

VORTEX GENERATORS FOR CONTROL OF SHOCK-INDUCED SEPARATION PART 3: EXAMPLES OF APPLICATIONS OF VORTEX GENERATORS TO AIRCRAFT

1. INTRODUCTION

This Data Item is Part 3 of a sequence dealing with the use of vortex generators for control of shock-induced separation. Part I (Data Item 93024) presents an introduction to vortex generators for control of shock-induced separation, together with a treatment of aerodynamic principles. In Part 2 (Data Item 93025), recommendations are presented relating to the application of vane-type vortex generators for control of shock-induced separation. In this Item, three Examples of applications of vortex generators to control shock-induced separation are presented, with the intention of giving practical background to the data presented in Parts 1 and 2. Two of the Examples (A and C) relate to actual aircraft applications, whilst Example B relates to a wind tunnel model of a swept-wing aircraft.

Example A deals with vortex generators applied to control shock-induced separation on the fuselage of the Gulfstream III aircraft – thereby rendering the cabin less noisy. In Example B, although vortex generators caused certain features in the measured aerodynamic characteristics to be deferred to higher incidences, the features were more severe when they were eventually encountered. In Example C, relating to the Canadair CF-100 MkIV A aircraft, the effects on the buffet boundary, of applying various vortex generators, are outlined and discussed.

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 3. Reference numbers used are those allotted in Part 1.

2. NOTATION

A_v	aspect ratio of vane vortex generator (<i>together with its reflection</i>), i.e. $4h_v^2/S_v$
b	wing span
c	local chord
$\bar{c}$	geometric mean chord
C_D	drag coefficient based on wing reference area
C_{D_v}	drag coefficient of vane, i.e. $2D_v/(\frac{1}{2}\rho_v U_v^2 S_v)$
C_L	lift coefficient based on wing reference area
C_{LL}	local lift coefficient (of aerofoil section)
C_p	pressure coefficient
D	lateral distance between neighbouring vortices (for co-rotating vortex generators), or least distance between vortices inducing rotational flow of same sense (for counter-rotating vortex generators)

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D_v	drag of vane, <i>i.e.</i> 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 main surface
h	local height of vortex
h_v	height of vane or other vortex generator device
K	local vortex strength
k_D	$D_v / \frac{1}{2} \rho_v U_v^2 h_v^2 = 2 C_{D_v} / A_v$
k_v	$K / h_v U_v$
l_v	length of vane vortex generator
l_p	streamwise distance over which full pressure rise downstream of shock occurs with vortex generators present
M_L	local Mach number on wing or aerofoil surface
$M_{L_{sh}}$	value of M_L immediately upstream of (unswept) shock
M_∞	freestream Mach number
S_v	planform area of vane vortex generator <i>together with its reflection</i>
t_v	thickness of vane vortex generator
U_v	local flow speed at edge of boundary layer at vane location
x	streamwise coordinate, zero at local leading edge, or at fuselage datum (see Example A)
x_{sh}	value of x corresponding to shock location
x_v''	value of x corresponding to vane leading edge
α	angle of incidence
α_v	angle of incidence of vane vortex generator relative to local stream direction
ΔC_D	increment in C_D due to vortex generators
η	spanwise position, expressed as fraction of semi-span from aircraft centre line
θ_{sh}	streamwise momentum thickness just upstream of shock location – corresponding to (or evaluated for) a boundary layer flow without vortex generators
λ_v	taper ratio of vane vortex generator

ρ_v density at edge of boundary layer at vane location

ϕ_v leading-edge sweep angle of vane vortex generator

Suffix

i denotes initial value. *i.e.* immediately downstream of vortex generator(s)

3. REFERENCES

Reference numbers are those allotted in Part 1.

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EXAMPLE A: GULFSTREAM III

A significant contribution to the cabin noise of the Gulfstream III, for a range of flight conditions, was due to boundary layer separation induced by a shock wave in the flow over the canopy, see Reference 36. Separation in this region resulted in what was termed a 'Mach rumble' type of interior noise. Aft of the separated flow region an externally mounted blade antenna generated a vibration with a frequency spike at 240 Hz. This vibration appeared to be produced by an unsteady vortex separation. Both the Mach rumble and the vibration were very effectively controlled by attaching a number of small ('sub-boundary layer') vortex generators to the fuselage upstream of the shock, and, in Reference 36, Holmes *et al* describe the processes of determining both the initial layout of vortex generators and its subsequent refinement. A number of important considerations for the application of vortex generators are illustrated by this work and are described in Sections A1 to A5.

