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NUMERICAL AND EXPERIMENTAL STUDIES ON LAMINAR FLOW CONTROL

DANIEL ARNAL*
CERT ONERA, Aerothermodynamics Department, 2 Avenue E. Belin, 31055 Toulouse Cedex, France

SUMMARY

This paper gives an overview of numerical and experimental investigations performed in the framework of laminar flow control studies. After a description of the transition mechanisms that are likely to occur on swept wings, different techniques to delay the onset of laminar–turbulent transition are presented. Application of these techniques is illustrated by numerical results, wind tunnel experiments and free flight tests performed on Falcon 50 and Falcon 900 aircraft. Copyright © 1999 John Wiley & Sons, Ltd.

KEY WORDS: boundary layer; laminar–turbulent transition; drag reduction; flight tests

1. INTRODUCTION

The drag of transport aircraft originates from various sources, such as induced drag, skin friction drag, interference drag, wave drag or parasitic drag. Over the past few decades, particularly since the 1970s, civil aircraft manufacturers have made great efforts to reduce aircraft drag. The objective is to decrease the specific consumption because the potential reduction of over 10% would represent savings of several million dollars for the airlines [1]. Although the importance of the different drag sources vary according to the type of aircraft and the type of flight mission, skin friction drag usually plays a dominant role. It represents about 50% of the total drag for a commercial transport aircraft of the Airbus type.

There are essentially two methods that can be used to reduce skin friction drag (a complete account of this problem can be found in [2]). The first method consists of modifying the structure of the turbulent boundary layer in such a way that the skin friction coefficient at the wall is reduced. This topic will not be covered in this paper but see [3] for complete information. The principle of the second technique is to maintain laminar flow on the surface, i.e. to control the laminar–turbulent transition mechanisms. The potential benefits are important, because transition separates the laminar flow region with low drag from the turbulent region, where skin friction dramatically increases. Research programs on laminar flow control are now being financed in a number of countries in Europe and in the US. In France, the DGAC and the STPA have initiated programs of this kind jointly with industries (Aerospatiale and Dassault Aviation) and ONERA. Airbus Industrie also supports research with its partners and national research centers. The EC has launched a research program too, a part of which is devoted to drag reduction. These are just a few examples to illustrate the importance of the problem.

* Correspondence to: CERT ONERA, Aerothermodynamics Department, 2 Avenue E. Belin, 31055 Toulouse Cedex, France.

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This paper reviews the various means used today to maintain laminar flow on swept wings. The main transition mechanisms will be first described, together with the available prediction methods. Then, recent studies on laminar flow control will be described; they will be illustrated by experimental and numerical results. The last part of the paper is devoted to examples of flight results obtained on Falcon 50 and Falcon 900 aircraft.

2. TRANSITION MECHANISMS AND PREDICTION METHODS

Before going into laminarization techniques, it is useful to list the main mechanisms that can cause turbulence. There are many such mechanisms, and each would require an appropriate description, but three sources of turbulence are most commonly encountered on a swept wing: surface roughness, leading edge contamination, and 'natural' instabilities.

2.1. Surface roughness

Surface roughness is created either by the wing (rivets or joints between moving parts) or by accidental occurrences in flight (insect impacts or ice crystals). These all generate disturbances that can trigger premature transition. There is no general theory for determining precisely the maximum size of roughness elements that will not adversely affect laminar flow, but only a few empirical criteria developed specifically for each type of surface imperfection. One correlation that is widely used is that of Von Doenhoff and Braslow [4], which applies to isolated three-dimensional roughness elements like rivets or insect impacts. Of course, laminar flow is compatible only with perfectly smooth surfaces.

2.2. Leading edge contamination

This is a phenomenon that may appear when a swept wing is in contact with a solid wall, such as the fuselage on an aircraft, or the test section wall in a wind tunnel. Turbulence from the wall may propagate along the wing's leading edge and may in some cases render it completely turbulent.

Many experiments have shown that the presence or absence of this *leading edge contamination* depends on the value of a Reynolds number Re computed on the attachment line. If Re stays below 250, the turbulence from the wall will damp out. Above this number, the leading edge is contaminated [5–7]. In the simple case of a swept circular cylinder, Re is expressed by:

$$Re = \left(\frac{V_\infty R}{\nu}\right)^{1/2}\left(\frac{\sin\varphi\,\tan\varphi}{2}\right)^{1/2}, \tag{1}$$

in which V_∞, R, ν and φ designate respectively, the free-stream velocity, the radius of the cylinder, the kinematic viscosity and the sweep angle. This relation shows that the risks of contamination increase with the unit Reynolds number V_∞/ν, the radius R and the angle φ. Strictly speaking, relation (1) is valid for low speed flows only, but the trends are the same in transonic conditions.

