OPPORTUNITIES FOR FUTURE IMPROVEMENTS
IN AIRCRAFT NOISE

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Abstract

Current status of theoretical understanding, experimental techniques and potential future noise reductions are examined for fan source noise, low frequency core engine noise, jet exhaust noise and airframe noise components. Potential future improvements in acoustic linings are also reviewed. It is concluded that there are a number of interesting possibilities for advancement of noise technology, but that energy, emissions and cost constraints will limit future noise reductions to relatively modest increments below the current wide-body fleet.

Introduction

Aircraft community noise continues to be an important issue for homeowners adjacent to airports, environmental groups, airport operators, airlines, engine and airframe manufacturers and governmental regulatory agencies. There is general agreement that improvements in noise would be desirable, and in many cases, essential to the growth and well being of the air transportation industry. However, there are many unanswered questions relating to the best ways of achieving further reductions of noise while at the same time recognizing the need to minimize engine exhaust emissions, conserve energy, and maintain sufficient economic health of the air transport industry to stimulate continued financial investment.

This paper will address one aspect of this very complicated situation by considering the key aircraft noise components that contribute to total community noise: fan noise, low frequency core noise, jet exhaust noise and airframe noise. The assumption is made that further noise reductions below the current widebody fleet must be achieved with reasonable energy or economic penalties; thus the range of improvements considered has been limited to those which are projected to be economically acceptable.

The following sections review, for each noise component, general background on the component, the current state of theoretical understanding, experimental techniques and potential reductions in component noise level.

Fan Noise

Background

The most widely used design criteria for fan-source noise control is that developed by Tyler and Sofrin who presented a method for the selection of the ratio of the number of blades and vanes of a fan stage so that, under certain circumstances, the aerodynamic interactions between these blade rows generate pressure fluctuations that decay very rapidly with distance and are theoretically incapable of carrying acoustic energy. This method has been used to select blade/vane ratios in high-bypass engines that power current wide-body aircraft and is partially responsible for the relative quietness of these engines. Unfortu-

nately, this method is limited in that other sources play a role when the rotor relative tip speed is supersonic, and structural and aerodynamic design constraints do not permit blade/vane ratios that cut off the overtones of the blade passing frequency, even at subsonic rotor tip speeds.

During the past decade much research has been directed toward finding additional methods for fan-noise control. Most of this research has been experimental and has attempted to find aerodynamic configurations in which surfaces experiencing large periodic force fluctuations are minimized. Configurations that have been tested include serrated leading and/or trailing edges, tilted or swept stators, large rotor-stator spacing and a wide variety of blade cross sections.

It has been found that the fan-stage of an engine operating at rest generates noise differently than when it is in flight. This will be discussed further in a later section; however for this reason it is obvious that current static experiments on fan configurations for noise reduction can be somewhat misleading. Prior to embarking on further testing, it is imperative to find methods that permit good simulation of the mechanisms for noise generation in flight.

Theoretical Understanding

The framework of a theoretical understanding of fan-noise generation is available as an extension of Lighthill's general theory of aerodynamic sound. One such extension is the analysis of Lowson wherein noise generation is related to rotor and stator force fluctuations [1]. These force fluctuations are derived from a combination of incompressible, unsteady aerodynamic theory and empirical turbulent flow descriptions. Herein lies the crux of the problem of developing an accurate model for calculation of absolute noise levels.

Boeing has developed a computer program which includes the Lowson source model as well as accounting for steady and unsteady inflow distortions. This program calculates modal amplitudes in either a circular or annular duct with steady mean flow and then calculates the radiated far-field noise. Only tones at the harmonics of blade-passing frequency are calculated by this program. A satisfactory model for broadband noise is not yet available; however, some attractive models which employ the pulse modulation theory for inflow distortion and rotor wake/stator interaction sources are available.

The major fan-source noise mechanisms are summarized in Figure 1. Except for noise caused by rotor leading-edge shocks, all the mechanisms are attributed to unsteady forces on the blade surfaces caused by unsteady flow. The resulting spectrum consists of both discrete (tone) and broadband frequency components. The tones are generated by periodic component forces while the broadband results from random forces. The noise intensity propagated to the far-field is influenced by the upstream and downstream duct geometric and aerodynamic characteristics.
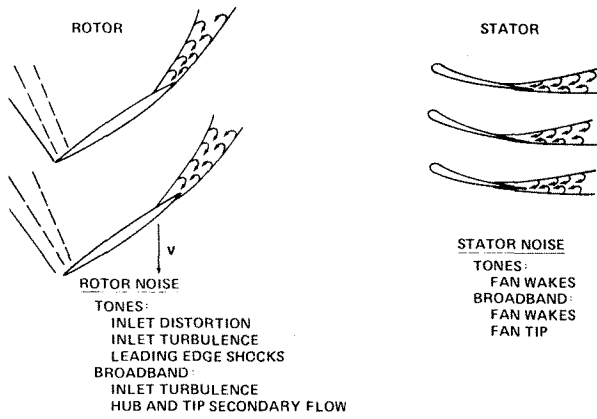
ROTOR

