Slipstream-Induced Pressure Fluctuations on a Wing Panel

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Propeller-induced pressure fluctuations have been measured on a wind-tunnel model. The results show that the main contribution on the wing panels can be attributed to the propeller tip vortex, which gives a pressure level at least 20 dB above the level from the inner parts of the propeller. The pressure fluctuations are predominantly periodic and the spectrum shows strong peaks at the blade passage frequency and its harmonics. The pressure level at the blade passage frequency is approximately the same on wing panel and fuselage, while the level of the higher harmonics is substantially higher on the wing panel than on the fuselage.

Nomenclature

c_L	= propagation speed of quasilongitudinal plate wave, m/s
f_H	= hydrodynamic coincidence frequency, Hz
f_p	= blade-passage frequency, Hz
f_r	= propeller rotational frequency, Hz
f_{res}	= resonance frequency, Hz
f_0	= frequency at zero phase angle, Hz
f_1, f_2	= half-power frequencies, Hz
$G_{xx}(f)$	= autospectral density of pressure at transducer x, Pa^2/Hz
$G_{xy}(f)$	= cross power spectral density of pressures at transducers x and y, Pa^2/Hz
h	= panel thickness, m
n	= harmonic number
n_d	= number of averaging segments in spectral estimate
Q	= magnification (quality) factor of resonator $= f_{res}/f_2 - f_1$)
U	= freestream speed, m/s
v	= convection speed, m/s
α	= angle of attack, deg
ϵ_r	= random error of spectral estimate
ρ	= coherence function
$\bar{\rho}$	= one-third octave average of coherence function
θ	= phase angle, rad
θ_{xy}	= phase angle of cross power spectral density for transducers x and y, rad

Introduction

ALTHOUGH there is significant literature on the technical aspects of noise in propeller-driven aircraft, little is devoted to the field of structure-borne noise compared with the vast amount of information on airborne noise. This indicates that the problem of structure-borne noise is not very pressing at this time. However, there are reasons to believe that this state of affairs may be changing in the near future. One factor, and perhaps the most important one, is the indication that the aircraft industry now regards a low noise level in the passenger cabin as a significant marketing factor. As the conventional measures for low noise levels are aimed primarily at airborne noise, it is thought that the importance of structure-borne noise will increase in the future.

This paper is devoted to one particular aspect of structure-borne noise: the noise caused by the fluctuating pressures induced by the propeller slipstream on the wing of an aircraft with wing-mounted engines. No attempt will be made here to cover the whole problem; the paper is restricted to an account of the magnitude and distribution of the fluctuating pressures on a stiff wing, obtained from model-scale wind-tunnel measurements.

Even if the literature is scarce on the subject of slipstream-induced noise and vibrations, some material does exist. One important paper is that by Johnstone et al.,[1] where results from flight tests with a Lockheed P-3C are reported. In this work, the propeller signatures were used to identify the distributions from the four propellers to the total noise level in the cabin. It was then found that at the blade passage frequency (68 Hz), the contributions from the outboard propellers were substantially larger than what could be expected assuming airborne sound. This is, of course, a strong indication of the importance of the structure-borne sound.

Another series of full-scale experiments has been described by Metcalf and Mayes.[2] The aircraft in this case was a de Havilland DHC-6 Twin Otter without trim panels. The measurements were carried out during ground run-ups with one engine at different torque levels. Sound insulation material was applied step-by-step to the relevant parts of the fuselage and the wing surfaces. This procedure made it possible to quantify the different contributions from airborne and structure-borne paths, respectively. The results indicate that the structure-borne noise gave a significant contribution to the interior noise. It was also shown that the structure-borne noise could be attributed to the propeller wake and tip vortex interaction with the wing surface.

Model-scale measurements on a prop-fan-type propeller in a wind tunnel have been reported by Miller et al.[3] This propeller model has eight swept blades and a diameter of 0.61 m. Pressure transducers were mounted on both sides of an airfoil-shaped vane, which was movable in the radial direction, and with the transducers located one propeller diameter downstream of the propeller. Results are presented for tunnel Mach numbers of 0.6 and 0.8. At Mach 0.6, where the propeller tip is operating subsonically, the measured fluctuating pressure (rms) on the advancing propeller side of the vane shows a sharp peak at a radial position corresponding to the propeller tip, thus indicating the presence of a strong propeller tip vortex. A narrow-band spectrum is also presented. This spectrum shows strong peaks at the propeller blade-passage frequency and its harmonics. The roll-off at higher harmonics is rather slow; the level of the twelfth harmonic is less than 15 dB below that of the fundamental. The noise level measured at a distance of 1.5 times the propeller diameter from the propeller tip (in the propeller plane) was substantially lower than the level of the pressure fluctuations on the vane in the slipstream. At the blade-passage frequency, the difference was about 15 dB, still at a tunnel Mach number of 0.6.