2.3. 'Natural' instabilities

Even if the surface is perfectly smooth and even if leading edge contamination is avoided, the laminar boundary layer is endlessly subjected to excitations generated by the free-stream flow (noise, residual turbulence), or by the surface (vibrations, micron-sized roughness).

Boundary layer eigenmodes then appear and develop, amplifying downstream, which will cause transition. These eigenmodes are waves that are modeled by an expression of the following type in their initial phase of linear amplification:

$$r' = r(y)\exp(\sigma x + \tau z)\exp[i(\alpha x + \beta z - \omega t)]. \tag{2}$$

Here, r' designates a velocity, pressure, temperature or density fluctuation, x and z are two orthogonal directions on the wing surface, σ and τ are spatial amplification components, α and β are the wavenumber components, and ω a circular frequency.

With the mean flow field known, the range of unstable frequencies and their characteristics can be determined by linear instability theory [8]. The results show that the wavenumber vector of the most unstable waves in an accelerated flow (for instance, near the leading edge of a swept wing) is about normal to the external streamline: this is the so-called *cross-flow instability*. When the flow is decelerated (downstream of the point of minimum pressure), the wavenumber vector returns toward the free-stream flow direction, and one speaks of a *streamwise instability*. The transition finally occurs after a series of complex non-linear interactions among different unstable waves.

To predict the transition abscissa in practice, the method most often used is the e^N method based exclusively on linear theory. This consists in computing the total amplification of the instabilities along the body to be studied, and assuming that transition will occur for a predefined value N of this quantity. At least two problems arise in the case of swept wings.

The first is choosing a strategy for integrating local growth rates. With ω fixed, relation (2) shows that a wave is characterized by four parameters (σ, τ, α, β); but the solution to the stability equations provides only two conditions, so two other conditions have to be set more or less arbitrarily. One possibility is to assume that the amplification vector $\vec{A} = (\sigma, \tau)$ is collinear with the external streamline, which provides a relationship between σ and τ. One then scans β at a given station so that the amplification rate can be determined as a function of the propagation angle, which is defined as:

$$\psi = \tan^{-1}\left(\frac{\beta}{\alpha}\right). \tag{3}$$

Then, the direction ψ_M in which the amplification is maximum is sought, and it is this quantity that is integrated along x to get the total amplification. This is the principle of the 'envelope strategy'. Other strategies have been proposed (see overview in [9]), but there is no way of telling which is better.

The second problem in determining the transition location on a swept wing is that of the N factor value at the transition point. In two-dimensional flow, values of N of about 10 correlate rather well with flight test data. But the dispersion is much greater in three-dimensional flows. Of course, the result depends first of all on the integration strategy chosen for computing the N factor, but even with a fixed strategy, the tendencies are not yet completely clear and the dispersion remains. One explanation for this is that the streamwise and cross-flow disturbances are not generated by the same type of forced excitations. The streamwise eigenmodes are mainly excited by noise or residual turbulence, whereas the amplitude of the cross-flow eigenmodes depends on a large part on the surface polishing of the leading edge [10]. The dispersion of the N values on a swept wing also stems from the major role of the non-linear mechanisms, which seems to be much greater than in two-dimensional flow. So it is not surprising that a single parameter derived from a linear approach—the N factor—cannot correlate such a multitude of phenomena.

Despite its insufficiencies, the practical interest of the e^N method is still undeniable, if only for parametric analyses. More sophisticated approaches aimed at modeling the weakly non-linear interactions are currently being developed (parabolized stability equations, see [11]), but they do not yet seem to be ready for industrial purposes.

Any of the various sources of disturbance described in Section 2 can be influenced in order to delay the transition abscissa downstream. Assuming that surface polishing is compatible with laminar flow development, we therefore have to (i) avoid leading edge contamination and then (ii) limit the amplification of the 'natural' instabilities.

