Figure 2.2: Propeller disk and co-ordinates system

For observer positions near the propeller plane, the above integrals are solved numerically. A Fortran program (Appendix A) have been written to perform the required integrations. A program "Prop Noise" was used to estimate the pressure-time history at various locations near the propeller disk. Results of theses two programs are incorporated in the discussion of results.

2.2 The phase velocity

Consider a plane harmonic wave propagating in the direction of increasing X in a medium with sound speed c . The pressure perturbation has the form $p = Ie^{i\omega(t - \frac{X}{c})}$ and assumed to be directed at surface at an angle α . If the co-ordinate system is chosen so that the surface lies in the plane $x=0$, then X , the co-ordinate in the direction of propagation is $X = x\cos\alpha + y\sin\alpha$. By replacing X in the above equation it becomes

$$p = Ie^{i\omega(t - x\cos\frac{\alpha}{c} - y\sin\frac{\alpha}{c})} \quad (2.4)$$

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Measurements of Unsteady Propeller Noise
Induced On A Model Fuselage

by

Hien Duc Duong

A thesis submitted to
the Faculty of Graduate Studies and Research
in partial fulfillment of
the requirements for the degree of
Master of Engineering

Department of Mechanical
and Aerospace Engineering
Carleton University
Ottawa, Ontario
Canada

September 1989

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Abstract

Airline deregulation and the high cost of fuel have stimulated renewed interest in propeller driven aircraft as a replacement for some existing turbofan airliners. Since passengers on current commercial aircraft are accustomed to relatively low interior noise, there is a continuing interest in the control of interior noise levels.

This study describes the apparatus and presents data on the unsteady pressures and trace velocities on a surface of a simulated circular fuselage due to the near-field of a two bladed propeller. The test were conducted in an acoustically treated windtunnel at advance ratios spanning the range 0.0 to 0.8. Pressure amplitudes and axial phase velocities were measured by means of microphones in the fuselage wall.

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Nomenclature

a_p	phase velocity
c	velocity of sound
D	propeller diameter
J	advanced ratio
$k=\omega/c$	wave number
M	Mach number
m	order of harmonic
n	propeller rotation speed , Hz
p	pressure
p_{inc}	incident pressure
p_Q	pressure due to torque
p_s	scattered pressure
p_T	pressure due to thrust
ΔQ	torque
R	propeller radius
r	radius to a blade element
r, θ	polar co-ordinates in yz-plane
θ	is also circumferential angles of fuselage
ΔT	thrust
t	time
V	forward velocity
x,y	Cartesian coordinates
u	surface velocity
Z	surface impedance
Ω	angular velocity
ω	frequency of m harmonic, $mB\Omega$
ϕ	phase readings of microphones
λ	wavelength

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Chapter 1

Introduction

In the last several years considerable interest has risen again for the use of advanced turboprop (prop-fan) aircraft as potential primary long distance transportation. This is due to the recognized fuel efficiency of the propellers and to the recent deregulation activity which has stimulated substantial expansion of commuter airlines.

However, before the potential fuel savings can be realized for large commercial prop-fan aircraft, several important technological problems must be solved. One of the serious problems is aircraft cabin noise. This subject has received attention because of its direct impact on passenger comfort, and therefore the acceptance of large propeller aircraft. The cabin or interior noise of propeller driven aircraft is significantly more annoying than the noise environment inside a fan-jet or a jet powered aircraft. This is because of the so-called tonality of the propeller noise; harmonic signals are perceived to be more annoying than broad band signals of equal sound pressure level. A study by NASA has indicated that levels of less than 82 dBA in cabin are considered comfortable by travelling public. While the cabin noise for the short range commuter turboprop aircraft have reached acceptable levels, the problem remains for large 150-200 passenger turboprop aircraft. Furthermore, some aircraft manufactures are trying to reduce interior noise levels

to about 70 dBA. In order to achieve such reductions, new approaches to interior noise treatment may be necessary.

An effective control of noise for the propeller driven aircraft is a major obstacle, mainly because of the high energy levels of low frequency discrete tones generated by propellers. It is seen that a satisfactory cost effective solution leading to an acceptable degree of passenger comfort is one of the major factors facing the design, certification and sale of large propeller driven aircraft.

1.1 Review of some acoustical treatments of cabin noise

Several solutions for the control of noise inside a propeller aircraft are: treatment of fuselage wall (by adding more sound insulation panels), moving the engines away from the fuselage (to increase tip clearance), moving the engines backward (to increase distance between passengers and propellers), synchrophasing of propellers, shielding with opposite propeller's rotation, and lastly a method called active noise control.

The first solution may only be effective over a certain range of frequencies. The installation of more sound proofing panels in the cabin may actually increase rather than decrease noise [1]. This is because the added mass of panels and other conventional acoustic treatments are only effective at frequencies higher than dominant tones of propellers. In addition, the sound proofing panels add more weight to the aircraft hence increase its fuel consumption.

Moving the engines outward or backward may encounter design problems, due to aircraft stability or dynamics. At the present time a tail-mounted pusher configuration, the General Electric Unducted Fan (UDF) engine, has been flown on both a Boeing 727 test bed and a McDonnell Douglas DC-9 derivative. This type of installation is limited to moderate size aircraft.

The last three options appear to be free from some of the above problems. Synchrophasing sets the rotational phase of each propeller (i.e. relative angular

Figure 1.1: Typical effect of synchrophasing on cabin noise (ref.1).

Figure 1.2: Opposite rotation cabin noise reduction concept (ref.1).

position) until the interior (or exterior) noise levels are at minimum. The idea is to control the rotation of the propellers so that at a given field point the pressure produced by one propeller is at the 180 deg. out of phase to that produced by the other propeller. If both pressures are of comparable magnitude the noise reduction can be sizeable. Analysis and some experiments have been performed by Fuller, Jones & Magliozzi [2],[3],[4]. As fig.1.1 shows the synchrophasing method does not in general, reduce absolute sound levels. By assuring that all propellers are at the same rotational speed and relative angular position, the characteristic beating signal, which is caused by two tones of slightly different frequencies (i.e mismatched engine speed) is diminished. The steady sound is less annoying than the periodic pulsations which typically have period of 5 sec.

The second concept which has not been exploited widely for noise reduction is the use of opposite rotation. The propellers on one side of aircraft rotate clockwise while the propellers on the other side rotate counter clockwise (and vice versa). As figure 1.2 indicates, the approaching blades create the noise heard by listener in the cabin. By installing propellers with direction of rotation such that the

blades move up as they approach the fuselage in a low wing installation the noise generated must pass through the area below before reaching the cabin. Thus, the advantage of double wall construction for reduced noise may be achieved with no additional mass penalty.

Finally there is noise control. The basic principle is to reduce the noise generated from a primary source by using a secondary source. If the secondary source signal can be made identical in amplitude but opposite in phase to the primary signal, then complete cancellation can be achieved near the control source. Commercial systems are now available to attenuate noise inside airducts. Application in noise reduction for propellers has been examined in the laboratory [5].

1.2 The scope of the thesis

The last three methods for noise control for propeller-driven aircraft are expected to be fully investigated in the future. However, a few problems remain to be solved: the lack of physical understanding of the suppression mechanism, the need to predetermine the phase relationship as in the case of synchrophasing, the hardware problems for the active control method. In the design of a fuselage wall, the noise reduction requirement must first be established. This depends on the exterior noise field, which must be defined in terms of amplitude (due to discrete nature of propeller noise) and a phase distribution. The proper acoustic design of the interior of a propeller driven aircraft requires not only an understanding of the propeller noise but also its propagation path characteristics, since both produce structural responses resulting in vibration and noise in the cabin.

