DESIGNING A HYBRID LAMINAR-FLOW CONTROL EXPERIMENT – THE CFD-EXPERIMENT CONNECTION

C.L. Streett
NASA Langley Research Center
Hampton, VA, USA

Abstract

The NASA/Boeing hybrid laminar flow control (HLFC) experiment, designed during 1993-1994 and conducted in the NASA LaRC 8-foot Transonic Pressure Tunnel in 1995, utilized computational fluid dynamics and numerical simulation of complex fluid mechanics to an unprecedented extent for the design of the test article and measurement equipment. CFD was used in: the design of the test wing, which was carried from definition of desired disturbance growth characteristics, through to the final airfoil shape that would produce those growth characteristics; the design of the suction-surface perforation pattern that produced enhanced crossflow-disturbance growth: and in the design of the hot-wire traverse system that produced minimal influence on measured disturbance growth. These and other aspects of the design of the test are discussed, after the historical and technical context of the experiment is described.

Background

Laminar Flow Control and HLFC

The application of laminar flow control (LFC), the delay of transition to turbulence, on subsonic transport aircraft for reduction of fuel burn has been the subject of numerous system and design studies. Some studies that include the effect of aircraft downsizing as a result of the reduction in fuel load required indicate a 20-30% drop in block fuel (BF) is possible, with an accompanying 10% reduction in take-off gross weight. Such numbers come from optimistic estimates of the amount of aircraft surface area that can be laminarized. However, system studies assuming that only the first 20% of the wing upper surface is laminarized show up to 5% reduction in BF.

As the subject of this paper is narrowly focused, no attempt at a comprehensive review of LFC work is attempted. The reader is directed to the excellent overview by Joslin,[1] and the annotated bibliography by Tuttle.[2]

Interest in laminar flow control has waxed and waned twice in the last 60 years. From about 1950 to 1965, numerous flight tests were conducted, with confusingly inconsistent (at that time) results. Researchers appear to have placed blame for these results on their inability to manufacture wing surfaces of consistent high quality, and interest was temporarily lost. Of course, we now know that while surface quality was indeed a factor, the state of knowledge at the time did not include the entire suite of disturbances that were potentially responsible for transition to turbulence in flight, particularly on swept wings. With the oil embargo of the early 1970's, fuel prices became so high (up to 30% of direct operating cost in 1980-81, as opposed to about 10% currently) that interest in the potential fuel savings of LFC was rekindled. The understanding of the physics of transition on swept wings was vastly expanded as well, and strategies for maintaining significant runs of laminar flow with minimal suction power required were developed. The most widely discussed concept is called hybrid laminar flow control (HLFC), wherein suction is applied only near the wing leading edge (ahead of the main spar), and is combined with the principles of natural laminar flow (NLF) to delay transition farther down the chord. This strategy results from considering the two main families of boundary-layer instability mechanisms that dominate transition on swept wings in flight. See c.f. ref. 3 for more details regarding boundary-layers disturbances.

The first family of these disturbances, known as "Tollmien-Schlichting" (TS) waves, are unsteady traveling ("convective") waves; in low-speed flat plate boundary layers, they are two-dimensional with their disturbance vorticity axis parallel to the surface and perpendicular to the oncoming flow. As the freestream speed goes up, this disturbance family covers a broad range of possible wave angles, and is only modestly sensitive to wing sweep. They also are most unstable (e.g., having high downstream growth) in adverse mean pressure gradient, and are stabilized by favorable pressure gradient. The nature of these disturbances has been studied since about 1930.

The second family of disturbances, known as crossflow (CF) disturbances, was first uncovered in the early 1950's by use of flow visualization in variable-sweep wind-tunnel transition experiments. These disturbances, which only appear in swept-wing boundary layers, have low frequencies and large wave angles (roughly parallel to the freestream), and are relatively insensitive to Mach number. Actually, it is the non-alignment of the boundary-layer edge velocity vector and the pressure-gradient (adverse or favorable) vector that produces the skewed BL profile in which the family of crossflow disturbances grow. The member of this family that is most relevant to transition on swept wings in the quiet flight environment in fact are stationary (frequency zero), and appear as nearly-streamwise vortices. This member of the crossflow-disturbance family is dominant in quiet environments due to issues of receptivity;[4] stationary CF disturbances have a strong initial-amplitude advantage over traveling CF in a quiet environment. This fact is pertinent to the experiment that is the main focus of this paper.

In HLFC, both suction and pressure-gradient tailoring are used to control both disturbance families on swept wings. In the region of rapid acceleration near the leading edge, CF disturbances grow strongly, and suction is used to control them. After this initial rise in velocity, the wing shape is tailored so that only a slightly favorable chordwise pressure gradient is maintained to 60-75% chord, whereupon a weak shock is designed to occur in transonic flow, and pressure recovery to the trailing edge commences. Of course, to reduce the chance of shock-induced separation and buffeting, it is necessary that the boundary layer be turbulent ahead of the shock. Suction is terminated near where the chordwise pressure gradient drops, since CF disturbance growth is greatly reduced. The small favorable pressure gradient is required, however, to reduce the growth of TS disturbances, but this gradient must be minimized to give the wing good off-design characteristics with a turbulent boundary layer.

The Need for Better HLFC Data

On the surface, it would appear that a wing pressure distribution (upper and lower surfaces) having a moderately rapid Cp rise, followed by a much more gradual rise to somewhere aft of mid-chord, is the ideal design pressure distribution for application of HLFC. However, in practice in transonic flight, such a pressure distribution is difficult to achieve and maintain; a leading-edge peak in Cp virtually always occurs due to the sensitivity of the flow there. This was proved in the Boeing 757 HLFC flight test experiment,[5] which was intended as a validation and calibration test of existing transition-prediction codes for HLFC design. In this test, a 757 transport aircraft was fitted with an upper-surface leading-edge glove covering a portion of the span of one wing. Suction flutes and ducts under a perforated titanium skin provided suction capability ahead of the main spar - see Fig. 1 (from Joslin[1]) - and it was found that the existing production aluminum skin over the wing box met quite rigid surface roughness and waviness criteria with only minor clean-up. Actual flight testing began in March 1990 and lasted until August 1991. In this test, it was found that a minor late-stage leading-edge contour modification resulted in a significant suction overshoot near the leading edge and a greatly reduced favorable pressure gradient over the wing box at design conditions. Despite this, it was found that the suction required to maintain laminar flow back to 65% chord or more was about one-third of that predicted during design. Indeed, post-test analysis using the measured pressure distribution as input to the existing transition prediction codes, showed significant scatter in flight-prediction correlation. The scatter amounted to an uncertainty in transition prediction from these tools of about ±0.07c at 95% confidence level.

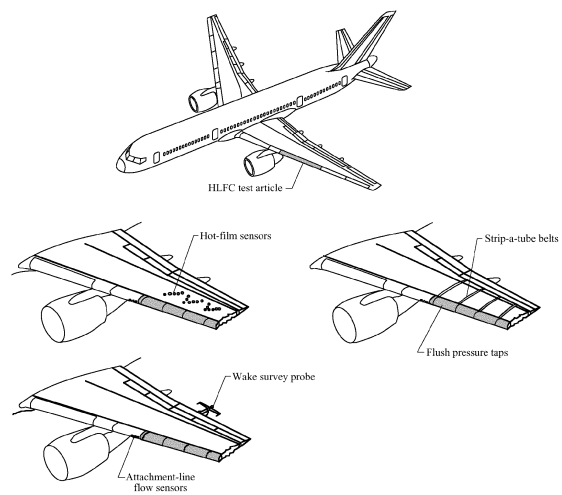

Figure 1. Boeing 757 flight test aircraft with HLFC test article and instrumentation (From Joslin[1])

During this time, Boeing was considering the application of HLFC in their design studies of large, long-range subsonic transports. Their finding was that while design predictions indicated that HLFC was an addition that was quite favorable in terms of direct operating cost (even considering the additional costs to maintain the suction surface and equipment), the uncertainty in the design predictions could reduce this advantage to negligible in the worst case. This is illustrated in Fig. 2; the notional graph plots the incremental cost advantage of adding a technology to a design against its increment in initial cost of a manufactured unit. If the change in initial cost is greater than the value of the incremental benefit, that is, a point which falls below the "break-even" line, the technology is a loser. If however the incremental value benefit is sufficiently above the break-even line, then the technology should be seriously considered for

application. Note that it is possible for the increment in initial cost to actually be **negative**; that is, the technology offers both initial and operating cost benefits – a clear winner. This is possible for application of HLFC, wherein the reduction in take-off fuel load results in a significantly reduced aircraft size over its all-turbulent counterpart for the same mission. However, design prediction uncertainty clouds this analysis, as indicated in Fig. 2, and can potentially reduce a clear economic advantage to a deficit if the design predictions are sufficiently wrong. In such a case of uncertainty, it is doubtful that an airframe manufacturer would risk employing such a technology.

