Laminar Flow Control by Suction as Applied to the X-21A Airplane

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After many years of preparatory work in wind-tunnel testing and analysis, the X-21A, an 80,000-lb jet airplane with a laminar flow control wing, was designed and built by Northrop. Laminar flow control reduces wake drag on the wing by surface suction. The suction system consists of turbocompressor units that remove boundary-layer air from the wing surface by many fine slots in the skin. Laminar flow on the outer halves of the wing was established very early in the flight test program and, after eliminating spread of turbulent disturbances along the attachment (stagnation) line, essentially the whole "possible" area was made laminar with predicted suction quantities. The gain due to laminar flow control on the wing of a big, long-range transport airplane is expected to be 35% in range at the same payload and a similar percentage in payload at the same range.

Introduction

IN the effort to reduce the skin friction, the last remaining major part of airplane parasitic drag, by "laminar flow control" full chord laminar flow in free flight, was demonstrated several years ago on a wing glove of an F-94 airplane. To demonstrate laminar flow by suction on a full-scale high-subsonic airplane with swept wings, the X-21A was designed and built by Northrop. This paper gives, in a very abbreviated manner, the background, the X-21A concept, and some of the flight test experiences and results to date.

Background

Airfoil Shape

In the years around 1940, efforts were made in different countries to achieve laminar flow by another form of "streamlining." An attempt was made to maintain laminar flow through appropriate airfoil shaping. However, it was soon found that the stability of the natural laminar boundary-layer profile at higher Reynolds numbers became too low to withstand even very small disturbances.

Suction

Unswept wings

From theoretical analysis, it was learned that removing the innermost part of the boundary layer with very small amounts of suction would increase the stability of a laminar boundary-layer profile substantially. The highest degree of stability could be obtained, according to this theoretical analysis, by continuous suction through a porous skin. Recognizing that a porous skin would be a major structural obstacle, researchers investigated the possibility of accomplishing suction in discrete steps. The objectives of these investigations were to determine the number, spacing, and shape of openings, and the amount of boundary-layer air to remove.

An investigation of these questions was made in the early 40's by Pfenninger in the Aerodynamic Institute of the Technical University at Zurich. He obtained full chord laminar flow on both sides of a 10.5%-thick airfoil with 14 suction slots on the upper side and 10 slots on the lower side. Laminar flow was recorded up to a maximum chord Reynolds number of 2.5×10^6 with a drag reduction of about 60%, including the pumping power, as compared to a turbulent airfoil.[1] In spite of the relatively low limiting Reynolds number, the fact that laminar flow was maintained through the strong pressure rise at the rear of the airfoil is significant.

Tests with a similar arrangement in the NACA Langley Field tunnel yielded full chord laminar flow on a 15% thick airfoil up to a Reynolds number of 17×10^6. The lower surface had 13 slots and the upper surface had 17 suction slots covering the area from the 40% chord to the trailing edge.[2]

A NACA test was conducted on a 10%-thick airfoil with 41 slots on the upper and lower sides. Nearly full chord laminar flow was obtained up to a Reynolds number of 10×10^6.[4]

A wind-tunnel test on a 4%-thick airfoil of 17 ft chord concluded the experiments on laminarization of unswept wings.[3] Full chord laminar flow at a Reynolds number of 25×10^6 was obtained in the Northrop Norair wind tunnel. The limiting factor was probably tunnel turbulence.

F-94 tests

A later flight test with a wing glove of 7.5 ft chord on the upper side of an F-94 airplane with 12 slots from 40% chord to the trailing edge provided full chord laminar flow up to a Reynolds number of 30×10^6.[5] A later version with a greater number of slots, 69 instead of 12, resulted in full chord laminar flow to a Reynolds number of 36×10^6.[6] During these tests locally, the speed of sound was exceeded by 5 to 8%, demonstrating that the stability of the laminar boundary-layer profile was high enough to overcome the disturbances of the weak shock. The 69-slot glove for the first time incorporated very narrow rectangular suction slots normal to the surface instead of the wide forward-pointing diffuser-type slots used in the early versions. Table 1 gives a summary of the tests discussed so far.