A1. Vortex Generator Configuration

It was judged important that any vortex generators applied to the fuselage of the aircraft should be very small, mainly so as not to detract from its appearance. A system of counter-rotating vortex generators was adopted, with each individual device (which generates a counter-rotating pair of vortices) consisting of two low aspect ratio vanes attached to a base plate, see Figure A1. The geometric parameters of the vortex generators, including their spanwise spacing, were selected on the basis of experience gained through prior, unreported, wind tunnel and full-scale test programmes. The selected device height of 1/8 in (3.2 mm), which includes the base plate thickness, is about one eighth the predicted boundary layer thickness. It had earlier been determined that such sub-boundary layer vortex generators could remain effective at this size relative to the boundary layer. With a height of 1/8 in the vortex generators had little impact on appearance. A small vortex generator height also leads to low additional drag in those flight conditions where the vortex generators are not needed, but this did not seem to be an important consideration for this aircraft.

The counter-rotating arrangement of vortices chosen was known to yield a high degree of separation control, albeit for a relatively restricted streamwise range of shock locations. To achieve a relatively high vortex strength with the small vane height adopted, the chord of the rectangular vanes was chosen to be relatively long (at about 1.1 in), giving a vane aspect ratio of about 0.2. Also, the vane incidence was high, at 30 deg. Reference to Part 2, Figure 1 would suggest that these vane parameters could yield a value for k_v of the order of 2.0, although data from that figure apply to vanes in uniform flow. Vanes submerged in a boundary layer would lead to lower values of k_v , and close proximity to neighbouring vanes is likely to reduce k_v further. A prudent estimate might be $k_v \approx 1.0$. No direct measurement of vortex strength is reported by Holmes *et al*³⁶.

A2. Initial Application of Vortex Generators

It was recognised that the locations and lateral extent of the vortex generators applied to the fuselage would be critically important for the effectiveness of the separation control achieved. For a range of flight conditions, the vortex generators would need to be upstream of the shock wave producing the separations that had to be controlled, and by a distance that would probably need to be quite close to the optimum, (see Part 1, Section 6.2, relating to the mixing induced by counter-rotating vortex pairs.) This aspect is considered further in Section A4.

The decision on the initial placing of the vortex generators utilised flow field computations made using 'transonic small disturbance' flow codes. These results were used to predict the extent, location and strength of the canopy shock for a number of flight Mach numbers between 0.78 and 0.86 and for incidence values of 0 deg and 4 deg. According to these predictions, the canopy shock increased in strength and moved aft

with increasing Mach number. The aft movement amounted to about 8 in (200 mm) as the Mach number increased from 0.78 to 0.86. The shock strength and location were predicted to be only slightly affected by incidence. The shock strength was highest on the fuselage centre line and, although the shock extended over somewhat more than a sixth of the fuselage circumference, its strength diminished quite rapidly with increasing distance from the centre line. Values of shock-upstream Mach number (converted from the data presented in Reference 36) and shock location are shown in Figures A2 and A3 for the fuselage centre line and for two further lateral positions. So far as predicted shock locations are concerned, the longitudinal cell length in the transonic flow code used for these predictions (close to the shock the length was 3.4 in) indicates the likely tolerance as ± 1.7 in.

The initial choices of extent and location of the vortex generators were made on a very conservative basis, since it was highly desirable to have evidence of the effectiveness of the vortex generators from the first test flight. The choice of a conservative lateral extent was straightforward, and this choice was achieved by placing the vortex generators so that their mixing effects would encompass the full lateral extent of the predicted shock wave, corresponding to a flight Mach number of 0.85, with a few inches 'to spare' on either side.