3. HOW TO AVOID LEADING EDGE CONTAMINATION?

As typical values of Re near the wing root range between 300–400 for small aircraft up to 800–1000 for large transport aircraft, it is necessary to develop specific tools to delay the onset of leading edge contamination. This is in fact the first problem to be solved for maintaining laminar flow on a swept wing: what good would it do to minimize the development of 'natural' disturbances if the wing is submerged in turbulent structures right from the leading edge?

The first idea is to lower the value of Re below the critical threshold of 250, at least in the wing root region. As relation (1) shows, this can be done by reducing locally the sweep angle φ or the leading edge radius R. Modifications like this are usually difficult to implement technically, which is why appropriate devices have been tested.

A successful device to prevent leading edge contamination is the Gaster bump [12]. It consists of a small fairing that is placed on the leading edge close to the wing root. It is shaped in such a way that the contaminated turbulent boundary layer is brought to rest at a stagnation point on the upstream side, whilst a 'clean' boundary layer is generated on the downstream side, see Figure 1. Tests performed in ONERA's wind tunnels (F2 at Le Fauga-Mauzac and T2 at CERT ONERA) made it possible to increase the onset of leading edge contamination up to $Re \simeq 350–400$. Numerical investigations related to these studies are described in [13]. Recent flight tests on the Falcon 50 and Falcon 900 business jets or future flight tests (A320 fin) have or will be using this device.

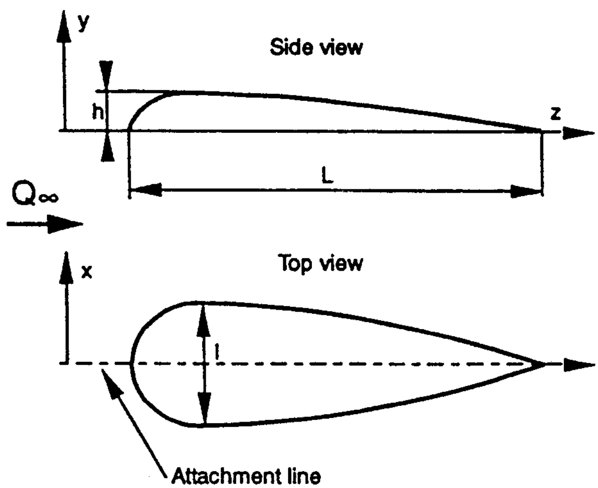

Figure 1. Sketch of a Gaster bump.

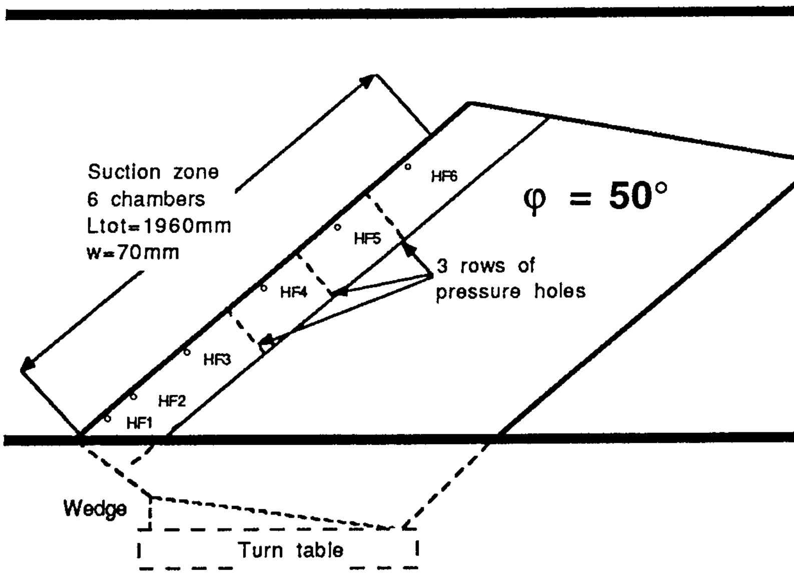

Figure 2. Experimental set-up without bump. HF, hot films.

