A review of recent developments in flow control

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ABSTRACT

This paper reviews and highlights recent developments of certain aspects of flow control concerned with reducing the drag of, and delaying flow separation on, wings and bodies over which the flow is turbulent. The study is restricted to devices that extend beyond the viscous sub-layer but are on a smaller scale than geometric features of the aircraft (e.g. wing chord).

The review is mainly concerned with developments within the UK, although significant developments in other countries are discussed.

The review discusses types of flow that need to be controlled, basic features of flow control devices and applications. It concludes with recommendations for future research.

NOMENCLATURE

alpha	vane toe angle or deflection
AJVG	air jet vortex generator
c	aerofoil chord
CFD	computational fluid dynamics
C_D	drag coefficient
C_{DW}	wave drag coefficient
c_f	local skin friction coefficient except in Fig. 10 where it is flap chord
C_L	lift coefficient
C_N	normal-force coefficient
c_n	chord normal to sweep
C_p	surface pressure coefficient
C_μ	jet momentum coefficient, $= \frac{M_j}{\frac{1}{2}\rho_e U_e^2 S}$
d	diameter of either perforations in surface, air jet or mass-less jet
D	diameter of plenum chamber of mass-less jet (Section 3.2.3)
f	frequency of vibration of mass-less jet
G	vortex strength function, Equation (3)
H	boundary-layer shape factor (Section 3.2.1)
H	depth of plenum chamber of mass-less jet (Section 3.2.3)
h	maximum height of SBVG above wall, mm
h_e	SBVG effective height
h^+	Reynolds number based on h and u_τ
h_s	spacing between limiting streamlines
h_v	height of vortex above wall
I	initial point on vortex trajectory in cross-flow plane
L	vane length (Table 1)
M	point at which vortices of counter-rotating vanes merge
MEMS	microfabricated-electro-mechanical-systems
M_j	jet momentum
M_s	Mach number at displacement surface normal to and just upstream of shock wave
M_∞	free-stream Mach number
NLF	natural laminar flow
n	integer used to define spacing between vanes (Table 1)
P	point on leading edge denoting onset of leading-edge separation
p	pressure
R	Reynolds number based on chord
RANS	Reynolds averaged Navier Stokes
r	leading-edge radius of aerofoil made non-dimensional by aerofoil chord

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S	saddle point (Fig. 3) or reference area for calculation of aerodynamic coefficients
SBVG	sub boundary layer vortex generator
S_T	Strouhal number, fd/U_e
t	thickness of perforated surface in aerofoil (Fig. 17)
tc	thickness of plate covering mass-less jet plenum (Section 3.2.3)
u_τ	wall friction velocity, Equation (4)
U_e	main-stream speed, ms^{-1}
VG	vortex generator
x	chordwise or streamwise distance from wing leading edge
y	lateral distance from plane of symmetry vertical distance from wall
z	vertical distance from wall

Greek symbols

α	angle of incidence
δ (or delta)	boundary–layer thickness, except in Fig. 10 where it is flap deflection
Δ	increment
ε	volume flow along streamtube
Γ	vortex strength
η	non-dimensional spanwise position, i.e. spanwise distance from wing root divided by wing span (Fig. 21)
κ (or k)	surface curvature
ν	kinematic viscosity
ω_w	magnitude of wall vorticity

Subscripts

e	main stream conditions
H	hole where jet emerges into flow
t	trailing edge of device
0	stagnation conditions (Fig. 9)
0·5	0·5h downstream of SBVG
5	5h downstream of SBVG
v	pertains to the vortex

1.0 INTRODUCTION

Since the first manned flight, efforts have been devoted towards providing efficient control of aircraft flight. The Wright biplane used wing warping to provide lateral control[1] — later to be superseded by ailerons on other aircraft. Following loss of life during the First World War from aircraft stalls due to leading-edge flow separation, thought was given to ways of preventing this from happening[2]. This resulted in the development of the Handley Page slot or leading-edge slat[1, 2].

As aircraft weight and speed increased during the 1920s and 30s, the need for additional forms of lift enhancement was recognised. Handley Page developed the slotted flap[1] and not long afterwards the Fowler flap was invented[1]; this combined the idea of the slotted flap with the use of increased wing area by translating the flap aft as it was deployed. For further details of flap designs developed in this period the reader is referred to Ref. 1.

The appearance of jet propulsion in the 1940s increased aircraft speed still further. This led to new problems due to the appearance of shock waves on wings and from their interaction with the boundary layers. The increase in drag due to the shock waves meant that either the aircraft maximum speed was limited or that there was a limit on maximum range. Shock/boundary-layer interaction could, in an extreme form, lead to shock-induced separation and to buffet. Increasing wing sweep was an effective way of delaying these problems but they remained as important limiting factors. Consequently, activities were undertaken subsequently to understand these flows and to find ways of controlling them. Among the many suggestions considered are: surface suction, tangential blowing and vortex generators, which were extensively studied, notably by Pearcey[3] and his colleagues at the National Physical Laboratory (NPL), Teddington.

Developments in the understanding of transonic flows over aerofoils led in the 1960s to the appearance of the supercritical aerofoil, which employed a relatively flat upper surface along with increased rear camber. This enabled increased speed and range or, for a given Mach number, increased wing thickness, thereby increasing fuel volume compared with those obtained with conventional aerofoils. However, this approach is based on design for a single point within the flight envelope, and attempts to improve performance for a number of flight conditions by designing for multiple points results in a compromise design. Thus there is scope for flow control in improving the performance of wings with supercritical aerofoils at off design conditions.

After the fuel crisis of 1973, efforts were redoubled towards reducing drag more drastically by promoting laminar flow on wings, using much earlier work on the subject as the basis. The requirements for laminar flow aerofoils differ from those for supercritical aerofoils in that the former requires largely favourable pressure gradients upstream of the shock whereas the latter feature adverse pressure gradients. The result is that the shock waves on laminar-flow aerofoils increase rapidly in strength with changes in speed or incidence. Hence this type of aerofoil also benefits from flow control to improve off-design characteristics.

During the second half of the 20th century interest increased in methods of controlling flow separations on swept-back wings at manoeuvre conditions, prompted by developments in methods for calculating 3D boundary-layer flows and a better understanding of 3D flows. Three forms of flow control were proposed and investigated:

1) *Design the lifting surfaces so that flow separations are avoided.* This can be done to some extent by minimising adverse streamwise pressure gradients on the lifting surfaces, such as by using camber or variable camber.
2) *Ensure that any separations occur smoothly from sharp edges.* This is the philosophy underlying the choice of sharp trailing edges for aerofoils, and it ensures that the separation line is fixed, hence providing a substantially steady flow. Maskell and Küchemann[4] advocated this philosophy for leading edges of highly swept wings on supersonic aircraft and it was used effectively on Concorde. A significant disadvantage is that flow separation from sharp leading edges implies a loss of leading-edge thrust and hence high drag at manoeuvre conditions.
3) *Utilise flow control devices to control flow separations.* This philosophy has the potential to provide the aircraft designer with the means to relax aerodynamic constraints, for example on wing taper ratio, thickness and loading. In addition, it allows the possibility of designing lifting surfaces of relatively simple shape compared with those of 1) and 2) above. It may also allow simpler flap systems to be designed. This means that flow control may be an effective means of reducing aircraft first cost.

This paper is intended to review recent developments in flow control applied to external flow surfaces (say within the last 10-20 years). For the purpose of this paper we define flow control to mean devices that do not materially alter the shape of the aircraft. More particularly these are devices that largely reside within the boundary layer and cause barely visible or no changes in surface shape (e.g. bumps, wall transpiration or vortex generators (VGs) and sub boundary–layer vortex generators (SBVGs)), but which extend outwards beyond the viscous sub layer.

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With this definition, devices such as flaps and slats are excluded from the discussion, as are micro grooves or 'riblets' for the control of turbulence near the wall.

We also exclude from the discussion laminarisation of boundary layers, which is a field that merits its own review.

The emphasis of the paper is on controlling two aspects of flows over modern, high-speed aircraft, namely, shock waves and their interaction with the boundary layer as well as separations from smooth surfaces due to adverse pressure gradients. By so doing it is possible to reduce drag or delay the onset of buffet.

The application of flow control may be seen in either of two ways:

i) To provide a 'fix' to cure problems that were not foreseen in the original design and that only emerged late on in the development process;
ii) As part of a multi-disciplinary design process.

It might be argued that the first of these ways has been made redundant by developments in computational fluid dynamics. However, experience would not support this suggestion, with problems being reported during flight trials of a number of fairly recent projects. It is the view of the authors that flow control will remain as a means to aid the designer as a 'fall back' position.

The second way of using flow control has yet to be fully exploited, partly, perhaps, because the science is not yet considered sufficiently mature and maybe because some designers view flow control only in the first of the two ways, i.e. something to fall back on if all else fails. This attitude (if it still exists) misses the many opportunities for multi-disciplinary design that flow control affords. For example, flow control can be used to enable a thicker wing to be designed, which leads to lighter wings with greater fuel volume. The third author, K.C. Hackett, in some unpublished work at QinetiQ, has shown that SBVGs on the outer panels of a swept wing for a civil transport aircraft would allow the wing to be more highly loaded in this region, enabling a more tapered wing. This would result in a wing design that is lighter and with higher fuel volume.

This paper will show that flow control is a maturing (although not fully matured) science, and has reached the stage where it can be considered part of a multi-disciplinary design process. As well as a number of wind-tunnel studies, some flight experience has been obtained and computational methods are beginning to be developed that model flow control.

The paper will be divided into a number of sections. Section 2 will include discussion of transonic flows with shock waves (wave drag and shock-induced separation) and low-speed flow separations, emphasising the importance of understanding the flows to be controlled. This will be followed in Section 3 by a review of the fundamentals of flow control devices and then by a discussion of the applications and effects on the flows controlled (in Section 4). The paper concludes in Section 5 with some recommendations for further work.

Although work outside UK is referred to and discussed, greater emphasis is placed on studies done in UK. This is because the material is readily available to the authors, some of which has not been published before. For international perspectives the reader is recommended to read the excellent review papers by Stanewsky[5] and Lin[6]. This review also complements these reviews by discussing research performed since they were written.

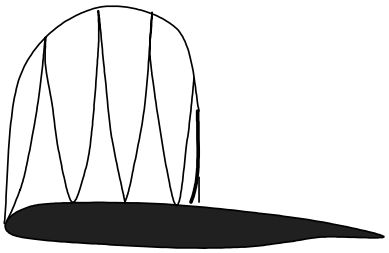
2.0 FLOWS THAT NEED TO BE CONTROLLED

On the basis that it is important to understand the flow that is to be controlled before controlling it[7], we introduce and describe flows that benefit from some form of control in this section.

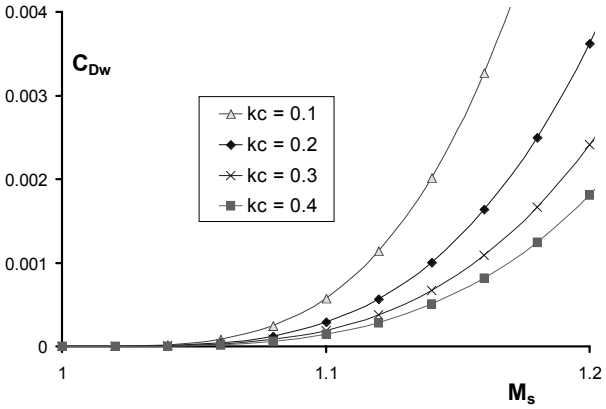
2.1 Shock waves

Shock waves may appear on subsonic aircraft when the local Mach number of the flow normal to isobars exceeds unity. The consequent increase in entropy or loss of total pressure through the shock wave results in an increase in drag, the component of which is referred to as 'wave drag'. Sufficiently strong shock waves, with a Mach number normal and just upstream of the shock wave, M_s, in excess of about 1·2, say, may lead to flow separation.

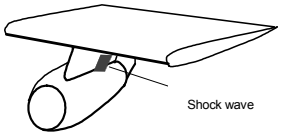
Fig. 1 illustrates examples where shock waves may be found. The first example, Fig. 1a, is the transonic flow over an aerofoil. For a given free-stream Mach number and angle of incidence, the Mach number just upstream of a shock on an aerofoil is determined by the position of the sonic point on the surface and the shape of the upper surface between there and the shock.



(a) Transonic aerofoil with shock wave



(b) Lock's approximation for aerofoil wave-drag coefficient



(c) Wing/pylon/nacelle combination with shock wave in the gully



(d) Shock wave on fuselage above cockpit

Figure 1. Examples of flows where shock waves may occur.

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A simple, approximate formula for the wave-drag coefficient of a two-dimensional aerofoil that represents the effect of surface shape and shock strength[8] points to possible ways of reducing wave drag. This formula, which is based on reasonable assumptions about the flow just upstream of the shock wave, may be expressed as:

$$C_{DW} = \frac{0\cdot 243}{c\kappa}\left[\frac{1+0\cdot 2M_\infty^2}{M_\infty}\right]^3 \frac{(M_S-1)^4(2-M_S)}{M_S(1+0\cdot 2M_S^2)} \quad \ldots (1)$$

Here κ, surface curvature, is the reciprocal of surface radius of curvature, M_∞ is free-stream Mach number and c is aerofoil chord.

Figure 1(b) shows results of calculations using Equation (1) and indicates that wave drag coefficient can be decreased either by reducing shock upstream Mach number, M_s, at constant surface curvature or increasing surface curvature at constant shock Mach number. In reality, the two parameters are not independent of one another, since increasing surface curvature in the convex sense usually increases shock Mach number. However, as shown in Fig. 1(a), shock upstream Mach number is clearly an important parameter, since wave drag increases very rapidly with increases with shock Mach number above a value of 1·1.

Lock[8] showed how Equation (1) may be extended to allow for sweep. Among the changes to the formula to allow for sweep, the term M_s becomes the shock Mach number normal to sweep.

Shock strength and wave drag may therefore be influenced by careful design of the aerofoil or wing upper surface. Alternatively, flow control can be used to replace the shock, at least partially, by multiple shocks or isentropic compression waves, which lessen the entropy rise or stagnation pressure losses directly. Such control strategies, both 'active'† and passive, will be considered in some detail in Section 3. A way of modifying Lock's formula to allow for the effect on the shock structure of a bulge or bump in the surface is described by Ogawa and Babinsky[9].

Other examples of where shock waves are found on external surfaces shown in Fig. 1 include wing/pylon/ nacelles, part c), and above or near the cockpit on fuselages, part d).

2.2 Flow separations

Examples of flows with separations from smooth surfaces are shown in Fig. 2; these include leading and trailing edge separations on aerofoils and wings (Fig. 2(a)), separations on the rear of bodies, at wing-body junctions and for wing-pylon-nacelle configurations (Fig. 2(b)). Flow separations are caused by adverse streamwise pressure gradients, an extreme example of which is the shock wave, as noted above (Fig. 2(a)), but they may also be provoked by flow convergence and cross flow, as found, for example, on afterbodies and swept wings.

The basic principles underlying flow separation were set out by Maskell[10] and Lighthill[11], and are illustrated by the sketches in Fig. 3. They deduced a relationship between the volume flow, ε, along a streamtube of rectangular cross section, the distance between the limiting streamlines forming the base of the streamtube at the wing surface, h_s, the variable height of the streamtube, z, and the magnitude of the vorticity at the wall, ω_w,

$$\frac{1}{2}\omega_w z^2 h_s = \varepsilon \quad \ldots (2)$$

Equation (2) reveals that there are two possible mechanisms for flow separation, which occurs when the streamlines greatly increase their distance from the surface. The first mechanism occurs when the magnitude of the wall vorticity vanishes. This form of separation results in a bubble in which the separation surface encloses fluid that is not of the main stream (Fig. 3(a)). The separation has one singular or saddle point[11], *S,* where $\omega_w = 0$; here the behaviour is similar to that of flow separations in two-dimensional flow on aerofoils.

It follows that this type of flow separation can be prevented by increasing the magnitude of the vorticity at the wall near the singular point. This can be done in a number of different ways. Examples are given below:

a) increased mixing between upper and lower layers of the boundary layer (e.g. by vortex generators);
b) decreasing adverse pressure gradient upstream of separation (e.g. body shaping);
c) increasing energy of lower layers directly (e.g. by tangential blowing near wall);
d) introducing vorticity in or just above the boundary layer that is of opposite sign to that required at the wall to give increase velocity in this region (e.g. small aerofoils above the wall).

Three-dimensional flow forms of flow control can also be used to break up the bubble, since it is known that separation bubbles are sensitive to three-dimensional flow influences.