The choice of a conservative fore and aft vortex generator location was not so simple. Based on unreported wind tunnel and flight tests it was assumed that the vortex generators would produce their best mixing effects, and separation control, over a streamwise extent having its fore and aft limits at about $x_v'' + 12h_v$ and $x_v'' + 64h_v$ respectively, where x_v'' denotes the position of the upstream extremity of the vortex generators. It was desirable for x_v'' to be not greater than the value of x_{sh} corresponding to the shock formed at a flight Mach number of 0.78. With this shock at the forward limit of its tolerance the consequence was that the vortex generators were placed so that x_v'' lay 118 in aft of the fuselage nose datum. If the shocks formed at their aft predicted tolerance limits, this vortex generator placing would still result in some cases of effective mixing in the regions of shock compression, but only for the shock locations corresponding to the lower cruise Mach numbers, *i.e.* up to about 0.81, see Figure A4.

A3. Initial Flight Test Results

With the initial configuration of vortex generators, the noise reduction was apparent to the flight crew without the aid of test instrumentation while analysis of recorded accelerometer and microphone signals enabled the benefits to be quantified. Outputs from an accelerometer, taken before and after the vortex generators had been applied, are shown in Figure A5. It is evident that the initial configuration of vortex generators significantly reduced the severity of separation induced by the fuselage shock, for a range of flight Mach numbers from about 0.78 to more than 0.85. In addition it was found that the blade antenna no longer had a strong vibration spike at 240 Hz, a result ascribed to a marked reduction in the flow turbulence exciting the vibration.

A4. Refinement of Vortex Generator Configuration

Following the success with the initial configuration of vortex generators, a systematic program of flight tests was conducted to determine whether the initial choices for the extent and location of the vortex generator row could be improved. The lateral extent was first investigated. In separate flights, two vortex generator pairs were removed successively from each side of the initial configuration until some degradation of the accelerometer data was noted. At that stage a vortex generator pair was added back to each side. In this way, flight tests were made with 21, 17, 13 and finally 15 vortex generator pairs installed. With fifteen vortex generator pairs installed, no measurable degradation was found in vibration levels, relative to the levels found with 21 vortex generator pairs installed.

Holding the lateral coverage with fifteen vortex generator pairs constant, the fore and aft position was then varied. The row of vortex generators was moved aft in steps of 4 in (102 mm) until a degradation in effectiveness was found. At that point the row was moved forward by 2 in for a final test point. This procedure entailed three additional flight tests, with the vortex generators positioned 4, 8 and then 6 in aft of the initial (baseline) placing. The location 4 in aft of the baseline was the one finally selected. Results of accelerometer outputs for the baseline position and for 4 and 8 in aft of this position are shown in Figure A6, in addition to the data with no vortex generators applied. It is seen that the selected location gave the lowest accelerometer output levels at flight Mach numbers up to about 0.82, but, at higher Mach numbers, a further aft location performed rather better.

A5. Discussion

The application of vortex generators to a fuselage flow is not specifically considered in Part 1, but some results, although determined from tests on lifting surfaces, may be utilised. It may be supposed that the fuselage shocks are, as predicted, almost normal to the local stream so no allowance for shock sweep need be made. Among relevant wing data that may be utilised are the correlations presented by Fulker and Ashill²⁶, for the lengths of flow separation induced by shocks of various strengths. It is found from these data that, for values of $M_{L_{sh}}$ above 1.45, the lengths of induced separated flow exceed $600\theta_{sh}$. Computed boundary layer data indicate that a value for θ_{sh} of about 0.09 in (2.3 mm) may be typical. Thus the shock strengths predicted to occur on the fuselage (Figure A3) would probably lead to lengths of separated flow in excess of 50 in (1270 mm), in the absence of any separation control.

It is of interest to consider the data presented in Parts 1 and 2 in relation to the vortex generators selected and the separation control achieved. Part 1, Figure 14 presents data for l_p , the length required for the pressure to recover to its attached flow form (with vortex generators present). It shows that l_p is of the order of $0.1c$ to $0.2c$ for values of $M_{L_{sh}}$ up to about 1.55 (with unswept shocks), with an indication that the higher values of l_p correspond to cases where the underlying attached flow has a relatively severe adverse pressure gradient. To apply the data of Part 1, Figure 14[†] to flows over a fuselage, it is necessary to choose a scaling length equivalent to c in the parameter l_p/c so that the pressure distributions and boundary layer characteristics of the test data are representative of those on the fuselage. The values of the data in Part 1, Figure 14[†] for l_p/c ranging between 0.1 and 0.2 correspond to values of l_p/θ_{sh} ranging between 120 and 240, approximately, and it is suggested that for fuselage configurations (in the absence of more definitive data) this range of l_p/θ_{sh} be used to estimate values of l_p .