STATOR

STATOR NOISE
TONES:
  FAN WAKES
BROADBAND:
  FAN WAKES
  FAN TIP

ROTOR NOISE
TONES:
  INLET DISTORTION
  INLET TURBULENCE
  LEADING EDGE SHOCKS
BROADBAND:
  INLET TURBULENCE
  HUB AND TIP SECONDARY FLOW

CF6–50 ENGINE
AIAA PAPER 75-465 BY
J. P. ROUNDHILL AND
L. A. SCHAUT

Fig. 2C.   Experimental Observations

Fig. 1.   Major Fan Noise Mechanisms

Future improvements of the analytical methods will include
urate aerodynamic calculations for the high-speed compressible
ee-dimensional flows that exist in turbofan engines. This may
the result of analytical developments or a more unified empiri-
data base. The extremely complex flow structures that exist
vnstream of the rotor tip or in the wake at the junction of the
nping shroud and the rotor blade must be measured and ade-
tely described so that they can be modeled in the rotor/stator
:raction source.

)erimental Techniques

During the past three years, literature concerning static-to-
it effects on fan noise has enlarged tremendously. Experimen-
results are available comparing static and in-flight noise gener-
1 by the current family of high-bypass ratio engines, all of
.ch have a large single-stage fan without inlet guide vanes.
ind pressure data taken with microphones mounted flush to the
t diffuser wall are compared in figure 2. It is obvious that

when the rotor-tip speed is subsonic, a dramatic difference be-
tween static and in-flight noise levels exists. Also, the static-to-
flight noise difference is not equal for the engines shown. When
the rotor-tip speed is supersonic, there is no clear result; two
measurements show no static-to-flight effect while the third shows
a large effect. There is a consensus of opinion that the excess
blade-passing tone noise measured in the static tests at subsonic
tip speeds is generated in a large part by atmospheric turbulence
interacting with the rotor. Direct measurements of the ingested
turbulence structure have been made and are summarized in
figure 3, where it can be seen that the prime conditions for noise
generation, significant transverse turbulence fluctuations with
small transverse scale and large axial scale, exist during typical
static-noise tests.

Variation of BPF Fundamental Level with Tip Relative Mach Number, Static and In-Flight.

TONE-LEVELS TAKEN FROM 32 Hz FILTER-BANDWIDTH
SPECTRA

RB211 ENGINE
ASME PAPER 73-WA/GT-4
BY N.A. CUMPSTY AND
B. W. LOWRIE

KEY
  □ STATIC
  x FLIGHT 140 kts } @ 1500-2000 FT
  ∇ FLIGHT 170 kts }
  ○ PREVIOUS DATA STATIC
  ● PREVIOUS DATA FLIGHT
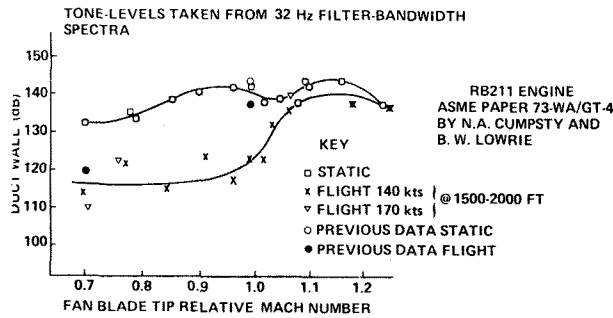
FAN BLADE TIP RELATIVE MACH NUMBER