Swept wings

The need for high-subsonic speed for high jet engine propulsive efficiency makes a swept wing almost mandatory. However, flight experience has shown that on swept wings the transition point is considerably farther forward than on unswept wings. This phenomenon was investigated in detail by British observers Gray[7] and Owen and Randall.[8] They concluded that the cause for the earlier transition on a swept wing was an unstable boundary-layer profile in the direction normal to the potential flow streamline. This unstable profile is created by a crossflow normal to the potential streamlines in a plane tangential to the wing surface. The crossflow is strongest where there is a high curvature of the streamlines

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in the tangential planes, i.e., in the immediate neighborhood of the leading edge and in the region of the rear pressure rise. This is illustrated in Fig. 1. Brown of the Northrop Norair Boundary-Layer Research Group computed analytically the stability limits of these crossflow boundary-layer profiles and the amount of boundary-layer air which must be removed to stabilize these profiles.[9] The boundary-layer development on swept laminar suction wings was calculated by means of Raetz's method.[10]

A 30° swept wind-tunnel model with 7 ft chord, 12% thickness, and 93 suction slots from the 0.5% chord to the trailing edge was tested in the Northrop Norair 7 × 10 ft wind tunnel and in the Ames 12-ft pressure tunnel [11,12] to verify the calculations of Brown and Pfenninger. At lower Reynolds numbers, very little or no suction near the nose was necessary. With increasing Reynolds numbers, more and more suction near the leading edge was needed to stabilize the boundary-layer profiles. The amount of suction required on a swept wing was not dictated so much by the boundary-layer profiles in the direction of the potential flow (as in the case of the straight wing), but by the crossflow profiles. Fortunately, the additional suction required was not substantial. In the Ames wind-tunnel test, it was possible to maintain full chord laminar flow up to a Reynolds number of 29×10^6 with a C_Q of 0.0007 for upper and lower sides combined; that is, a band of air of the span of the wing and a thickness of 0.07% of the chord had to be sucked into the wing in order to achieve a drag coefficient of 0.001, including the pumping work; that is, a fifth of the turbulent friction drag of a flat plate wetted on both sides. It can be supposed that laminarization during these tests was limited, not by the fundamental behavior of the flow but by the wind-tunnel noise, or more precisely, by the wind-tunnel turbulence as expressed in noise.

Since noise is nothing more than presssure fluctuations in air, fluctuations that require the transport of air particles in a certain time (depending on frequency) over a certain distance (depending on amplitude), the presence of a specific noise in a flow is representative of a certain turbulence $\Delta V/V$. This turbulence may become pronounced enough to cause transition from laminar to turbulent flow. Comprehensive investigations and tests[13–15] have helped to clarify the relationship between stability of the laminar boundary layer and noise. On the strength of these investigations and tests, limits can be established for the sound pressures on laminarized parts. As a guide, it might suffice if it is said that at a chord Reynolds number of 30×10^6 a noise level of 110 db is tolerable. Present investigations seem to indicate that the limit might be higher. Major noise sources are the powerplant (compressor, turbine, and jet), the turbulent boundary layer, areas of separated flow, and possibly the suction flow inside ducts and valves.

X-21A Program

The stage was now set for an application of laminar flow control on a wing with 30° sweep and about 30×10^6 Reynolds number under cruise condition. Under government contract, two WB-66 airplanes were adapted to a 30° sweep aspect ratio 7 wing with an area of 1250 ft². The wings incorporated narrowly spaced fine laminar flow control suction slots. These aircraft are designated X-21A.

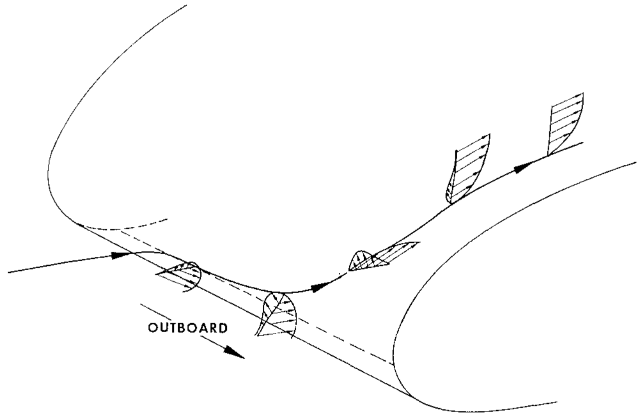

Fig. 1 Streamline and boundary-layer profiles.

Figures 2 and 3 show the arrangement of the airplane which, in many details, is heavily compromised because of the use of existing parts and equipment. The main propulsion engines, nonafterburning versions of the J79-3A engine, are suspended on pylons on the rear fuselage. This location was chosen to reduce the noise and the ensuing turbulence level on the wing. A fan engine on pylons on the wing would have resulted in approximately the same sound pressure level on the wing, but fan engines of the appropriate size were not in the Air Force inventory at the time this program was initiated.

The suction compressors, laminar flow control (LFC) pumps, are located in nacelles underneath the wing at 32% half-span. The boundary-layer air is removed from the wing surface by approximately 120 slots on the upper surface and 120 slots on the lower surface. The slots vary in width from about 0.0035 to 0.01 in. and discharge through 48 collector ducts into the suction compressors. The installation of the suction nacelles at about one-third half-span permits the collector ducts to be relatively short and the duct velocities low. It also permits separate adjustment of the suction quantities for the inboard and outboard wing.