In the case of the Gulfstream III fuselage, a calculated value of $\theta_{sh} = 0.09$ in, combined with the quoted range of l_p/θ_{sh} yields values of l_p in the range 10 in $< l_p < 20$ in, approximately. For vortex generators to be effective and yield these predicted values of l_p , they would need to produce vigorous mixing effects over comparable distances. In the case of the height h_v of the vortex generators applied to the Gulfstream III, this corresponds to vigorous mixing effects being required to extend for distances of $80h_v$ to $160h_v$ downstream of the shock waves tending to induce separation.

In Part 2, where recommendations concerning vane vortex generators are presented, it is suggested that, for counter-rotating vane vortex generators, at the 'lower level of confidence', separation corresponding to a range of shock locations extending over $60h_v$ can probably be successfully controlled, provided $h_v \geq 5\theta_{sh}$. The small vortex generators used here have a height which is no more than $2\theta_{sh}$ (it is probably only about $1.4\theta_{sh}$), and are therefore only some thirty per cent of the height recommended (even at the 'lower level of confidence') for the initial choice of vanes. However, it must be noted that the vanes used here have a much lower aspect ratio (and are therefore relatively longer), and are also set at a much higher incidence, than the ones recommended (for initial application) in Part 2. Thus the small height vanes should generate relatively strong vortices, with values for k_v possibly up to about 2.0, although, as noted in

[†] The same data appear in Part 2, Figure 1.

Section A1, a value for k_v of about 1.0 is probably more likely. This takes into account the fact that for turbulent boundary layers that are not close to separation, which is the situation upstream of the shock waves, even the lowest 10 per cent of the turbulent boundary layer still has a relatively high mean flow speed. However, the range of effectiveness of vortex generator vanes is influenced by their height, and for small devices the range of effectiveness is inevitably reduced below the ranges that could be achieved with a greater height. For the vane generators applied to the aircraft here, Figure A6 shows that, at $M_\infty > 0.82$, approximately, the effectiveness of the vortex generators is increased by moving them aft of the selected location, at least by 4 in, indicating that their control effectiveness (when placed at the selected location) is diminished significantly so far as the shock waves occurring in the more downstream locations are concerned. Higher vanes would have given an increased range of effectiveness, and could possibly have avoided the need to accept a reduced effectiveness of control at the higher flight Mach numbers.

If the predicted shock positions (see Figure A3) are accepted as roughly[†] correct, then the effectiveness of the vanes has fallen significantly for shocks occurring some 6 in, i.e. $50h_v$, downstream of the leading edge of the vanes. Since the vanes extend about 1 inch in the streamwise direction, this is equivalent to a distance of about $46h_v$ aft of the mid-chord location of the vanes, and $42h_v$ aft of their trailing-edge location. The evidence thus suggests that the region of vigorous mixing downstream of the vortex generators fails to extend to a distance much more than $200h_v$ aft of the trailing edges of the vanes.

As explained in Part 1, Section 6.2, the effectiveness of counter-rotating vortex generators is limited in its streamwise extent by the migration of the vortices away from the surface. Increasing the strengths of the vortices, whilst keeping other parameters unchanged, results in the migration commencing nearer to the generators, and in a corresponding reduction in the streamwise extent of effectiveness. Conversely, reducing the vortex strengths may increase the streamwise extent of effectiveness.

