Laminar Flow Control and Drag Reduction for Supersonic Aircraft Configurations

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Abstract

This paper describes a method of controlling transition in order to reduce drag for improving the aerodynamic performance and hence reducing the engine emissions of supersonic aircraft configurations using a baseline configuration developed within the EU FP6 Integrated Programme HISAC (High Speed Aircraft). The attachment line contamination of the highly swept wing could be controlled by a new leading edge control device, Gaster's bleeding slot. A transition control methodology of applying surface suction at the leading edge for stabilising crossflow instabilities and surface suction or cooling further downstream for stabilising Tollmien-Schlichting waves has shown that a large region of laminar flow can be achieved. The effect of external pressure gradient on the suction distribution in the leading edge region can be controlled by using a single plenum chamber with varying porosity. The transition control technique established from the baseline aircraft configuration has been applied to a full low-supersonic-boom configuration and a significant improvement in the overall aerodynamic performance due to the transition delay has been obtained. The possibility of controlling transition on the fuselage and the potential on drag reduction has also been investigated.

1 Introduction

The EU FP6 HISAC project is aimed at assessing the feasibility of an environmentally friendly and economically viable small size supersonic transport aircraft. The increasing concerns for the environmental effects of engine emissions, particularly at high altitude, have driven the need to reduce drag and, hence, fuel burn and emissions. Part of the HISAC project was to investigate the possibility of applying flow control techniques to delay transition in order to reduce drag. The Aircraft Research Association (ARA) has been involved in the investigation of transition control techniques and the assessment of aerodynamic performance. The main study has been carried out using a HISAC baseline configuration, where the results have been transferred to other partners within the HISAC consortium to investigate the control system required and its associated weight penalty. The overall information obtained has been used as input to the MDO (Multi-disciplinary Design Optimization) process for the overall aircraft configuration and engine integration. The transition control techniques and improved aerodynamic performance achieved due to transition delay for the baseline aircraft configuration are presented in this paper. The technique has also been applied to a low-supersonic-boom aircraft configuration with an installed engine powerplant and the results are included.

2 Aircraft Configuration and Pressure Distributions

The HISAC generic baseline configuration which has been used for the transition control study is shown in Figure 1. The pressure distributions across the wing span, η, for a range of lift coefficients, C_L, at the cruise Mach number of M=1.6, predicted by an Euler code are shown in Figure 2. The pressure distributions obtained are used as input to a 2.5D swept and tapered boundary layer code, which provides the velocity profiles required for the stability analysis method for transition prediction. The transition control investigation has been carried out for the wing upper surface only and the cruise altitude for the transition control study is 50,000ft.

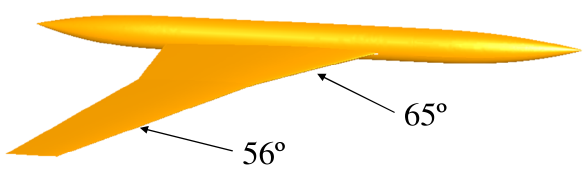

Figure 1 HISAC baseline configuration

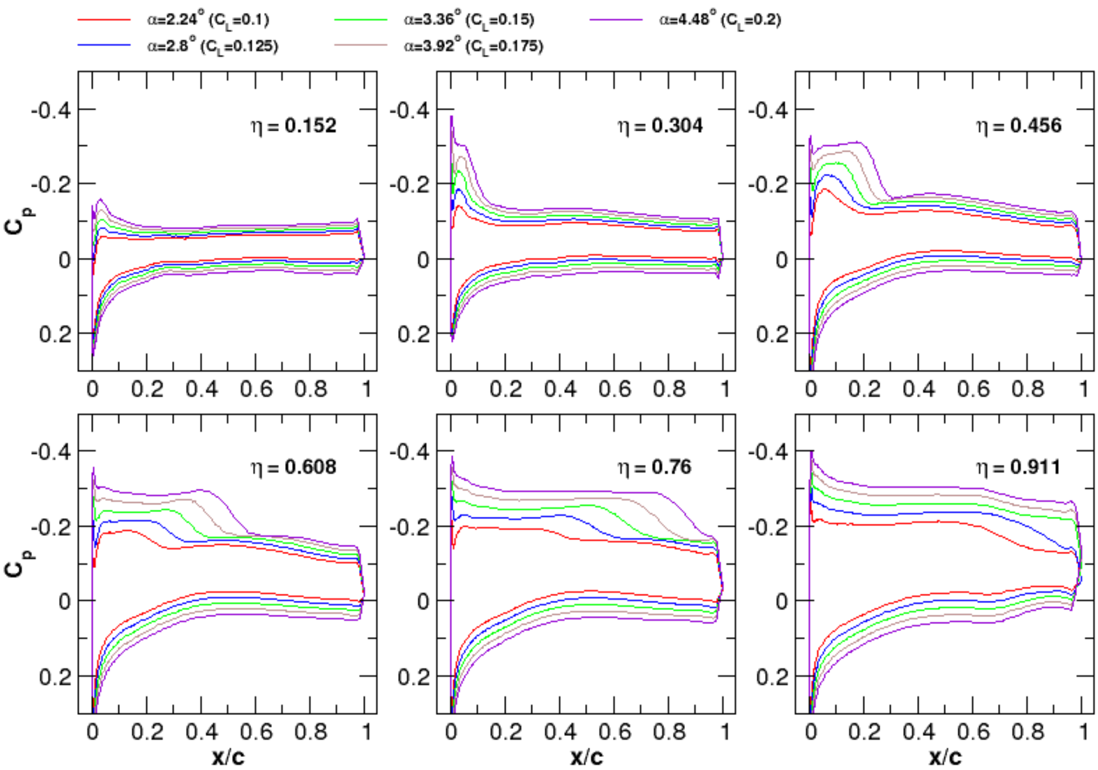

Figure 2 Wing pressure distributions for M=1.6

The pressure distributions in Figure 2 show that as C_L increases, the upper surface shock moves downstream, with a higher suction peak at the leading edge. From a transition control point of view, there may be a problem with laminar separation associated with the suction peaks. The pressure distributions on the main part of the wing are quite flat and this is considered to be beneficial for stabilising the Tollmien-Schlichting waves.

