

VORTEX GENERATORS FOR CONTROL OF SHOCK-INDUCED SEPARATION

PART 2: GUIDE TO USE OF VANE VORTEX GENERATORS

1. INTRODUCTION

This Data Item is Part 2 of a sequence dealing with the use of vortex generators for control of shock-induced separation. It provides a Guide to the initial selection of an array of vane vortex generators for control of shock-induced separation. Advice is also given on the subsequent evaluation and ‘tuning’ of the vortex generator array so as to arrive at an optimum. The Data Item is presented *on the assumption that the vanes are to be applied to a wing where the flow is quasi two-dimensional*, although similar flows over other surfaces can be treated in a comparable manner. Part 1 (Data Item No. 93024) presents an introduction to vortex generators for control of shock-induced separation, together with a treatment of aerodynamic principles, whilst in Part 3 (Data Item No. 93026), some example applications of vortex generators are described in such a way as to demonstrate the aerodynamic principles involved and their practical consequences.

The Guide takes the form of a step-by-step procedure which is given in Section 12. In order to follow these steps, and arrive at a specification for a suitable array of vanes, it is necessary to allow for a number of aerodynamic effects, some of which interact. To do this, Section 12 refers to specific Sections which contain the detailed analyses and considerations on which the procedure is based. Co-rotating and counter-rotating arrays are considered. To a large extent the present recommendations for vane height, spacing, aspect ratio and vane incidence are in close correspondence with those given by Pearcey in Reference 13 – both for co-rotating and counter-rotating arrays. This close correspondence to recommendations that have been proved reliable in many applications adds confidence in the present recommendations.

The first few steps of Section 12 deal with the assessment of whether a single array of vanes is likely to be effective in providing full* control of shock-induced separation for the particular case under consideration. Subsequent steps enable the defining parameters of a vortex generator array providing the required level of control to be found. Although not the only way of determining a vortex generator layout adequate for a particular application, Section 12 is a systematic procedure for using data currently available.

In deciding on the vanes to be specified for a particular application, much may depend on the extent of any subsequent testing or investigation that can be undertaken in order to improve on the first specification and ‘tune’ it so as to match the requirements in an optimal manner. If little or no such tuning is permissible, then the specification should err on the side of vanes that provide a high level of confidence of effective control, rather than smaller or more widely spaced vanes, or ones producing weaker vortices, since these may fail to achieve the effectiveness desired. If further development, or tuning, is permissible then the initial specification is clearly not critically important. Each of these options is considered here. If the latter applies, it is necessary to decide whether to start with vortex generators that provide a high level of confidence of effective control and subsequently investigate arrays which are less extensive (and/or comprise weaker or sparser vortices), or to start with an array for which there is a lower level of confidence of effective control and then, if necessary, strengthen the array.

Some of the steps in Section 12 make use of aerodynamic data which relate to the particular application, such as shock locations, certain boundary layer parameters, and flow directions. The details of the data requirements are given in Section 4. Carrying out tests and/or conducting calculations to produce such data could entail a considerable amount of additional work. However, most of the data would be available at the outset, as they would be required for other design purposes. Since this may well not apply in the case of

* See Section 3.

the flow direction data, special consideration is given to their provision in Section 4.3.

The complete list of References, covering Parts 1–3, appears in Part 1. For convenience, the References used in this Part are listed in Section 11. Reference numbers used are those allotted in Section 1.