The suction pump developed and manufactured by AiResearch is shown in Fig. 4. It consists of a small compressor with a nominal compression ratio of 1.2 and a large compressor with a nominal pressure ratio of 1.65. The former sucks the air from the low-pressure area of the wing, i.e., just behind the stagnation point to about 70% chord on the upper side. This air is discharged together with the large amount of boundary-layer air from the remainder of the wing into the mixing chamber in front of the large compressor. The total boundary-layer airflow is ejected through a nozzle at the end of the suc-

Table 1 Laminar flow control tests

Year	Test facilities	Airfoil thickness ratio	Chord Reynolds number	Unit Reynolds number	References
		Straight wing			
1944–1945	Zurich 7 × 10 ft	0.105	2.5×10^6	1.74×10^6/ft	1
1951	Langley TDT 3 × 7½ ft	0.15	17×10^6	3.4×10^6/ft	2
1951	Langley TDT 3 × 7½ ft	0.10	10×10^6	3.3×10^6/ft	4
1954	Flight	0.13	30×10^6	4×10^6/ft	5
1956	Flight	0.13	36×10^6	4.8×10^6/ft	6
1959–1960	Northrop 7 × 10 ft	0.04	25×10^6	1.47×10^6/ft	3
		30° Swept wing			
1956–1959	Ames 12-ft pressure tunnel	0.12	29×10^6	4.5×10^3/ft	11 and 12

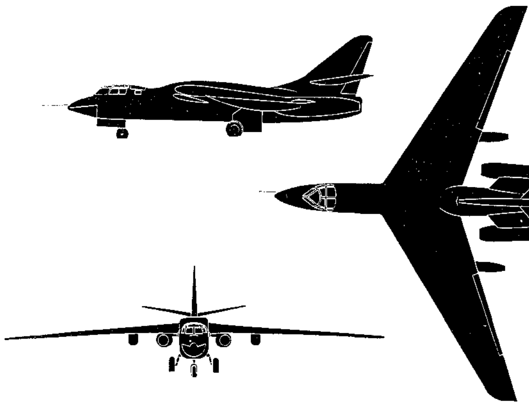

Fig. 2 3-view LFC general arrangement.

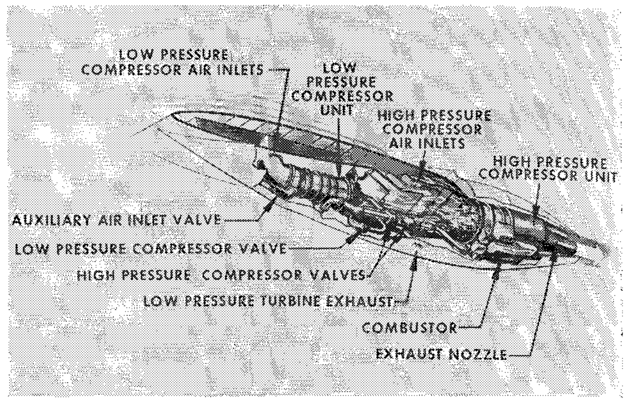

Fig. 4 Laminar flow pumping systems.

tion nacelles with an exhaust speed of about 70% of the flight speed. The pumping units are driven by compressor bleed air from the J79 engines. In the high-pressure compressor the bleed air is heated in a combustion chamber. Because of the lower power level and in the interest of simplicity, the low-pressure unit uses bleed air that is not reheated. This arrangement gives a relatively high degree of flexibility and insures a high propulsion/suction system efficiency.

The LFC compressor system in effect is a fan converting a jet-propulsion engine into a turbofan. In the case of a fan engine, the LFC compressor system in effect increases the bypass ratio. It has been shown that the compressor bleed drive is thermodynamically identical to an additional turbine stage.[16] Reference 16 also shows that if the pumping compressor were driven with its own independent powerplant a lower total efficiency would result. The use of a fan for expelling boundary-layer air for propulsion provides the same advantages as in the case of a ship and its propeller operating in its wake as treated first in clear mathematical terms by Fresenius.[17] In this way, laminar flow control by suction improves the propulsive efficiency in addition to reducing friction drag.

Because of the relationship of drag and propulsion, it is convenient in the performance analysis to treat the LFC wing as a jet-propulsion system. The ram drag of the wing is computed from LFC mass flow and flight velocity, and LFC gross thrust is computed from LFC mass flow and exhaust velocity; external drag is treated as a wake drag, and total fuel flow required is computed by combining fuel flow for pumping with the fuel flow of the main propulsion systems including the penalties for compressor bleed.

The data given earlier for airfoil drag, consisting of wake drag and pumping drag, cannot be applied directly for this type of analysis. They are derived with the assumption of equal efficiency for propulsive and pumping system and are given for comparison purposes only.