Euler computations have also been performed for a range of lower Mach numbers and various C_L conditions. For these flow conditions, the suction peak in the leading region is much higher and with a more adverse pressure gradient in the region upstream of the shock location compared with that for the cruise condition. This type of pressure distribution is not suitable for laminar flow control. Hence the laminar flow control study was mainly focused on the cruise Mach number condition, M=1.6.

3 Transition Control Methodology

On swept wings, transition from laminar to turbulent flow may be caused by one of three principal mechanisms which depend mainly on streamwise pressure gradient, Reynolds number and sweep angle. These three mechanisms are attachment line contamination, crossflow (CF) instability and Tollmien-Schlichting (TS) instability. The occurrence of attachment line transition and CF instability is mainly associated with the initial pressure gradient, though favourable ‘rooftop’ gradients will allow CF instability to persist. TS instability generally becomes dominant further aft, particularly for adverse gradients. Given that the various instability modes can be confined to different regions of the wing, a control methodology which treats each mode separately may be employed.

The attachment line contamination can be controlled by the use of surface suction or devices such as a Gaster bump. CF instabilities are due to the inflection point in the crossflow mean velocity profile. These instabilities are strongly dependent on wing leading edge sweep and the initial flow acceleration corresponding to a steep favourable pressure gradient. To suppress CF instability, surface suction is applied over the initial steep pressure gradient region with the aim of reducing the N-factor below a certain value in the linear stability analysis method. With the assumption that CF instability has been controlled and the onset of transition delayed to aft of the minimum pressure point, then TS instability would become the dominant mode for transition. For suppressing TS instability, the level of control required is much less than for CF and it may be feasible to use the technique of surface cooling instead of suction further downstream on the wing surface [1,2].

3.1 Attachment Line Contamination

The stability of the attachment line can be determined by the magnitude of the attachment line Reynolds number, $\overline{R}$. The gross contamination due to the turbulence originating from the fuselage is likely to occur when $\overline{R}$ exceeds a value of around 245. The attachment line boundary layer is also susceptible to TS disturbances. These waves first appear when $\overline{R}$ reaches a value of approximately 580, the neutral stability limit. If $\overline{R}$ exceeds 580 then the waves amplify as they travel along the attachment line and ultimately reach some threshold condition beyond which the waves break down to form localised turbulent spots. However, if $\overline{R}$ drops below 580 then the waves will be damped and eventually die out.

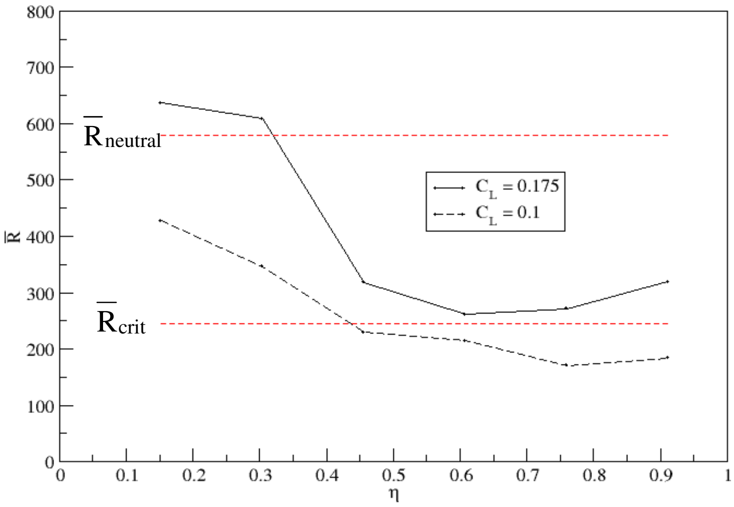

Figure 3 Attachment line Reynolds number M=1.6, Altitude=50,000 ft

The values of $\overline{R}$ across the span of the wing of the HISAC baseline configuration for the M=1.6 case are shown in Figure 3. It can be seen that the wing is susceptible to gross contamination as $\overline{R}$ is higher than 245. For the C_L=0.175 condition, the flow inboard of the wing crank (η=0.38) may become turbulent since $\overline{R}$ is higher than 580.

Recent research by Gaster [3] has shown that the attachment line instability can be controlled effectively using a device known as a 'bleeding slot'. The device is shown in Figure 4, taken from Reference [3]. This device can raise the critical value of $\overline{R}$ for attachment line contamination from 245 to about 600. In principle, from the values of $\overline{R}$ shown in Figure 3, the instability of the attachment line of the baseline configuration should be within the control limit of the device; however, further research may be required to develop the device for supersonic aircraft application.

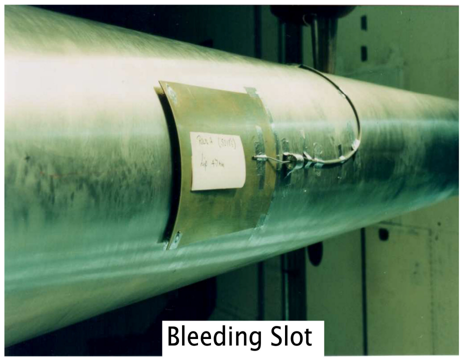

Figure 4 Gaster patented leading edge device.

3.2 Crossflow and Tollmien-Schlichting Instabilities

The pressure distributions obtained for a range of flow conditions have shown that laminar flow may only be achievable for M=1.6. An example of the variation of the N-factors predicted by linear stability analysis using a constant spanwise wave number integration strategy for an inboard station at η=0.304, without and with transition control, is shown in Figure 5. The magnitude of the N-factor is a measure of the amplification of the instability waves and each N-factor curve corresponds to a different spanwise wave number. The N-factor curves close to the leading edge are associated with CF instability, while those further aft are due to TS waves. According to linear stability theory, transition will occur when the disturbed wave has been sufficiently amplified, i.e. when the N-factor exceeds a certain value. This critical value may be determined by correlation with experiment. The N-factor for transition onset has been correlated against the experimental transition data for an ONERA swept panel wing model with a supersonic profile employed in the European project SUPERTRAC [4]. This correlation exercise was part of a work package within the HISAC programme. From a set of experimental data provided by ONERA, it was found that an appropriate N-factor value for transition is about 10 for the current method. For the inboard wing without transition control as shown in Figure 5, transition would occur at the leading edge as the magnitude of the N-factor is above 10.

