Some Aspects of Fan Noise Suppression Using High Mach Number Inlets

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The results of a systematic study on the suppression of transonic fan noise by high velocity inlets or the so-called accelerating inlets are presented. The issues pertinent to the design of such fan-inlet systems are first examined in terms of a simplified model evolved from the well-known Morfey and Fisher theory.[4] These issues are then examined in terms of the aero-acoustic data obtained from three distinct fan test facilities. It is found that the noise reduction due to an accelerating inlet correlates best with the geometric throat one-dimensional Mach number rather than, say, the peak wall Mach number. A typical reduction of 5 PNdb was measured with an inlet operating at a Mach number of about 0.72.

Nomenclature

B = number of blades on the fan
c_o = speed of sound
d = tip diameter of fan
L = overall length of inlet
M = Mach number
M_a = flow axial Mach number at fan face
M_T = fan tip Mach number
M_{throat} = geometric throat one-dimensional Mach number
P = integrated total pressure at a given axial station
r = local radius of inlet cross-section
R_o = fan tip radius
s = rotor blade spacing
Z = normalized shock strength parameter $\Delta p/P_o$ [see Eq. (1)]
γ = specific heat ratio of a gas
η = integrated total pressure recovery of an inlet
λ = sawtooth (shock) wave length
ϕ = wave angle

Introduction

It is an almost paradoxical feature of the transonic isolated rotor, whose noise spectrum is dominated by the blading bow shock pattern, that, unlike the subsonic rotor, a significant reduction in the forward arc radiated sound power can be achieved even with a hard-walled inlet. Such inlets, sometimes referred to as accelerating inlets, are simply hard walled inlets contoured to accelerate the flow at some station ahead of the fan to a sufficiently high, but still subsonic, Mach number. The phenomenon being exploited is the enhanced rate of decay of the rotor bow shock pattern in a flow of high subsonic velocity counter to the direction of shock propagation. The intent is for the average throat Mach number to remain below 1.0 to avoid severe fan operational penalties.

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The mechanism of the rotor bow shock pattern attenuation is viscous dissipation but can be conveniently explained in terms of repeated interaction of the bow shocks with expansion fans. This is depicted schematically in Fig. 1, where for the purpose of illustration, we have used a cascade plane model with a leading edge centered expansion fan system (the broken lines). The picture is simplified but representative of fans once the relative Mach number of the flow with respect to the blade tips exceeds unity. For the typical blade, the initial waves of the expansion fan intersect the bow shock, bend it, and weaken it. The process is enchanced, as illustrated in Fig. 1, by the intersection of the trailing expansion waves of a preceding blade with the shock. As the upstream Mach number is increased, bow shock pattern bends even more sharply further intensifying these interactions. The accelerating inlet exploits this increased attenuation rate by increasing the flow Mach number at some station ahead of the fan, to a high but still subsonic value.

In the aforementioned picture it has been implicitly assumed that all the blades are exactly alike, and the upstream bow shock pattern repeats itself exactly once every blade spacing in the circumferential direction. Actually, in practice, due to manufacturing tolerances there are small variations blade to blade which results in a nonuniform shock strength pattern upstream of the rotor. The picture in the immediate vicinity of the rotor remains almost unchanged, with pressure peaks and valleys repeated at an interval of a blade spacing.

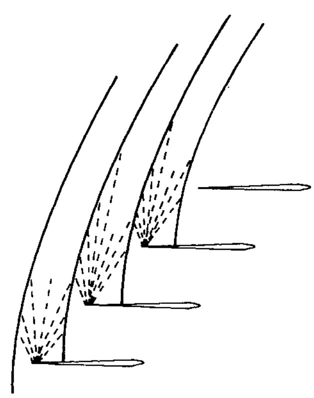

Fig. 1 Schematic of a rotor bow shock pattern (solid and arcs) interacting with tip centered expansion fan systems (broken lines).

However, since the shocks now propagate with varying speeds, some axial distance ahead of the rotor the sawtooth pattern becomes irregular and repeats on the whole only once in every revolution. Thus the fundamental harmonic is now once per rotor revolution, and the spectrum contains harmonics of the shaft frequency including the one at the blade passing frequency. The nonlinear propagation process together with the nonuniform shock structure results in the transfer of energy from the blade passage frequency (BPF) into subharmonics representing multiples of the shaft frequency. Indeed it generally happens that the spectral signature of the transonic fan is dominated by these shaft harmonics sometimes referred to as multiple pure tones or MPT's for short.

The concept of the accelerating inlet has been known for sometime, and one early variant of it was reported on by Chestnutt and Stewart.[1] While not strictly an inlet, they found up to 25 db noise reduction with the elimination of pure tones by allowing the flow in the inlet guide vanes (IGVs) to approach choking conditions. However the noise reduction was accompanied by a significant reduction in the compressor efficiency. More recently, there have appeared the papers by Klujber[2] and Mathews and Nagel.[3] Klujber's paper presents the results of an anechoic chamber study of sonic inlets using several different concepts including the translating centerbody. The latter paper by Mathews and Nagel[3] is of particular interest since it is the first published piece of work to attempt an analysis of the accelerating inlet problem. Indeed they anticipate some of our results, especially those in extension to the Morfey and Fisher[4] supersonic rotor acoustic analysis.