To date only pressure amplitudes are used in the design in the specification of acoustic treatment. It is well known, that the response of a panel to an incident sound wave is a function of angle of incidence. In fact, when the so-called trace velocity of the disturbance matches the flexural wave speed of the panel, no sound attenuation is observed. The trace velocity is the velocity at which a wave front sweeps over the surface. For all cases but grazing incidence, this velocity is supersonic. Flexural wave speeds in metal sheets tend to be higher than the speed of

sound in air. It is seen that the information about the trace velocity is important. As the flexural wave speeds in the panels are function of panel geometry and material, a designer may use the information about the trace velocities to design a fuselage skin whose flexural wave speed is always mismatched to that of the local trace velocity.

Clearly such a fuselage is likely to be inhomogeneous. Still, there may be a weight savings, if the structure is optimised in some way. These aspects are beyond the scope of this thesis.

The object of this thesis is to provide a more complete noise description of the propeller noise on a fuselage under simulated flight conditions. Detailed "maps" of rms pressure on a fuselage wall at different advance ratios are obtained. Calculations of point-to-point trace velocity were also made in the hope of providing starting data for advanced fuselage wall soundproofing methods. Finally, a propeller driven-aircraft usually involves two or more propellers a calculation for the synthesis of pressure due to 2 propellers were also included. The data obtained can serve as input to analytical models for the synchrophasing methods [2], the active control method [5] or for the study of simulating model of noise impact on fuselage [6].

Chapter 2

Propeller noise

2.1 Steady air loads

A propeller is a device which produces a controlled amount of force (thrust) in fluid medium. The principal mechanism is lift produced by a moving airfoil. Consider a section of the propeller as illustrated in figure 2.1. The blade element moves forward with a speed V and rotates about the propeller axis with a velocity of ωr . As is the case for a conventional wing, there is an induced velocity $\bar{w}$. The blade element is set at an angle θ with respect to the propeller disk, and the propeller experiences an effective angle of attack α . The resultant lift and drag components can be resolved into a thrust force (ΔT) (axial) and a radial force which induce a torque ΔQ on the propeller shaft. The calculations of ΔT and ΔQ as well as the total thrust and torque is normally performed using methods due to Goldstein [7], or Theodorsen [8].

For most part these theories use incompressible aerodynamics, with "compressible correction". Thus they cannot predict sound radiation. It is well known that surfaces in motion generate sound. A simple example is the case of a loud speaker. For a propeller, the blade moves and imparts a force on the medium. To an observer the force will appear to be in motion, thus it is not steady in a general sense ($F = F(\bar{x}(t))$). Lamb [9] provides a relationship for the unsteady pressure at $\bar{x}$ due

Figure 2.2: Propeller disk and co-ordinates system

For observer positions near the propeller plane, the above integrals are solved numerically. A Fortran program (Appendix A) have been written to perform the required integrations. A program "Prop Noise" was used to estimate the pressure-time history at various locations near the propeller disk. Results of theses two programs are incorporated in the discussion of results.

2.2 The phase velocity

Consider a plane harmonic wave propagating in the direction of increasing X in a medium with sound speed c . The pressure perturbation has the form $p = Ie^{i\omega(t - \frac{X}{c})}$ and assumed to be directed at surface at an angle α . If the co-ordinate system is chosen so that the surface lies in the plane $x=0$, then X , the co-ordinate in the direction of propagation is $X = x\cos\alpha + y\sin\alpha$. By replacing X in the above equation it becomes

$$p = Ie^{i\omega(t - x\cos\frac{\alpha}{c} - y\sin\frac{\alpha}{c})} \quad (2.4)$$

Figure 2.3: An obliquely incident wave and co-ordinate system

The pressure on the surface at $x=0$ is

$$p = Ie^{i\omega(t-y\sin\alpha/c)} \quad (2.5)$$

Hence there is a wave propagating along the surface in the y -direction, at a velocity $\sin\alpha/c$ and is called the phase velocity.

For simplicity a simple form of flexible surface is given as

$$p = Zu = Ie^{i\omega(t-y\sin\alpha/c)} \quad [12] \quad (2.6)$$

where u is the normal surface velocity measured in the direction leading into the surface from the half space containing the incidence wave and Z is called the surface impedance and is determined by the mechanical properties of the surface (mass, rigidity, damping, natural frequency...). Therefore, it can be seen from the above equation that the pressure is a function of the normal velocity, which both in turn are related to the amount of sound being reflected or transmitted through the relation

$$R_f = \frac{Z - \rho_0 c / \cos\alpha}{Z + \rho_0 c / \cos\alpha} \quad (2.7)$$

where

R_f = Reflected wave pressure

T_t = Transmitted wave pressure

Figure 2.4: Defined co-ordinates for phase calculations

In order to evaluate the effectiveness of a soundproofing method, the amount of sound being transmitted through the surface is required. Since the sound field induces vibration in the surface, it can be argued that the maximum vibration (and maximum transmission) occurs when the solid structure is excited at a resonance frequency.

This vibrational behaviour of structures is often expressed in terms of “modes”. A modal pattern of vibration may be thought of as a standing wave pattern which is the result of a set of properly matched flexural waves.

If the incident sound field matches these flexural waves in term of wave length and phase velocity, the sound energy can pass through the structure without significant loss of energy.

If region of potential matching are known during the design phase, appropriate measures should be taken to render the modified structure un-responsive to the sound-field.

2.2.1 The calculation of the phase velocities

A disturbance with frequency ω is measured at $\bar{x}_j$ and $\bar{x}_i$. The measured relative phase is $(\phi_i - \phi_j)$. The signal must travel along the ray of length s . At point $\bar{x}_j$ the signal is proportional to $e^{i\omega t}$ while at $\bar{x}_i$ $e^{i(\omega t + (\phi_i - \phi_j))}$ applies. The time interval $\Delta\tau$ required to obtained a phase of $(\phi_i - \phi_j)$ is given by $|\phi_i - \phi_j| = \omega\Delta\tau$ where the absolute is used to simplify the interpretation. The defined co-ordinates are shown in Figure2.4

Now the wave travels a distance s in the time $\Delta\tau$. It follows that $s = a_p \Delta\tau$ where a_p is the phase velocity

$$a_p = \frac{s}{\Delta\tau} = \frac{s\omega}{\phi_i - \phi_j} \quad (2.8)$$

where $(\phi_i - \phi_j)$ is measured in radians. If $(\phi_i - \phi_j)$ is given in degrees then $(\phi_i - \phi_j)$ is multiplied by $\frac{\pi}{180}$. For a_p based on degrees phase difference

$$a_p = \frac{s\omega}{(\phi_i - \phi_j) \pi} \frac{180}{\pi} \quad (2.9)$$

or

$$a_p = sf \left(\frac{360}{\phi_i - \phi_j} \right) \quad (2.10)$$

if $\omega = 2\pi f$ and f in Hertz, s in metres and a_p in m/sec.

The acoustical wavelength then can also be calculated from the relation $\lambda f = c_p$, where f is the frequency. The matching of a propeller λ with a modal wavelength can enhanced the modal response and lead to a more efficient coupling between the sound field and the fuselage structural response.

Chapter 3

Experiment

3.1 Overview

In order to provide information about the nature of pressure field induced on a fuselage, a scale model experiment has been performed. To keep the data base in a manageable size, a single propeller is used. It will be shown in sections below that the sound field due to additional propellers can be accounted for by some simple transformations of the data.

The novel feature of this work is an attempt to measure the trace velocities of the pressure field on a fuselage. To this end the pressure signal must be measured with respect to a known absolute phase or time reference. Here the propeller shaft position provides a convenient reference time. This then allows one to construct an image of the instantaneous pressure field on the fuselage.