Another solution is to relaminarize the turbulent boundary layer developing along the leading edge by applying suction along the attachment line. The efficiency of this process was first demonstrated by Spalart's direct numerical simulations [14]. These computations showed that contamination could be delayed up to $Re \simeq 350-400$ for $K = -1$. K is a dimensionless suction parameter:

$$K = \frac{V_w}{W_e} Re, \tag{4}$$

where V_w is the vertical mean velocity at the wall (it is negative for suction) and W_e is the free-stream velocity component parallel to the leading edge (for low speed flows $W_e = V_\infty \sin \varphi$). A first series of experiments carried out at CERT ONERA were performed on a small model [15]. With $K = -1.15$, contamination first appeared at $Re = 470$, but the small dimensions of the wind tunnel did not allow higher values of Re to be investigated. Therefore, ONERA decided to perform tests in the F2 wind tunnel at Le Fauga-Mauzac in order to study this phenomenon at large values of Re [16].

The chosen experimental support was a constant chord swept wing model generated from a symmetrical airfoil with a radius of 0.2 m near the leading edge. The phenomenon of leading edge contamination was studied at sweep angles of 40° and 50° by fixing the model to the wind tunnel wall.

The objective of the tests was to delay leading edge contamination either by the use of a Gaster bump or by applying suction along the leading edge or a combination of both. Figures 2 and 3 show the two leading edges that have been tested; the first one consists of six independent suction chambers fitted along the leading edge and the second one combines a Gaster bump with three leading edge suction chambers downstream of the bump. The chordwise width of the suction panel was about 70 mm, i.e. 35 mm on each side of the attachment line. The titanium perforated panel was laser drilled by AS&T company and the mean diameter of the holes was about 50 μm. The model instrumentation consisted of three

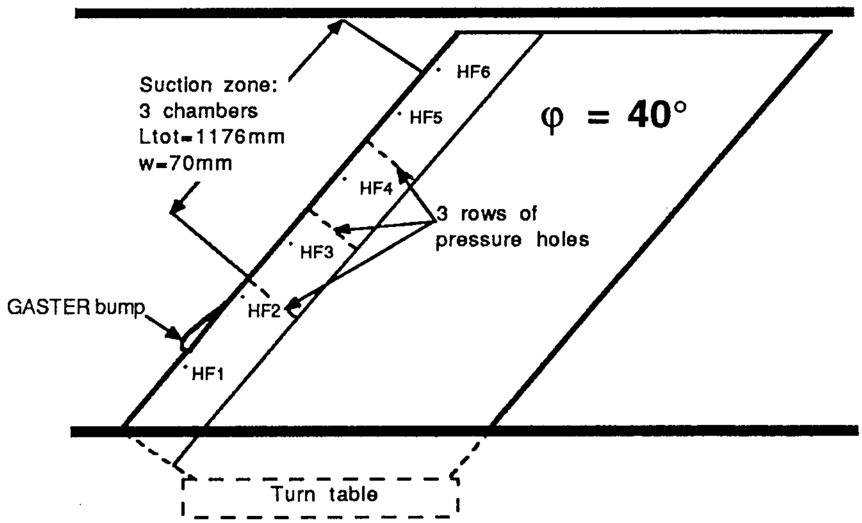

Figure 3. Experimental set-up with bump. HF, hot films.

rows of surface pressure taps aligned normal to the leading edge. Leading edge contamination was detected by flush-mounted surface hot films. The position of the hot films is shown in Figures 2 and 3.

Figure 4 shows the evolution of *Re* corresponding to the onset of leading edge contamination (first spots) as a function of the suction parameter *K*. The results obtained without Gaster bump at $\varphi = 50°$ are compared with the DNS results by Spalart [14] and with the experimental data currently available [15,17]. Without suction, leading edge contamination occurs for