In this paper, the results of a largely experimental study of accelerating inlets are presented. In the process such questions as how effective are the accelerating inlets in attenuating various shaft orders of MPT's including the one at the blade passing frequency (BPF) will be examined. To cast the results into a better perspective, a simple heuristic extension of the Morfey-Fisher[4] acoustic theory shall be developed to provide a framework for understanding the experimental results. The analysis will account for both a varying axial flow Mach number and the outer cowl wall radius distribution. Prior to developing this heuristic modification, the approximation to MPTs by comparisons with a more detailed theory shall be justified. The test results were based on several different fan and inlet test configurations, thus some generalizations of considerable interest in the acoustic design of such inlet systems accrue. Among these results, it will appear that the noise reduction of an accelerating inlet correlates best with the "throat one-dimensional" Mach number and that for a given axial through flow, larger insertion losses occur for the higher tip-speed fans.

II. Analysis of Rotor Bow Shocks Propagating Into an Axially Nonuniform Flow

To better understand the experimental results and offer a rational basis to extend those results, a mathematical model for the decay of shock-originated MPTs through a region of varying axial Mach number shall be attempted. In essence the procedure involves the tracing of the rotor bow shock trajectories, given the appropriate flow and geometric information. In an effort to render the problem tractable, accuracy shall be sacrificed in favor of simplicity and our attention will be concentrated on trends.

The transonic rotor acoustic problem has been treated in several recent papers. A particularly attractive analysis, in view of its simplicity, is due to Morfey and Fisher.[4] The analysis is a cascade plane analysis restricted to the cascade equivalent of a uniform rotor (without any blade to blade variations). However, in view of the nearly two-dimensional nature of the rotor tip shock behavior, it clearly represents a plausible model for a cylindrical inlet. It would therefore also be an attractive model on which to base the propagation of MPTs in ducts with variable Mach number. The alternative of a truly three-dimensional propagation faces an almost impossible task. This poses two issues which need to be resolved: first, even in the cascade plane, how well does the Morfey-Fisher model represent propagation of MPTs, and secondly what effect does the variable area flow duct have on the decay rate of the sawtooth We cannot answer the second question, for which we will supply a posteriori heuristic justification, but we can examine the first question.

Although restricted to a cascade plane approximation, Kurosaka[5] developed an analysis capable of tracing the evolution of MPTs from a started rotor given the blade-to-blade nonuniformity. The shock trajectories are constructed geometrically accounting for repeated interactions with expansion wavelets using the oblique shock relations. The model suggested that the strongest mechanism for the generation of MPTs of a rotor with attached shocks is the blade to blade stagger variations. This was recently extended to include detached shocks[6] using Moeckels results.[7] This latter analysis now takes into account throat area variations due to stagger errors, leading to varying spillage flow and consequently the shock strength and wave angle. Omitting calculational details, typical calculations resulting from this analysis are presented in Fig. 2.

Figure 2 depicts the evolution of MPT's of a 53-bladed rotor operating at a relative Mach number of 1.135 with a nominal stagger angle 65°. The topmost line, with the encircled calculated points, represents the uniform rotor. Below this are the calculated amplitudes of selected shaft harmonics (MPT's) as they evolve. The most striking feature of this calculation is the implication of a "frozen" MPT pattern after an evolutionary process lasting over several blade spacings ahead of the rotor. Physically this is perhaps not surprising since in the farfield the shocks should, after they have sufficiently weakened, asymptote to the Mach lines. This feature was verified experimentally and is to be reported on in detail in an upcoming paper.

The implication of the frozen pattern is two fold. First, within a relatively short distance of the rotor, the decay rate of all shaft orders becomes the same as the Morfey-Fisher uniform rotor rate. This suggests that Morfey-Fisher model can be profitably employed to examine some aspects such as decay rates, of even MPT's, provided the calculation is interpreted sufficiently far from the rotor. A second, and perhaps more startling, result is that one therefore expects, barring any change in the evolutionary process, that the accelerating inlet will attenuate all shaft orders more or less uniformly. This is a rather important point since in linear acoustic liners, attenuation rate is a strong function of frequency.

We return now to the Morfey-Fisher acoustic theory and modify it first to account for a varying axial Mach number distribution, which is designated by M_x, x being the axial direction. To relate the current calculations to the thorough going discussion of that paper, a large part of their nomenclature will be adopted. Hence designate by Z the normalized shock strength parameter $\Delta p/p_o$, and t is the time, and Δp the pressure rise across the shock, and p_o the undisturbed static pressure.