The experiments were conducted in the closed wind-tunnel at Carleton University. The test section has a length of 72 inches and a cross-section of 20in. by 30in. All surfaces in the test section have been covered with a 1in. thick acoustic foam. This rather crude wall treatment provides sufficient reduction in the reverberant levels inside the test section. It is not possible to simulate free-field conditions. This does not present a serious limitation; as the amplitudes on the direct waves are likely to be much larger than those of any waves reflected from the wind tunnel walls. Test were conducted at several simulated flight speeds.

The principal test object, the fuselage, was constructed from two cylindrical

plexiglass tubes. The nominal dimensions of the tubes and the test propellers are such as to provide a 1:10 scale of a modern commuter aircraft such as the deHavilland Dash 8 aircraft.

One section of tube, having a diameter of 5in. consists of a faired nose, and has provision for accepting up to 10 microphones which can be flush-mounted. The microphones are arranged along the axial direction and have a nominal separation of 1in. This section fits snugly into a second plexiglass tube. The latter is anchored securely to a sub-floor which fits inside the wind-tunnel. This arrangement allows the test section to slide as well as rotate inside the support tube, thereby permitting a survey of the entire sound field.

Eight electret microphones were installed. Although the Tandy type 270-092B condenser microphones are not laboratory grade, they have been used successfully at this laboratory. Furthermore, careful calibration of both amplitude and phase permit one to compensate for any deviation from the desired 'flat' frequency response.

The microphones require a 10V DC bias voltage, and their outputs are coupled to signal amplifiers via 10 mF capacitors. The signal leads of the microphones are gathered by a 25 pin connector. A suitable shielded cable then transmits the signal to a microphone preamplifier system. Each microphone is connected to a buffer amplifier with a nominal signal gain of 67. A rotary switch selects two microphones at a time. The switch was wired in such a manner as to select only immediate neighbours. The selected outputs are buffered by unity gain amplifiers and available for further signal processing.

3.2 Test set up

The typical propellers installed on commercial aircraft feature three or more blades. Since the noise is governed by the blade passing frequency, it is not necessary to match the precise number of blades. Here a wooden 2 blade propeller was used. The wooden scale model propellers are stiffer than the plastic ones. This feature avoids possible resonance effects, where the propeller blades vibrate at one of the

blade passing frequencies, which would result in poor simulation.

The propeller was driven by an Astro-10 electric motor. The recommended 2 blade 7-4 propeller was used. The motor was enclosed in a fairing. Figure 3.1 & figure 3.2 show the model set up. Without overloading, the motor can drive propeller at rotational speeds up to 160 rps. However, to be able to stay away from signal contamination with line frequency harmonics, and yet at the same time maximize the motor speed without overload, a propeller speed of 145 rps was chosen. A voltmeter was used to monitor the input power to the motor to assure operation at constant power. An internal motor cooling system was also added inside the fairing to avoid overheating during extended runs. The motor speed was controlled by a two power transformers. The first is an autotransformer which is used to adjust power levels. The second is a step down transformer reducing the input AC voltage by a factor of 10.9. This provides the nominal 11 volts needed by the motor. A full bridge power rectifier convert the AC voltage into a nominal DC voltage (appendix B). This simple system is more effective than a feed back control system used in previous studies [13].

The propeller speed is monitored using a sensor unit (appendix C). The unit contains a light emitting diode and a phototransistor. Light emitted from the diode is reflected from an aluminum foil glued to the motor shaft, and is picked up by the transistor. The remainder of the shaft is covered with black tape to suppress spurious reflections. The speed of the propeller was checked by reading the signal of any microphone at the 3rd. harmonic of the blade passing frequency. The reading from frequency counter connected to the output of the sensor unit was used as a double check. The output of the sensor unit serves as a reference time base in this experiment which will be discussed later.

3.2.1 Acoustic instruments

Figure 3.3 shows the measurement and the signal processing set up. The unsteady pressures are measured by eight 0.375 in. microphones. As discussed above the microphones were embedded in the fuselage wall at 1.0 in. apart, with microphone 2 at the propeller plane. The signals from all the microphones were routed via the pre amplifier channel selector (appendix D), to a HP3582a Dual Channel Spectrum Analyzer.

The sensor unit (appendix C) is used to trigger the spectrum analyzer. In the experiments the spectrum is triggered as the propeller blade passes by microphone 2 at $\theta = 0$ deg. The co-ordinate system used in data collecting is defined in figure 3.4 with Mic.2 is in the propeller plane and microphones are placed 1 in. apart. The propeller is rotating counter-clockwise as seen from the front.

Each run represents an average of 16 events. As each individual record is triggered by a reference, the resulting average is equivalent to a signal or "phase" averaging. This type of averaging is known to reduce contribution from random noise, while "enhancing" or retaining information which is periodic with respect to trigger signal. Also, as was pointed out, all data is now referenced to a common time datum. To maximize the resolution of the spectrum analyzer only the first three harmonics were taken. The analyzer provides a 256 point estimate of the spectrum; signals were analyzed over the nominal frequency band of 80 to 1080 Hz, with an effective resolution of 12 Hz.

Figure 3.1: Test set up plane view

Figure 3.2: Test set up front view

3.2.2 Measurements

Experiments were conducted at a constant propeller speed of $N=8,700$ rpm (145 rps) which results in a tip Mach number of $M_T = 0.24$. The free stream velocities were chosen to be 0.0 m/s, 10.3m/s, 20.7m/s, and the resultant advance ratios were 0.0, 0.4 and 0.8. At each advance ratio the microphones were rotated from $\theta = 0$ deg. to 360 deg. to obtain a complete set of noise measurements of unsteady pressure on the fuselage at several longitudinal (or axial) and the circumferential (or radial) positions. The measurements were repeated one more time for each case to check for repeatability. The same procedure was followed for a reverse thrust case.

The entire measurements were performed under the following assumptions :

- (i) the amount of sound reflected from the absorbing material is small and can be neglected.
- (ii) The sound reflected from the motor casing was small enough that it did not contaminate the sound field.
- (iii) The sound generated by the motor was mostly random noise and did not contribute to any discrete tonal noise produced by the propeller.

3.2.3 Typical time trace pressure waveform

Figure 3.5 shows a typical time trace of pressure recorded at microphone 7. As the propeller passes by the fuselage at 0 deg. angle, the data acquisition is triggered. The sharp peak of the waveform is seen as the wave front from one propeller blade arrives at the microphone. The peak decays before it rises again as the wave front from the other blade arrives. The cycle is completed when the first blade returns. The time trace pressure waveform is a combination of signal at different harmonics. Each harmonic is characterized by an amplitude and a phase angle. The sum over all harmonics is given by

$$p(t) = \sum_{n=1}^{\infty} a_n \cos(\omega_n t + \phi_n)$$

As seen from figure 3.6, here the DC contribution has been removed since the measured signal has zero average. Although the time history of the pressure illustrates

Figure 3.3: Instrument set up

θ = Circumferential angles
of fuselage

Front view of fuselage

← Longitudinal (axial) direction →

Side view of fuselage

Figure 3.4: Defined co-ordinates system used for data collection.

Figure 3.5: Typical time trace of pressure. (Mic.7 at $\theta = 0$ deg, reverse thrust).

$t=0$ (arbitrary)

Figure 3.6: Fourier's components of time trace signal.

the evolution of unsteady pressure, it is customary to analyze each harmonic on an individual basis. This implies that the underlying processes are linear. The harmonic analyses is performed (in an approximate manner) by the spectrum analyzer.

3.2.4 Typical amplitude spectrum

Figure 3.7 shows a typical amplitude spectrum of microphone 7 for the first three harmonics. Although the rotational speed is 145 Hz, there are 2 propeller blades, hence the frequency spacing between harmonics is 290 Hz. To minimize the bandwidth of the spectrum analyzer only the first three harmonic were recorded. It is in this frequency range that the innovative sound control methods are likely to be used. Also the energy level of the higher harmonics is quite small and they can be absorbed by conventional treatments.