The lower drag penalty of a laminarized wing compared to a turbulent wing permits use of a larger wing area or a larger span at the same aspect ratio; i.e., the induced drag can be reduced, or with the same induced drag the flight altitude can be increased and, therefore, the dynamic pressure reduced. The optimization process of wing area and span is, as with a turbulent airplane, a rather complex process of trading off drag and weight penalties.

The weight penalty for the laminarization of the X-21A wing was kept low by using the duct material for structural strength. Figure 5 shows a typical cross section of the wing. The front and rear parts of the wing are of continuous truss construction and weigh approximately the same as a wing of conventional skin and rib construction. The main box between front and rear spar is structurally very similar to a conventional main box with heavy skin and stringers.

The additional inner skin that separates suction ducts and fuel tank also contributes to the strength and stiffness of the wing. With this type of design, the wing weight penalty for full chord laminarization of a transport-type airplane with a wing loading of about 100 psf is between 0.7 and 0.8 psf for the projected wing area.

The weight of the pumping equipment, i.e., turbines, compressors, bleed pipes, ducts, elbows, valves, and suction pods, is estimated to be between 1.3 and 1.4 psf of the projected wing area for laminarization of a wing of about 100 psf wing loading for a transport-type airplane in the 300,000-lb class.

Performance analyses show that the required engine size is smaller than that required for a turbulent counterpart, considering both cruise and takeoff. This smaller engine size for a given gross weight or the higher gross weight for a given engine size results in a weight-saving that offsets the weight penalty for the pumping equipment.

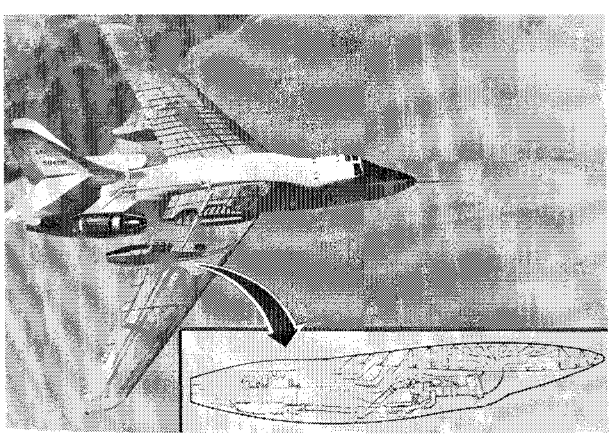

Fig. 3 X-21 airplane.

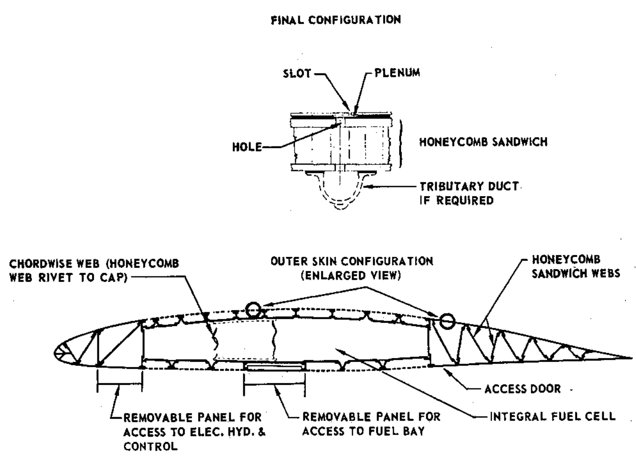

Fig. 5 LFC typical wing section.

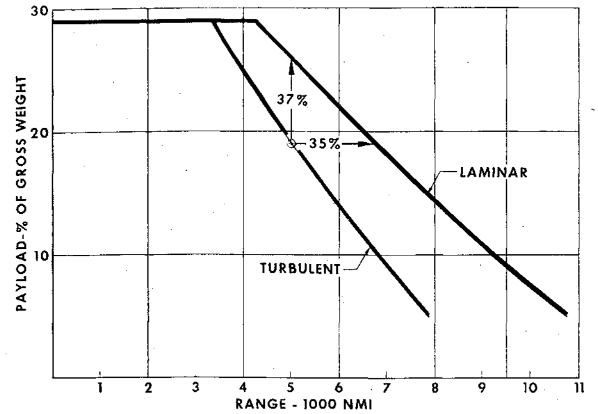

Fig. 6 Payload vs range.

Balancing the variations of weight, drag, and specific fuel consumption results in an optimized design with an increase in range of one-third or an increase in payload of one-third at a design range of 5000 naut miles for a laminar transport-type airplane (with turbulent flaps and ailerons) against its turbulent counterpart. This is shown in Fig. 6.