The Morfey-Fisher theory is devised around the well-known result of nonlinear acoustics for the decay rate of a repeated sawtooth waveform propagating in one dimension

$$(d(1/Z)/d(c_o t/s)) = (\gamma+1)(s/2\gamma\lambda) \tag{1}$$

where c_o, λ, γ, s are respectively the undisturbed speed of sound, sawtooth wave length, gas specific heat ratio and anticipate the application to rotors by normalizing against the rotor blade spacing s. Recall now that if M_u is the upstream "phase speed" of the shock system in the cascade plane (see Fig. 3), the time of passage of the shocks to traverse a distance x may be related to M_u by

$$(c_o M_u t/s) = x/s \tag{2}$$

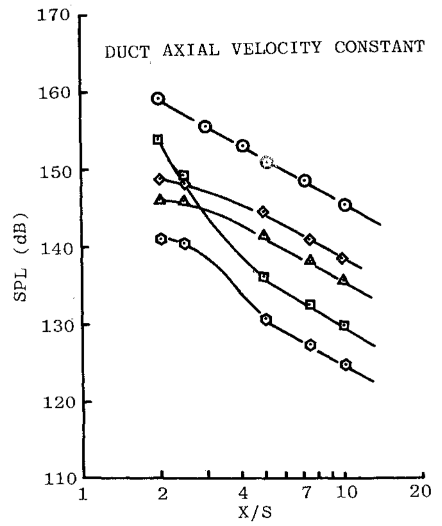

Fig. 2 Evolution of a rotor (cascade) MPT pattern using the nonlinear model of Kurosaka.[6] $B=53$ blades. $M_\infty = 1.135$. Blade to blade spacing, $S=1.09$ in. Incidence for nominal stagger = 1.05°. Nominal stagger 65°. x = axial distance; ○ = uniform rotor; ◇ = 25th shaft harmonic; △ = 13th shaft harmonic; □ = BPF; ⬡ = 41st shaft harmonic.

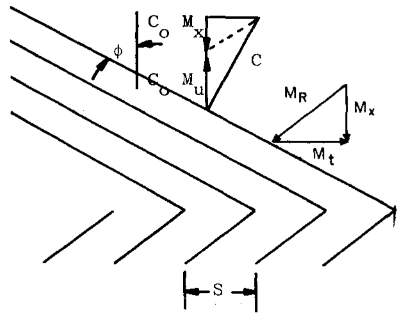

Fig. 3 Kinematic picture of upstream bow shock pattern. M_x is axial velocity and M_u axial phase speed.

Since the axial Mach number is varying with x, so is M_u. Moreover, for the rotor we may write

$$\lambda = s\cos\phi \text{ and } s = 2\pi R_o/B$$

where, as depicted in Fig. 3, ϕ is the wave angle with respect to the axis. Now within the acoustic approximation, the wavefront travels at the speed of sound. Hence, with reference to Fig. 3 we may write

$$M_u = \sin\phi - M_x$$

From the geometry of the flow the wave angle relates to the velocity parameters by the relation

$$\sin\phi = [M_x + M_t(M_R^2 - 1)^{1/2}]/M_R^2$$

and consequently obtain

$$M_u = ([M_x + M_t(M_R^2 - 1)^{1/2}]/M_R^2) - M_x \tag{3}$$

where M_t is the local tangential speed of the wave pattern, and M_R is the relative speed given as $(M_x^2 + M_t^2)^{1/2}$.

The objectives of the steps between Eqs. (2) and (4) is to replace the time element in the decay law, Eq. (1), with the axial distance parameter x. In the uniform flow case of Morfey-Fisher, this involved the direct substitution previously spelled out. In the case of the nonuniform flow such a substitution is not strictly correct. However, provided the axial velocity gradient is sufficiently shallow compared with the sawtooth wavelength, the decay law should not change up to the order of the approximation, and such a formal substitution is a reasonable approximation. In the case of MPTs, partial justification for such a step can be offered in terms of the frozen pattern, suggesting that the proper length parameter does indeed remain the initial sawtooth wavelength rather than one based on, for example, low order shaft harmonics. Applying the chain to M_u, obtain for this modified decay law

$$d\left(\frac{1}{Z}\right) = \frac{\gamma+1}{2\gamma}\,\frac{1}{\cos\phi}\left[\frac{M_u - x(dM_u/dx)}{s\,M_u^2}\right]dx \tag{4}$$

Hence, we may integrate Eq. (4) to obtain

$$1/Z - 1/Z_0 = \frac{\gamma+1}{2\gamma}\int_0^{x/s}\frac{1}{\cos\phi}\left[\frac{M_u - x(dM_u/dx)}{M_u^2}\right]d(x/s) \tag{5}$$

Z_o being the initial shock strength. Note that we have assumed M_u to be a function only of x. The aforementioned integral may now be evaluated numerically once the axial distribution M_x is specified as a function of x. In the subsequent discussion we shall need to identify the velocity triangle at the rotor or the cascade, therefore designate the values of M_x and M_t at the rotor or cascade plane by M_a and M_T, respectively. Suppose one were to ignore all other effects of varying duct cross-sections except to note that we need preserve the frequency. Then, restricting attention to the streamline passing through the rotor section represented by the cascade plane, the preservation of the frequency is equivalent to the statement (for the outermost streamline):

$$M_t = M_T r/R_o$$

where r is the local radius, R_o the rotor tip radius. If now $r=r(x)$, introducing the aforementioned in Eq. (5) we may again numerically carry out the integration indicated. Physically, this is the observation that through the variable area duct, if the variation is large enough, M_R at some station might become less than 1.0 and the signal suffers cutoff. In actuality this is most likely to occur near the rotor cutoff speed of $M_T^2 + M_a^2 \sim 1$, and its effect for a given inlet is to shift the cutoff to a higher relative Mach number.

