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THE PROBLEM OF TURBINE NOISE IN THE CIVIL GAS TURBINE AERO ENGINE

by

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ABSTRACT

The significance of noise from the turbine of a turbojet engine is examined in the context of the other component noise sources. A study is made of its generation, propagation and radiation, and the question of suppression is related to current research work.

INTRODUCTION

During early work on the noise problems associated with jet engines in use with civil aircraft operators attention was centred almost entirely on the jet as the main noise producer. Over the years however it has become quite clear that the changing design philosophy in going to the use of higher and higher bypass ratios has thrown up both the compressor and the turbine as significant, and within the next few years, the predominant noise producers. When the coming generation of high bypass ratio turbofan engines are in service this difference in source importance will be strikingly apparent, although today it is possible to cite clear differences caused by more modest changes in bypass ratio. This is illustrated in figure 1, which is a plot of Perceived Noise Level against time for four flyovers of two aircraft. These aircraft would normally be identical

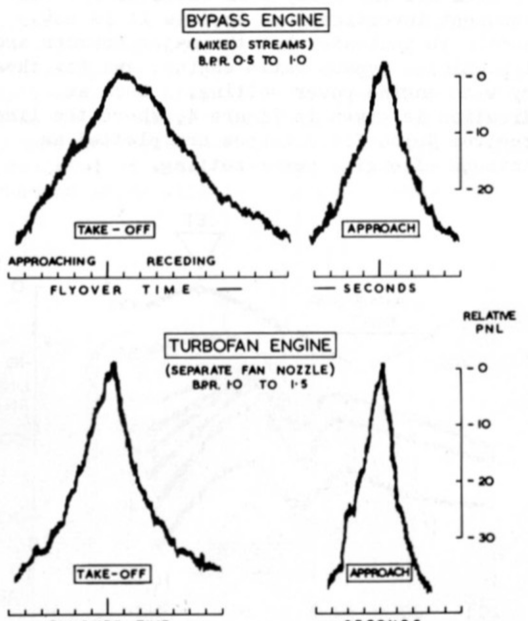

FIGURE 1. COMPARISON OF TAKE-OFF & APPROACH FLYOVER TRACES. LOW BYPASS RATIO MIXED ENGINES & UNMIXED TURBOFANS IN SAME AIRFRAME.

if it were not for their engines having slightly differing bypass ratios and dissimilar nozzle geometry. The actual bypass ratio difference is from just less than 1 on the one hand to about $1\frac{1}{2}$ on the other, and the flyovers chosen are directly comparable in that there is both an approach and takeoff trace presented for each aircraft.

In the case of the lower bypass ratio installation, the engines of which have relatively high jet velocities and fully mixed nozzle configurations, the over-riding noise contribution is caused by the mixing of the jets with the atmosphere. This is clearly evident in the takeoff trace where the post overhead peak is seen to be an extended exposure typical of the radiation pattern of jet noise, whereas the equivalent trace for the higher bypass ratio engine shows a much quicker decay of noise versus time. In the approach traces there is not such an obvious difference, since the jet noise of even the low bypass ratio engines is reduced sufficiently to reveal both a pre-overhead and post-overhead peak, which we now know is caused by the blading of both the compressors and the turbines. The higher bypass ratio installation exhibits a rather sharper peak, since the compressors are of a high tip speed high relative velocity design, and as such are the over-riding noise producers. Furthermore this noise is allowed to radiate directly through a short unmixed fan outlet nozzle.

These are but two of many in-service engine types, each of which has a turbine noise content which, depending on the design philosophy and actual power setting, may or may not affect the overall level. To consider more exactly the significance of the noise caused by the turbine on these and other engines the subject will obviously need amplification, and it is worth devoting a little time to a discussion of the way in which turbine noise was first noted and isolated.

SOURCE DETECTION & DEFINITION

As already pointed out, 10 or 15 years ago jet noise was the major problem with civil aero engines, and an intensive research and development programme was carried out by the major aero-engine manufacturers to establish both how the noise was generated and what steps could be taken to reduce it. (A number of publications exist dealing solely with jet noise and, for further consideration, reference works may be consulted elsewhere, including the selection quoted at the end of this paper). Towards the end of the development programme on jet noise suppressors it became clear that at low power settings on even pure jet and low bypass ratio engines some other sources of noise were contributing to the overall engine level. Since these other sources were not obeying accepted jet noise laws, and furthermore were not responding to normal external jet noise suppression techniques, it was assumed that they were the result of generating mechanisms contained within the engine casing.

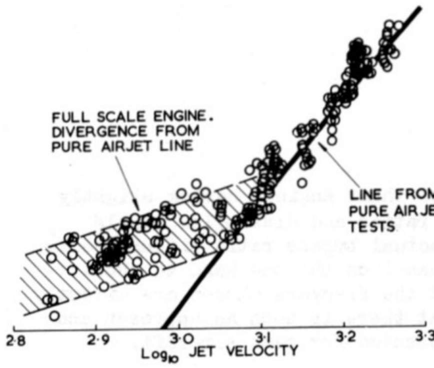

FIGURE 2. VARIATION OF POLAR PEAK REAR ARC NOISE LEVEL WITH JET VELOCITY. FULL SCALE ENGINES COMPARED WITH PURE AIRJETS.

Figure 2 illustrates how, in plotting the overall noise level from a series of pure jet and low bypass ratio engines, the observed laws of jet noise generation were not accounting for the measured noise levels at low jet velocities. The correlating parameter in this illustration is that usually accepted, the velocity of the jet relative to the outside environment, and established theory suggests that the noise level variation with this parameter should follow a power law with an index of somewhere in the region of 8. The theory is backed up by small scale pure jet test evidence which allows the construction of the experimental jet line indicated. Clearly at lower jet velocities there is a significant divergence from this law in the case of the full-scale engine results. For a long time it was felt that this divergence could be accounted for by errors in the theory, but closer examination of the results revealed other differences that led to the conclusion that there was one or more other noise sources contributing to the overall level at the lower velocities.

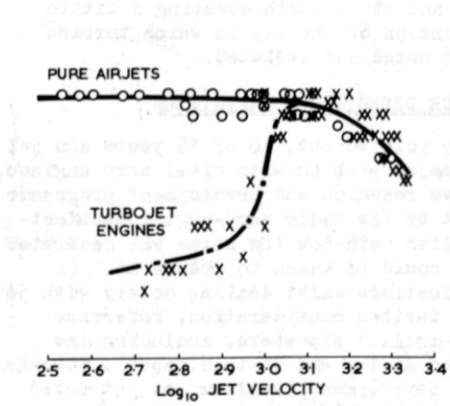

FIGURE 3. VARIATION OF ANGLE OF RADIATION OF REAR ARC PEAK NOISE LEVEL WITH JET VELOCITY. FULL SCALE ENGINES COMPARED WITH PURE AIRJETS.

