INSTITUTE OF AEROPHYSICS

UNIVERSITY OF TORONTO

CANADIAN RESEARCH IN AERODYNAMIC NOISE

BY

B. ETKIN and H. S. RIBNER

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SUMMARY

Canadian research on flow noise and some aspects of the aircraft noise problem is described. The work was done at the Defence Research Board, the University of Toronto Institute of Aerophysics and A.V. Roe (Canada) Ltd.

Specific experimental and/or theoretical investigations include: Aeolian Tones; Boundary Layer Noise (rigid wall and flexible wall); Effects of Boundary Layers and Noise on Aircraft Structures; Distribution of Noise Sources Along a Jet; Ground Run-up Mufflers; Transmission of Sound from, and Acoustic Energy Flow in, a Moving Medium; Sound Generated by Interaction of a Vortex with a Shock Wave.
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INTRODUCTION

This paper describes fundamental and applied research that has been conducted in Canada during the last few years on flow noise, and on some aspects of the aircraft noise problem. The work has been carried out at the Defence Research Board, the University of Toronto Institute of Aerophysics, and A.V. Roe (Canada) Ltd. Financial support for the work at the Institute of Aerophysics has come from the Defence Research Board, the United States Air Force, and A.V. Roe.

The research described embraces both fundamental and applied, and both theoretical and experimental investigations. These relate to the generation of sound by jets, wakes and boundary layers; to the suppression and reduction of such sound; and to its effects upon structures.

A.1 - Jet Noise

Strength Distribution of Noise Sources along a Jet

Progress in the development of mufflers for jet noise has proceeded largely by empiricism, (cf Ref. A1) guided to a certain extent by Lighthill's basic theory (Refs. A2, A3). Curiously enough, it is only recently that the theory has been applied (in different ways, Refs. A4, A5; see also Ref. A6) to assess the distribution of noise-source strength along a jet; yet this information is basic to an understanding of jet-muffler action.

In the Canadian study (Ref. A5) Lighthill's theory is applied to the two regions of 'similar' profiles in a jet. The analysis refers to the noise power emitted by a 'slice' of jet (section between two adjacent planes normal to the axis) as a function of the distance $x$ of the slice from the nozzle. It is found that this power is essentially constant with $x$ in the initial mixing region ($x^0$law), then farther downstream (about 8 or 10 diameters from the nozzle) falls off extremely fast ($x^{-7}$ law or faster) in the fully developed jet (Fig. A1). Because of this striking attenuation of strength with distance, it is concluded that the mixing region produces the bulk of the noise and must dominate in muffler behaviour; conversely the 'fat' part of the jet must contribute much less to the total noise power than is commonly supposed.

Powell's experiments (Ref. A7) on the effects of nozzle velocity profile on total noise power can thereby be interpreted qualitatively. The behaviour of multiple-nozzle or corrugated mufflers, both as to overall quieting and as to frequency-shifting, are also interpreted in the light of the results. Because the noise is concentrated near the nozzle (Fig. A-1) the possibility emerges that such mufflers may be improved without serious thrust loss by the addition of a sound-attenuating shroud (Fig. A2).
Transmission of Sound to or from a Moving Medium

The details of the transmission of noise from a high speed jet to the surrounding quiescent air are not known. Lighthill (Refs. A2, A3) circumvents the details in terms of his Mach number factors for 'self-propelled' noise sources (his analysis does not imply convection); however, the factors are open to suspicion in their prediction of infinite energy radiation at $M = 1$. Thus some examination of the mechanism of the sound transmission is in order, starting with a much simpler case. The problem of the transmission of plane sound waves to or from a moving stream has been investigated in Ref. A8. The waves are specified as originating in air at rest and impinging obliquely on a plane interface with a moving stream. The analysis and the physical interpretation are both simplified by using axes moving with the ripple that must develop in the interface. The acoustic problem is thus changed into the aerodynamic problem of the flows above and below a wavy wall - the rippled interface. (Fig. A3).

In this view the angles of incidence, reflection, and refraction are regarded as Mach angles in two supersonic streams. The velocity difference between these two streams is just the velocity of the original moving stream. The known angle of incidence thereby leads immediately to an equal angle of reflection and a simply-related angle of refraction (Fig. A4). The angle of refraction is imaginary when the associated velocity, calculated by the velocity-difference rule, is subsonic. This is a condition of total reflection.

The amplitude relations (coefficients of reflection and transmission) are evaluated in closed form. In a graph three zones can be distinguished in the plane of angle of incidence $v.$ Mach number of the moving medium: ordinary reflection and transmission, total reflection, and amplified reflection and transmission (Fig. A5). Included are three loci of infinite reflection: i.e., self-excited waves. The energy balance is examined, and the source of amplification is concluded to be the energy of the moving stream.

Acoustic Energy Flow in a Moving Medium

It was remarked in the last section that Lighthill's Mach number factors for the moving sources in a jet lead to the unacceptable prediction of infinite energy radiation at a jet Mach number of unity. Thus the whole question of acoustic energy flow in a moving stream must come under scrutiny.