As pointed out above the waveform is sampled only when the propeller shaft has a certain known angular position. It is assumed that the acoustic signal measured contains both random noise and periodic contributions.

$$p(t) = S(t) + \sum_{j=1}^{\infty} a_j \cos(\omega_0 j t + \phi_j)$$

Consider now M distinct events

$$p_m(t) = S(t + m\tau) + \sum_{j=1}^{\infty} a_j \cos(\omega_0 j(t + \tau M) + \phi_j), m = 1, M$$

where τ is the sampling interval. A simple shift in time converts the sequential events into M simultaneous events:

$$\hat{p}_m(t) = p_m(t - m\tau)$$

Now the sampling interval τ is an integral multiple of the time required to complete one revolution ($\frac{1}{n}$). Noting that $\omega_0 = 2\pi nB$ one finds

$$\hat{p}_m(t) = S(t - m\tau) + \sum_{j=1}^{\infty} a_j \cos(\omega_0 j t + \phi_j)$$

on the range $t=0$ to τ . If all M records are added one finds

$$\sum \hat{p}_m(t) = M \sum_{j=1}^{\infty} a_j \cos(\omega_0 j t + \phi_j) + \sum_{m=1}^M S(t - m\tau)$$

division by M provides an estimate of the average waveform. If the sampling interval τ is sufficient long, the one may assume that the m noise signals $S(t)$ are un-correlated with each other, their effective sum tending to diminish. It can be shown [14] that the residual value of the summation will take on an average value of $\sqrt{\frac{S^2}{N}}$.

Since the propeller harmonics are quite dominant, there is no need to average over a large number of events. It was observed that the spectral display did not change appreciably after 8 events. Thus 16 events was chosen as a reasonable number of averages.

3.2.5 Typical phase angle spectrum

Figure 3.8 shows a typical phase angle as a function of frequency for microphone 7. The spectrum analyzer displays phase measurements from $+200$ deg. to -200 deg. with the accuracy of ± 10 deg. Phase angle outside this window are wrapped around and are displayed in terms of $\phi = \phi_{display} \pm 360K$ (K is an integer number). Under favourable circumstances the value of K can be deduced; it is not possible

(or necessary) with harmonic data. Only readings at harmonics are recorded, other values are difficult to interpret since they are due to random noise.

The readings of both amplitude and phase are taken directly from the spectrum analyzer. The alphanumeric CRT annotation & marker readout that are available on the spectrum allow for such task. The amplitude and phase spectra can also be plotted using a 7044 HP plotter.

Fig.3.7.a : Typical amplitude spectrum. Mic. 7, reverse thrust.

Figure 3.8: Typical phase spectrum. Mic.7 . Reverse thrust.

Chapter 4

Results

4..6 Unsteady pressures induced on a fuselage

Fig.4.1 shows the measured pressure waveforms at different microphone positions for circumferential angle $\theta = 0$ deg. and advance ratio $J=0$. The waveforms are low pass filtered by the spectrum analyzer. Thus are "cleaner" than those displayed by an oscilloscope. The pressure waveforms indicate that the two propeller blades are not perfectly matched as each registers a different signature.

The pressure waveforms are observed to be functions of location along the fuselage. Although all waveforms were taken at the same propeller position, it is apparent that there are phase shifts as the waveforms crossed the axis at different locations. Proceeding from mic.1 to mic.8, one observes a change in signal character. Starting with mic.1's signal, it is characterized by positive peaks, with the positive portion exceeding the negative one. The signal from mic.2 shows a more symmetrical positive and negative energy distribution with spikes appear in both positive and negative portions. For mic.3's signal, except for a small change in amplitude, it appears to be that of mic.1 with a complete phase shift. The last four signals, from mic.5 to mic.8, display the same saw-tooth type waveform with almost no change in signal's characters.

The differences in signal characteristics can be related to the idealized directional sound patterns around a propeller. Without considering the interference of the fuselage the pressure is given by Garrick and Watkins [10] as in equations

2.2 & 2.3 of chapter 2. Using these two equations, the pressure-time histories at different microphones for $\theta = 0$ deg. are plotted in fig.4.1.b. The figures are plotted using only the first three harmonics. As seen, there is a phase change as the wave arrives at microphones at different time. The pressure waveforms at mic.1 and mic.2 again display positive peaks while for that of mic.3 & mic.4, the positive and negative portions are more equally distributed. Note there is almost a complete phase change between waveforms of mic.1 and mic.3 (although their amplitudes are not nearly equal as seen in fig.4.1). As the distance is further increased (mic.6 & mic.7), the pressure waveform decays into almost a straight line, with little change of pressure over time. The effect of increase in distance allows the pressure waveform to remain the same as seen from fig.4.1 from microphones 5 to 8. The changes in pressure waveforms going from mic.1 to mic.8 are under the influences of two changes (i) change in distance and (ii) the directionality. For far field radiation, the free field pressures are given by Gutin's equation [11] as follows

$$p = \frac{\omega m B}{2\sqrt{2}\pi c r} \left(-T \cos \nu + \frac{cQ}{\omega R^2} \right) J_{mB}(kR \sin \nu) \quad (4.1)$$

where

B = no. of blades; m = harmonic no.

r = distance from propeller hub to field point

T = thrust; Q = torque; ω = angular frequency

$k = \frac{\omega}{c}$; c = speed of sound; ν = angle made with respect to propeller axis of rotation

For the case of acoustic far-field the directionality of the torque component is determined by the function $J_2(kR \sin \nu)$ and that has the same phase all around. The thrust component is governed by $-\cos \nu J_2(kR \sin \nu)$ and phase therefore reverse for angles between $\pm \pi/2$. The directionality patterns of the thrust and

torque are shown in sketch below. The phase change between $\pm\pi/2$ around the

propeller plane of the directional sound pattern may account for the complete change in phase between signals of mic1. and mic3. It can also be said that signal at mic2 is dominated by torque component, while that of mic1 and mic3 are contributed by both torque and thrust components.

For the case of the near field, one must integrate the complete set of equations. Since both spectra and pressure-time histories were measured several waveforms have been synthesized. Each waveform contains but 3 harmonics, these qualitative images are compared with measured pressure-time history.

Also, since the microphones/hub distance varies from about $.64 D$ to $1 D$, where D stands for the propeller diameter, the effects of directivity and near field/far field transition are intermingle. The saw-tooth type signals persist for the last four microphones; this indicates that the far field conditions have been reached. Also, from sketch above as the axial distance increases, the contribution of the torque component decreases while that of the thrust component increases. The result magnitude is a compromise, and hence as seen from fig.4.1 the magnitude of the signal remains almost the same from mic.5 to mic.8.

At higher wind tunnel speeds the random background noise of the wind tunnel becomes quite large and is superimposed onto the waveforms (Fig.4.2 for

mic.4 and mic.6 at $\theta = 60$ deg. and $J=4$). The magnitudes of the waveforms decrease (few dB for the first harmonics) as the speed of the wind tunnel increases.

Fig.4.3 shows the pressure waveforms for the reverse thrust case at $\theta = 0$. deg. Note that in this reverse thrust case the propeller is rotated in opposite direction and this is achieved by switching the power leads at the motor power supply terminals. Hence, the leading edges of the propeller now in effect, is the trailing edges. The flows around the propeller blades in forward and reverse thrust cases can be sketched below. As seen from sketch, it is possible that in the reverse

thrust case the propeller is stalled and separation on the upper section may occur. If any part of the propeller blade is stalled the local drag fluctuation is set up by the separated flow, and this may well effect the signal's characteristics. However, because of the low Reynold no. (approx. 91,200 based on max.blade chord and tip speed) it is difficult to predict the behaviour of the propeller. From fig.4.3, signal at mic.2 is still probably dominated by the torque term and displaying positive and negative spikes, while signals at mic.1 and mic.3 are not exactly the inverse of each other. It is also noted that the saw tooth type signals again appear from mic.5 to mic.8, indicating the possibility of the same far field waveform characteristics for

both forward and reverse thrust case.

