TM 85835, October 1984.

(CIRCA 1940)



- NACA 35-215 Airfoil
- Wooden Glove
  (± 1/1000 inch Waves)
- Transition Reynolds No.
  ~ 11.3×10^6
  at 42.5% Chord
- Engine/Propeller Noise
  Reduced Laminar Run
  by 3% Chord

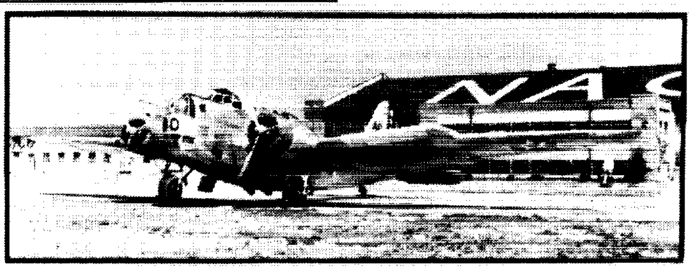
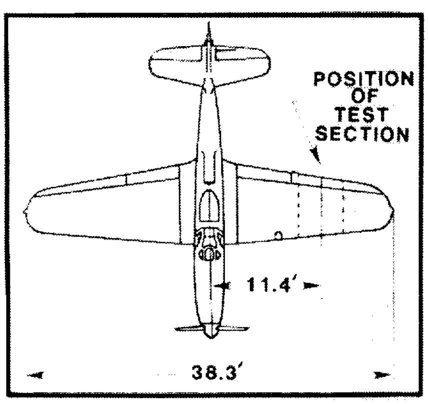


Figure 1. - The B-18 NLF Glove Flight Test.

(CIRCA - 1945)



- NACA 662X-116 section
- Polished and filled production
  wing surface (± 1/1000 inch waves)
- Transition Reynolds No. ~11×10^6
- Transition beyond $C_{p_{min}}$.
- Operational evaluations
  - Insects
  - Dust & rain
  - Maintenance
  - Surface quality deterioration

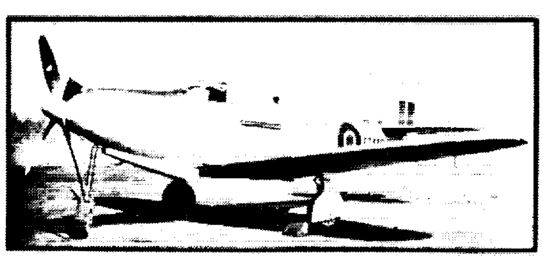


Figure 2. - The King Cobra NLF Flight Test.

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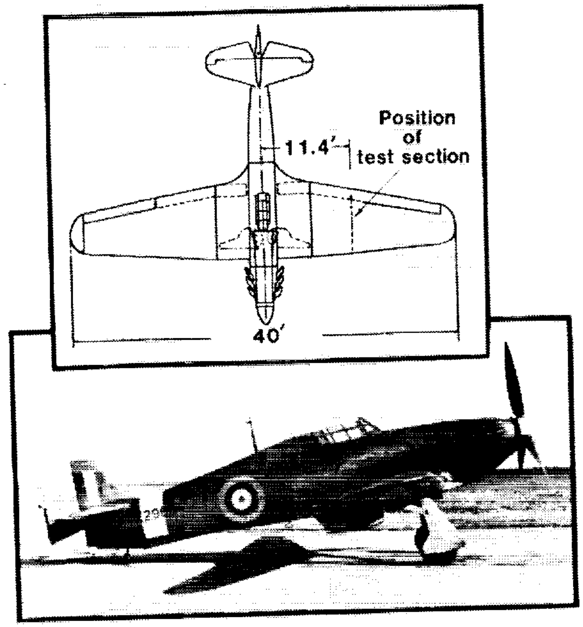
(CIRCA - 1945)



- N.P.L. Section
  $C_{p_{min}}$ @ 50% c
- Polished, filled special "low drag construction" wing (± 1/1000 inch waves)
- No spars, stringer stiffened thick wing skins
- Expected low drag performance achieved
- Laminar flow to 50 – 60%c

Figure 3. - The Hurricane NLF Flight Test.

(CIRCA 1955)

- Multiple Suction Surfaces:
  - Porous Monel Cloth (Nylon Covered)
  - Perforated Sheet (0.007 & 0.020 inch dia.)
  - Porous Strips (Sintered metal)
- Full Chord LFC at $M_\infty = 0.7$ and $R_c \sim 30\times10^6$ with:
  - Monel/Nylon Cloth
  - 0.007 inch Perforations

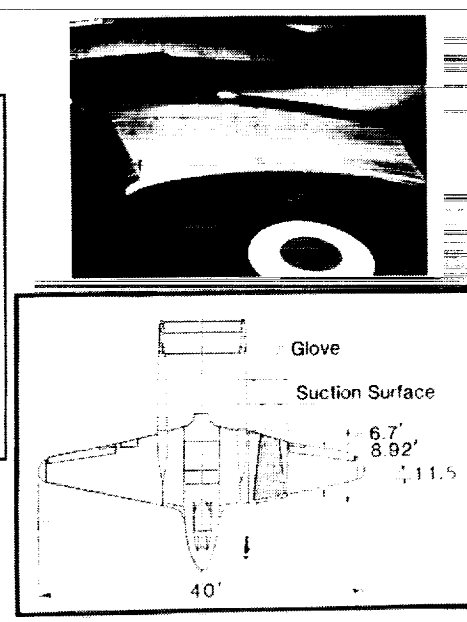


Figure 4. - The Vampire LFC Flight Experiments.

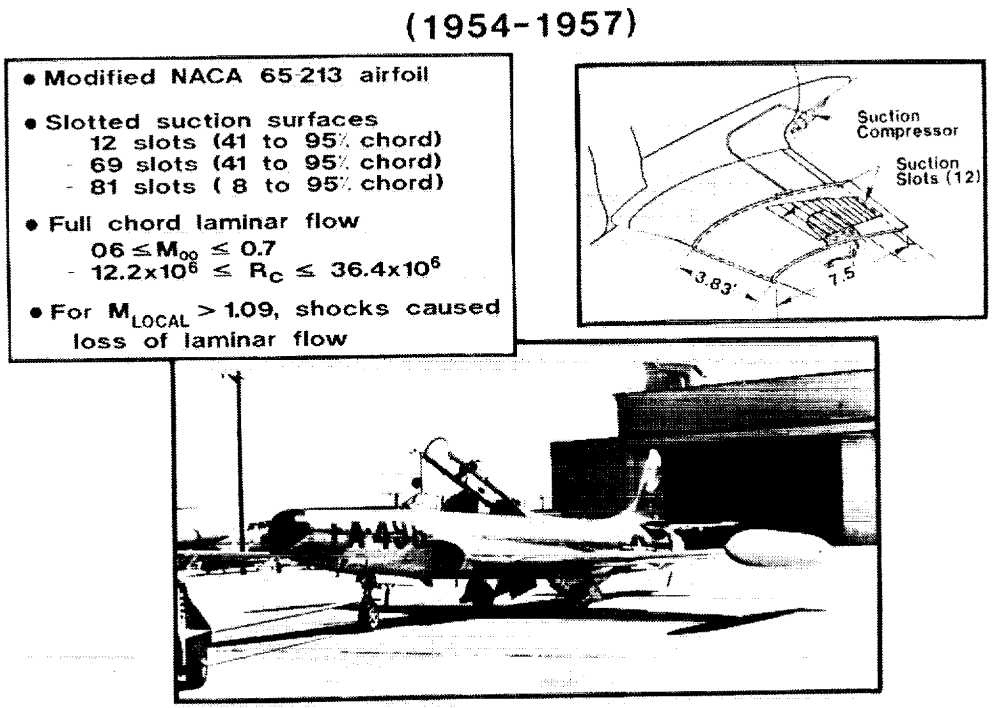


Figure 5. - The F-94 LFC Flight Experiments.

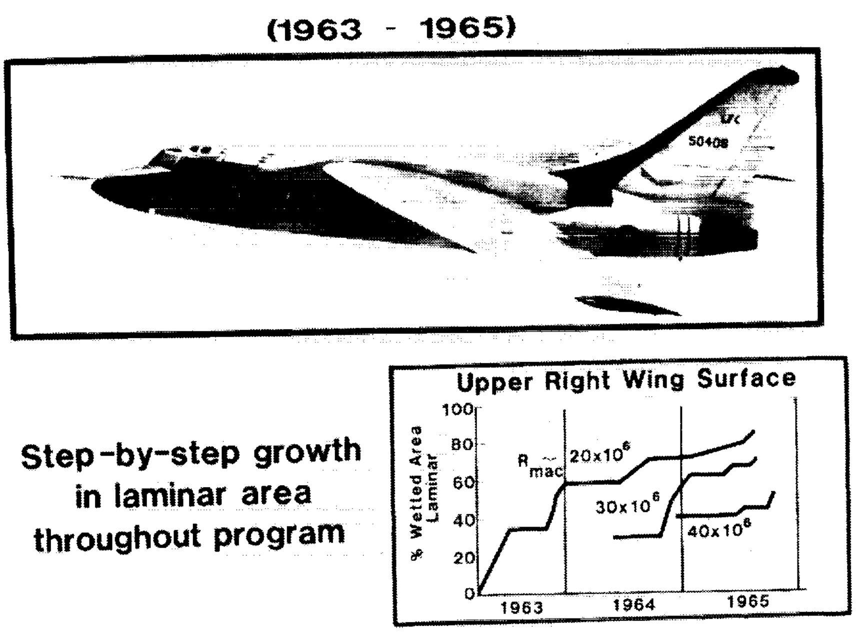


Figure 6. - The X-21 LFC Flight Program.

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Figure 7. - The Lancaster Swept LFC Wing Flight Experiment.