It is known that both acoustic energy density and energy flow are modified by motion of the medium. The classical relation, acoustic energy flow equals pressure times velocity, applies only when the medium is at rest. The derivation of a corresponding relation for a moving medium offers some difficulty. Comparison has been made (Ref. A9) of similarities and discrepancies in the formulas of
three investigators (Refs. A8, A10, A11) in order to infer a correct formulation.

A convenient form of such a formulation for plane sound waves is found to be:

\[ \varepsilon = \frac{P^2 V_f}{\rho c^3} \]  

('linear theory')

energy flow/unit area = \( \varepsilon \overrightarrow{V}_s \) (a vector)

(area taken \( \perp \) energy flow)

This formalism is that of Ref. A10; that of Ref. A8 is compatible, and that of Ref. A11 errs only by a multiplicative constant.

These formulas for a moving stream can be illustrated with the aid of Fig. A6. The planes of constant phase move with the 'phase velocity' \( V_f \) different from the ordinary speed of sound \( c \). The energy flows with the 'ray velocity' \( \overrightarrow{V}_s \). Fig. A7 shows how, for a given wave pattern, progressive variation of the stream velocity can change the 'linear theory' acoustic energy density - which is proportional to \( V_f \) - from positive through zero to negative, with corresponding changes in the energy flow. For the particular case of zero acoustic energy density and energy flow the convection has cancelled out the propagation so that the sound waves are stationary; they are the familiar 'Mach waves' of steady supersonic flow.

Sound Generated by Interaction of a Single Vortex with a Shock Wave

It is well known that with the appearance of shock waves in a supersonic jet the noise emission is greatly increased, exceeding the \( U^8 \) law. (cf. Ref. A1). Theoretical studies have emphasized on the one hand a resonance effect (Ref. A12), and on the other hand the interaction of convected turbulence (created by the mixing) with the shock waves (Refs. A13, A14, A15).

The turbulence analyses do not provide a physical picture of what happens when a localized "eddy" passes through a shock wave. To simulate such a situation Hollingsworth and Richards made a schlieren study of the passage of a single columnar vortex 'broadside' through a shock wave (Ref. A16) and also presented an heuristic theory (Ref. A17). The work to be discussed (Ref. A18) is an attempt at a quantitative theory of such an interaction.

The analysis exploits the concept that a vortex can be decomposed by Fourier methods into plane shear waves disposed radially like the spokes of a wheel (Fig. A8). Each of the shear waves interacts with the shock to produce a refracted shear wave and a plane sound wave according to previous work (Ref. A13) (Figs. A9, A10).
The plane sound waves emanate, with varying angles, all along the shock front.

The plane waves possess an envelope that is essentially a growing cylindrical sound wave partly cut off by the shock (Fig. A10); that is, the pressure pattern peaks sharply at a cylindrical front (Fig. A11). The cylindrical wave is centered at the transmitted (and modified) vortex core and its peak attenuates inversely as the square root of the growing radius. The strength varies smoothly around the arc, from compression at one shock intersection to rarefaction at the other general shock intersection. These calculated characteristics appear to conform in a general way with a schlieren photograph of the interaction process obtained in Ref. A16.

Ground Mufflers (UTIA Research)

One means of muffling a jet on the ground is to expand it in a diffuser. The small-diameter high-velocity jet is thereby converted into one having a large diameter and a low velocity. The eighth-power law indicates that the final jet will produce much less noise than the original one. Of course, the noise generated inside the diffuser may be very intense, and must be absorbed by a suitable acoustic treatment. Since high pressure-recovery in the diffuser is not a factor, the application of short wide-angle diffusers appears attractive. One investigation along these lines was reported by Green and Lilley (Ref. A19) who found that the diffuser would run full if screens were used to provide enough internal resistance. The UTIA investigation was independently conceived, although information about the Green-Lilley work was received during the course of the research. The UTIA experiment (Ref. A20) differed in one significant respect from that carried out at Cranfield. This difference was the inclusion of air augmentation in the model configuration (Ref. A12). This feature might be necessary in a full-scale muffler for cooling purposes.

After a number of trials, satisfactory aerodynamic performance was achieved with two screens \( \Delta p/\rho = 3.4 \) as shown in Fig. A13. It can be seen that the diffuser did achieve the objective of producing a fairly uniform low-velocity exit jet. The induced flow achieved was 13.4% of the jet flow. A theoretical analysis given in Ref. A20 indicates that the velocity profile at the diffuser inlet is an important factor in determining the magnitude of the induced flow, and that a longer mixing length would increase it.

The acoustic performance of the model muffler tested was poor, in that more noise was generated with it than without it. This is attributed to inadequate acoustic design of the muffler walls. The Green-Lilley experiment was quite successful in this regard.