Returning to Eq. (6), let us examine its implication for an inlet whose length L is the same as the fan diameter d. We shall also employ this constraint experimentally since the limitation is of practical value. In each case the integration will be carried up the the "throat." Consider first the effect of varying the axial Mach number schedule.

A calculation representing three different axial Mach number schedules is presented in Fig. 4. Several other schedules have also been tested but these summarize the essential result. Since we are interested in the effectiveness of accelerating inlets, the results are presented in terms of a calculated noise reduction. In each case, the rotor, tip Mach number, number of blades, axial Mach number into the fan (=0.55) were held fixed. In each case the throat Mach number was kept at 0.75. The reference was taken to be uniform flow of Mach number 0.55. These situations are illustrated in Fig. 4. The calculated noise reductions are given to the left of the applicable velocity schedules. It is interesting to note the narrow spread resulting, especially between Cases 2 and 3. The significance of this result is obvious, since the major weakness of accelerating inlets is the loss penalty paid on the diffuser performance downstream of the throat. Indeed a Mach number schedule such as Case 2 introduces severe design difficulties in trying to accommodate a reliable diffuser.

On the basis of the previous calculations, we shall restrict our attention to linear axial profiles in the calculations to follow. The object of these calculations is to test the effects of various fan parameters when coupled to an accelerating inlet.

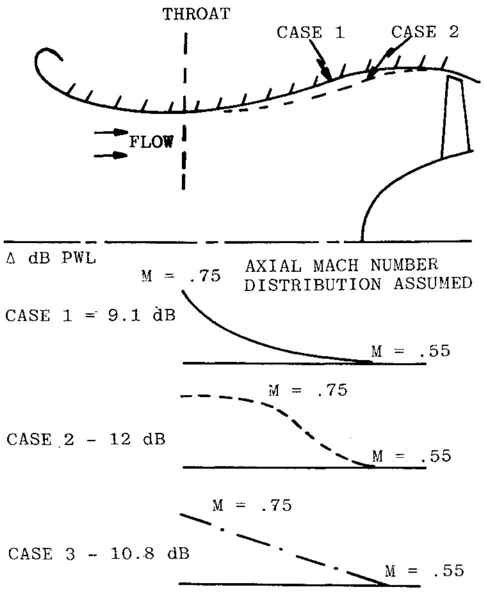

Fig. 4 Theoretical evaluation of the axial Mach schedule influence. The lower straight line represents the "baseline" inlet assumed.

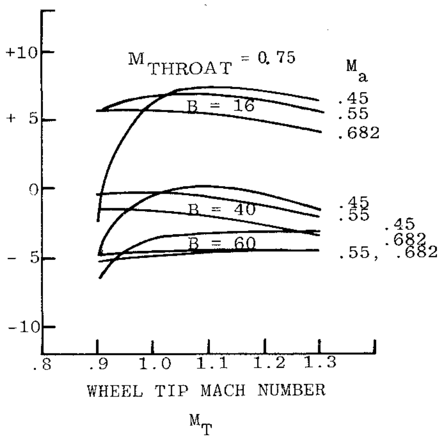

Fig. 5 Calculated radiated sound power for representative fan diameters. B is the blade number and M_a the flow axial Mach number into the fan.

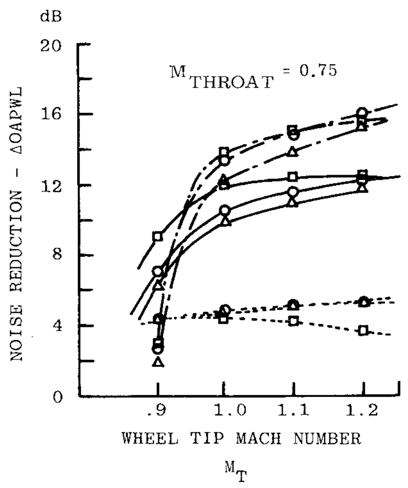

Fig. 6 Calculations based on Morfey-Fisher model modified to account for axial Mach number distribution.

B		M_a
△ = 16	– – –	0.45
○ = 40	——	0.55
□ = 60	-------	0.682

For this reason all of the remaining calculations will be carried out for a throat Mach number of 0.75. Figure 5 presents the relative radiated sound power for fans of varying velocity triangles and three representative blade numbers. The trend of these calculations is to indicate lower radiated sound power as the blade number increases and that except near cutoff there is not a strong influence of the tip Mach number in the (practical) range tested.

Calculations for the noise reductions referenced against a uniform flow inlet whose Mach number is everywhere M_a are presented in Fig. 6. The curves now group with respect to the flow axial Mach number M_a into the fan, and the influence of blade numbers is weak. More importantly, near cutoff there is a strong dependence on the tip Mach number. This latter suggests that for a given throughflow, the accelerating inlet would be more effective on higher tip speed fans. The dependence on the difference between the throat and fan axial Mach number is more obvious, so that as this difference increases, noise reduction increases. Briefly in summary then, we expect the accelerating inlets to be uniformly effective across the frequency spectrum and that larger reductions in noise would accrue to the higher speed fans.