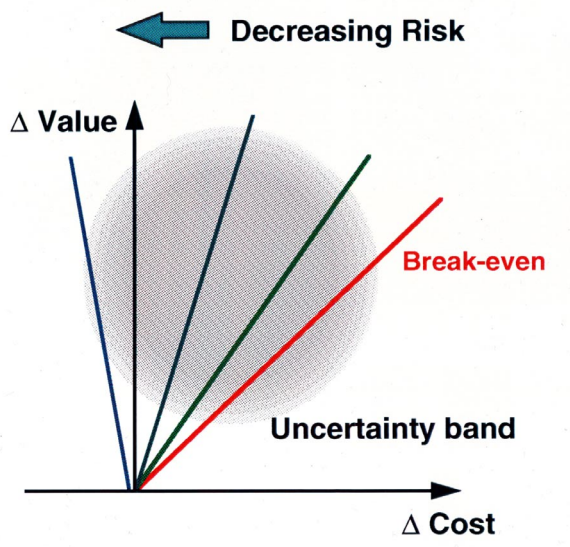

Figure 2. Notional plot of incremental benefit vs. incremental initial cost.

Additionally, it became clear during Boeing's design studies that the design of the titanium perforated suction surface was a strong economic driver in the application of HLFC. In the lack of detailed data and physical understanding, the design concept used for the 757 HFLC flight experiment was that the suction hole size needed to be as small as the laser-drilling process would allow, with the hole spacing correspondingly small to provide the required porosity. This was believed to be a conservative design, as the understanding at the time of the impact of receptivity on initiating stationary CF disturbances indicated that the hole size/spacing had to be an order of magnitude or more smaller than the disturbances that are near their onset-of-stability point at that suction location. This was thought to prevent individual suction holes from acting as direct receptivity sites, a phenomenon blamed for the adverse "oversuction" effects observed in earlier tests. But such a conservative design is costly, and there was strong impetus to know just how large the holes/spacings could be without this phenomenon occurring.

The NASA/Boeing HLFC Experiment

Experiment Design Factors

During the late 1980's to mid-1990's, the Transition Physics and Prediction group within the Theoretical Flow Physics and Experimental Flow Physics Branches of NASA Langley Research center was among the world centers of transition physics and prediction studies. Following their findings from the 757 HLFC flight experiment, and recognizing that the advanced transition-prediction tools being developed at NASA could potentially reduce the design uncertainty, Boeing contacted LaRC in March 1993, with the proposal that their complementary capabilities could be pooled to conduct an HLFC experiment. Such a shared experiment would be large in scale, and be designed provide detailed and accurate benchmark data for the calibration of prediction tools for swept-wing transition control by perforated-surface suction at high Reynolds numbers.

It was decided quite early in the design process of the experiment that it would be conducted at low, non-transonic speeds, taking advantage of the relative insensitivity of CF disturbances to Mach number and the relative quiet of low-speed wind tunnels versus transonic ones, and avoiding the choking problems encountered during the NASA transonic swept-wing LFC experiment conducted during most of the 1980's[6].

In addition, the experiment was to provide high-quality data for elucidation of transition physics in such conditions, in order to guide the development of next-generation transition prediction tools. In particular, the issue of suction-hole receptivity was to be studied in detail. This required that Reynolds number based on the hole size/spacing had to be representative of that seen in flight; at flight conditions of about 30kft. altitude and Mach number about 0.85, the unit Reynolds number is roughly 2.5 million per foot. Recall that the "conservative" panel design, which would form the baseline case for this test, was dictated by the smallest hole size that the best technology could produce in titanium sheet over a large enough area. This combination of factors lead to the requirement that the tunnel in which the test would be conducted would have to be capable of flight unit Reynolds numbers, a condition reached at about 2 atm. static pressure in a low-speed (Mach about 0.24) flow.

This combination of requirements – low turbulence and noise, size, and pressure capability – lead to the choice of the NASA Langley 8-foot Transonic Pressure Tunnel for this experiment. Pre-test risk-reductions experiments showed that this tunnel, operating in its low-speed mode, had freestream turbulence levels that rivaled the best wind tunnels in

the world, including NASA Langley's heralded Low-Turbulence Pressure Tunnel (LTPT). Such risk-reduction experiments and tests were a central aspect of the nearly two years of concurrent design and fabrication of this experiment. Virtually every system was tested in a realistic environment prior to wind-on, to address hardware design issues, ensure interoperability of systems, define limits of variability and uncertainty, and even to verify the model installation approach. The crucial need to quantitatively establish acceptance criteria for limits and uncertainty was met by extensive use of CFD in this experiment, as will be discussed later.

A "Designer Flowfield"

It was recognized early in the design of the NASA/Boeing HLFC experiment that in order to meet the goal of quantifying variability in transition-prediction tools, the experiment would have to be quite "pure"; that is, it would have to focus on a single particular physical mechanism of interest, while minimizing the competition from other transition mechanisms. Indeed, transition physics can be extraordinarily sensitive; it is characterized by multiple families of exponentially growing and decaying disturbance modes, often with large sensitivity to subtle details of the boundary-layer profile and surface condition. To achieve this desired "purity", it was decided that instead of the design process being one where an airfoil profile is chosen and its stability characteristics analyzed, the aero lines design would be conducted in reverse. That is, knowing what transition physics was to be highlighted, the designer is lead to a set of boundary-layer characteristics that produce this stability behavior. This step in the process was aided greatly by the fact that much of this characterization work had already been carried out in unpublished research of the physics of crossflow disturbances by the author. The assumption of an "infinite swept wing" was made, in which the three-dimensional Navier-Stokes equations are simplified by the assumptions that the geometry and flowfield are invariant in the direction of the wing surface generators, which are swept at a constant angle to the freestream direction. This results in a geometry that is characterized by a single airfoil profile taken in a cut normal to the swept leading edge, and a flowfield (both mean and disturbance) that has three components of velocity but are computed in just two dimensions. The disturbance field is further assumed to have a harmonic behavior in the spanwise direction, with a fixed spanwise wavelength. Using a numerical tool known as the harmonic linear Navier-Stokes method[7], Streett performed CF disturbance-growth simulations on a series of analytically defined normal-chordwise boundary layer edge velocity distributions, cataloguing boundary-layer and stability properties against the parametric definition of the mean flow. Some examples are shown in Figs. 3, 4, and 5, with a sweep angle of 35-deg chosen for all. The first model flowfield is designated "FL1" (Fig. 3); it is representative of some early concepts of the required pressure distribution for HLFC. It is characterized by a long run of favorable pressure gradient for strong TS disturbance stabilization. Shown in Fig. 1a are the chordwise-normal BL edge velocity near the leading edge, and the corresponding Hartree pressure gradient parameter β, plotted against the chord-length-normalized surface arclength. While the favorable wing-box pressure gradient is advantageous in terms of reduced TS growth, it results in continuous and strong CF growth over a wide range of spanwise wavelengths, as observed from Fig. 3c. The mechanism for this can be discerned from Fig. 3b, wherein is shown the evolution of BL crossflow velocity component; it is seen to continue at significant magnitude far down the airfoil. Strong leading-edge suction is required to prevent very early transition in such pressure distributions, but the boundary layer eventually transitions due to the continual CF growth.

Such pressure-distribution design concepts had been more or less rejected by the mid-1990's, since their off-design behavior was poor, and the run of laminar flow was limited at flight-scale Reynolds numbers due to the long continuous CF growth over the mid-chord region. Of more interest for application were pressure distributions with flatter "rooftop" regions aft of the leading edge. As noted above, a leading-edge overexpansion is usually unavoidable for such pressure distributions, so that feature needed to be considered as well. Such pressure distributions are represented in Fig. 4a, designated "FL5" and FL5c", the former having a leading edge suction peak, and the latter without. In the case of FL5c (without the peak), it is seen from Fig. 4b that the BL crossflow component decays in the chordwise direction, leading to the stabilization of short-wavelength CF disturbances aft of the initial leading-edge expansion, and greatly reducing the growth of longer wavelength disturbances (Fig. 4d).