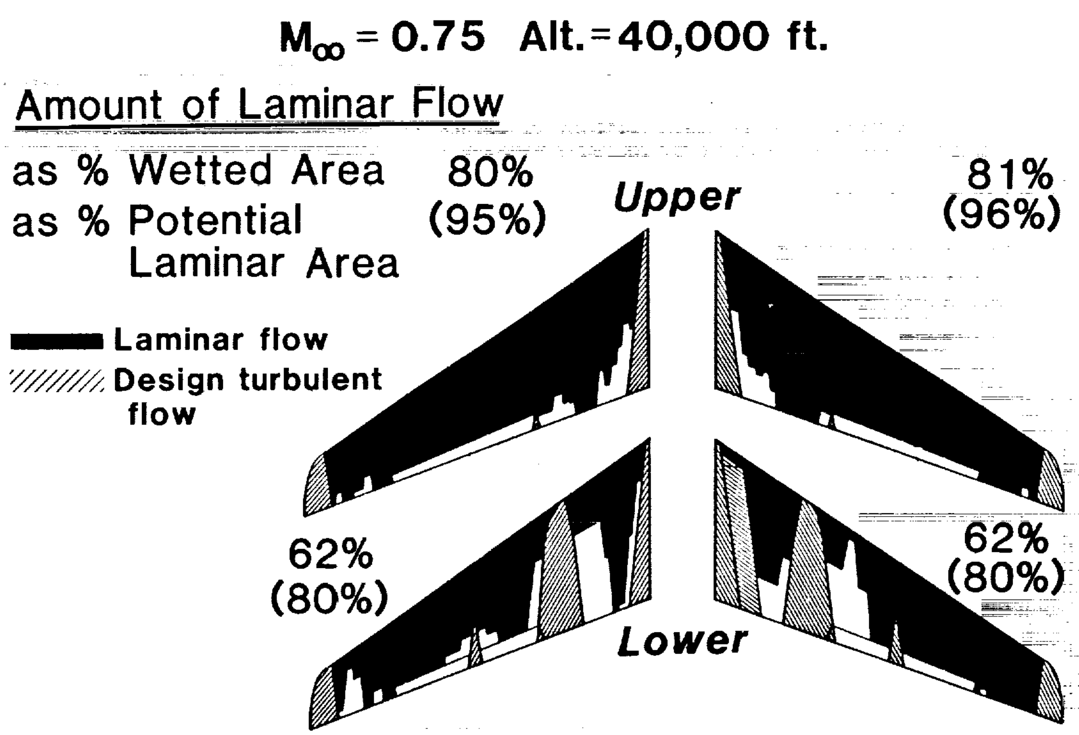

Figure 8. - The X-21 maximum laminar flow areas.

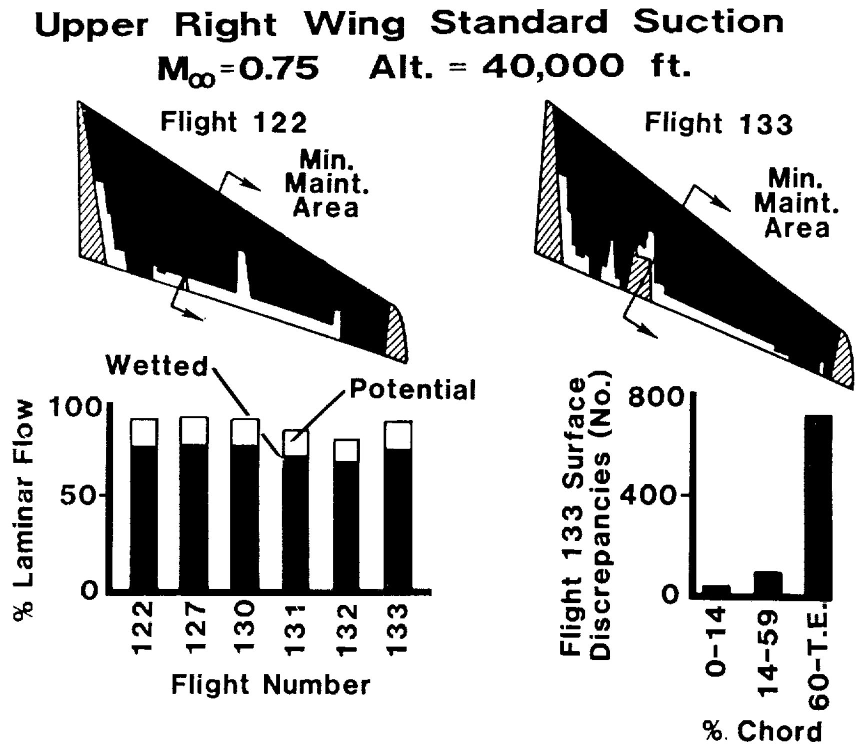

Figure 9. - Maintenance effects on the repeatability of laminar flow on the X-21.

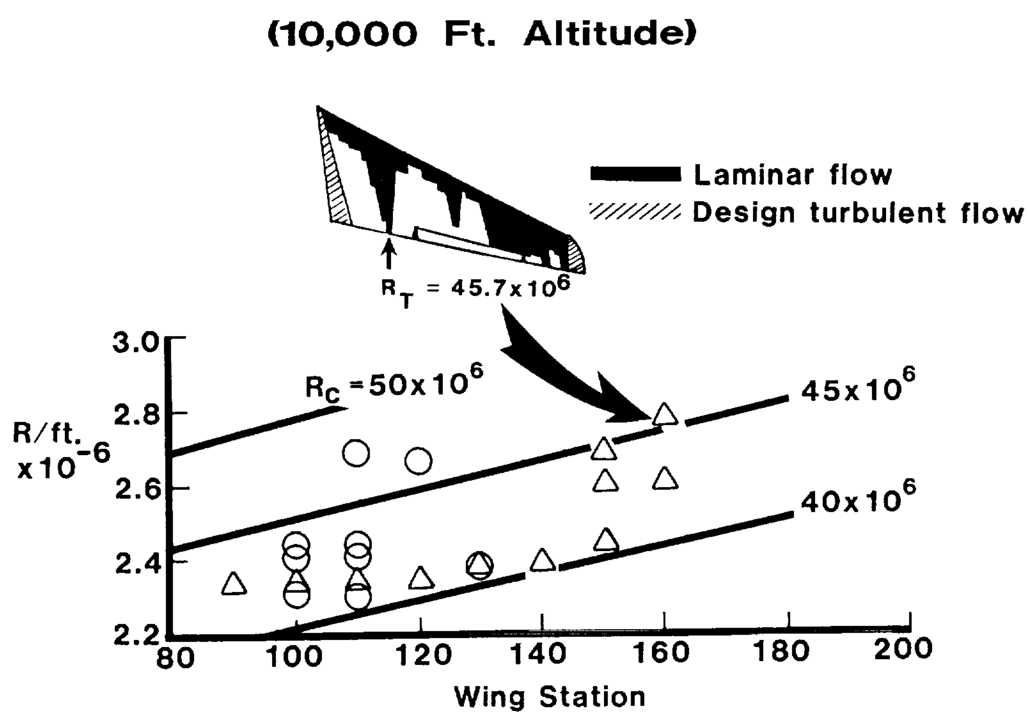

Figure 10. - X-21 high chord Reynolds number results.

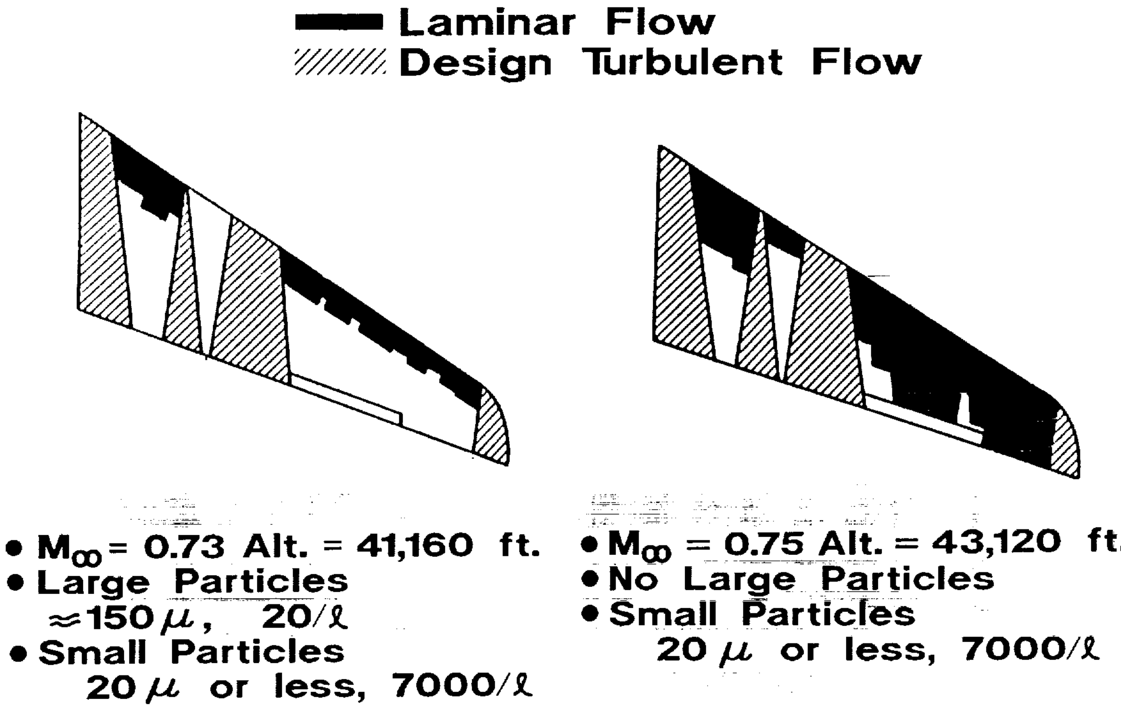

Figure 11. - Effect of ice particle encounters on X-21 results.

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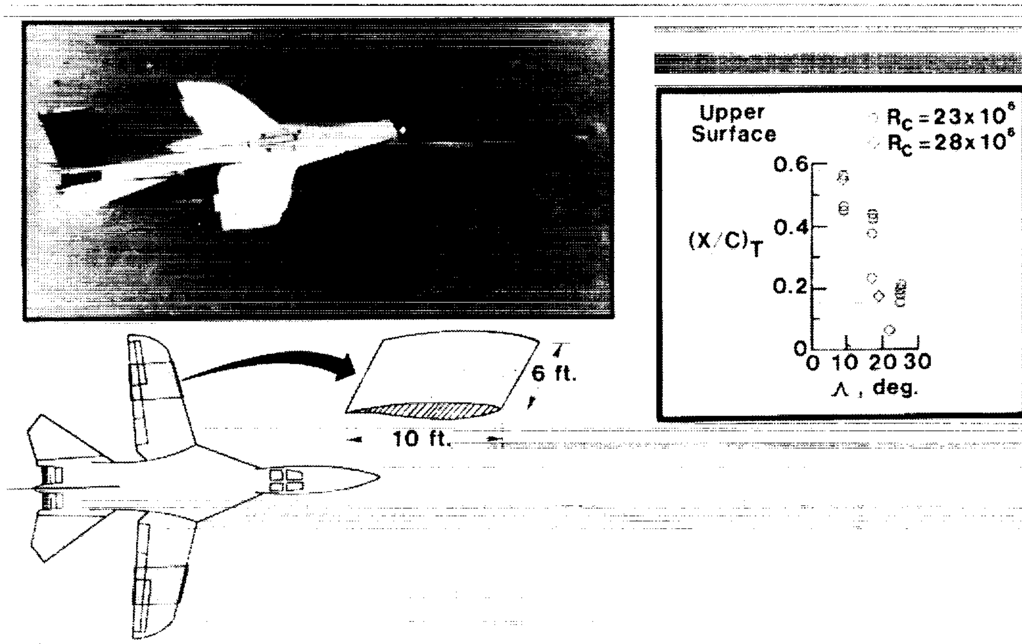

Figure 12. - The F-111/TACT NLF Glove Flight Experiment.