The second type of flow separation takes place when the distance between the limiting streamlines becomes very small. This type of flow separation (Fig. 3(b)) is sometimes referred to as a separation along a line of ordinary points or nodal points[10], such as occurs on highly swept wings. The best example of such flow separations is the leading-edge separation on aircraft such as Concorde. For convenience this is referred to in this paper as 'ordinary separation'. In controlling separations of this sort it is necessary to prevent narrowing of the gap between the limiting streamlines just upstream of separation. This can be done by promoting motion of the fluid close to the surface in a downstream direction that is normal to the (original) separation line and by opposing it along the same line. Expressed in a different way, this can be achieved by rotating the wall vorticity vector so that there is a non-zero component in the direction of the flow along the original separation line. This can be done in one or other of the following ways:

a) introducing vorticity of an appropriate sign and direction into the boundary-layer (e.g. co-rotating vortex generators (see Fig. 4));
b) increasing mixing in the boundary layer;
c) decreasing the magnitude of the streamwise adverse pressure gradient;
d) increasing energy of the lower part of boundary layer directly.

Examples will be given of methods that set out to achieve the types of control referred to above.

3.0 FLOW CONTROL DEVICES – BASIC STUDIES

The flow control devices to be reviewed in this paper can be divided into two broad groups:

i) Ones that have a favourable direct effect on the main or inviscid flow outside the boundary layer (e.g. by reducing shock strength), and influence the boundary layer indirectly by altering the streamwise pressure distribution on the surface of the body. Any 'direct' boundary-layer control effect (that is favourable) is slight.
ii) Others that affect the flow directly within the boundary layer, and hence influence the main flow only by changing the boundary-layer displacement surface. Any favourable direct effect on the main flow is small.

† In this paper 'active' is used to refer to devices whose shape is changed to provide the type of steady flow control needed.

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Leading-edge separation



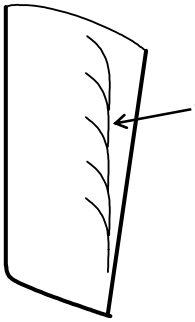
Separations from flap 'knuckles'



Trailing edge separations on rear cambered aerofoils



Trailing edge separation on slotted flaps

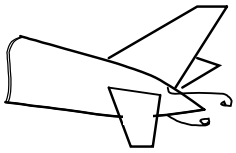


Swept wing separation

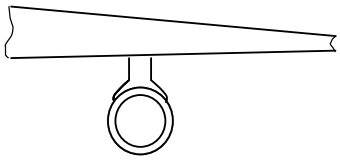


Shock-induced separation

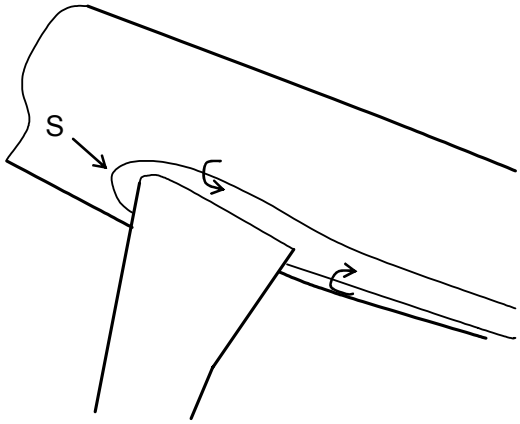
(a) Flow separations on aerofoils and wings



Afterbodies



Gullies between various surfaces

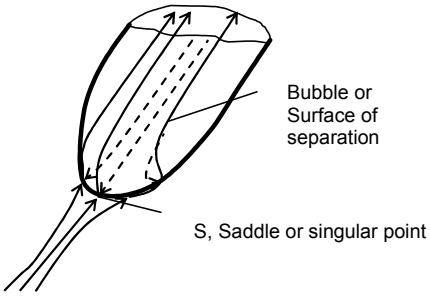


Corner flows

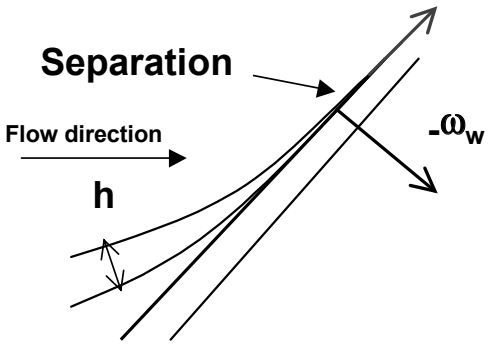
(b) Flow separations at bodies and junctions

Figure 2. Examples of flow separations on wings and bodies.

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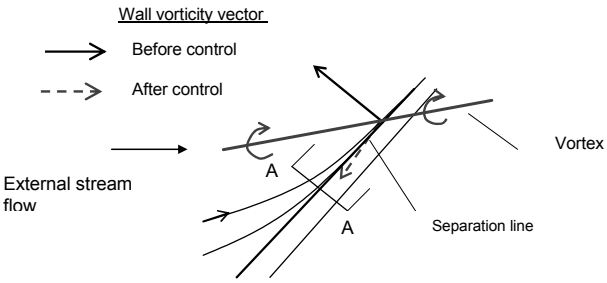


(a) Singular separation

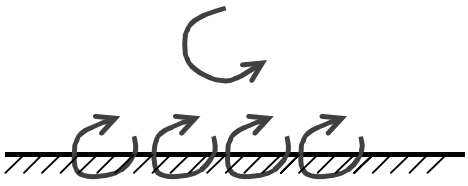


(b) Ordinary separation

Figure 3. Types of flow separation.



(a) Plan view



b) View on Section AA.

Figure 4. Vorticity induced at wall by single vortex (as generated by a co-rotating VG) near an ordinary separation.

For convenience we classify these two sets of devices as 'Inviscid flow control' and 'Boundary layer or wake control'.

3.1 Inviscid-flow control

Within the definition of flow control used in this paper, we are concerned with types of devices that cause significant changes in the flow outside the boundary layer by exploiting the sensitivity of the flow to small changes to the wall boundary condition†. Figure 5 shows four examples of such devices on an aerofoil, which are discussed in turn below.

3.1.1 Devices for changing leading-edge radius or camber

a) Two-dimensional low-speed flows
Detrimental leading-edge separation may occur by either of two mechanisms:

- Turbulent separation, or
- The bursting of a laminar separation bubble.

These separations depend on the severity of the adverse pressure gradient near the leading edge, as well as Reynolds number and boundary-layer state. One way of reducing the adverse pressure gradient in this region is to reduce the magnitude of the suction peak at the leading edge.

Results of some calculations made using a simple (inviscid, incompressible) method for the flow over an aerofoil are shown in Fig. 6 for an angle of incidence of 10°. In this figure surface pressure coefficient is plotted against $\pm\sqrt{x}$, where x is chordwise distance downstream of the leading edge. Here the positive root signifies the upper surface and the negative root the lower surface. The figure shows that small changes in leading-edge radius can cause large changes in the suction peak near the leading edge particularly for aerofoils of small leading-edge radius. Thus a device that can bring about this change could possibly be used to control leading-edge separation.

Similar control effects can be expected from devices that create small changes in the effective camber at the leading edge (see Part 1) Fig. 5). Either way, changes could be made pneumatically (e.g. by blowing through the aerofoil surface), although the authors are unaware of such a technique being tried in practice. Another approach is to use mechanisms to change the surface shape locally.

b) Effect of wing sweep and compressibility
Although wing sweep and compressibility may affect the nature of flow separation near the leading edge, the flow control exerted by change in aerofoil nose radius or camber is not expected to alter in character with increases in sweep or free-stream Mach number.

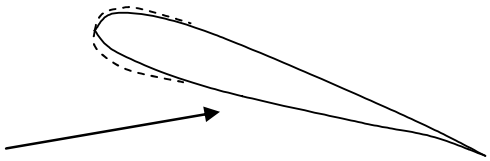
3.1.2 Changes in effective shape of wing in region of a shock wave

a) Two dimensional flows
As noted in Section 2, decreasing the Mach number just upstream of the shock reduces wave drag, and this can be achieved by compressing the flow in this region by a bump or bulge in the aerofoil surface. However, there are additional effects as discussed below.

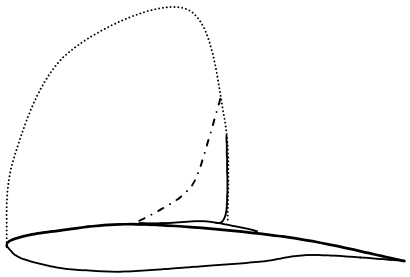
The effect of a bulge in the boundary–layer displacement surface in the region of a shock wave is illustrated schematically in Figs 7(a), (b) and (c). Shown there is the effect on the flow at the model surface (indicated by surface Mach number distributions) and in the field of altering the streamwise position of a 'bump' in the displacement surface relative to the shock at a given aerofoil Mach number and lift

† Here 'small' means that the change in shape, typified by change in local ordinate, is small compared with a characteristic dimension, e.g. aerofoil chord for flows over aerofoils.

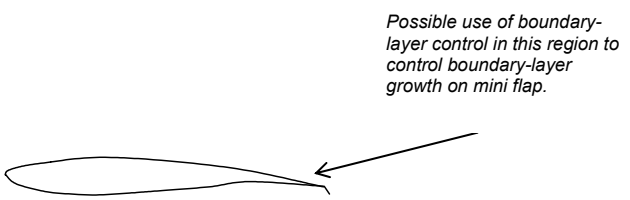
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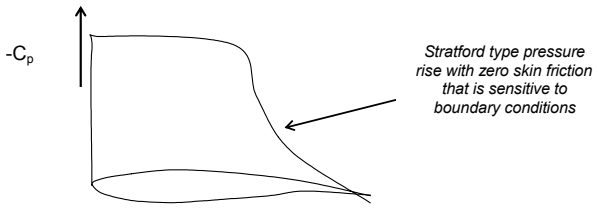
1) Devices for changing leading-edge radius or camber to alter the suction peak on the aerofoil near the leading edge.



2) Devices that change the effective shape of an aerofoil in the region of a shock wave.



3) Methods of altering the effective camber near the trailing edge.



4) 'Compliant' flow.

Figure 5. Four methods for controlling main flow using small changes in surface shape.

coefficient. Typically, for all the types of shock control devices to be discussed in this paper, the amplitude of the 'bump' is of the order of 0·1% aerofoil chord. The bump shape is taken to be such that it does not alter the surface curvature significantly in the region between the bump leading edge and a point just upstream of the crest of the bump.

Part a) of the figure shows the flow pattern when the shock wave is close to the leading edge of the 'bump'. In this case the windward part of the 'bump' acts as a ramp; the compression waves formed upstream of the ramp reduce the Mach number just upstream of the normal shock near the bump surface. The reduction in pressure rise through the normal shock has the effect of reducing the growth of the boundary layer, hence delaying the onset of a shock-induced separation bubble to higher angles of incidence or lift. This type of shock control offers possible benefits for delaying the onset of buffet onset, which suggested the name of 'buffet bump' in a previous paper[12]. The reduction in shock strength close to bump surface also leads to reduced wave drag.

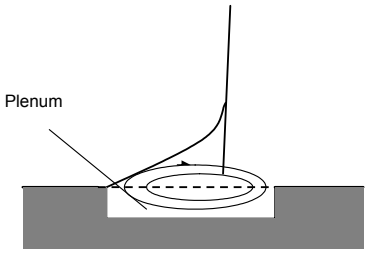
Part b) shows the result of moving the bump further upstream so that the shock sits downstream of the leading edge of the 'bump' but upstream of the 'bump' crest. A weak oblique shock is formed at the leading edge of the bump, extending into the flow field. Depending on the shape of the 'bump' between the oblique shock and the main shock, the pressure may either rise or fall on the aerofoil surface towards the shock. However, in the case considered, the pressure and Mach number stay roughly constant between the bump leading edge and the shock. Thus the bump reduces shock strength at the wall, as indicated in the pressure distribution. It also reduces shock strength on that part of the normal shock between the bump and the 'triple point', i.e. the point at which the two shocks meet. This results in reduced wave drag. This can be reconciled with Lock's formula[8] on the basis that surface curvature is unchanged by the bump at the normal shock but shock upstream Mach number is reduced (see Fig. 1(b)). However, this neglects the increase in entropy across the oblique shock, which increases as the ramp angle of the bump increases. Ogawa and Babinsky[9] allowed for this effect in their modification of Lock's method, and showed that the reduction in wave drag reaches a maximum at a ramp angle of between 2° and 4°, depending on bump length.

Further movement of the 'bump' upstream, so that the main shock is just downstream of the crest (Part c)), results in an increase in the length of the normal shock affected by the compressive effect of the 'bump'. However, the normal shock is downstream of a region where expansion waves are formed, which *increases* the Mach number just upstream of the shock. In the example shown this effect more than counters the compressive effect of the 'bump'. This effect is partially mitigated, as far as wave drag is concerned, by the local curvature of the surface, which reduces the normal gradient of Mach number just upstream of the normal shock.

In scenarios a) and b) of Fig. 7 the part of the normal shock that is close to the surface is significantly weakened by the bump. In scenario c), on the other hand, the shock is strengthened in this region. Therefore it might be expected that in the first two scenarios, impermeable, solid surface devices would decrease viscous drag while in the last it would increase it. This is borne out by calculations described in Ref. 13 using the transonic aerofoil CFD code BVGK[14], especially for strong shocks. However, as implied from previous remarks, this effect is small compared with that on wave drag, although there may be extreme examples where the bump causes the boundary layer to separate causing large increases in boundary-layer drag.

As before, this form of control may be achieved either by surface shape changes or pneumatically. Pneumatic devices involve either the passive or active transfer of air through the wing or aerofoil surface. A sketch illustrating the flow over a passive system is given below.

This transfer becomes possible for passive, pneumatic systems only when there is a pressure difference between the front and rear



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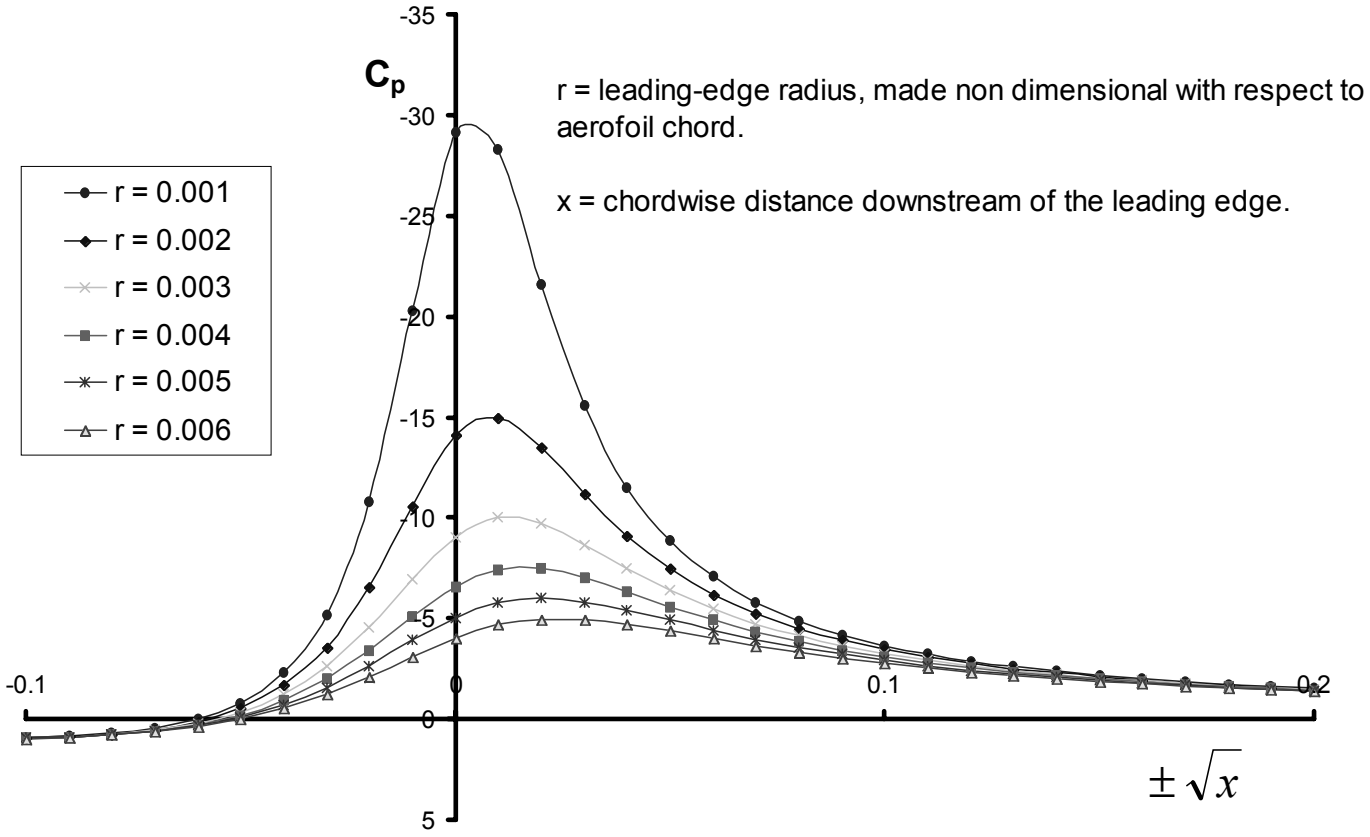
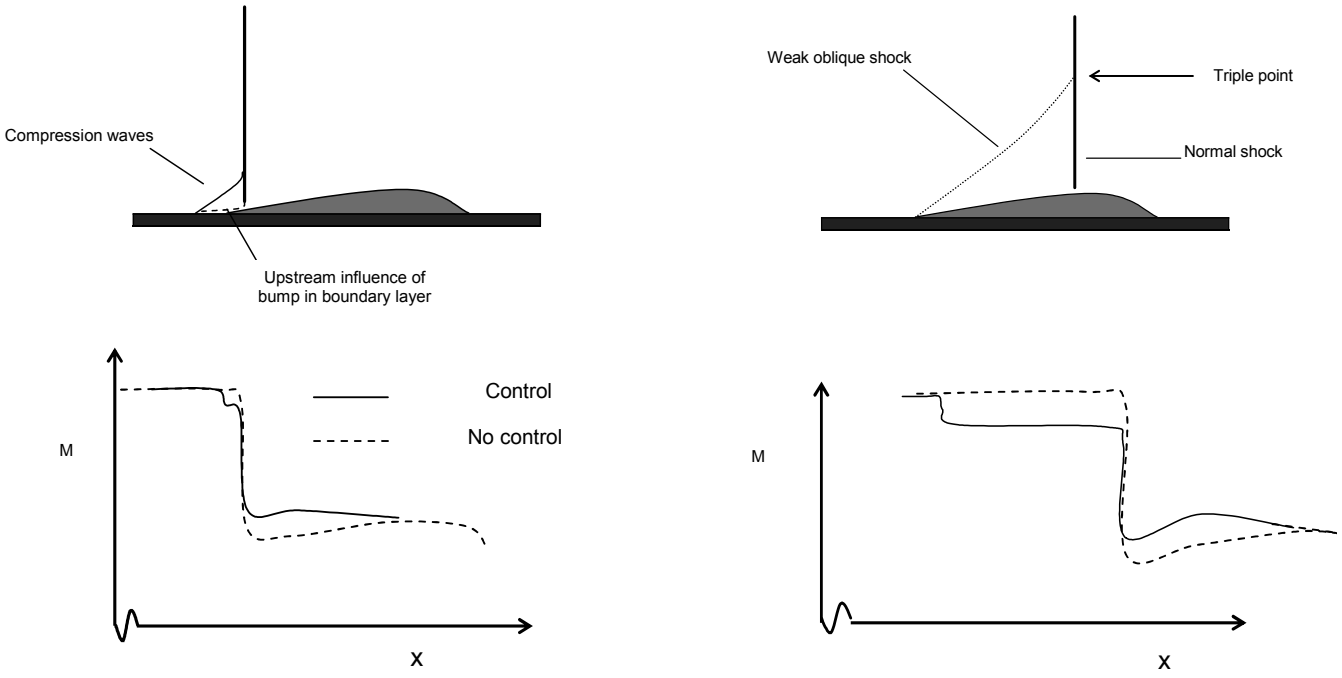