2. NOTATION

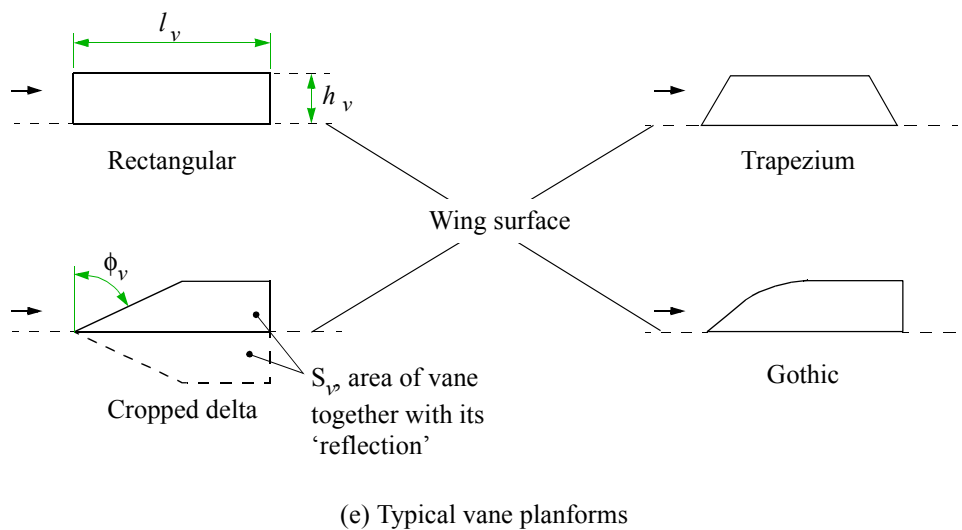
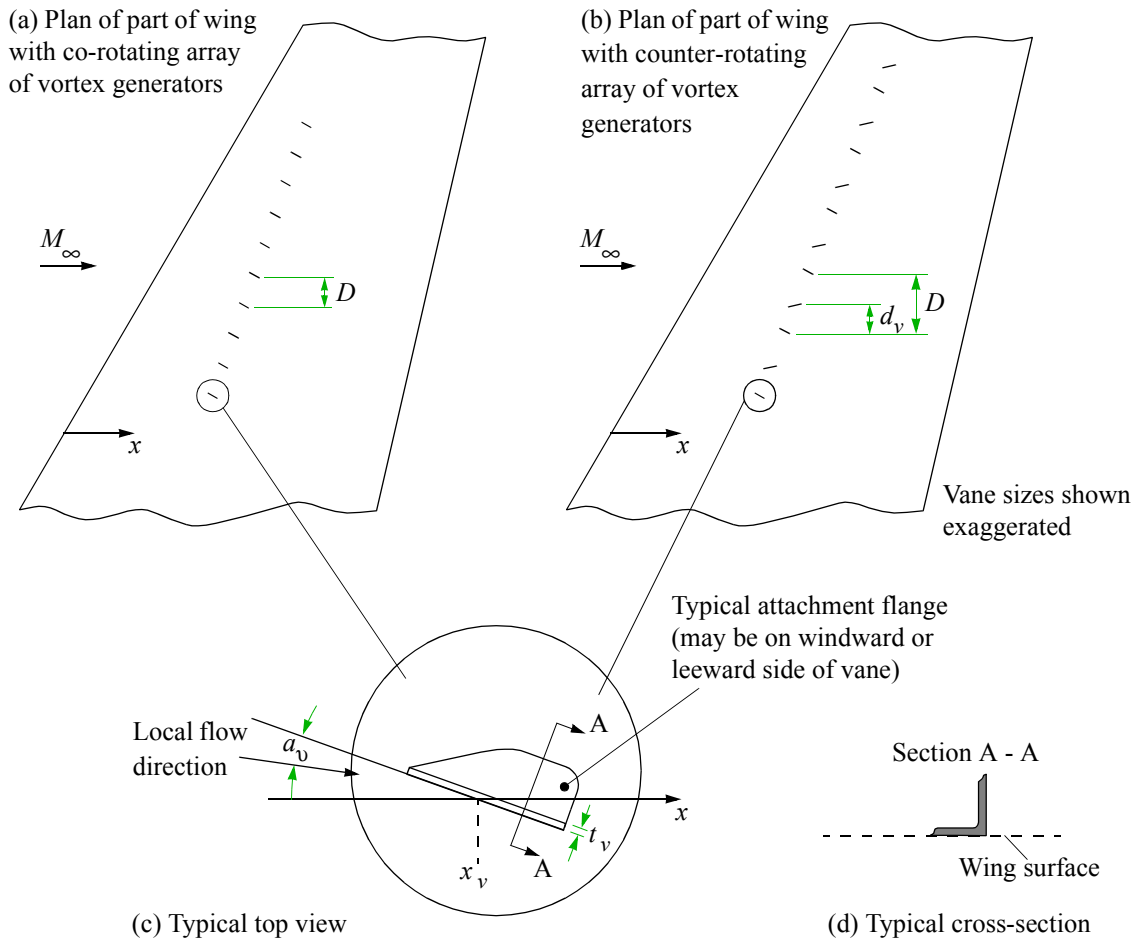
A_v	aspect ratio of vane vortex generator (<i>together with its reflection</i>), i.e. $(4h_v^2)/S_v$, see Sketch 2.1
C_L	lift coefficient, based on wing reference area
C_{L_v}	lift coefficient of vane, $= 2L_v/(\frac{1}{2}\rho_v U_v^2 S_v)$
C_p	pressure coefficient
c	local chord
D	lateral distance between neighbouring vortices (for co-rotating arrays of vortex generators), or least distance between vortices inducing rotational flow of same sense (for counter-rotating arrays)
D_v	drag of vane, i.e. component of aerodynamic force acting on vane in direction of local stream
d	lateral distance between pair of counter-rotating vortices, selected so that induced flow between them is towards surface
d_v	lateral distance between trailing edges of pair of counter-rotating vane vortex generators, selected so that induced flow between them is towards surface, see Sketch 2.1
F_{TI}	aerodynamic force acting on toed-in vane
F_{TO}	aerodynamic force acting on toed-out vane
h	local height of vortex
h_v	height of vane
K	local vortex strength
k_D	$D_v/\frac{1}{2}\rho_v U_v^2 h_v^2$
k_v	$K/h_v U_v$, see also Section 7.2
L_v	lift of vane, i.e. component of aerodynamic force acting on vane normal to local stream
l_p	streamwise distance over which full pressure rise downstream of shock occurs with vortex generators present, see Section 5