III. Diffuser Design Considerations

The central issues of this paper are acoustic, and the critically important concern of the inlet diffuser design has been mentioned only in the passing. Nevertheless, the feasibility of the accelerating inlets hinges crucially on the ability to minimize the total pressure recovery penalty for inlets constrained to an $L/d \lesssim 1$.

The design procedure was based on the Stratford's separation criterion[8] and an inviscid streamtube curvature calculation routine devised by Keith et al.[9] The particular feature of Stratford's model which was extensively exploited was that it represented a nearly lossless (zero wall shear) flow, except for mixing losses. From the point of view of diffuser design this suggested that instead of a straight walled conical diffuser, a preferable concept would be rapid diffusion early, followed by reduced rate of pressure recovery in the latter part. Such an approach would reduce the skin friction loss. This is plausible from simple boundary-layer considerations, since the initially thin boundary layer in the diffuser can sustain high adverse pressure gradient without separation.

IV. Experimental Program for Accelerating Inlets

A. Experimental Facilities and Test Outlines

This paper will summarize test data from three fan test vehicles. These are briefly described in Table 1. The first two of the three fans were tested in anechoic chamber facilities and the third at an outdoor site. The first, or the high-speed fan was tested at the General Electric Corporate Research and Development Center (CR&D) in the recently installed anechoic fan test facility. The special feature of this facility is that the chamber walls, in addition to including GP-2 Sound-

Table 1 Test fan data

	GE CR&D[a]	NASA Langley	GE-½ scale fan
Fan diam.—in.	20	12	35
100% fan tip speed—fps	1424	1310	1020
100% fan flow rate/unit frontal area—lbm/ft^2 sec	40.6	44.6	39.9
Number of blades	44	19	16
Number of OGVs	86	30	60
Hub to tip ratio	0.5	0.5	0.65
Inlet guide vanes	None	None	None
Design Pressure ratio	1.57	1.33	1.25

[a] Corporate research and development.

form wedges, are made porous so that the flow approximates a true sink flow into the fan.

The test program for the GE CR&D tests was structured around four inlets. Of this, inlet 1 was designed to achieve a nearly constant Mach number axially and was used to represent the baseline against which the accelerating inlets were assessed. This inlet, inlet 1, was also contoured and in terms of the flow path it is not too dissimilar to the current generation of high bypass ratio fan inlets. Inlets 2 and 3 were similar to each other in design except for the peak throat Mach numbers, inlet 2 Mach numbers being the highest. The region of high Mach numbers in these inlets was relatively short. The purpose of these inlets was to provide a cross-check on the influence of the Mach number. Inlet 4 had a peak Mach number similar to inlet 3, however the high Mach number region was stretched out over a longer region. The performance of this inlet relative to inlet 3 was to test the influence of the axial Mach number schedule. The wall contours and the wall Mach number distributions of these inlets are illustrated in Fig.7 for a 90% inlet flow condition. The wall Mach numbers were closely verified by aerodynamic data. Inlet 3 contour is omitted since it was hard to clearly distinguish it in drawing the figure. It should be noticed that all inlets had an L/d of 1.

The joint GE-NASA Langley tests were conducted in the Langley anechoic fan test facility using a NASA 19 bladed 12 in. fan. Three inlet configurations were tested including one which served as a baseline low Mach number inlet. The low speed fan was a 36" experimental fan which was tested outdoors. Tests of this fan were carried out with a single accelerating inlet. The baseline of these tests were run with a low Mach number bellmouth. Care was taken to suppress the fan exhaust noise from interfering with the tests by using a long acoustically treated annular duct containing an acoustically treated splitter.

In reporting the data to be examined next, we shall present both the model data and PNL data corresponding to the model data scaled up to full scale of the current generation of single-stage fans. While the data trends are the same in both cases, the latter data are perhaps more meaningful from the point of view of applications.

B. Experimental Results

The experimental results can be cast into two subgroups, aerodynamic and acoustic. The primary objectives of this paper are the acoustic issues, but as pointed out earlier the feasibility of the accelerating inlet concept hinges on the aerodynamic performance. More precisely the concern is the penalty to be paid for the overacceleration of the flow and whether the penalty can be minimized. Consider first sample aerodynamic results.

There are a number of questions related to the aerodynamic performance of the accelerating inlets. These can be partially summarized by examining the total pressure recoveries. The definition of inlet (or diffuser) recovery η we use is the integrated total pressure profile at the fan referenced against the upstream constant total pressure integrated over the same cross-section.

$$\eta = [2\pi \int_{r_{\text{hub}}}^{r_{\text{tip}}} P(r) r \, dr] / [P_o \pi (r_{\text{tip}}^2 - r_{\text{hub}}^2)]$$

where P_o is the ambient static pressure.

The total pressure measurements were recorded with a Kiel probe traversed continuously at a sufficiently slow speed across the inlet annulus. The bold arrow in Fig. 7 indicates the point of the total pressure recovery measurements.