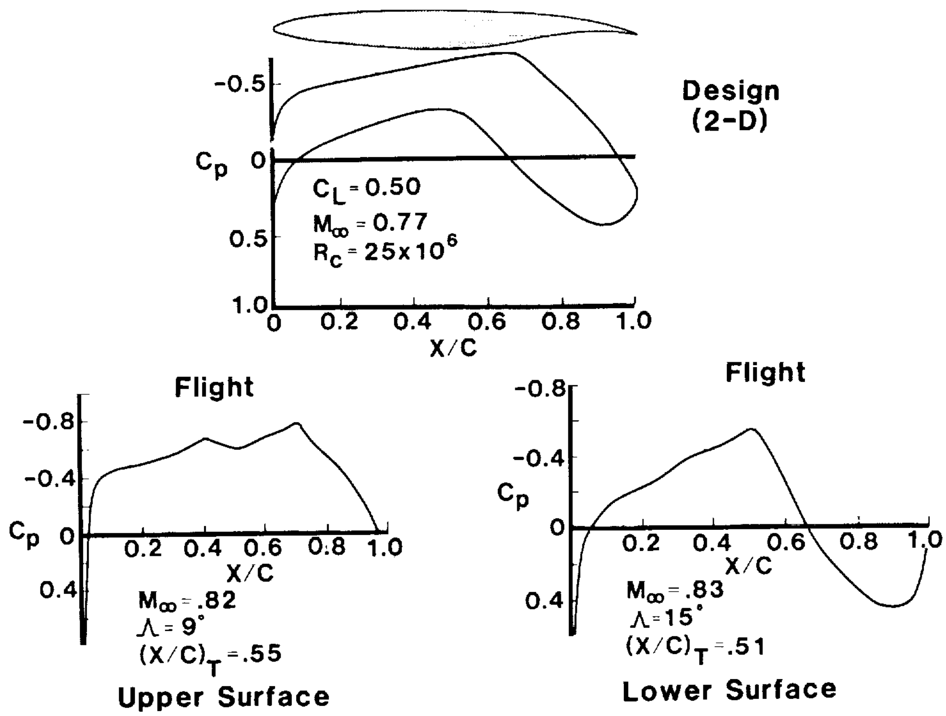

Figure 13. - Design and flight pressure distributions for the F-111/TACT NLF glove.

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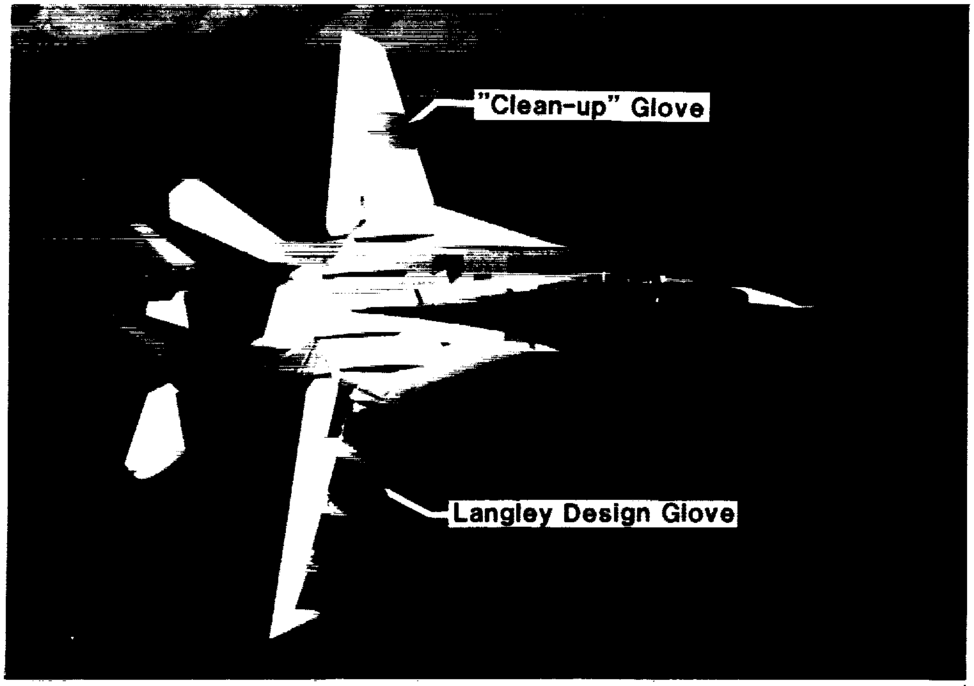

Figure 14. - F-14 test-bed aircraft for the Variable Sweep Transition Flight Experiment (VSTFE).

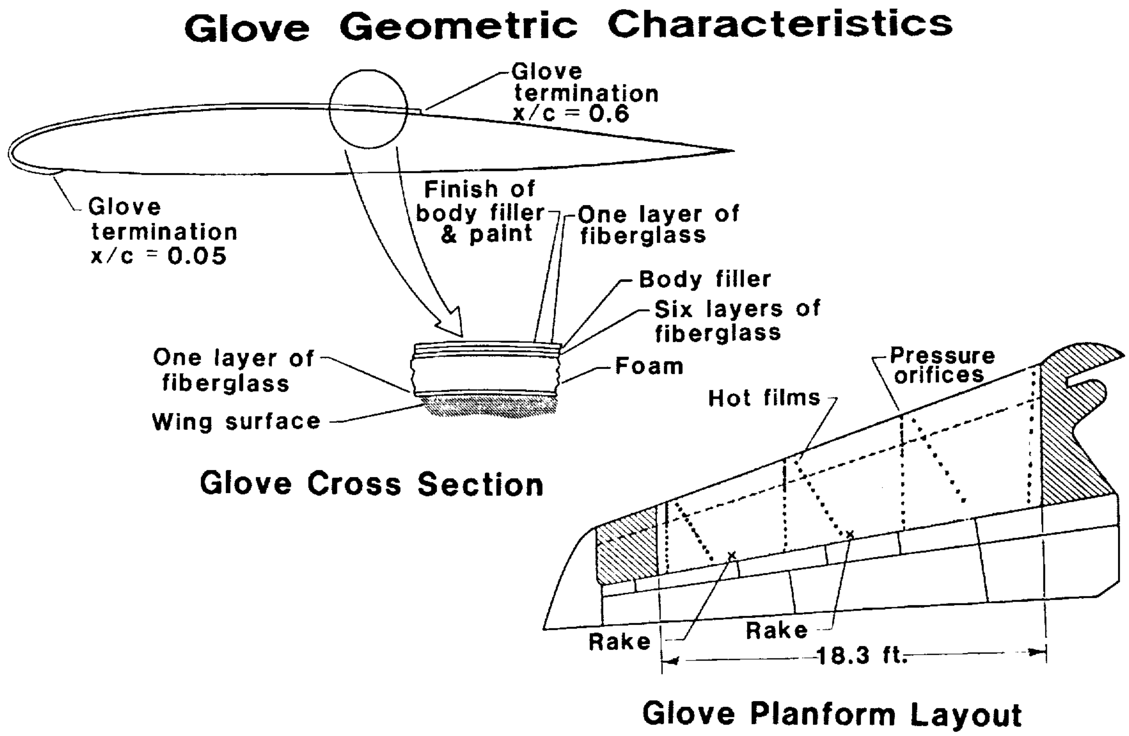

Figure 15. - F-14 VSTFE glove details.

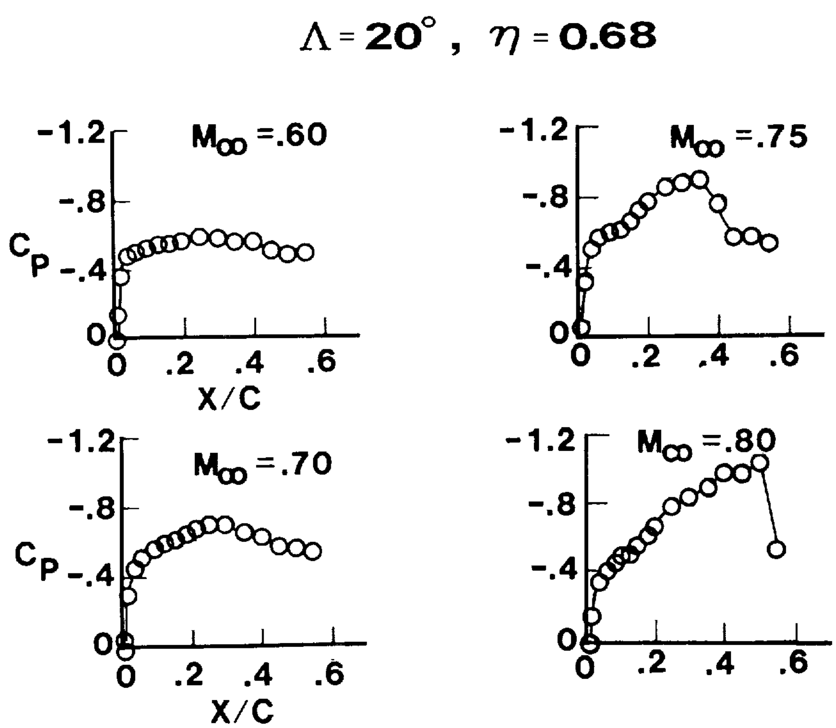

Figure 16. - F-14 VSTFE "clean-up" glove pressure distributions.

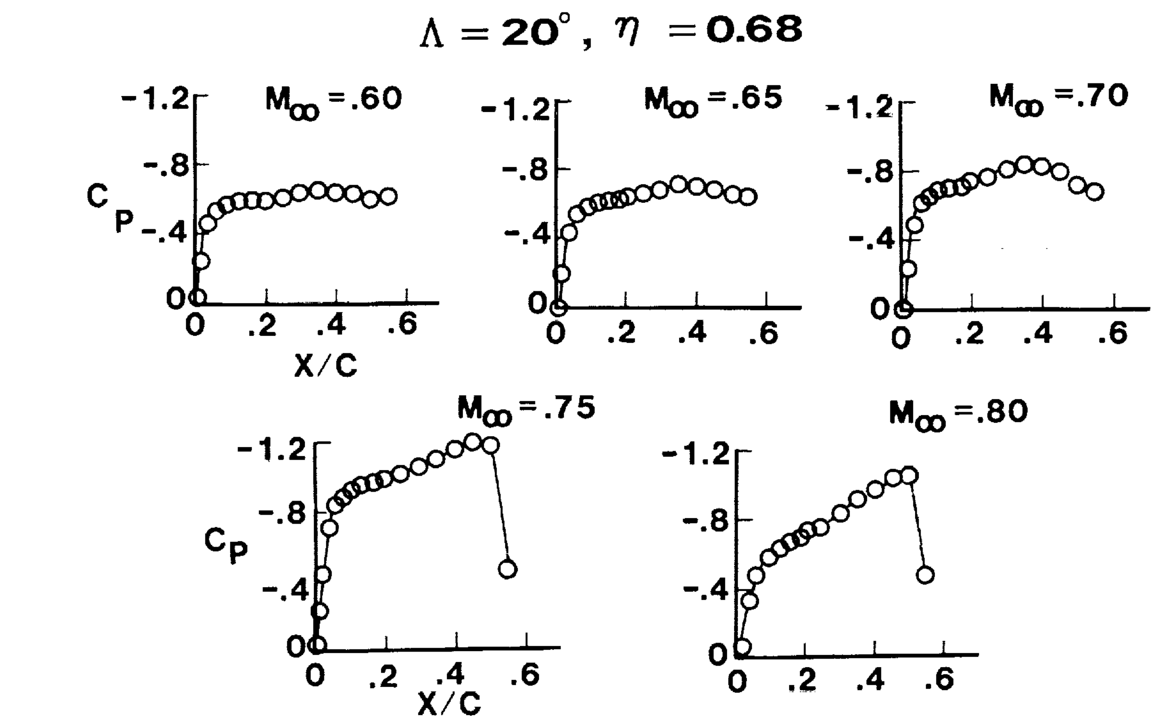

Figure 17. - F-14 VSTFE "Langley-design" glove pressure distributions.