Figure 6. Calculated pressure distributions on surface of aerofoil showing the effect of leading-edge radius.



(a) 'Buffet bump'

(b) Shock just upstream of bump crest

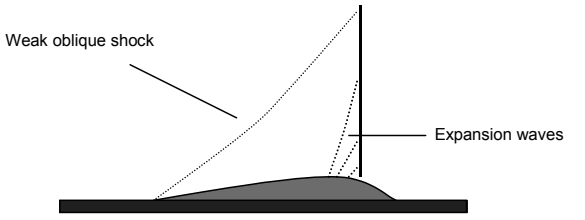
Figure 7. Surface shapes and pressure distributions, illustrating the controlling effects of bumps on shock waves.

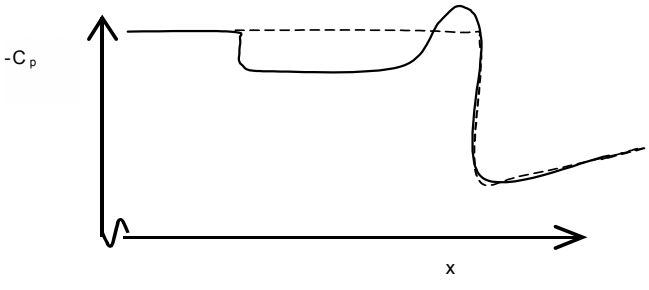
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of the control region, and this can only occur if the shock wave is directly above the control region. Hence scenario a) in Fig. 7 is probably not realistic for passive pneumatic systems. Pneumatic devices will also have an effect on the boundary layer, and could therefore also be considered to be boundary-layer control devices. However, as will be discussed later, they appear to be more successful in reducing shock strength than in controlling the boundary layer. Indeed, there is evidence that passive, perforated surfaces exert an unfavourable effect on the boundary layer[15] by thickening it downstream of the control region and increasing viscous drag.

Another, more recent, proposal for a pneumatic device, based on the use of streamwise slots or grooves in the aerofoil or wing surface may offer improved performance in this respect[12]. To gain a basic understanding of the flow generated by streamwise slots beneath a shock wave, tests have been performed in a two-dimensional insert in a supersonic wind tunnel at the University of Cambridge[16-18] These experiments have shown that, as well as reducing shock strength, streamwise slots introduce trailing vortices into the flow. These vortices are likely to control shock-induced separation and any separations that might occur further downstream. In addition, any unfavourable effect of the slots on the boundary layer is localised in the region of the slots[18]. Thus slots are hybrid devices in that they exert some direct control both on the inviscid flow and the boundary layer, although it is thought that the former effect is the more significant. In the outer or 'inviscid' flow the effect of streamwise slots is surprisingly two-dimensional. The overall effect of streamwise slots is well summarised by a conceptual sketch taken from Ref. 19 and shown in Fig. 8.

As well as streamwise slots with a common plenum, grooves or slots without a plenum (of the same open area ratio) have been studied at Cambridge[19]. Grooves were investigated because they may be more practicable than slots with a plenum on wings for operational aircraft. The effect of tapering the grooves and adding baffles within the grooves was studied. The baffles were arranged to obstruct the streamwise movement of the flow within the grooves.





(c) Shock downstream of crest of bump.

Figure 7. cont.

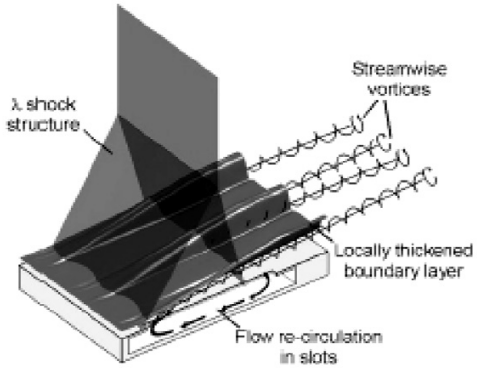


Figure 8. Conceptual sketch of shock/boundary-layer interaction with streamwise slots[19].

The grooves were found not to be as effective as the slots in controlling the flow through the shock wave. However, introducing a baffle into the groove produced a shock structure that is very similar to that with a slot. This is illustrated by pressure distributions, taken from Ref. 19, in Fig. 9.

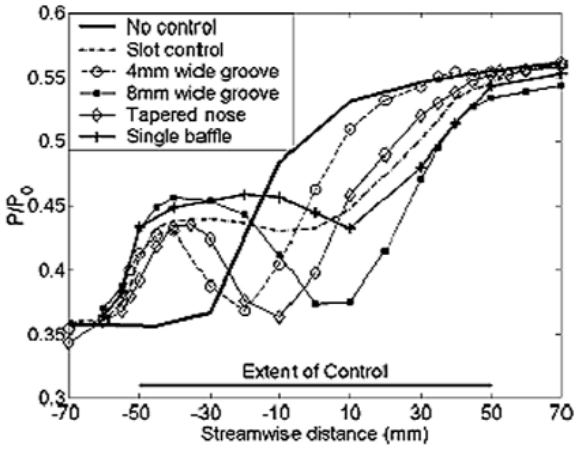
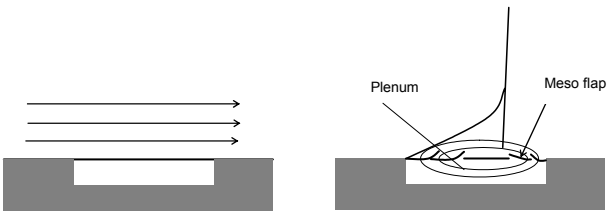
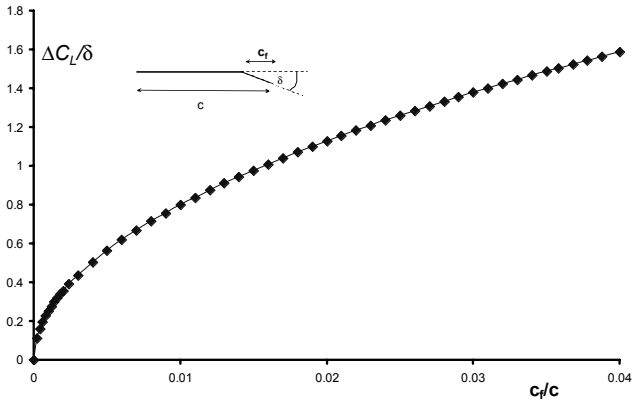


Figure 9. Wall pressure distributions for various shock control devices, $M_s = 1{\cdot}3$[19].

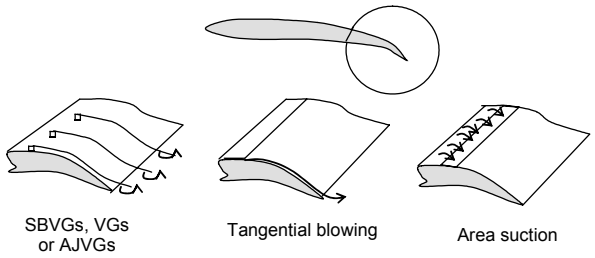
Basic studies of a form of passive control of normal shocks using 'meso flaps' have been undertaken mainly in the USA[20] but also as a collaboration between the Australian Defence Force Academy and the University of Cambridge[21]. In this scheme an array of aeroelastically tailored flaps are placed beneath the shock. As shown in the sketch below these flaps remain closed where there is no shock and deflect under the action of the pressure difference across caused by the presence of a shock. By so doing they cause the recirculation that is necessary to control the shock.



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(a) Lift coefficient increment due to a trailing edge flap.



(b) Means of reducing excessive boundary-layer growth over the rear of the aerofoil by boundary-layer control.

Figure 10: Lift coefficient increment due to a trailing edge flap and means of reducing excessive boundary-layer growth or flow separation over the highly-cambered rear of an aerofoil.

Hence, in principle, these devices do not increase drag when a shock is not present.

b) Effects of sweep
Sweep effects on the performance of solid cylindrical bumps are expected to be straightforward as far as wave drag is concerned, and can be assessed by applying simple sweep theory to 2D results. It is expected that the effect of sweep on the influence of solid bumps on viscous drag will be small as long as the bumps are properly positioned in relation to the shock.
Sweep effects on the performance of pneumatic devices have been investigated[22] and shown not to be significant when the Mach number normal to the shock wave is kept the same and the flow is attached.

3.1.3 Methods of altering the effective camber near the trailing edge

a) Two-dimensional low -speed flows
Glauert's thin aerofoil theory[23] for aerofoils of small camber reveals that the effect of camber on lift coefficient is heavily weighted towards the trailing edge. This is brought out well by calculations using this method for the case of a small flap. Figure 10a shows that significant increases in lift coefficient are predicted for quite small flap chords. This classical result is repeated here to emphasise that camber is potentially most effective when applied very close to the trailing edge. The effectiveness of such camber changes will, of course, be adversely affected by viscous effects, occurring in the form of increased boundary–layer growth or even separation in this region†. If necessary, boundary-layer growth could be reduced by a boundary-layer control device, as shown in Fig. 10(b) and to be discussed in more detail below.

b) Sweep and compressibility effects
Sweep effects may alter the nature of any flow separations in this region but the character of this inviscid flow control is not expected to change. Compressibility will not be a significant, direct factor in designing this type of flow control.

3.1.4 'Compliant' flows – flows close to rear separation

Fulker and Alderman[24] have noted that for flows close to rear separation, it is possible to achieve large changes of lift by very small changes in boundary conditions. Fulker and Alderman refer to these flows as *compliant*.

Basic aspects of how this form of flow control is affected by wing sweep are not yet understood.

† For example, separation could be expected at the leading edge of the simple flap, although having a cambered surface with continuous curvature would perhaps reduce this tendency.

3.2 Boundary-layer or wake control

There are a number of ways of exerting direct control of turbulent boundary–layers, including by suction, tangential blowing and by increasing the mixing in the boundary layer. These control methods generally work by increasing the 'fullness' of the boundary-layer velocity profile, making the boundary layer less prone to separation. Blowing normal to the surface reduces skin friction and hence skin-friction drag. However, normal blowing increases boundary-layer momentum thickness and consequently also the total viscous drag (skin friction plus form drag). It also increases the tendency towards flow separation.

Most of the work in recent years on boundary-layer control has been devoted to vortex generators (VGs) to increase entrainment into the boundary layers. VGs may take the form of vanes or obstacles, air jets or mass-less jets. Therefore this section is almost wholly devoted to this form of control, although some mention is made of simulated bird feathers and slot suction, in the latter case to control shock waves.

3.2.1 Vortex Generators (vane or obstacle type)

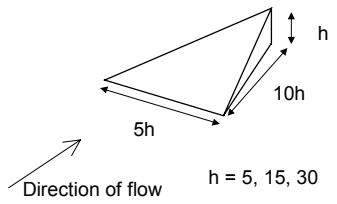
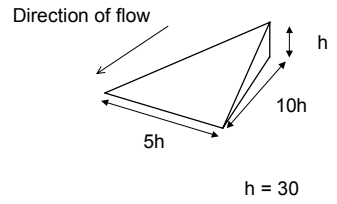
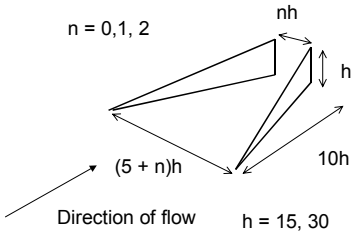
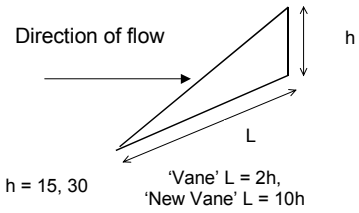
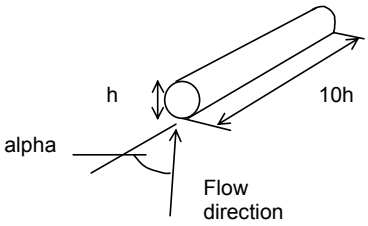
a) Two -dimensional low-speed flows
VGs have been in use for aeronautical applications for many years, and Pearcey[3] and Freestone[25] (ESDU) have comprehensively reviewed the subject. The concept is therefore well understood. However, there have been more recent developments concerned with the use of micro or Sub Boundary-layer VGs (SBVGs)[6, 26-30] that need to be appreciated.

Lin[6] pointed out that 'micro' vortex generators have been given different names in various publications on the subject. He used the name 'low profile vortex generators'. It should be mentioned, however, that a device that might be termed 'low profile' is not necessarily completely submerged within the boundary layer. For example, a device that is 'small' and within the boundary layer near the trailing edge of a wing, say, may be largely outside the boundary layer if placed near the attachment line. For the purpose of this review we will take SBVGs to have a height to boundary-layer thickness ratio (h/δ) of less than 0·5.

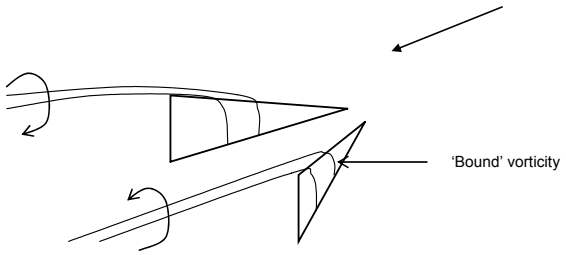
The main function of SBVGs, like VGs, is to increase the mixing in turbulent boundary layers between the high-energy layers in the

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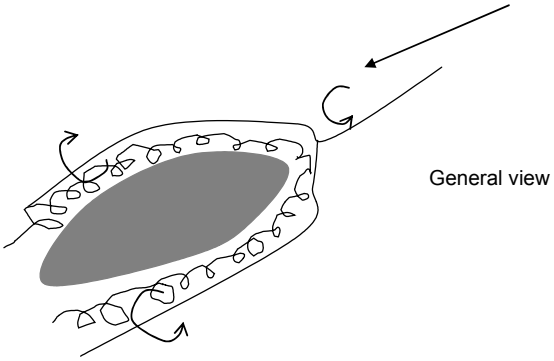
Table 1
Types of SBVGs studied at QinetiQ[32,33]

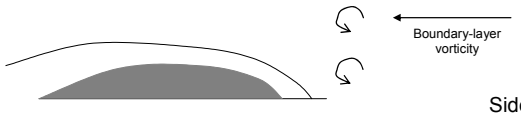
Device type	Sketch
Forwards wedge	
Backwards wedge	
Counter-rotating vanes n= 0 joined vanes	
Single rotation vanes	
Dowel	

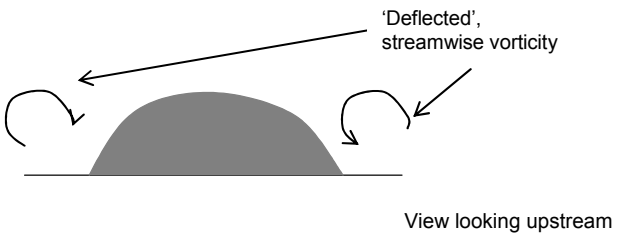
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(a) Vane class SBVG







(b) 'Obstacle' class SBVG

Figure 11. Two classes of SBVG.

outer part of the boundary layer and the low energy regions near the walls. Since SBVGs are smaller than conventional VGs, it follows that their ability to entrain the flow into the boundary layer is unlikely to be as effective as that of VGs. On the other hand, they exert an influence on the flow in the region where it matters (i.e. within the boundary layer) and can be expected to have lower drag. It is important, therefore, to understand the way SBVGs affect the flow downstream of them while knowing their drag.

