Passive control of transition in three-dimensional boundary layers, with emphasis on discrete roughness elements

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A brief review of laminar flow control techniques is given and a strategy for achieving laminarization for transonic transport aircraft is discussed. A review of some flight-test results on swept-wing transition is presented. It is also shown that polished leading edges can create large regions of laminar flow because the flight environment is relatively turbulence free and the surface finish reduces the initial amplitude of the stationary crossflow vortex.

Keywords: laminar flow control; three-dimensional boundary layers; transition; discrete roughness

1. Laminar flow control

The principal interest in laminar flow control (LFC) is the reduction in turbulent skin friction, which is 50 per cent of the drag budget on a modern transport aircraft such as an A320. The skin friction on the wings, fin and H-tail represent 40 per cent of that value. A number of different techniques have been used to delay transition through extended regions of laminar flow. Although some coverage is given to active control techniques, this paper concentrates on passive techniques that can sustain laminar flow.

Laminar–turbulent transition occurs as the result of the growth of unstable disturbances within the boundary layer. The principle behind LFC is to keep the growth of these disturbances within acceptable limits so that three-dimensional and nonlinear effects do not cause breakdown to turbulence. With this philosophy, it is possible to deal only with linear disturbances, the behaviour of which can be reliably calculated. Thus, the difficulties with transition prediction do not directly arise. The exception to this rule is crossflow, which will be discussed below. Bushnell & Tuttle [1], Bushnell [2] and Joslin [3] are good reviews of laminar flow control. A recent review by Arnal & Archambaud [4] brings the

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literature up to date and, in particular, covers current aircraft applications. There are four types of instabilities that can lead to transition in flight conditions, and these are discussed below.

(a) Streamwise instabilities: Tollmien–Schlichting (TS) waves

This instability occurs in two-dimensional flows and the mid-chord region of swept wings and is driven by viscous effects at the surface. The manner in which LFC works on TS waves can be described using the following example of Reshotko [5,6]. It is well known that the velocity-profile curvature term in the Orr–Sommerfeld equation (OSE), \( \frac{d^2 U}{dy^2} \), is an important driver of the stability behaviour. In fact, it is more important than immeasurable changes in the mean velocity itself. Solutions of the OSE demonstrate that the boundary-layer flow can be made more stable by making the curvature term more negative near the wall. This results in a fuller profile. The boundary-layer momentum equation, with \( \mu = \mu(T) \), can be evaluated near the wall, as shown in equation (1.1), and used to illustrate the stabilizing effects of different LFC techniques:

\[
(\rho V_0) \frac{\partial U}{\partial y} + \frac{\partial P}{\partial x} - \left( \frac{\partial \mu}{\partial T} \right) \left( \frac{\partial T}{\partial y} \right) \frac{\partial U}{\partial y} = \mu \frac{\partial^2 U}{\partial y^2} \quad \text{as} \ \ y \to 0. \quad (1.1)
\]

Here, \( V_0 \) is the wall-normal velocity evaluated at the wall (+ for blowing, − for suction), \( \rho \) is the density, \( U \) is the streamwise velocity component, \( P \) is the pressure field, \( \mu \) is the dynamic viscosity, \( T \) is the temperature, \( x \) is the streamwise coordinate and \( y \) is the wall-normal coordinate.

An evaluation of equation (1.1) shows that wall suction \( (V_0 < 0) \), an accelerating pressure gradient \( (\partial P/\partial x < 0) \), wall cooling in air \( (\partial T/\partial y > 0) \), wall heating in water \( (\partial T/\partial y < 0) \) all tend to stabilize the boundary layer by making the profile curvature term more negative near the wall.

(i) The case for pressure gradient

It should be pointed out that these are very sensitive mechanisms and that even weak suction or weak pressure gradients produce strong effects. For example, in a Falkner–Skan boundary layer, a Hartree parameter of \( \beta = +0.1 \) (which has only a 6.6 per cent change in the shape factor \( H = \delta^*/\theta \) from Blasius) increases the minimum critical \( x \) Reynolds number for stability by a factor of 9. Accelerating pressure gradients such as these are the basis for all natural laminar flow aerofoils that are subject to TS waves.