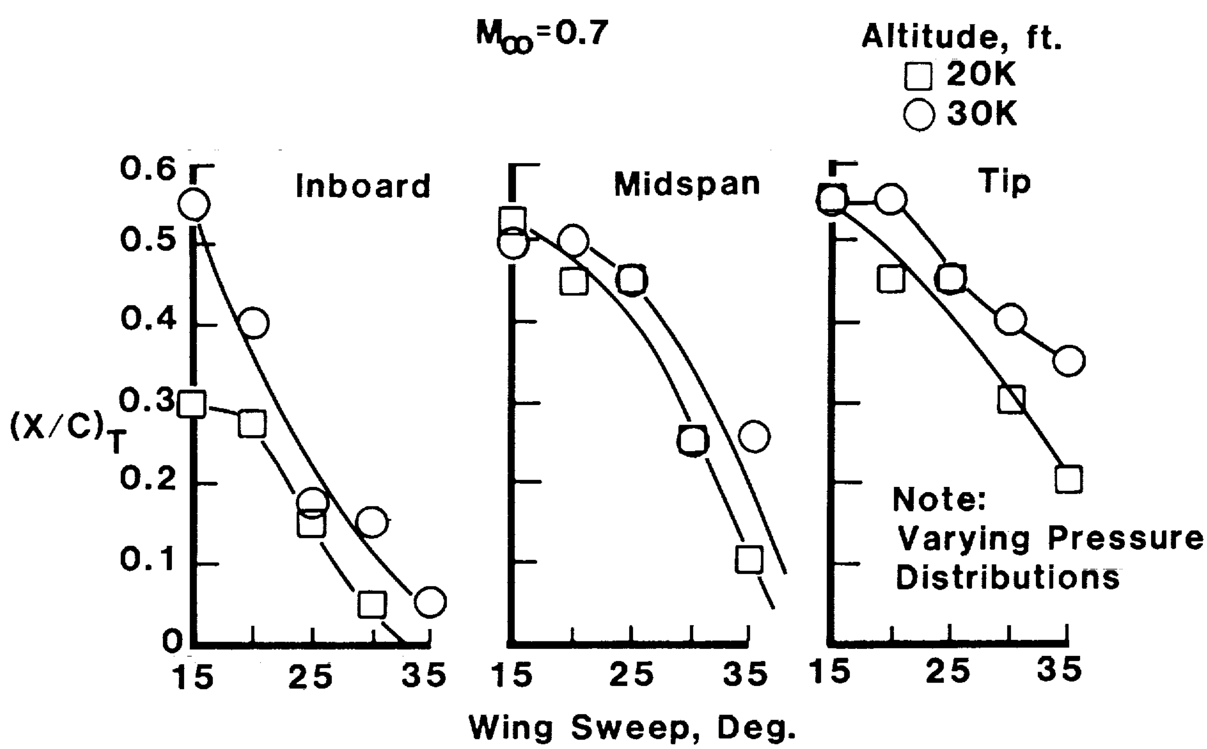

Figure 18. - Maximum transition location for the VSTFE "Langley-design" glove.

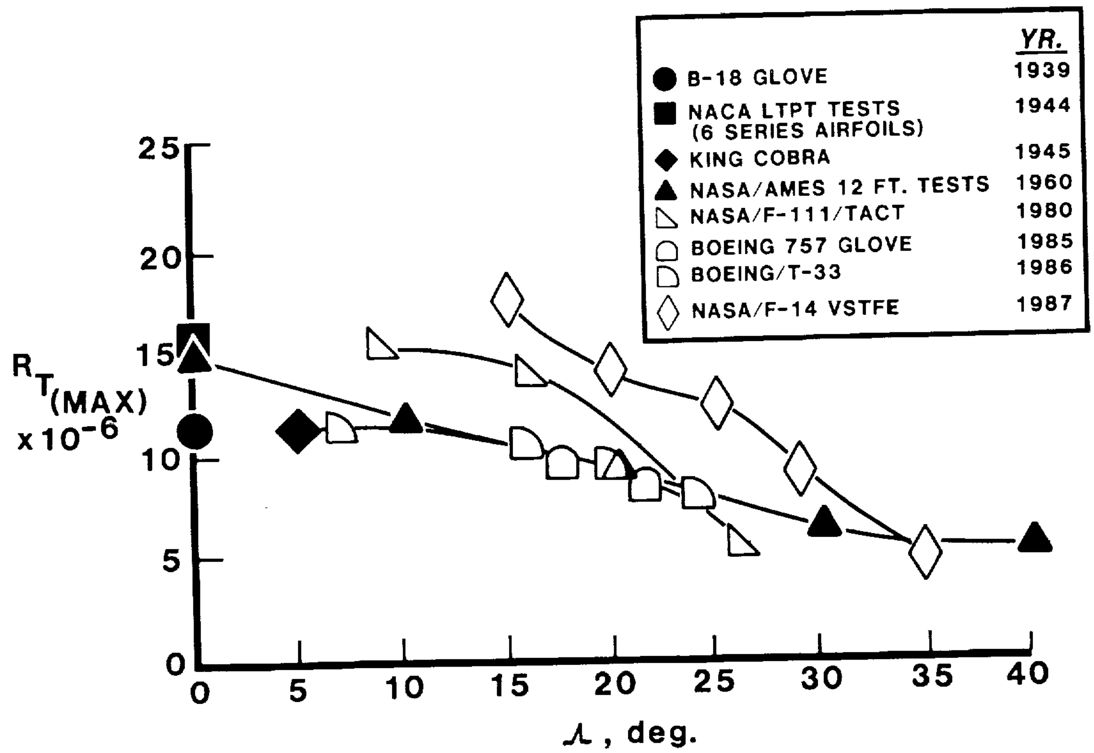

Figure 19. - Maximum transition Reynolds number for several natural laminar flow experiments.

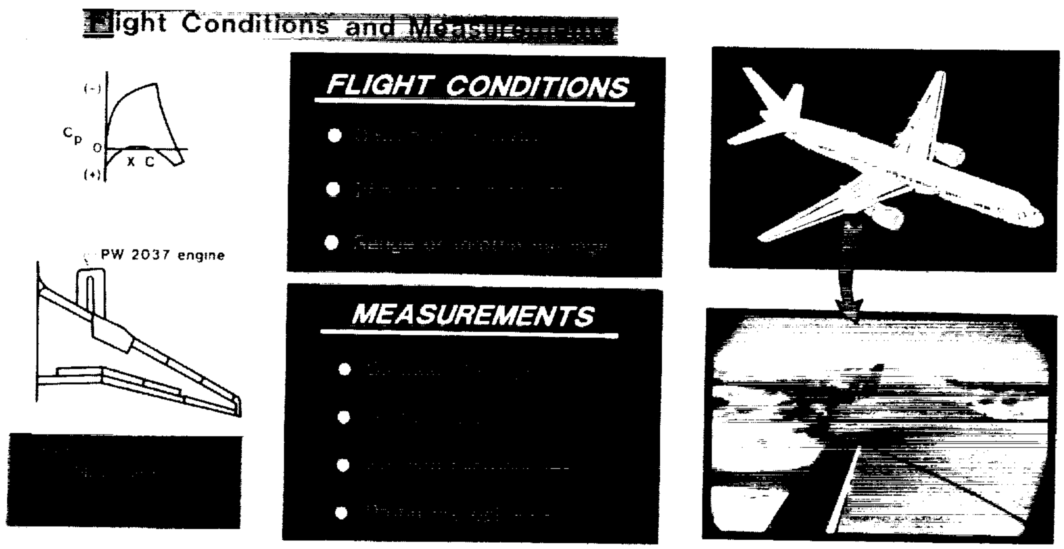

Figure 20. - The 757 Wing Noise Survey and NLF Glove Flight-Test Program.

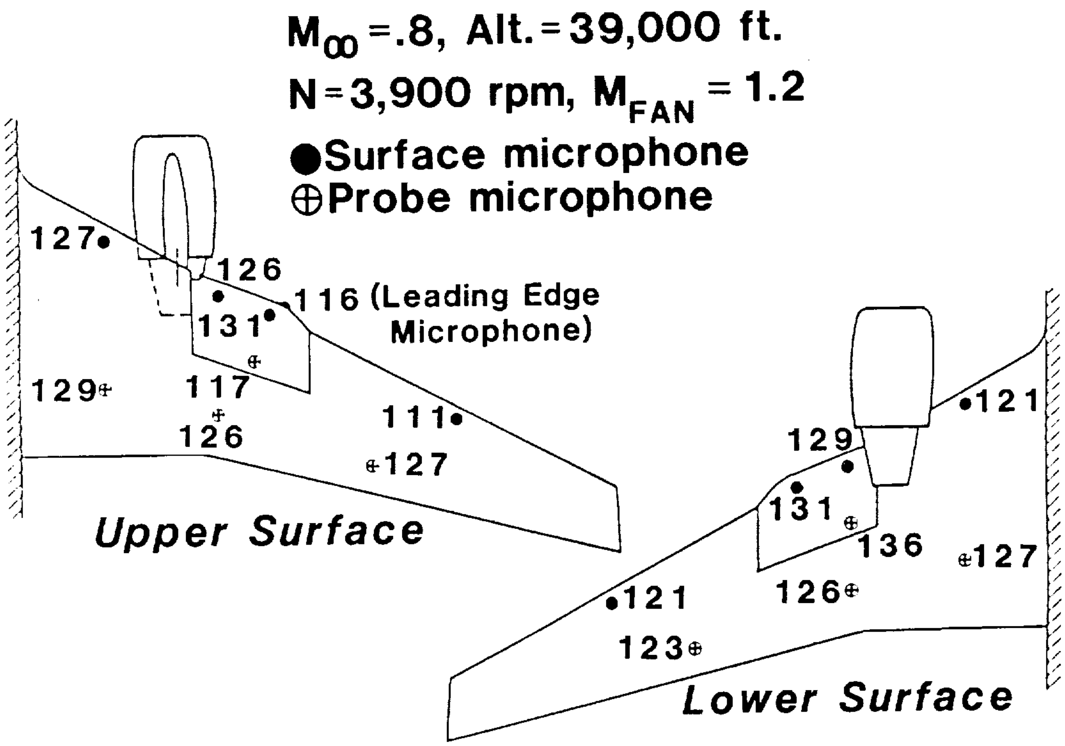

Figure 21. - Overall sound pressure level distribution on the 757 wing at cruise conditions.