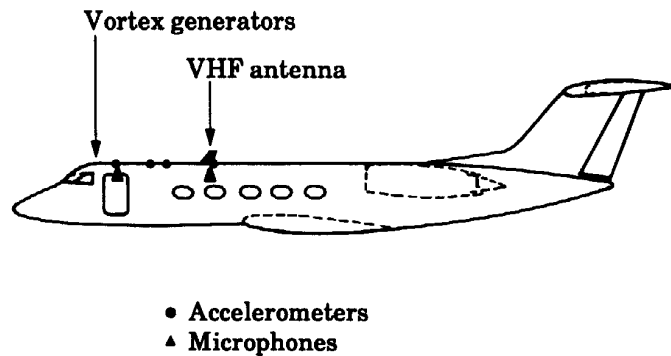
The evident limitation of the region of vigorous mixing afforded by the vortex generators applied to the Gulfstream aircraft may be considered further in the light of an analysis of the paths of vortices and fluid particles. Calculation of these paths, using the simple approach of Jones⁶ which is also presented in Part 1, Section 6, requires knowledge of the vortex strengths (or, rather, values of k_v) and the vortex positions at the 'points' where they are formed, which to an acceptable degree of approximation may be taken to be close to the trailing edges of the vanes. With k_v set equal to 1.0, and the initial vortex coordinates taken as indicated in Figure A7, the height of a typical vortex above the surface varies in the way shown in Figure A8. If the vortices forming a 'pair' are initially closer together than shown in Figure A7, then the vortex migration is delayed slightly, as indicated in Figure A8. It is found that little effective mixing is produced once the vortices have migrated to a height of $6h_v$. Thus, depending on the assumptions made, the vortex generator effectiveness may be expected to start to decline at a point which is some $120h_v$ to $150h_v$ downstream of the trailing-edge location of the vanes.

This range of values may be compared to the value of $200h_v$ determined above as the (upper limit) of the distance for a decline in effectiveness to have commenced. This latter figure was deduced on the basis of predicted shock locations and accelerometer outputs for different vortex generator locations, coupled with a presumed requirement (in this case) for the vigorous mixing level to extend $80h_v$ to $160h_v$ downstream of the shock location. Although there is a margin of uncertainty on these figures, their general agreement may be taken as an indication that the underlying assumptions and analysis of the vortex generator effects are at least roughly correct.

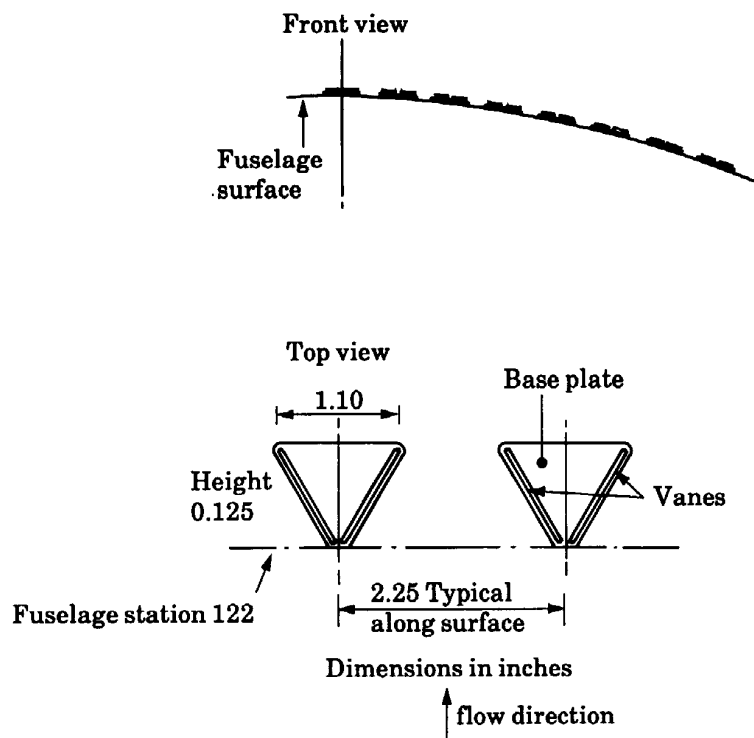
The application of vortex generators to this aircraft resulted in significant gains. However, on the basis of the analysis presented here, it is possible that a different form of vortex generator *might* have performed

[†] Positioning the vortex generators 4 in aft of their initial location led to an improvement in effectiveness of control, even at the lower flight test Mach numbers. Thus there is a strong indication that the actual shock locations are somewhat aft of the predicted locations (possibly by 2 or 3 in), for otherwise it would appear that the vortex generators are very effective even when the shock is located between the leading and trailing edges of the vanes.

rather better. In particular, although counter-rotating vortex generators undoubtedly produce high maximum effectiveness, this effectiveness is limited in its streamwise extent, and this limitation appears to be of some importance to the application to the Gulfstream aircraft. It seems likely that vortex generators with a somewhat greater height (for example, $3/16$ in or $1/4$ in) would have performed even better, but quite possibly these would have been deemed unacceptable on aesthetic grounds. It also appears possible that co-rotating vortex generators, no larger than the ones that were applied, might have given almost as large a maximum effect, but with a greater streamwise extent of effectiveness, thus providing rather lower levels of noise and vibration at the higher flight Mach numbers than were achieved with the counter-rotating vortex generators in their selected location.