On the basis of the work cited, it would appear that wide angle diffusers could be developed into successful mufflers.
Ground Mufflers (Orenda Engines Research)

Orenda Engines Ltd. undertook the development of a light portable ground muffler which would be suitable for use with afterburning engines and which would provide a moderate attenuation. The program began with experiments on models, and progressed to full-scale testing of the configuration which evolved (Ref. A21). The muffler tested is shown schematically in Fig. A14. It consists essentially of a diffuser, which contains hollow baffles through which outside air is drawn into the flow. The resistance of the baffles is such that the flow remains attached to the walls. In this respect it is similar to the screened diffuser described above.

Cooling is accomplished both by the air drawn through the baffles, and by the air aspirated through the inlet, which is about 60% greater in diameter than the jet exit. No deterioration of the muffler was observed after running with the afterburner lit.

The diffusion achieved is sufficient, on the basis of Lighthill's AV^2 law, to produce very large reductions (of the order of 30 db.) in the noise of the emergent jet. The actual reductions achieved are of the order of 15 db. at 100 ft. radius, and 45° from the jet axis. A typical sound polar is shown in Fig. A15. It shows the characteristic field of the unmuffled jet engine and the reduction achieved by the muffler - the peak SPL is reduced from 139 db to 127 db. The effect of the muffler on total sound power is shown in Fig. A16 - the power level is reduced by 9.3 db. in the afterburning condition. Additional development of the muffler is expected to produce further gains in performance.

A.2 - Boundary Layer Noise

Rigid Wall: Rotating Cylinder Investigation

The aerodynamic noise radiated by a turbulent boundary layer flowing over a rigid surface (one that cannot add to the sound by vibrating) is known to be relatively weak; thus it is easily masked by other noises, particularly in a wind-tunnel. An experimental arrangement that avoids this masking by other flow-produced noises utilizes the turbulent boundary layer on a rotating cylinder. An investigation of the noise produced by such a boundary layer has been under way for some time at UTIA, conducted by L.N. Wilson.

The rotating-cylinder arrangement is shown in Fig. A17. The cylinder is a thick-walled tube of aluminum 6" in diameter and 18" long, mounted in journal bearings. It is belt driven and has been run at speeds up to 16,000 rpm. The heavy supporting structure (Fig. A17) was enclosed in a heavy plywood sound-isolating box filled with sand for the tests reported below. The cylinder assembly is installed in a reverberant chamber that can be calibrated for measurements of total sound power.
A typical near field spectrum with a single microphone (Fig. A18) exhibits a number of sharp peaks projecting above a broadband continuous spectrum. These peaks occur at multiples of the rotation frequency. Cylinder unbalance and slight out-of-roundness would appear to account for such harmonic peaks. These effects and bearing imperfections, can lead to rigid-body modes of motion of the cylinder.

The sound pressures associated with these modes ("bearing noise") would be expected to be correlated over large areas of the cylinder whereas that from boundary layer turbulence would be correlated over relatively small areas, the size of an "eddy". Therefore two microphones were placed close together near the cylinder surface and one-half the square of the difference between the signals was measured. The correlated 'bearing noise' should then cancel out, leaving only the uncorrelated noise, which is presumably the true boundary layer noise.

Such two-microphone (anti-correlation) near-field measurements are shown in Fig. A18 together with single microphone measurements. One curve refers to the cylinder in the smooth condition, using two microphones and the others to the cylinder roughened with aluminum oxide particles of about .007" to .009" height (No. 80 Alundum grains) using one and two microphones. The particles were sprayed on the surface in a shellac vehicle. It is clear that roughening raises the levels substantially, as might be expected. The wide-band noise between the harmonic peaks can be identified as true boundary layer noise because of the agreement there of the single and double-microphone curves. The peaks, which appear only on the single-microphone curve, are the extraneous "bearing noise".

The rough-wall curve of uncorrelated pressure (true boundary layer noise) Fig. A18, has been replotted in Fig. A19 on the basis of db/cycle vs. nondimensional frequency fD/U (Strouhal number). This and other curves obtained for speeds from 7000 to 13000 rpm have been adjusted to the same effective speed (10,000 rpm) and superposed.

The relation $p^2 \sim U^4$ assumed in the adjustment is confirmed by the collapse of the points to a single curve in Fig. A19. Cross-plots of db/cycle vs. U for fixed fD/U further confirm a $U^4$ law. Apparently, then, the near-field boundary-layer noise is essentially the hydrodynamic (i.e. incompressible) pressure field associated locally with the eddying flow, for which $p^2 = \rho^2 U^4$.

Blokhintsev, who has discussed such pressure fields, calls them "pseudo-sound" (Ref. A10) (see also Ref. A22).
Boundary Layer Noise Theory for Flexible Skin

The weak noise radiated by turbulent flow past a rigid wall - e.g., the rotating cylinder described above - is greatly augmented if the wall can vibrate, like the skin of an airplane. At high subsonic speeds such skin vibration is, in fact, reported to be the major source of noise within the aircraft (Ref. A23).