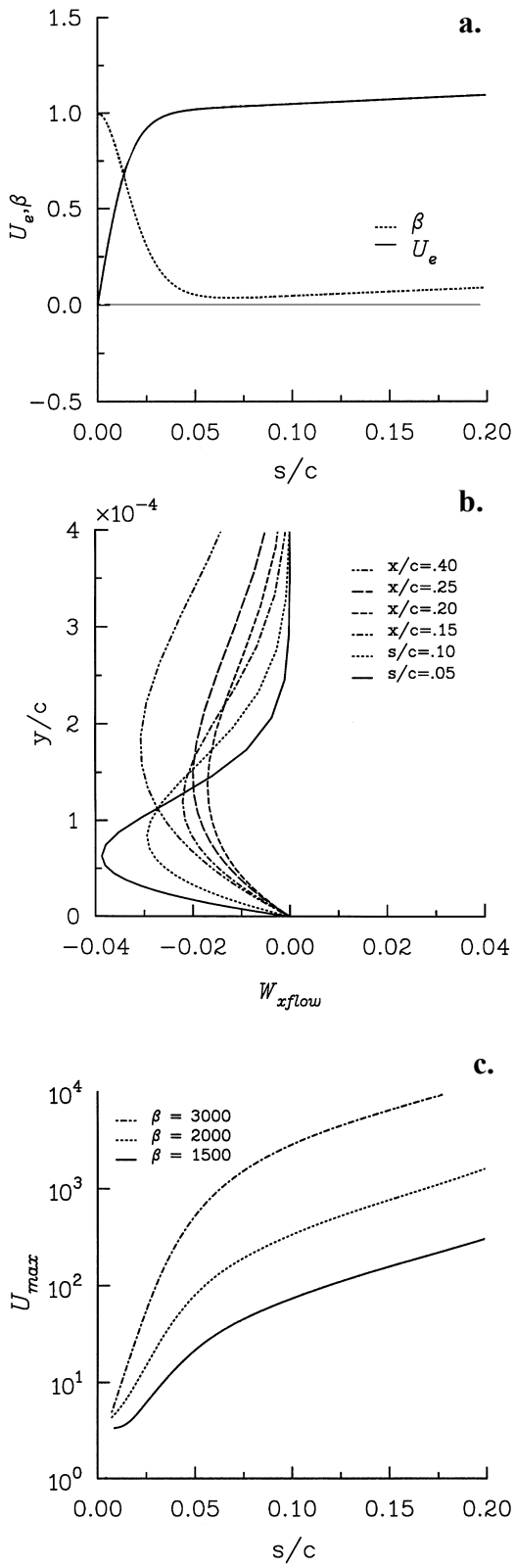

Figure 3. Characteristics of FL1 flowfield: a. edge velocity and Hartree parameter; b. boundary-layer crossflow velocity; c. amplitude of stationary CF disturbances with various spanwise wavelengths.

The addition of the leading-edge overexpansion peak (FL5) modifies the CF physics considerably. It can be seen in Fig. 4e that there is a short region of stabilization of moderate-to-long wavelength CF disturbances near the suction peak, but growth resumes strongly thereafter in the slight adverse pressure gradient region, continuing at growth rates far higher than those seen for FL5c (Fig. 4d). It is the physics of the resumption in growth of disturbances after the peak that was the primary issue behind the uncertainty in the application of transition prediction tools based on linear stability theory (LST). In the so-called "N-factor" method[8], local growth rates computed via LST are integrated under assumptions of the nature of disturbance growth to yield a relative disturbance amplitude. Values of this relative amplitude are correlated with experimental transition locations to yield a "critical" value, supposedly applicable to other similar flowfields. Unfortunately, the theory gives little guidance for the correct assumptions for such integration of growth rate. For instance, the ongoing argument at the time was whether of not the reversal in BL crossflow velocity component (see Fig. 4c), which resulted from the change in chordwise pressure gradient aft of the peak, should result in the amplitude integration starting afresh, as if a distinctly new CF disturbance began to grow aft of the peak. However, results from the HLNS method used here, which computes the disturbance growth directly rather than relying on integration of local growth rate, clearly suggests otherwise. This question, in the presence of suction and at flight Reynolds numbers, was one of the primary issues to be answered in the HLFC experiment.

Of course, such flat-rooftop pressure distributions result in the flow eventually transitioning due to TS disturbances. Since it was desired that the HLFC experiment be "pure" in its highlighting of CF disturbances, it was decided that the TS disturbances must be stabilized and that transition be due to CF alone. Beside the use of favorable pressure gradient, it is known that TS disturbances, even at low speed, are stabilized by wall cooling. However, it was found that such cooling would be excessive in the case of a pressure distribution like FL5, and not representative of that likely to be used in an HLFC application. It was decided therefore that a slightly favorable pressure gradient aft of the distinct suction peak would be needed. It was also found that the adverse pressure gradient region in FL5 was too unstable to TS disturbances to permit its use in the HLFC experiment, but that the near-reversal of the boundary-layer crossflow velocity component was sufficient to create the CF disturbance growth pattern that gave rise to the unresolved disturbance integration issue. The final leading-edge flowfield decided upon is shown in Fig. 5, and was designated "FL9". The near-reversal of the

crossflow velocity component is seen at s/c=0.20 and 0.25 in Fig. 5b. The CF amplitude evolution (Fig. 5c and 5d) shows the desired pattern of leading-edge growth, decay nearing the suction peak, resumption of growth in adverse pressure gradient region, a second stabilization at the end of the adverse pressure gradient, and resumption of CF growth in the aft wing box region.

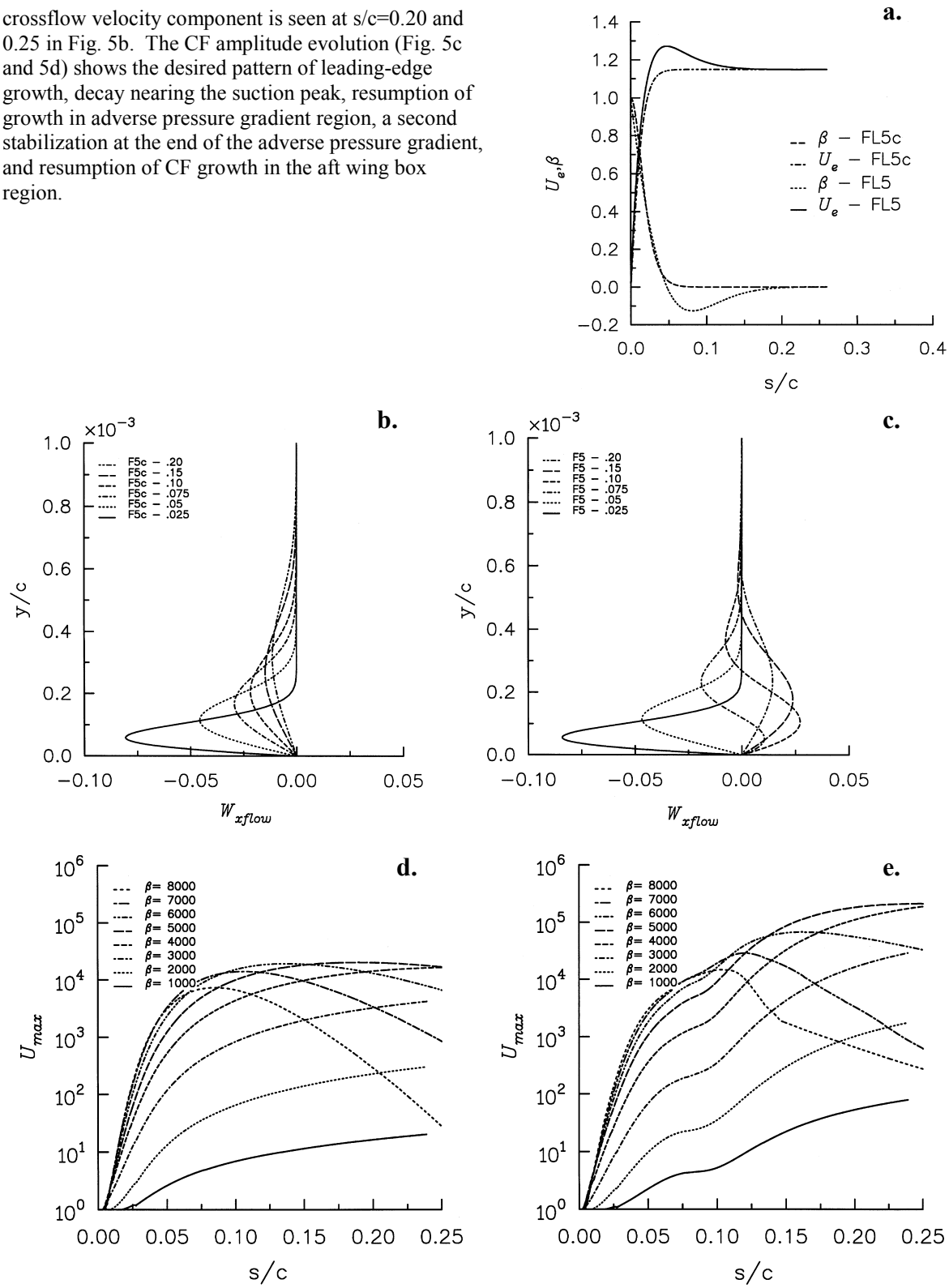

Figure 4. Characteristics of FL5c and FL5 flowfields: a. Edge velocity and Hartree parameter; b. boundary-layer crossflow velocity, FL5c; c. boundary-layer crossflow velocity, FL5; d. amplitude of stationary CF disturbances with various spanwise wavelengths, FL5c; e. amplitude of stationary CF disturbances with various spanwise wavelengths, FL5.