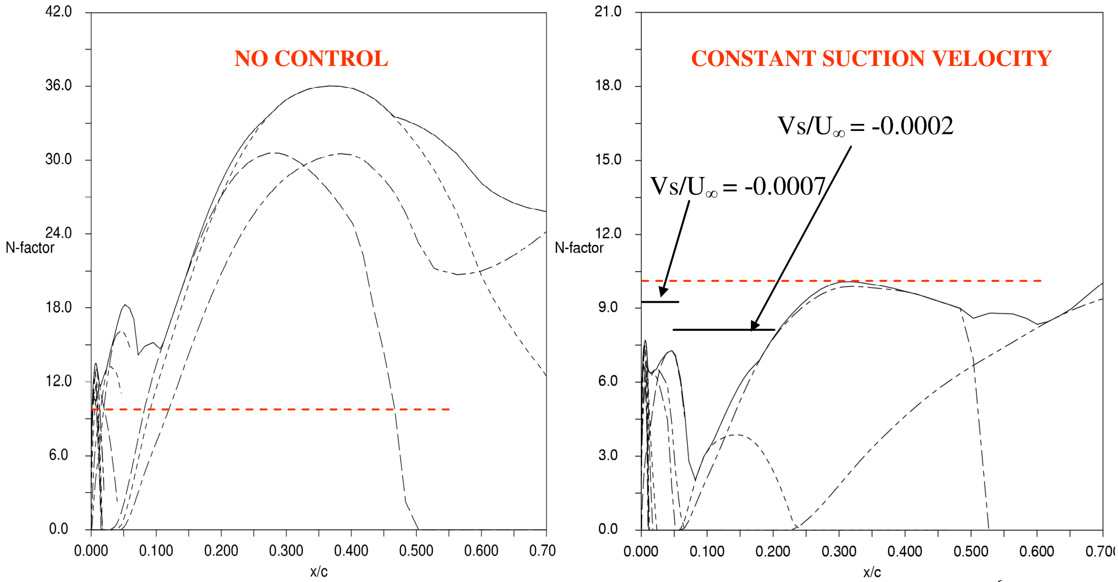

Figure 5 N-factor variation, inboard wing, η=0.304, M=1.6, C_L=0.1, Re=75.3x10^6

For the HISAC baseline configuration, it was shown that for C_L≤0.15 at M=1.6, by applying surface suction in the leading edge region upstream of 20% chord, forward of the wing front spar, transition can be delayed as far downstream as the shock location near the trailing edge. This is illustrated in the N-factor variation with transition control for a C_L=0.1 case shown in Figure 5. It can seen that by applying a surface suction velocity of V_s/U_∞=-0.0007 over the initial 5% chord for suppressing CF instabilities and V_s/U_∞=-0.0002 between 5% and 20% chord for suppressing the TS instability, transition is delayed to 70% chord. Transition can be further delayed to the trailing edge region if an additional suction or cooling panel is applied in the 70% chord region. For the higher C_L conditions, $C_L \geq 0.175$, laminar flow cannot be achieved due to laminar separation associated with the high suction peak at the leading edge as shown in the pressure distributions in Figure 2.

For stations outboard of the crank, for C_L≤0.15, transition is delayed to the shock location without the need for any control. This is illustrated in the results shown in Figure 6 for a C_L=0.15 case. For higher C_L cases, the adverse pressure gradients downstream of the leading edge suction peak are not as severe as those on the inboard part of the wing, see Figure 2. The pressure distributions on the main part of the wing are generally fairly flat and the shock locations are further aft compared with those for the inboard wing. The N-factors associated with CF instabilities in the leading edge region are not high enough to cause transition. The TS instability waves further downstream on the wing can be suppressed using surface suction or cooling.

The transition locations that can be achieved for the HISAC baseline configuration for the cruise Mach number condition are summarised in Figure 7.

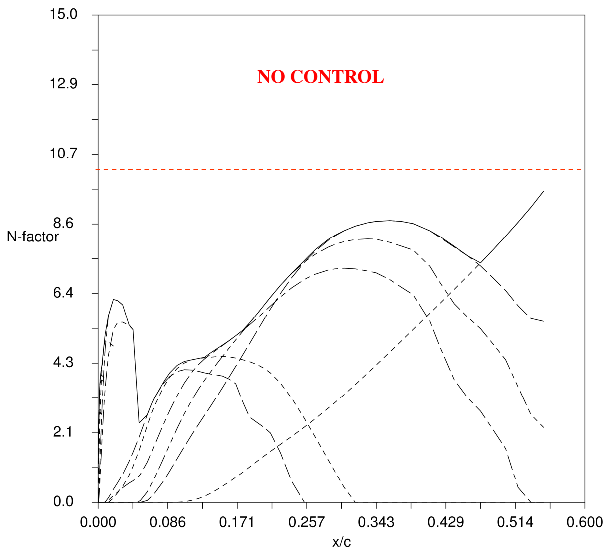

Figure 6 N-factor variation, outboard wing, η=0.760 M=1.6, C_L=0.15, Re=40.9x10^6