(ii) The case for suction

At the same time, average suction velocity ratios of \( V_0/U_\infty = 10^{-3} \) to \( 10^{-4} \), which are not unusual for LFC applications, can, for example, reduce relative amplitude growth from \( e^{26} \) to \( e^5 \) at \( F = 10 \times 10^{-6} \) for a flat-plate boundary layer. That the idea works is evidenced by the fact that the X-21 aircraft achieved laminar flow at transition Reynolds numbers of \( 27 \times 10^6 \) with a 20 per cent decrease in overall drag [7]. There is no dearth of data regarding the success of suction in the laboratory and in flight.

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(iii) The case for heating and cooling

Examples of the heating and cooling applications in LFC systems are reviewed by Reshotko [5,6,8,9]. Successful LFC by heating in water has been demonstrated [10–12], but the practicality of using the technique in other than particulate-free, fresh water is in question. Reed & Balakumar [13] calculate stability for cooling in air and show that the cooling stabilizes the usual supersonic TS mode but destabilizes the acoustic mode (second mode [14]). Reshotko [9] proposed a scheme for cooling in air, but transport LFC applications have not yet left the laboratory. A strong case cannot be made for heating or cooling LFC.

(iv) Hybrid laminar flow control

Present designs for energy-efficient aerofoils have LFC systems with a porous region near the leading edge. Generally, suction is applied near the leading edge of a swept wing in order to control leading-edge contamination and crossflow instabilities. Appropriate shaping of the pressure distribution stabilizes mid-chord instabilities. This arrangement is called a hybrid LFC system (HLFC) in that it combines steady-suction LFC with natural LFC (accelerating pressure gradient). A successful example of HLFC on a Boeing 757 was conducted by Boeing Commercial Airplane Group [15].

(v) Active control of transition

The idea of transition control through active feedback systems is an area that has received considerable attention [16–18]. The technique consists of first sensing the amplitude and phase of an unstable disturbance and then introducing an appropriate out-of-phase disturbance that cancels the original disturbance. In spite of some early success, this method is no panacea for the transition problem. Besides the technical problems of the implementation of such a system on an aircraft, the issue of three-dimensional wave cancellation must be addressed. Although Pupator & Saric [17] successfully cancelled broadband random two-dimensional waves with an active feedback system, Thomas [18] showed that, when the two-dimensional wave is cancelled, all of the features of the three-dimensional disturbances remain to cause transition at yet another location. Recently, Engert & Nitsche [19] showed a $10^{-1}$ reduction in two-dimensional TS wave amplitude (similar to the previous low-speed studies) that extended the laminar region by 10 per cent. The future of this work must address the three-dimensional cancellation as the $O(1)$ problem. Some clear advantage over passive systems has yet to be demonstrated for any active-feedback technique.

(b) Attachment-line contamination and stability

It is possible for the turbulence in the fuselage boundary layer to propagate from the wing-root junction along the attachment line and contaminate the boundary layer on a swept wing. Under these conditions, laminar flow cannot be maintained on the wing.

Pfenninger [7] developed a simple criterion based on an attachment-line Reynolds number, $\tilde{R}$, to avoid contamination. For a leading edge of a swept wing
with leading-edge-normal radius $r$, ellipse ratio $e$, sweep angle $A$, and free-stream speed $U_\infty$, the critical $\bar{R}$ is defined as

$$\bar{R} = \left( \frac{U_\infty r \sin A \tan A}{\nu (1 + e)} \right)^{1/2} < 250. \quad (1.2)$$

This seems to work under a remarkably broad range of flow conditions. If one cannot design a system with a subcritical $\bar{R}$, there are Gaster bumps, suction patches and other means described by Arnal et al. [20]. It should be mentioned that the linear stability limit of the attachment-line boundary layer is in the range $\bar{R} = 550 - 600$. The attachment-line problem is a manageable LFC issue.

(c) Curvature-induced instabilities: Görtler vortices

A shear layer over a concave surface exhibits a centrifugal instability of the Rayleigh type and when applied to boundary layers in open systems it is called a Görtler instability [21–23]. In some designs of supercritical aerofoils [24], regions of concave surfaces may be present, but typically they are not part of an LFC design or can be controlled by appropriate surface curvature or suction.