Flows downstream of various types of SBVGs were first investigated by Schubauer and Spangenberg[31] and more recently by Ashill *et al*[32,33] and Yao and Lin[34]. The SBVGs that have been studied in the wind tunnel in recent investigations[32,33] are illustrated in Table 1. As noted before, two main classes may be identified: the vane type and the 'obstacle' type. In the first type the streamwise vortices are joined to the bound vortices on the wing-like elements of the SBVG, as shown in Fig. 11(a). This class is basically similar to the conventional VG.

The second class, the 'obstacle' type converts boundary layer vorticity, which is normal to the axis of the main flow, into streamwise vorticity. The flow associated with the streamwise vorticity has been referred to as a 'secondary' flow[35]. An example is shown schematically in Fig. 11(b).

SBVGs that belong to the former class can easily be identified in Table 1†, while deciding which are members of the 'obstacle' class is perhaps less straightforward. The 'dowel' belongs to the obstacle class, while the wedge probably generates flows that would put the device in either class.

The ultimate aim of flow studies is to determine the relative effectiveness of these devices. 'Effectiveness' could be measured by the extent of the improvement in the performance of the vehicle; however, this is not particularly helpful in assessing devices because the answer depends on the application. For a given device drag more general measures of effectiveness are:

i) Strength of the vortices and the way the vortices decay downstream.

ii) The general level of decrease in boundary-layer shape factor (or fullness of the velocity profile) at a station close to where control is required (e.g. separation line).

The first of these is the more fundamental of the two but it is indirect in that it does not provide a measure of the effect of the device on the boundary layer (although this might perhaps be inferred using a CFD method).

The second measure relies on the idea that the devices have some sort of quasi-uniform effect on the boundary layer so that, downstream of the devices, the boundary layer behaves as if it had been in a more favourable streamwise pressure gradient further upstream. There is some evidence to support this concept from the early studies by Schubauer and Spangenberg[31], in which closely-spaced arrays of devices were placed some way upstream of where their effect on the boundary layer was measured. This conditioning of the boundary layer can only take place some distance downstream of the devices, probably of the order of ten boundary-layer thicknesses.

It seems likely that other flow mechanisms occur, particularly when the devices are close to the flow feature that needs to be controlled and/or the spacing between the devices is large. In these situations a quasi-uniform effect is unlikely and mechanisms of a three-dimensional nature occur that may be effective in controlling the flow.

In the remainder of Section 3.2.1a) consideration is given to efforts to understand the flow downstream of SBVGs, the way they affect the boundary layer and their contribution to drag. The behaviour of SBVGs when placed in an array is also briefly discussed.

† All SBVGs probably generate secondary flows near the wall but, for the vane type, these flows are of secondary importance to the main vortex flow.

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3.2.1.1 Vortex strength and decay

Parameters defining the vortices are a) the strength of the vortices just downstream of the device, b) the streamwise decay of vortex strength and c) the vortex trajectories. These parameters are of interest because the initial vortex strength determines the mixing effectiveness of the vortices just downstream of the device, while the decay and trajectories determine the streamwise extent of the effectiveness of the devices.

Additional parameters of interest are the maximum value of the streamwise vorticity in the vortices and the lateral dimensions of the vortex cores.

The strength of the vortices (or the circulation around them) at a particular streamwise station, x–x_t, downstream of the device clearly depend on the type of device. Dimensional analysis suggests that they can be related to other physical quantities by the expression

$$\frac{\Gamma(x)}{u_\tau h} = G\left(h^+, \frac{x - x_t}{h}\right) \quad \ldots (3)$$

Here, in incompressible flow,

$$u_\tau = U_e\sqrt{c_f/2} \quad \ldots (4)$$

is the wall friction velocity in undisturbed or uncontrolled flow, with U_e the local stream speed and c_f the local skin-friction coefficient. Suffix t refers to the trailing edge of the device. As well

$$h^+ = \frac{hu_\tau}{\nu} \quad \ldots (5)$$

where h is device height.

A correlation based on Equation (3) for the circulation at five device heights downstream of the various devices illustrated in Table 1, Γ_5, is shown in Fig. 12, taken from Ref. 32. Further details of the contours used in defining vortex strength may be found in Refs 32 and 33†. The figure shows that the non-dimensional circulation varies only slightly with h^+ for values of h^+ between 750 and 2,800. Measurements at lower values of h^+ are only available for the Forwards Wedge and these suggest that the non-dimensional circulation decreases as h^+ approaches zero.

The lines shown in Fig. 12 are derived from the correlation shown in Fig. 13, also given in Ref. 32. This correlation was obtained by noting from Fig. 12 that the maximum value of the non-dimensional circulation reached at high Reynolds number depends on the type of device, the largest value being achieved with the Joined Vanes (i.e. vanes without a space between them or n = 0 Table 1). While not surprising, this dependence on device geometry does suggest a framework for generalising the correlation by using the concept of a device effective height. For this purpose the effective height of the Forwards Wedge is taken to be equal to its actual height. The effective heights of the other devices are then selected by ensuring that the maximum value of non-dimensional

† Some care needs to be taken when determining the circulation around the vortex not to include the counter vorticity generated at the wall. This is done in the study described in Ref. 33 by determining and taking the maximum value of the circulation around a contour surrounding the vortex. For devices inducing counter-rotating vortices the value of circulation shown is the arithmetic mean of the absolute values of the circulations of the two vortices.

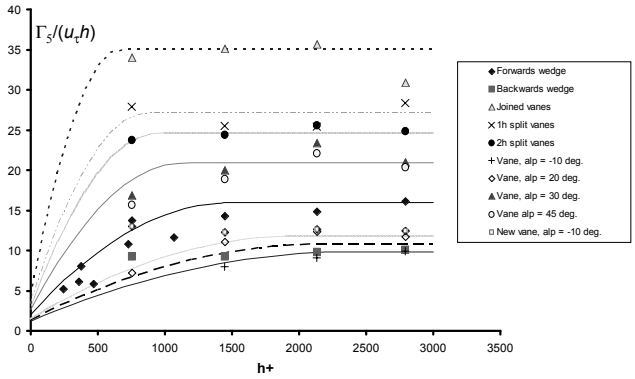


Figure 12. Correlation of vortex strength five devices downstream of various SBVGs against Reynolds number h^+.

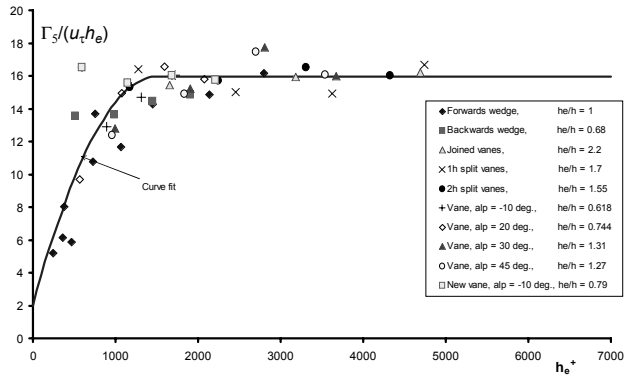


Figure 13. Generalised correlation of vortex strength against Reynolds number based on device 'effective height'.

circulation based on effective height is independent of geometry. The result of doing this is shown in Fig. 13, where the non-dimensional circulation based on effective height is plotted against the non-dimensional effective height.

While not entirely satisfactory, this correlation may be useful in determining the sensitivity of vortex strength to Reynolds number for particular devices.

An essential feature of the vortices induced by SBVGs is that they decay with distance downstream under the action of wall shear stress and mutual interference between vortices if they are sufficiently close together.

It has been suggested in Ref. 25 that the streamwise rate of change of vortex strength or circulation of a single vortex downstream of a VG (or SBVG) is proportional to the lateral (inviscid) velocity induced at the wall by the vortex and local skin-friction coefficient. Assuming that the flow in lateral planes is two-dimensional, this leads to the simple equation:

$$\frac{\mathrm{d}\Gamma}{\mathrm{d}x} = -Kc_f\frac{\Gamma}{h_v}, \quad \ldots (6)$$

where K is a constant, c_f is local skin friction coefficient in the uncontrolled flow† and h_v is the height of the vortex above the wall. It was further suggested in Ref. 25 that this height may be assumed

† Skin friction coefficient is included in recognition of the viscous shearing effect of the wall.

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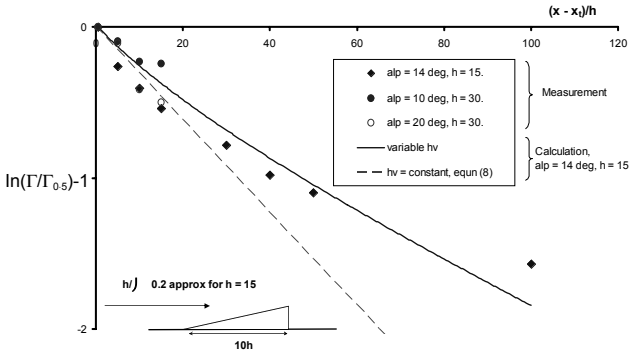
to vary so slowly with streamwise distance that it may be taken to be constant. Thus, with this assumption, the differential Equation (6) may be integrated explicitly to give

$$\Gamma / \Gamma_t = \exp\left(-Kc_f \left(x - x_t\right)/ h_v\right) \quad \ldots (7)$$

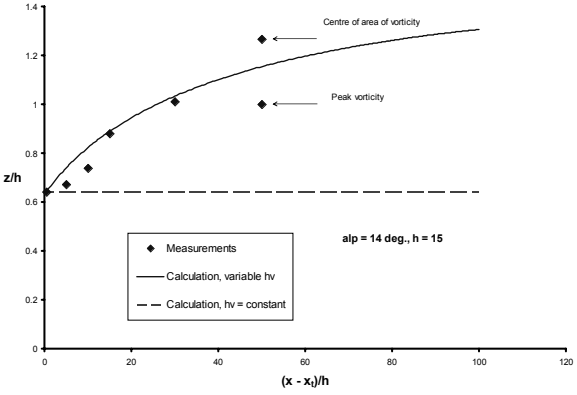
or

$$\ln\left(\Gamma/\Gamma_t\right) = -Kc_f \left(x - x_t\right)/ h_v \quad \ldots (8)$$

Supposing that the vortex strength at the trailing edge of the vane may be taken as equal to that at 0·5h further downstream, we may plot the logarithm of vortex strength against non-dimensional streamwise position. This is done for single ('New' or L = 10h) vanes (Table 1), as shown in Fig. 14(a) for two device heights (h = 15 and 30) and various deflection angles (alpha). This figure shows that Equation (8), with h_v equal to its value just downstream of the trailing edge of the vane, does not accurately reflect the decay of the vortex for the h = 15 device. Fig. 14(b) shows that the reason for this is that, while in the simple theory vortex height, z, is taken to be constant, it actually increases with streamwise distance.



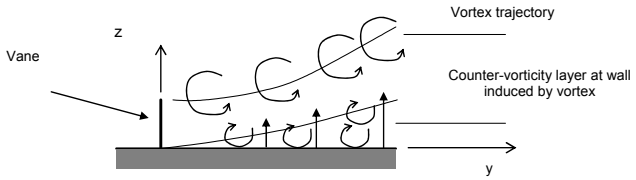
(a) Vortex decay for two vane heights



(b) Non-dimensional height of vortex against streamwise distance downstream of device.

Figure 14. Vortex decay and height of vortex above the wall for a single vane, U_e = 30.

The most reasonable explanation for this change in vortex height is that the vortex induces a vorticity layer at the wall (which is of opposite sign to that of the vortex) as a consequence of the no-slip condition at the wall. This layer induces a vertical velocity that displaces the vortex away from the wall. This is illustrated in the sketch of the y-z or cross-flow plane below.

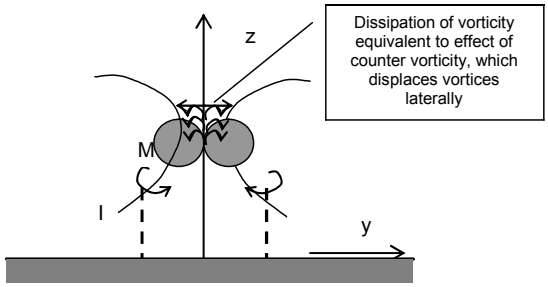


Any model of a vortex induced by a vane therefore needs to take this into account, and calculations by a method that does this (purely on an empirical basis) are shown in Fig. 14(a) to give improved agreement with measurement as regards vortex decay. The corresponding vortex height used in the calculations is shown in Fig. 14(b).

In their study of an SBVG vane of rectangular planform, Yao and Lin[34] confirmed that vortex height increases with streamwise distance, and they also associated this with the interaction between the vortex and the wall. They also found that the trends of vortex strength and height with streamwise distance are reasonably well predicted by the NASA OVERFLOW CFD code. CFD codes are likely to play an important part in the optimisation of the design of SBVGs. However, one of the problems that has been found to occur is that the need to have adequate grid density in the region of the device means that the grid is rather coarse further downstream, as found in studies by Press[36]. Hence CFD methods may not be able to predict accurately the behaviour of the vortices downstream of the device. Problems such as this have led to the suggestion that SBVGs might be modelled using a vorticity injection method[37]. In this approach the geometrical details of the SBVG are ignored, and empirical methods are used to prescribe the strength of the vortices and their position at a station downstream of the device.

Yao and Lin[34] compared results for a SBVG vane with a VG (h = δ) of the same planform, and found broadly similar characteristics in the two cases as regards vortex decay and trajectory.

The situation is more complicated for counter-rotating vortices, as in the case of the Spaced Vanes (Table 1). To the shearing effect of the wall must be added the effect of the neighbouring vortex on the flow within the vortex. The physics of the flow over split vanes are illustrated by reference to the sketch below, showing schematically the loci of the vortices in the y-z plane.



The shearing effect of the wall is reduced by the presence of the opposing vortex. Furthermore, between the points I and M, the vortices rise rapidly away from the wall due to the mutual effect of the vortices, thus reducing the wall shearing effect. Depending on the gap between the vanes and the vortices, the vortices overlap above the point M causing dissipation of the vorticity. As well as reducing the strength of the vortices, this has the effect of displacing the vortices laterally away from one another. This tendency, which is analogous to

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the effect of wall counter vorticity, seems to have been first observed by Wendt and his colleagues[38, 39], and these effects have been observed in the studies described in Ref. 33.

Results for vortex decay of other types of SBVG can be found in Refs 32 and 33. These references show that the Joined Vanes give higher vortex strength just downstream of the device than Spaced Vanes (as indicated in Fig. 12). However, the larger streamwise decay of the vortices of the Joined Vanes compared with that for the vortices of Spaced Vanes means that by about 50*h* downstream of the device, the vortex strengths for the joined and spaced vanes are similar.

Ref. 3, in comparing the advantages and disadvantages of co-rotating and counter rotating VGs, mentions that the vortices of the latter type of VG rise up out of the boundary layer. Once this occurs they are less effective in entraining air into the boundary layer. With SBVGs, on the other hand, the vortices of counter-rotating devices can remain in the boundary layer over a significant streamwise interval, depending on the ratio h/δ. Hence counter-rotating SBVGs may possibly be more effective in sustaining their ability to entrain air into the boundary layer than their VG counterparts.

The effect of adverse pressure gradient on the vortex characteristics of single and double vanes is discussed in Ref. 33. The main conclusions are as follows:

i) The effect on vortex strength just downstream of the device for single and spaced vanes can largely be explained by the change in wall friction velocity. However, when the vanes are joined (n = 0, Table 1) there is an additional adverse effect probably caused by the vanes separating the boundary layer upstream of the device.

ii) Adverse pressure gradient promotes the growth of the vortices, causing them to merge further upstream than in zero pressure gradient for counter-rotating devices. Increasing the spacing between the vanes reduces this adverse tendency.