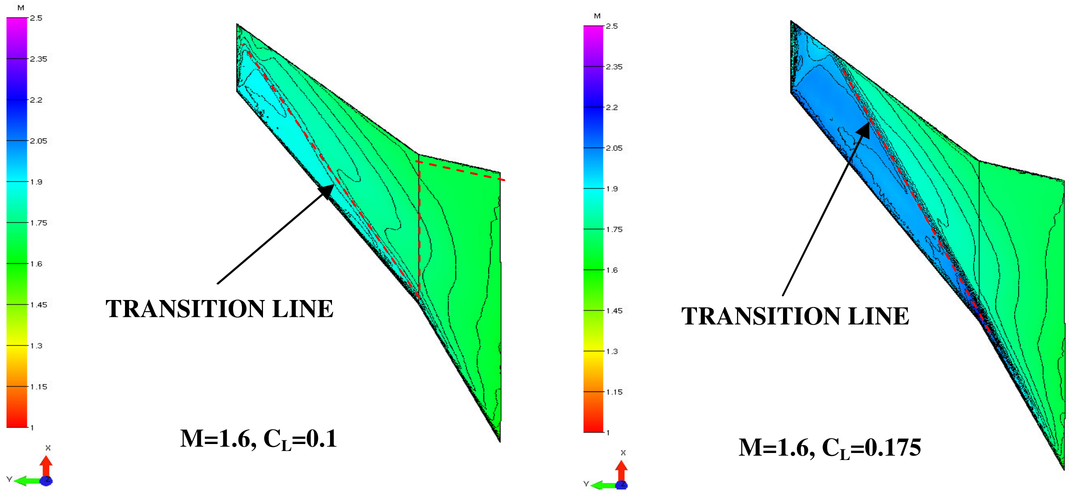

Figure 7 Surface Mach number distributions and transition locations with surface suction applied

An assessment of the effect of transition on the drag performance has been carried out by boundary layer viscous drag calculations based on the momentum thickness of the wake far downstream of the wing. For the transition locations shown in Figure 7, a reduction in the boundary layer viscous drag of 28.5% for C_L=0.1 and 11.7% for C_L=0.175 has been achieved.

3.3 Effect of Surface Cooling

For suppressing TS instability, the level of control required is much less than for CF instabilities and it may be feasible to use the technique of surface cooling instead of suction. Figure 8 shows the surface temperature distribution with various levels of surface cooling and cooling extents that have been applied in the study, for the outboard wing. The surface temperature needed to suppress TS instability is about 26K below the surface temperature of the non-cooling case. This is illustrated in the N-factor distributions shown in Figure 9. The level of surfacing cooling required could be further reduced by optimising the pressure distribution, the location and length of the cooling panel. The implication of the current results and the feasibility of the cooling system required needs to be assessed in the future.

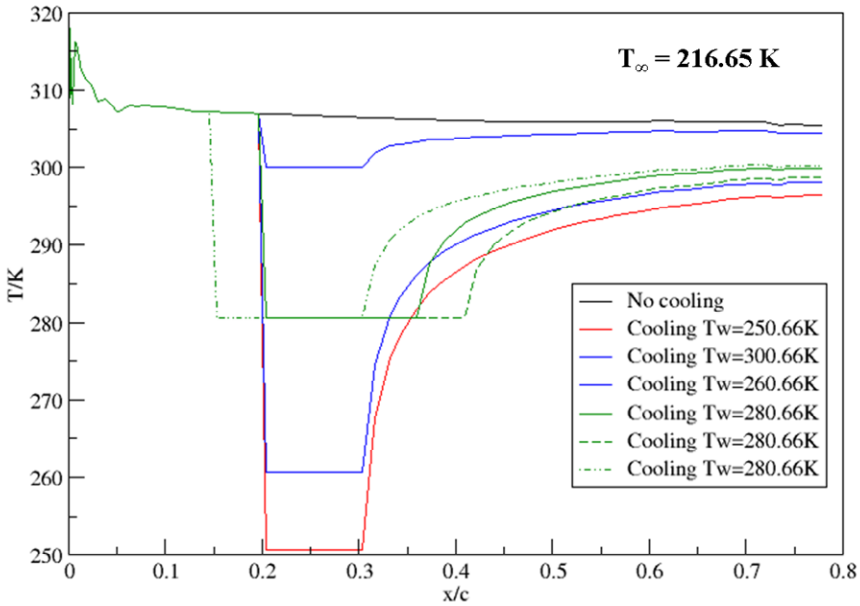

Figure 8 Wall temperature distributions, η=0.76, M=1.6, C_L=0.2

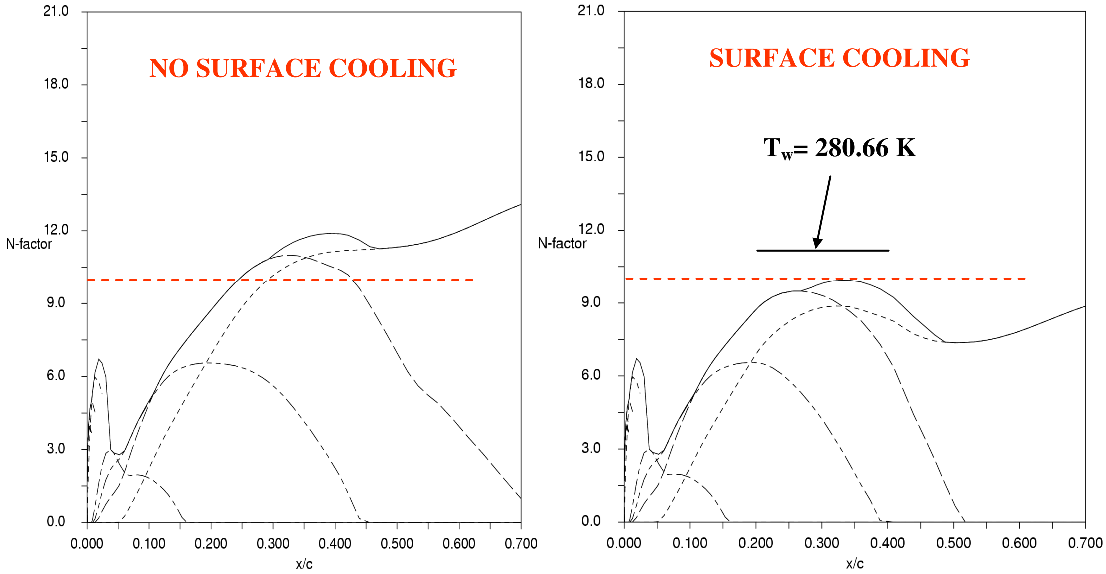

Figure 9 Effect of surface cooling on the N-factor distributions, η=0.76
M=1.6, C_L=0.2, Re=40.9x10^6