(d) Three-dimensional boundary layers: crossflow waves

The crossflow instability occurs in regions of pressure gradient on swept surfaces. In the inviscid region outside the boundary layer, the combined influences of sweep and pressure gradient produce curved streamlines at the boundary-layer edge. Inside the boundary layer, the streamwise velocity is reduced, but the pressure gradient is unchanged. Thus, the same balance between centripetal acceleration and pressure gradient that is present in the inviscid layer does not exist in the boundary layer. This imbalance results in a secondary flow in the boundary layer, called crossflow, which is perpendicular to the direction of the local inviscid streamline. A basic tutorial and review on the subject is given by Saric et al. [25].

When considering the control of instabilities, the crossflow is the most difficult. Whereas an accelerating $C_p$ is stabilizing to TS waves, it destabilizes crossflow. Because the mechanism is very different from that of TS and governed by the inflection point in the crossflow-velocity profile, the crossflow behaviour is often opposite to that of TS. For example, unlike TS instabilities, the crossflow problem exhibits amplified disturbances that are stationary as well as travelling. Even though both types of waves are present in typical swept-wing boundary layers, transition is usually caused by either one, but not both, of these waves. Although linear theory predicts that the travelling disturbances have higher growth rates, transition in many experiments is induced by stationary waves. Whether stationary or travelling waves dominate is related to the receptivity process. Stationary waves are more important in low-turbulence environments characteristic of flight, whereas travelling waves dominate in high-turbulence environments [26,27]. Therefore, one expects stationary waves in the flight environment.

Stationary crossflow waves often show evidence of strong nonlinear effects [28–35]. Because the wavefronts are fixed with respect to the model and are nearly aligned with the local potential-flow direction (i.e. the wavenumber vector is
nearly perpendicular to the local inviscid streamline), the weak motion of the wave convects $O(1)$ streamwise momentum, producing a strong distortion in the streamwise boundary-layer profile. This integrated effect and the resulting local distortion of the mean boundary layer lead to the modification of the basic state and the early development of nonlinear effects. As the distortions grow, the boundary layer develops a spanwise-alternating pattern of accelerated, decelerated and doubly inflected profiles. The inflected profiles are inviscidly unstable and, as such, are subject to a high-frequency secondary instability [36–39]. This secondary instability is highly amplified and leads to rapid local breakdown. Because transition develops locally, the transition front is non-uniform in span and characterized by a ‘saw-tooth’ pattern of turbulent wedges. This led to the realization that the boundary layer was ultra-sensitive to micrometre-sized roughness near the leading edge [40].

In subsequent roughness studies on a $45^\circ$ swept wing with a favourable pressure gradient, where the most unstable wavelength was 12 mm, Reibert et al. [34] used spanwise-periodic discrete roughness elements (DREs) to excite the most unstable wave. There are two important observations concerning these discrete roughness results: (i) unstable waves occur only at integer multiples of the primary disturbance wavenumber; and (ii) no subharmonic disturbances are destabilized. Spacing the roughness elements with wavenumber $k = 2\pi/\lambda$ excites harmonic disturbances with spanwise wavenumbers of $2k, 3k, \ldots, nk$ (corresponding to $\lambda/2, \lambda/3, \ldots, \lambda/n$) but does not produce any unstable waves with ‘intermediate’ wavelengths or wavelengths greater than $\lambda$.

Following this lead, Saric et al. [35] investigated the effects of spanwise-periodic discrete roughness, the primary disturbance wavenumber of which did not contain a harmonic at $\lambda = 12$ mm. By changing the forced fundamental disturbance wavelength (i.e. the roughness spacing) to 18 mm, the velocity contours clearly showed the presence of the 18, 9 and 6 mm wavelengths. However, the linearly most unstable disturbance (12 mm) was completely suppressed. Moreover (and consistent with all previous results), no subharmonic disturbances were observed, which shows that an appropriately designed roughness configuration can, in fact, inhibit the growth of the (naturally occurring) most unstable disturbance. When the disturbance wavelength was forced at 8 mm, the growth of all disturbances of greater wavelength was suppressed. The most remarkable result obtained from the subcritical roughness spacing is the dramatic effect on transition location. In the absence of artificial roughness, transition occurs at approximately 55 per cent chord. Adding roughness with a spanwise spacing equal to the wavelength of the linearly most unstable wave moves transition forward to 47 per cent chord. However, subcritical forcing at 8 mm spanwise spacing actually delays transition beyond the pressure minimum and well beyond 80 per cent chord (the actual location was beyond view).