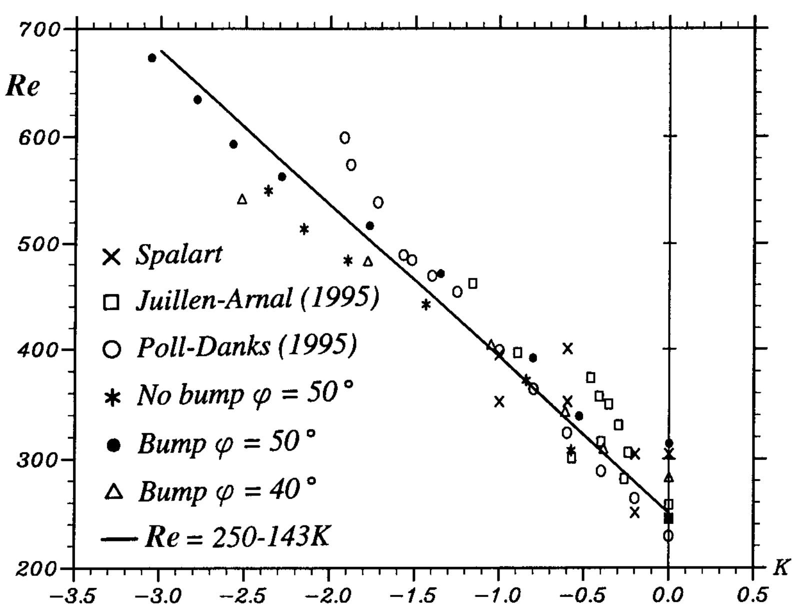

Figure 4. Leading edge contamination Reynolds numbers: summary of the results.

$Re \simeq 250$, as expected. Application of suction causes the onset of contamination to be delayed to $Re \simeq 550$ for the maximum suction rate attainable in the experiments.

For the leading edge fitted with a Gaster bump at $\varphi = 50°$, leading edge contamination in the absence of suction occurs at $Re \simeq 320$, a value that is lower than that obtained in other previous experiments. As soon as the flow over the bump is fully turbulent, the data with and without bump become close together (within the experimental uncertainty). The porosity of the porous leading edge fitted with the bump was larger than that of the leading edge without bump, so that the dimensionless suction parameter could be increased up to $K = -3.07$. This allowed the delaying of the onset of leading edge contamination up to $Re = 670$. A complete analysis of these results can be found in [16].

The conclusion of these studies is that with rather modest suction rates, boundary layers that are contaminated by turbulence at the wing root can be relaminarized and kept in the laminar state up to very large values of Re. This technique still remains to be validated under flight conditions.

In terms of Re, suction on the attachment line seems to be more powerful than the Gaster bump. Large insect debris on a bump will trigger transition irreversibly, whereas the same debris on a sucked leading edge will create a local turbulence wedge that the suction will take care of eliminating it. The suction system requires energy, though, and is not easy to implant on the attachment line (it is incompatible with the de-icing system, for example).

4. HOW CAN THE INSTABILITY AMPLIFICATION BE CONTROLLED?

Assuming that leading edge contamination is avoided, the boundary layer must now be influenced to delay the onset of transition. The techniques used in practical applications consist of modifying the shape of the mean velocity distribution so as to minimize the amplification of its eigenmodes.

4.1. Natural laminar flow

If one wants to obtain natural laminar flow, i.e. with no additional energy supply, one tries to optimize the outer flow velocity distribution. This is done by seeking a compromise between the positive and negative pressure gradients, which, as was said before, have contrary effects: roughly speaking, accelerated flows enhance cross-flow disturbances and damp streamwise disturbances, whilst the contrary is true in decelerated flows. The flight tests performed on the Falcon 50 fin (Section 5.1) have shown that this concept could yield good results. Nevertheless, it can be applied successfully only on wings of moderate sweep and/or small chord Reynolds numbers (sailplanes, small aircraft tail sections or rudders).

4.2. Laminar flow control

When natural laminar flow is too limited by the sweep or Reynolds number, one can turn to controlled laminar flow. The usual technique is to suck a small portion of the laminar boundary layer in order to increase its stability properties by changing the shape of the mean velocity profile. The effectiveness of suction has been known for a very long time, theoretically speaking, but its practical application has been slowed down by technological difficulties. One major problem is in manufacturing perforated walls with sufficiently small suction holes (a fraction of the boundary layer thickness) in order to keep from triggering transition by a 'roughness' effect. These problems have been solved today by the development of appropriate

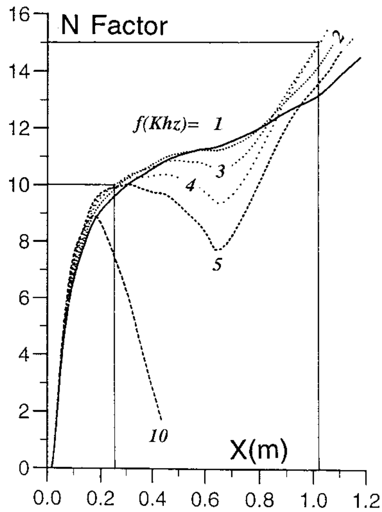

Figure 5. N factor on a swept wing with constant suction.

techniques, such as electronic or laser beam. However, it is not always easy to manufacture large suction panels with a uniform porosity. Pressure drop across the perforated wall also needs to be determined accurately in order to avoid local outflow in regions of strong chordwise pressure gradients.