A theoretical study has been made (Ref. A24). The skin vibration is considered to be excited by the fluctuating hydrodynamic pressures in the turbulent boundary layer. These 'pseudo-sound' pressures greatly exceed the associated compressibly-generated pressures that are radiated as sound in the rigid-wall case; this accounts for the amplified radiation provided by the sounding-board effect of a flexible wall.

The fluctuating pseudo-sound pressure distribution can be decomposed by Fourier methods into a pattern of sinusoidal pressure waves with various angles of yaw (Fig. A20). The pattern is idealized as rigid and moving uniformly by convection (the pressure fluctuation at any point is thus caused by the motion). A running ripple in the skin follows underneath each wave, and the noise is ultimately due to these ripples. The acoustic effects of the running ripples have been determined for an infinite plane sheet. Supersonically moving ripples radiate strong sound in the form of Mach waves (Fig. A21); subsonically moving ripples generate no sound unless the sheet is finite (or the ripples unsteady).

For an airplane fuselage, however, the infinite plate is replaced by a succession of finite panels. Successive panels are considered to be statistically independent because the running waves are interrupted by the frames and stringers supporting the skin. Moreover, multiple reflections at the frames and stringers convert the running waves into standing waves. An assumption is used to link the two kinds of waves, and this leads to provisional estimations of noise levels within aircraft. On this basis the mean square noise pressure is predicted to vary as $U^2 \delta^5 / h^3 \rho$ for thin boundary layers, changing progressively to $U^2 \delta^5 / h^7 \rho$ for thick layers or high speeds ($U =$ flight speed, $\delta =$ boundary layer thickness, $h =$ panel thickness, $\rho =$ panel damping coefficient). A common factor $\delta^5 / h^3 \rho$ has been omitted for simplicity from both formulas where $\rho =$ interior air density, $\rho_e =$ exterior air density, $\rho_p =$ panel density.

It is clear that increasing the panel damping $\rho$ is a powerful means for reducing the noise. For the fixed value $\rho = .01$ the noise level formulas are illustrated in Figs. A22 and A23. Shown also on Fig. A23 are some experimental data for actual aircraft, adjusted to apply to the same pressure altitude. The close agreement with the $\delta = 10''$ curve is perhaps fortuitous in view of the uncertain correspondence of the parameters (e.g. $\rho$ and $\delta$) and the approximations in the theory.
Boundary Layer Noise Experiments with Flexible Skin

An acoustically quieted air duct facility has been constructed at the Institute of Aerophysics, University of Toronto (UTIA) for the purpose of investigating noise generated by turbulent flow past a flexible panel; this is essentially the "boundary layer noise" the theory of which was discussed in the last section. In this duct the panel is to be fitted in a cutout portion of one wall flush with the inner surface.

The facility is basically an open circuit acoustically lined wind tunnel with a 33 foot duct section (Figs. A23, A24). Any one of four interchangeable sections can be used, with respective inside cross sections 12" wide and 8", 4", 2" or 1" deep to provide fully developed turbulent channel flow before reaching the test section. The test section passes through a reverberation chamber for measurement of noise power.

Maximum air speed with the 10 hp blower (4" duct) is 200 fps. Details of the design and aerodynamic performance - e.g., velocity profiles and pressure gradients - are given in Ref. A24.

A set of exploratory noise measurements have been made with a steel panel 12" x 12" x 0.002" installed horizontally in the 8" duct. No great care was taken, and the thin panel as finally tested (crudely supported in a wooden frame) contained numerous irregularities or wrinkles in the surface. Sound pressure level readings were taken at heights z = 1 1/2", 6", 12", 30", 36" and 42" vertically above the centre of the panel. The readings in db (re .0002 microbar) are plotted vs. air speed U at the duct inlet in Fig. A26. The curves seem not to be inconsistent with the theory (Ref, A23) which predicts a U^5 law at low speeds with a transition to a U^3 law at high speeds. The theory is inadequate, however, in its present form, to predict quantitatively the position of the observed knee of the curve.

Tentative measurements have also been made on a second .002" steel panel carefully mounted under tension in a machined steel frame and free of wrinkles. Spectra and overall pressure levels for various air speeds are given in Fig. A27. The overall levels approximate a U^5 law. It is thought that the absence of a transition to a lower power may be due to the initial tension in the panel: the increased effective stiffness simulates the properties of a thicker panel without tension, for which the transition point theoretically occurs at a higher air speed.

A.3 - Aeolian Tones

One of the goals of research in flow noise has been the attainment of quantitative experimental verification of the theory. The classical phenomenon of Aeolian Tones, being relatively simple, offers an attractive possibility for such a verification, and therefore has received some attention both in Canada and the U.K. Aeolian tones are
the almost-pure notes emitted by a wire or circular cylinder placed in an airstream; they are closely associated with the periodic shedding of vortices into the wake.