l_v	length of vane vortex generator, see Sketch 2.1
M_L	Mach number of local flow at edge of boundary layer on wing or aerofoil surface
M_{L_n}	Mach number of component of local flow normal to isobar, at edge of boundary layer
$M_{L_{nsh}}$	value of M_{L_n} immediately upstream of shock (quasi two-dimensional flow)
$M_{L_{sh}}$	value of M_L immediately upstream of shock (two-dimensional flow)
M_∞	freestream Mach number
P_v	k_v/k_D
p	local surface pressure
p_∞	freestream static pressure
S_v	planform area of vane vortex generator (<i>together with its reflection</i>), see Sketch 2.1
s	wing semi-span
T_e	local static temperature at edge of boundary layer
T_∞	freestream static temperature
t_v	thickness of vane vortex generator, see Sketch 2.1
U	local mean flow speed within boundary layer
U_e	local flow speed at edge of boundary layer
U_v	local flow speed at edge of boundary layer at vane location
U_∞	freestream speed
v'_n	local flow velocity perturbation at edge of boundary layer normal to isobar, divided by U_∞
x	streamwise coordinate, zero at local leading edge
x_R	value of x corresponding to point on upper surface where adverse pressure gradient becomes stronger, relatively suddenly, see Section 5.2.1
x_{sh}	value of x corresponding to shock location
x_v	value of x corresponding to vane mid root chord point, see Sketch 2.1
x_0	value of x corresponding to effective origin of vortex

α	angle of incidence of wing
α_v	angle of incidence of vane vortex generator relative to local stream direction at edge of boundary layer, positive when vane is toed-out relative to this direction, see Sketch 2.1
β	difference in angle between external and surface streamlines, see Section 10.3
δ	boundary layer thickness (99 per cent thickness) at vortex generator location
δ_{sh}	boundary layer thickness just upstream of shock location – corresponding to (or evaluated for) boundary layer flow without vortex generators
θ	local streamwise momentum thickness
ΔD_v	contribution to aircraft drag from vane vortex generator
θ_{sh}	streamwise momentum thickness just upstream of shock location – corresponding to (or evaluated for) boundary layer flow without vortex generators
Λ	local angle of sweep of isobar
λ_v	taper ratio of vane vortex generator
ρ_v	density at edge of boundary layer at vane location
ξ_v	vane setting angle, see Section 10.1
ρ_∞	freestream density
σ	angle between undisturbed local stream direction (<i>i.e.</i> in absence of vanes) and freestream direction, positive when local flow is directed inboard
σ_{datum}	angle between airframe geometric datum and freestream direction, see Section 10.1
ϕ_v	leading edge sweep angle of vane vortex generator, see Sketch 2.1

Suffices

i	denotes initial value, <i>i.e.</i> immediately downstream of vortex generator(s)
f	denotes most forward shock location
r	denotes most rearward shock location
te	denotes value at local trailing edge



Sketch 2.1 Vane vortex generator features and notation

3. GENERAL CONSIDERATION OF CONTROL OF SHOCK-INDUCED SEPARATION USING VANE VORTEX GENERATORS

Vane vortex generators have been successful in controlling shock-induced separation in a variety of different situations – see Part 1, Section 3. Shock-induced separations on aircraft wings comprise a common and important application, with the aircraft flight Mach number then being generally in the range 0.6 to 0.9. Both swept and unswept wings have been treated. For the former, depending on the design, the flight Mach numbers need not be less than one. In addition to the application to wings, successful control has been achieved of separations occurring in flows over other surfaces (*e.g.* tailplane, fin and fuselage) and in junction regions (see Part 1, Section 8.5).