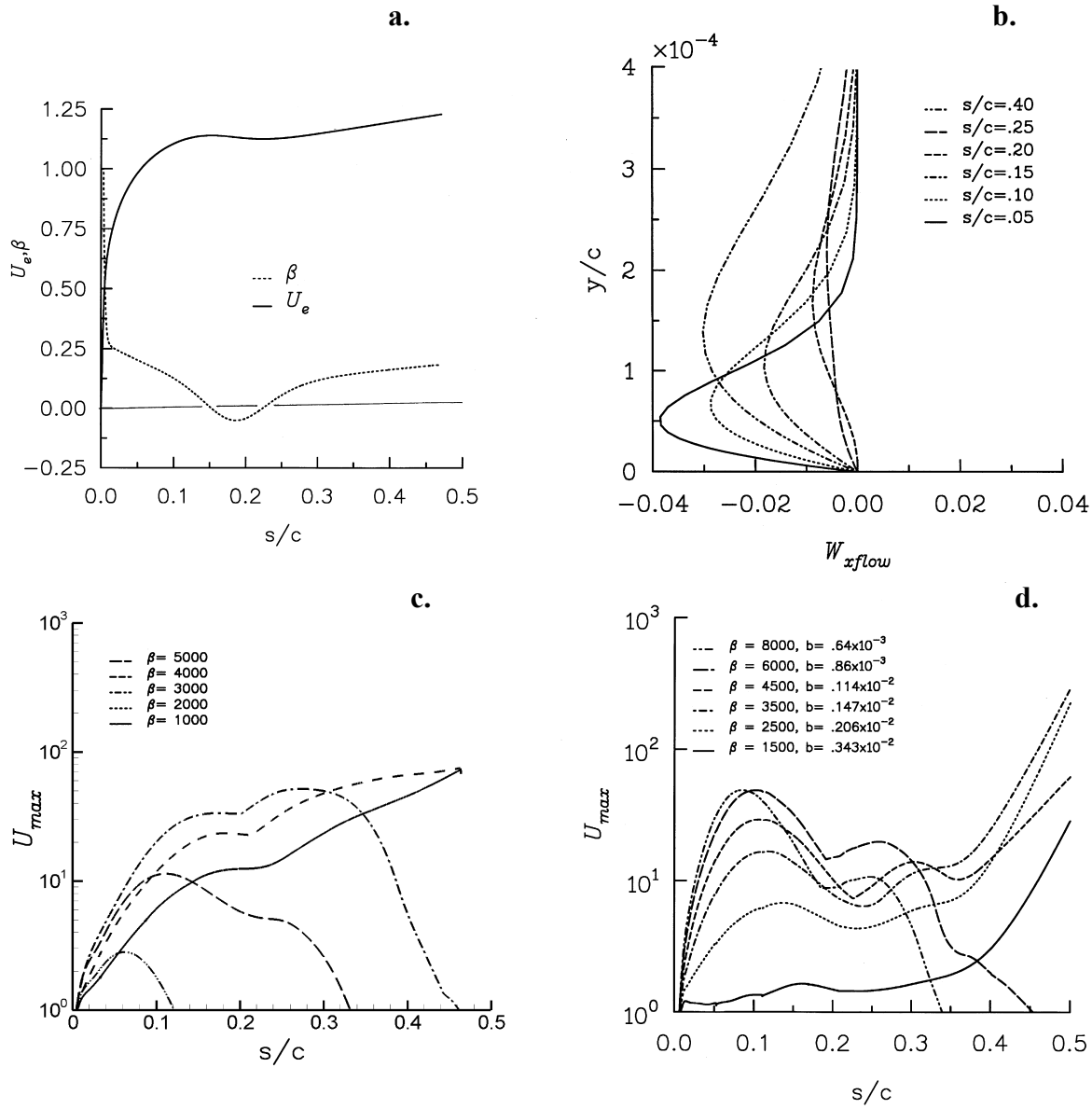

Figure 5. Characteristics of FL9 flowfield: a. edge velocity and Hartree parameter; b. boundary-layer crossflow velocity; c. amplitude of stationary CF disturbances with various spanwise wavelengths, no suction; d. amplitude of stationary CF disturbances with various spanwise wavelengths, with suction.

While detailed hot-wire measurements were to produce the majority of the data from this experiment, mean transition locations were measured to provide an initial correlation database for simple transition prediction tools like N-factor methods. These transition location measurements were made using an IR camera; the IR images show a difference in surface temperature at the transition location, due to the difference in skin friction between a laminar and turbulent boundary layer. In addition, stationary CF disturbances could frequently be seen in the images, and wedges of premature transition due to a stray bit of roughness near the leading edge could be quickly located and eliminated at the start of a test run. The transition location database provided a rigorous and sensitive discriminant for calibration of transition-prediction codes, due to the nature of the designed CF growth pattern. As a result of the growth-stabilization-growth disturbance evolution, the transition location was designed to first move forward slowly with increasing Reynolds number for a fixed suction distribution, then rapidly jump forward by a significant increment when it passed the region of stabilization, and finally resume its slow forward movement. This behavior was indeed observed to occur consistently, and provided a fallback position for success of the experiment if the design of the hot-wire traverse mechanism failed in some way. Examples of fore- and aft-transition IR images are shown in Fig. 6.

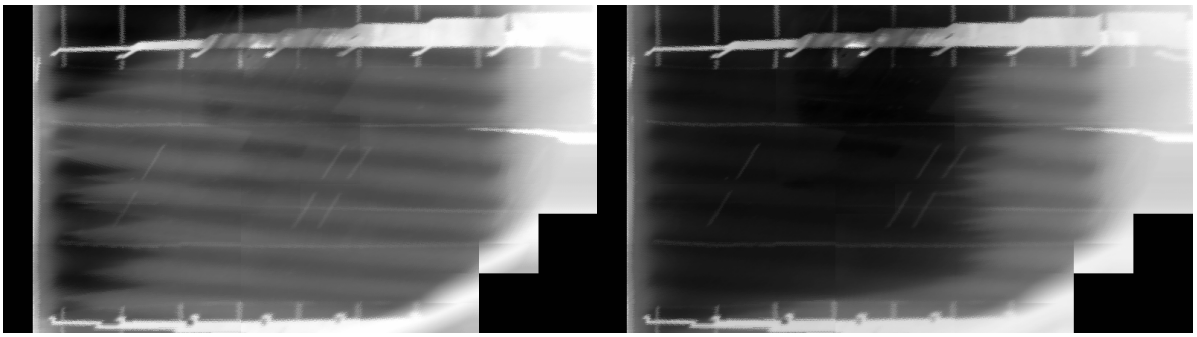

Figure 6. IR photographs showing transition front (unswept image geometry): a. forward transition location; b. aft transition location.

As a result of the massive number of stability computations made using a variety of methods in the design of the HLFC experiment, something of a discovery was made. As a result of sweeping a very broad disturbance parameter space of frequency, spanwise wavelength, and wave angle, a hitherto-unknown (to the researchers and advisors involved in the experiment) family of disturbances was uncovered. These disturbances are low frequency – much lower than the expected TS disturbances – of moderate wave angle – lower than CF but much higher than the expected TS family – and develop in a favorable pressure gradient flow. Computations indicated that these disturbances could potentially cause early transition (relative to the desired CF-dominated transition) for some conditions. The connection of this modal family to other disturbance modes was unclear and was not investigated fully, but computations further indicated that, like TS, they were strongly stabilized by surface cooling. It was found that 25° of wall cooling was adequate to stabilize these disturbances sufficiently so as to not compete with the desired CF disturbances, and a cooling system was implemented in the model design.

Note in Fig. 5 that the FL9 velocity distribution is defined to be considerably more "spread out" than that of FL5; the suction peak of FL9 occurs at s/c=0.250, as opposed to 0.05 for FL5. It was deemed desirable for the experiment that hot-wire measurements be made at or even ahead of the leading-edge suction peak. This led to a design with a relatively large leading-edge radius and gentle curvature change in that region. It was decided that the HLFC experiment was to elucidate the transition physics characteristics of the flow back to about 25-30% chord, including a suction panel covering the first 10% chord, of the wing of a subsonic transport. Thus for the HLFC experiment, in order to fit a flight-article size suction leading edge, to achieve boundary-layer Reynolds numbers up to those of 30% aircraft wing chord, and to have minimum airfoil length in the wind tunnel, it was necessary to design the test article to have most of one surface having the velocity distribution equivalent to the first 30% of the flight article. The remainder of the test airfoil would merely be support and fairing to give the desired velocity distribution over the surface of interest. Using a full-potential equation inverse airfoil design code, it was found that the equivalent of 30% of the flight article could cover about 65% of the test article, and still leave sufficient distance for pressure recovery to the trailing edge at a rate that would not separate a turbulent boundary layer. An additional constraint of zero net lift was added to the design process, which started with an NACA 0012 profile. The final airfoil profile and pressure distribution for the test article are shown in Fig. 7.