(a) Location of vortex generators, antenna and noise and vibration instruments



(b) Location and geometry of vortex generators

FIGURE A1 Vortex generators on fuselage of Gulfstream III

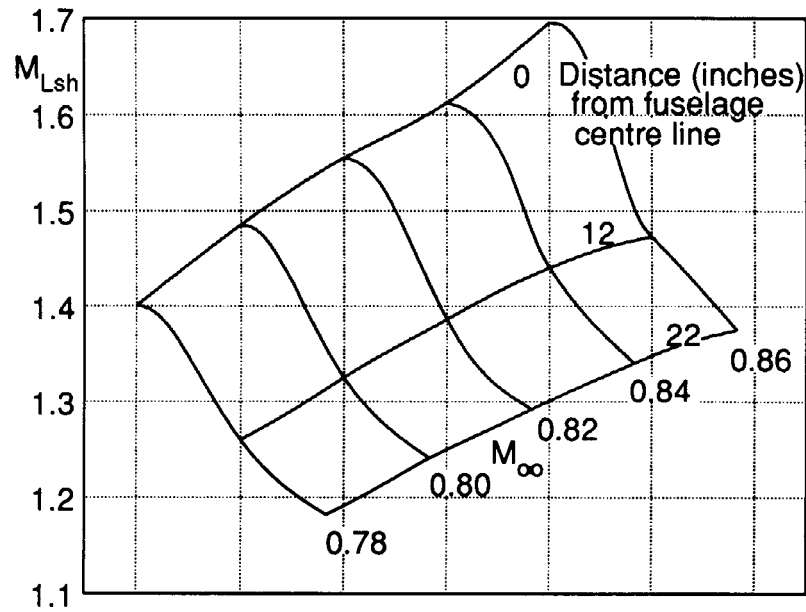


FIGURE A2 Shock strength predicted by 'transonic small disturbance' flow code – variation with M_∞ and distance from fuselage centre line

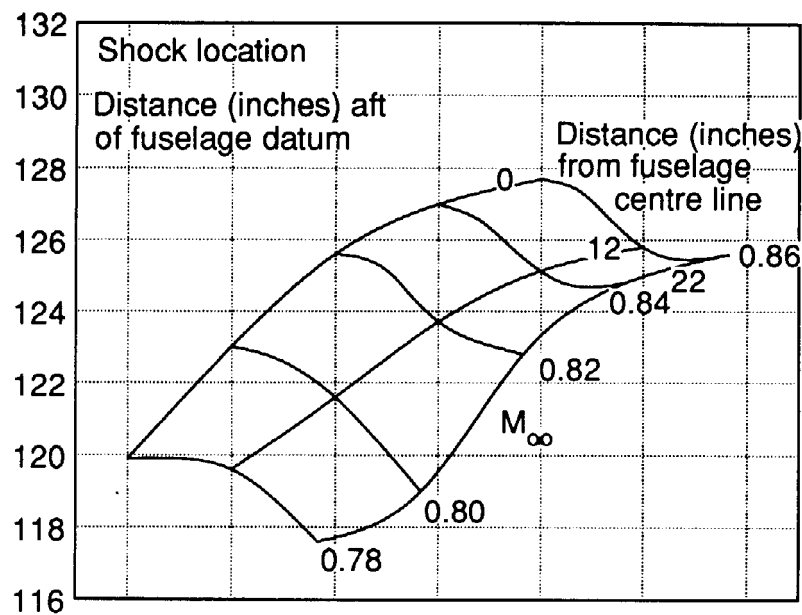


FIGURE A3 Shock location predicted by 'transonic small disturbance' flow code – variation with M_∞ and distance from fuselage centre line