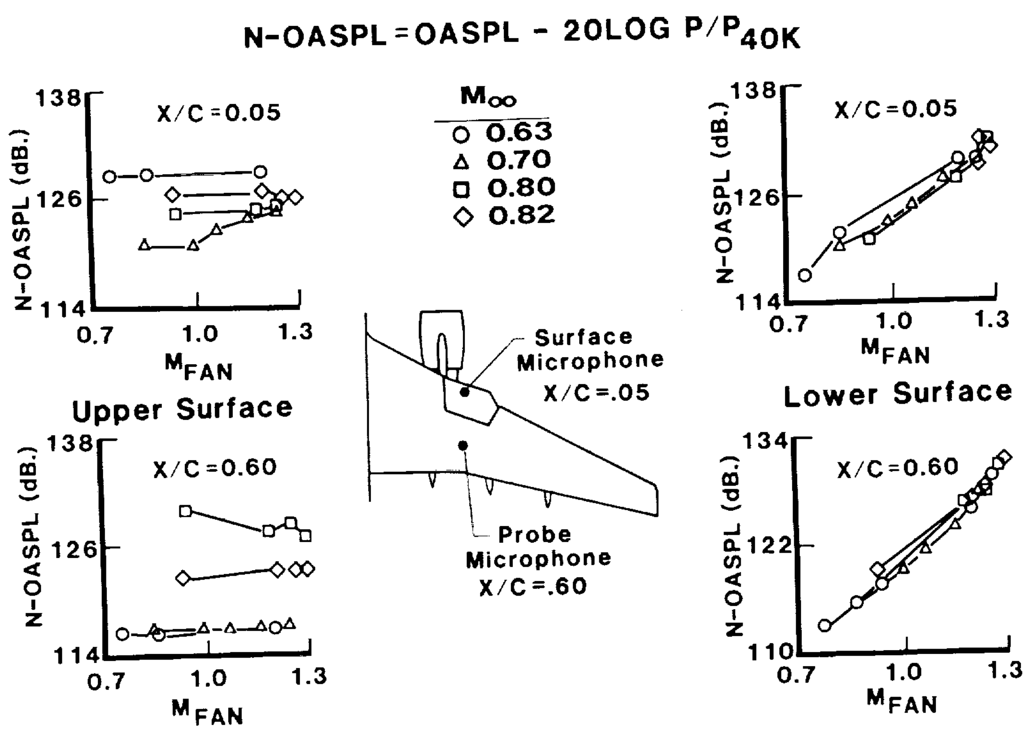

Figure 22. - Normalized overall sound pressure level versus fan Mach number for the 757 wing.

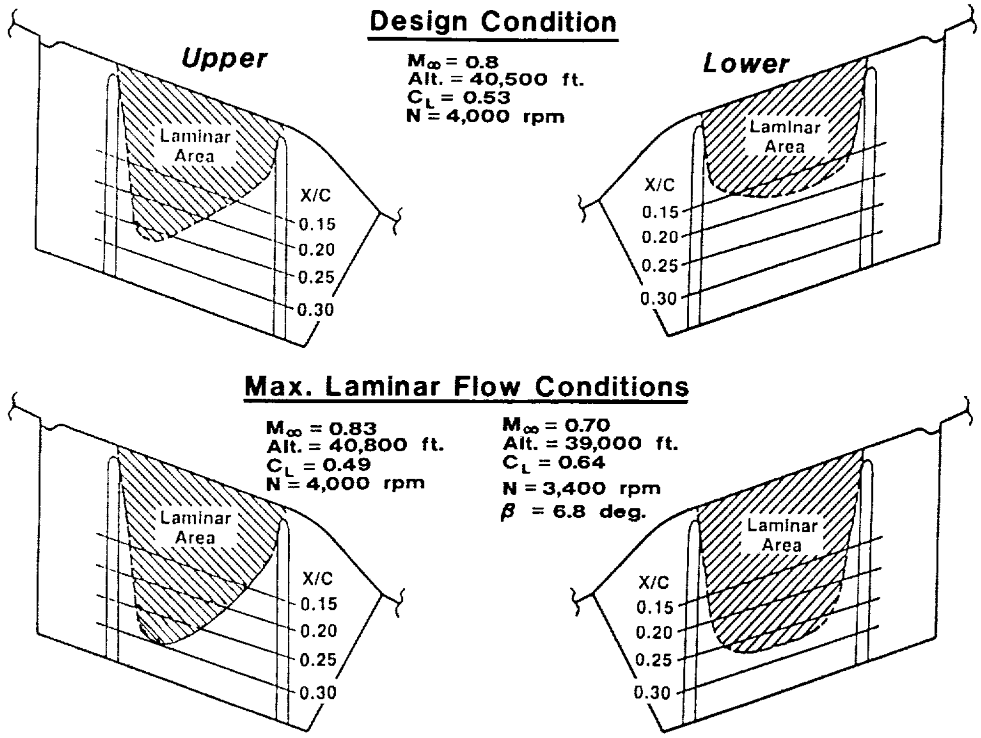

Figure 23. - Measured extent of laminar flow on the 757 NLF glove.

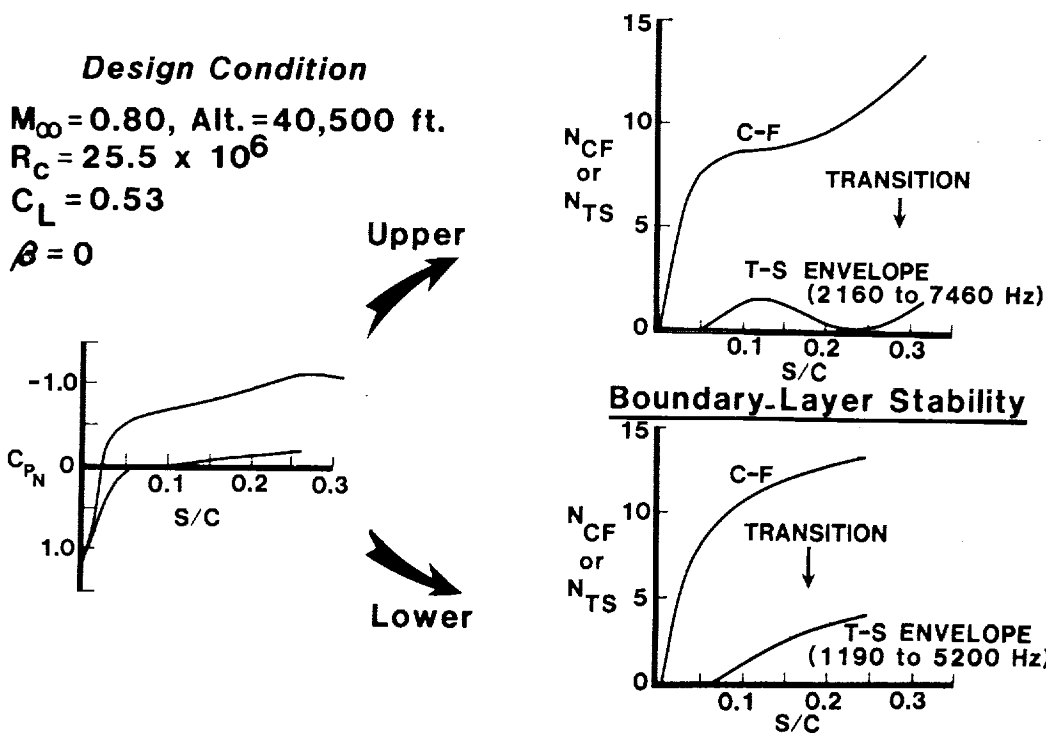

Figure 24. - Boundary-layer stability calculation at the design condition for the 757 NLF glove.

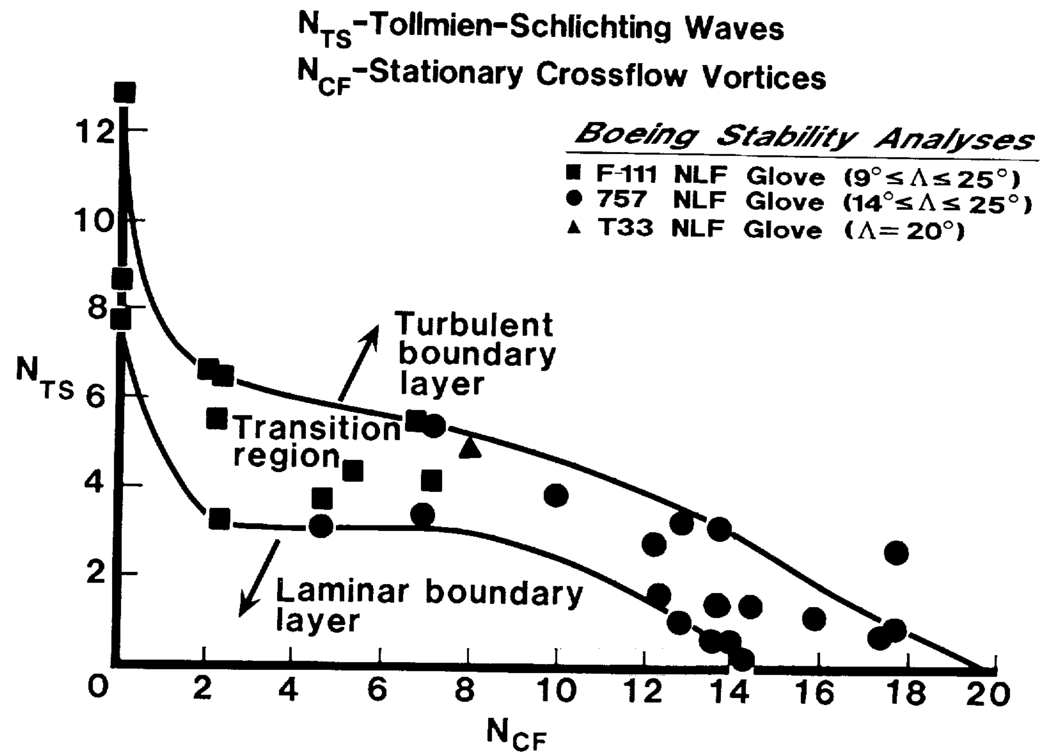

Figure 25. - Boeing stability analyses correlation of the Tollmien-Schlichting vs. crossflow N-factors with flight transition data.