4..7 Power spectrum

A typical power spectrum has been shown on section 3.6 last chapter. However, it is observed that the power spectrum would display different characteristics if it is taken at the time when the propeller has not reached a constant RPM. Fig. 4.5 shows some typical characteristics of power spectrum recorded under such conditions (for mic.2, .14D distance from blade tip in the propeller plane). The tonal components in the propeller plane are up to 30dB higher than the broad-band noise floor which is in part due to the background noise of wind tunnel. At higher harmonics the power spectrum levels decrease. While it is known that the harmonics spectra of the propeller in addition in showing the blade harmonic frequency also exhibits subharmonics (between blade harmonic frequencies). This is tied to unavoidable manufacturing differences of blades. However, if the amplitudes of such harmonics are small compared to the blade harmonic frequency (which can be seen from fig.4.5 and is true for all of the tests) they will be ignored.

Fig.4.6 and 4.7 show the spectra obtained at mic.2 (in the propeller plane) and mic.8 (6in. behind the propeller plane) at $\theta = 0$ deg., $J=.8$ the figures exhibit the differences in tonal peaks; also the higher harmonics of blade passing frequency are also masked by background noise. This indicates that the rotational noise levels at the side of the fuselage decrease with the distance from the propeller plane.

4..8 Dependence of pressure levels on the distance from the propeller plane

Figures 4.8, 4.9, 4.10, 4.11 illustrate the variation of the first harmonic of the blade passing rate with the axial distance at the same radial position $d=.14D$ from the propeller tip.

For the static case, fig.4.8, the maximum pressure levels do not occur in the propeller plane (at low θ angle) but slightly ahead of propeller plane. At high θ angles the maximum seems to occur in the propeller plane and the level decays as

the axial distance increases.

It can be observed from figures 4.8, 4.9, 4.10 that in general there is a reduction of noise as the advance ratio is increased. Since the pressure is a function of thrust, the reduction of pressure is due to the decrease in thrust as the forward velocity is increased. For constant power operation the thrust decreases with increasing forward speed, thus the noise should decrease.

Also, the pressure inversely scales with distance from the source. The variation of pressure as a function of (1 / distance) is shown in figures 4.10a and 4.10b. Here the pressure level at mic.2 is used as reference (0 dB), and the notation is defined as in the sketch below

In fig.4.10.a, the dash line is the same as that in fig.4.8 for $\theta = 0$ deg., except now the level at mic.2 is referenced as 0 dB., and the other microphone levels either fall below or above the 0 dB line. The other two curves are defined on plots.

As seen from fig.4.10a, the curves all seem to fall close to each other from mic.1 to mic.4, and then depart greatly from one another. This suggests that the region spanned by mic.1 to mic.4, (where the angle ν is small), the effect of a change in distance ($1/r$ term) is bigger than the change in directivity and hence dominates. As $\hat{r}$ increases, the change in ν becomes dominant, and the effects of both distance and directivity (plus other interference) play equally important roles. Fig.4.10.b shows the same relationship between axial distance and pressure levels for the case

of $J = 0.8$. Here the plotted lines follow more closely to each other from mic.1's position to mic.8's position. Recalling the results of Garrick and Watkins, it is noted that the forward speed affects the torque term by replacing $S = \sqrt{x^2 + y^2}$ (for $M=0$) by a smaller distance $S = \sqrt{x^2 + \beta^2 y^2}$. The thrust term is strongly affected by this substitution. The resulting effect of forward velocity is difficult to see but it may have modified the acoustic path in such a way the pressure level falls off at the same rate as the log of the increase in distance.

Fig.4.11 illustrates the pressure variation in the axial direction for the reverse thrust case. The maximum levels occur in the propeller plane at low θ angles. The sound pressure levels at $\theta = 90$ deg. and $\theta = 270$ deg. are different (about 2 dB), which indicate a higher pressure produced on the approaching side of the propeller blade.

The data is also compared with the predictions based on model of Garrick and Watkins. Figure 4.11.a shows the predicted pressure variation in the axial direction obtained by using Garrick and Watkins model. Figures 4.11.b and 4.11.c show the predicted pressure values plotted in dB., using pressure at mic.2 as the reference. The measured data are also plotted on the same figures. Since there is no diffraction correction, the agreement is not expected to be excellent. Note that the ratio of the fuselage diameter and a characteristic wavelength is small ($D/\lambda \approx 0.11$). Thus the obstacle is but a small fraction of the incident wavelength. As a first approximation the ratio of the incident and scattered pressures is

$$\frac{p_s}{p_{inc}} \approx 0.38\sqrt{n} \quad [15]$$

where n is the harmonic index. Since this is based on a far field assumption the prediction is matched to the measured values at microphone position 6. Here the non-propagating nearfield is small compared to the acoustic pressure. The results are shown in figures 4.11.b and 4.11.c. It is seen that the non-propagating nearfield does induce additional pressures, while the levels further aft of the propeller plane are reasonably well modelled.

4..9 Pressure variation in the circumferential direction

Figure 4.12 shows polar plots for typical variation of ratio of pressure for mic.7 at $J=0$ in the circumferential direction, with the reading at $\theta = 0$ deg. used as reference level. Figure 4.12 shows high ratio readings in the fuselage side with propeller and a drastic decrease on the far side of the fuselage. The same characteristics are also observed for other advance ratios (Fig.4.12.a). Besides the effect of distance from source the decrease in pressure may also be due to the acoustic shadowing of the fuselage (i.e the far side of the fuselage does not receive direct noise from the propeller) and the directional nature of the propeller noise source.

4..10 1st. Harmonic contour plots

Figures 4.13, 4.14, 4.15, 4.16 show the measured pressure contours on the "un-wrapped" fuselage surface. The contour plots are obtained directly from experimental data in dB. The reference dB. level is set by the spectrum analyzer. The graphics program applies linear interpolation between points and then uses ~~spline~~ fitted curves for joining between points. In figures 4.19, 4.20 the contours seem to be symmetrical about the axial direction for areas near the propeller plane. However they appear not to be symmetrical about the fuselage center line for the rest of the surface. Also note that the contours are more closely spaced below the fuselage center line. It is probably due to the acoustic wave incidence

on the fuselage (as in sketch below).

The four figures also show that there is a region of high pressure region near the propeller plane as more contour lines are packed. Lines of constant pressure then begin to run axially as they move away from the propeller plane. This indicates a lesser change in pressure levels as the distance from the propeller plane increases.

In fig.4.15 at high advance ratio, because of the high background noise which are masked onto the microphone signals at high circumferential angles the contour plot exhibits local circular contours at high circumferential angles.

4.11 The trace velocity

Figures 4.17,4.18,4.19,4.20 show the trace velocities between adjacent microphones at different circumferential angles under different advanced ratio. The trace velocities a_p are calculated using equation below

$$a_p = \frac{2\pi fl}{\Delta\phi} \quad (4.2)$$

where

f = frequency (Hz)

l = Distance between adjacent microphones (.0254m)

$\Delta\phi$ = phase difference between two adjacent microphones (deg.)