Flight Test Experiences

Full chord laminar flow was established with comparative ease on the outer third of the X-21A wing very early in the flight test program. The observed turbulent flow over the inboard two-thirds of the wing was caused by spanwise turbulent contamination of the flow along the front attachment line, sometimes called stagnation line, of the wing, originating from the turbulent boundary layer at the wing fuselage intersection, as first suspected by Pfenninger.[18] Spanwise turbulent contamination along the leading edges of swept wings was discovered first by Gray[19] in flight at the Royal Aircraft Establishment (RAE) in 1951 and was further studied by Gregory[20] at the National Physical Laboratory (NPL) on 60° swept wings. Installation of leading-edge fences stopped the spanwise turbulent contamination along the front attachment line of the wing; however, the fence boundary layer then had to be removed at the intersection of the fence and the wing nose.[21] This was accomplished by the gap visible in Fig. 7 which was vented to a low-pressure point on the wing. Also, appropriately sized chordwise suction slots in the stagnation region, as proposed in Ref. 21 and investigated by Pfenninger on a 45° swept wing in the wind tunnel, were used with success for removal of the boundary layer flowing

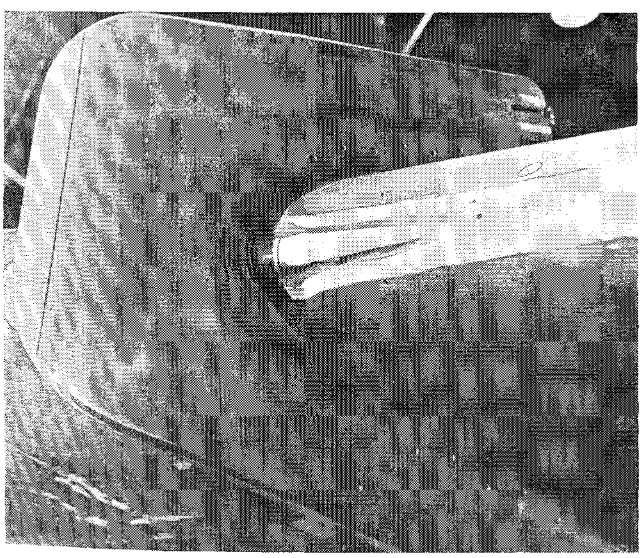

Fig. 7 Slotted fence.

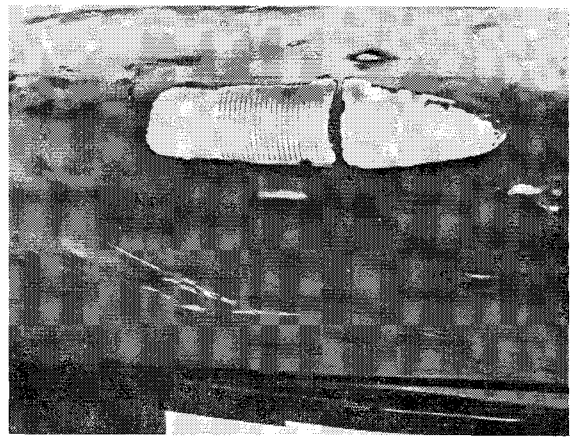

Fig. 8 Suction patch.

along the leading edge of the wing. This is shown in Fig. 8. But even with these devices, extensive laminar flow over the inboard part of the wing up to 30 × 10^6 transition length Reynolds number was obtained only during a rather limited number of test flights.

Comparison of all information from flight and wind-tunnel experiments indicated that the phenomenon was connected with the leading-edge radius or with the suction arrangement at the leading edge, or both. In Fig. 9, two streamlines are shown starting at the stagnation line of the 30° swept wing, one for a 4.8-in. leading-edge radius and the other for a 1.20-in. leading-edge radius representative of the 12% and 90% half-span of the X-21A wing. It can be seen that at the larger nose radius the streamline travels a rather long distance, in this case more than 15 in., before crossing the first suction slot, and that the amount of disturbed flow developed would possibly be more than could be removed in the spanwise slot. It is also seen that the divergence angle of the streamline from the stagnation line is rather small, smaller than the spreading angle of a disturbance in a laminar flow. This causes a disturbance to propagate itself along the stagnation line.

According to Pfenninger,[21] the boundary-layer Reynolds number at the attachment line, stagnation line, of a swept wing is a useful parameter in defining whether or not spanwise turbulent contamination may develop on a swept wing.