A theoretical and experimental investigation was started at the UTIA in 1955 (Refs. A25, A26, A27, A28). Initially it comprised an adaptation of Lighthill's theory (Ref. A2) for application to the case in hand, and some measurements of the intensity and frequency of the radiated sound. It was soon found that there was insufficient information available on the unsteady pressures and forces acting on a cylinder to provide a good quantitative check on the theory. The program therefore continued with measurements of the pressures, the two-point correlations of the pressure, and the forces. Some details of the theoretical and experimental work follow.

**Theory** - Lighthill's equations (Ref. A2) could not be applied directly to the problem because of the presence of a solid boundary in the flow (the surface of the cylinder). The equations were adapted to apply to this case by imagining that the solid cylinder is replaced by a column of fluid which is maintained at rest by a suitable distribution of body forces. The radiated sound field is then found to be that of a distribution of quadrupole sources associated with the turbulence in the wake, and of dipole sources associated with the body forces. The latter are uniquely determined by the surface pressures. This result is exactly the same as that obtained by application of Curle's theory (Ref. A29). It was assumed that in the experiments the quadrupole sound would be negligible compared to that from the dipoles at the cylinder surface. The theory then predicts that the principal radiation is a note at the fundamental frequency of the wake, radiating as a dipole with its axis cross-stream. This note is associated with the alternating lift force on the cylinder. A second note, at double the frequency, and associated with the drag fluctuations, radiates as a dipole with axis in the stream direction. The intensity of the sound in the far field varies approximately as the sixth power of the speed, and depends upon the magnitudes of the fluctuating forces and on their correlation along the length of the cylinder. No theory exists for predicting these forces and correlations.

**Measurements of Intensity and Frequency**

The experiments were carried out in the UTIA subsonic wind tunnel. A number of cylinders were used, varying from 1/8 to 1 3/4 inches in diameter, and the speed range was from about 100 fps to 225 fps. The results of the frequency measurements are shown in Fig. A28. They are in general agreement with results obtained by other investigators.

Intensity measurements were made with the microphone both inside and outside the tunnel. A typical spectrum with the microphone upstream of the cylinder is shown in Fig. A29. The sound from the cylinder is seen to be significantly louder than the tunnel background
and the peaks at the fundamental and second harmonic frequencies are clearly evident. Interpretation of these intensity measurements is somewhat difficult, because of an unknown reverberation effect. However, the order of magnitude is given by these measurements.

**Measurements of the Pressure**

The pressure fluctuations at the cylinder surface were measured by means of a condenser microphone installed internally. Typical results are shown in Figs. A30, A31, A32. These show the nature and magnitude of the rms pressure, and how it varies around the circumference. The fundamental component dominates at the sides, and the second harmonic at the back. This is consistent with the presence of a lift at the fundamental frequency, and a drag at double that frequency. The curve labelled "theory" in Fig. A32 was obtained from a simple potential-theory model of the flow incorporating a periodic circulation (Ref. A26). It gives the shape of the pressure curves with rather surprising agreement.

**Measurement of the 2-point Correlations**

A series of measurements were made of the correlations of the pressure along the cylinder. Condenser microphones were used as pressure transducers, and the quarter-square method was used to obtain the correlation coefficients. A typical result, with the holes at 90° to the stream, is shown in Fig. A33. The correlation does not fall off to zero at large hole separations as was expected. Further experimentation is required to explain this result. Until such an explanation is forthcoming, it has been tentatively assumed that the effect is caused by an unknown extraneous influence. (Possibly a tunnel wall effect). The correlation curve appropriate to free-field conditions has therefore been assumed to be given by the one shown dotted in Fig. A33. The area under this curve defines the effective correlation length. Fig. A34 shows how this correlation length varies with Reynolds number.

**Measurement of the Forces**

A strain gage transducer has been designed and built for the measurement of the forces acting on a short section of the cylinder (one diameter in length). It is shown in Fig. A35. The transducer has been found to have satisfactory linearity, sensitivity, and repeatability. In order to calibrate it over the whole frequency range it has been necessary to use both mechanical and acoustical means to supply the calibrating forces. The acoustical loading required the development of a special calibration device.

Fig. A36 shows the results obtained so far with this instrument. It can be seen that they are in fair agreement with those estimated from the pressure distributions, and with those of Phillips, obtained at a much lower Reynolds number. The results of Bingham
et al (Ref. A31) are in strong disagreement, however. Unpublished results obtained by Gerrard at Manchester are also in disagreement with those shown here. Gerrard's experiments were similar to McGregor's (Ref. A26), and this disagreement has not yet been explained.

Comparison of Theory and Experiment

The calculated values of the sound pressure in the far field are compared with measured values in Fig. A37. The data is presented in the form $p^*$ as a function of Reynolds number. $p^*$ is a non-dimensional sound pressure given by

$$p^* = 4\sqrt{2} \frac{a_0}{\rho U^3} \frac{r}{ld}$$

where $p$ is the rms sound pressure, $a_0$ is the speed of sound in the undisturbed air, $\rho$ is the density, $U$ is the speed, $r$ is the distance from the cylinder to the microphone, and $l$ and $d$ are cylinder length and diameter. The calculated value is given by

$$\sqrt{2} \lambda (C_L)_{rms} S_t$$

where $\lambda$ is the effective correlation length in diams. $(C_L)_{rms}$ is the rms value of the lift coefficient, and $S_t$ is the Strouhal number. The values of these quantities used for the calculated curve are those obtained in the UTIA measurements and those given by Phillips (Ref. A30). The intensity measurements shown are those of Keefe, Gerrard, and Phillips.