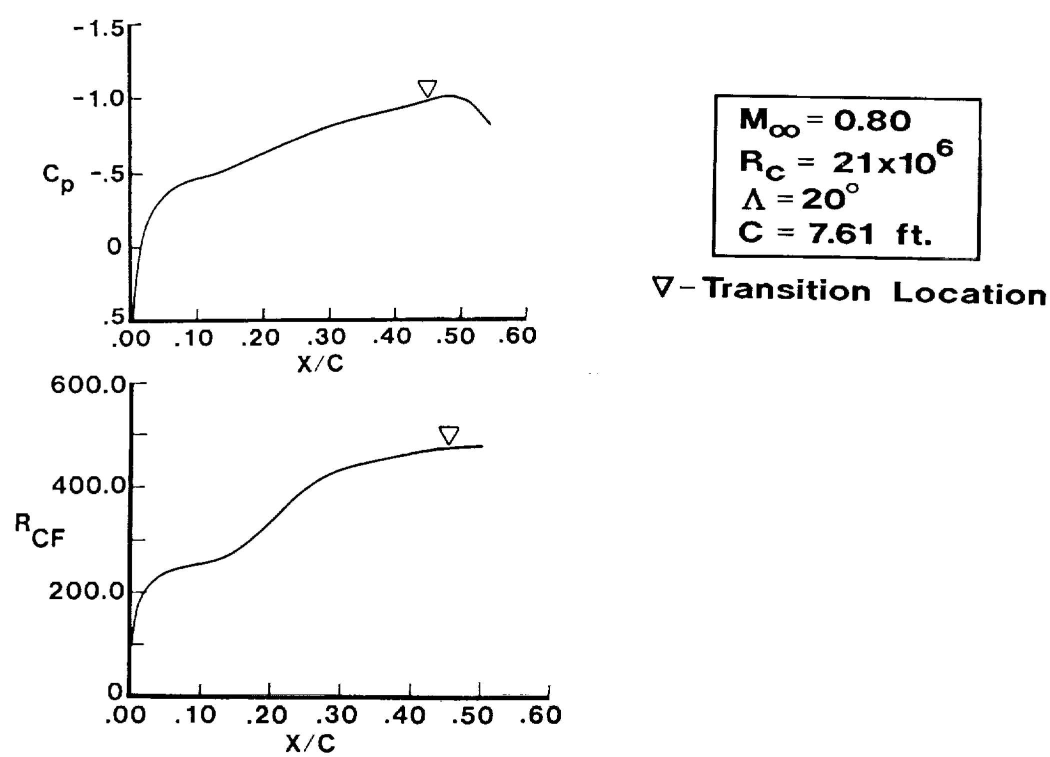

Figure 26. - F-14 "clean-up" glove pressure and crossflow Reynolds number distributions at a Mach number of .80 and a chord Reynolds number at 21 X 10^6.

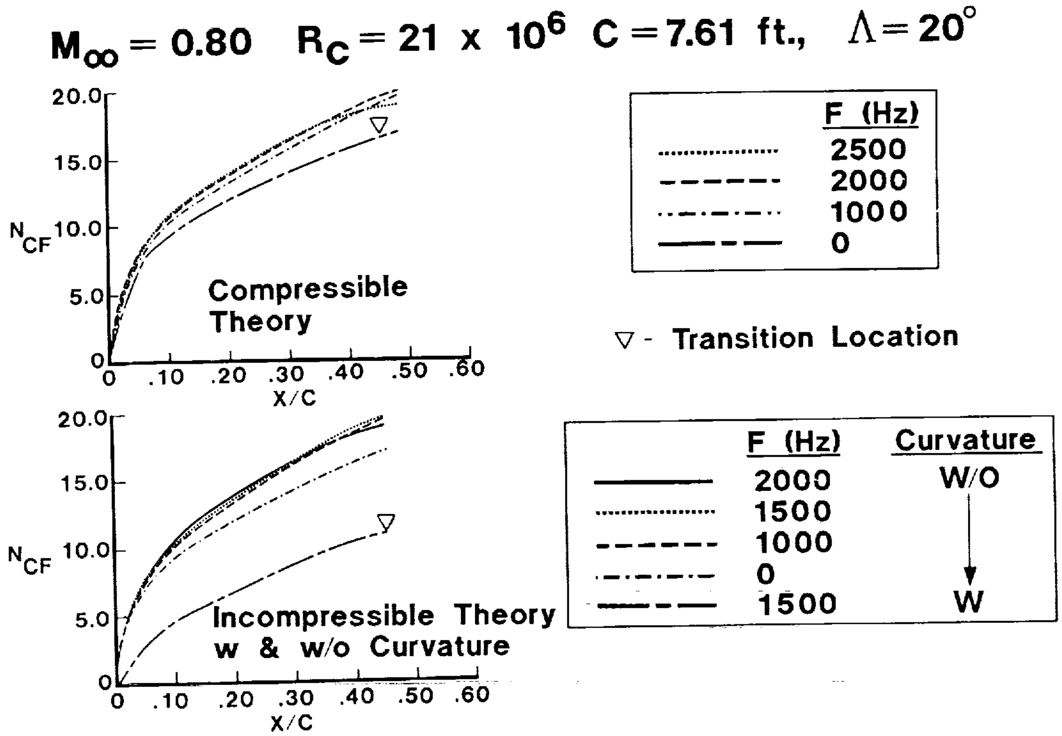

Figure 27. - Crossflow N-factors (with and without curvature effects) for the F-14 "clean-up" glove at a Mach number of .80 and a chord Reynolds number of 21 X 10^6.

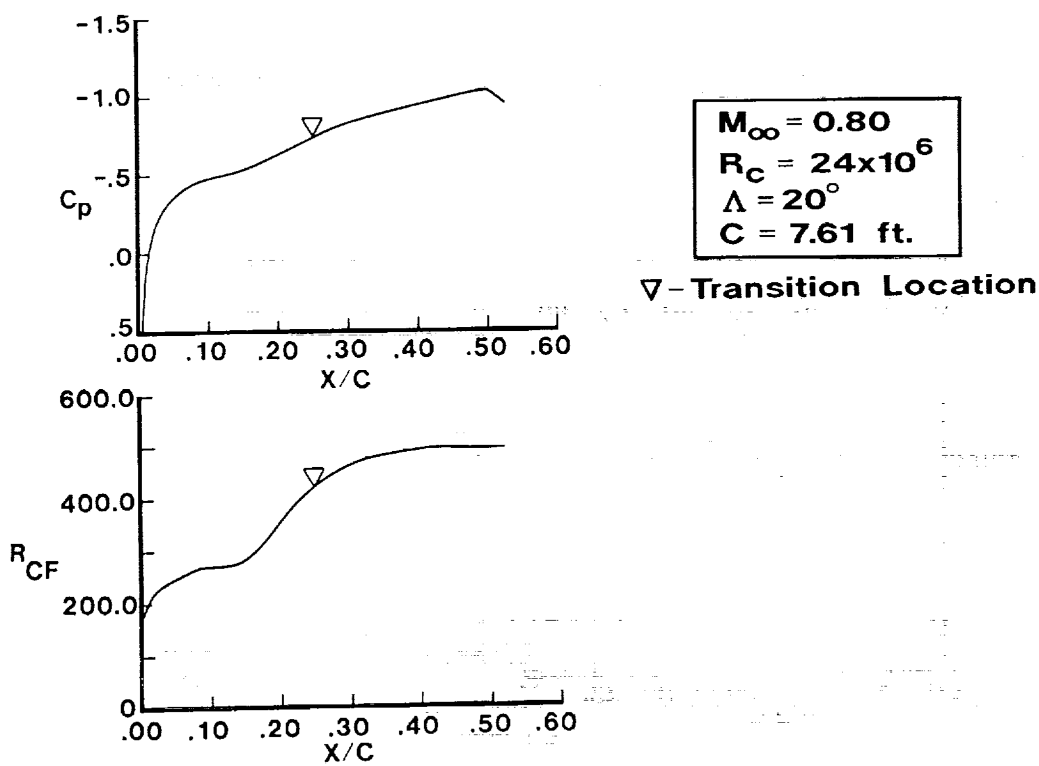

Figure 28. - F-14 "clean-up" glove pressure and crossflow Reynolds number distributions at a Mach number of .80 and a chord Reynolds number of 24 X 10^6.

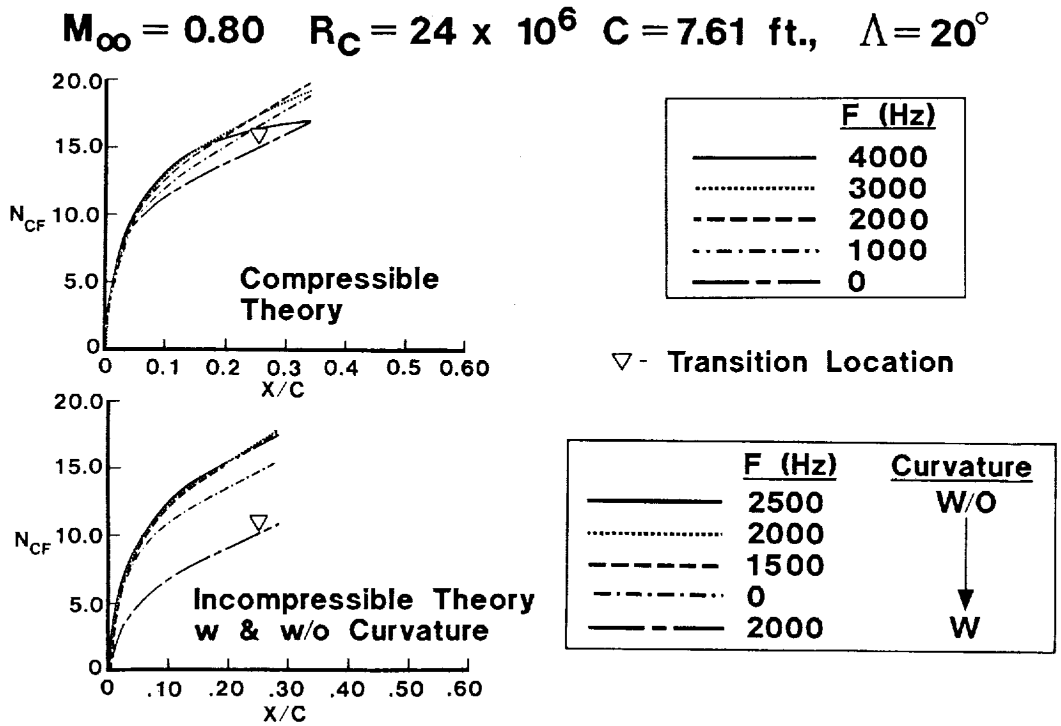

Figure 29. - Crossflow N-factors (with and without curvature effects) for the F-14 "clean-up" glove at a Mach number of .80 and a chord Reynolds number of 24 X 10^6

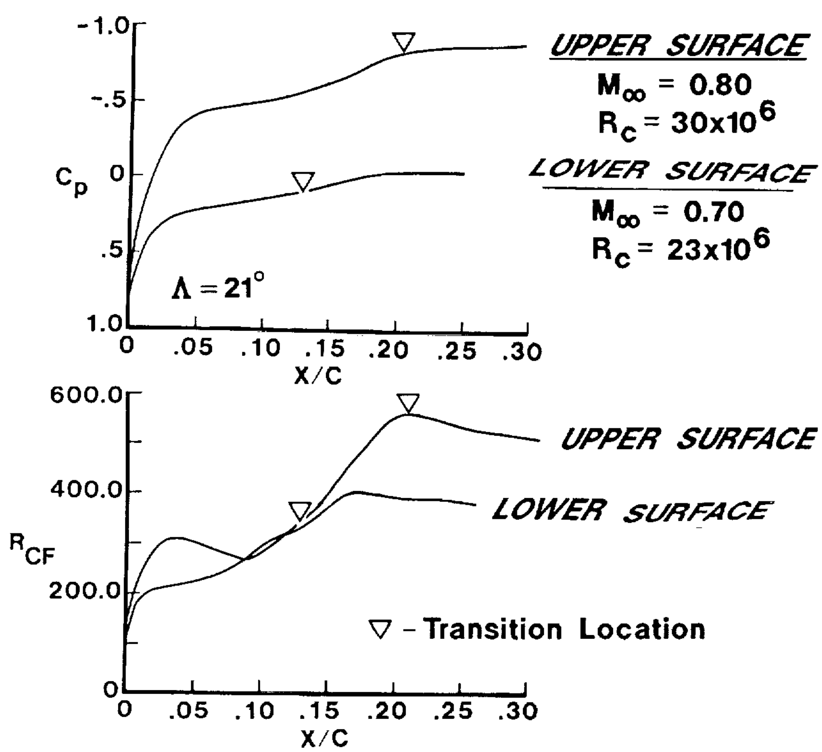

Figure 30. - 757 NLF glove pressure and crossflow Reynolds number distributions on the upper and lower surface at two flight conditions.