One significant piece of evidence arose from consideration of the angle of radiation of the peak noise. It was found that in plotting angle of peak radiation against jet velocity the pure air jet and full-scale engine results diverged significantly, and this is illustrated in figure 3. Clearly at high jet velocities the engines and the airjets agree in radiation angle, just as they do in peak intensity, but as the reduced velocities corresponding to the area of divergence in peak level are approached, there commences a significant divergence in peak angle. This is a gradual process, until the point is reached where the anticipated intensity of the jet noise is well below that of the observed results. Here the peak angle of radiation from the full-scale engines settles to an almost constant difference of something like 40° from the pure air jet angle.

Another piece of evidence was that exposed by close inspection of the spectra associated with actual engines at very low velocities. This revealed that there was a clearly defined discrete tone content resulting from turbine blade interaction, and therefore it was concluded that the turbine was probably a major contributor to the overall noise level. It was felt at the time of these early conclusions that the compressor was not a significant contributor in the rear arc, since it was not possible to detect any compressor discrete tones even with the aid of fairly narrow band analysis. Tests on later engines, which were subjected to more critical analysis, subsequently showed this conclusion to be quite incorrect, as will be discussed, due mainly to the fact that long bypass duct and mixer system on a low bypass ratio engine causes an attenuation of discrete tones. The conclusion was presumably right however in the case of a pure jet engine, where there is of course no bypass system to carry and radiate the downstream compressor noise.

From all the early work outlined and the subsequent investigations to date it is now possible to indicate what the major sources are in a typical low bypass ratio engine, and how these vary with engine power setting. Such an indication is given in figure 4, where the linear Perceived Noise field shapes are plotted as functions of engine power setting.

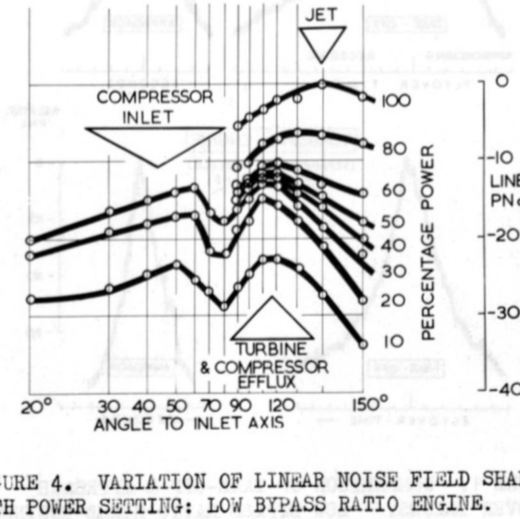

FIGURE 4. VARIATION OF LINEAR NOISE FIELD SHAPES WITH POWER SETTING: LOW BYPASS RATIO ENGINE.

Forwards of the engine the compressor noise is seen to be the dominant source, whilst rearwards of the engine the peak is attributable to either the combined effect of the compressor and the turbine or to the jet, depending upon the absolute power setting of the engine. At the higher power settings, in this case above 60%, the jet is of course the dominant feature of practically the whole noise field.

EXTRACTION OF TURBINE NOISE AS A UNIQUE SOURCE

If we consider the nature of the low power noise at the angle of peak radiation in the rear arc noise field of the low bypass ratio engine considered in figure 4, we see the type of spectrum illustrated in figure 5.

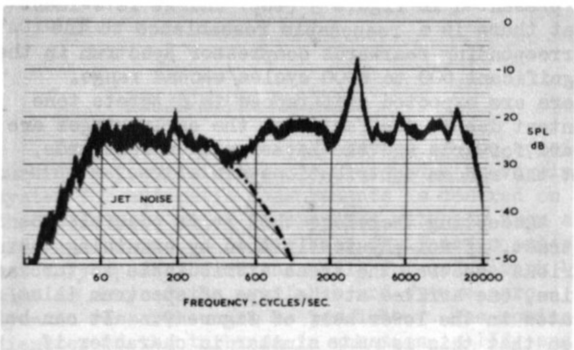

FIGURE 5. PEAK REAR ARC SPECTRUM AT LOW POWER LOW BYPASS RATIO MIXED ENGINE. 6% BANDWIDTH ANALYSIS.

Here jet noise is very low but nevertheless is still a contributor to portions of the overall spectrum. Its contribution is quite readily predictable from model jet evidence and is restricted to frequencies below approximately 600 cycles/sec. Above this frequency the whole spectrum is a combination of the turbine noise and the compressor noise carried down the bypass duct, both radiated at substantially the same angle, for reasons which will be discussed later

It is therefore not possible to simply deduce and extract the separate turbine and compressor components from such a combined spectrum.

To extract the individual components it is necessary to measure them individually, and to do so on an engine of this type, where both the compressor and gas generator flows are mixed before emission through the propulsion nozzle, it is necessary to carry out modifications to the engine. This in fact was done early in 1967 on a Conway engine, which was re-engineered to provide separate compressor bypass and gas generator flow exit nozzles. To accomplish this it was necessary to extend the gas generator jet pipe and provide a separate annular exit for the bypass flow. Noise measurements were then taken alternately with the gas generator flow carried to a detuner in one case, and then with the bypass flow deflected and screened from the noise measuring area in the other. An illustration of the rig arrangement for the first of these tests is shown in figure 6. Here the large open air test bed at Hucknall is seen with its movable detuner rolled up behind the gas generator nozzle, and a screening system built around the inlet of the engine to provide isolated measurements on the bypass flow. Both this configuration, and the configuration for gas generator flow measurement alone is illustrated diagrammatically in figure 7.

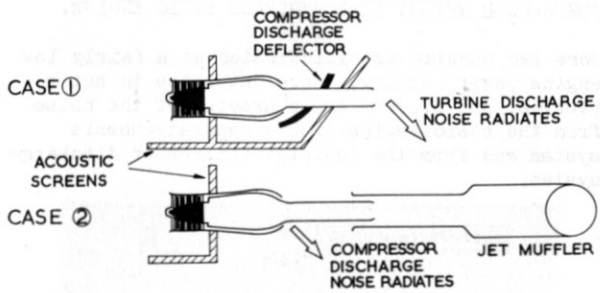

FIGURE 7. SCHEMATIC ARRANGEMENT FOR ISOLATION OF COMPRESSOR AND TURBINE NOISE SOURCES IN LOW BYPASS RATIO ENGINE.