3.4 Surface Suction Velocity Distribution Control

In the earlier transition control studies, the suction requirements were obtained by assuming that the suction velocity was constant along the porous panel. Due to the effect of the external pressure gradient, a uniform suction distribution cannot be achieved when using a single suction chamber. This effect is most pronounced in the leading edge region due to the steep pressure gradient associated with the initial flow acceleration. One approach to resolving this problem is the use of a number of separate suction chambers within a double skin suction surface [5]. An alternative approach is to use a single plenum chamber and a porous surface panel with varying porosity to alleviate the effect of pressure gradient. This approach has been investigated for pressure distributions and flow conditions relevant to military and civil transport aircraft [6,7]. This concept has been extended to supersonic configurations using the HISAC baseline configuration. The study was carried out for the inboard part of the wing at M=1.6 and C_L=0.1, where laminar flow could be controlled using surface suction as shown in Figure 5.

It is assumed that suction is provided using a single chamber with the porous panel applied from the wing leading edge to 20% chord downstream as in Figure 5. The chamber pressure must be set at an appropriate level in order to avoid blowing for the constant porosity case; however, the resultant suction velocity presented in Figure 10, is insufficient to control transition as shown in Figure 11. The porosity of the panel is defined by the hole spacing to the hole diameter ratio, S_d. The effect of the pressure gradient on the suction velocity, can be controlled by reducing S_d in the initial accelerated flow region, and then increasing it downstream of the minimum pressure point linearly with surface distance. This is illustrated in Figure 10 where S_d is reduced linearly with distance by a factor of a half from 0 to 1% chord and then increased it linearly by a factor of a half from 1% to 20% chord. The effect of this variation in porosity on the surface suction velocity distribution, assuming the same chamber pressure, is also shown in Figure 10.

The effect of the porosity of the suction panel on the N-factor distribution is shown in Figure 11. It can be seen that using the varying porosity technique, a significant extent of laminar flow has been achieved if transition onset is assumed to occur at N-factor of 10. The results obtained show that the technique of varying the panel porosity can be used to alleviate the effect of external pressure gradient on the suction velocity distribution for controlling transition for supersonic aircraft configurations.

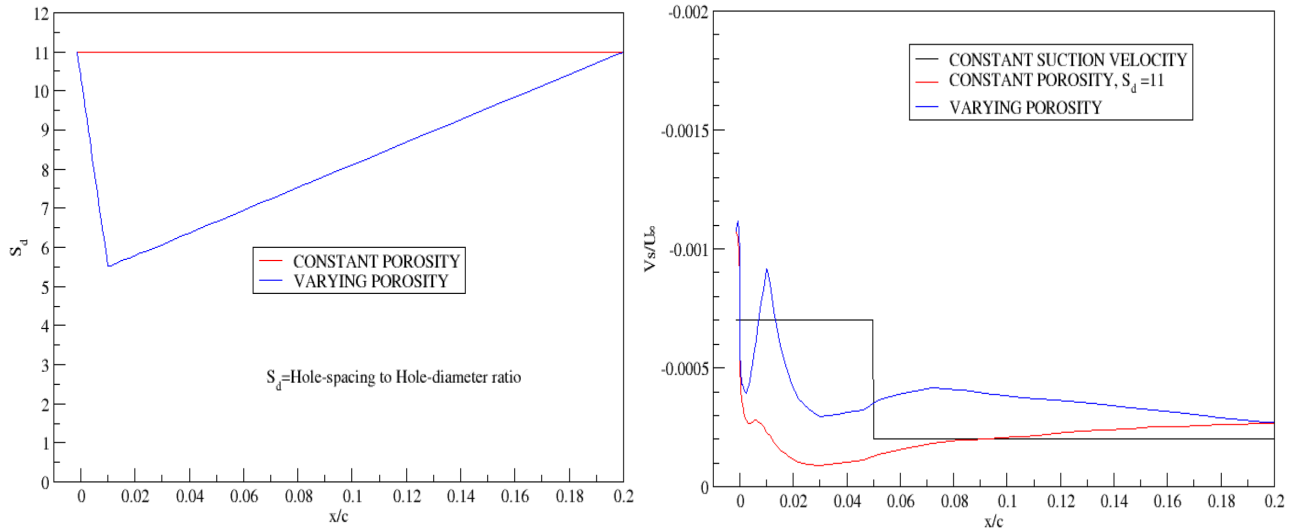

Figure 10 Effect of porosity variation on suction velocity distribution

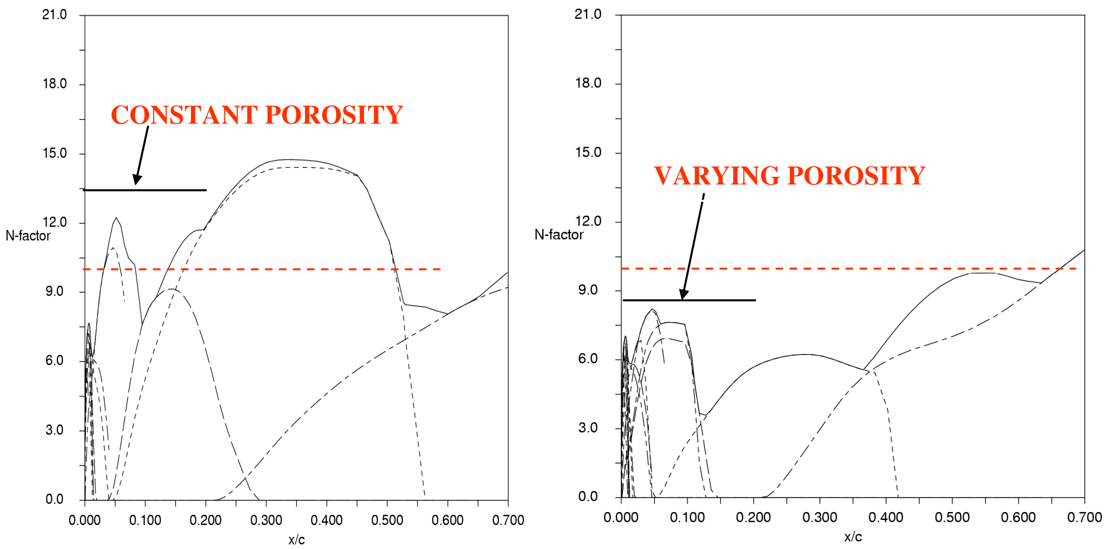

Figure 11 Effect of porosity distribution on the N-factor distributions
Inboard wing, η=0.304, M=1.6, C_L=0.1, Re=75.3x10^6