In the light of the results obtained in low-speed studies of the flow downstream of SBVGs, it may be remarked that vortex strength just downstream of the device is directly proportional to the product $u_\tau h$ for a given set of ambient conditions (see Equation (1)). This implies that an SBVG placed in a high-shear region need only to have a small height compared with an SBVG of similar geometry in a low shear region to achieve the same vortex strength. In determining where to place SBVGs, consideration needs to be given to whether or not the device is either too far upstream of or too near the region where flow control is required. This will depend on the form of control required, i.e. whether it is required to 'condition' the boundary layer or to create three-dimensional flows to break up regions of separation (see below).

3.2.1.2 *Effect on boundary-layer shape factor*

An illustration of the way the vortices induced by counter-rotating SBVGs change the effect on the boundary layer shape factor with streamwise distance is shown in Fig. 15. The data are for 1*h* vanes (h^+ = 2,130). The figure shows the increment in boundary-layer shape factor ΔH plotted against non-dimensional spanwise position (y/h). Also shown are the non-dimensional heights of the vortex cores, z/h, as well as insets showing plots of cross-flow streamlines and colour plots

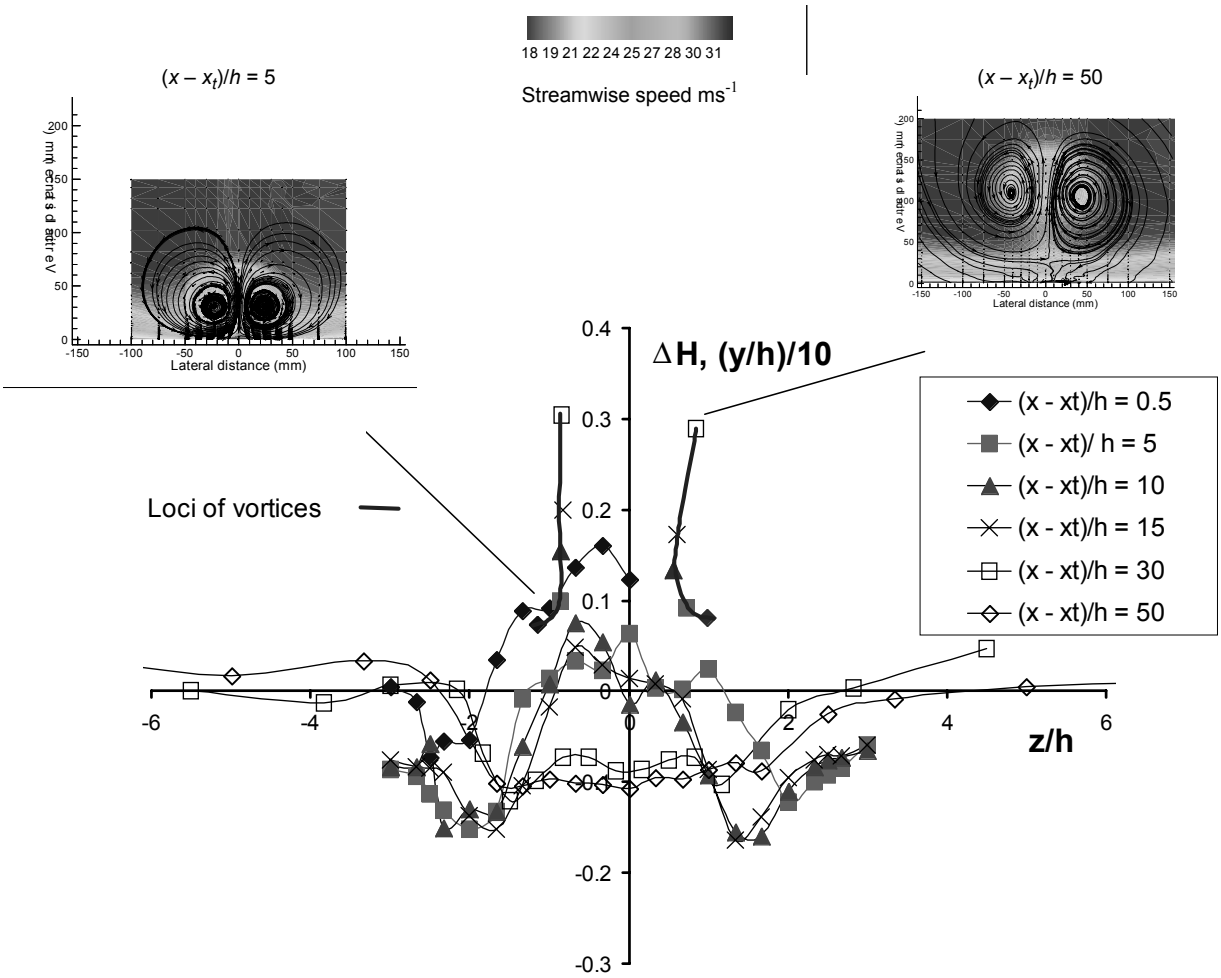


Figure 15. Spanwise variation of increment in boundary-layer shape factor and vortex height for 1*h* spaced counter-rotating vanes, U_e = 30, h = 30, $h/\delta \approx 0{\cdot}4$.

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of streamwise velocity for $x - x_t = 5h$ and $50h$. The height of the device is 40% of the undisturbed boundary-layer thickness i.e. $h/\delta \approx 0{\cdot}4$.

Just downstream of the vanes, at $x - x_t = 0{\cdot}5h$, where the vortex height is similar to vane height, the vanes cause an increase in the shape factor on the plane of symmetry. This may be explained by the vortices sweeping low energy air from either side of them into the boundary layer on the plane of symmetry.

Further downstream, as the vortices rise up under mutual induction, the shape factor in the region between the vortices decreases as the vortices pull in higher energy air and the low energy air is swept out into the main flow.

Figure 15 shows that at $x - x_t = 30h$ and $50h$ (12δ and 20δ, respectively), the increment in shape factor is fairly uniform over the interval $-2 < z/h < 2$, suggesting that a quasi-uniform effect of the SBVG on the boundary layer can be envisaged in this region. However, further upstream for $x - x_t < 15h$, say, the increment in shape factor is much less uniform with spanwise distance. In this region the controlling effect is likely to be much more three-dimensional.

The effect of device height on the spanwise variation of shape factor is shown in Fig. 16 for $x - x_t = 50h$, indicating that at the lower of the two values of h^+ or h/δ, the spanwise distribution of shape factor increment is less uniform than at the higher value. This is perhaps to be expected since, as hinted before, the conditioning effect on the boundary layer is likely to depend on boundary-layer thickness rather than device height. Hence the larger of the two devices will condition the boundary layer more effectively for a given non-dimensional streamwise distance $(x - x_t)/h$ than the small device.

Nothing can be said at this stage as to which of the two mechanisms (three-dimensional or conditioning effects) is the more effective for SBVGs. However, the need to provide control over a range of flow conditions probably means that the conditioning mechanism is preferable in order to provide a sufficient range of effectiveness for the device. To illustrate the point, if devices were placed close to a shock wave so as to create a 3D effect, the devices would cease to be effective if the shock wave moved upstream of the device. This might happen, for example, as a result of a small reduction in either Mach number or angle of incidence.

The conditioning mechanism is similar to that already described for VGs[25], the main difference between SBVGs and VGs being that wall shear stress plays a more important role in determining vortex strength and decay for the former than it does for the latter.

3.2.1.3 *Device drag*

Lin *et al*[40] and Lin[41] have investigated the drag of SBVGs, showing that for $h/\delta > 0{\cdot}2$, device drag increases with device height without any significant increase in effectiveness in terms of reducing the streamwise extent of flow separation. This corresponds approximately to $h^+ > 1{,}000$, above which vortex strength is largely independent of device height (see Fig. 12). Lin's measurements were made using a force balance. Other workers have used boundary-layer traverses, together with the momentum equation[31,33], to determine drag. The latter method has the advantage over the balance method in that it includes all the effect of the device on the skin friction of the wall, not just on the plate under which the balance is mounted. On the other hand, the wake survey technique gives relatively large errors, since it involves the subtraction of two large numbers, owing to the way the device affects the spanwise distribution of boundary-layer momentum thickness.

Comparisons between measurements using the wake survey technique and predictions by the CFD method of Press[36] for Joined vanes, $1h$ vanes and the Forward Wedge were found to be in moderately good agreement[33].

Recent measurements of device drag using a skin-friction balance have been made at Cranfield University[42]. These were performed using vane devices of similar geometries to those studied previously[32, 33], and the experiments were performed in nominally zero streamwise pressure-gradient. The measurements reveal that increasing the spacing

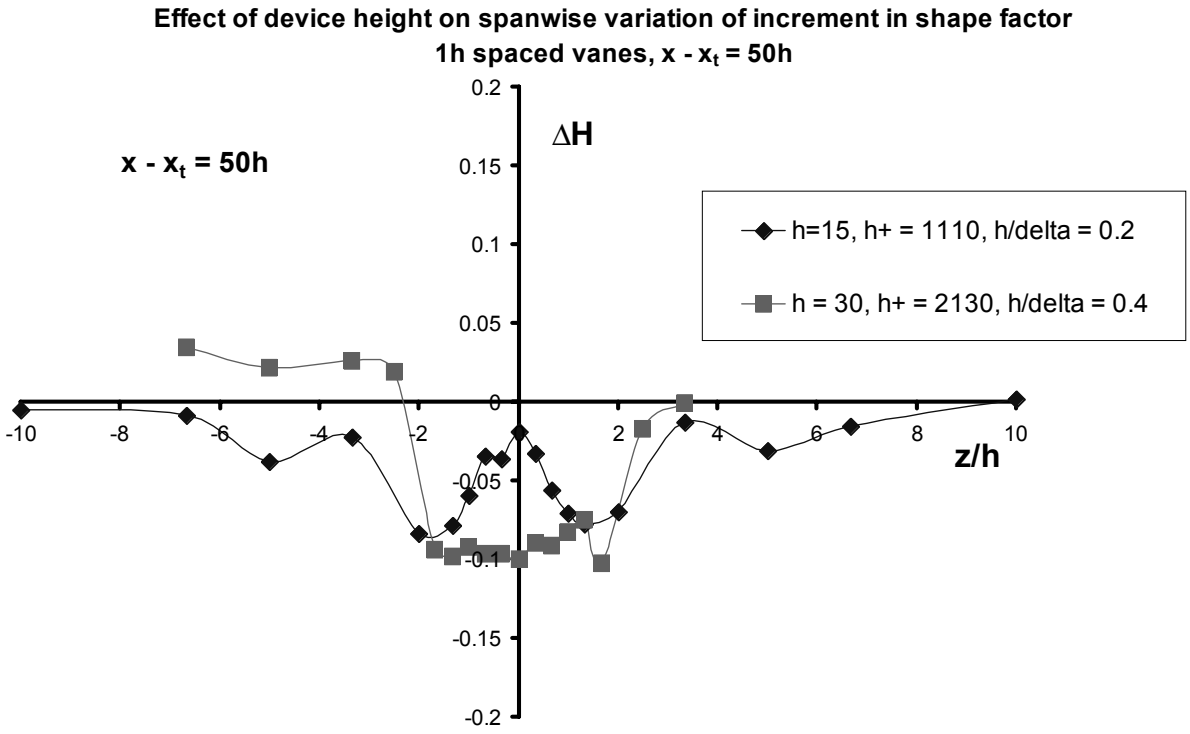


Figure 16. Effect of device height Reynolds number on increment in boundary layer shape factor.

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between counter-rotating vanes (of the geometry shown in Table 1) from zero to 1*h* and then to 3*h* reduces drag by about 20% and 50%, respectively. The Cranfield University tests also show that the drag of the 3*h* Spaced Vanes is less than twice that of a single vane at the same deflection or toe angle, showing that there is a favourable influence between the two vanes in the counter-rotating vane.

Comparisons between the skin-friction balance measurements made at Cranfield and those referred to in Ref. 33 on the same configurations using the wake-survey technique are difficult because of differences between the values of h/δ in the two sets of tests. However, in terms of relative magnitude of drag, the two sets of results are consistent, bearing in mind the lack of accuracy of the wake survey technique.

The Cranfield study was also concerned with the effect of yaw on device drag: it measured the drag of two co-rotating vanes of 4*h* spacing and studied the effect of toe angle on the drag of co-rotating vanes.

3.2.1.4 Mulitiple devices

The discussion so far has been concerned with single devices. As far as the authors are aware, no experimental work has been published on the effect of lateral spacing on the flow downstream of SBVGs. However, Figures such as 15 and 16 suggest that a spacing of between 4*h* and 12*h* is likely to lead to satisfactory results for counter-rotating vanes with a 1*h* gap between them. Lin[41] investigated devices with a spacing of 9*h*.

Using a CFD method, Wik[43], calculated the spanwise variation of the increment in boundary–layer shape factor downstream of co-rotating vane SBVGs. He found that the overall reduction in shape factor was greater with 10*h* spacing than with 4*h*.

b) Swept wings

Pearcey[3] pointed out that co-rotating VGs have an additional role for swept wings, namely to control the outflow in the boundary layer. Co-rotating SBVGs may be expected to have similar tendencies to those mentioned by Pearcey. Indeed, they could be more effective than VGs in this respect, since the outflow is a maximum deep within the boundary layer, typically at about 0·1, i.e. about half height for a typical SBVG.

It will be recalled from Section 2 that inserting vorticity of the appropriate sign and direction can control 'ordinary' separations. Ref. 44 considered the strength required for a single streamwise vortex to ensure that a 3D or 'ordinary' separation is moved downstream. It concluded that SBVGs with $h/\delta > 0{\cdot}2$ should be able to provide vortices of sufficient strength for this purpose. They are likely to be most effective in regions of high local velocity such as near the leading edge.

The outflow increases as ordinary separation is approached. Thus SBVGs should not be placed so far upstream that the vortices have decayed by a substantial amount in the region of the flow separation where the control is required. In determining this position, account needs to be taken of the increased lateral shear (due to sweep) that will tend to increase the dissipation of the vortices[25].

c) Effects of compressibility

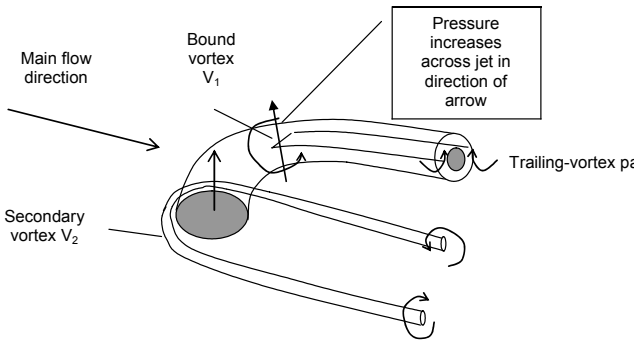
Referring to results from wind tunnel tests on a vane VG ($h/\delta = 0{\cdot}75$) of low aspect ratio (0·5) Bray[45] pointed out that vortex strength just downstream of a vane device is directly proportional to Mach number. Based on this result, Bray was able to justify the observation that vortex trajectory is insensitive to Mach number. He also found that increasing Mach number enhances the vortex decay slightly for Mach numbers up to 0·75. These results are likely to apply to SBVGs. However, there is a need for further studies in this area, extending the ranges of non-dimensional heights and Mach numbers above the maximum, 0·75, Bray investigated. Of special interest is the behaviour of SBVGs in supersonic main stream flow, since this is the type of flow in which SBVGs will be placed when upstream of shocks on transonic aerofoils. The effect of SBVG planform on the effect of the device on the boundary layer is of particular interest in this connection.

3.2.2 Air jet vortex generators

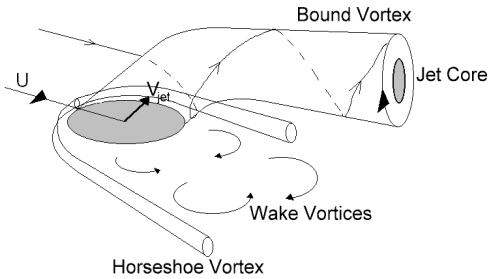
a) 2D flows

Beginning with the work of Wallis[46] in the 1950's and following that of Pearcey[3], Air-Jet Vortex Generators (AJVGs) have been subjected to extensive study, as reviewed by Bray[45] and Lewington[47]. The attraction of this type of device is that AJVGs need only be operated when required, and so this type of device is unlikely to be a significant source of parasitic-drag. Of course, when they are operated, some source of pressurised air is needed but the typical mass flow rates are well below those required for area blowing.