The boundary-layer profile at the attachment (stagnation) line of a yawed cylinder and a yawed ellipse was first computed by Sears[22] and Wild[23], respectively, in 1947. This attachment line boundary-layer profile is very similar to a Blasius profile. With a momentum thickness

$$\theta_{a.l.} = 0.407 \cdot \nu^{1/2} \cdot (dU_n/ds)^{-1/2}$$

and a momentum thickness Reynolds number

$$Re_\theta = U_\tau \cdot \theta/\nu = 0.407 \cdot U_\tau \cdot \nu^{-1/2}(dU_n/ds)^{-1/2}$$

ν is kinematic viscosity, U_τ is flight velocity component parallel to the leading edge, U_n is flight velocity component normal

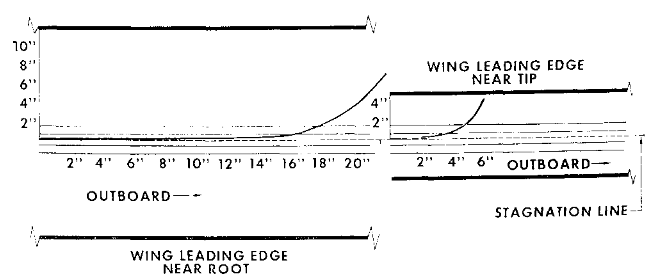

Fig. 9 Path of streamlines.

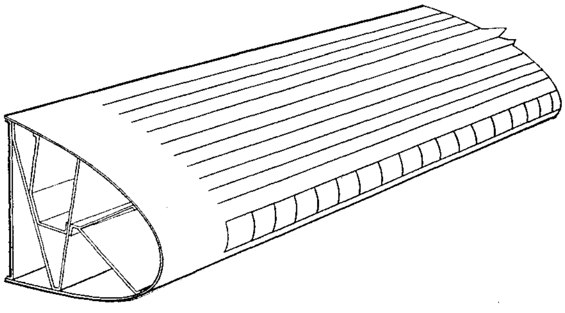

Fig. 10 Leading edge configuration.

to the leading edge, and s is running coordinate along the airfoil surface normal to the leading edge.

For an ellipse with the stagnation point at the apex of the major axis, this stagnation-point velocity gradient is

$$dU/ds = U\cdot(1/\rho)\ (t + 1)$$

where ρ is radius of curvature of the ellipse at the apex of the major axis, and t is thickness ratio of the ellipse (ratio of minor axis to major axis). Thus, it can be stated that for an ellipse and, therefore (also in the first approximation for airfoils which can conveniently be replaced in the front part by an ellipse) for a given sweep angle, flight velocity and kinematic viscosity, the momentum loss Reynolds number is proportional to the square root of the nose radius, as long as the "stagnation" point is in the immediate neighborhood of the minimum radius of curvature.

Simple reasoning shows that the front attachment line boundary-layer profile may not have sufficient stability if it exceeds certain Re_θ values. The stability of the laminar boundary layer at the front attachment line of swept wings has been discussed by Pfenninger in Ref. 24. To reduce the thickness of the front attachment line boundary layer, it is necessary to reduce the nose radius or to provide suction in the immediate neighborhood of the stagnation line. A solution, chordwise slots covering the area between the first upper and lower spanwise slots (Fig. 10), as proposed by Pfenninger,[24] was investigated in the wind tunnel by Pfenninger[25] and Bacon.[26]

Both means to reduce the thickness of the attachment line boundary layer, i.e., reduced leading-edge radius and chordwise slots, have been incorporated in a makeshift arrangement installed between the body and the pumping pods. The fences as shown in Fig. 7 were eliminated in connection with this modification.

Spanwise spread of turbulence originating from the body was successfully prevented at least at lower unit Reynolds numbers, say, 1.5×10^6/ft as occurring at cruise conditions. Laminar flow over the full chord or up to the aileron hinge has

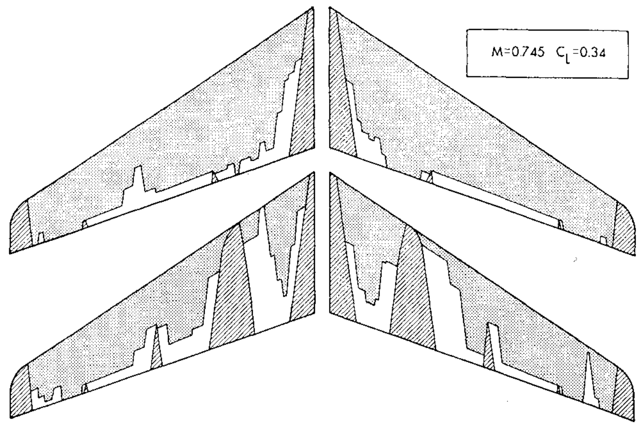

Fig. 12 Laminar area flight 120.

been achieved at cruise conditions with good repeatability over the entire upper surface of the wing, outboard of the first chordwise slotted leading edge station, which (because of structural restrictions of the mentioned makeshift device) is 48 in. outside of the body mold line or at 8½% half-span.