Subsequent to the experiments, the nonlinear parabolized stability equation (NPSE) results [41] confirmed this effect. In a direct numerical simulation (DNS) solution, Wassermann & Kloker [38] have shown the same stabilization due to subcritical forcing. Using the same independent approach in terms of the calculation of the basic state, they demonstrated the stabilization due to subcritical roughness.
2. Strategies for laminar flow control

As with any flow control implementation, it is assumed at the outset that one is not trying to retro-fit an existing system, but rather flow control should be considered an integral part of the design process. In this way, the actual control outcomes are better achieved. Because of the difficulties in laminarization of swept wings, a list of strategies is presented that begins with the highest technical readiness level (TRL) and ends with the lowest TRL.

(a) Use of a suction system

If one had a real and immediate requirement for LFC on a transonic transport, we should recommend first the use of weak wall suction. Pfenninger [7] made it work on the X-21. In one of the recent flight examples, Boeing Commercial Airplane Group [15] successfully demonstrated suction HLFC on the 757 up to a transition $Re_x = 16 \times 10^6$. Everyone agrees that suction works for the attachment line, Görtler, TS and crossflow. It is well understood, and has a strong fundamental basis. Wagner et al. [42] and Maddalon et al. [43] made the case for suction LFC working in an operational environment. However, despite its successes, objections have been raised regarding the reliability, complexity, certification, cost and structural compromises of a suction system.

(b) Two-dimensional wing

If suction is impracticable, we should recommend, as the second option, reducing the wing sweep to $10^\circ$–$12^\circ$ and eliminating crossflow instabilities. Then use an accelerating pressure gradient to stabilize TS waves. Determine the requirements for efficient pressure recovery and place the pressure minimum as far aft as possible. Pressure gradient stabilization of TS waves is a reliable passive technique that has been proven to work and is the conventional design philosophy of two-dimensional natural laminar flow wings. Richard Tracy and his colleagues at Reno Air and Aerion have been making this case for years [44,45]. However, despite its successes, objections have been raised regarding wave-drag penalties, pressure recovery problems and aircraft handling qualities.

(c) Polished leading edge

If a two-dimensional wing is impracticable, we should then recommend, as the third option, having a polished leading edge. The stationary crossflow wave draws its initial amplitude from surface roughness. We have demonstrated that, with a leading edge polished to $0.3\mu m$ r.m.s., laminar flow can be achieved on a $30^\circ$ swept wing with $N$-factors up to 14 or so (see summary in §3(b) below). Thus, laminar flow can be obtained to at least the pressure minimum. One can use a leading-edge Krüger [15] to protect the polished surface during landing and take off. If a material harder than aluminium is plated on the leading edge, the surface finish can be improved and higher $N$-factors for transition can be achieved. The conjecture is that one would have no difficulty achieving natural laminar flow with a highly polished leading edge. However, despite its successes, objections have been raised regarding the operational requirements of maintaining a polished leading edge.
(d) Spanwise-periodic distributed roughness elements (DREs)

If a highly polished leading edge is impracticable, we should then suggest the use of spanwise-periodic DRE technology because there is nothing else left on the table to try. The TRL for periodic DREs is not high and significant research remains to be done. The technique has been demonstrated in flight at \( Re_c = 8 \times 10^6 \). Planning is under way to conduct additional tests on a Gulfstream III at transport unit Reynolds numbers \((M = 0.75, H = 40\,000\,ft (\sim 12\,200\,m))\) with an \( Re_c \) in the range of \((22–30) \times 10^6\). These flights will occur during mid 2012.

Wind-tunnel tests are not recommended because of the higher turbulence levels and a greater sensitivity to \( Re_k \).

3. Flight-test results for discrete roughness elements

The original wind-tunnel experiments and computations were done in a modest chord Reynolds-number range \(((2.2–3.5) \times 10^6)\) and the goal has been to extend this to higher chord Reynolds numbers more typical of flight systems. Because of the sensitivity of the crossflow instability to free-stream turbulence [46], it appears to be difficult (if not impossible) to do laminar crossflow experiments at higher Reynolds numbers \((>8 \times 10^6)\) in wind tunnels because of free-stream turbulence apparently inherent in all facilities operating above Mach 0.3.

Flight tests can be very difficult since one does not have the collection of instrumentation available to a wind tunnel. However, if one follows the guidelines of the Transition Study Group for transition research in flight and takes the care outlined by Saric [47], there is a chance for success.