Two sketches illustrating the principle are shown below. The first, sketch i), shows a round jet emerging normal to the wall. Owing to the curvature of the jet, the pressure increases in the direction of the blocked arrow. Hence, since pressure is continuous at the edges of the jet, the pressure must rise between the two sides of the jet. From Bernoulli's equation this implies a decrease in speed between the lower and upper sides of the jet, which further implies circulation in the counter-clockwise sense with the main flow taken from left to right. This suggests representing each axial element of the jet by a bound vortex (V_1), i.e. a vortex with its axis normal to that of the jet†. As the curvature of the jet decreases with downstream distance, the vortex strength decreases. Each of these elementary bound vortices is accompanied by a pair of trailing vortices as shown in the sketch. As well as the bound vortex V_1, there is a secondary vortex V_2, and its accompanying trailing vortices, resulting from the jet inducing a three-dimensional separation of the boundary layer just upstream of itself. These trailing vortices are of opposite sense to the trailing vortex pair associated with the vortex V_1.



i) Jet normal to surface



ii) Jet at angle to surface and rotated (after Bray[48])

The second sketch, Bray unpublished[48], illustrates the flow when the jet is pitched and rotated. This flow is more complicated than for the jet normal to the surface, as discussed by Bray[45]. The jet

† Strictly, the strength of the bound vortex varies across the span of the jet, and hence there is a trailing vortex sheet which probably rolls up to form two trailing vortices.

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deflects the boundary-layer streamlines, with the result that vorticity within the boundary layer is converted to essentially streamwise vorticity. The result is trailing vorticity at the edges of the jet giving a strong trailing vortex. Thus the jet behaves in the same way as an obstacle type SBVG. This effect of pitching and rotating the jet appears to have first been observed by Wallis[46]. It follows that AJVGs are likely to be most effective when placed in regions of high local wall shear or vorticity.

In addition to converting boundary-layer vorticity into streamwise, AJVGs increase the streamwise momentum of the flow, thereby providing another source of boundary-layer control.

The general conclusions from various investigations reviewed and summarised in Refs. 45 and 47 are, first, that the longitudinal streamwise vortices in the downstream direction are sufficiently strong to control turbulent separated flow. Secondly, the effectiveness of the device is improved with the jet pitched at an angle of 30° or lower and a skew angle of no less than 60°. This produces an increase in skin friction, which implies a thinning of the boundary layer and hence a delay in the onset of separation. Thirdly, it appears that increasing the relative jet velocity increases the strength of the resulting vortices. Although this does enhance the desired effects, placing the AJVGs in a counter-rotating array also results in the vortices moving away from the surface more rapidly. Finally, AJVG performance generally decreases with increasing distance upstream of the separation line. The ability to control flow separation remains effective for AJVGs located as far as 40δ. upstream of the uncontrolled separation line.

According to Lewington[47], Freestone[49] has studied AJVGs of non-circular cross-section, and he showed that rectangular cross section devices give higher trailing vorticity than either round-hole devices or vane VGs. Lewington also reported that Compton and Johnston[50] concluded that the peak vorticity and vortex strength of AJVGs decay much less rapidly than with VGs. This is to be expected since the jet energises the core of the vortex. However, it seems likely that the streamwise decay of vortex strength may be represented by an equation similar to that for VGs or SBVGs, i.e. Equation (6).

Detailed studies of the flow downstream of AJVGs have been reported by Zhang[51].

Dynamic similarity suggests that the parameters of interest for AJVGs are:

- jet pitch angle and rotation or yaw angle,
- jet momentum coefficient

$$C_\mu = \frac{M_j}{\frac{1}{2}\rho_e U_e^2 S}$$

where S is a reference area,

- jet pressure ratio, jet velocity ratio or jet Mach number ratio and
- hole 'Reynolds number'

$$d^+ = \frac{du_\tau}{\nu}$$

or the ratio of jet diameter to boundary-layer thickness, d/δ. For rectangular air jets the jet diameter will be replaced by some other hole dimension.

It is reasonable to suggest that for a given AJVG geometry, vortex strength of AJVGs may be described by an equation of the form

$$\frac{\Gamma}{u_\tau d} = J\left(d^+, C_\mu, \frac{x - x_H}{d}\right) \quad \ldots (9)$$

where suffix H refers to the hole where the jet emerges into the flow. As far as the authors are aware, no attempts have been made to study the validity of correlations of this form for AJVGs.

Recognising that jet diameter needs to be very small in regions of high shear, such as a leading edge of a wing (see Equation (9)), Warsop[52] prescribed the use of Microfabricated-Electro-Mechanical Systems or MEMS technology to develop pulsating AJVGs on a micro-millimetric scale. Laboratory experiments have shown that such AJVGs provide a reduction in boundary–layer shape factor in the region of 10%.

b) Swept wing flows

No work has come to the authors' attention of studies of a basic nature to understand the effect of boundary-layer cross flow on AJVG vortices or the way AJVGs affect boundary layers on swept wings.

c) Effects of compressibility

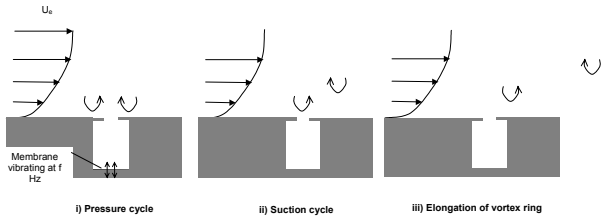
Bray[45] found that the effects of compressibility on vortex strength and trajectory are similar to those of a vane VG. Again, this is a field that needs to be explored further, notably to higher Mach numbers than those studied by Bray, including within the supersonic range.

3.2.3 Mass-less or synthetic jets†

The basic geometry of the mass-less or synthetic jet is illustrated in Fig. 17. A diaphragm at the base of a chamber is vibrated harmonically. Hence air is expelled from the nozzle in sympathy to form vortices that rise under self-induction.

Mass-less jets are of interest because they do not require the transport of pressurised air to wherever control is needed. Furthermore, the power requirements are expected to be small, and, as with AJVGs, they need only be switched on when needed.

The process involved in the control effect of a mass-less jet in a boundary layer is illustrated in the sketch below, showing how the vortices distort under the action of the onset flow. This interpretation by Wood and his colleagues[53-55] at The University of Manchester is based on their water-tunnel studies of synthetic jets.



In the first stage, the pressure cycle, a ring vortex is produced above the orifice. At the second stage, the suction cycle, the ring vortex moves downstream under the action of the main flow and the leeward part of the vortex is attracted towards the orifice, tilting the vortex ring. Thus the downstream end of the vortex is in a higher velocity flow in the boundary layer than the leeward part, with the consequence that the vortex ring becomes elongated. This elongation is an important feature because it extends the region of control in the streamwise direction. According to Wood *et al*[53], subsequent cycles create trailing vortices that combine to enhance the effect.

With reference to the symbols of Fig.17, dynamic similarity suggests that the following parameters are important in the production of the trailing vortices: jet velocity ratio, d/D, H/D, tc/D, $d^+ = du_\tau/\nu$ or d/δ (as with AJVGs) and $S_T = fd/U_e$.

The latter three parameters need to be borne in mind when

† The authors have been informed by Dr David MacManus of Cranfield University that a comprehensive review of basic work on this subject up to 2002 is given by A. Glezer and M. Amity in Annual Review of Fluid Mechanics, 2002, 34:503 – 29.

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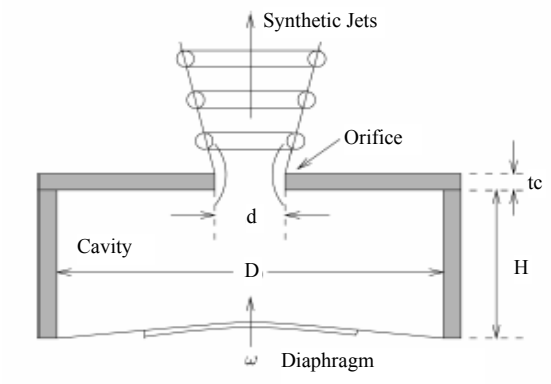


Figure 17. Mass-less or synthetic jet, taken from Ref. 54.

considering using laboratory experiments as a guide to what might be expected at 'full scale'.

According to Crooke and Wood[55] it is important to pay close attention to the geometry of the mass-less jet, since failure to do so could lead to incoherent structures being ejected into the surrounding flow. In addition, they remarked that operation at too high a Strouhal number, S_T, or at too high a jet velocity ratio will also have this effect.

Recently, Liddle and Wood[56] have studied the effect on the flow of having two mass-less jets at a fixed distance apart but at different angular orientations. The effects of phase difference between the excitation frequencies of the two jets were studied and shown to have a significant effect on the flow induced by the devices.

3.2.4 Devices simulating bird feathers

Observations of the behaviour of bird feathers near the trailing edges of their wings as they approached stall suggested a novel form of boundary-layer control[57,58]. It was noted that the feathers pop up when the bird is at high angles of attack, and it was thought that this was a way the bird unconsciously controls the forward movement of separation on the upper surface of the wing. To simulate this effect, wind tunnel tests at low speed were made with an aerofoil with an upper-surface split flap, as illustrated in the sketch below.

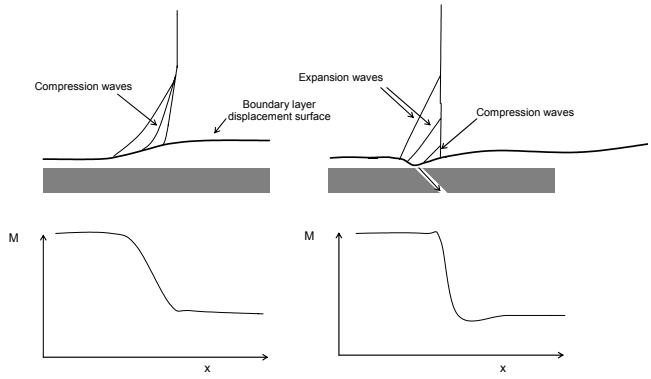


The device is autonomous and is restrained by a flexible link to limit the movement of the flap. The flap responds when flow separation reduces the pressure above it.

It seems likely that the split flap fixes the separation at the join between the flap and the wing, thus preventing it from moving further upstream and stalling the flow over the whole of the upper surface.

3.2.5 Slot suction

The benefits of slot suction have been known for some time but some interest in the idea has been rekindled in recent years as a way to control shock/boundary layer interactions. From this point of view the main benefit is in improving the 'health' of the boundary layer upstream of the shock. The sketch below illustrates possible effects of suction applied just upstream a normal shock/boundary layer interaction on the main flow and streamwise Mach number distributions inferred from surface pressures.



Without a slot, illustrated on the left, the boundary–layer displacement surface grows through the shock as a result of the upstream influence of the shock. This growth of displacement thickness induces compression waves that provide a form of natural flow control, which tends to reduce shock strength. When suction is applied just upstream of the shock, there is some reduction in displacement thickness or in the growth of displacement thickness. This boundary-layer growth causes expansion waves that tend to strengthen the shock in the field. Furthermore, since the length of the region over which there are compression waves is reduced, the shock wave becomes stronger. One possible adverse effect of suction is likely to be increased wave drag that offsets any reduction in boundary-layer drag.

Important parameters in determining the effect of suction on the interaction are: a) distance of the slot upstream of the shock made non dimensional by undisturbed displacement thickness, δ^*, and b) the ratio of suction mass flow per unit span to boundary–layer displacement flux ($\rho_e U_e \delta^*$).

4.0 APPLICATIONS AND EFFECTS ON WING AND BODY FLOWS

4.1 Control of shock waves on aerofoils and wings

4.1.1 Inviscid control devices

Devices for controlling shock waves appear to be particularly attractive for aerofoils or wings on which the shock waves have a tendency to increase in strength rapidly with changes in flight conditions. A good example is the natural laminar flow aerofoil, which has an upper surface pressure distribution with a favourable pressure gradient upsstream of the shock in order to preserve laminar flow. In this supersonic flow region there is a tendency for the expansion waves generated at the surface to become stronger with increase in either Mach number or angle of incidence. Thus, with laminar aerofoils, there is a rapid increase in shock strength with increase in Mach number or lift. The shock wave moves position slowly with change in flight conditions, making this type of aerofoil suitable for shock control.

Another example of a wing section suited to shock control is a thick aerofoil designed for low subsonic flight. This type of aerofoil will be severely limited in maximum speed by the rapid increase in

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shock strength with Mach number, making this type of section suitable for shock control.

On a 'turbulent' aerofoil the expansion waves on the upper surface are countered by compression waves formed by the reflection from the sonic line of expansion waves from near the leading edge[3]. These waves weaken the expansion waves further downstream, with the result that the shock wave is weakened. On the other hand, owing to the subtle balance between expansion and compression waves at the design condition, the shock wave tends to move streamwise considerably with changes in Mach number or lift coefficient. In some cases, two shocks may form on the upper surface. This class of aerofoil is therefore less amenable to shock control than the laminar aerofoil.

A number of inviscid methods have been considered for controlling shock waves on aircraft wings. These include

- Perforated surface passive control[59-61].
- Slotted surface passive control (spanwise slots[59,60] and streamwise slots and grooves[12,16-19]).
- Bumps (two-dimensional[5,12,13,62-65] and three-dimensional[12, 19]).

The first two types involve the passive transfer of air from the region downstream of the shock wave to the relatively low-pressure region upstream of the shock wave. Contrary to what was mentioned in Section 3.2, initial studies[59,60] suggested that perforated control gave significant drag reductions even at low lift. However, more recent work[5,61,62] confirms a widely held view that the overall effect on drag of perforated passive control is adverse.

The effect of various types of control on a transonic aerofoil flow can be seen in Fig. 18. This figure shows drag polars taken from tests on the aerofoil model M2303 (with the Natural Laminar Flow (NLF) aerofoil section RAE 5243) in the 8ft × 8ft High Speed Tunnel formerly at QinetiQ, Bedford. Data for the perforated surface have been published in Refs 5 and 63; some data for the 'buffet bump' have appeared in Ref. 12 and data for the streamwise slots were published in Ref. 66. Tests were performed on various devices, including those shown in Fig. 18a and b, i.e.:

- a perforated surface of 8% open area between 50% and 60% chord;
- a surface with streamwise slots of rectangular planform, of open area 4% also between 50% and 60% chord;
- a 'buffet bump' (i.e. one placed just downstream of the shock) placed between 55% and 70% chord.

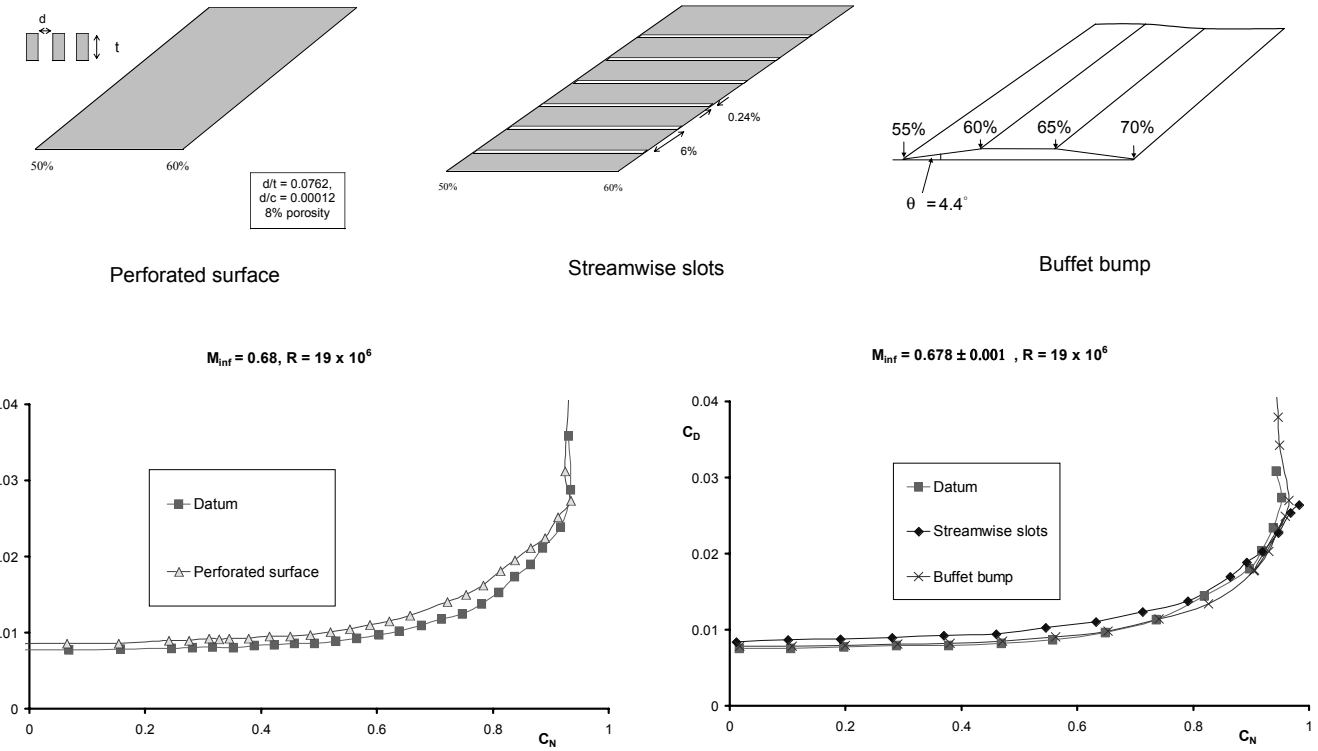
Both passive flow transfer devices had a common plenum as illustrated in Ref. 63.