At higher unit Reynolds numbers as occurring in lower altitudes, desired for research and test purposes, the available suction quantities through the vertical slots of the makeshift arrangement were apparently not sufficient to eliminate all disturbances from the body. In consequence, both a small fence and a notch were tested in positions immediately adjacent to the fuselage (11 in. outboard of the mold line) to shield the wing-leading edge from the fuselage boundary layer. Approximately equally good results were obtained with both devices. Full-chord laminar flow has been obtained up to a maximum Reynolds number of 45.7×10^6.

The reduction of leading-edge radius should not be alarming to the aerodynamicist responsible for $C_{L\max}$ and stall characteristics of airplanes. In the modification of the X-21A wing, the leading-edge radii of the outer third of the wing (most responsible for pitchup and rolloff) will remain practically unchanged; only the very inner part of the wing will experience a significant reduction of the nose radius.

At this place, a word about the general aerodynamics of the X-21A wing is in order. The foremost task in developing the aerodynamic lines of the X-21A wing was, first, not to exceed a local Mach number of 1 normal to the isobars at cruise conditions ($M = 0.8$, $H = 40{,}000$ ft, and $C_L = 0.35$) and, second, to create straight isobars following element lines or suction slot lines. The intent of the requirement for straight isobars is to create as nearly as possible two-dimensional flow conditions and to create a high critical flight Mach number.

The well-known tendency of sweptback wings to "unload"

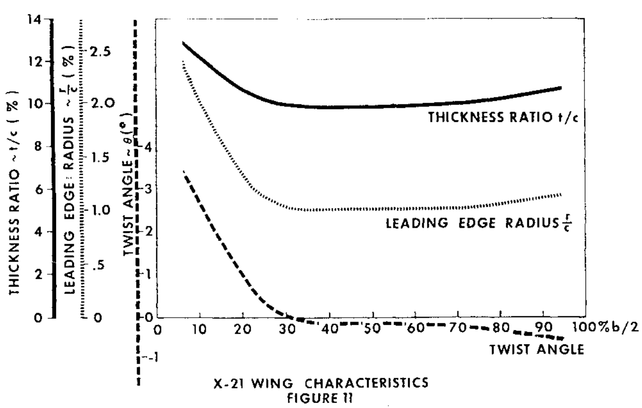

Fig. 11 X-21 wing characteristics.

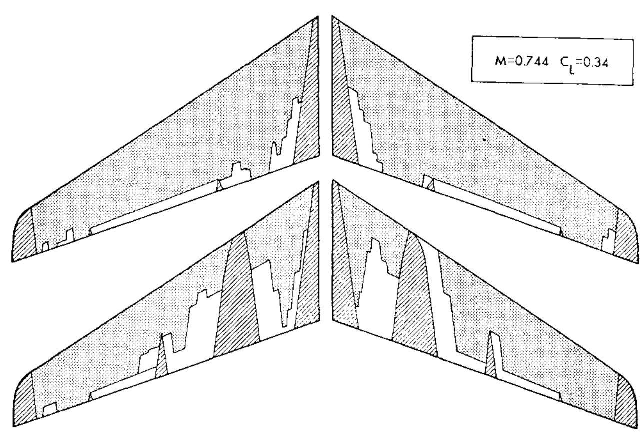

Fig. 13 Laminar area flight 121.

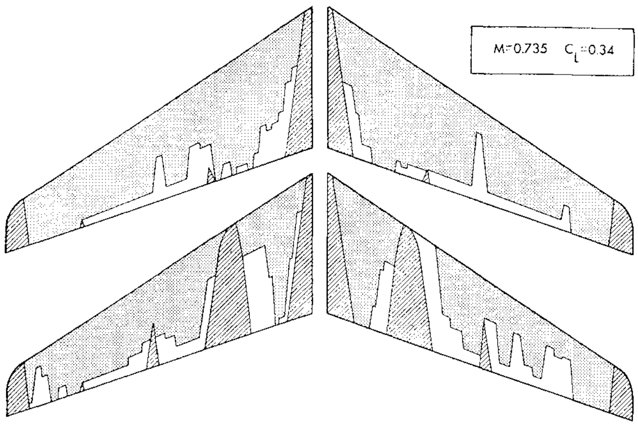

Fig. 14 Laminar area flight 122.