The effects of the pressure gradients or suction on the transition location can be quantified by the linear instability theory and the e^N method. While these theoretical tools may not predict the transition abscissa with the desired accuracy, they are still of great value in optimizing the shape of the pressure distribution or defining the most efficient suction distribution. For instance, Figures 5 and 6 show the theoretical N factors computed in the case of the wing of a supersonic aircraft in cruise conditions (qualitatively similar results could be obtained at transonic speed). In Figure 5, a constant suction velocity $-V_w = 0.47$ m s^{-1} is applied from the attachment line to 20% chord. For $N = 10$, transition is located at $X = 0.25$ m. In Figure 6, the same total mass flow rate is used, but the suction velocity $-V_w$ is now linearly decreasing from 0.94 m s^{-1} at the attachment line to 0 at 20% chord. In this case, the transition location predicted with $N = 10$ is delayed up to $X = 1.1$ m. These numerical results demonstrate that a uniform suction distribution performs less well than intense suction near the attachment line, with decreasing suction velocity farther downstream. But it is thought that perforated walls can decrease the value of the transition N factor by creating a multitude of microdisturbances at the surface. This is a crucial problem that calls for detailed experimental studies.

5. FLIGHT TESTS ON FALCON 50 AND FALCON 900 AIRCRAFT

The final purpose of the theoretical and experimental research described above is to develop the methods for practical application of the laminar flow concept to civil aircraft. Flight tests are the final judge of the validity of these techniques. Over the past few years, several series of experiments were conducted on flight demonstrators, essentially in Europe and in the US. References [18,19] provide a review of these investigations. In this section, attention is focused on flight tests performed in France by Dassault Aviation (with state aid and in close co-operation with ONERA) using Falcon 50 and Falcon 900 demonstrators. A more complete description of these flight tests can be found in [20].

5.1. Falcon 50 experiments

A first phase (1985–1987) of the flight tests on a Falcon 50 aircraft was aimed at demonstrating the feasibility of the natural laminar flow (NLF) concept. For this purpose, a wing section was installed on the fin of the aircraft and tested in transonic conditions for two sweep angles ($\varphi = 25°$ and 35°). Due to the rather low values of the chord Reynolds number, streamwise disturbances dominated for $\varphi = 25°$, while transition was induced by cross-flow disturbances for $\varphi = 35°$. The measured transition locations agreed well with the theoretical predictions.

In addition to the need for a good prediction of the transition position, applying laminarity techniques requires a correct evaluation of the benefits that can be expected by reducing the skin friction drag and consequently, a good determination of the skin friction coefficient variation. So it is essential to prove that the available models are representative of these variations, not only in the laminar and in the turbulent regions, but also in the transition region where the skin friction varies greatly. The flight experiments validated the existing methods and especially the intermittency model developed at CERT ONERA, by comparison of the theoretical predictions with the experimental values measured by hot films glued on the wing section (Figure 7).

The second phase (1987–1990) of these flight tests was much more ambitious, since its objective was to use hybrid laminar flow control (HLFC), i.e. a combination of natural laminar flow and of laminar flow control by suction, on the inboard right wing of the same aircraft. To perform these tests, Dassault Aviation designed a new wing shape and developed a suction system, as well as a leading edge cleaning and anti-icing system. Transition was detected with 36 hot films flush-mounted on the wing up to 30% chord, downstream of the suction panel, and a Gaster bump was installed close to the wing–fuselage junction to prevent leading edge contamination. For sweep angles around 30° and weak suction rates, laminar

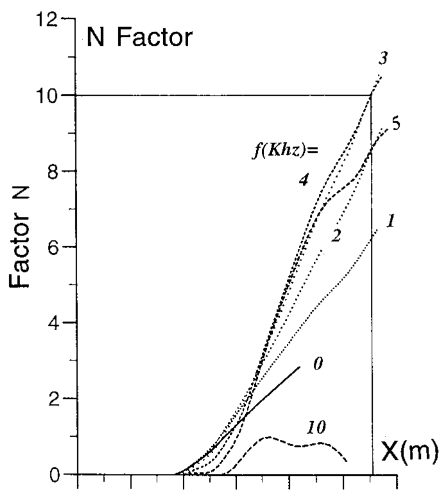

Figure 6. N factor on a swept wing with linearly decreasing suction.