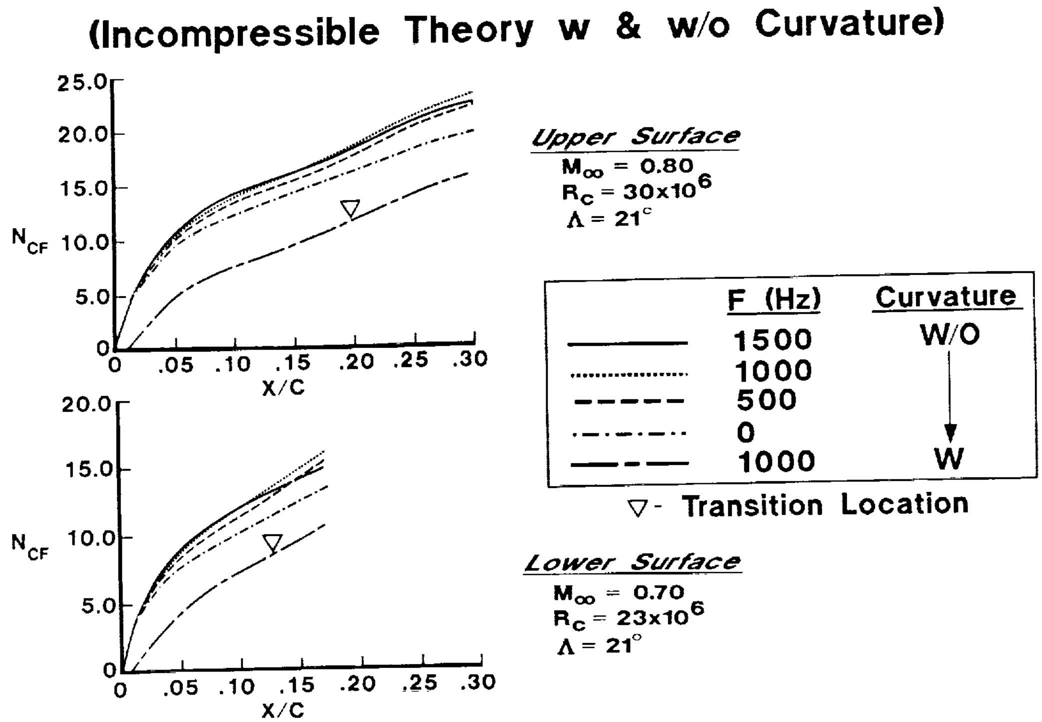

Figure 31. - 757 NLF glove crossflow N-factors (with and without curvature effects) on the upper surface and lower surface at two flight conditions.

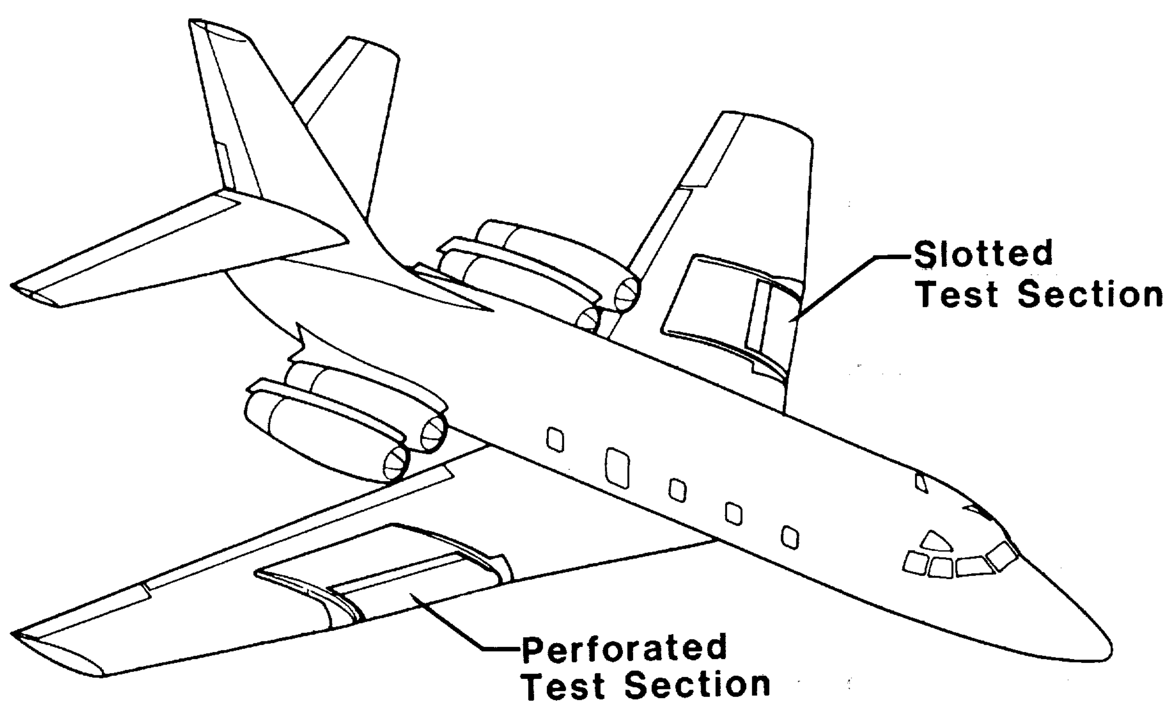

Figure 32. - JetStar test-bed aircraft for the NASA leading-Edge Flight Test Program.

- Suction on upper surface only
- Suction through electron-beam-perforated skin
- Leading-edge shield extended for insect protection
- De-icer insert on shield for ice protection
- Supplementary spray nozzles for protection from insects and ice

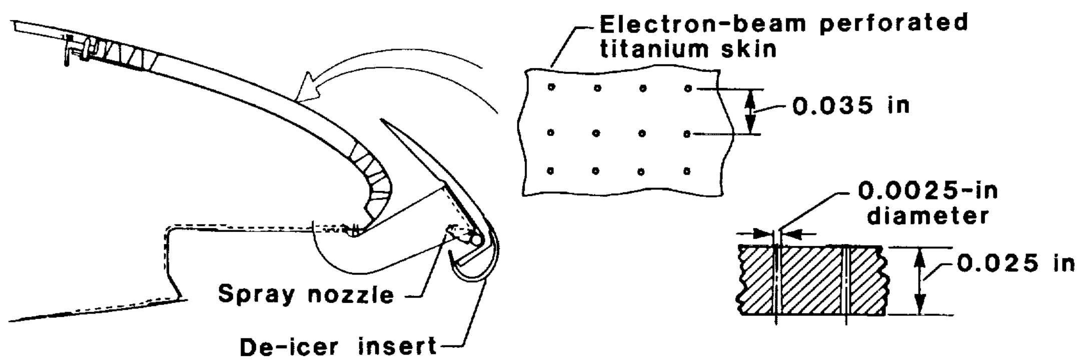

Figure 33. - The Leading-Edge Flight Test Program perforated test article.

- Suction on upper and lower surface
- Suction through spanwise slots
- Liquid expelled through slots for protection from insects and icing

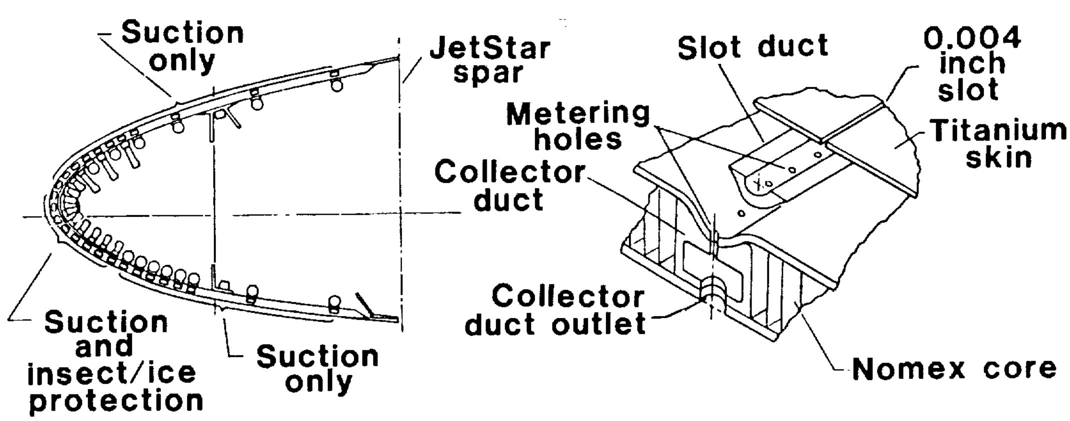

Figure 34. - The Leading-Edge Flight Test Program slotted test article.

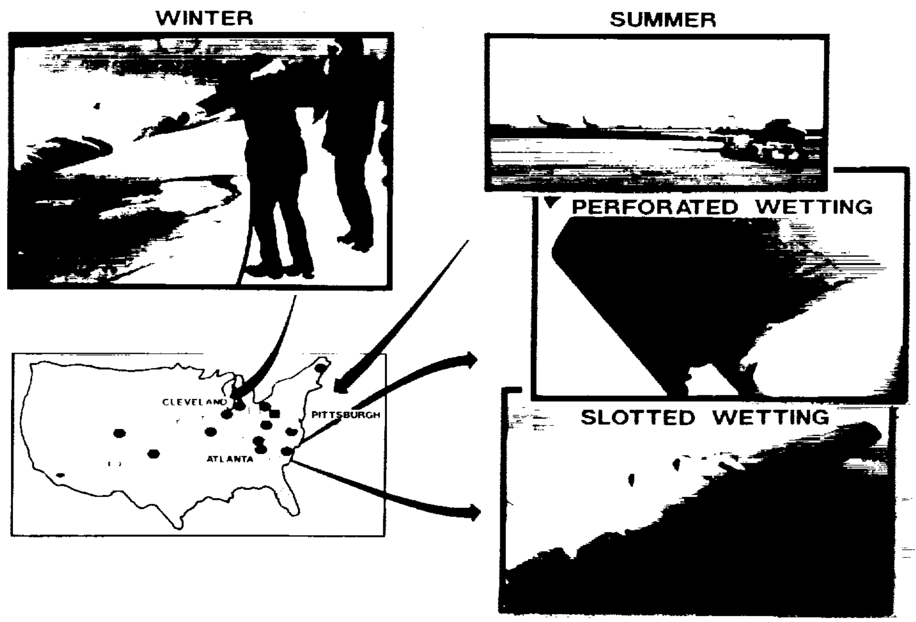

Figure 35. - The Leading-Edge Flight Test Program simulated airline service.