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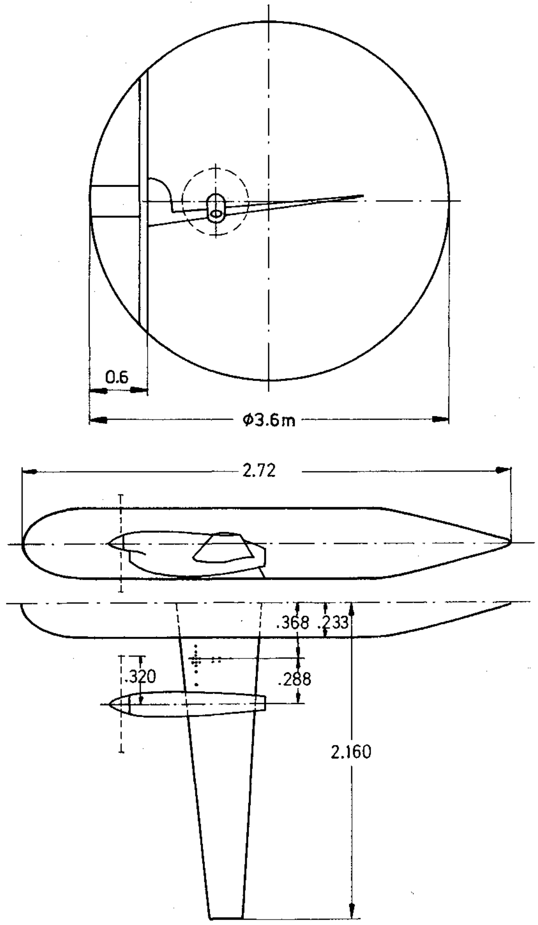

Fig. 1 Wind-tunnel model.

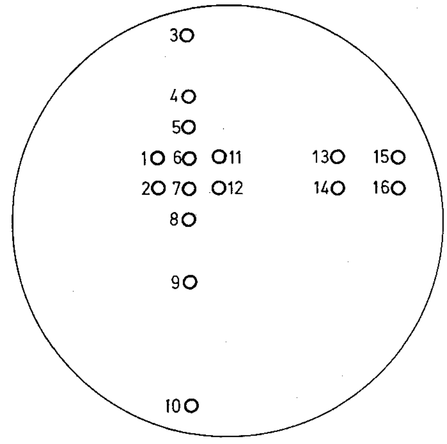

Fig. 2 Positions of wing-panel pressure transducers.

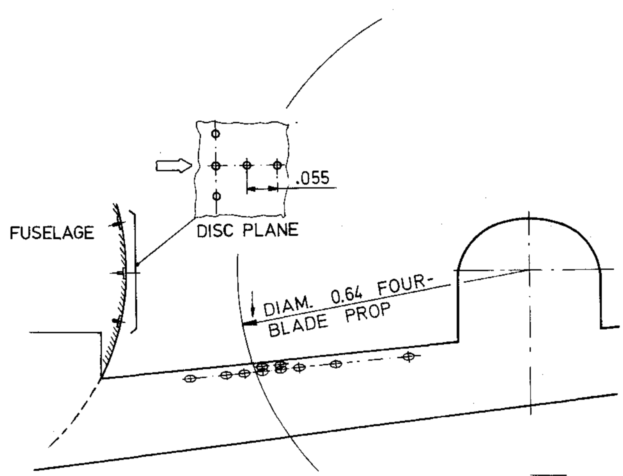

Fig. 3 Positions of fuselage pressure transducers.

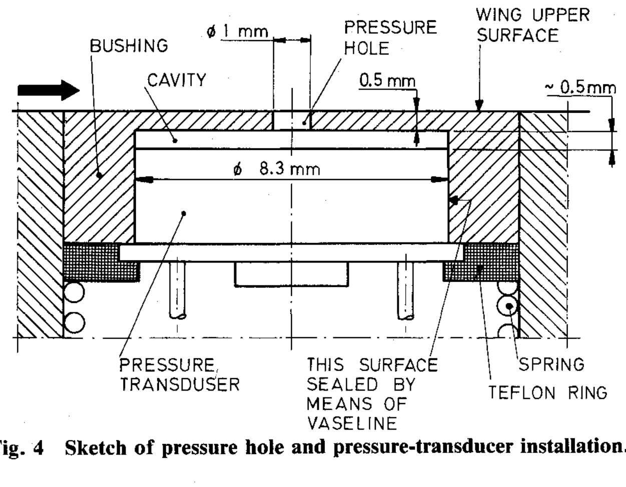

Fig. 4 Sketch of pressure hole and pressure-transducer installation.