The influence of free-stream disturbances must be resolved, and an important step is to carry out careful stability and transition experiments in flight, where the disturbance levels are indeed low. Such experiments should form the base state for then determining the influence of roughness. A well-known and very successful flight programme was conducted by Dougherty & Fisher [48]. Since an identical model was taken to every supersonic facility, this work actually provided a means to evaluate flow quality in high-speed tunnels. Since then, the achievements have been meagre, for a variety of reasons—not the least of which is the cost of doing flight experiments.

(a) Objectives

With the objective to investigate DRE technology in a low-disturbance, flight-test environment, a subsonic swept-wing test article was designed to be consistent with a SensorCraft-type wing section (30° leading-edge sweep). The goals are: to quantify the effectiveness of spanwise-periodic DREs in increasing the extent of laminar flow (i.e. transition location in chordwise direction) on the suction and/or pressure sides beyond the baseline (no-control) case; to investigate the robustness and utility of DREs in maintaining laminar flow over the SensorCraft flight envelope (i.e. variations in test-article angle of attack \( AoA \)) at chord Reynolds numbers, \( Re_c = 7.5 \times 10^6 \); to gain insight into conducting boundary-layer transition control experiments in a flight environment versus a wind-tunnel
environment; and to obtain a database that provides additional insight into boundary-layer stability and transition and for validation of prediction tools. The AoA for the test article was nominally set at $0^\circ$ but was adjusted to as much as $\pm 4^\circ$ using sideslip.

The program planning objectives were: (i) to measure the free-stream disturbance environment and establish that the flight test has an acceptable disturbance environment within which one can conduct boundary-layer stability and transition measurements; (ii) to develop a map of breakdown due to isolated roughness as a function of $Re_k$ and roughness location ($Re_x$); (iii) to develop the laminarization technology with DREs and determine the sensitivity of the roughness at higher unit Reynolds numbers; (iv) to determine how the low-disturbance environment of flight can validate (or invalidate) wind-tunnel experiments; (v) to complement the experiments with stability computations; and (vi) to provide programme guidelines for laminarization and long-range flight. All six objectives were met.

(b) Test results

The primary objective for phase I testing was to determine whether the in-flight turbulence intensities were low enough to proceed with the swept-wing experiment. A value less than 0.08 per cent for $u'/U_\infty$ was expected. Experimental results show that the nominal value is between 0.05 and 0.06 per cent of $U_\infty$.

Basically the target conditions for achieving 60 per cent laminar flow were a chord Reynolds number of $Re_c = 7.5 \times 10^6$, at model angle of attack of $AoA = -4^\circ$ and a sweep angle of $\Lambda = 30^\circ$. The model (see figure 1) was flown on a Cessna O-2 as an external store.

The swept-wing model was designed with an accelerated flow to 70 per cent chord. The intent was to make the boundary layer subcritical to TS waves but rather unstable to crossflow instabilities. One of the principal results is that we achieved 80 per cent laminar flow with a polished leading edge (LE) at $Re_c = 8.1 \times 10^6$, $AoA = -4^\circ$ and $\Lambda = 30^\circ$. This corresponds to linear stability $N$-factors of well over 14 for the most unstable disturbance. Background roughness was 0.3 $\mu$m r.m.s. with 2.2 $\mu$m average peak-to-peak. The $N$-factor is the log of the unstable disturbance amplitude ratio: $N = \ln(A/A_0)$, where $A_0$ is the initial amplitude at the first neutral point and $A$ is the amplitude at transition. Thus an $e^{14}$ growth is an amplitude ratio of $10^6$. The IR thermography for this case is shown in figure 2.

The colder area denoted by the dark orange colour indicates laminar flow, while the lighter area denotes turbulent flow. The flow in all figures is right to left. These conclusions were confirmed by placing large roughness elements on the model and tripping the boundary layer. The white marks at the bottom and top of the model are pieces of aluminium tape denoting 40, 60 and 80 per cent chord, respectively. The light orange colour near the top of the model is due to the cabin IR reflection. The diagonal line across mid-span is the reflection of the bottom of the aircraft. The bright area near the top is the forward propeller and forward engine exhaust reflections.