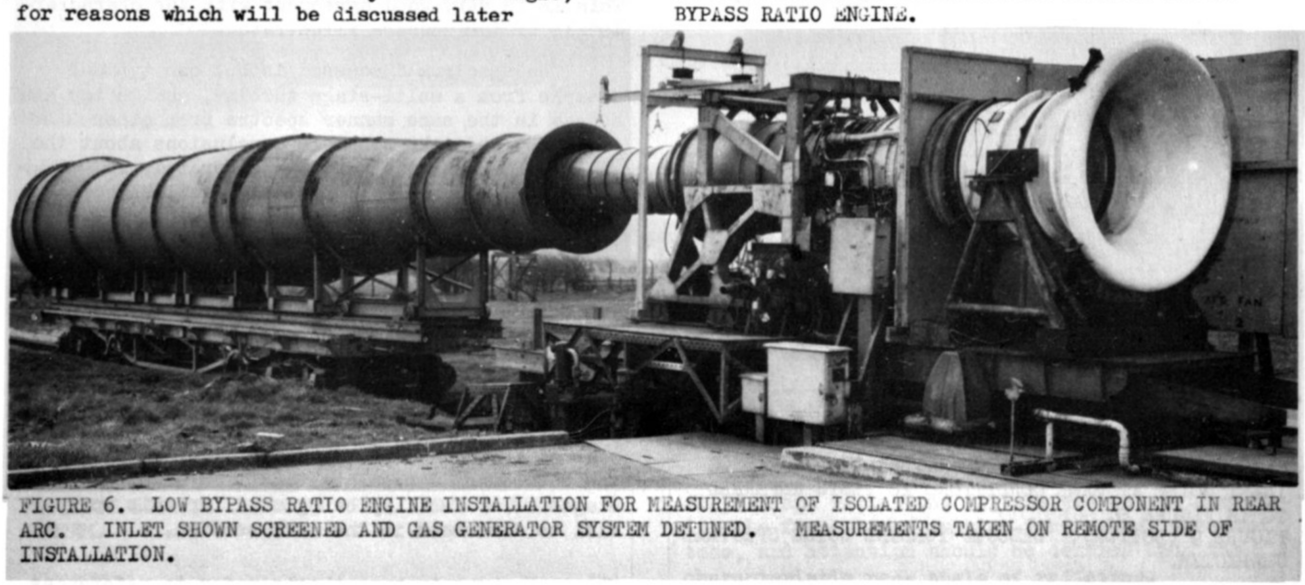

FIGURE 6. LOW BYPASS RATIO ENGINE INSTALLATION FOR MEASUREMENT OF ISOLATED COMPRESSOR COMPONENT IN REAR ARC. INLET SHOWN SCREENED AND GAS GENERATOR SYSTEM DETUNED. MEASUREMENTS TAKEN ON REMOTE SIDE OF INSTALLATION.

The results of these tests, which were extensive and covered the whole operating range of the engine, are typified by the sample spectra shown in figure 8.

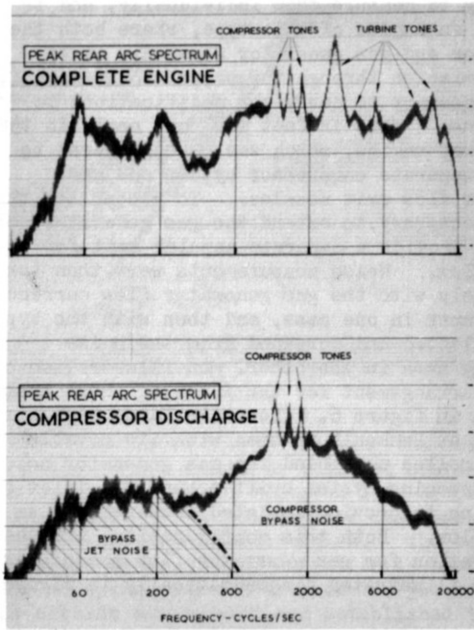

FIGURE 8. ISOLATION OF SPECTRAL CONTENT OF COMPRESSOR EFFLUX IN LOW BYPASS RATIO ENGINE.

Here two spectra are illustrated at a fairly low engine power setting, where jet noise is not a problem, and express the character of the noise from the basic engine with a separate nozzle system and from the isolated compressor discharge system.

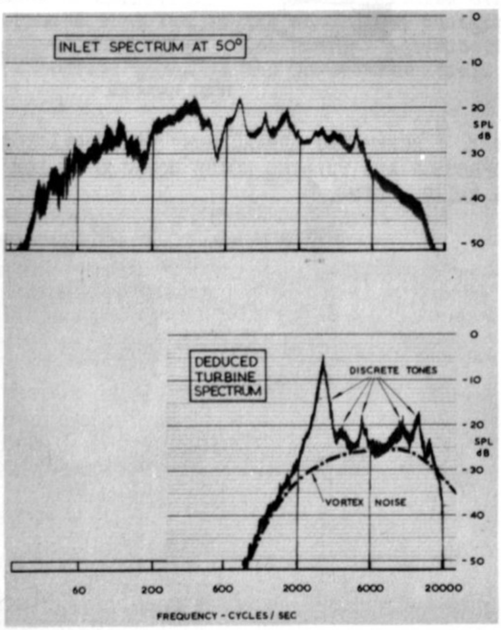

FIGURE 9 (TOP). SPECTRAL CHARACTER OF COMPRESSOR INLET NOISE AT SAME CONDITIONS AS TESTS REFERRED TO IN FIGURE 8.
FIGURE 9 (BOTTOM). DEDUCED TURBINE NOISE SPECTRAL CHARACTER.

Considering the spectra individually, the first point is that the basic engine without any mixer system exhibits a significant discrete tone content, not only from the turbine but also from the compressor. This justifies the conclusion that there is a significant contribution from the compressor in the overall level to the rear of the engine, whereas earlier tests with a mixed engine had suggested the opposite conclusion. The spectrum from the tests with the gas generator system fed to a detuner reveals just how much of this peak is in fact compressor noise, and by deduction the contribution from the turbine.

Since the rearwards propagated compressor noise is clearly a significant part of the total turbomachinery noise in the rear arc, it is of interest to compare the spectrum obtained to the rear of the engine with the actual forward radiated noise at the same engine condition. Therefore a spectral analysis of the forwards compressor noise is presented in figure 9 (top) and it is evident that there is a reasonable resemblance to the corresponding rearwards compressor spectrum in the significant 600 to 6000 cycles/second range. There are expected differences in discrete tone content due to the fact that the early stages are heard forwards and the latter stages rearwards, but the random contributions are alike.

Accepting therefore that it is possible to extract, if not physically then by examining the various spectra, the areas attributable to turbine noise, one arrives at the type of spectrum illustrated in the lower half of figure 9. It can be seen that this is quite similar in character if not in frequency content to the compressor spectrum in the same illustration, and it is quite logical to suppose that it consists of the same component noises as the compressor spectra. Therefore in the same way that compressor noise is broken down into discrete and vortex or "white" components (6) (7) the turbine spectrum has been shown as the result of such components in the illustration. Discussion of the generation of these components will follow in due course, but it is of interest to note here that on a Strouhal number basis the blade dimensions and stream temperatures suggest a 4 : 1 characteristic vortex noise frequency separation between the turbine and compressor. This is in very good agreement with the characteristics of the spectra illustrated.