An analytical examination of the whole problem of generation of the fluctuating pressures in the slipstream, excitation of the wing, and transmission of the vibrations and radiation into the cabin was presented recently by Junger et al.[4] Theoretical values of the forces acting on the wings are presented for the same set of parameters as in the investigation by Miller et al.[3] There is a fairly good agreement between the computed and measured results, but the theory seems to give a higher roll-off rate of the harmonic levels.

Experimental Arrangements

Although the present investigation is of a principal character, and it therefore would have sufficed to use a schematic test configuration, the work was performed on the Saab SF-340 commuter aircraft configuration. There were two reasons for this: First, a semispan model in scale 1:5 was available from an earlier high-lift development work. Also, a nacelle with a hydraulic drive motor, a balance for thrust and torque measurements, and suitable propellers were available. Second, there was to be an opportunity in the near future to carry out flight tests with the SF-340, thus making it possible to compare the wind-tunnel results with the full-scale results.

The measurements were performed in the FFA LT1 subsonic atmospheric wind tunnel. This tunnel has a closed return circuit and a closed, circular test section with a diameter of 3.6 m. Figure 1 shows the half-model installed in the wind tunnel and the main dimensions of the model.

The propeller is direct driven by a hydraulic motor delivering about 40 kW at 6650 rpm. A two-component (thrust and torque) balance is housed in the propeller hub. A slip-ring unit connecting the motor and the propeller shaft transfers the signals from the balance.

The test was carried out with one three-blade and one four-blade propeller. The three-blade propeller has a diameter of 0.68 m, whereas the diameter of the four-blade propeller is 0.64 m. The four-blade propeller corresponds to the original SF-340 propeller.

The upper wing panel between nacelle and fuselage and also the fuselage side were instrumented with high-response pressure transducers. There are 16 transducers in the wing panel positioned close to the wing surface (Fig. 2). They are arranged in two spanwise rows in order to pick up the highest impact effects of the propeller blade tip vortices. The fuselage side is equipped with five transducers of which three are positioned in the propeller plane (Fig. 3). The purpose of these fuselage transducers is to make possible the comparison of the different characters of the pressures on the wing and on the fuselage.

The transducers were of the semiconductor type and mounted in cavities just under a 1-mm hole drilled through the wing surface (Fig. 4). The original housings of the transducers were cut down to the size indicated in the figure.

Measuring Technique and Data Acquisition/Reduction

Pressure Transducers

As the transducer arrangement with a cavity and a pressure hole can be regarded as a Helmholtz resonator, a limited test was carried out, along with a calculation, in order to estimate the resonance frequency of the resonator. A compressed air jet was directed over the pressure hole and the transducer response was frequency analyzed. All of the transducers indicated Helmholtz resonance frequencies above 5 kHz. The calculation of this frequency gave $f_{res} = 8$ kHz with the nominal dimensions shown in Fig. 4. The frequency response further indicated that the resonator Q values were in the range of 3–10. According to Kinsler et al.,[5] the Q value for the Helmholtz resonator (with nominal dimensions) can be estimated to be about 70. This high value, however, is derived on the assumption that there are no losses in the hole neck other than those resulting from acoustic radiation. For the small-hole diameter used (1 mm), it is believed that viscous losses in the neck flow account for the large Q value discrepancy.

The lowest observed Q value was about 3. This would mean that the data obtained are not quite correct for $f > 3$ kHz, both for the spectral density values and the phase angle of the cross spectral densities.

The response of the pressure transducers to structure-borne sound was tested by comparing the signals obtained during a test with open holes and holes sealed with an aluminum tape. The level obtained with closed holes was in the relevant frequency range more than 30 dB below that obtained with open holes, which shows that the structure-borne sound contribution can be neglected in this case.

Data Acquisition

Since the interest in the present investigation was to measure the pressure variations on the wing surface and not the absolute pressures, the pressure transducer outputs were ac-coupled, making the transducers acting effectively like microphones. The transducer signals were amplified, filtered, and either multiplexed, A/D (analog to digital) converted, and stored directly in a computer or recorded on a 14-channel magnetic tape recorder.

In the computer direct storage mode, each signal was sampled with about 10,000 samples/s and stored directly in the computer main memory. The data were then transferred offline to a mass memory disk. With the tape recorder, the signals were recorded in a run for 30 s at 30 in./s tape speed.