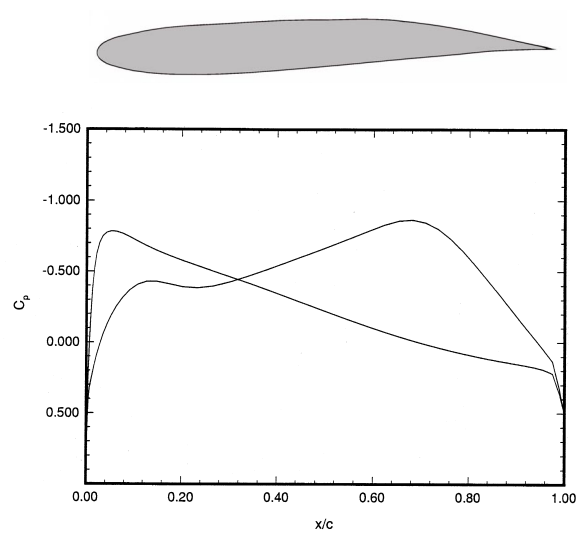

Figure 7. Final airfoil profile and complete pressure distribution.

Thus, the design process for the infinite-swept wing test article passed through the following stages:

1. Deciding upon the desired CF disturbance growth characteristics,
2. Defining the three-dimensional boundary-layer characteristics that would produce such a growth pattern,

3. Creating a leading-edge mean velocity distribution that would result in those boundary-layer characteristics,
4. Designing an airfoil profile that would create the desired velocity distribution over a large portion of the airfoil, and meet other criteria such as a minimum leading-edge radius and zero lift.

This is in essence the inverse of the path usually taken in designing a test airfoil for a transition study.

As indicated in the Background section, Boeing designers realized that the laser-drilled perforated panel characteristic could be a strong economic driver for application of HLFC technology to a new aircraft; this was one aspect to be investigated in this experiment. It was decided that leading edge units with four different suction-surface hole configurations would be constructed and tested. These were:

1. "Conservative" hole size and spacing based on the 757 flight test article; smallest hole size laser-drillable.
2. Maximum coupling to CF; large holes, and hole spacing/pattern angle set to match CF vortex evolution.
3. Relaxed criteria on hole size and spacing; lower manufacturing cost.
4. Solid surface.

The design of panel #2 was aided by extensive HLNS packet computations[9], which showed in detail the expected CF vortex pattern engendered by suction through a grid of circular suction holes. Iso-velocity surfaces for one example are shown in Fig. 8. Surface roughness measurements were taken post-test on panel #4, and highly detailed suction field measurements were made on the other panels, using a specially-designed 4-probe hot-wire traverse mechanism.

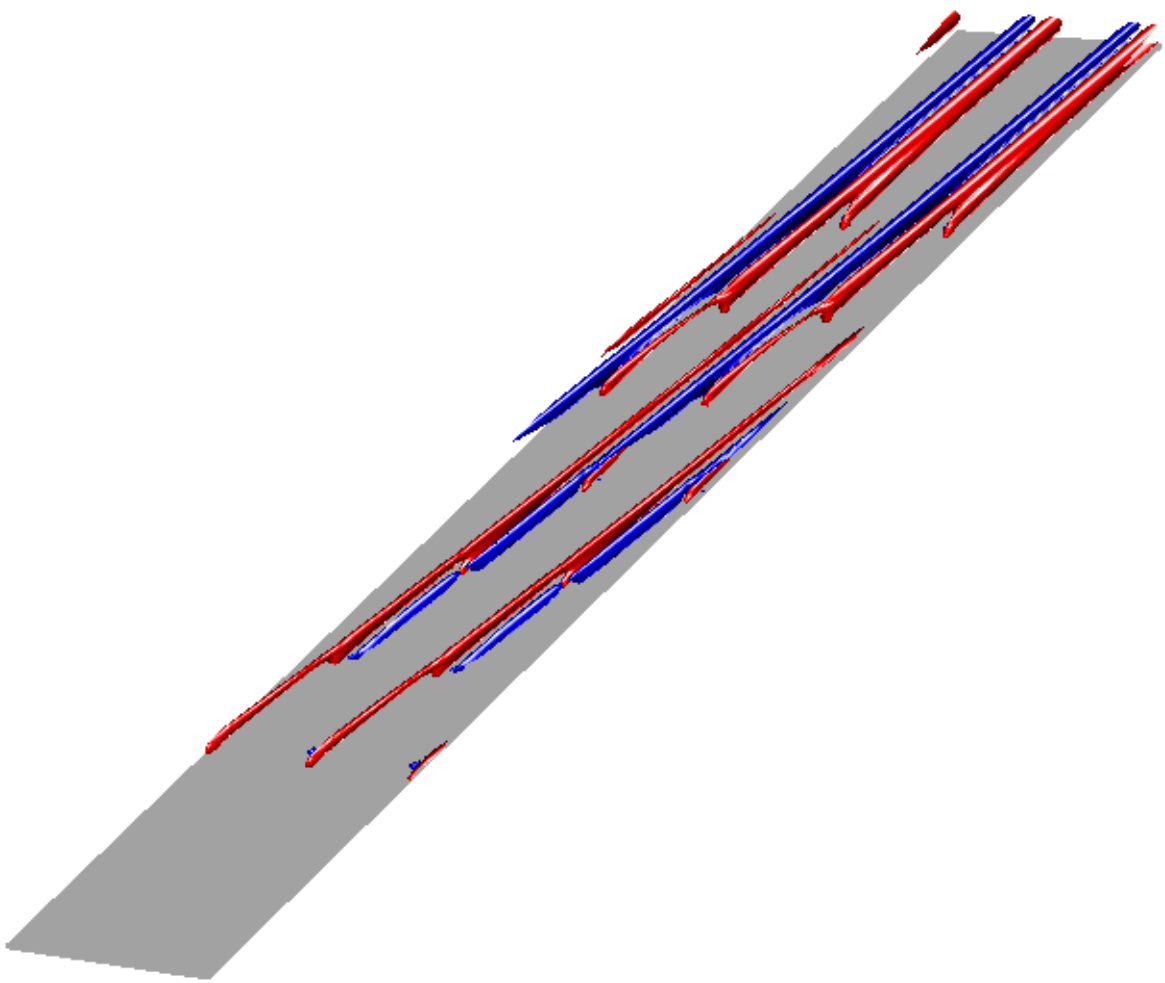

Figure 8. Iso-velocity surfaces of CF disturbances engendered by pattern of discrete suction holes.

There was one final aspect of the design of the aero lines for the test article, beyond the design of the airfoil geometry. In order to simulate an infinite swept-wing flow in a tunnel of finite spanwise extent, the practice is to use wall liners that are shaped as the streamlines would be in an ideal infinite-span flowfield[10]. It is surprising how severe the effect of finite span can be on a swept wing in a straight-walled wind tunnel, and how slowly the effect decays with increasing aspect ratio. Since it was necessary for the wing chord to be as large as possible for this test, the aspect ratio was limited, and a careful liner design was conducted, with multiple checks using different computational methods. Of course, allowance for the displacement effect of the boundary layer on the sidewall must be taken. These are relatively simple computations once the airfoil velocity distribution is defined. Since upper- and lower-surface streamlines at the edge of the boundary layer on a swept wing with lift will diverge as they pass from leading to trailing edge, the definition of zero net lift for the airfoil prevented a step from occurring in the liner at the trailing edge. Results of these liner design computations for the HLFC experiment are shown in Figs. 9, 10, and 11. Fig. 9 shows a comparison of the isobars on the model with straight tunnel walls and with the liner. Note the severe nonuniformity of the isobars in the straight-wall case. In Figs. 10 and 11 are shown drawings of the liner geometry, first as a streamwise cut through the model, then in an isometric view. A sectioned flap was provided on the model in case a small adjustment in its pressure distribution was required, but as can be seen in Fig. 12, no adjustment was required to achieve the target pressure distribution on the model. The measured wind-tunnel wall pressures also agreed with the design computation, as can be seen in Fig. 12. An additional use of the model-in-tunnel design computations was to locate the tunnel reference static pressure port; a region of relatively flat pressure distribution was located on the liner upstream of the model, and the predicted pressure there was referenced to the tunnel mean freestream velocity.