Achieving an $N$-factor greater than 14 with the polished leading edge demonstrates the low-turbulence environment of flight. Results such as these have never been obtained in wind tunnels, where $N$-factors of 8–9 have been achieved, with $N = 6$ being more common. With 80 per cent laminar flow, there is not much
Figure 1. The swept-wing model hung on a Cessna O-2. A black powder-coat finish was used to enhance the IR image. The IR camera was mounted in the cabin.

Figure 2. IR image at 170 knots, true airspeed (KTAS), $Re_c = 8.0 \times 10^6$, $AoA = -4^\circ$, $A = 30^\circ$; 3500 ft (~1070 m) above mean sea level (MSL), polished LE, no DREs; roughness: peak-to-peak = 2.2 μm, r.m.s. = 0.3 μm; $N$-factor > 14, $(x/c)_{Tr} = 80\%$.

Figure 3. IR thermography at 173KTAS, $Re_c = 8.0 \times 10^6$, $AoA = -4^\circ$, white painted LE, no DREs, $(x/c)_{Tr} \approx 30\%$.

that can be done with DREs for LFC. However, the polished leading edge with 0.3 μm r.m.s. can be considered a base state. A more realistic, operational surface would be painted.

(c) Laminarization results with a painted surface

The model surface was painted to achieve a background roughness level of 1.0 μm r.m.s. with a 3.8 μm average peak-to-peak. In this case transition moved forward to 25–30 per cent chord under conditions of $Re_c = 8.0 \times 10^6$, $AoA = -4^\circ$,
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Figure 4. IR image at 180 KTAS, $Re_c = 8.13 \times 10^6$, $AoA = -4^\circ$, white painted LE, DREs $\times 2$ placed at 1% $x/c$ at inboard pressure row, and at 1.3% $x/c$ at outboard pressure row, $d = 1$ mm, $A = 2.25$ mm, transition moved to 60% $x/c$.

Figure 5. Representative NPSE for figure 4 showing the subcritical wavelength DREs to be effective in modifying crossflow disturbance growth. Amplitudes of the ‘critical’ 4.5 mm mode are shown with and without control by a 2.25 mm disturbance with initial amplitude of $8 \times 10^{-3}$ (filled blue diamonds, $7.5 \times 10^{-4}$ initial-amplitude critical mode and no DRE control; filled pink squares, $7.5 \times 10^{-4}$ initial-amplitude critical mode and DRE control; green crosses, $1 \times 10^{-4}$ initial-amplitude critical mode and no DRE control; blue dashes, $1 \times 10^{-4}$ initial-amplitude critical mode and DRE control).

$L = 30^\circ$ and an $N$-factor $= 7$. This is shown in figure 3 and is the new base state. In this case transition moved forward to 30 per cent chord and this is our new base state.

When a double layer of DREs (12 $\mu$m high) was applied at a 2.25 mm spacing (one-half the critical wavelength), the transition location moved back to 60 per cent chord under conditions of $Re_c = 8.0 \times 10^6$, $AoA = -4^\circ$, $A = 30^\circ$ and an $N$-factor $= 12$. This is shown in figure 4. Thus, the region of laminar flow was doubled from the base state and, according to linear theory, the disturbance amplitude was reduced by $e^{-5}$ or $<10^{-3}$. This rather remarkable result demonstrates the DRE technology in flight at a chord Reynolds number of $8 \times 10^6$.

(a) Nonlinear parabolized stability equation results

Companion NPSE studies by Rhodes et al. [49] applied to the full-aircraft basic state show that the crossflow instability can be stabilized; see, for example, figure 5. Roughness receptivity studies are also presently under way.
under flight conditions in order to quantify the role of roughness amplitude in generating crossflow waves [50]. As a companion to this effort, Rizzetta et al. [51] have successfully applied accurate Navier–Stokes solutions of the disturbances generated by the micrometre-sized roughness as upstream conditions for the NPSE. Sensitivity to element configuration is predicted, and roughness receptivity is shown to be nonlinear with element height. Being able to relate roughness features to the resulting initial amplitude of the instability will benefit aircraft designers in that the goal is to provide the critical connection between stability analysis design tools and transition location prediction.

4. Conclusions

Flight experiments, supported by NPSE and DNS computations, have shown that spanwise-periodic DREs can stabilize swept-wing boundary layers at modest Reynolds numbers and provide an alternative to other means of passive LFC. The TRL of this technology can be raised by extending the work to Reynolds numbers that are representative of medium-sized transport aircraft.

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