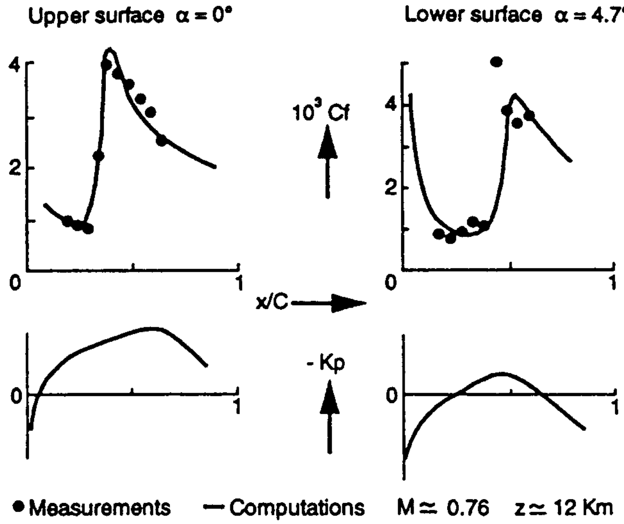

Figure 7. Comparison of skin friction coefficient

flow was maintained over nearly the whole test surface. Figure 8 gives typical examples of the results.

5.2. Falcon 900 experiments

While the previous experimental results did validate laminarity application methods in the specific framework of test aircraft, these technologies were not validated in an operational framework, considering especially the additional constraints related to business aircraft. This is why the Falcon laminar (FLAM) operation was launched. The purpose of this demonstration was to design, manufacture and certify an aircraft with hybrid laminar flow control using industrial methods, with consideration of weight and cost constraints, and then to put the aircraft into service to analyze the robustness of the laminar flow devices in operation.

For this purpose, the two inboard wings of the Falcon 900 were modified according to a principle similar to what was used in the experiments on the Falcon 50. After a new inboard wing was optimized in order to meet the laminarity objectives over the entire range of lift

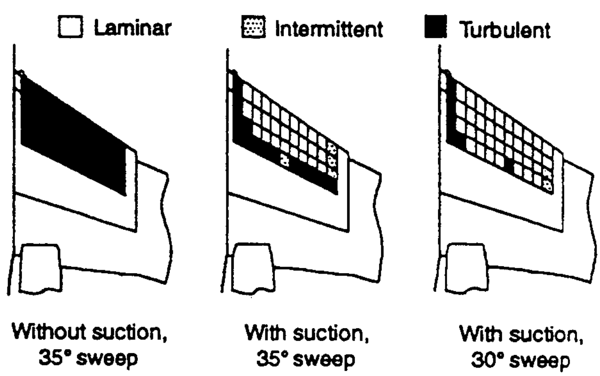

Figure 8. Laminar flow extent indicated by hot films

coefficients used by the Falcon 900 in cruise conditions, a leading edge and an upper surface panel were designed using production type solutions. Special attention was also paid to the design of the de-icing system on the leading edge, where a Gaster bump was used to delay leading edge contamination.

The flight tests were performed between April 1994 and February 1995, in two separate phases:

- the systems were first optimized and the results qualified in terms of laminar flow extent by using essentially hot film sensors;
- certification followed, with a demonstration of safety performance, flying qualities and especially certification flights under icing conditions, to fully validate the final design.

The results of the first phase demonstrated the efficiency of the suction and leading edge decontamination systems over a broad range of lift coefficients and Mach numbers. Since its certification in February 1995, the Falcon 900 'FLAM' has been in operation at Dassault Falcon Service.

6. CONCLUSION

Maintaining laminar flow on an aircraft wing first of all requires an understanding of the transition mechanisms and the development of prediction methods that are as reliable as possible. With the large number of theoretical, numerical and experimental studies conducted over these past years, our knowledge of the physical phenomena involved has increased considerably. Linear instability theory is still a very useful tool though, and the e^N method is widely used industrially, despite its shortcomings. Future improvements will relate to the development of more sophisticated methods, such as parabolized stability equations.

On the practical level, it has been proven that today's manufacturing techniques are compatible with the tolerances required for sustained laminarity. The suction systems have also shown their efficiency, both in laboratory and in flight conditions. The tests conducted on demonstrators show that delaying the transition is a realistic target today for reducing skin friction drag.

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