4 Transition Control for a Low-Supersonic-Boom Aircraft Configuration

The transition control methodology established using the HISAC baseline aircraft configuration has been applied to a low-supersonic-boom aircraft configuration. However, although in principle, the technique of varying the porosity distribution could be used for this aircraft, the study considered here assumes a constant suction velocity distribution. The study has been carried out for the wing upper surface at the cruise Mach number condition. The flow solutions have been obtained using a RANS code, with the Spalart-Allmaras turbulence model. An example of the Mach number distribution for M=1.6, α=3° case at 50,000ft is shown in Figure 12. The pressure distributions for the inboard and outboard part of the wing for various angles of incidence are shown in Figure 13.

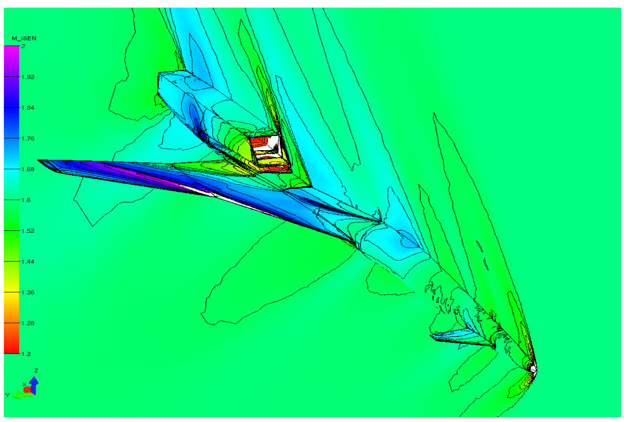

Figure 12 Mach number distribution, M=1.6, α=3°, altitude=50,000 ft

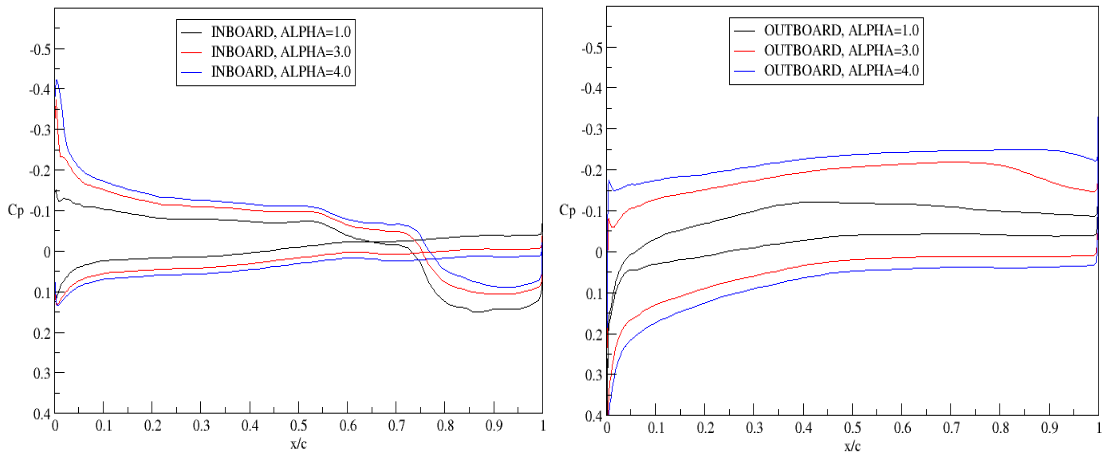

Figure 13 Pressure distributions, M=1.6, altitude=50,000 ft

For the inboard wing at lower angle of incidence, i.e. α=1°, transition can be delayed using surface suction in the leading edge region. For the higher angle of incidence cases, it is not possible to delay transition for the inboard wing due to laminar separation associated with the high suction peak, as apparent in the pressure distributions in Figure 13. With the assumption that surface suction can be applied from leading edge to 20% chord, forward of the wing front spar, transition can be delayed to 44% chord for N-factor of 10. This is illustrated in the N-factor distributions shown in Figure 14 for the α=1° case.