The spectrum discussed is but one typical example from a multi-stage turbine, and to try and deduce in the same manner spectra from other turbines, and make sensible conclusions about the way in which turbine noise is generated, is virtually impossible if the only available data is that from full scale multistage engine tests. It was therefore decided that the only way to pursue turbine noise research satisfactorily was to commission a special facility, so that work could be carried out on less complex noise sources. This involves going to a minimum of a single stage turbine with the appropriate guide vane systems. It would probably be sensible to go even further and work with an isolated rotor or aerofoil, but at this point in time work is centred around a representative single stage with the subsequent incremental addition of other stages. The facility being used for this work will be discussed in the next section.

(*) Numbers in paranthesis refer to references at the end of the paper.

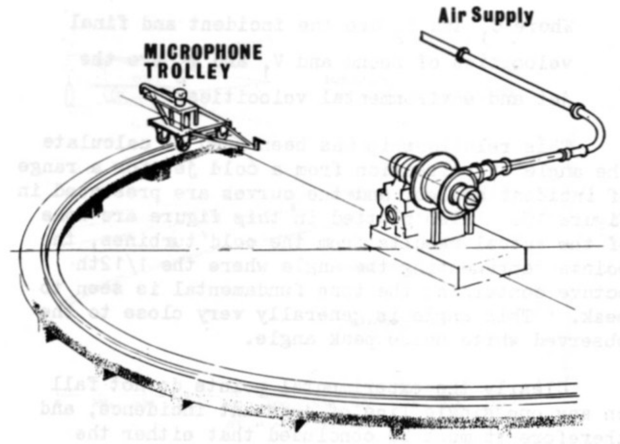

FIGURE 10. DIAGRAM OF TURBINE NOISE RESEARCH FACILITY.

NEW TURBINE FACILITY

Figure 10 indicates the layout of the special turbine noise facility. A microphone traversing system for automatic measurements is centred on the test unit, which is supplied with air from an external source. It has been designed to accept not only special turbines built for noise work but also as many of the available and future aerodynamic research units as possible. The nominal diameter of the turbines in question is 15", and the facility is run either hot or cold. The initial part of the programme, which started late last year, has been confined to cold systems, but the addition of a combustion chamber has made it possible to work on existing small engine turbines.

Figure 11 is a photograph of the actual site. The turbine and microphone traversing track is shown inside a "rosebowl" screening system, which has been erected for two reasons. One is to avoid the reflection of turbine noise from adjacent buildings back into the measuring area, and the other to reduce wind effects.

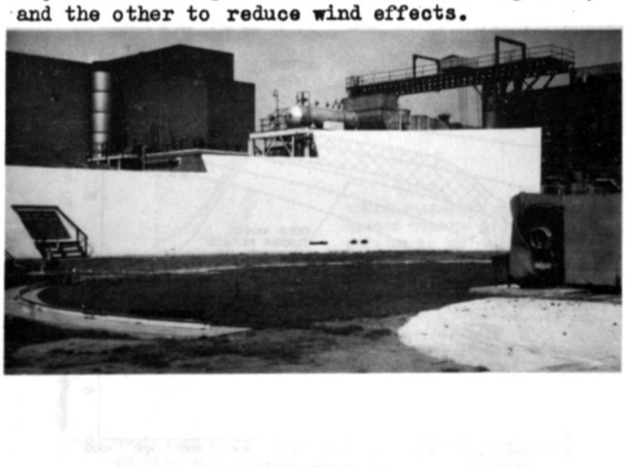

FIGURE 11. PHOTOGRAPH OF TURBINE NOISE RESEARCH FACILITY SHOWING TRAVERSE TRACK AND SCREENING SYSTEM.

RESULTS FROM TURBINE RIG

At the time of writing the results available for open discussion are limited to those from a single and a two stage cold turbine. These will be discussed in greater detail in a later section but at this point it is of interest to consider the types of peak spectrum being observed. The two spectra shown in figure 12 were obtained from the single and two stage units with the common first stage running at identical conditions. Since great care was taken to avoid high jet velocities by the use of a large final nozzle behind the turbine, they are substantially devoid of jet mixing noise. The character of these spectra is very similar to the full scale engine spectrum deduced earlier, except for frequency differences caused by the different blade numbers, the scale of blades used and the different temperatures. There is as expected a clearly defined vortex noise level with the rotor passage discrete frequencies superimposed. Unfortunately there is very little difference between the blade numbers on the first and second stages, and therefore separate tones from the two stages are not apparent.

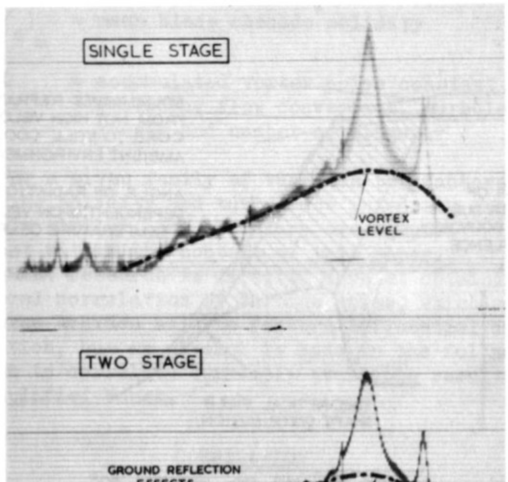

FIGURE 12. SAMPLE SPECTRA FROM SINGLE AND 2 STAGE COLD TURBINES. 6% BANDWIDTH ANALYSIS.

RADIATION PATTERNS

Before considering the mechanisms and generation of the turbine noise, and discussing the way in which intensity varies with the changing parameters during operation, one point well worth discussion is the question of how the noise generated at the blades radiates through the jet to the outside atmosphere.

Figure 13 shows several full scale engine field shape variations for the $\frac{1}{3}$rd octave containing the final stage turbine fundamental discrete tone, and attention should be centred upon the characteristic peak angle of radiation.

It is fairly easy to postulate the way in which a noise source generated upstream of a nozzle is propagated through the jet to the outside atmosphere, but it is by no means certain that one can account absolutely for the measured angles of radiation.

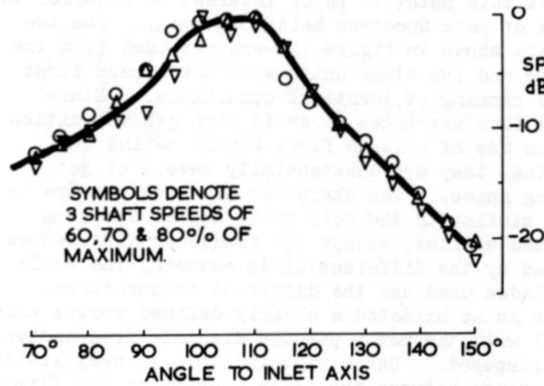

FIGURE 13. LINEAR FIELD SHAPES FOR FULL SCALE ENGINE TURBINE NOISE. ⅓rd OCTAVE CONTAINING FINAL STAGE FUNDAMENTAL ZONE.