Because the phase data were taken as pairs between adjacent microphones, and because of the orientation of the microphones, that the calculation for the trace velocity is only possible for the component in the axial direction. The axial trace velocities are defined as negative if the phase difference ($\phi_i - \phi_j$) is negative, where ϕ_i = phase readings of microphones 2, 4, 6, 8 and ϕ_j = phase readings of microphones 1, 3, 5, 7. It can be seen from figure 4.17 that the axial trace velocity between mic. 1 and mic. 2 remains the same for all circumferential angles while the axial trace velocities for others pairs vary with the circumferential positions. At $J=4$ and $J=8$ (figures 4.18, 4.19) the trace velocities are all dependent on the circumferential angles, with the maximum velocities occur between angles θ of 40 and 80 deg., on the approaching side of the blades.

For the reverse thrust case, the axial trace velocities between microphones 1 and 2, 3 and 4 remain the same for all circumferential angles while the phase velocities between microphones 5 and 6, 7 and 8 vary with circumferential positions. Hence, the point to point trace velocities in the longitudinal direction vary with the radial and the axial positions.

The trace velocities are all subsonic, except at few points of microphones positions. The expected supersonic results are not obtained, and a possible explanation is due to the interference of waves. Since when the cylinder is excited, the circumferential waves of propeller propagate in both directions around the cylinder, combining to create an interference pattern or standing wave. With the presence of the fuselage the microphones will experience an outgoing wave from the propeller and a standing wave created by previous outgoing waves. The interaction between these two types of waves will result in some trapped or confined waves. This causes the waves to travel at low velocity on the fuselage wall. Hence, due to the fuselage circumferential standing waves and the presence of the contained wave field, the plane wave model for calculating the incidence wave angles may not be able to provide accurate results. Another possible hypothesis to explain ~~for~~

these subsonic velocities consists of a pressure field that rotates as a rigid body with the propeller at the propeller speed. As one moves close to the propeller, one would experience different pressures when the propeller blades are at different positions. However, as the propeller blades return to the same positions in which the pressures were taken, one would again experience the same pressure. Hence, there is a pressure field that is locked to the propeller, and the acoustic waves that originates from the pressure field propagate in the axial direction and could impinge on the microphones at almost the same time (as in sketch below) resulting in low axial trace velocity between adjacent microphones. Hence, the axial trace

velocity between adjacent microphones can be subsonic, but the radial component of the wave may be supersonic. During the experiments it was observed that there was a change of 1 to 2 degrees in wind tunnel air temperature. Since the trace velocity is a function of sound velocity, which in turn is a function of temperature, therefore a change in temperature can affect the resulting trace velocity. The relationship between trace velocity (or phase angle ϕ) and temperature can be seen through the equation

$$\Delta\phi = \frac{360}{C} f l; \phi = \text{deg.}, f = \text{hz}, l = \text{m.}, C = \text{m/s.}$$

Thus, the phase angle ϕ (and hence the trace velocity) are sensitive to change in temperature (C), frequency change (Δf), and position error (Δl). The relationship can be rewritten as follow

$$\Delta\phi = \frac{360(f + \Delta f)(l + \Delta l)}{K R_s (\sqrt{T + \Delta T})} \quad (4.3)$$

where Δf = change in frequency, Δl = position error, ΔT = temperature change, $R_c = 287 \text{ KJ/ kg } K^0$ = gas constant, and $K = 1.4$ ratio of specific heat of air. Assuming that there is no changes in frequency and position, the phase angle (or trace velocity) then only be a function of temperature. During the experiments, the changes in wind tunnel velocity temperature was observed to be 1 or 2 deg. c in magnitude. However, assuming that f (frequency) and l (position) remain the same, a change of $\Delta T = 2$ deg. from $T = 20^0 \text{ c}$ to $T = 22^0 \text{ c}$ would result in a change in $\Delta\phi$ (between two adjacent microphones) of

$$\Delta\phi = \frac{360fl}{\sqrt{KR_c}} \left(\frac{1}{\sqrt{20}} - \frac{1}{\sqrt{22}} \right) \quad (4.4)$$

With $f = 145 \text{ Hz.}$, $l = .0254\text{m}$, the change is less than 1 deg. Hence, the effect of temperature on the trace velocity was hardly observed; this is probably due the nature of the inverse square root relationship and the phase resolution (± 10 deg.) of the Spectrum Analyzer.

4.12 The synthesis of pressure on fuselage due to 2 propellers

Since an aircraft usually has 2 or more than propellers, the data obtained in the experiment may be extended for the case of two propellers rotating in the same or different direction.

The sketch below shows a typical propeller arrangement. Let first assume that

$\beta = 0$ deg., and let the pressure at the point a due to propeller 1 be $p_1(\theta_a)$, and due to propeller 2 be $p_2(\theta_a)$. Then the total pressure p_{Ta} at the point a due to the two propellers will be

$$p_{Ta} = p_1(\theta_a) + p_2(\theta_a)$$

If the two propellers are rotating at the same speed in the same direction then

$$p_2(\theta_a) = p_1(\theta_a + \pi) = p_1(\theta_b)$$

or

$$p_{Ta} = p_1(\theta_a) + p_1(\theta_b) = p_1(\theta_a) + p_1(\theta_a + \pi)$$

In the case of two propellers rotating in different direction then

$$p_2(\theta_a) = p_1(\pi - \theta_a) = p_1(\theta_b)$$

Thus

$$p_{Ta} = p_1(\theta_a) + p_1(\pi - \theta_a)$$

In the case the propeller is at an angle α with the fuselage (as in sketch above), the angle θ will then be shifted by an angle of α . The synthesis of pressure distribution on the fuselage due to 2 propellers (rotating in the same direction and different direction) using experimental data are shown on fig.4.21, & fig.4.22. The figures show the pressure distribution in the circumferential direction for mic.7 under static and reverse thrust conditions.

From both figures, it is observed that, for the case of propellers rotating either in the same or opposite direction, the radial pressure distribution remain almost the same. The figures indicate that in the region of $\theta = 0$ or 180 degrees the fuselage would experience high pressures, while that at $\theta = 90$ or 270 degrees it would experience low pressures. The high pressures region at $\theta = 0$ or 180 degrees is expected since at these regions the fuselage pressure is dominated by the nearby propeller (it can also be seen from fig.4.12).

As a reference to the synchrophasing technique, because of the domination of pressure distribution from each individual propeller in the regions of $\theta = 0$ and 180 deg., possible cancellation then only occurs only where the contribution of each propeller is relatively equal i.e at $\theta = 90$ or 270 deg., where the pressures are already low (compared to that at $\theta = 0$ or 180 deg.). The technique, hence has little effect at the high pressure regions.

Mic.1, $\theta = 0$

Mic.2, $\theta = 0$

Mic.3, $\theta = 0$

Mic.4, $\theta = 0$

Mic.5, $\theta = 0$

Mic.6, $\theta = 0$

Mic.7, $\theta = 0$

Mic.8, $\theta = 0$

Figure 4.1: Pressure waveforms of microphones at different locations, $\theta = 0$, static ; Angle θ in degrees

Fig. 4.1.a : Pressure time histories at $\theta = 0$ deg. (using Garrick and Watkins equation). Static.

Mic. 4 , $\theta = 60deg$.

Mic. 6 , $\theta = 60deg$.

Figure 4.2: Pressure waveforms of mic. 4 and mic. 6 at $\theta = 60$. and $J = .4$

Mic.1, $\theta = 0$

Mic.2, $\theta = 0$

Mic.3, $\theta = 0$

Mic.4, $\theta = 0$

Mic.5, $\theta = 0$

Mic.6, $\theta = 0$

Mic.7, $\theta = 0$

Mic.8, $\theta = 0$

Figure 4.3: Pressure waveforms of microphones at different axial locations for the reverse thrust case at $\theta = 0$; Angle θ in degrees

Mic.1, $\theta = 90$

Mic.2, $\theta = 90$

Mic.3, $\theta = 90$

Mic.4, $\theta = 90$

Mic.5, $\theta = 90$

Mic.6, $\theta = 90$

Mic.7, $\theta = 90$

Mic.8, $\theta = 90$

Figure 4.4: Pressure waveforms of microphones at different axial locations for the reverse thrust case at $\theta = 90$.