Examine the measured recoveries of inlets 1, 2, 3, 4 shown graphically in Fig. 8. The data have been plotted against the one-dimensional throat Mach numbers. The data have separated into two distinct curves, and the recoveries of inlets 1, 2, 3 appear to fall along a single line. Recall the definitions of these inlets, and note that increasing the throat Mach number simply shifts downward on this curve. Yet note that the recoveries remain high and indeed are of the type typically measured in a flight configuration. This suggests that the major losses of inlets 1,2, 3 are the skin friction component, and the profile mixing losses are minimized. These recoveries, of course, do not account for the loss in fan performance due to thickened boundary layers.

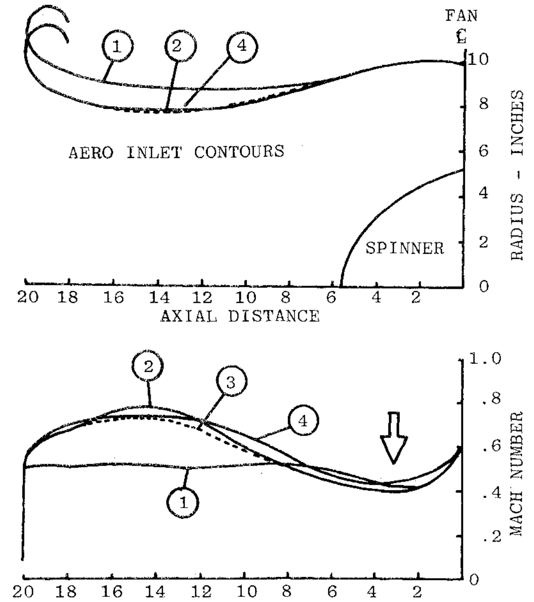

Fig. 7 GE-corporate research & development high speed fan test inlet contours and wall Mach number distribution. 90% design air flow rate depicted. Encircled numbers correspond to appropriate inlets. Inlet 3 contour omitted. The bold arrow indicates the station at which the inlet recoveries were measured.

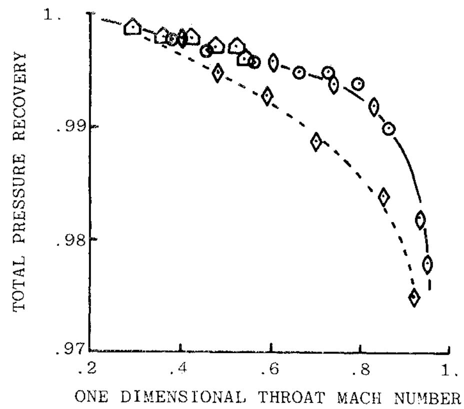

Fig. 8 Aerodynamic behavior of accelerating inlets—total pressure recovery correlated with one-dimensional geometric throat Mach number. -inlet 1 (baseline); -inlet 2; ○ -inlet 3; ◊ -inlet 4. All inlets $L/d = 1$. Inlet 4 has an extended high Mach number region and the shortest diffuser.

Inlet 4 referenced to inlet 3 is suppose to test the influence of a severe axial Mach number distribution. Not surprisingly the recoveries of this very short diffuser design have suffered in relation. The acoustic benefits will not be found to justify such a penalty.

We turn our attention now to the acoustic results. Figure 9 summarizes the behavior of the accelerating inlets, this representing the overall sound power radiated by the 20 in. rotor in the high-speed fan tests (GE anechoic chamber). The figure brings out several interesting features. First and foremost, the accelerating inlets do induce a net noise reduction, which effect comes into play once the relative Mach number of the rotor goes transonic. However, the inlets do exact a penalty in the sense of enhanced inlet turbulence—rotor scattering noise due to the thickened boundary

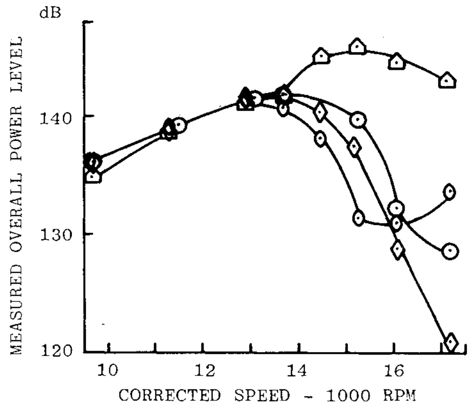

Fig. 9 Measured radiated sound power vs wheel speed for the GE-corporate research & development tests. o -inlet 1 (baseline); ⟨ ⟩ -inlet 2; -inlet 3; -inlet 4. Note the breaks in curves for inlets 2 and 3 which appear to be traceable to effects of unsteady shock boundary-layer interaction in the inlet throat.