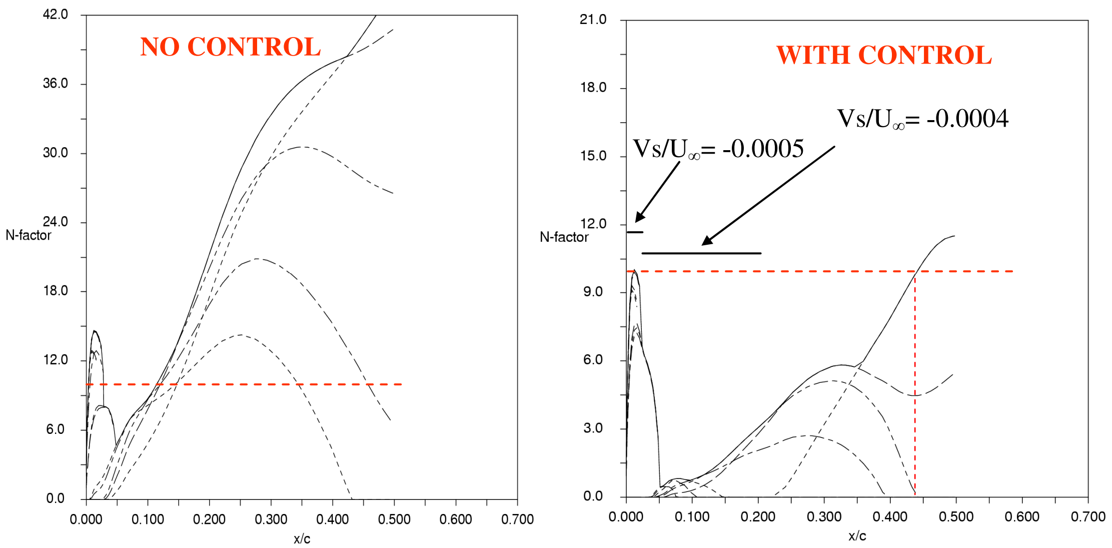

Figure 14 N-factor distributions, inboard wing, M=1.6, α=1°

For the outboard wing where the pressure gradient is favourable, transition can be delayed to about 44% chord using surface suction for a range of angle of incidences. The level of suction required is less than that for the inboard wing. Figure 15 shows the N-factor distributions for the M=1.6 and α=4° case, where a suction velocity of -0.0002 has been applied in the initial 20% chord region.

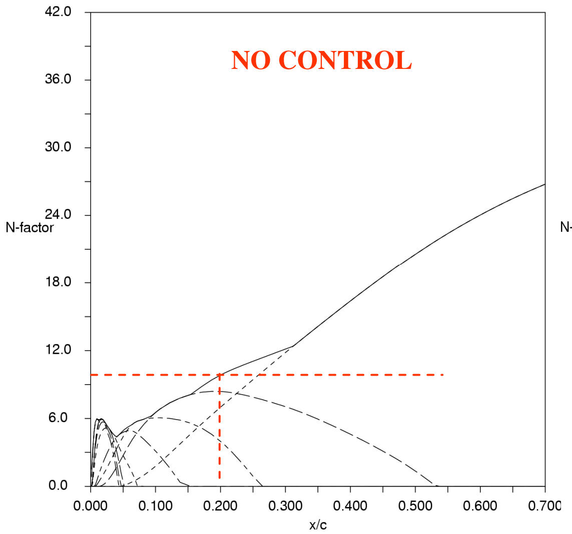

Figure 15 N-factor distributions, outboard wing, M=1.6, α=4°

The drag reduction due to the transition delay compared with the fully turbulent flow cases for the complete aircraft is about 3.6% of the total aircraft drag for α=1° and 1.4% for α=4°. These results were achieved with the constraint of limiting the transition control to within the first 20% chord region. The aircraft was designed for fully turbulent flow and the pressure distribution has not been optimised for laminar flow. Further laminar flow extent could be achieved by optimising the pressure distribution and the application of surface suction or cooling downstream on the wing to suppress the Tollmien-Schlichting instability waves.

5 Future Laminar Flow Control Study

The drag of the fuselage contributes a large percentage of the aircraft total drag. The possibility of controlling transition for the fuselage in order to reduce drag has been investigated. This was carried out using the HISAC natural laminar flow configuration, since the fuselage geometry is identical to that of the low-supersonic-boom configuration but without canards. The study was carried out for a flow condition of M=1.6, C_L=0.18 at altitude of 50,000 ft. The surface Mach number contours and the pressure distribution along the top and bottom side of the fuselage are shown in Figure 16.

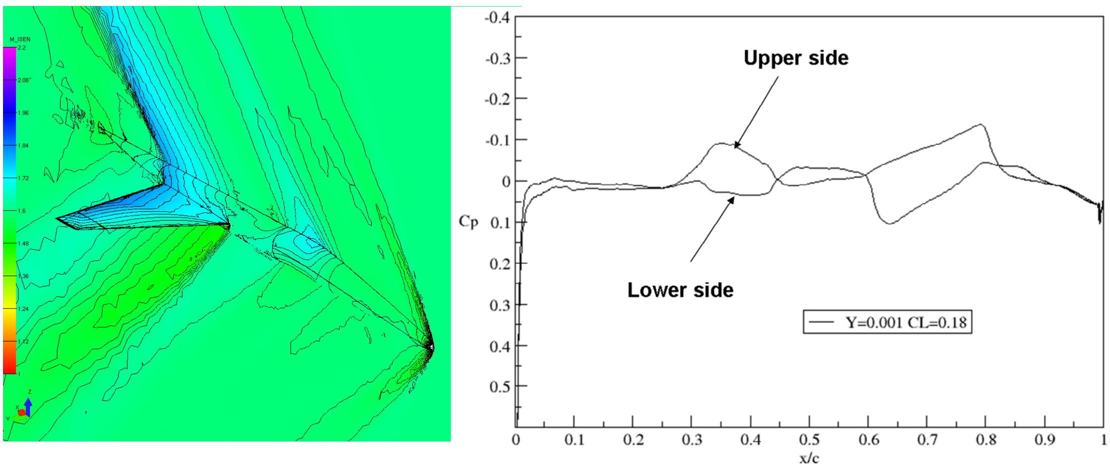

Figure 16 Mach number contours and fuselage pressure distributions
M=1.6, C_L=0.18, altitude=50,000ft

The N-factor distributions for the fuselage upper and lower sides without transition control are shown in Figure 17, which indicate that transition would be at the nose of the fuselage if a N-factor of 10 for transition is assumed. The results presented in Figure 17 assumed an effective sweep of 10º in the boundary layer calculation and transition prediction to take account of some degree of 3D effects on transition. A higher effective sweep of 20º has also been investigated but the effect on transition was found to be negligible. This may be due to the dominant effect of the extremely high Reynolds number for the fuselage. It is emphasised that further research in this area is required and the need to use a fully 3D boundary layer and transition prediction method [8].