Fig. 4.5 : Typical near field noise spectra (mic.2 at $\theta=35^\circ$ & $J=0.4$).

Figure 4.6: Spectrum obtained for mic.2 (in the propeller plane) at $\theta = 60deg$. $\bar{J}=.8$

Fig. 4.7 : Spectrum obtained for Mic.7 (5" behind propeller plane) at $\theta = 60$ deg. & $J = 0.8$.

Fig 4.8 : Pressure distribution in axial direction (Propeller plane at Mic.2; Static case)

Fig. 4.9 : Pressure distribution in the axial direction (Propeller plane at mic.2;

J=0.4)

Fig. 4.10 : Pressure distribution in the axial direction (Propeller plane at mic 2; $J = 0.8$)

Fig. 4.10.a : Variation of pressure as function of $1 / \text{distance}$. Static.

Fig. 4.10.b : Variation of pressure as function of $1 / \text{distance}$. $J = 0.8$.

Fig. 4.11 : Pressure distribution in the axial direction (Propeller plane at mic.2; Reverse thrust)

Fig. 4.11.a : Axial pressure variations at different radial positions (using Gar-
rick & Watkins equation).

Fig. 411.b : Pressure variation in the axial direction at two different radial positions. Static.

$20 \log (p_{mic}/p_{(2)})$

(dB)

10

0

-10

-20

Mic. # . 1

2

3

4

5

6

7

8

Measured

Calculated

$\theta = 90 \text{ deg.}$

$20 \log (p_{mic}/p_{(2)})$

(dB)

10

0

-10

Mic. # . 1

2

3

4

5

6

7

8

Measured

Calculated

$\theta = 180 \text{ deg.}$

Fig. 4.11.c : Pressure variation in the axial direction at two different radial positions . Static.

Mic.7 , Static condition.

Mic.7 , Reverse thrust

4.12: Polar plots for typical Variation of pressure ratio (p_θ/p_{ref}) in the circumferential direction ($p_{ref} = p_{\theta=0}$)

Fig. 4.12(a) : Polar plots for typical variation of pressure ratio (p_a/p_{ref}) in the circumferential direction ($p_{ref} = p_{ref}$)

Chapter 8

APPENDIX B

B.1 Power Supply Unit

Figure B.1 shows the connection for the power supply of the motor. A variac is used to control the input voltage to the power supply unit. The switch and the fuse are installed for safety reasons. The transformer changes the alternating high voltage at the input to an acceptable level. For an ideal transformer the product of the voltage times current is constant at the input as well as the output. Thus the output provides a nominal 11 VAC at approximately 15 Amp. This is well within the power requirements of the electric motor.

Since the motor operates on a direct current (DC), the output of the transformer is connected to a full wave rectifier. Because of the large current draw of the motor a filter capacitor was not installed.

The bridge rectifier is used to convert the AC signal to a DC signal. A heat sink is also connected to the bridge rectifier, and a heat sink compound is used to enhance the heat transfer between the two surfaces.

The voltage required by the motor is between 8 and 10v, thus, a Hammond 165-U11 transformer and a Motorola bridge rectifier were chosen. The power switch is a lighted AC switch (5 amp at 250 Vac) . The fuse is rated at **2 Amp.** . The entire unit is enclosed in a ground- chassis.

Fig. 4.14 : Sound spectrum amplitude contours. (in dB, re arbitrary reference level)($J=0.4$)

Fig.4.15 : Sound spectrum amplitude contours. (in dB., re arbitrary reference level)($J=0.8$)

CONTOUR KEY	
1	-20.0
2	-25.0
3	-30.0
4	-35.0
5	-40.0
6	-45.0
7	-50.0
8	-55.0
9	-60.0

Fig.4.16 : Sound spectrum amplitude contours. (in dB., re arbitrary reference level)(Reverse thrust)

Fig.4.17: Trace Velocities between pairs of microphones at different radial positions (m/s). (J=0)

Fig.4.18: Trace velocities between pairs of microphones at different radial positions (m/s).

Fig. 4.19 : Trace velocities between adjacent microphones (m/s). ($J=8$). (Based on absolute phase difference). (m/s.)

Fig. 4.20 : Trace velocities between adjacent microphones (m/s.) (reverse thrust) (m/s).

Fig. 4.21 : Pressure ratio at different radial angles (due to 2 propellers rotating in the same direction). Pressure at $\theta = 0\text{deg.}$ as reference.

fig. 4.22 : Pressure ratio at different radial angles (due to 2 propellers rotating in different direction). Pressure at $\theta = 0\text{deg.}$ as reference.

Chapter 5

Conclusions

This thesis presents a description of the acoustic field generated by a two blade propeller on a simulated fuselage wall. A series of 8 flush-mounted microphones to measure the impinging noise and point to point trace velocities at four flow conditions. The following conditions can be drawn as the results of this study :

1) At any axial distance, the high pressure side is the one nearer to the propeller. At any radial position the high pressure region is experienced near the propeller plane.

2) The synthesis of pressure due to two propellers shows that at the regions of $\theta = 0$ deg or $\theta = 180$ deg. there is a pressure dominated region aft of the propeller.

3) The effect of increasing in forward velocities is to reduce the pressure noise levels. This reduce in noise pressure levels at low harmonic numbers is due to the decrease in thrust.

4) Examination of trace velocities between pairs of microphones led to description of a rigid body acoustic pressure field rotating with the propeller, and acoustic waves originating in the propeller region, propagating longitudinally, and impinging on the fuselage wall at almost zero degree angle.

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Chapter 7

APPENDIX A

A.1 Program for Garrick & Watkins Equations

```

C *****
C
C THE FOLLOWING PROGRAM APPLIES SIMPSON RULE FOR INTEGRATION
C A,B :LIMITS OF INTEGRATION ; C = REAL PART OF INTEGRATION
C D = IMAGINARY PART OF INTEGRATION ; Q = TORQUE ; T = THRUST
C W = MACH # ; XX = AXIAL DISTANCE ; ANGLE = PHASE ANGLE
C F1,F2,F3,F4,F5 ARE INTEGRATION FUNCTIONS ; N = # OF DIVISIONS
C ALL UNITS USED ARE SI UNITS
C *****
C      IMPLICIT REAL*8 (A-H,O-Z)
C      DOUBLE PRECISION XX,BE,W,RE,DE,VM,XT
C      DIMENSION Y(12),Z(12)
C      DATA Y(1)/.1143/,Y(2)/.1191/,Y(3)/.1329/,Y(4)/.1535/,
C * Y(5)/.1778/,Y(6)/.2227/,Y(7)/.2413/,Y(8)/.2227/,Y(9)/
C *.1778/,Y(10)/.1535/,Y(11)/.1329/,Y(12)/.1191/
C      DATA Z(1)/0.0/,Z(2)/.0243/,Z(3)/.0449/,Z(4)/.0587/
C      DATA Z(5)/.0635/,Z(6)/.0449/,Z(7)/0.0/,Z(8)/-.0449/
C * ,Z(9)/-.0635/,Z(10)/-.0587/,Z(11)/-.0449/,Z(12)/-.0243/
C      S(X)=(XX**2+BE*((Y(K)**2)+RE**2+(Z(K)**2))-2.*BE*RE*(Z(1)*
C * DCOS(X)+Z(K)*DSIN(X)))*.5
C      O(X)=(W*XX+S(X))/BE
C      F1(X)=DCOS(2.*X+(5.3123*O(X)))/(S(X))
C      F2(X)=DSIN(2.*X+(5.3123*O(X)))/(S(X))
C      F3(X)=DCOS(2.*X+(5.3123*O(X)))/(S(X)**2)
C      F4(X)=DCOS(2.*X+(5.3123*O(X)))/((S(X)**3))
C      F5(X)=DSIN(2.*X+(5.3123*O(X)))/((S(X)**2))
C      F6(X)=DSIN(2.*X+(5.3123*O(X)))/((S(X)**3))
C
C THE DO LOOP CALCULATES THE INTEGRATION FOR DIFFERENT
C VALUES OF X
C      DO 105 K=1,12
C      DO 99 J=1,8
C      XT=(DBLE(J)*1.-2.)
C      XX=(XT/12.)*.3043
C      P=0.0
C      C=0.0
C      D=0.0
C      T=8.62
C      Q=.1524
C      V=2.6523
C      W=0.00
C      Y=.1143