A.4 - Underwater Noise

Studies of sound propagation in water have been carried out at the Pacific Naval Laboratory of the Defence Research Board. These have involved some investigations of flow noise, particularly in relation to the use of hydrophones being towed through the water. The noise associated with the wakes of connecting cables was found to be important. Studies have also been made of the noise produced by the flow over the hydrophone itself, and of the noise recorded by one hydrophone mounted in the wake of another.

A.5 - Noise Induced Fatigue

Effect of Boundary-Layer Noise

A previous section discussed how the turbulent boundary layer covering much of an airplane in flight can give rise to substantial noise in the interior by exciting vibration in the skin. The question has been raised whether the skin stresses associated with these vibrations are sufficient at the higher speeds to bring in the possibility of fatigue failure. A theoretical investigation of these skin stresses was therefore undertaken at UTIA under the sponsorship of Avro Aircraft, Canada, Ltd. (Ref. A32).
For all but very thick boundary layers the analysis was based on the idealized boundary-layer noise theory discussed earlier (Ref. A23). The procedures were modified and adapted to yield a tentative expression for the mean square stress in the skin as a function of flight speed, boundary layer thickness, panel thickness, panel fundamental resonant frequency, damping coefficient, etc. A numerical example was worked out for .032" thick dural skin panels of three sizes (fundamental frequencies 37.5, 150, and 600 cps., respectively), in which the flight speed was varied from 400 fps to infinity with neglect of air damping and certain idealizations of the boundary layer. The boundary layer thickness was held constant at about 4 inches.

The rms stress for these cases was found to be nearly constant over this entire speed range, showing a flat maximum in the range 700 to 1100 f.p.s. (see Fig. A38). The maximum was far below typical 'infinite life' endurance limits for dural type alloys, (e.g. 10,000 p.s.i. for an unfavorable static loading), never exceeding 170 p.s.i. A supplementary study for extremely thick boundary layers (turbulence scale comparable with panel dimensions) was made by a one-dimensional approach developed from Miles ideas (Ref. A33). Indications of rms stress exceeding 10,000 psi were found for boundary layers several feet thick at supersonic speeds.

**Effect of Jet Noise**

Experimental and theoretical investigations of noise-induced fatigue of built-up structures and simple flat panels have been undertaken at Avro Aircraft. The experimental work has been directed both at obtaining ad hoc fixes, and at verifying the theoretical calculations. The theoretical work has been directed at predicting the life of flat panels subjected to plane waves, normally incident.

The testing has been carried out in a specially designed enclosure, using a siren to provide acoustic pressures of the order of 170 db. The tests conducted on built-up structures indicated that the failures tended to occur in the supporting structure, at joints in the skins through rivet lines (see Fig. A39). Generally, effective means of increasing the life were found to be: reduction of panel size, application of doublers (see Fig. A44) and introduction of additional damping.

The theoretical work (Ref. A34) has indicated that thin flat panels subjected to intense acoustic loading will achieve large enough amplitudes that both "plate-like" and "membrane-like" displacements and stresses coexist. The result is a stress-time curve such as that shown in Fig. A40(a). A non-linear theory has been developed to describe this phenomenon (Fig. A40(b)).

Comparison of Figs. A40(a) and (b) shows that the qualitative features of the panel behaviour are correctly predicted. A
comparison has been made of the calculated peak-to-peak stress interval with measured values. The result is shown on Fig. A41, and the agreement is seen to be very good. The comparison of measured and predicted fundamental panel frequency is not so good, however (Fig. A42).