The A/D conversion of the data was carried out without any sample-and-hold of the multiplexed signals, which meant that the data were not taken simultaneously, thereby causing phase shifts. However, the effect of this was corrected for in the subsequent calculation of the phase angles of the cross spectral densities.

Data Analysis

The scheme of calculations performed on the time-signal series data was as follows: after some data editing (i.e., removal of mean values, conversion to pressure units, etc.), the data are input to a FFT algorithm, the output from which is the desired spectral quantities for the time series (autospectral density, magnitude, and phase angle of the cross spectral density). The next step is to calculate a coherence function, denoted by $\rho(x,y;f)$ and defined as

$$\rho(x,y;f) = Re\{G_{xy}(f)\}/\sqrt{G_{xx}(f)G_{yy}(f)}$$

The coherence function is further analyzed in standard one-third octave bands (with center frequencies from 10–5000 Hz) group $\bar{\rho}$.

Each time series, consisting of 12,288 data points, was divided into 12 segments of 1024 data points each. The spectral quantities were then obtained as the average of these 12 segments. The resulting analysis bandwidth was about 10 Hz with 512 lines over the frequency range up to 5 kHz.

Sources of Error

Some of the possible sources of error in the pressure measurements are 1) nonlinear pressure transducer frequency response; 2) presence of aliasing in sampled data; 3) random errors in the spectral estimates; 4) pressure transducer calibration drifts; and 5) nonconstant propeller rpm, unequal blade setting, etc. Of these, the nonlinear transducer frequency response is thought to be the potentially most serious source of error. As pointed out before, the smallest observed Q value for transducer response is about 3. This particular value is also reported by Goldstein,[6] where, according to that author's experience, it is stated that the ratio of maximum pressure response at resonance frequency and the static response is about 3. It is also pointed out that this kind of pin-hole cavity arrangement is to be used for frequencies less than half the resonance frequency. Of course, the useful frequency range is also dependent on phase-angle-accuracy requirements. For instance, for a resonance frequency of 6 kHz and phase shift < 10 deg ($Q = 3$), the useful range would be < 3 kHz.

The presence of aliasing in the sampled data is thought to be of minor importance for the present test, since the signals had very little power contents above the Nyquist frequency. As can be shown (see, for example, Bendat and Piersol[7]), the random error ϵ_r in the spectral estimates is $\epsilon_r = 1/\sqrt{n_d}$, where n_d is the number of averaging segments. In the present case, the number of segments is rather low ($n_d = 12$), so $\epsilon_r = 0.29$, which, expressed in decibels, corresponds to $+1.1$ dB and -1.5 dB.

It was found, over the extended test period, that the (static) calibration of the pressure transducers varied somewhat. The variation of the calibration constants was about 10% and is thought to be of minor importance in the present investigation, since this amounts to less than 1 dB in the spectra.

The observed maximum variation of propeller rpm was rather small ($< 1/1000$ of the nominal rpm), which meant that the effect on, for example, harmonic peak values should be neglible.

The presence of side lobes in spectra at $\pm f_r$ from the harmonic frequencies, where f_r is the propeller rotational frequency, could be interpreted as an effect of unequal pitch setting of the blades. However, for the first harmonic the ratio of power contained in the side lobe to the power in the harmonic is typically of the order of magnitude 0.25% (-26 dB), which is a rather small amount. It is thus believed that the effect of unequal blade setting is small and that it does not significantly distort the spectral results.

Measurement Results

The major part of the measurements were performed with the four-blade propeller. For this case, the following parameters were evaluated: time histories and power density spectra from each transducer, magnitude and angle of the cross-spectral functions, $|G_{xy}(f)|$ and θ_{xy}, respectively, and the one-third octave-band coherence function $\bar{\rho}(f)$. No attempt was made at this stage to evaluate all possible combinations of transducer signals. For practical reasons, the evaluation was restricted to the following pairs: 1 with 6, 11, 13, and 15; 2 with 7, 12, 14, and 16; and 7 with 3, 4, 5, 6, 8, 9, and 10. All measurements with the four-blade propeller were carried out at five different angles of attack $\alpha = 3$, 4, 5, 6, and 7 deg.

Time histories and spectra were also evaluated for a set of transducers during the measurements with the three-blade propeller. In this case, α was chosen to be 5 deg, i.e., nominal climb.

All of the measurements were done with a freestream speed of 50 m/s in the wind tunnel.