```

```

RE=((3.5*.8)/12)*.3048
Z=0.0
E=2./(RE**2)
A=0.0
BE=1.-(W**2)
B=2.*3.141592653589793
S1=0.0
S2=0.0
S3=0.0
S4=0.0
S5=0.0
S6=0.0
N2=600
N=N2/2

```

```

PRINTING HEAD LINE

```

```

WRITE(6,11)
FORMAT(//,° *** ***** ***)
WRITE(6,11)
FORMAT(//,° STATIC CASE°)
WRITE(6,11)W,XT
FORMAT(//,° FOR EACH I = °,F5.2,3X,°AT A ( in. ) = °,F5.2)
H=(B-A)/OBL E(N2)
S1=F1(A)+4.*F1(A+H)
N1=N-1
DO 2 I=1,N1
S1=2.*F1(A+H*DBLE( 2*I))+4.*F1(A+H*DBLE( 2*I+1))+S1
S1=H*(F1(B)+S1)/3.
S2=F2(A)+4.*F2(A+H)
S3=F3(A)+4.*F3(A+H)
S4=F4(A)+4.*F4(A+H)
S5=F5(A)+4.*F5(A+H)
S6=F6(A)+4.*F6(A+H)
DO 3 I=1,N1
S2=2.*F2(A+H*DBLE( 2*I))+4.*F2(A+H*DBLE( 2*I+1))+S2
S2=H*(F2(B)+S2)/3.
DO 4 I=1,N1
S3=2.*F3(A+H*DBLE( 2*I))+4.*F3(A+H*DBLE( 2*I+1))+S3
S3=H*(F3(B)+S3)/3.
DO 6 I=1,N1
S4=2.*F4(A+H*DBLE( 2*I))+4.*F4(A+H*DBLE( 2*I+1))+S4
S4=H*(F4(B)+S4)/3.
DO 7 I=1,N1

```

```

7      S5=2.*F5(A+H*DBLE( 2*I))+4.*F5(A+H*DBLE( 2*I+1))+S5
      S5=H*(F5(B)+S5)/3.
      DO 8 I=1,N1
3      S6=2.*F6(A+H*DBLE( 2*I))+4.*F6(A+H*DBLE( 2*I+1))+S6
      S6=H*(F6(B)+S6)/3.
      C=(T*XX*S4)+(T*V*W*S2/(BE))+(T*V*XX*S5/BE)-(Q*S2*E)
      D=(T*V*XX*S3/BE)-(T*XX*S6)-(Q*S1*E)+(T*V*W*S1/BE)
      P=((C**2)+(D**2))**.5
      P=P*(2.**.5)/(8.*(3.14159**2))
      WRITE(6,22)XX,Y(K),Z(K)
22     FORMAT(/,4X,' AT X = ',F7.4,3X,' Y = ',F7.4,3X,' Z = ',F7.4)
      ANGLE= ( DATAN(D/C))*360./B
      WRITE(6,9)C,D
9      FORMAT(/,4X,' REAL = ',F14.7,2X,' IMAGINARY = ',F14.7)

      WRITE(6,77)P,ANGLE
77     FORMAT (/,4X,' PRESSURE = ',F14.7,3X,' PHASE ANGLE = ',F7.3)
      VM=(P*.1)
      DB=20.*DLOG10(VM)
      WRITE(6,100)DB
100    FORMAT(/,4X,' THE PRESSURE IN db IS = ',F14.7)
C PRESSURE AT MIC.2 (X=0,Y=.0143,Z=0) IS USED AS REFERENCE
C
      RATIO= 2/(10.626)
      WRITE(6,101) RATIO
101    FORMAT(/,4X,' PRESSURE RATIO = ',F14.7)
C      IF (J.EQ.11) GO TO 91
      VM=0.0
      XT=0.0
39     CONTINUE
      IF (K.EQ.12) GO TO 91
103    CONTINUE
91     STOP
      END

```


Chapter 8

APPENDIX B

B.1 Power Supply Unit

Figure B.1 shows the connection for the power supply of the motor. A variac is used to control the input voltage to the power supply unit. The switch and the fuse are installed for safety reasons. The transformer changes the alternating high voltage at the input to an acceptable level. For an ideal transformer the product of the voltage times current is constant at the input as well as the output. Thus the output provides a nominal 11 VAC at approximately 15 Amp. This is well within the power requirements of the electric motor.

Since the motor operates on a direct current (DC), the output of the transformer is connected to a full wave rectifier. Because of the large current draw of the motor a filter capacitor was not installed.

The bridge rectifier is used to convert the AC signal to a DC signal. A heat sink is also connected to the bridge rectifier, and a heat sink compound is used to enhance the heat transfer between the two surfaces.

The voltage required by the motor is between 8 and 10v, thus, a Hammond 165-U11 transformer and a Motorola bridge rectifier were chosen. The power switch is a lighted AC switch (5 amp at 250 Vac) . The fuse is rated at **2 Amp.** . The entire unit is enclosed in a ground- chassis.

Fig. B.1 : The Power Supply Unit.

Chapter 9

APPENDIX C

The Trigger Control Unit

Figure C.1 illustrates the trigger control unit. It consists of a reflective object sensor and a TL704 operational amplifier. The sensor contains a light emitting diode and a phototransistor. The diode emits infrared light which is reflected off a target. The target is made of aluminum foil attached to the back of the propeller. The beam is reflected back to the sensor only when the propeller is aligned with the sensor. The reflected beam is picked up by the transistor side, which then supports a large current flow and the output voltage "goes low" (high = 5V, Low = 0 V). The sensor is positioned behind the propeller in such a way that the transistor will only conduct if the propeller blade passes by the fuselage at $\alpha = 0$ deg. At other times the transistor does not conduct and the output is "high".

The operational amplifier is used to condition the signal from the output of the object sensor and to produce a clean 0-5 V TTL compatible signal. The op. amp. has two inputs. One, the reference is held at 2.5V, the other is supplied by the sensor output. The op. amp. "compares" this to the 2.5V. If the signal is above 2.5V it will output a signal as "high" (5V), and if not it will output a low (0V). The result is a clean 0-5 V TTL compatible signal.

Fig. C.1 : The Trigger Control Unit.

Chapter 10

APPENDIX D

D.1 Pre-amplifier/Channel Selector Unit

Figure D.1 is the circuit diagram for the pre- amplifier/channel selector unit. The unit contains a dual position rotary switch which steps through the microphone channels, 2 channels at a time. Each channel is designated to receive a microphone signal , the signal is amplified by non inverting operational amplifier with gain of 67. The operational amplifiers used are low noise TL704.

$$R_1 = 0K\Omega$$

$$R_2 = 680K\Omega$$

$$C_1 = .1\mu F$$

Pre-amplifier unit.

$$R_4 = 82K\Omega$$

$$C_3 = 18pF$$

Each Channel is connected to a pre-amplifier unit seen on the left.

Fig. D.1 : The pre-amplifier / Channel Selector Unit.

END

140990

FIN