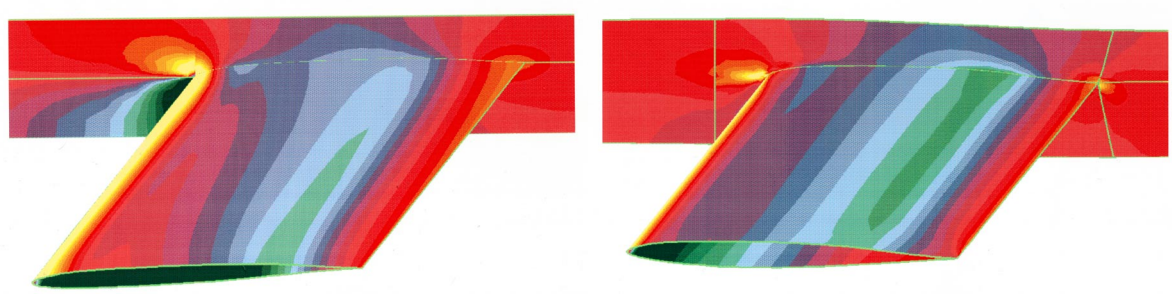

Figure 9. Comparison of swept wing isobars, straight tunnel walls vs. liner.

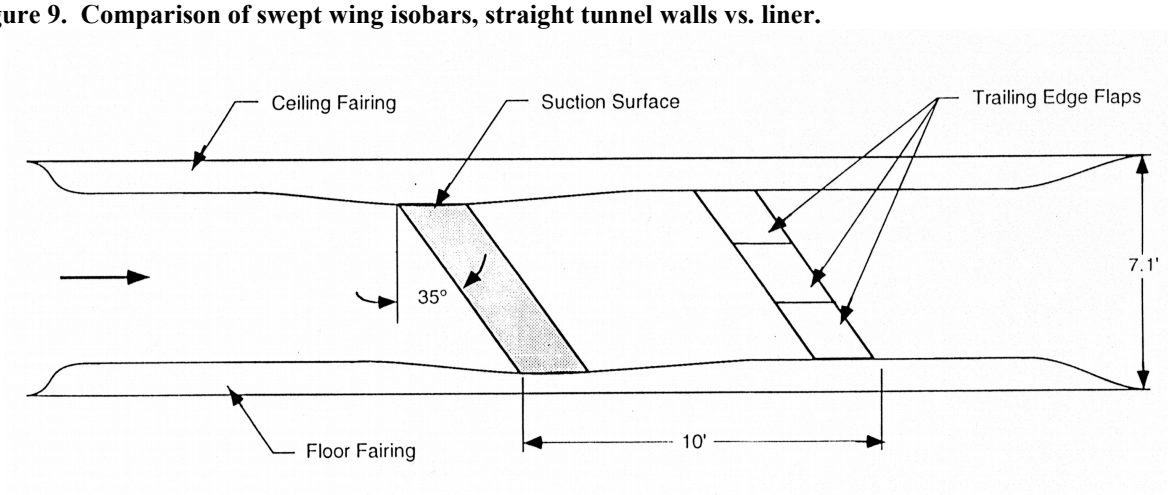

Figure 10. Streamwise cut through tunnel with liner in place.

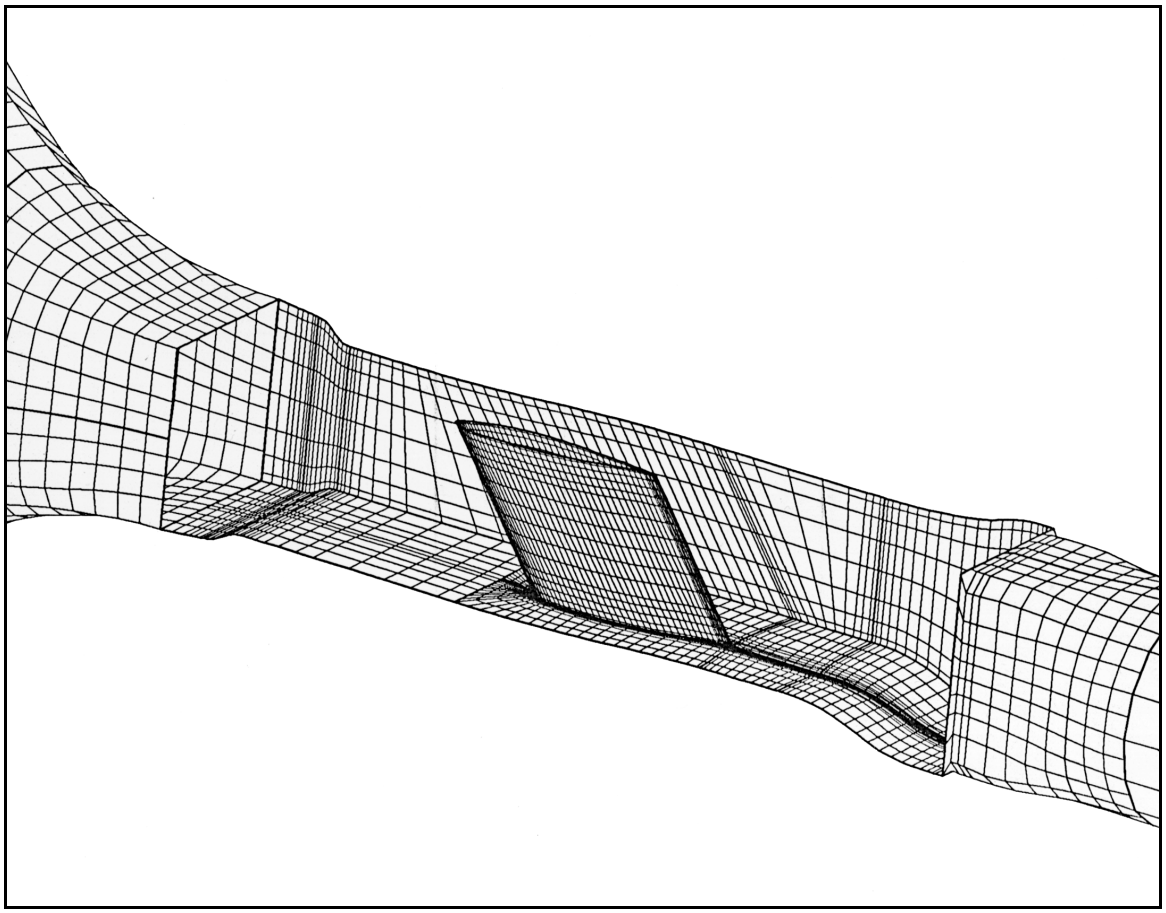

Figure 11. Three-dimensional view of wing in wind tunnel; ceiling and near wall removed.

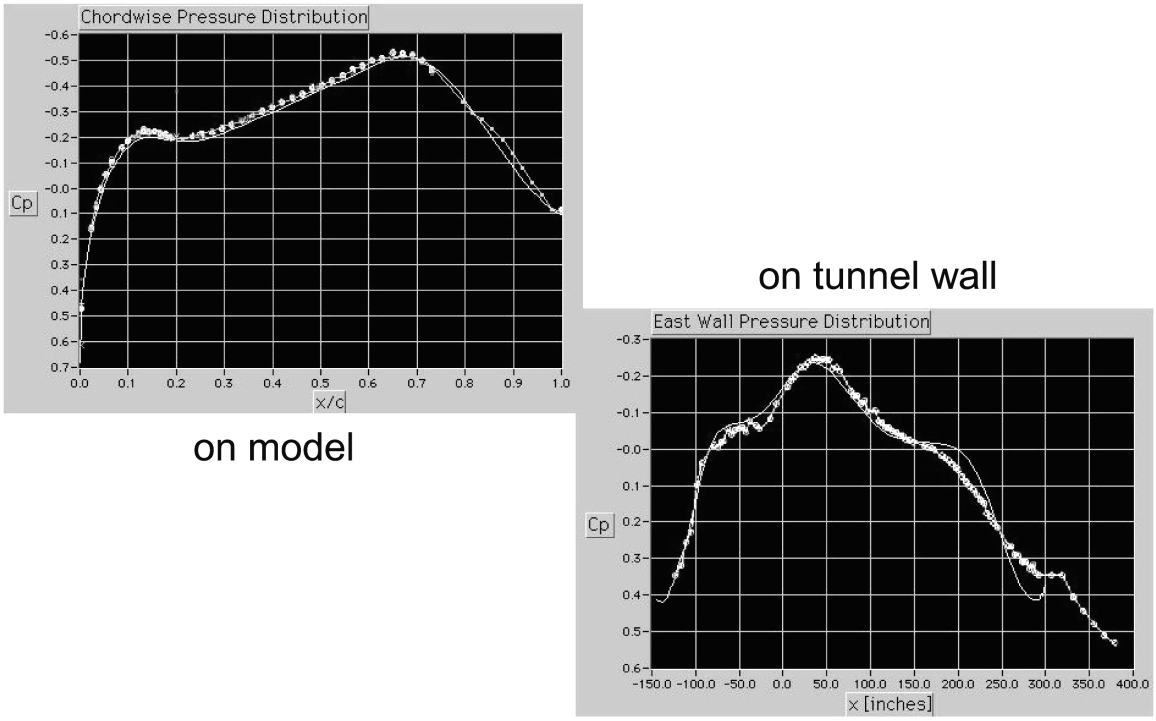

Figure 12. Comparison of predicted and measured surface pressures.