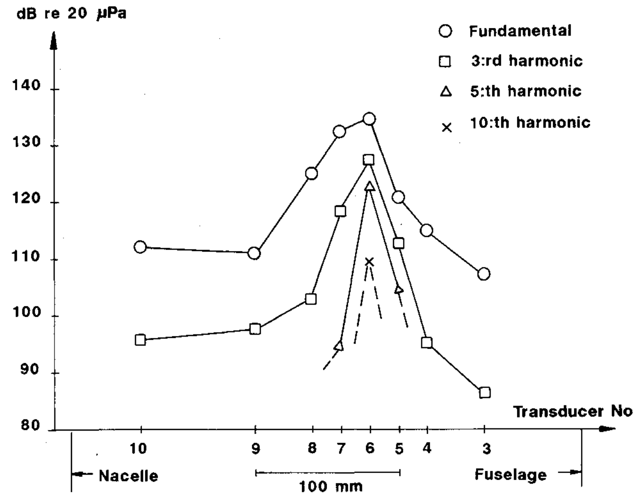

Fig. 5 Spanwise variations of the pressure measured on the wing; four-blade propeller ($\alpha = 5$ deg).

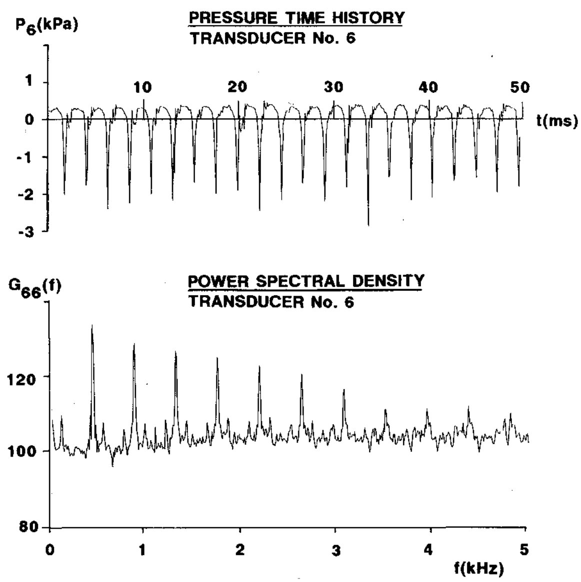

Fig. 6 Pressure-time history and power spectral density from transducer 6.

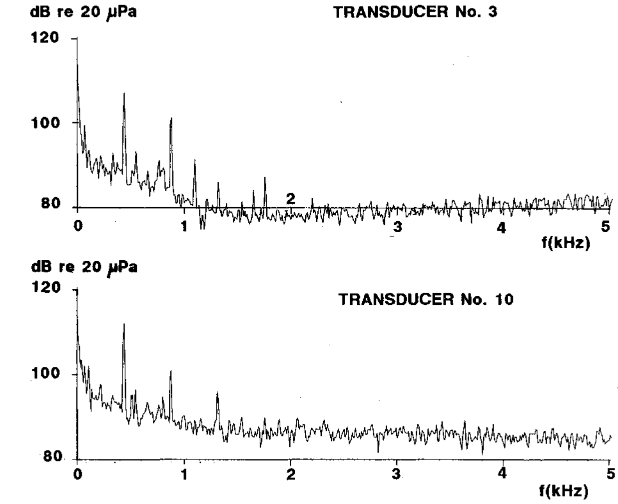

Fig. 7 Power spectral density at points 3 and 10.

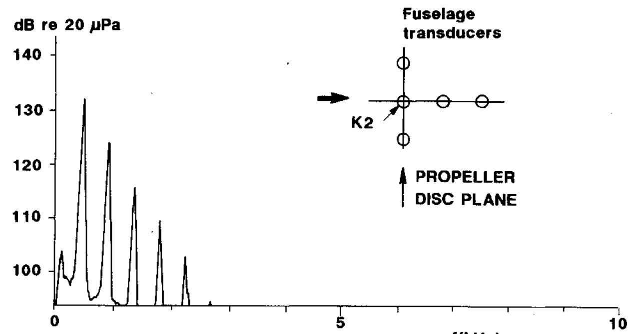

Fig. 8 Power spectral density of the sound measured on the fuselage by transducer K2.

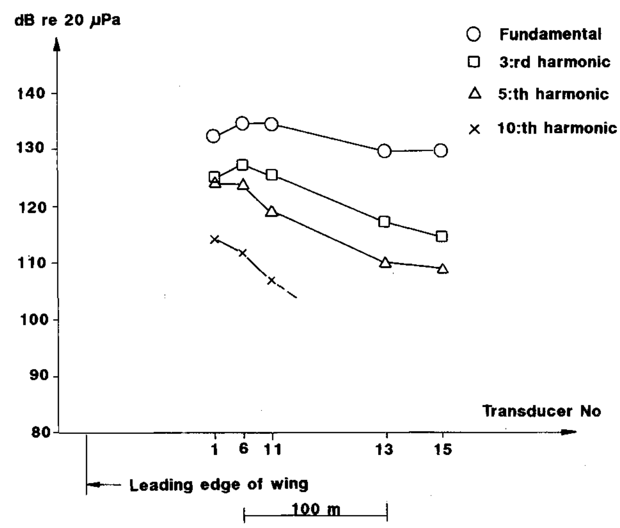

Fig. 9 Chordwise variations of the pressure measured on the wing.