Figure 18(a) shows that the perforated surface gives larger drag at a given lift than the datum aerofoil, the increment diminishing with lift at high lift as the favourable effect of the control on wave drag is felt.

There is some evidence that perforated surface shock control has a favourable effect on buffet by reducing the tendency for the shock to oscillate backwards and forwards[5].

Figure 18(b), illustrating data taken from the same model but in a different tunnel entry and at a slightly lower Mach number, shows that streamwise slots have a similar effect on drag as the perforated surface at low lift $C_N < 0{\cdot}8$. However, the slots give lower drag than the perforated surface at higher normal-force coefficients and a higher maximum lift than both the perforated surface and the basic aerofoil.

Further analysis of data taken from the tests on the M2303 aero-



a) Effect of perforated surface

b) Effect of streamwise slots and buffet bump

Figure 18. Comparison of drag polars for various shock control methods.

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foil with streamwise slots is presented in Ref. 66. This shows that the slots delay the onset of shock-induced separation and subsequent flow breakdown, as well as reducing considerably the movement of the shock wave with angle of incidence at constant free-stream Mach number. However, still to be resolved is the relative importance of the streamwise vorticity induced by the slots compared with their 'inviscid' effect on the shock wave in determining flow breakdown.

A basis for assessing and designing wing surfaces with slots and grooves has been described by Smith *et al*[67], who used a RANS CFD method to calculate the shock/boundary–layer interaction with a grooved surface.

As discussed in Section 3.1, two-dimensional bumps have a major advantage over perforated or slotted surfaces in that they either reduce or only increase boundary–layer drag slightly, at least as long as they are properly positioned in relation to the shock wave. This is illustrated in Fig. 18(b) by results obtained with the 'buffet bump' on the M2303 (RAE 5243) model in the 8ft × 8ft tunnel. This gives virtually the same drag as the datum at low lift but significantly lower drag at lift coefficients above about 0·75.

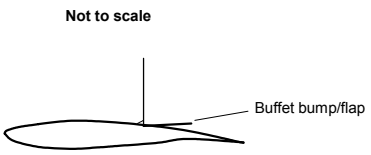
Thus, on a NLF aerofoil section such as RAE 5243, bumps could be used either in an active sense to improve performance off design, using variable geometry, or as a 'fix' if the performance of a design needed to be improved at high lift or at higher Mach number. The possibilities for variable geometry bumps were briefly referred to in Ref. 13 and explored in much more detail in Refs 5, 64 and 65.

Reference 63 showed that it is possible to predict accurately the aerodynamic performance of both bumps and perforated passive devices by using a variant of the BVGK code.

The need for variable geometry is more acute for turbulent-flow type aerofoils, since, as mentioned above, the chordwise position of the shock wave is more responsive to changes in free-stream conditions than for the laminar flow aerofoil. This is illustrated in Fig. 19 for the aerofoil RAE 5225, which is taken from Ref. 63. This shows the effect on drag of various simulated bumps of circular-arc cross section. For Mach numbers between about 0·72 and 0·735 and a lift coefficient of 0·75 (Fig. 19(a)) and at lift coefficients above about 0·7 for a Mach number of 0·725 (Fig. 19(b)) the effect is favourable but outside these ranges of conditions there is a pronounced adverse effect.

A variant of the 'buffet bump', intended only to control shock-induced separation, might take the form of a split flap with its leading edge just downstream of the shock as illustrated in the sketch below.

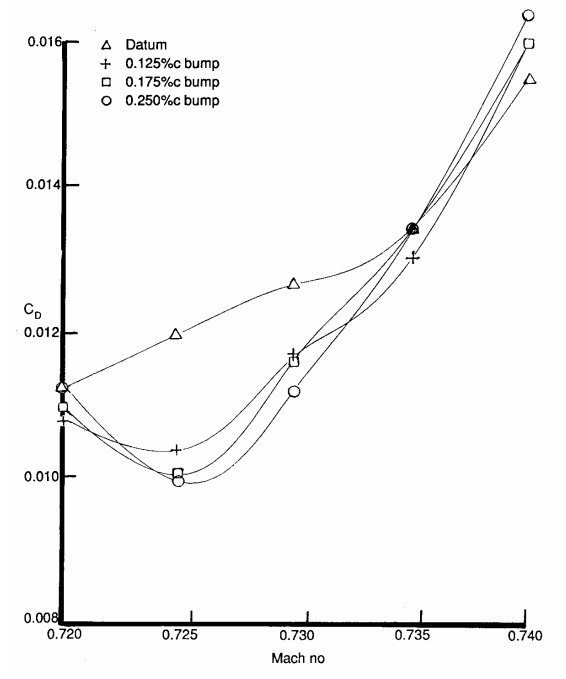
The flap would be deployed only when needed to delay shock-



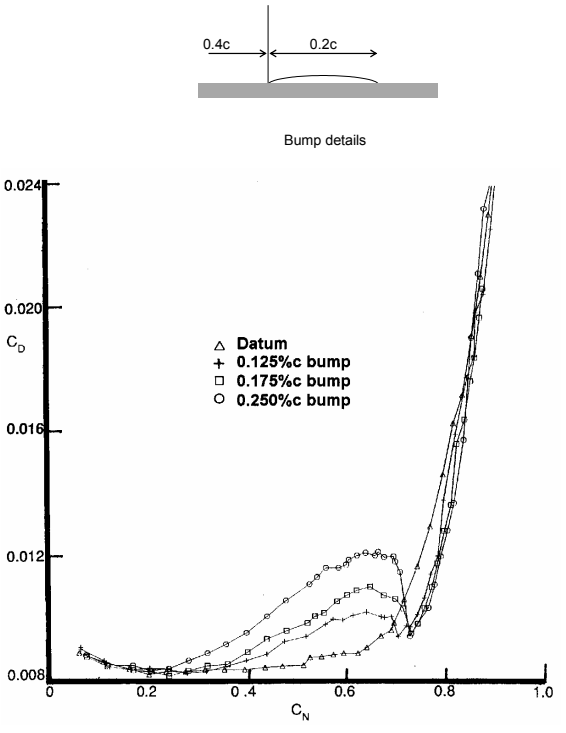
induced separation and hence extend the buffet boundary to higher lift coefficients and Mach numbers. This device might conceivably also have a role as a spoiler for low-speed lift control.

As with slots, three-dimensional bumps offer the possibility of reducing shock strength while introducing streamwise vorticity into the boundary layer downstream of the shock†. In addition, they are easily applied as a retrofit to wings or other external surfaces. To provide the deflection needed in the far field to reduce shock strength, these devices need to have a higher maximum ordinate than the corresponding 2D bump. Therefore they need to be carefully designed to ensure that they do not affect the boundary layer adversely (e.g. by provoking separation on their leeward surfaces or between them). Holden and Babinsky[19] studied the effect on shock

† Therefore these devices, like streamwise slots, are hybrid devices, although the significance of the streamwise vorticity induced may not be significant.



(a) Variation with Mach number, C_N = 0·75



(b) Variation with normal-force coefficient M_∞ = 0·725

Figure 19. Drag characteristics of bump on RAE 5225 aerofoil, Ref. 63.

structure for a shock with M_s = 1·3 of various 3D bump designs and concluded that the three-dimensional shape of the bump is important in determining their turning effect on the supersonic flow upstream

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of the shock wave. Qin *et al*[68] used a RANS solver to predict flows over aerofoils with 3D bumps. Their calculations indicate that the increase in maximum lift to drag attainable with their particular 3D bumps is just under half that achieved with 2D bumps. However, further studies are needed to optimise the design of 3D bumps.

As far as the authors are aware, no attempt has been made to study 'inviscid' devices to control flows in highly three-dimensional regions, such as e.g. wing/pylon/nacelles or wing/body junctions. Although outside the scope of this review, there would also appear to be scope for the use of this class of device to control shock waves in intakes.

4.1.2 Boundary-layer control

a) SBVGs

Studies were undertaken by McCormick[69] with the aim of comparing SBVGs with a passive (perforated surface) device to control shock/boundary-layer interaction. These investigations showed that the former would be preferable if the aim were to control the shock/boundary-layer interaction. However, the latter should be used if wave drag needed to be reduced.

Holmes *et al*[70] were probably the first to make use of SBVGs to control shock-induced separation on the external surface of an aircraft. They successfully applied SBVGs to control a shock-induced separation occurring on the canopy of the Gulfstream III.

Investigations of the effect of SBVGs on the transonic flow over the aerofoil RAE 5243 have been described by Ashill *et al*[32]. Results for lift curves at M_∞ = 0·71, are given in Fig. 20 showing that 1h split or spaced vanes are more effective than vanes in increasing lift at high incidence. A significant increase in maximum lift is obtained with both devices, suggesting that SBVGs could have a useful role in improving manoeuvre performance of combat aircraft. The study also provided limited evidence showing that wedges placed upstream of the shock reduce shock strength and increase lift to drag ratio. However, this conclusion was not borne out in a subsequent basic study in the Cambridge transonic facility[71]. Nevertheless, in the latter study both wedges and counter-rotating vanes were found to reduce the extent or prevent shock-induced separation for strong shock waves (M_s = 1·5).

An investigation of the effect on the transonic flow over a swept, untapered wing of wire SBVGs of 45º sweep at high Reynolds number (17×10^6 based on chord normal to sweep) and at high subsonic speeds has been described in Ref. 44. The wires were stuck to the wing upper surface at up to four spanwise rows: 2%, 10%, 20% and 30% chord along lines that were normal to the wing leading edge. There was a slight increase in zero lift drag even with a single row but the overall effect of the wires was to reduce drag at high lift. Results for pressure distributions for at M_∞ = 0·85 and C_L = 0·6 at two spanwise stations with a single row of wires at 30% chord are shown in Fig. 21, indicating that the wires reduce the extent of the shock-induced separation on the wing upper surface.

b) AJVGs

Pearcey[3] and Pearcey *et al*[72] studied the effect of AJVGs on shock/boundary-layer interactions. Comparable results to those obtained with VGs were found with AJVGs in terms of the pressure recovery at the trailing edge of an aerofoil. Tilmann[73] investigated the effect of AJVGs on transonic flows over the aerofoil RAE5243. He found that while they gave a significant improvement in trailing-edge pressure recovery and in maximum lift, they caused an increase in the drag measured by a wake rake. Pulsating the jet at frequencies above about 300Hz reduced the drag increment but the drag was higher and the lift/drag ratio lower than for the basic aerofoil. This is an interesting but somewhat surprising result.

Raghunathan *et al*[74] have studied a novel form of AJVG, which is illustrated in Fig. 22. The device is similar to the passive control devices mentioned in Section 4.1.1. However, the air from the slot downstream of any region of pressure rise feeds air to vortex

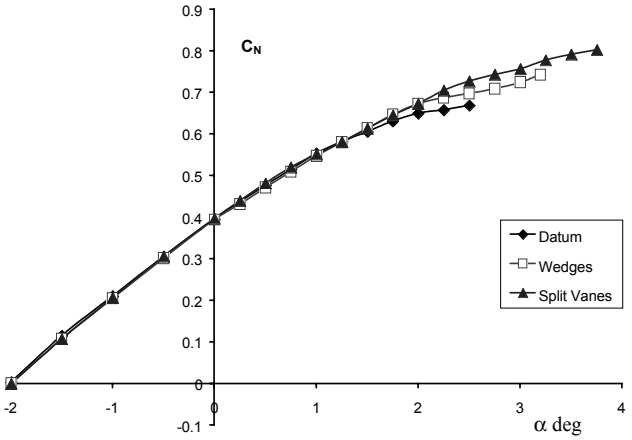


Figure 20. Normal force coefficient versus angle of incidence, M_∞ = 0·71, RAE 5243 aerofoil (Model 2303). Ref. 32.

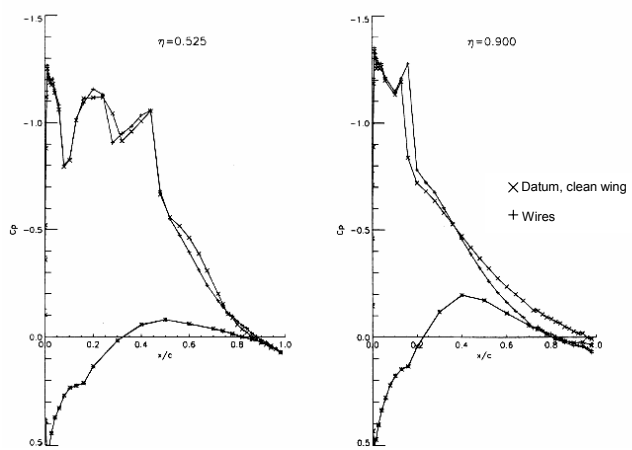


Figure 21. Effect on pressure distributions of wire SBVGs on a 45° swept wing at M_∞ = 0·85 (Ref. 44).

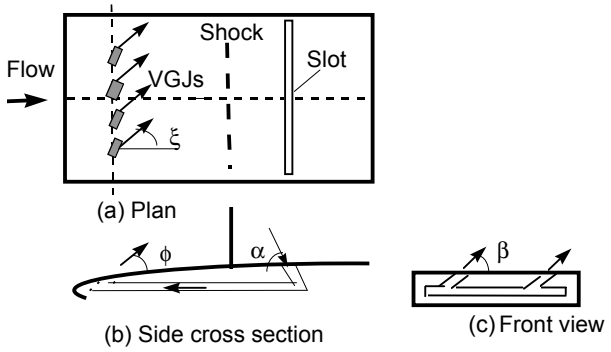


Figure 22. Novel passive AJVG system investigated by Raghunathan *et al*, Ref. 74.

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generators further upstream. Raghunathan *et al* applied this device to the control of separation in wide-angle diffusers, where it appears to have been successful. Such a device could equally be used to control shock-induced separation on wings, although it is difficult to say with certainty what the viscous drag penalty would be with this system.

c) Mass-less jets
Seifert and Pack[75] used a synthetic jet, generated from a slot placed at 10% chord on the upper surface of an aerofoil, to control shock-induced separation†. They found that the control was most effective when the slot was just upstream of the shock wave.

d) Suction
Fulker and Simmons[76] performed experiments in which they sucked boundary-layer air from the flow upstream of a shock wave on the aerofoil model M2303 (RAE5243) referred to previously. They found that quite small amounts of suction caused significant reductions in drag. Although suction increases wave drag, as discussed in Section 3.2.5, it reduces viscous drag, via its favourable effect on boundary–layer form or normal-pressure drag.

4.2 Rear separations

a) SBVGs
Experiments have been performed in the Boundary–Layer Tunnel (Formerly at QinetiQ, Bedford) using a bump on the roof of the test section to simulate a flow with a rear separation[32]. Single arrays of various SBVGs spaced apart by 1h were placed upstream of the separation. All the devices reduced the length of the separation region but the split vanes were the most effective in this respect. This is consistent with the observation that the vortex decay of the split vanes is much lower than that of the other two types tested (Forwards wedges and joined vanes). It was also found that although the vortices were relatively weak in the separation region, their favourable effect on the boundary layer was clearly evident.

As noted in Section 3.1, there are opportunities for control of flows over aerofoils and wings by using flaps close to the trailing edge, which give large changes in lift for such small chord devices. However, sections that are highly cambered near the trailing edge may have significant flow separations in this region. Thus this 'inviscid' form of control may need to be supplemented with some form of boundary-layer control, as noted before. Hackett[77] has pointed to the possibilities in this direction in his wind-tunnel study at high subsonic speed of the control of the flow over the rear of a 'novel' section with pronounced rear camber. Tests were also performed on a reference or 'conventional' aerofoil (without rear camber in the close vicinity of the trailing edge) (see Fig. 23(a)).

Results for drag increments due to split vane SBVGs measured and predicted by a modified version of the transonic aerofoil code BVGK[14], are shown in Figs 23(b) and (c) plotted against normal-force coefficient. The figure shows that the measured drag increment is larger than that predicted for the reference and 'novel' aerofoils with SBVGs at 55% chord. However, the predicted drag does not contain an allowance for the base thickness of the SBVGs nor the thickness of the glue used to attach them to the aerofoil surface. Figure 23(b) shows an estimated increment in drag due to these sources for the SBVGs positioned at 55% chord, and this goes a substantial way to explaining the differences between measurement and prediction with the SBVGs at this station. For the other cases with SBVGs at 60% and 70% chord on the 'novel' aerofoil, inclusion of this correction leads to reductions that are significantly larger than predicted. However, it should be noted that the correction for spurious sources would become smaller as the SBVGs are moved downstream. Nevertheless, it may be inferred that significant

† The authors are grateful to Dr David MacManus for drawing their attention to this reference.