Hot-Wire Traverse Design

Another design challenge for which CFD was essential was the traverse mechanism for the hot-wire probe that was used to make detailed measurements of the HLFC wing's mean boundary layer and its disturbance field. The conflicting requirements of rigidity and minimization of aerodynamic interference required extensive team brainstorming, CFD, and risk-reduction testing to fulfill the design goals. The requirement that accurate measurements be made in a boundary layer as small as 0.02" (0.5 mm) thickness lead to a criterion that the probe head would have a positional uncertainty of 0.0005" (0.0125 mm). This uncertainty covered both vibration and probe-position measurement error. The former was measured by taking hundreds of samples at a fixed location within a laminar boundary-layer velocity gradient, and the latter by incorporating a sensitive laser positioning sensor in the traverse probe-head fairing. Of course, the flowfield around the probe-head fairing and the swept traverse strut itself (which provided 6" of spanwise traversing capability to the probe head) had to be fully attached and free from even the slightest unsteadiness. This was confirmed by extensive flow visualization and accelerometer risk-reduction testing prior to the actual HLFC experiment. Figs. 13and 14 are photographs of the traverse system in one such risk-reduction test, showing tail-on and plan views.

Of course, this rigidity and stability of the probe and support strut had to be embodied in a design that offered minimal influence on the CF disturbance growth. Again, maximum-deviation criteria were set for the impact of the probe/traverse on this growth, and extensive computations were made to ensure that these were satisfied. Given a rigidity-based size requirement for the traverse strut, steady CFD was used to compute and optimize the surface pressure footprint of the strut; it was found that a small negative net lift was advantageous. Then the computations were used to locate the distance upstream of the strut that was required to place the hot wire itself ahead of the upstream effect on CF growth from the local velocity distortion of the strut. Again, the HLNS simulation method was used extensively, as the effect of small-scale local mean velocity distortions violate the assumptions inherent in LST. The result of one pressure-footprint computation is shown in Fig. 15; the magnitude and shape of the predicted pressure footprint is confirmed by data from the HLFC test itself, as shown in Fig. 16. Finally, pre-test HLNS predictions of the influence of the strut pressure footprint on the growth of CF disturbances are shown in Fig. 17; in this figure, the hot wire is located $\Delta s/c=0.09$ ahead of the center of the pressure distortion – $s/c=0.30$ and 0.40 in this figure. As can be seen, the strut was expected to have minimal influence on disturbance amplitude at the hot wire.

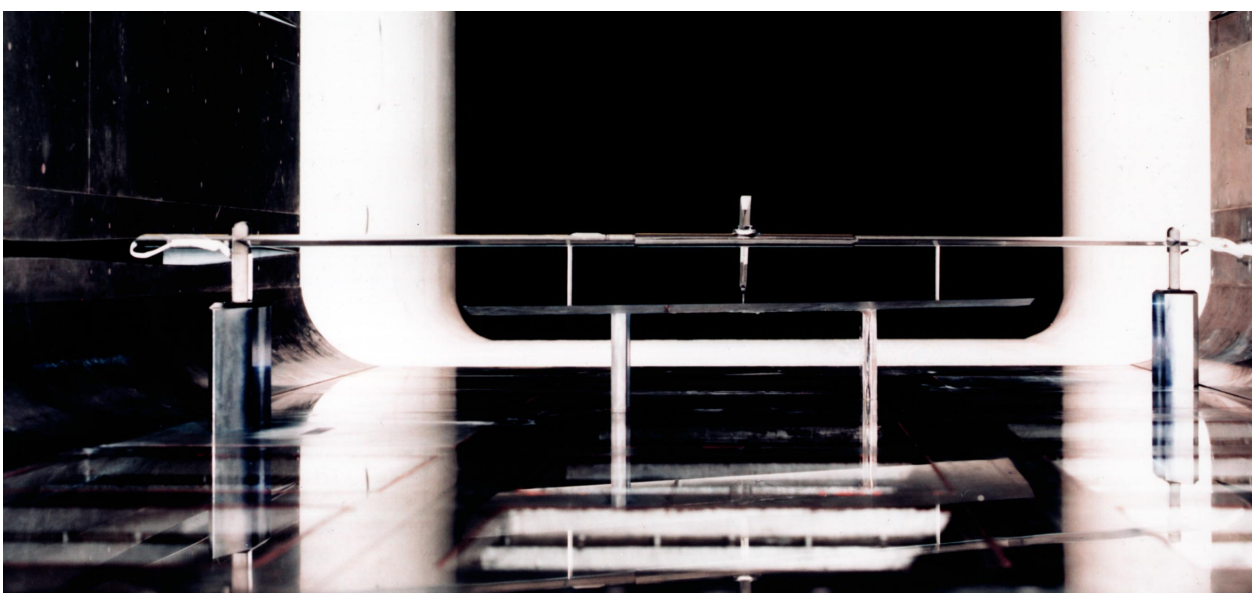

Figure 13. Tail-on view of swept traverse strut and hot-wire probe fairing, shown during risk-reduction test.

Figure 14. Plan view of swept traverse strut and hot-wire probe fairing, shown during risk-reduction test.

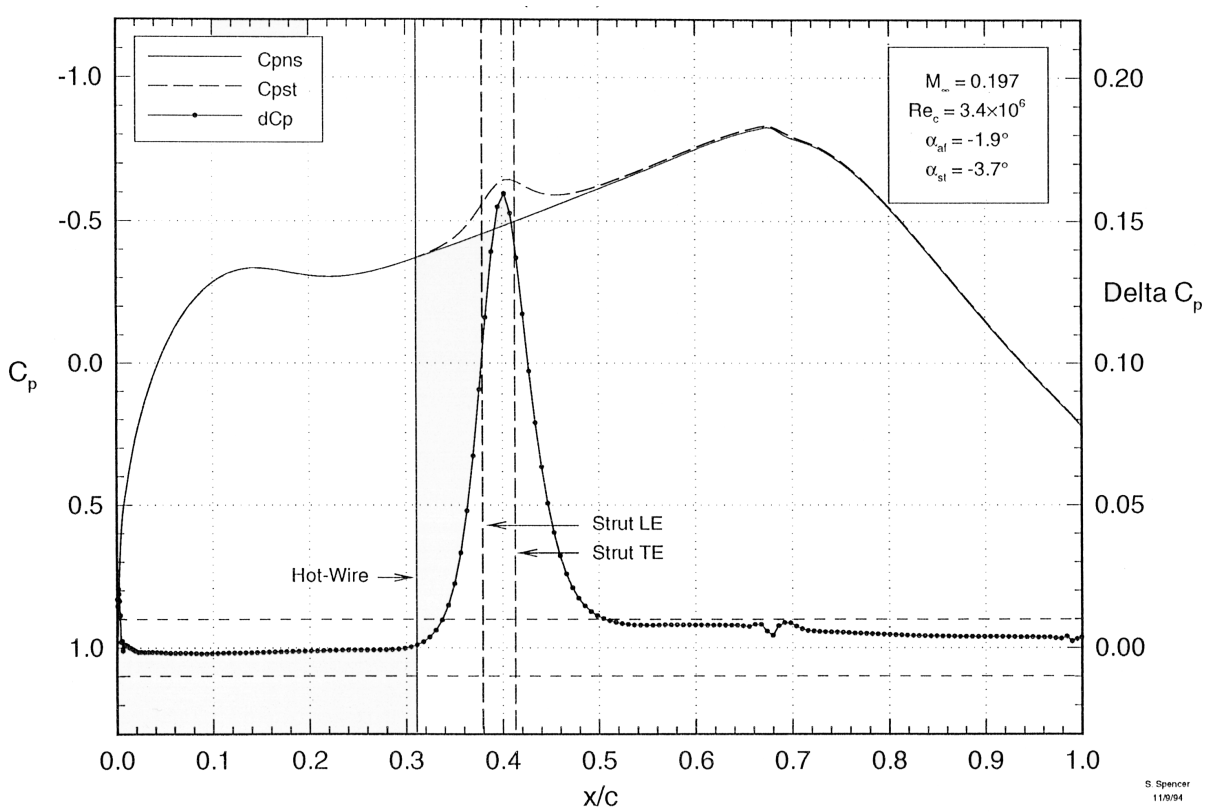

Figure 15. Influence of traverse strut on pressure distribution; computed.