Time Histories and Frequency Spectra

For the case of $\alpha = 5$ deg, the spanwise variations of the measured pressure (fundamental, 3rd, 5th, and 10th harmonic) are given in Fig. 5. These results show that the center of the propeller tip vortex meets the wing near transducer 6.

The time history of the pressure at transducer 6 and the corresponding spectrum are shown in Fig. 6. The time history shows a periodic behavior with very sharp downward peaks with a magnitude of about 2 kPa. The periodic character of the signal is again evident from the spectrum, where the fundamental has a level of about 30 dB above the noise floor and where the very sharp peaks in the time history correspond to a large number of harmonics with substantial levels.

The spectra obtained from the transducer closest to the nacelle (no. 10) and from that closest to the fuselage (no. 3) are given in Fig. 7 and for the same operating conditions as Fig. 6. Not only the fundamental but even more the harmonics have much lower levels at transducers away from the propeller tip vortex, which shows the importance of the propeller tip vortices with regard to the structure-borne noise.

It is interesting to compare these pressure levels with the corresponding sound pressure level at the fuselage. The highest level on the fuselage was obtained at the transducer K2 (about 40 mm from the propeller tip), and the corresponding spectrum is shown in Fig. 8. A comparison with the spectrum in Fig. 6 shows that the fundamental of the sound pressure has a level that, within a few decibels, is the same as the maximum level of the slipstream. There is, however, a large difference at higher harmonics. The fuselage sound pressure spectrum at higher frequencies rolls off faster than the propeller tip vortex induced wing pressure spectrum.

The chordwise variations of the measured pressure on the wing are illustrated in Fig. 9. These results are valid for $\alpha = 5$ deg, so this figure can be compared directly with Fig. 5. As expected, the variations in the chordwise direction are much smaller than in the spanwise direction. The time history is,

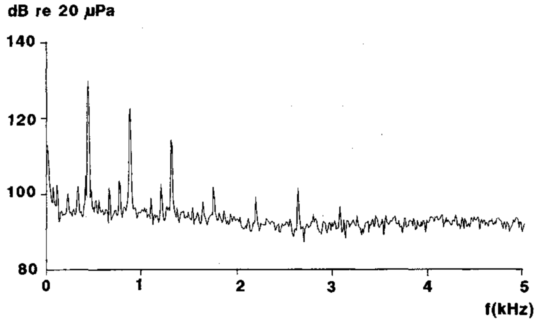

Fig. 10 Power spectral density from transducer 15.

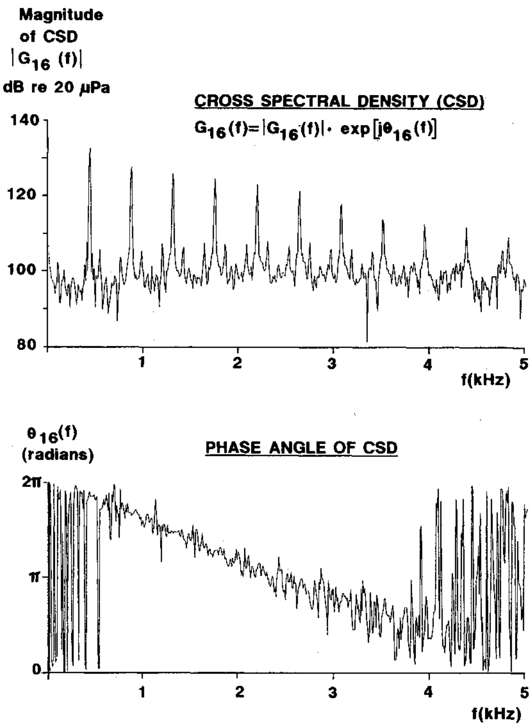

Fig. 11 Magnitude and phase angle of cross spectral density measured at points 1 and 6.