The theory is applied to predict the fatigue life of flat panels subjected to periodic loading - an example is shown in Fig. A43. No experimental confirmation of these predictions is yet available. A result of some importance is the prediction that the use of a damping-tape plus edge doubler system (Fig. A44) can increase the sound pressure level for infinite life by 30 db.
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FIG A1 COMPARATIVE NOISE POWER DISTRIBUTION OF A SINGLE JET AND A MULTIPLE-JET OF THE SAME TOTAL THRUST.
FIG. A2  MULTIPLE-NOZZLE JET MUFFLER PLUS
SOUND-ATTENUATING SHROUD
FIGURE A3. SOUND REFLECTION AND TRANSMISSION BY A MOVING MEDIUM IN TERMS OF MACH WAVES GENERATED BY RIPPLES RUNNING ALONG THE BOUNDARY.
FIGURE A4 ANGLE OF REFRACTION $\alpha''$ VS. ANGLE OF INCIDENCE $\alpha$ FOR A SERIES OF MACH NUMBERS OF THE MOVING MEDIUM
FIGURE A5. CONTOURS OF CONSTANT REFLECTION COEFFICIENT, $R$ IN A GRAPH OF ANGLE OF INCIDENCE vs MACH NUMBER OF THE MOVING (SECOND) MEDIUM.
FIGURE A6  PLANE SOUND WAVE IN A HORIZONTAL STREAM. THE PLANES OF CONSTANT PHASE MOVE WITH THE 'PHASE VELOCITY' $\vec{V}_f$ DIFFERENT FROM THE ORDINARY SPEED OF SOUND $c$. THE ACOUSTIC ENERGY FLOWS WITH THE 'RAY VELOCITY' $\vec{V}_s$. 
FIGURE A7  PLANE SOUND WAVES IN A MOVING STREAM: EFFECT OF STREAM VELOCITY $U$ ON PHASE VELOCITY $V_f$ AND RAY VELOCITY $V_s$. ACOUSTIC ENERGY DENSITY $\sim V_f$; ACOUSTIC ENERGY FLOW $\sim V_f$ TIMES $V_s$. 
**FIG A8** Synthesis of vortex from radially disposed shear flows (physical interpretation of Fourier integral).

**FIG A9** Convection of vortex through shock wave, I: focusing of the refracted shear waves.
Fig. A 10  CONVECTION OF VORTEX THROUGH SHOCK WAVE, II: FORMATION OF ENVELOPE BY PLANE SOUND WAVES GENERATED AT SHOCK
Upper and lower bounds to radial pressure profile of cylindrical sound wave. (Shaded area for $\sigma > 1$ gives trend only, not bounds.)
FIG A12  AXIALLY SYMMETRIC MUFFLER MODEL
FIG A13 VELOCITY PROFILES
FIGURE A14 FULL SCALE SILENCER
FIGURE A15 POLAR DIAGRAM OF CF100/ORENDA 11R NOISE FIELD WITH AND WITHOUT SILENCER
FIGURE A16  SOUND POWER LEVEL VS. ORENDA IIR
THRUST WITH AND WITHOUT SILENCER

THRUST - lb

SOUND POWER LEVEL - db (re 10^{-13} watts)

- x AFTERBURNING
- O UNSILENCED
- * (136 kW)
- O (16 kW)

SILENCED
FIGURE A17

ROTATING CYLINDER AND BOUNDARY LAYER TRAVERSE GEAR
FIGURE A18 SPECTRA OF NEAR FIELD BOUNDARY LAYER NOISE PRESSURES FOR SMOOTH AND ROUGH 6'' ROTATING CYLINDERS, MEASURED WITH A SINGLE MICROPHONE ($p_1^2$) AND WITH ANTICORRELATED MICROPHONES $\frac{1}{2}(p_2 - p_1)^2$ SPACED 7/8'' APART. DISTANCE FROM SURFACE 2''. 3% BANDWIDTH ANALYSIS.
FIGURE A19—SPECTRUM OF NEAR-FIELD BOUNDARY LAYER NOISE FOR ROUGH CYLINDER WITH ANTI-CORRELATED MICROPHONES 2 in. FROM SURFACE. DATA FOR 7000 rpm TO 13,000 rpm ADJUSTED TO 10,000 rpm ON BASIS $\overline{P^2} \sim U^4$
FIG A20 - MOVING PRESSURE WAVE. AN ASSEMBLAGE OF SUCH WAVES OF ALL ORIENTATIONS, WAVELENGTHS, AND SPEEDS CAN REPRESENT A RANDOM FLUCTUATING DISTRIBUTION OF SURFACE PRESSURE.

FIG A21 - GENERATION OF MACH WAVES BY FLEXURAL WAVE TRAVELLING WITH SUPersonic SPEED ALONG PLATE. (UPPER FAMILY OMITTED)
Boundary-Layer Thickness, $\delta$, in.

FIG A22  BOUNDARY-LAYER-INDUCED NOISE PRESSURE LEVELS CALCULATED FOR EXAMPLE AIRPLANE ALONG CENTERLINE OF AFT END OF FUSELAGE. HIGHLY REVERBERANT CONDITIONS, NO INSULATION. ALTITUDE 18,000 FT., FUSELAGE PRESSURIZED TO 8,000 FT.
FIGURE A 23 BOUNDARY-LAYER-INDUCED NOISE PRESSURE LEVELS CALCULATED FOR EXAMPLE AIRPLANE ALONG CENTERLINE OF AFT END OF FUSELAGE. HIGHLY REVERBERANT CONDITIONS, NO INSULATION. ALTITUDE 18,000 FT., FUSELAGE PRESSURIZED TO 8,000 FT. EXPERIMENTAL POINTS ARE CORRECTED FOR ALTITUDE.
FIGURE A24

GENERAL ARRANGEMENT

BOUNDARY-LAYER-NOISE DUCT
\[ \frac{1}{3} \text{ OCTAVE ANALYSIS} \]