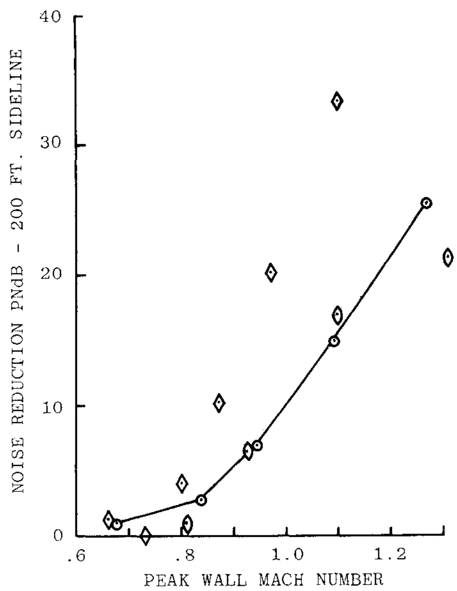

Fig. 10 Noise reduction for the GE-corporate research & development tests. Peak 200 ft sideline PNL tested for correlation with peak wall Mach number. All data referenced to inlet 1. O -inlet 2; o -inlet 3; ◇ -inlet 4.

layers. This can best be seen in the subsonic part of Fig. 9. Thus the noise reduction of the accelerating inlets measured is the net of the shock propagation attenuation and the increased broadband generation. This can also be partly seen in a spectral evaluation such as given in Fig. 13.

The curious breaks in the curves for inlets 2, 3 at high-speed operation appear to result from an unsteady shock boundary-layer interaction in the throat region. The severest problem being experienced by the highest Mach number inlet, inlet 2. The far field spectrum at this point shows a strong low frequency peak (see, for example, Fig. 13).

In the earlier calculations and the discussions to this point, we have not clearly isolated the parameter of importance to the noise reduction of an accelerating inlet. On the one hand, the argument of two-dimensionality of the shock pattern seems to suggest a parameter which is some measure of the wall Mach number. On the other hand, the fan source distribution is radially nonuniform and, the thickened boundary layers together with turbulent diffusion, would tend to further refract a part of the acoustic energy away from the walls towards the inlet axis. This question can be settled by plotting the noise reduction data against both the peak wall Mach number and the geometric throat one-dimensional

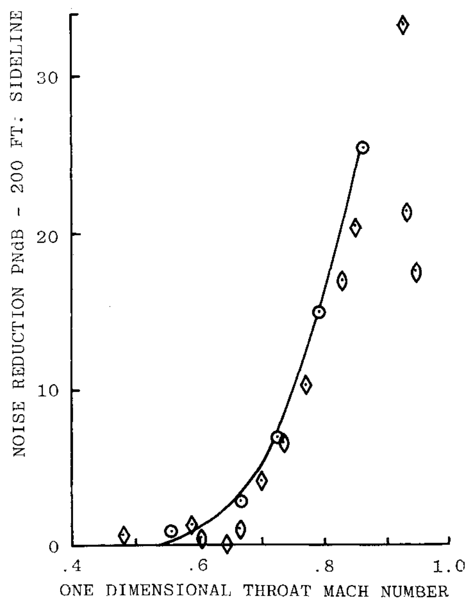

Fig. 11 Noise reduction for the GE-corporate research & development tests in terms of Peak 200 ft sideline PNL. Correlation with respect to one-dimensional geometric throat Mach number. All data referenced to inlet 1. O -inlet 2; o -inlet 3; ◇ -inlet 4.

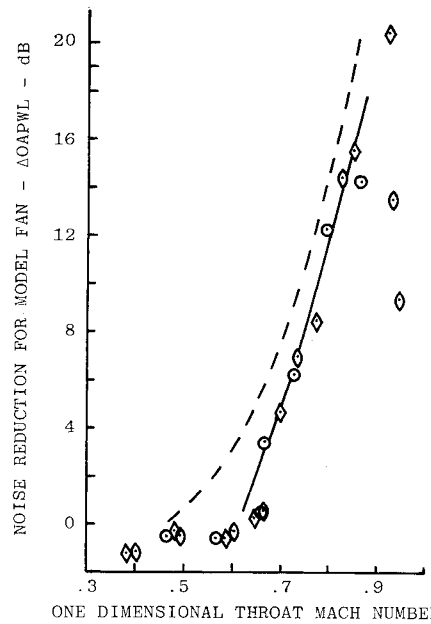

Fig. 12 Noise reduction data for the GE-corporate research & development tests (high-speed fan) in terms of radiated sound power level. O -inlet 2; ◇ -inlet 3; O -inlet 4. —calculations using inlet 3 Mach number distribution. All data reference to inlet 1.

Mach numer as in Figs. 10 and 11. These data are plotted in terms of 2000 ft sideline noise reduction measured in the peak PNdb sense as referenced against the baseline in inlet 1. Similar results are evident in a plot based on OAPWL insertion loss (Fig. 12). Clearly, the noise reduction appears best to correlate with the simple one-dimensional geometric throat Mach number. The breaks in the curves are again due to what seems to be the unsteady shock boundary-layer interaction in the throat.