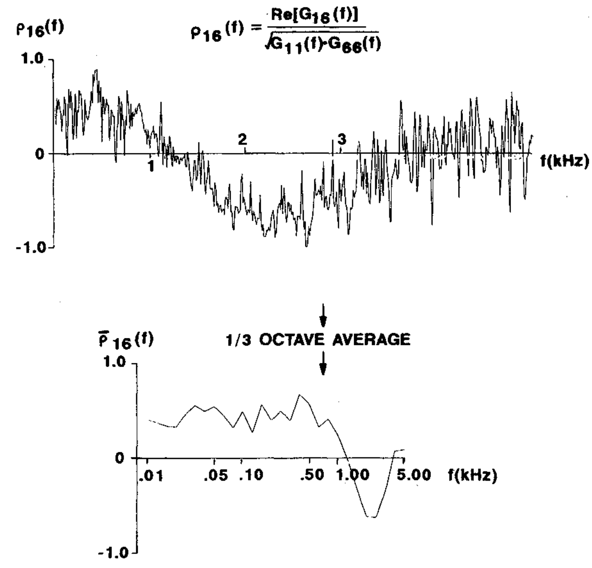

Fig. 12 Coherence function obtained from transducers 1 and 6.

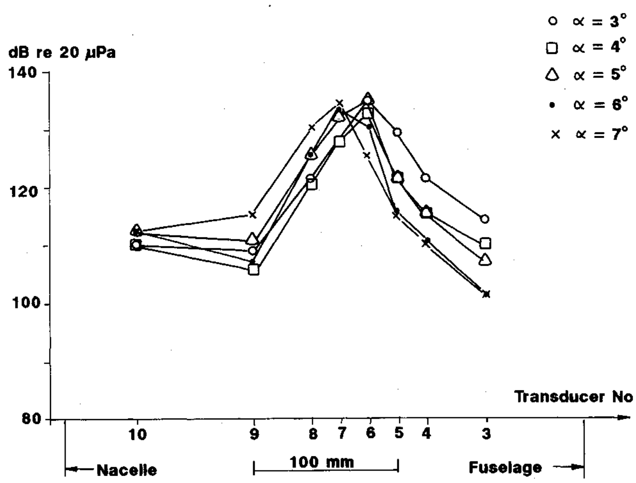

Fig. 13 Spanwise variations of the pressure measured at different angles of attack.

however, somewhat smoothed out during the chordwise transit over the wing, which results in a decrease in the level of the harmonics; see Fig. 10.

Cross-Spectral Density Functions

The magnitude and phase of the cross-spectral density (CSD) function for points 1 and 6 are given in Fig. 11 (point 1 is situated near the leading edge of the wing and near the center of the propeller tip vortex; point 6 is 20 mm downstream from point 1). The phase angle is specially interesting. In the middle part of the spectrum, the curve is rather close to a straight line that crosses the abscissa at 4.7 kHz and the ordinate at 2π. This straight line shows that the pressure field above the wing is convected downstream with an average speed of $4700 \times 0.02 = 94$ m/s.

The spectrum of the magnitude of the cross-spectral density function is very similar to the autospectra of transducers 1 and 6 (the spectrum of transducer 6 is given in Fig. 6). This fact, together with the possibility to ascribe a straight line to the phase angle, shows that it is meaningful to regard the fluctuating pressure field as a "frozen" convected pattern; see, for example, Ribner.[8]

An important parameter for describing the coupling between a load moving over a thin plate and the resulting bending vibrations in the plate is the hydrodynamic coincidence frequency f_H (see, for example, Strawderman[9]). This frequency can be calculated from the convection speed v as [10]

$$f_H = v^2/(1.8c_L h) \tag{1}$$

If the plate is made of aluminum with the thickness of 1 mm, the measured speed of 94 m/s corresponds to a hydrodynamic coincidence frequency of 970 Hz. The corresponding "acoustic" coincidence frequency is much higher, approximately 12 kHz. The presence of a convected frozen pattern together with a fairly low coincidence frequency shows that the coupling between the propeller wake and the bending vibrations of the wing panel can be comparatively strong.

Coherence Functions

Figure 12 shows the coherence function for transducers 1 and 6. The resolution of this spectrum is the same as in the power density spectra, i.e., 512 lines over the frequency range of 5000 Hz. The appearance of the curve immediately leads to the suggestion that a smoothed version could be useful. Such a smoothing is shown in the lower part of Fig. 12, where the coherence function is given in standard one-third octave bands.