Fig. 2A.   Experimental Observations

Comparison of Fan Fundamental Tone Levels Measured on The Inlet Wall
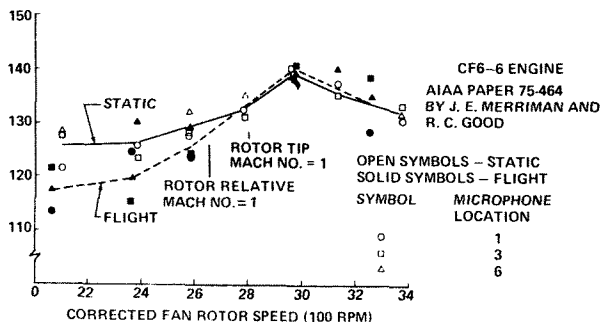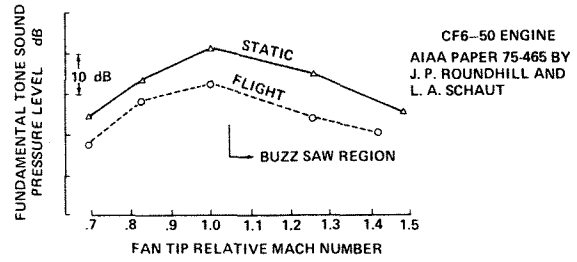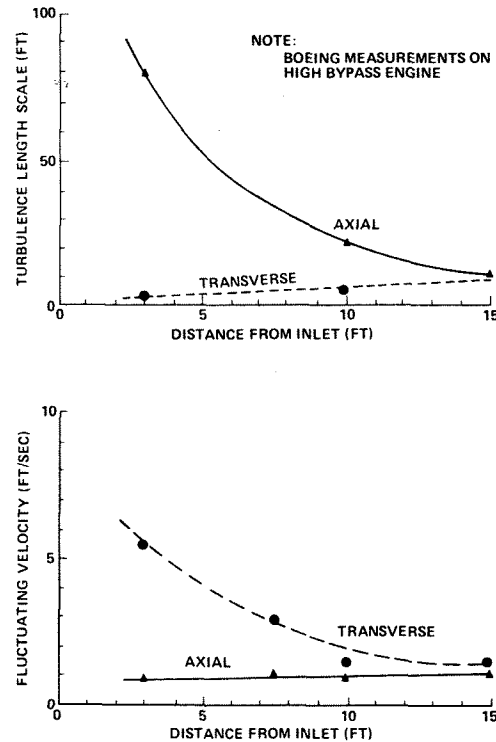of The CF6-6 Engine During Static and Flight Conditions.

CF6–6 ENGINE
AIAA PAPER 75-464
BY J. E. MERRIMAN AND
R. C. GOOD

OPEN SYMBOLS – STATIC
SOLID SYMBOLS – FLIGHT

SYMBOL	MICROPHONE LOCATION
○	1
◻	3
△	6

ROTOR TIP MACH NO. = 1
ROTOR RELATIVE MACH NO. = 1

CORRECTED FAN ROTOR SPEED (100 RPM)

Fig. 2B.   Inlet Fan Noise Static-To-Flight Effect

NOTE:
  BOEING MEASUREMENTS ON
  HIGH BYPASS ENGINE

AXIAL

TRANSVERSE

DISTANCE FROM INLET (FT)

TRANSVERSE

AXIAL

DISTANCE FROM INLET (FT)

Fig. 3.   Static Inflow Turbulence

The goal of Boeing research, to find a method of conducting
static tests which closely simulate actual flight conditions, has
been previously described [2]. This program is continuing and is
expected to produce a viable test technique for outdoor engine rig
use in the future.

Potential Improvements

As we move forward from the current comparatively quiet fan-engine installations it is natural to ask, "What source noise reduction methods will become available for the fans of future engines?" A discussion of this subject has been presented by Schairer and Gerend [2].

One of the concepts discussed in reference (2) has been pursued further during the past year. This concept, called the stator-bypass system, is shown schematically in figure 4 and is envisioned to reduce fan-stage noise by utilizing a ducting system to extract approximately 10% of the fan flow from behind the rotor and prevent it from interacting with the stator row. The result is tone and broadband noise reduction since the mean flow velocity incident onto the stators is decreased and the highly turbulent flow caused by the rotor-tip vortices and secondary flows does not interact with the stators. Further, depending on the requirements of the particular installation, the bypassed flow may be discharged from a separate nozzle causing the rotor to be slightly unloaded.
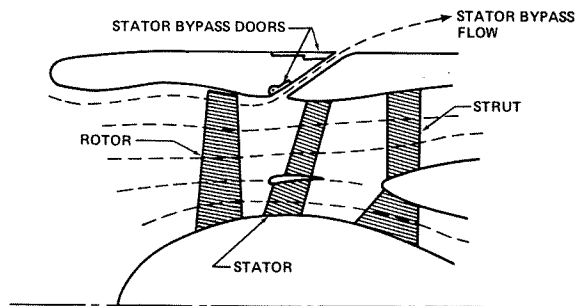

Fig. 4. Fan Stator Bypass Door Schematic

Figure 5 shows some experimental measurements taken on a Boeing 15 in. dia. model fan that is aerodynamically similar to fan stages on current high-bypass ratio engines, although the acoustics are somewhat different since the rotor/stator interaction is acoustically cut on at the blade-passing frequency. It has been verified that the residual rotor-inflow distortion tone at blade-passing frequency is reduced by an amount similar to that when the stators are not present. It can be seen that the substantial noise reduction is obtained across the entire spectrum. For this particular case, unloading the rotor provides some noise reduction in addition to the bypass system.