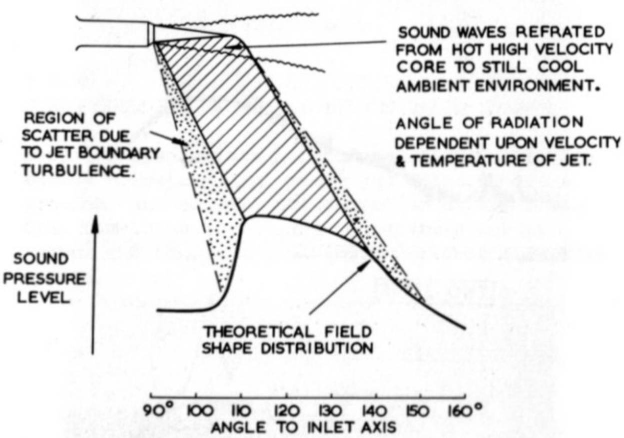

FIGURE 14. SIMPLIFIED ILLUSTRATION OF REFRACTION AND RADIATION OF NOISE ORIGINATING WITHIN JET POTENTIAL CORE.

Figure 14 shows diagrammatically how it is felt the noise is refracted and radiated from the potential core to the outside environment, and it should of course be possible to calculate a theoretical angle of refraction knowing the governing variables. Such calculations do not however appear to agree simply with the observed angles typified by figure 13. This is especially true of the case of the cold turbine, where any refraction from the potential core will mainly be a factor of the velocity distribution in the jet. Using simple refraction theory, the refracted angle ϕ is related to the incident angle θ by:

$$\sec\phi = \frac{C_1}{C_2}\sec\theta + \frac{V_1 - V_2}{C_2}$$

Where C_1 and C_2 are the incident and final velocities of sound and V_1 and V_2 are the jet and environmental velocities.

This relationship has been used to calculate the angle of refraction from a cold jet for a range of incident angles, and the curves are presented in figure 15. Also plotted in this figure are some of the actual results from the cold turbines, the points representing the angle where the 1/12th octave containing the tone fundamental is seen to peak. This angle is generally very close to the observed white noise peak angle.

Clearly the experimental points do not fall on any one single line of constant incidence, and therefore it must be concluded that either the refraction supposition is wrong, or that the effective incident angle within the jet is not a constant value with the velocity. The second supposition is probably the more viable, since the discrete tones are, like compressor tones, generated as helical waves and the effective incident angle must vary with varying turbine conditions. In practice very low velocities are not of any real interest, and therefore a consideration of the points above 200 or 300 feet per second jet velocity is probably sufficient at this stage. Here the incident angle is apparently around 50°, some of which may be accounted for by the cone angle of the jet core, but the rest must be associated at this time with the helical nature of the wave fronts. No complete analysis is attempted here but the deduced incident angle is applied to the "real life" case, where turbines operate at temperatures substantially above ambient.

This is done in figure 16, for a range of jet temperatures up to 1000°K. Plotted on here also are a number of actual engine results for which the average temperature is in the region of 500 to 600°K. The observed results are seen to be in fairly good agreement with the "theoretical" lines which have been calculated on the basis of a 50° incident angle within the jet.

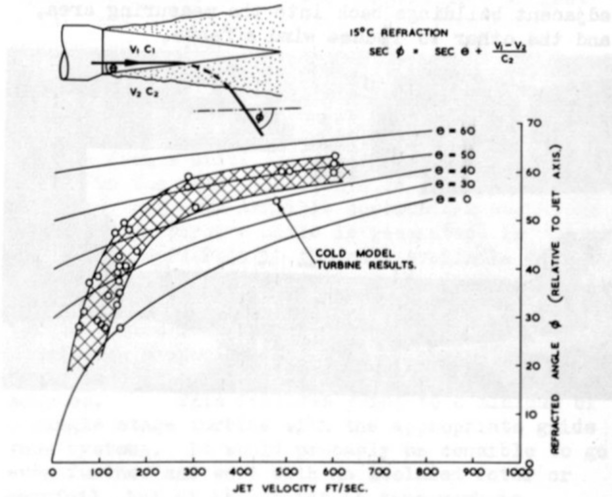

FIGURE 15. COMPARISON OF COLD TURBINE PEAK RADIATION ANGLE WITH THEORETICAL CURVES.

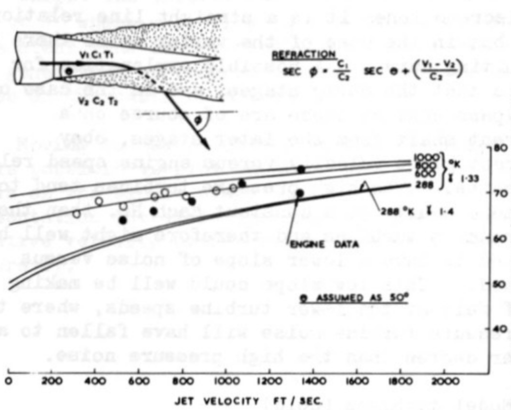

FIGURE 16. COMPARISON OF FULL SCALE ENGINE PEAK RADIATION ANGLES WITH THEORETICAL CURVES.

MECHANISM OF GENERATION

The two component forms of noise, discrete tones and broad band vortex noise, are of a completely different nature, and the mechanisms of generation are different. They are however directly comparable with the corresponding mechanisms of generation of fan or compressor noise, the more significant of which are illustrated in figure 17. The discrete tones result from the cyclic interception of the nozzle guide vane wakes by the rotor blades, whilst the vortex noise may result from random lift pressure fluctuations on both rotors and stators. The pressure fluctuations are caused by wake and freestream turbulence, the latter resulting from both combustion system mixing and also the burning process.

Factors contributing to the intensity of generation are considerable in number, and have been fairly fully outlined in an earlier paper on compressor noise(7). The most important is undoubtedly the blade relative velocity, whilst turbulence intensity, stream density, blade incidence, cascade solidity and number of stages all contribute, with nozzle guide vane to rotor

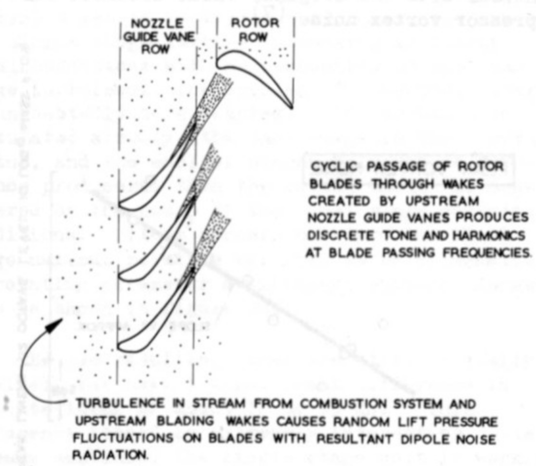

FIGURE 17. SIMPLIFIED EXPLANATION OF GENERATIVE MECHANISMS OF TURBINE VORTEX AND DISCRETE TONE NOISE.

separation having an independently important bearing on discrete tone intensity. The relevant expression for dipole noise developed from Lighthills Theory of Aerodynamic Sound is as follows:-

$$PWL = 50 \log_{10} V_{rel} + 30 \log_{10} \left(\frac{1116}{a}\right) + 10 \log_{10} M \phi^2 \left(\frac{\overline{v}^2}{V^2}\right) - 10 \log_{10} \left(\frac{C}{s}\right)_m \left(\frac{V_a}{V_{rel}}\right) + K$$

where PWL = vortex or "white" noise total acoustic power

V_{rel} = relative blade velocity

M = mass flow

V_a = mean axial velocity

ϕ = lift curve slope

$\overline{v}^2/V^2$ = turbulence intensity

a = local speed of sound

$(C/s)_m$ = mean blade cascade solidity

and K = accumulated vortex noise constant embracing flow convection, turbulence growth and number of stages.