Where the wing (or other lifting surface) has a high aspect ratio, shock-induced separation is usually associated with a single shock wave, see, for example, Reference 44, with separation occurring first well outboard near the tip. With lower values of aspect ratio, or high values of sweep, the shock pattern is generally more complex, and a ‘three-shock’ pattern is common (see Reference 44). In these cases the separation again first occurs outboard as a result of the effect of the strong combined shock on the outer part of the wing. In both situations, increasing the incidence or Mach number results in the development of more severe and extensive separations.

It is usual for separation control to be needed for a range of flight conditions, with the controlled flows corresponding (at any spanwise location) to a range of shock strengths, shock locations and boundary layer conditions. Finding a vortex generator array (including its location) so as to provide the desired control for such a range of flight conditions imposes further conditions, and might not actually be possible. If it does prove to be possible, then some freedom of choice of the array parameters would probably remain. In addition to controlling separation for a range of flight conditions, any array of vortex generators adopted would need to give acceptably small adverse effects under those conditions where no separation control is needed, and also (possibly) under conditions where the vortex generators no longer offer separation control. In these circumstances the vortex generators preferred would probably be those that gave an adequate control with close to the minimum adverse effects.

In Section 5 the onset of adverse effects of shock-induced separation requiring control is assumed to be indicated by the behaviour of the wing trailing-edge pressure coefficient, $C_{p_{te}}$, – specifically by the divergence of $C_{p_{te}}$ from its attached flow trend with increase in either M_∞ or α (see Part 1, Section 7.1 and References 10 and 14). *Full separation control* is considered to be achieved when the divergence behaviour of $C_{p_{te}}$ is eliminated from the range of relevant flight conditions. In this sense, full separation control corresponds to the flow processes associated with the shock pressure rise (*e.g.* separation, enhanced mixing and re-attachment) being completed upstream of the trailing edge, for then the full pressure rise occurring downstream of the shock is achieved without significant influence on $C_{p_{te}}$.

In certain cases, and/or flight conditions, a greater degree of control may be required than that corresponding to the elimination of the divergence behaviour of $C_{p_{te}}$. This would be the case if the complete suppression of the shock-induced separation were required – for example, to prevent a local vibration. For such cases the high degree of control (including effective separation suppression) that may be available locally from suitable counter-rotating arrays of vortex generators, makes this type of array particularly deserving of attention (see Part 1, Section 6).

There may be some benefit from control of shock-induced separation even where the aim of *full separation control* is not achieved. For example, buffet might be reduced to an acceptable level, even without $C_{p_{te}}$ divergence being completely avoided.

4. FLOW DATA

4.1 General Flow Data Requirements

Various flow data are required at different stages of the method of this Data Item and the overall requirements are summarised in Table 4.1. In particular, to make informed decisions on the features of the vortex generator array to be utilised in a particular application it is important to assess parameters for both of the following cases.

- (a) ‘Uncontrolled flows’, *i.e.* the flows containing the shock-induced separations that it is intended to control.
- (b) ‘Controlled flows’, *i.e.* the flow situations obtained with vortex generators present.

It is to be expected that these data would be based on computations (possibly as suggested in Reference 23 or 47) or on wind tunnel observations. If the latter are used then some allowance should be made for the effects of differences between flight and test Reynolds numbers, see Part 1, Section 7. A further alternative for case (b), above, is to use extrapolation of data for case (a) – see Section 4.2.