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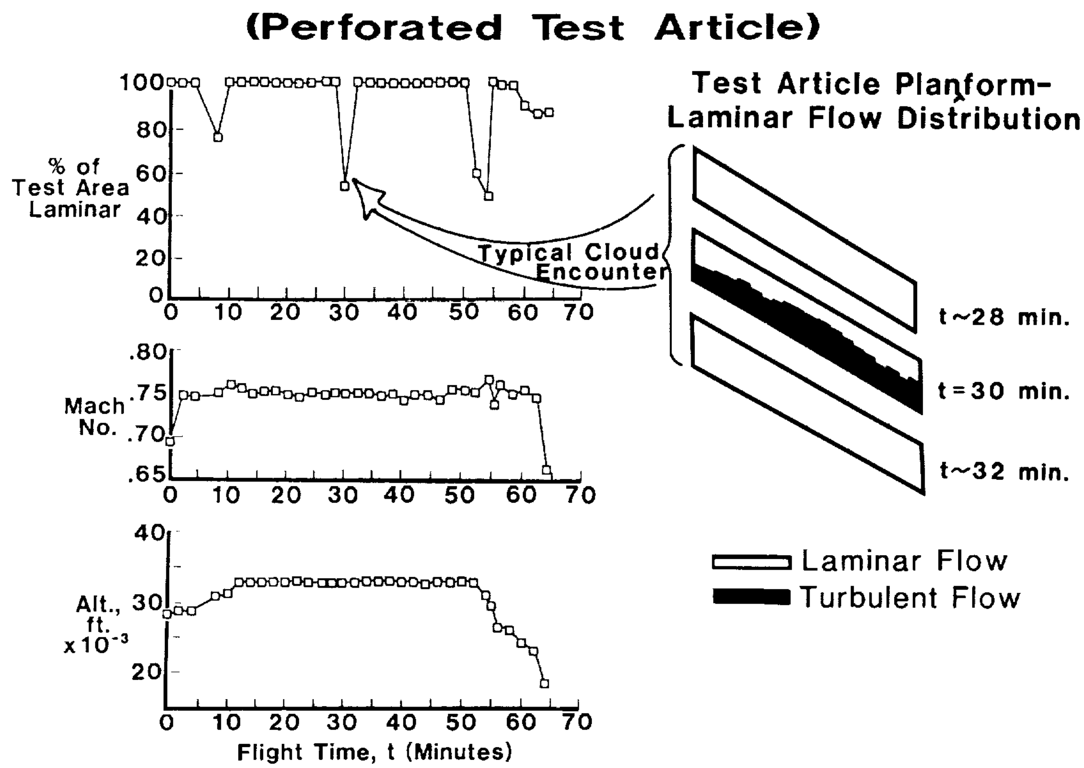


Figure 36. - Typical flight profile from the LEFT Program simulated airline service.

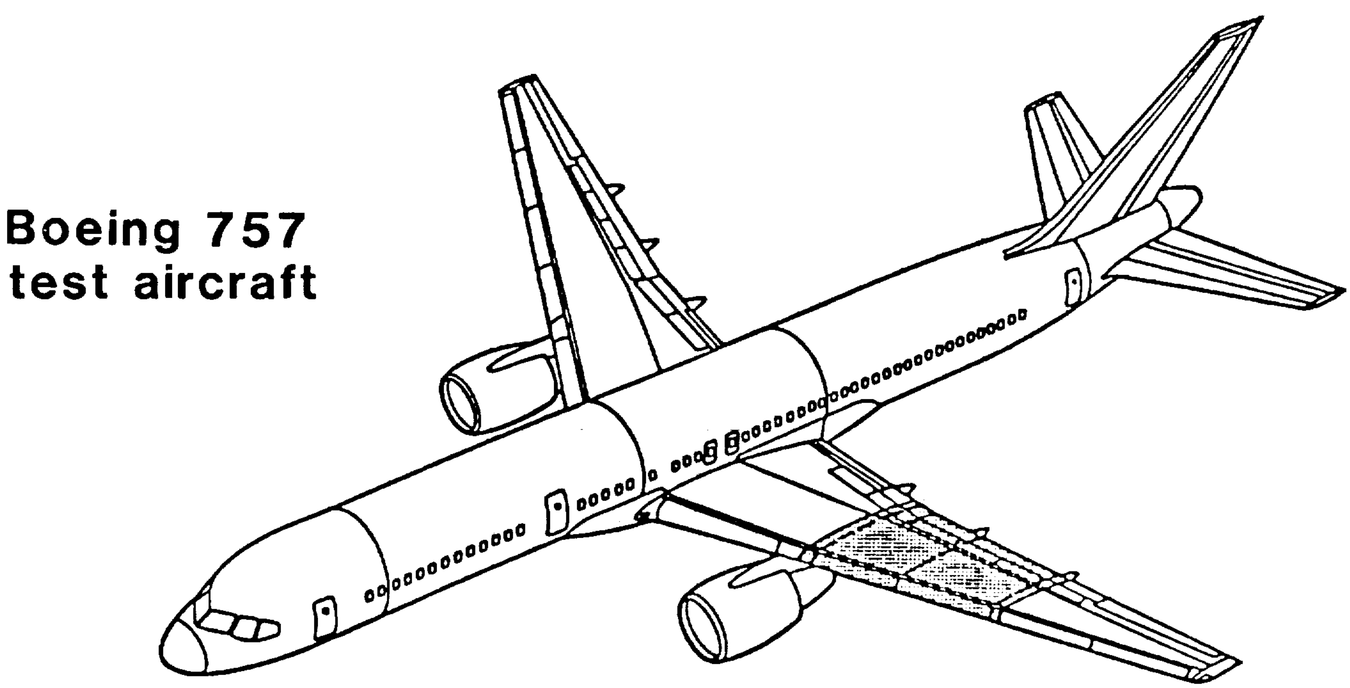
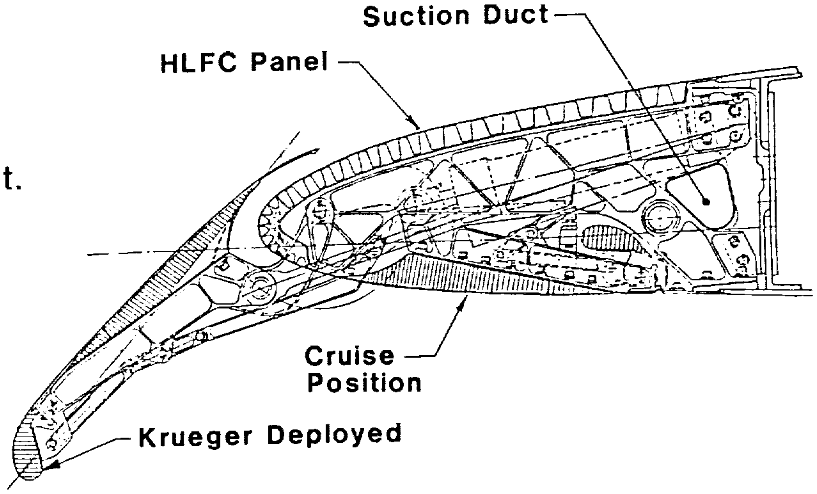


Figure 37. - The 757 test-bed aircraft for the Hybrid Laminar Flow Control (HLFC) Flight Experiment.

Technical Features:

- Cruise Conditions
  - $M_\infty = 0.8$
  - $R_C \geq 30 \times 10^6$
  - Alt.=34K to 42K ft.
- Laminar Flow, 38 to 62% chord
- Suction to Front Spar
- Microperforated Titanium Suction Surface
- All Metal Construction
- Operational Systems
  - Leading Edge Krueger/Insect Shield
  - Anti-icing



Milestones:

• Contract Award	Nov '87
• Critical Design Review	Dec '88
• First Flight	Feb '90
• Complete Program	Nov '90

Figure 38. - The HLFC Flight Experiment technical features and milestones.

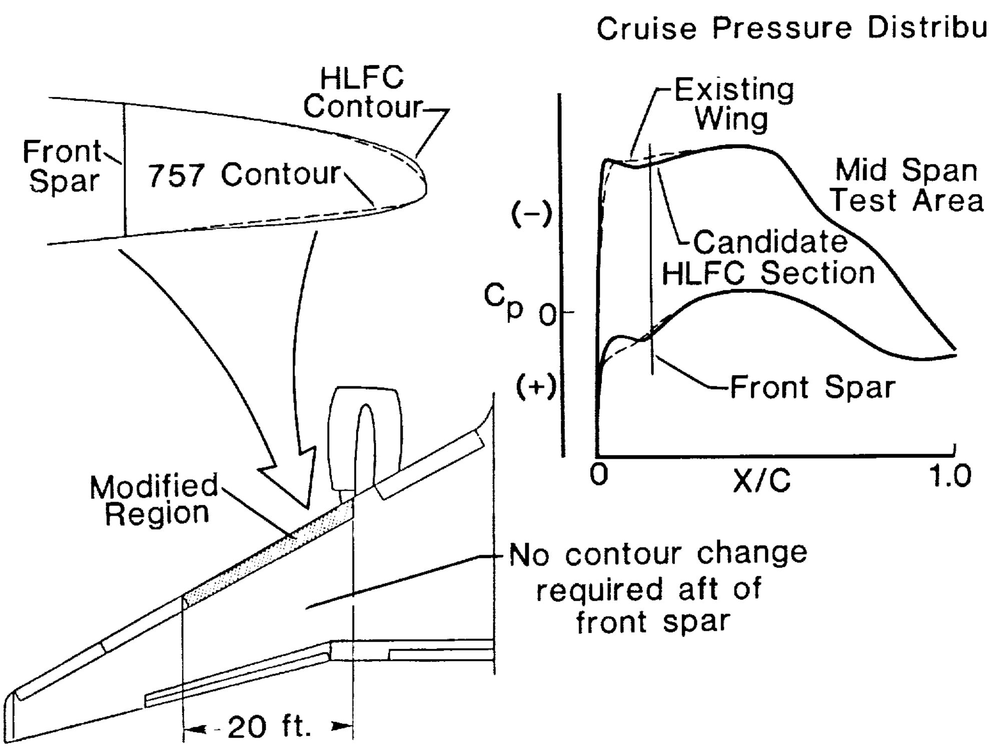


Figure 39. - The 757 wing leading-edge modification for the HLFC Flight Experiment.