For a given family of turbines the factors involving solidity and turbulence intensity are probably similar, and therefore are assumed constant within the context of this paper. Subsequent research is directed more at obtaining a general correlation of turbine noise, which must of course embrace all the factors influential in generation, but an attempt is made in the following section to correlate currently available results on simplified bases.

CORRELATION

At the outset it must be admitted that correlation of turbine noise, and in particular the vortex component, is probably a different procedure than correlating fan and compressor noise. The way in which turbines work, for instance the high pressure system and low pressure system having significantly different working lines and frequently being on separate shafts, and the fact that almost all the noise must be propagated in the downstream direction, makes the deduction of the exact contributions to the overall level extremely difficult. Moreover the opinion has been expressed that the early stages of turbine may only be acting as turbulence producers, and perhaps not direct contributors to the overall radiated noise level, and if this is the case it is not merely a question of how much noise is produced by each stage, but primarily of which stages produce noise.

Faced with these difficulties it is perhaps sensible to look firstly at some of the experimental evidence in the context of "correlations attempted to date" and then consider if it is possible at this stage to derive some overall form of analysis.

(a) Full scale engine turbines

Using the results of the separate nozzle tests on a low bypass ratio engine that were outlined earlier as a basis for the extraction of turbine noise from the peak rear arc spectra of other engines, two simple correlations have been made. There refer firstly to the peak 1/12th octave vortex noise level and secondly to the final turbine stage fundamental discrete tone level. These are presented as figures 18 and 19. The basis for both the correlations is turbine outlet tip speed, that is to say the actual blade absolute tip speed, although since the noise produced should be a function of relative velocity this is not the correct fundamental parameter. However it is a sufficiently good basos as to show that, after normalisation of all the results by removal of mass flow, there is some sort of relationship with tipspeed in both cases.

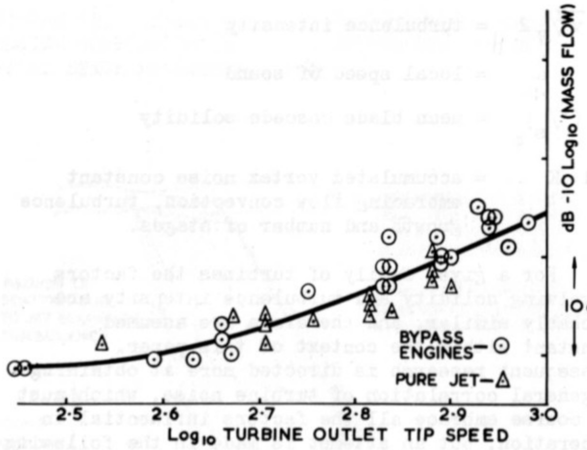

FIGURE 18. CORRELATION OF LINEAR PEAK VORTEX NOISE LEVEL. FULL SCALE ENGINES.

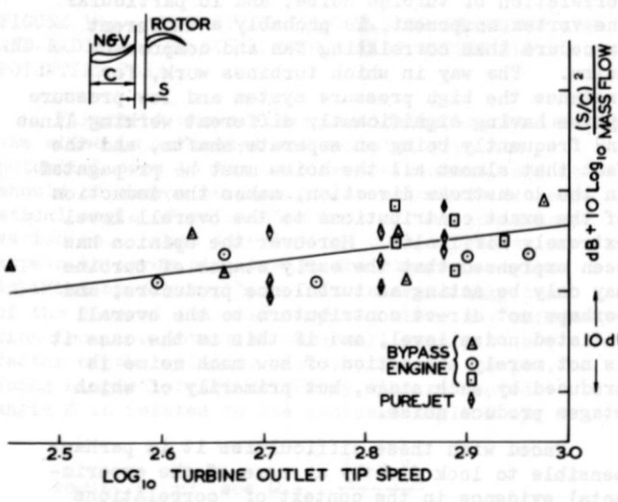

FIGURE 19. CORRELATION OF FINAL STAGE FUNDAMENTAL DISCRETE TONE LEVELS. FULL SCALE ENGINES.

For discrete tones it is a straight line relationship, but in the case of the vortex noise there is a resulting curve. A possible explanation for this is that the early stages, and in the case of the bypass engines these are of course on a different shaft from the later stages, obey different blade velocity versus engine speed relationships. The high pressure turbines tend to work more nearly at a constant Mach No. than the low pressure turbines and therefore might well be expected to have a lower slope of noise versus tipspeed. This low slope could well be making itself felt at the lower turbine speeds, where the low pressure turbine noise will have fallen to a greater degree than the high pressure noise.

(b) Model turbines (cold)

The available work to date on model turbines is limited to cold tests on a single and two stage unit. The two stage unit consists of a second stage added to the single stage unit, so it is possible to compare the results from the two turbines directly by choosing comparable operating conditions of the common stage.

The method of testing on both turbines followed a procedure whereby a series of constant non-dimensional speeds were held for a variety of pressure ratios. In this way operating conditions well removed from the normal operating conditions of a full scale turbine were obtained, and thereby large variations of blade relative velocity and incidence recorded.

Neglecting these wide variations of incidence for the moment, the same procedure adopted for compressor correlations can be used for the single stage unit. That is to say a correlation of the peak vortex noise level can be carried out using some measure of blade relative velocity as a basic parameter. This is attempted in figure 20, where the maximum rotor inlet relative velocity is used and the peak vortex levels are normalised by removal of mass flow. Whilst the scatter of the results might well emerge as an effect of blade incidence, as it did in compressor correlations, it is interesting to note at this stage that the mean line drawn has a slope of about V^3. This is almost identical with the original value observed for compressor vortex noise (7).

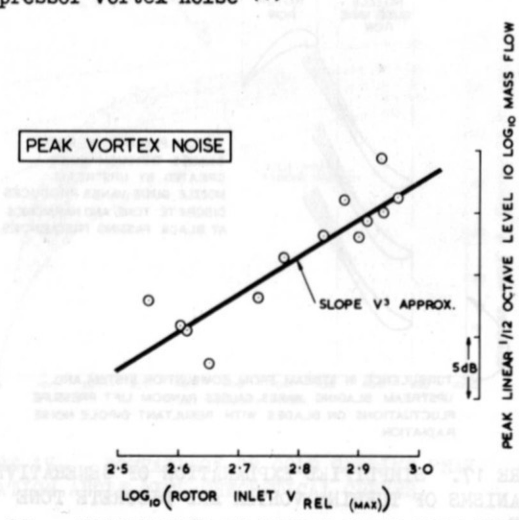

FIG.20. CORRELATION OF SINGLE STAGE COLD TURBINE PEAK VORTEX NOISE.