This figure shows that the coherence function is rather far from 1.0 even at low frequencies. This may at first seem somewhat surprising in view of the very small change of the peaks

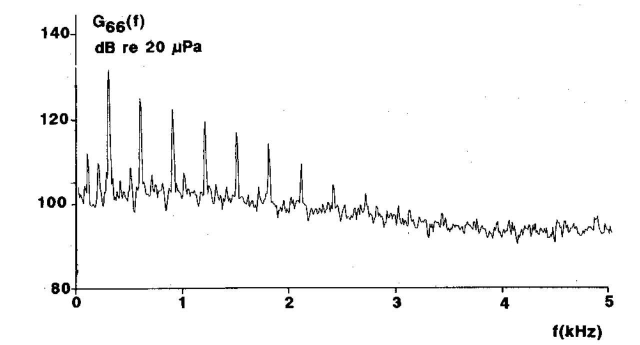

Fig. 14 Power spectral density measured at transducer 6 with a three-blade propeller.

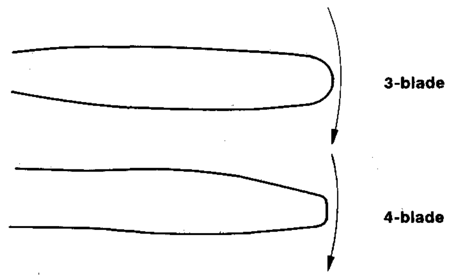

Fig. 15 Propeller tip shapes (three- and four-blade propellers, respectively).

Table 1 Measured levels in decibels re 20 μPa at the harmonics of the blade passage frequency

	Harmonic				
	1	2	3	4	5
Level on fuselage	133	125	117	110	104
Level on wing panel	134	129	127	126	123

in the power spectral density (PSD) during the chordwise passage, which has just been pointed out. It should be remembered, however, that the coherence function is normalized in such a way that every part of the spectrum in the relevant frequency band is equally important. As the bandwidth of the peaks in the power density spectra is very small, it follows that the frequency average of the coherence function is mainly determined by the coherence of the noise floors of the power density spectra.

An estimate of the correlation coefficient of the peaks can be calculated by normalizing the magnitude of the peaks of the cross spectrum with respect to the two power densities and then multiplying this expression with the cosine of the angle of the cross-spectral density. In the case of transducers 1 and 6, the normalized magnitude of the cross-spectral density is well over 0.9 for the peaks in the entire frequency range. As the angle θ of the cross-spectral density (see Fig. 11) can be approximated to

$$\theta = 2\pi(1 - f/f_0), \qquad f_0 = 4700 \text{ Hz} \tag{2}$$

it is clear that the coherence function for the harmonics of the blade-passage frequency as a first estimate can be taken as

$$\rho(f) = \cos(2\pi f/f_0), \qquad f = nf_p \tag{3}$$

Influence of the Angle of Attack

As the propeller axis of the model is situated above the wing, it could be expected, for geometrical reasons, that the propeller tip vortex should meet the wing at a point that is nearer to the engine nacelle if the angle of attack α is increased. Figure 13 shows that this is also the case. This effect also seems to be the most important one if α is changed; the influence of α on power density and cross-spectral density functions is small.

Different Propellers

In addition to the four-blade propeller, some measurements have also been performed with a three-blade propeller model. Apart from the different blade-passage frequencies, the main difference between the propellers is found in the more rapid trailing-off of the level of the harmonics of the three-blade propeller. As an example, the spectrum in Fig. 14 can be compared with that in Fig. 6 as the spectra are obtained from the same transducer (no. 6) and during approximately the same operating conditions. It is interesting to note that the tip of the four-blade propeller is more square than the smoothly rounded-off tip of the three-blade propeller, whereas the thickness is about the same; see Fig. 15. This result can be compared with the generally held view that both the thickness and the geometry of the tip are important for the near-field noise; see, for example, Metzger.[11]

Comparison of Pressure Levels on Wing and Fuselage

As stated in the Introduction, the fundamental question that prompted the present investigation is: Will the effect of the slipstream on the wing panels give a substantial contribution to the cabin noise? The present study gives no definitive answer, but at least allows a direct comparison of the maximum pressure level on the wing panel with the maximum level on the fuselage (see Table 1).

Table 1 shows that the level of the fundamental tone is approximately the same on the wing panel as on the fuselage, while the level of the harmonics is substantially higher on the wing than on the fuselage; from the third harmonic and upwards, the difference is more than 10 dB.

Concluding Remarks

The model measurements show that the level of the fundamental tone is about the same on the wing as on the fuselage and that the levels of the harmonics are higher on the wing than on the fuselage. Of course, these results alone are insufficient to establish the importance of the noise due to the propeller slipstream; the transfer functions describing the transmission from the measurement points to the interior of the aircraft are also needed. A wealth of material can be found in the literature for the transfer function between fuselage wall and cabin interior, but very little is known about the transfer function between wing panel and cabin. It is therefore difficult to establish the importance of the noise due to the slipstream at the present stage. However, it is hoped that the information on the excitation reported in this paper will promote measurements in the near future.

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