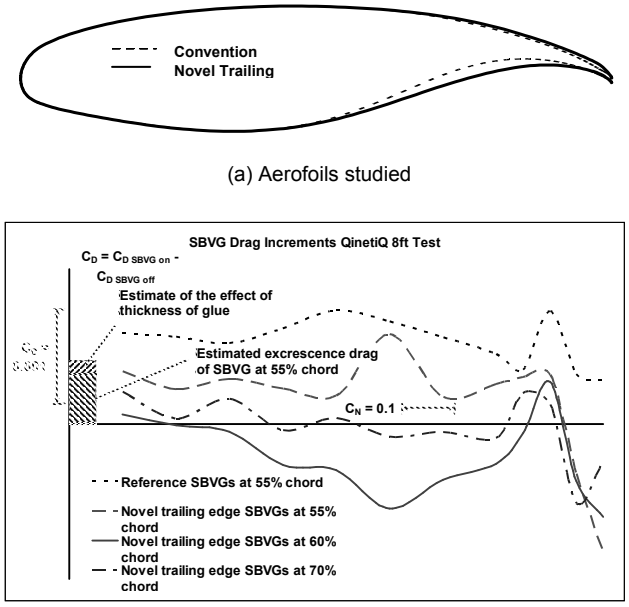


Figure 23 Measured and predicted increments in drag coefficient due to SBVGs on 'novel' rear-cambered and 'conventional' aerofoils[77].

reductions in drag are possible for aerofoil sections with large rear camber by using appropriately placed SBVGs.

The flow model used in these calculations exploits the idea that closely spaced SBVGs have an effect on the boundary layer that can be simulated as:

i) An increase in boundary-layer momentum thickness at the position of the SBVGs.
ii) An increase in entrainment coefficient downstream of the SBVGs.

Unpublished calculations[78] show that this crude model gives satisfactory predictions of the effect of SBVGs on overall forces, pitching moment and pressure distributions for the aerofoil RAE 5243. However, the comparison between prediction and measurement for

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the 'novel' aerofoil shown in Fig. 23 suggests that further refinements may needed to the method, although some of the differences between measurement and prediction may be due to the inadequacy of the correction for the spurious effects mentioned above.

The need to use SBVGs, or indeed any other form of boundary-layer control, to control rear separations on the upper or lifting surfaces of aerofoils with locally high camber near the trailing edge is avoided in the device proposed by Gurney[79] and that suggested by Henne[80]. However, both of these devices feature base flows with an associated drag penalty, which might be reduced by some form of flow control of the separated flow in the base-flow region.

A similar approach has been to use SBVGs to control the flow over slotted flaps. Further details may be found in the review paper of Lin[6]. The SBVGs are stowed in the main aerofoil when the flap is retracted, and so any adverse effect on drag of the SBVGs is not felt at cruise conditions. More recent work is described in Ref. 81, in which results of experiments on swept wings over a range of Reynolds numbers were discussed. Results for a simple tapered wing with a slotted flap reveal the following key features:

- For a landing configuration with a flap angle of 40°, SBVGs reduce the sensitivity of lift to flap gap by between 50% and 60% at angles of incidence below those for C_{Lmax}, and the optimum flap gap increased. These factors make the positioning of the flap less critical than without control.
- At the same flap angle the SBVGs are much less effective at C_{Lmax} for most combinations of flap gap and Reynolds number owing to the dependence of maximum lift on factors other than the flap flow. However, increases in C_{Lmax} of 0·1 (about 0·3%) are observed at 3% gap.
- The SBVGs do not significantly affect the lift/drag ratio for take-off configurations.

Broadley and Garry[82] have described a wind-tunnel study of the effect of various SBVGs on rear separations on swept wings.

b) AJVGs
Fairly extensive wind-tunnel studies have been performed at City University, London on aerofoils with slotted flaps by a number of workers, including Innes[83], Innes *et al*[84] and Lewington[47]. These studies have shown that air jets placed on the upper surface of the main aerofoil of a slotted-flap configuration give increased maximum normal force coefficient. These increases are found to be greater than those obtained with vane VGs.

Tilmann[73] described a series of experiments using pulsed AJVGs to control flow separations on the upper surfaces of wings and an NACA-4412 aerofoil. Significant improvements in lift and lift/drag ratio were obtained.

Singh *et al*[85] used pulsed AJVGs to control the flow separations on helicopter blades, demonstrating that pulsed jets could provide the same enhanced performance as steady AJVGs, with significantly reduced mass flow requirements.

c) Mass-less jets
Stanewsky[5] reviewed wind-tunnel studies of the application of mass-less jets to control the flow over a slotted flap, over which the flow was wholly or almost wholly separated. Increases in maximum lift of up to 30% were obtained with the highest flap angle tested.

d) Devices simulating bird feathers
Bechert *et al*[58] discussed the application of this type of device to a glider where an increase in maximum lift coefficient of 18% was obtained.

e) Streamwise grooves
Lin *et al*[86] used streamwise grooves to control the separation on a rearward-facing ramp in low-speed flow, and found that they were able to reduce the length of the separation region by 66%.

f) Change in effective surface shape
As noted in Section 3.1, Fulker and Alderman[24] have investigated the use of effective shape changes in regions where the boundary layer is close to separation to effect large changes in lift. They have applied this concept to the lateral control of tail-less aircraft, making the effective change in shape by blowing through the wing surface.

4.3 Leading-edge separations

4.3.1 Wings of low sweep (<40°)

A review of the state of art in this area has been provided by Heap and Crowther[87], and they suggest that this is an area that is ripe for further research.

a) Mechanical or pneumatic forms of control
As already shown in Section 3.1, variable effective leading-edge radius is a powerful way of controlling leading-edge suction peak, as is a device for increasing nose camber but both may involve significant structural and or mechanical complications. An example of a camber-altering device is that developed by Pierce at RAE and known as RAEVAM[88]. More recent developments in materials and micro mechanisms might make it possible to refine such a scheme.

Pneumatic devices, such as slot blowing, require air to be supplied to the leading-edge, which leads inevitably to complications and losses in thrust.

b) SBVGs
No published information is available to the authors on the use of SBVGs or low profile VGs to control leading-edge separations on aerofoils or wings of low sweep. Such devices would be used to control high-lift or high angle of incidence performance, and, as such, would need to be located between the attachment line (on the lower surface of the wing) and the suction peak. In this region the flow accelerates rapidly, giving a very favourable pressure gradient. The effect of favourable pressure gradient on the flow downstream of SBVGs has not yet been studied. However, based on studies of the effect of adverse pressure gradients on counter-rotating spaced vanes[33], it is expected that favourable pressure gradients will delay the merging and subsequent dissipation of the vortices, making them more persistent. As well as the effect of the favourable pressure gradient, the change in surface flow angle between the attachment line and the suction peak needs to be taken account for swept wings.

SBVGs could be used to supplement the control by small changes in leading edge shape described under the heading of 'Inviscid' flow control devices in Section 3.1.

b) AJVGs
AJVGs were first proposed for the control of leading edge flows by Wallis[46]. More recently, it has been suggested that micro-scale pulsating jets could be used for this purpose since they are likely to be more effective than macro devices in the very thin boundary layers in this region[52].

4.3.2 Highly-swept wings

The use of wire SBVGs to control leading-edge separations on highly-swept, cambered wings at low speed has been described in a series of papers[89-91]. The types of flow studied are shown in Fig. 24(a) and (b). In both types of flow there is an ordinary separation on the wing upper surface S_1, which at low incidence ($C_L < 0{\cdot}4$, Fig. 24(a)), is roughly parallel to the leading edge. The leading edge separation occurring near the tip is distinct from this separation. This type of leading-edge flow separation is referred to as 'locally-

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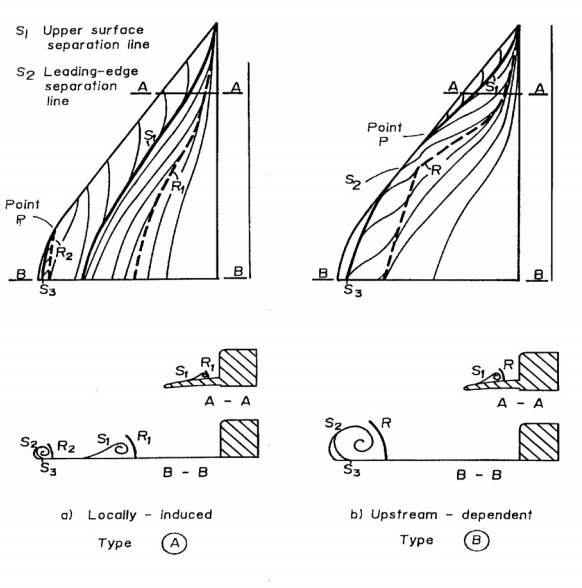


Figure 24. Types of flows studied over highly swept cambered, delta wing[89, 90, 91].

induced'. At higher angles of incidence (Fig. 24(b)), the upper-surface separation intersects the wing leading edge at the point '*P*', inducing a leading-edge separation further downstream. This leading-edge separation is described as 'upstream dependent'.

A wire SBVG placed upstream of the separation line S_1 moves this separation downstream, in the way discussed in Section 3.2.1, thus shifting the position of the point '*P*' denoting the onset of leading-edge separation downstream. The effect is to increase leading-edge thrust and reduce lift-dependent drag.

Increasing the number of wires results in two additional effects, one favourable and the other unfavourable. The first comes about because the wire SBVGs shift the upper surface separation downstream, thereby allowing other SBVGs, previously downstream of the separation line, and therefore ineffective, to become effective. Each vortex that is upstream then displaces this separation downstream by an amount that is accumulative. The second effect also arises from the downstream displacement of the upper surface separation. As the distance of the separation line downstream of a given wire increases, due to the effect of wires further upstream, so the decay of the vortex generated between the wire and separation line increases. This ensures that the strength of this vortex at the separation line is reduced.

As well as reducing lift-dependent drag wire SBVGs decrease wing-tip buffet intensity[44].

4.4 Vortex wakes

Highly-swept lifting surfaces with sharp leading edges feature vortical flows. These flows are basically stable and an efficient way of providing lift on wings of low aspect ratio, although they do not provide leading-edge thrust. However, at high angles of attack and/or some distance downstream of the lifting surface, the vortex may break down causing a high degree of flow unsteadiness, leading to buffeting[92,93]. A great deal of effort has been expended in understanding this process and finding ways of controlling it[5]. Among a number of methods considered, is that due to Mabey and Pyne[94], where air is blown tangentially along the leading edge to energise the leading-edge vortex.

A related problem is the control of vortices shed from slender forebodies at high angles of attack (between about 20° and 60°)[5]. Here the problem arises from asymmetrical shedding of the vortices from the forebody, resulting in unwanted lateral sideforce and yawing moments. Wood and Crowther[95] showed that this phenomenon could be prevented by blowing through two slots along generators either side of the axis of the nose. They also showed that blowing at different levels could provide control of the yawing moment. In this respect their system is a genuine flow control device rather than some other device that merely fixes separation lines on the forebody.

5.0 CONCLUDING REMARKS AND RECOMMENDATIONS

5.1 Inviscid flow control

5.1.1 Devices for changing leading-edge radius or camber

Mechanical devices for controlling the shape of the leading edge are feasible with current technology. Pneumatic methods of changing effective shape have yet to be studied.

5.1.2 Changes in effective shape of wing in region of a shock wave

Shock control devices have reached differing degrees of maturity. The aerodynamic characteristics of 2D bumps on aerofoils are well understood and can be modelled without difficulty with current CFD methods. This form of shock control has been found to give significant reductions in drag at drag-rise conditions. As regards the implementation of these devices, considerable efforts have been expended in Germany on structural/mechanical/systems issues, suggesting that this method is mature enough to be considered for aircraft. The method is particularly attractive for wings of laminar-flow aircraft, on which the shock wave moves little with changes in flight conditions. For this application either fixed or variable-geometry bumps can be contemplated. The situation is more complex for turbulent-flow wings; in this case the supercritical flow on the upper surface is sensitive to flight conditions. Hence either a sophisticated form of bump control would have to be used or the control would need to be restricted to certain flight conditions.

Buffet bumps or bumps applied just downstream of the shock wave provide the facility to control shock-induced separation as well as reducing wave drag.

The prospects for '2D' bumps being effective on swept wings are good, purely on the basis of simple-sweep theory. Unpublished calculations by Dr A. Shires at QinetiQ suggest that they also give significant drag reductions on practical designs suitable for civil transport and military combat aircraft.

3D bumps offer the possibility of providing some boundary-layer control as well as 'inviscid' control of a shock wave. However, they are less well understood aerodynamically than 2D bumps and further wind-tunnel studies are justified to provide improved understanding.

The application of 2D or 3D bumps to 3D flows, such as in the region of the nacelle/wing junction, has yet to be considered.

The viscous drag of perforated-surface, passive-control devices has been found to be so high that it is unlikely that this form of shock control will be used to reduce overall drag, even where there is a strong shock wave. However, these devices might have a role in improving buffet performance.

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Streamwise slot methods have lower drag than the perforated surface method and have been shown to increase maximum lift of an aerofoil. However, despite significant efforts, more work of a basic character is needed to understand the effects on the control behaviour of spacing and slot or groove shape. Furthermore, there are likely to be issues regarding how to implement this technology on wings. The effect of sweep on this type of shock control has yet to be studied.

5.1.3 Methods of altering the effective camber near the trailing edge

Considering the problem purely on an inviscid basis, rear camber applied close to the trailing edge gives large increments in lift. However, the 'indirect' effects of the surface shape change on boundary-layer thickness can be large, with the consequence that the predicted increases will perhaps not be realised in practice, and the camber change may give rise to a large drag increase. Thus these devices are likely to benefit from some form of boundary-layer control, and SBVGs appear to be particularly suited to this task.

5.1.4 'Compliant' flows – flows close to rear separation

This novel concept has only recently been proposed, and it appears to offer real benefits for stealthy combat aircraft, although it is unlikely to find civil application because of the high drag associated with operating close to separation. Furthermore, the flows that are to be controlled are likely to be sensitive to changes in flow conditions. Hence further work is needed to investigate the viability of this concept.

5.2 Boundary-layer or wake control

5.2.1 Vortex generators

a) Two-dimensional flows
The aerodynamic performance of VGs is well understood. Although SBVGs are a more recent concept, their flow characteristics bear many similarities to those of VGs. Similar rules apply as regards spacing between devices in terms of device height. However, the distance the SBVGs need to be placed upstream of the region to be controlled should be scaled by boundary-layer thickness rather than device height if the aim is to 'condition' the boundary layer.

CFD methods, in which the geometry of the SBVG is represented, have an important role. However, the massive demands on computer storage by such methods means that they are unlikely to become a routine design tool. Vorticity injection methods, in which details of the SBVG geometry are ignored, offer better prospects in this respect. Simple models representing arrays of SBVGs in a 2D manner are also worth considering.

Wind-tunnel studies have been made of the effects of adverse pressure gradient on the flow downstream of SBVGs, and a good understanding has been achieved. However, no work has yet come to light on the effect of favourable pressure gradients, as found, for example near leading edges of wings at high lift. As far as is known, no studies have been made into the possible benefits of SBVGs on flows over leading edges.

The drag of SBVGs has been studied experimentally and good progress is being made. However, further studies to establish figures of merit (e.g. mean reduction in boundary-layer shape factor divided by drag increment) for SBVGs are justified.

b) Swept wings
The effects of co-rotating, wire SBVGs on the flow over highly-swept wings have been studied in some detail. It is interesting to enquire if similar or improved benefits would be obtained with vane type SBVGs.

Studies of a basic nature are needed to assess the effect of boundary–layer cross flow on the vortex decay characteristics associated with SBVGs.

c) Effects of compressibility
There is scope for investigating the effects of compressibility on the flows induced by SBVGs, particularly in the supersonic range. Studies of the effect of SBVG planform on the flow induced by the SBVG would be worthwhile in this connection.

5.2.2 Air jet vortex generators

The control effect of AJVGs on the flow over aerofoils, in some cases with transonic flows, has been studied in some depth but only a limited amount of work has been done to study their effect on swept-back wings. It is possible that they could provide significant improvements in lift to drag ratio for highly swept wings typical of combat aircraft. Therefore this is an area in need of attention, and it is understood that such work is proceeding in UK.

The use of microfabricated systems to actuate pulsed jets of micro-millimetric size should be explored for the control of leading-edge flows. Systematic studies of the effect of air-jet hole size relative to a boundary layer length scale are justified.

5.2.3 Mass-less or synthetic jets

Although fundamental aspects of mass-less jets are well understood, only a very limited amount of work has been undertaken towards their application to the control of the flow over wings.

5.2.4 Miscellaneous devices

Some interesting research has been done in recent years on a number of devices, including bird feathers, slot suction on aerofoils, slot blowing on bodies and blowing into vortices. It remains to be seen if these devices will be applied to aircraft.

5.3 Conclusion

In conclusion, Flow Control promises significant improvements in overall performance where good off-design behaviour is required. Such a requirement is likely to be important for certain stealthy, combat-aircraft configurations, which have to combine 'stealth' with good manoeuvre performance. Civil transport aircraft already benefit from flow control by increasing the lifting efficiency of slotted-flap configurations at high lift. The future possible requirement for transport aircraft to be able to operate over a wide range of altitudes and speeds may make Flow Control necessary for cruise configurations. The research done to date, together with additional research suggested here, will provide increased confidence in design methods that include Flow Control from the onset rather than as an afterthought.

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