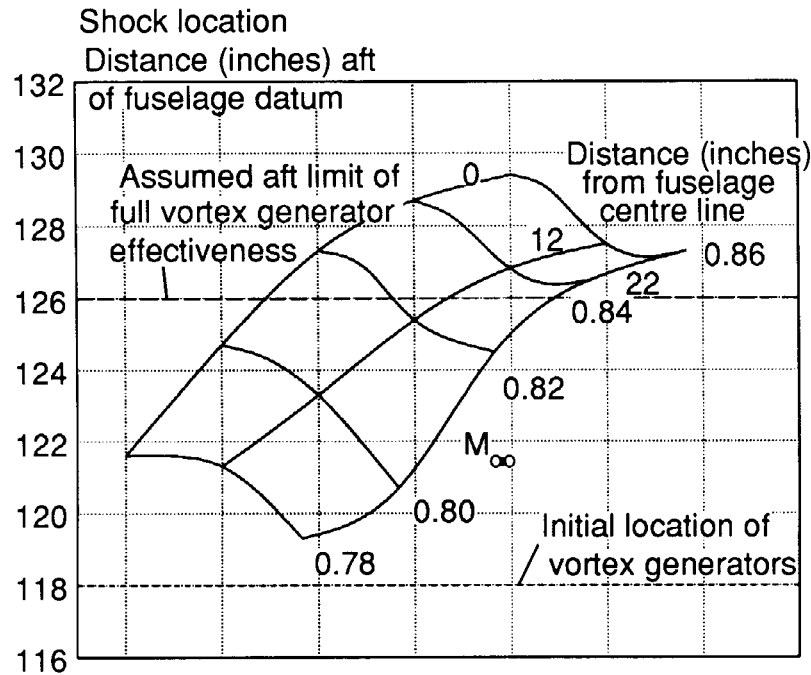


FIGURE A4 Initial location of vortex generators in relation to shock locations at their aft predicted tolerance limits

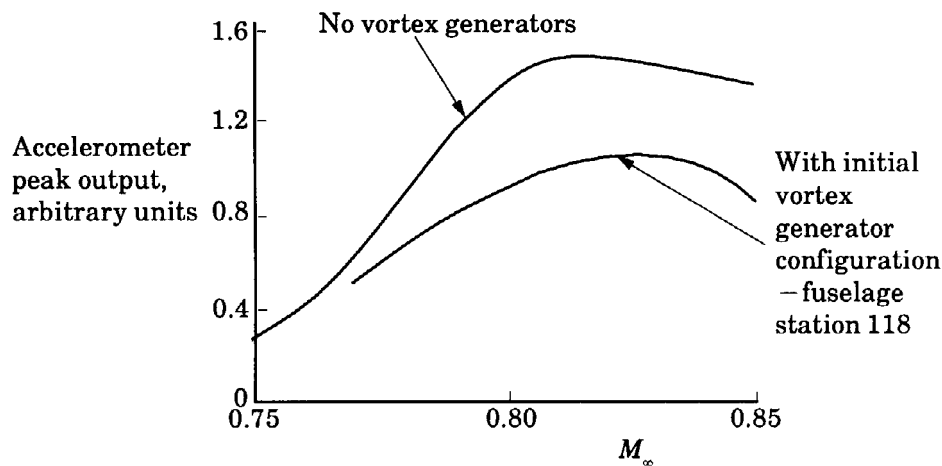


FIGURE A5 Effect of initial configuration of vortex generators on recorded accelerometer output