DISTANCE FROM MICROPHONE TO PANEL \( z = 16 '' \)

8'' DUCT

.002'' PANEL

OVERALL LEVELS

FIGURE A27. FAR FIELD MEASUREMENTS OF ACOUSTIC RADIATION FROM .002 INCH STEEL PANEL.
FIGURE A.28

VARIATION OF FUNDAMENTAL STROUHAL NUMBER WITH REYNOLDS NUMBERS

- UTIA - 1955
- Roshko - 1953
- Kovasznay - 1949 (after Lehnert)
- Relf - 1924
- DVL Hiebtone - 1919 (after Lehnert)
- Strouhal - 1878 (after Lehnert)
FIGURE A 29

SPECTRUM OF CYLINDER NOISE

$\frac{1}{3}$ octave band analysis uncorrected for microphone response, $U = 225$ ft./sec., microphone in settling chamber (Fig. 10, Configuration C)

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<tr>
<td>O - 100</td>
<td>cylinder in tunnel</td>
</tr>
<tr>
<td>O - 101</td>
<td>tunnel empty</td>
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FIG A30  OSCILLOGRAPH PICTURES OF OVERALL PRESSURE FLUCTUATIONS AT VARIOUS LOCATIONS AROUND THE CYLINDER. U = 75 ft/sec., BAND WIDTH: ZERO SELECTIVITY
FIG A31

PRESSURE SPECTRA AT VARIOUS LOCATIONS AROUND THE CYLINDER. U = 211 FT./SEC. BAND WIDTH: 1/3 OCTAVE
FIG A32 COMPARISON OF THEORETICAL AND EXPERIMENTAL PRESSURES.

$C_{p_{fun}} = \text{r.m.s. value of pressure fluctuations at fundamental frequency} \approx (1/2) \rho U^2$
FIGURE A33

2 POINT PRESSURE CORRELATION ON A 1 IN. DIA. CIRCULAR CYLINDER.  U=100 fps, Re=60,000.
FIGURE A 34

CORRELATION LENGTH vs REYNOLDS NUMBER
FOR A 1 IN. DIA. CIRCULAR CYLINDER.
FIGURE A35 STRAIN GAUGE BALANCE FOR MEASURING UNSTEADY COMPONENTS OF LIFT AND DRAG
FIGURE A36: RMS LIFT COEFFICIENTS
FIGURE A37. INTENSITY OF AEOLIAN TONES
FIG A38 Skin stress associated with boundary-layer noise according to example computations based on an idealized model. Dural skin 0.032" thick, boundary layer 4.27" thick, three sizes of panel (specified by resonant frequency).
FIGURE A39: FATIGUE CRACK THROUGH RIVET LINE (ACOUSTIC LOADING)
FIGURE A40. CYCLIC VARIATION OF FIBER STRESS AT CENTRE OF PANEL WITH AMPLITUDE
THEORETICAL STRESS FOR SAWTOOTH INPUT ($\mu = 0.31$)

ESTIMATED RANGE OF STRESS TO BE EXPECTED FROM TEST DUE TO VARIATION IN TEST INCIDENT WAVE FORM

- PANEL 8-CENTRE GAUGE
- PANEL 8-GAUGE AT $x/D = 0.042$
- PANEL 12-CENTRE GAUGE

FIGURE A41 COMPARISON OF THEORY WITH TEST RESULTS
FIGURE A42 COMPARISON OF THEORY WITH TEST RESULTS — FUNDAMENTAL RESONANT FREQUENCY VS. SPL

- THEORETICAL FREQUENCY FOR PURE SAWTOOTH INPUT ($\mu = 0.31$)
- RANGE TO BE EXPECTED FROM TEST DUE TO VARIATION IN TEST INCIDENT WAVE FORM

CLASSICAL LINEAR THEORY
FIGURE A43. CALCULATED TEST PANEL LIFE VERSUS INCIDENT SOUND PRESSURE LEVEL

SCATTER BAND FOR PURE SAWTOOTH WAVE
ESTIMATED SCATTER BAND FOR TEST CONDITIONS
THIN LAYER OF
VISCO-ELASTIC MATERIAL

DOUBLER

PLATE

FIGURE A44
DOUBLER-PLUS-TAPE DAMPING SCHEME
This review consists of material assembled for presentation at the "Jet Engines and Noise" session of the ICAS meeting in Madrid, Sept. 1958. Canadian research on flow noise and some aspects of the aircraft noise problem is described. The work was done at the Defence Research Board, the University of Toronto Institute of Aerophysics and A.V. Roe (Canada) Ltd. Specific experimental and/or theoretical investigations include: Aeolian Tones; Boundary Layer Noise (rigid wall and flexible wall); Effects of Boundary Layers and Noise on Aircraft Structures; Distribution of Noise Sources Along a Jet; Ground Run-up Mufflers; Transmission of Sound from, and Acoustic Energy Flow in, a Moving Medium; Sound Generated by Interaction of a Vortex with a Shock Wave.

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