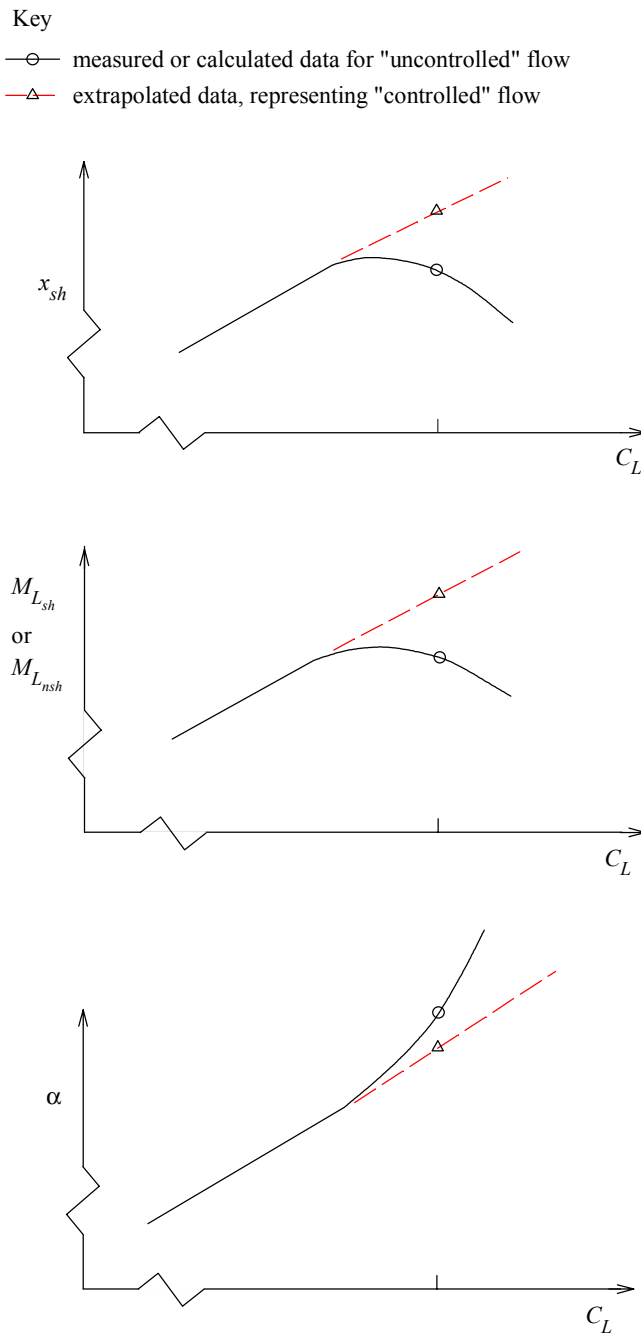
The extent of the flow data required depends on the task involved. In the case of a wing, the separations to be controlled are likely to relate to a range of flight conditions each of which may be denoted by a particular combination of freestream Mach number and lift coefficient (M_∞ , C_L). Then, for a representative selection of conditions (M_∞ , C_L), the flow field about an adequate number of chordwise sections over the span should be assessed so that, at least roughly, the parameters listed in Table 4.1 are found for each such section.

TABLE 4.1 FLOW DATA REQUIREMENTS

Flow parameter			Use of flow data	Source of flow parameter values	
				Uncontrolled flow	Controlled flow
(i)	Shock location	x_{sh}	Section 5 Section 8	From measured or calculated distribution of C_p	From calculated distribution of C_p with separation suppressed, or by extrapolation of values obtained for uncontrolled flow, see Section 4.2.
(ii)	Shock strength parameters	$M_{L_{sh}}$ $M_{L_{nsh}}$	Section 5		
(iii)	Boundary layer properties upstream of shock: Streamwise momentum thickness	θ_{sh}	Section 8	Calculation or measurement at x_{sh} .	Calculation.
(iv)	Boundary layer thickness	δ_{sh}	Section 8		
(v)	Local flow direction (at vane)	σ	Section 10	Flow visualisation techniques, see Section 4.3.1, or computational fluid dynamics, see Section 4.3.2, or approximate method, see Section 4.3.3	Same as for uncontrolled flow.
(vi)	Local isobar sweep angle	Λ	Section 4.3.3	From measured or calculated distributions of C_p .	Same as for uncontrolled flow.
(vii)	Local surface pressure coefficient	C_p	Items (i), (ii), (v), (vi) above, and Section 4.3.3.	Measurement or calculation.	Calculation with separation suppressed.

4.2 Approximate Method for Derivation of Shock Location and Strength in 'Controlled' Flow

Application of the procedures of Sections 5 and 8 requires shock location and shock strength data for the 'controlled' flow. In the absence of computed flow data, approximate estimates of x_{sh} and $M_{L_{sh}}$ (or $M_{L_{nsh}}$) can be made by extrapolation of the data for the uncontrolled flow as illustrated diagrammatically for a wing in Sketch 4.1. Clearly, this approach becomes less acceptable as the extent of the extrapolation increases but, in general, it is still preferable to working only with the data for the uncontrolled flows.