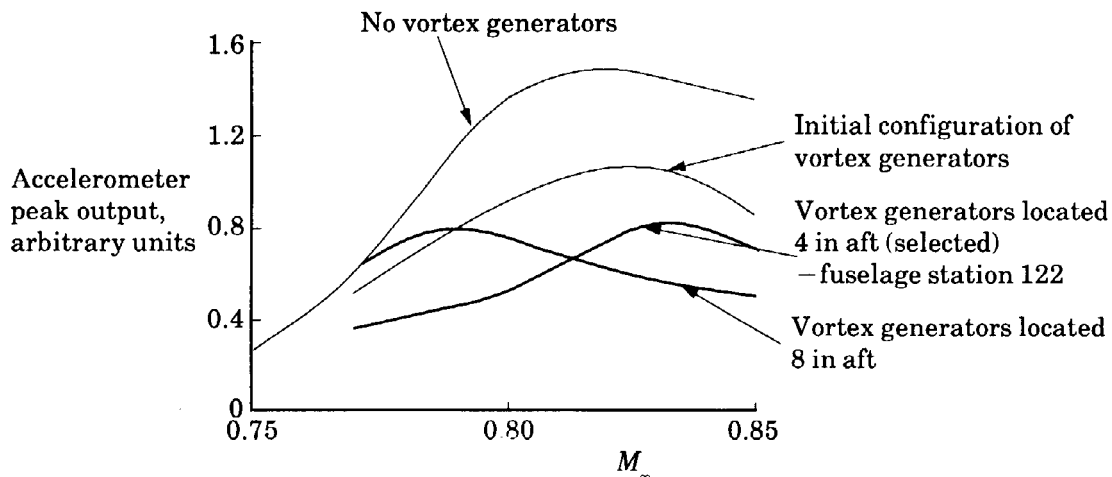


FIGURE A6 Effects of aft locations of vortex generators

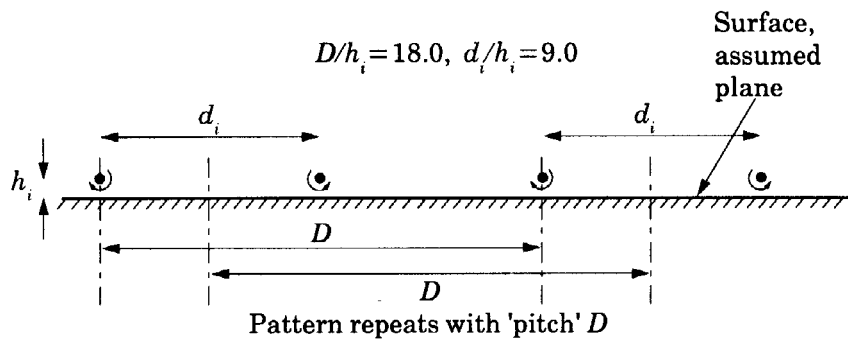


FIGURE A7 Assumed initial locations of vortices in crossflow plane

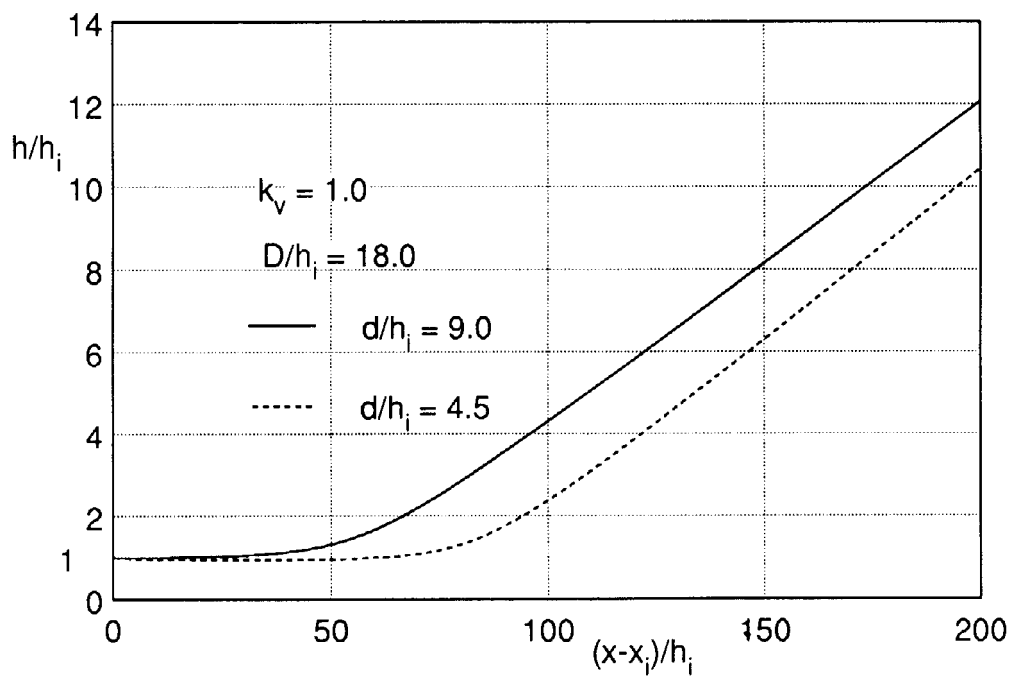


FIGURE A8 Calculated variations of vortex height downstream of vortex generators