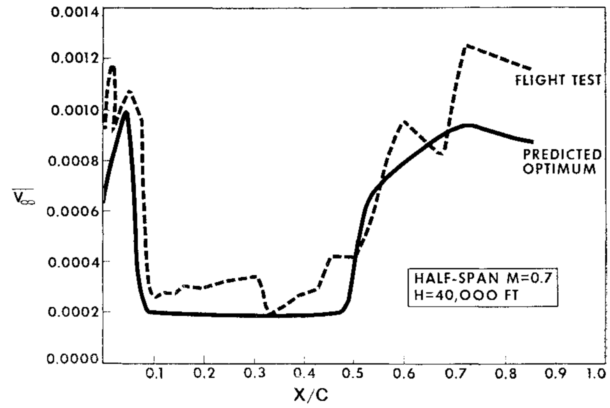

Fig. 15 Inflow velocity distribution at 28% half-span.

the forward part of the wing near the root while "loading" the nose outboard can be countered by twist.[27] Figure 11 shows how this technique is used on the X-21A. The use of twist alone would cause the surface pressure distributions to deviate from what was thought to be desirable in 1960 when the X-21A wing was being developed. Today, it is believed that a steeper pressure slope on the nose followed by a small suction peak is more desirable than the flat root-top pressure distributions originally desired. Consequently, a smaller nose radius is desirable from this standpoint as well as from the standpoint of reducing stagnation line boundary-layer thickness.

A word should be said about the "camel's hump" on the body in the wing area. In a low-wing airplane, the body acts as a wall and, with proper shaping of the wall, a nearly two-dimensional streamline could be created next to the body. For a high-wing airplane, the streamlines over the center of the wing would be straighter than those near the body unless the body were shaped in such a way as to create the desired curvature. Also, if the body intersected the rear part of the wing in the manner commonly used on high wing airplanes, a stagnation point on the body ahead of the trailing edge would occur. This would cause the isobars in the rear part to bend forward. The "camel's hump" of the X-21A airplane was designed to eliminate the condition that causes the isobars to bend.

Figures 12–14 show the laminarized areas during three successive flights at one flight condition $M = 0.75$ at 41,000 ft.

The suction distributions for the presented flight results are generally in good agreement with the predictions made on the ground of boundary-layer crossflow stability calculations and boundary-layer development calculations based on the work of Refs. 9 and 10. In areas where the pressure gradient was close to zero, i.e., where there was no appreciable boundary-layer crossflow and, consequently, the criteria of the crossflow stability limit did not apply, an arbitrary suction distribution was set, based on past experience with straight wings. The influence of this part on the total suction quantity is very minor as can be seen from Fig. 15. This shows a comparison of prediction and flight test results at 28% half-span for the flight condition $M = 0.7$, $H = 40{,}000$ ft length Reynolds number about 20×10^6. The suction distribution is shown in terms of v_{inflow} over v_{flight}, which is equivalent to the local C_Q.

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Some Recent Aerodynamic Advances in STOL Aircraft

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A brief review is presented of the results of some of the developments carried out recently by the de Havilland Aircraft of Canada Ltd., in two broad areas: 1) basic STOL performance, and 2) low-speed control and handling. In connection with basic STOL performance, the case for the classical STOL deflected slipstream configuration and some of its limitations are reviewed. Some evolutionary improvements to this basic design approach are outlined. In addition, the possibility of replacing the classical propeller installation with a true jet STOL aircraft is briefly discussed. Some improvements in longitudinal low-speed control are discussed. Other criteria are reviewed and some observed inherent limitations noted. Finally, the flight-test results obtained with an experimental aircraft incorporating a modified longitudinal control system are discussed.

Nomenclature

a_g = average ground deceleration, ft/sec^2
C_D = drag coefficient
C_J = jet thrust coefficient
C_L = lift coefficient
C_{L_A} = approach lift coefficient
C_T = thrust coefficient
C_X = horizontal force coefficient
C_{X_A} = approach horizontal force coefficient
C_μ = blowing coefficient
q = dynamic pressure, psf
M = mass flow, slugs/sec
S = wing area, ft^2
T_C = propeller thrust coefficient
T_P = propeller thrust, lb
T_J = jet augmenter thrust, lb
V_J = isentropic exit velocity, fps
V_V = vertical descent velocity, fps
α = angle of attack, deg
δ = aileron, flap elevator deflection, deg
φ = flight trajectory angle relative to horizon, deg

Subscripts

a = condition for attached flow
A = aileron
e = elevator
F = flap
0 = condition at freestream
disk = condition at propeller disk
ss = based on dynamic pressure in the slipstream

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1. Limitations of Conventional STOL Transport Aircraft

Existing propeller-powered STOL transport aircraft are characterized by a number of basic compromises that severely interact with the attainable performance levels. It will be of value to review briefly a few of the more important of these compromises.

Current STOL transport configurations normally combine a high level of static thrust with a somewhat oversized wing employing extensive high lift features. For the purposes of this paper and to establish a datum configuration against which certain aircraft improvements may be evaluated, those STOL configurations employing both a compromised (low) wing loading and thrust loading will be termed "conventional" STOL aircraft. Certain mandatory STOL characteristics (i.e., cargo density and deployment and an adequate rough field capability) significantly compromise the attainable