Sketch 4.1 Estimation of x_{sh} and $M_{L_{sh}}$ (or $M_{L_{nsh}}$) for wing with 'controlled' flow by extrapolation of 'uncontrolled' flow data

4.3 Determination of Local Flow Direction

Knowledge of local flow direction is needed for the optimum orientation of vane vortex generators. Sections 4.3.1 and 4.3.2 describe briefly the use of flow visualisation and computational techniques while Section 4.3.3 gives an approximate method for estimating local flow direction on a swept wing.

4.3.1 Use of flow visualisation techniques

Flow direction indications revealed by patterns of oil-flow or mini-tufts (*e.g.* References 21, 33 and 59) applied to a surface relate to the local directions very close to the surface and, at some locations on the wing planform, may differ markedly from the flow directions within the majority of the boundary layer and at the edge of the boundary layer – see Sketch 10.2. This possible marked change in flow direction has implications for the setting angles of vanes which have a height very much less than the boundary layer thickness. Further remarks on this effect are made in Section 10.5.

4.3.2 Use of computational fluid dynamics

The flow velocity at the boundary layer edge can be calculated using a computational procedure such as one for solving either the full potential equation, or the Euler equations (ideally incorporating a scheme for viscous effects) for flows over finite swept wings, if necessary with allowance for fuselage effects. If such calculations are available for the flow cases of interest, then the flow directions would also be available, although it might be necessary to undertake post-processing to extract these flow direction data in a suitable form.

4.3.3 Approximate method for estimating local flow direction on swept wings

In the absence of an alternative source of data, the flow direction at the edge of the wing boundary layer may be estimated from knowledge of M_∞ and the values of C_p and Λ .

The governing assumptions and derivation of this method are given below while a graphical presentation of the results is given in Figure 4.

Assumptions

Flow outside the boundary layer is isentropic from conditions far upstream to the vanes (since the vanes are to be placed upstream of any strong shock wave).

The flow perturbation velocity is normal to the direction of the local isobar and lies in a plane containing the local isobar and the freestream direction. (This assumption is correct for uniform thin swept wings and is a fair approximation for swept tapered wings except close to the root or the tip, or in any situation where the wing flow is not quasi two-dimensional*.)

Derivation of method

Denoting the local flow velocity perturbation normal to the local isobar (taken to be swept through an angle Λ) by $U_\infty v'_n$, and, using the second assumption above, the flow velocity, U_e , is found from

$$U_e^2 = U_\infty^2 (v_n'^2 + 2v'_n \cos \Lambda + 1). \quad (4.1)$$

* The wing flow is quasi two-dimensional where it exhibits only weak spanwise variations in the various flow parameters. Generally, if the spanwise variations of C_p are weak this is taken to be an adequate indication that the flow is quasi two-dimensional.

The angle between the flow velocity vector and the freestream direction is σ , where

$$\tan \sigma = v'_n \sin \Lambda / (1 + v'_n \cos \Lambda). \quad (4.2)$$

The following isentropic relation applies:

$$\frac{p}{p_\infty} = \left(\frac{T_e}{T_\infty} \right)^{7/2}, \quad (4.3)$$

where here (and in Equations (4.4) to (4.6)) the ratio of specific heat capacities is taken as 1.4.

Also,
$$\frac{T_e}{T_\infty} = 1 - \frac{M_\infty^2}{5} \left(\frac{U_e^2}{U_\infty^2} - 1 \right), \quad (4.4)$$

and
$$C_p = \frac{1}{0.7 M_\infty^2} \left(\frac{p}{p_\infty} - 1 \right). \quad (4.5)$$

In addition, the local Mach number, M_L , is found from

$$M_L^2 = 5 \left\{ \left(1 + \frac{M_\infty^2}{5} \right) \left(\frac{T_\infty}{T_e} \right) - 1 \right\}. \quad (4.6)$$

It is simpler to derive σ and C_p for various assumed values of M_∞ , Λ and v'_n , rather than attempt to find an explicit relation between C_p , M_∞ , Λ and σ . Relationships between σ and C_p are shown in Figure 4 for ranges of values of M_∞ and Λ .