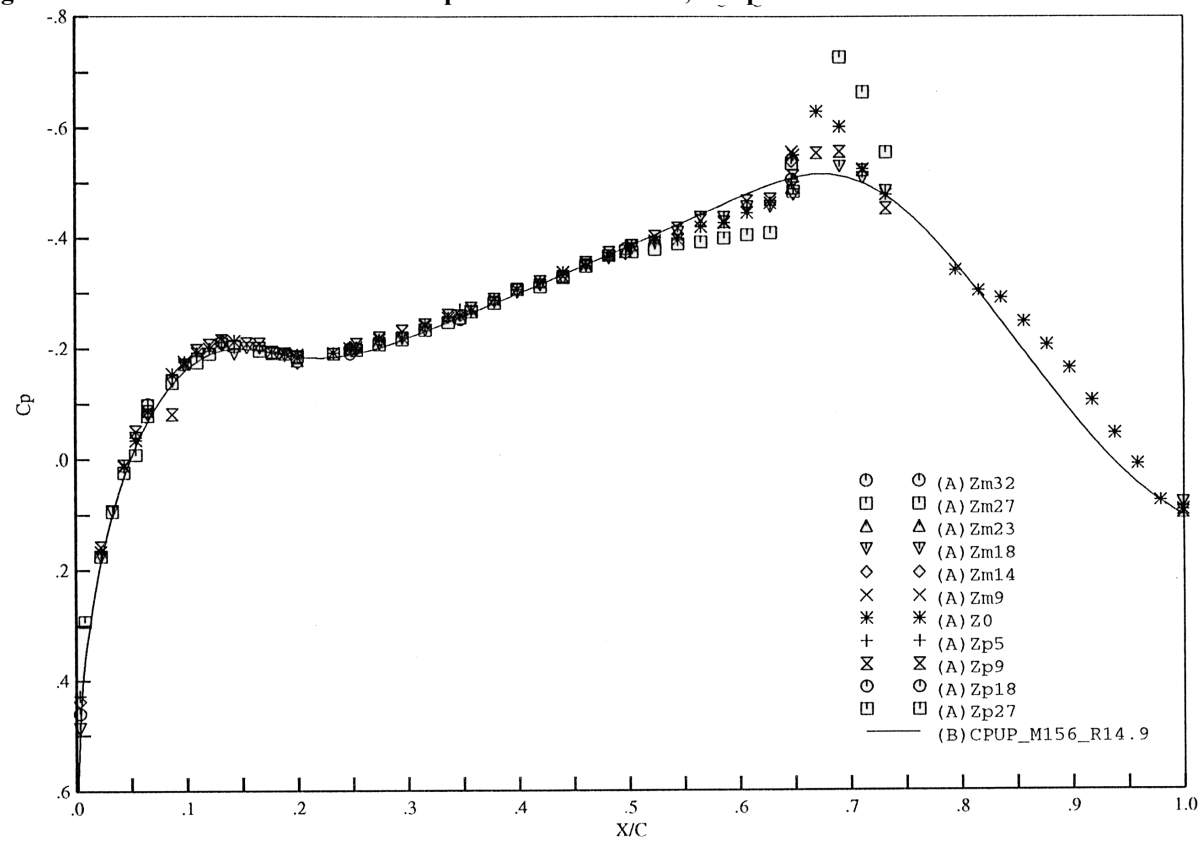

Figure 16. Influence of traverse strut on pressure distribution; measured.

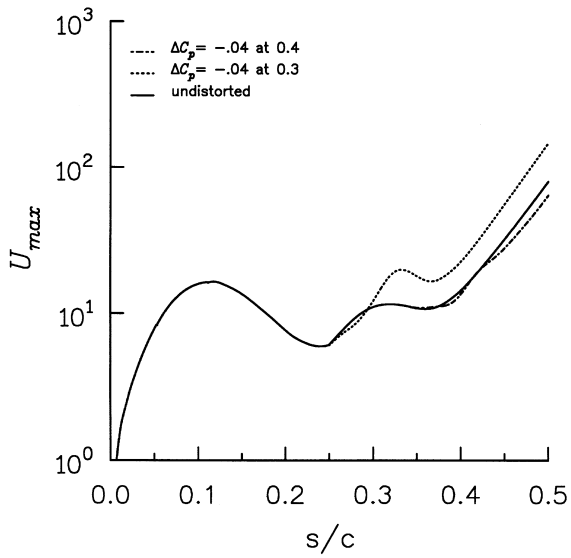

Figure 17. Predicted influence of distorted pressure distribution on CF disturbance growth.

Conclusions

The NASA/Boeing HLFC experiment, designed in 1993-1994 and conducted in the NASA LaRC 8-foot Transonic Pressure Tunnel in 1995, produced a one-of-a-kind dataset for CF-induced laminar-turbulent boundary-layer transition on a swept wing with various perforated suction panels at nearly flight Reynolds numbers. Efforts to mine the 6000+ hotwire profiles covering hundreds of flowfield and suction conditions have presently merely scratched the surface of the dataset. This database will provide detailed challenges to numerical simulations for many years to come, and will also provide extensive and accurate calibration information for present and future transition prediction methods for use in HLFC-aircraft design. The success of the experiment is in large part due to the extensive use of CFD in the following areas, some of which were not discussed here:

1. The design of the test wing, carried from definition of desired disturbance growth characteristics, through to the final airfoil shape that would produce those growth characteristics,
2. Identifying aspects and issues for which risk-reduction experimentation was required pre-test,
3. Design of the wind-tunnel liner surface to maintain uniformity to infinite-swept wing flow,
4. Design of the suction-surface perforation pattern that produced enhanced CF growth,
5. Design of the hot-wire traverse system that produced minimal influence on measured CF growth,
6. Provision of sample datasets to pre-test check out on-line hot-wire data reduction software,
7. Provision of a discriminant based on movement of the IR-measured transition location, which would provide a sensitive calibrator for transition-prediction tools based on linear methods like LST.

Acknowledgements

This experiment could not have been conducted without the dedication and attention to detail of the over 200 researchers, designers, machinists, and technicians who worked on this unique, once-in-a-lifetime effort. There are a few colleagues that stand out in the author's mind, and he deeply apologizes in advance for leaving out someone who rightfully belongs on the list due to the author's failure of memory of the events of some six or more years ago. On the NASA LaRC side, in no order: Jerry Kegelman, Rich Wlezien, Pete Jacobs, Cuyler Brooks, Mujeeb Malik, Meelan Choudhari, Chau-Lyan Chang, Shelly (Spencer) Sadowski, Chuck Harris, Ray Dagenhart, Latunia Pack, Mike Langford, Butch Watkins, Cathy Cronin, Kennie Jones, Brian Howerton, and especially the 8-foot TPT crew. For Boeing: Paul Johnson, Liam Ng, Jeff Crouch, Mark Anderson, Alex Krynytzky (buy a vowel, Alex!), Guenter Brune, Ron Radeztsky, Harry Topness, Frank Lyman, Frank Davenport, and all of the mechanical designers and machinists/artisans whom I never got to meet. Thanks also to Ajay Kumar for believing in us in the early days. This paper is dedicated in memoriam to Dr. Werner Pfenninger, who was a pillar in LFC research for decades.

References

1. Joslin, R.D.: "Overview of Laminar Flow Control," NASA TP-1998-208705, 1998.
2. Tuttle, M.H.: "Laminar Flow Control, 1976-1982: A Selected Annotated Bibliography," NASA TM-84496, 1982.
3. Reed, H.L., Saric, W.S., and Arnal, D.: "Linear Stability Theory Applied to Boundary Layers," in: Annual Review of Fluid Mechanics, vol. 28, Annual Reviews Inc., pub, 1996.
4. Choudhari, M.M., and Streett, C.L.: "Boundary-Layer Receptivity Phenomena in High-Speed and Three-Dimensional Boundary Layers," AIAA Paper 90-5258, 1990.
5. Maddalon, D.V.: "Boeing 757 Hybrid Laminar-Flow Control Flight Tests," Langley Aerospace Test Highlights, NASA TM-104090, p. 159, 1990.
6. Bobbitt, P.J., Ferris, J.C., Harvey, W.D., and Goradia, S.H.: "Hybrid Laminar Flow Control Experiment Conducted in NASA Langley 8-foot

Transonic Pressure Tunnel." NASA TP-3549, 1996.
7. Streett, C.L.: "Direct Harmonic Navier-Stokes Methods for Computation of Transition Physics," Invited paper, First AFOSR International Conference on Direct Numerical Simulation and Large-Eddy Simulation, Ruston, LA, August 4-8, 1997.
8. Malik, M.R.: "Stability Theory Applications to Laminar Flow Control," Research in Natural Laminar Flow and Laminar-Flow Control, J.N. Hefner and F.E. Sabo, compilers, NASA CP-2487, Part 1, pp. 219-244, 1987.
9. Streett, C. L.: Direct Harmonic Navier-Stokes Methods for Efficient Simulation of Weakly-Nonlinear Wave Packets. AIAA 98-0784, 1998.
10. Dagenhart, J.R.: "Amplified Crossflow Disturbances in the Laminar Boundary Layer on Swept Wings," NASA TP-1902, 1981.