Whilst the theoretical dipole index would be V^6, or V^5 where mass flow is removed as variable, some clear link is nevertheless probable between the generation of compressor and turbine vortex noise on the basis of the test results.

Moving to the results from the two stage unit, it is possible to plot these on exactly the same basis as the single stage data. As a first presentation therefore the same maximum rotor inlet relative velocity (first stage) has been used in figure 21.

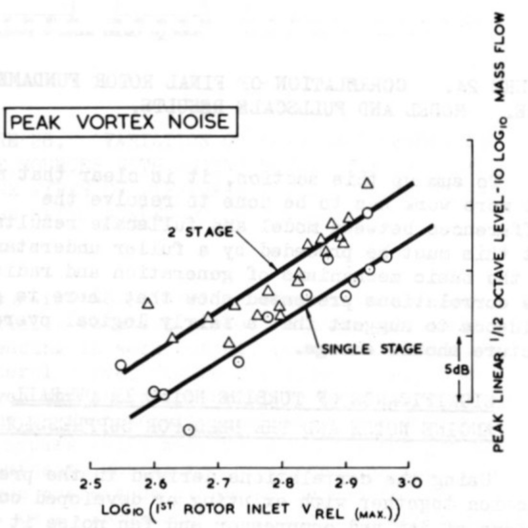

FIGURE 21. CORRELATION OF SINGLE AND 2 STAGE VORTEX NOISE USING FIRST ROTOR STAGE VELOCITIES AS BASIC PARAMETER.

The two sets of results fall on almost parallel lines and, as might be expected, the two stage levels are higher than the single stage levels, by in fact about 3 or 4 dB. This observation immediately suggests a $10 \log_{10} (N)$ relationship, where N is the number of stages. Unfortunately no 3, 4 or more stage results are available yet to verify this approach, but since the single stage turbine is working in fairly ideal conditions with no combustion or upstream stage turbulence, the problem of subsequent stages is undoubtedly more complex. If, as has been postulated already, the last stage is the governing factor, and the earlier stages only act as turbulence producers, then the levels should be considered on the basis of the final stage operating conditions. If we choose therefore the final stage maximum relative velocity as an alternative correlating parameter a different picture emerges. This is shown in figure 22.

The two resulting lines are still virtually parallel, but have a significant difference in absolute level of about 12 or 13 dB. This difference itself is extremely interesting. As already mentioned the single stage unit is working under almost ideal conditions. The nozzle guide vanes are fed with clean, smooth unburnt air from a large settling chamber, whereas the second stage of the two stage unit must be receiving fairly

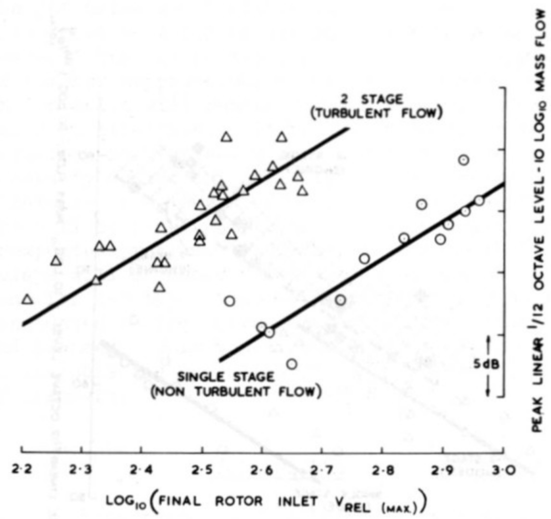

FIGURE 22. CORRELATION OF SINGLE AND 2 STAGE VORTEX NOISE USING FINAL STAGE VELOCITIES AS BASIC PARAMETER.

turbulent air as a result of traversing the first stage. Compressor work and small scale aerofoil work has led to the conclusion that the difference in the basic generative mechanisms involved in turbulent and turbulent free conditions is of the order of 10 to 15 dB, the turbulent conditions producing the more intense noise. Clearly, if it is true that a significant difference in turbulence exists between the first and second stages, the order of difference observed in the test results is in very good agreement with that anticipated.

Taking these results a stage further, the next logical step is to compare on a common basis the cold model results with fullscale hot engine turbine results.

(c) Model and fullscale turbines

The essential differences between the model and fullscale turbine tests lie in the region of operating temperatures and blade velocities. The use of blade relative velocity as the prime correlating parameter caters for changes in this function, but a correction must be made for temperature. Another difference between the test results lies in the mode of measurement. In the case of the model results polar traversing was carried out at about 40 ft but in the case of the engine results traversing was parallel to the engines at 100 ft. The correlation presented in figure 23 therefore expresses all the results on a common basis, temperature having been corrected to ambient conditions in the case of the full scale engines and the distance having been extrapolated to 100 ft in the case of the model results.

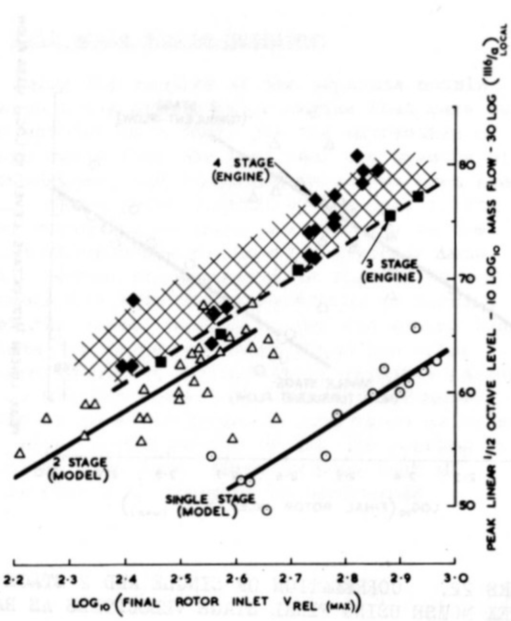

FIGURE 23. CORRELATION OF MODEL AND FULLSCALE VORTEX NOISE USING FINAL STAGE VELOCITIES AS BASIC PARAMETER.

The degree of agreement between the 2 stage model and the 3 and 4 stage fullscale turbine results is surprisingly good for a first approximation. The increase over and above the two stage results for adding subsequent stages and going to hot conditions appears to be of a sensible order of magnitude. The three stage turbines lie at the bottom of the scatter band of fullscale results, and the overall scatter is no more than 5 dB, the mean lying about 5 dB above the two stage results. This is very nearly a 10 $\log_{10}(N)$ relationship with number of stages.