5. APPROXIMATE LIMITS TO FULL SEPARATION CONTROL FROM VORTEX GENERATORS

Before commencing the process of selecting vortex generators, it is useful to assess whether *any* array of vortex generators is likely to provide the degree of separation control required. To help in making such an assessment this Section provides guidance based on the concept of 'full separation control' (*i.e.* elimination of $C_{p_{te}}$ divergence – see Section 3). Section 5.1 presents the available experiment data; Section 5.2 draws inferences from those data as to the approximate limits to full separation control achievable with vortex generators. As indicated in Section 3, there are situations where a greater degree of control can be required and where a lesser degree of control might be adequate.

Where the requirement is for full separation control, assessments can be based on observations of l_p , the streamwise distance required, with vortex generators present, for the pressure downstream of the shock to recover to the attached flow form. In practice, this requirement is satisfied if the distance between the shock location, x_{sh} , and the trailing edge is greater than or equal to l_p , *i.e.*

$$c - x_{sh} \geq l_p. \quad (5.1)$$

To achieve the limits suggested in Section 5.2, the vane parameters must be chosen appropriately, and the array suitably positioned relative to the shock wave. Such choices would not in general result in a particular array, and a whole range of different arrays could be selected.

The approximate nature of the suggested limits is due in part to the restricted range of flows yielding observations of l_p , as a result of which there is no means of assessing the influence of various factors, such as the nature of the pressure field downstream of the shock, the flow structure upstream of the shock, and the flow Reynolds number, each of which could have a significant influence.

Adopting l_p as the only parameter indicative of separation control is an over-simplified approach. It gives no information about the nature of the pressure rise in the vicinity of the shock, and no indication of the extent of the reduction in length of the separated flow region, even though such results are desirable.

5.1 Experimental Data for Full Separation Control

In Figure 1 values of l_p/c deduced from wind tunnel experiments (References 13 and 42) and flight determination (Reference 2) are plotted as a function of $M_{L_{sh}}$. These data relate to full- and half-aerofoils fitted with various vortex generator arrays indicated in the key to the Figure. The following comments refer to the data in Figure 1 and provide a background to the inferences drawn in Section 5.2.

- (i) All the data in Figure 1 have been determined from tests relating to quasi two-dimensional flows (*i.e.* the flow parameters have varied only slowly in the spanwise direction).
- (ii) The data relate to co-rotating vortex generator arrays (vanes or jets) except for one set for a counter-rotating array (Reference 2), and one set for a biplane array. The mixing effects (and thus the influence on separation control) of counter-rotating arrays and biplane arrays may be superior, locally, to those of any co-rotating array, see Reference 13.
- (iii) Certain data points are indicated with 'flags'. Two types of flag are used; one indicates that there is doubt as to whether full separation control has been achieved, the other that the vortex generators are so far upstream of the shock wave and region of adverse pressure gradient as to significantly diminish their effectiveness.
- (iv) The data fall into a number of bands, each linked to surface shape and characterised in the Figure by a rough mean 'curve'. This fact indicates that l_p/c is influenced by the shape of the model or wing surface. However, the limited amount of data available does not permit the factors producing the changes in l_p/c values between the bands to be fully identified. For each band of data, or curve, l_p/c appears to be insensitive to the shock strength, except at the 'upturn' of the curve near the highest values of $M_{L_{sh}}$. Examination of the data sources indicates that these upturns are associated with the most downstream shock locations, but they are probably due more to a decrease of vortex strength (and a consequent reduction of mixing effectiveness) in the critical regions of high adverse pressure gradient for these downstream shock locations than to the shock strengths and positions as such. The sensitivity of the values of l_p/c to surface shape is believed to be due to the latter's influence on the underlying (*i.e.* attached flow) adverse pressure gradient, aft of the shock. In general terms, the higher this gradient, the greater the pressure rise required to return to the attached flow form, and the greater the consequent length of the pressure recovery region – for a given effectiveness of shear-layer mixing.

5.2 Approximate Limits to Full Separation Control on Wings

Approximate limits to full separation control on wings are given in Sections 5.2.1 and 5.2.2 for co-rotating and counter-rotating vortex generator arrays. The limits relate to quasi two-dimensional flows; highly three-dimensional flows such as shock-intersection regions are not considered. In applying the limits with a swept shock, $M_{L_{nsh}}$ should replace $M_{L_{sh}}$. In evaluating shock locations for use in Equations (5.1) and (5.2), data corresponding to the controlled flows should be used (see Section 4.1).