The issue of the axial Mach number schedule may also be examined with the previous data. The calculations discussed in Sec. II suggested a marginal benefit in going from Case 3 (approximating inlets 2, 3) to Case 2 (approximating inlet 4). This was of course within a constraint of $L/d \leq 1$. Not even this small benefit is evident in these data. The reason appears to be a combination of a possible change in the source charac-

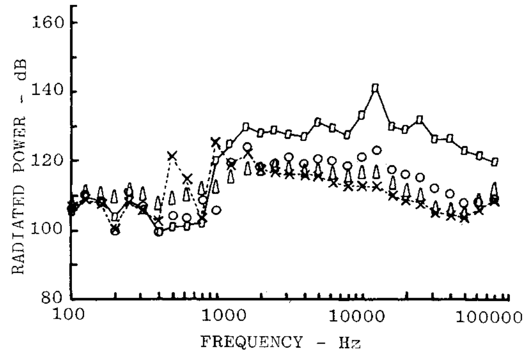

Fig. 13 1/3 octave radiated power level spectra for high speed fan operating at 100% design tip speed (GE corporate research & development tests). □ -inlet 1; × -inlet 2 (high Mach number); ○ -inlet 3 (intermediate Mach number); △ -inlet 4 (stretched high Mach number region). Note these spectra do not correspond to the same throat one-dimensional Mach number.

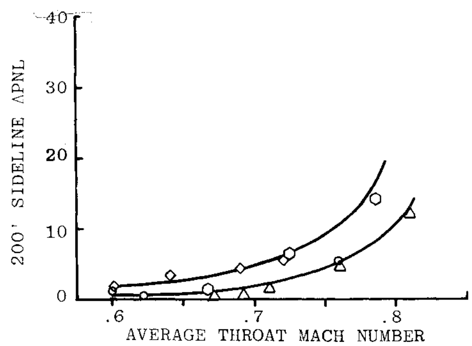

Fig. 14 Composite of accelerating inlet noise reduction data—influence of wheel tip speed. ○ -½ scale fan data; ◇ -Langley high tip speed; △ -Langley low tip speed; ◊ -GE-corporate research & development tests (high-speed fan).

ter, refraction by the thickened boundary-layer (especially in inlet 4). The net practical result, within the length constraint employed is that there isn't a measurable benefit of the varied axial velocity profile. The implications of this in terms of diffuser design are obvious: One need not strain the diffuser considerations and achieve all or most of the noise reduction by employing the simplest velocity schedules, maintaining simply a sufficiently high one-dimensional throat Mach number.

As already indicated, there is no change in the conclusions if one examines the results in terms of the integrated radiated sound power. This is illustrated in Fig. 12 in which the noise reduction measured in terms of OAPWL referenced against inlet 1 are plotted. With the issue of the reference parameter settled, one may also test the calculations based on the modified Morfey-Fisher model discussed earlier. Calculated noise reductions based on that two-dimensional theory are also shown in Fig. 12 as the broken line. These calculations actually correspond to inlet 3 using a linear axial profile and one-dimensional Mach numbers. Within the limits of expected accuracy, there is little effect of changing the distribution to correspond to inlets 2 or 4. Despite the extreme assumptions involved, there does appear to be a measure of agreement and certainly a good one on the trend. The calculations do, however, generally overestimate the insertion loss. This is not surprising since the two-dimensional calculations based on the tip parameters cannot reproduce the actual three-dimensional distributions. One could use the "strip theory" approach, however this requires the definition of fictitious acoustic stream tubes. Considering the gross limitations of the theory the model appears to represent the phenomenon quite well. It should also be noted that the test inlets appeared to be effective over the entire radiation quadrant although in some instances their effectiveness appeared to be somewhat less on the axis than on the off axis angles.

One of the major issues outlined in the earlier discussion of MPT propagation was the spectral effectiveness or frequency dependence of accelerating inlets. The issue is examined for a sample case in Fig. 13 which was obtained in the GE anechoic facility with the high-speed rotor. Plotted are the 1/3 octave power level spectra. Except for the blade passing frequency and the lowest frequencies (below 1000 Hz), the distributions for the accelrating inlets are generally parallel to the baseline. This appears to bear out the theoretical prediction of frozen MPT patterns. Note also that the frozen MPT patterns were verified independently by wall Kulite measurements further supporting the previous conclusion.

Finally there is the question about the influence of fan speed on the effectiveness of accelerating inlets. The calculations of an earlier section suggested that higher noise reduction should occur for the higher speed fans compared with the low speed fans. To test this we present the composite of the noise reduction data from all three fan test systems in Fig. 14. It would appear, that within the scope of the limited data examined, the predicted influence of the fan tip speed in the near range beyond cutoff is reproduced.

V. Conclusions

In this paper the effectiveness of the accelerating inlet concept in suppressing transonic rotor noise has been examined. In terms of the issues we set forth in the beginning it would appear that: a) The noise reduction of the accelerating inlets best correlates with the one-dimensional geometric throat Mach number. b) There is not a measurable influence of varying axial Mach number schedule within an $L/d \leq 1.0$. c) The accelerating inlets are effective more or less uniformly across the MPT spectrum. d) Accelerating inlets appear to be more effective on higher speed fans, near cutoff, than the low-speed fans. e) The penalty paid in terms of the inlet recovery can be minimized and recoveries in excess of 99% over the operating range can be achieved. In addition to the aforementioned experimental conclusions, a simple theory by which to explain the behavior of accelerating inlets has also been presented and a surprisingly good agreement between theory and experiment especially in terms of trends has been found.

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