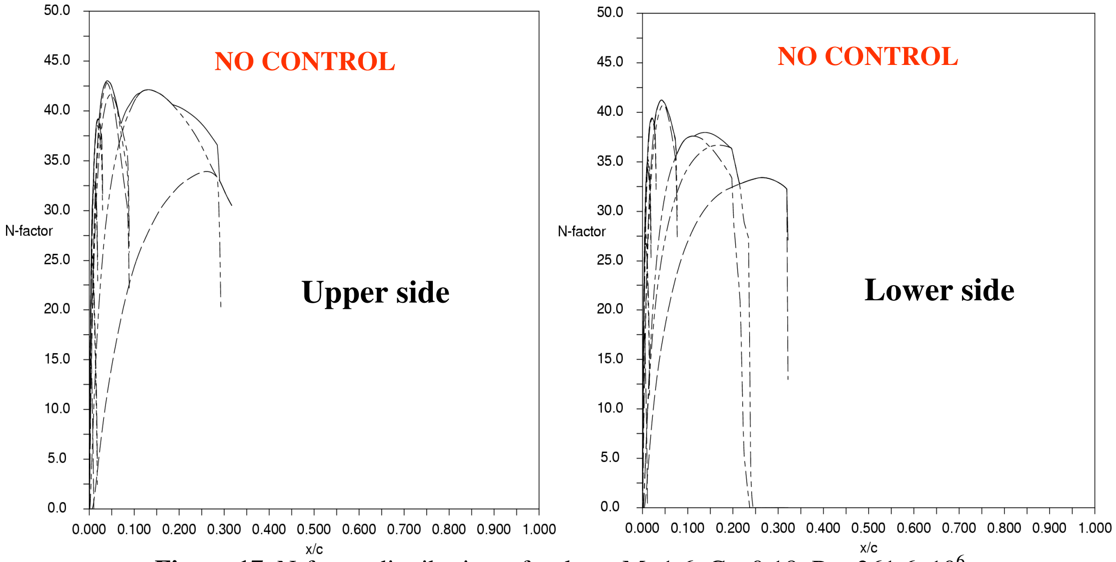

Figure 17 N-factor distributions, fuselage, M=1.6, C_L=0.18, Re=261.6x10^6

However, the current study found that transition on the fuselage may be delayed using a discrete distribution of surface suction at various locations along the fuselage as shown in Figure 18. The suction panel lengths for the case shown are 2% fuselage length. It can be seen that transition is delayed to about 32% fuselage length for an N-factor of 10, where laminar separation is predicted to occur due to the adverse pressure gradient as shown in Figure 16.

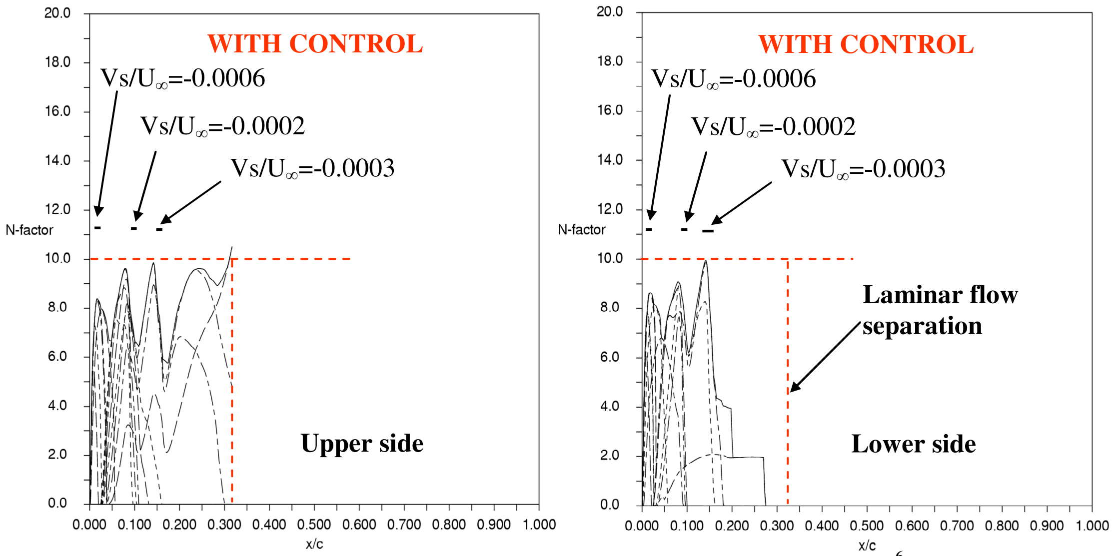

Figure 18 N-factor distributions, M=1.6, C_L=0.18, Re=261x10^6

The above results suggest that there is a potential for achieving a greater length of laminar flow on the fuselage with an appropriate pressure distribution. The delay in transition has reduced drag by 11.3% for the fuselage relative to that for the turbulent flow case. The potential of laminar flow control for other components such as the canard and nacelle can be exploited in the future.

5 Conclusions

A significant extent of laminar flow can be achieved by the hybrid laminar flow control technique for supersonic aircraft configurations. For the HISAC baseline configuration, a drag reduction of 28.5% for C_L=0.1 and 11.7% for C_L=0.175 has been achieved for the cruise Mach number condition, M=1.6. For the low-supersonic-boom configuration, a drag reduction of about 3.5% of the total aircraft drag for α=1° and 1.4% for α=4°, at M=1.6, has been obtained.

The level of surface cooling required for suppressing Tollmien-Schlichting waves has been found to be about 26K below the surface wall temperature. The use of surface cooling would reduce some of the problems of structural constraint and maintenance of the porous holes associated with surface suction. The practicality of the cooling technique will need to be assessed in the future.

The technique of varying the suction panel porosity can be used to alleviate the effect of external pressure gradient on the suction velocity distribution for controlling transition. The ability to use a single suction chamber would reduce the structural complexity and weight penalty associated with the suction system.

It may be possible to delay transition on the fuselage which would lead to further drag reduction. Initial predictions have shown that a reduction of fuselage drag by 11.3% could be achieved. The results obtained in the current investigation have shown the potential of transition control and drag reduction for performance improvements for supersonic aircraft configurations.

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