Moving briefly to the topic of discrete tones, a similar correlation procedure has been adopted for the final stage turbine fundamental tones. This is shown in figure 24, where the normalising parameter other than mass flow and temperature is nozzle guide vane/rotor separation. This has been taken as $(S/C)^2$ since this is the only available parameter for use at this time, and results from work on compressors and fans. S is the separation between the nozzle guide vane and rotor, C being the guide vane axial chord.

Here again, as is the case in the vortex correlation, the agreement between model and fullscale is seen to be fairly good, There is rather more scatter than in the case of vortex noise levels, but this has always been the case in the context of discrete tone measurement, and a number of possible reasons have been discussed elsewhere.

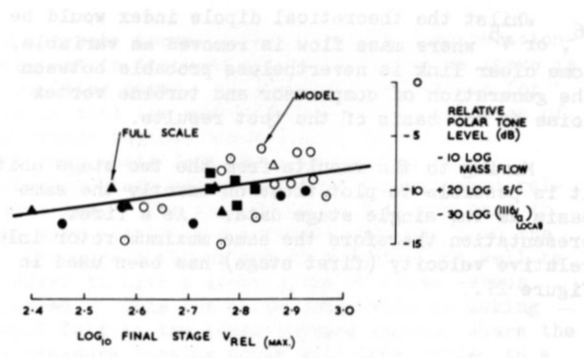

FIGURE 24. CORRELATION OF FINAL ROTOR FUNDAMENTAL TONE. MODEL AND FULLSCALE RESULTS.

To sum up this section, it is clear that a lot more work has to be done to resolve the differences between model and fullscale results, but this must be preceded by a fuller understanding of the basic mechanisms of generation and radiation. The correlations presented show that there is good evidence to suggest that a fairly logical overall picture should emerge.

SIGNIFICANCE OF TURBINE NOISE IN OVERALL ENGINE NOISE AND THE NEED FOR SUPPRESSION

Using the correlations derived in the previous section together with existing or developed correlations of jet and compressor and fan noise it is possible to generalise on the significance of each noise source in overall engine noise levels.

For engines designed to produce the same overall thrust, an increase in bypass ratio causes the jet component to fall in intensity due to the lowering of jet velocity, and the compressor or fan noise to increase due to increasing work.

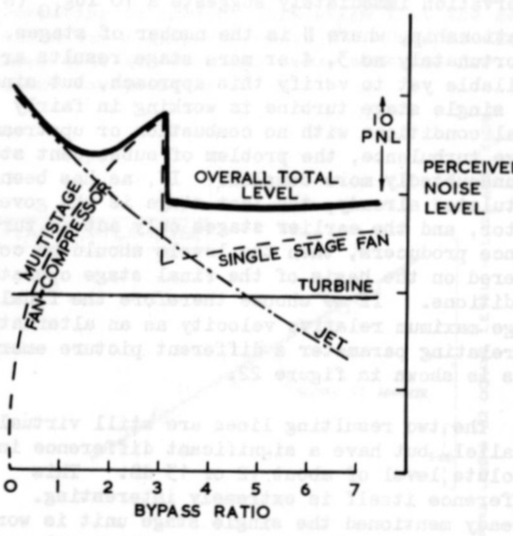

FIGURE 25. VARIATION OF COMPONENT NOISE SOURCES AS BYPASS RATIO IS VARIED AT CONSTANT THRUST.

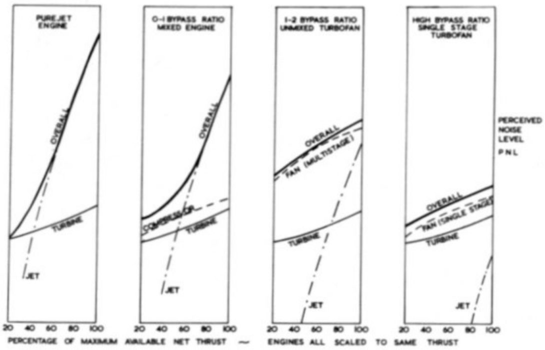

FIGURE 26. VARIATION OF REARWARDS PROPAGATED NOISE SOURCES WITH ENGINE POWER SETTING. FOUR ENGINE TYPES OF DIFFERENT DESIGH PHILOSOPHY AND BYPASS RATIO SCALED TO SAME THRUST.

The turbine noise would appear to remain substantially constant since the unit, although increasing in work output, reduces in massflow. A general curve, figure 25, illustrates these changes and shows the relative importance of each source in the overall level at engine design point. From bypass ratio zero to around $1\frac{1}{2}$ the jet noise is dominant, whilst the compressor or fan noise assumes precedence at higher bypass ratios. There is a discontinuity in the compressor noise curve where design capability allows the use of a single stage fan [11]. The turbine noise, in remaining substantially constant, begins to assume real importance beyond the point where the compressor noise discontinuity occurs.

This, however, is only half the story. For power settings well below design point, i.e. those used during noise abatement throttling after take-off and those used during approach to land, the turbine noise assumes much more importance. Figure 16 illustrates this fact, plotting in four sections the variations of intensity of the rearwards propagating sources for three existing in-service engine types and a high bypass ratio engine type. For the purejet engine the jet noise is all important, except at very low powers where the turbine noise is a contributory factor in the overall level. For the mixed bypass engine of up to bypass ratio 1, the jet is again most significant, but only at half power or more. Below half power the combined effect of compressor bypass and turbine noise is very significant. For the turbofan, of short cowl or coplanar exit design, and of bypass ratio between 1 and 2, the fan noise is dominant at almost all power settings. Because of the extreme intensity of fan noise the turbine noise, like the jet noise, is relatively insignificant.

However, turning to the advanced technology engine of bypass ratio of 3 or more, where there is a significant reduction in fan noise due to the use of a single stage, the turbine noise becomes of importance. Over the whole power range it is only marginally less than the fan noise, and since the jet noise is of little significance the turbine noise must be afforded the same attention as the fan noise. This is particularly true in the context of further suppression, since even a small reduction in fan noise will render further suppression less and less effective. It is therefore clear that research into fan noise suppression must be accompanied by equivalent activity in the turbine area, otherwise in a very short space of time manufacturers will be faced with a sudden and possibly unexpected problem. This may well mean that delays will be encountered when application is made for the type of noise certificate now being considered in the United Kingdom, United States and France. This would have serious repercussions in the sale and operation of the next generation of aircraft.

CONCLUSIONS

(1) Turbine noise has been exposed as a significant noise source in the gas turbine engine. Its real importance in the context of the high bypass ratio engine has been indicated.

(2) The composition of turbine noise has been shown to be very similar to fan and compressor noise. The two significant forms, discrete and vortex, have been defined and are expected to vary basically with blade relative velocity.

(3) Correlations of both components have been attempted on the basis of blade relative velocity and a reasonably sensible relationship between cold model and fullscale hot engine turbine levels has been established.

(4) The need for suppression of turbine noise has been expressed from a consideration of the relative levels of fan and turbine noise on the high bypass ratio engine. It is imperative that turbine noise suppression research proceeds alongside the current programmes on fan noise suppression.

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