Predicted thrust loss for the stator bypass system is a maximum of 2% during takeoff operation and not a factor at approach conditions since engine speed can be adjusted for loss produced by the bypass system. Since this system will not be operated during climb or cruise, the takeoff and approach noise reduction is obtained without impacting the rest of the mission.
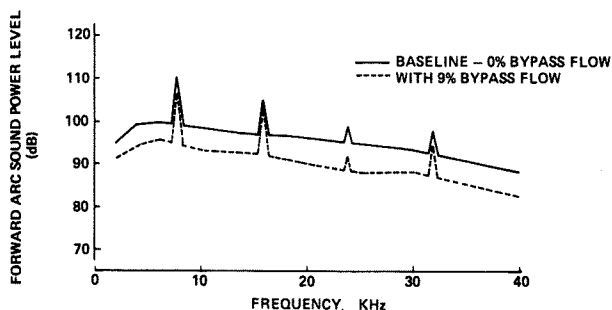

Fig. 5. Stator Bypass System Noise-Reduction at Approach Power

It appears that other potential noise reduction methods are "acoustically" available but require development of structural and manufacturing technology. These include such items as high-tip speed fans with variable stagger for improved off-design aerodynamics, rotors incorporating blown trailing edges to eliminate the wakes, and rotors with extremely small tip clearance. It is thought that a tip-shrouded rotor without part-span damping rings may also provide reduced fan-noise levels. Finally it is suspected that either the existence of and/or the mean asymmetry of the inlet cowl boundary layer may be a cause of fan noise. Some data indicate that this source may be dominant in low-speed fans. One may therefore conclude that noise reductions may be possible by controlling the inlet boundary layer.

Low Frequency Core Noise

Background

Core noise may be defined as the portion of the engine exhaust noise which is attributed to a number of independent sources located within the engine primary stream tailpipe. The general characteristics such as level, spectrum shape and directivity of core noise have been identified in several engines. It has been observed that core noise has a broadband spectrum with most of its energy below 1000 Hz.

Although core noise is thought of as being a potential problem at low power settings, in many high-bypass ratio turbofan engines it may also become a key noise component even at takeoff power settings. In other gas turbine applications, such as the auxiliary power units which have relatively low exit velocities, nearly all low-frequency exhaust-radiated noise is core noise. Since stringent regulations on airport ramp noise are anticipated, the APU noise reduction effort has assumed considerable importance.

A number of empirical prediction methods have been developed based on data from full-scale engine tests [3]. These methods provide an interim solution to the need in the aircraft industry for estimation of core noise levels. They are, however, inadequate for understanding the generation mechanisms and developing reduction techniques.

In addition, since core noise sources are located within the engine, the duct effects on noise sources, noise transmission in complicated flow passages and radiation of noise out of the engine exhaust also play an important role.

Potential sources of core noise include:

● Direct noise from the burner resulting from the pressure unsteadiness which accompanies turbulent combustion.

● Indirect burner noise from interactions of burner generated velocity and temperature fluctuations with the turbine.

● Noise generated further downstream at the exhaust struts caused by turbulence and/or swirl in the exhaust flow.

● Noise generated at the nozzle lip by interaction with flow turbulence.

A detailed description of each of the above is given in reference (4). It is essential to understand the mechanism of generation of each one of these sources in order to develop efficient core noise reduction devices.

Experimental Techniques

Direct combustion noise has been recognized as a dominant core noise source. Recent Boeing tests on a high-bypass ratio engine have shown that at some power settings there is a good correlation between the combustor and primary tailpipe pressure fluctuations. In another test of a turboshaft engine, an even stronger correlation was obtained between far-field sound pressure and combustor exit pressure fluctuations. Extensive combustion noise research has been done in the past few years. Theoretical approaches to combustion noise have been presented by Strahle[5] and Chin and Summerfield[6]. The theories, at present, can predict the trends reasonably well although they are incapable of predicting absolute noise levels. Experimentally, noise from various combustor designs, ranging from simple laboratory type burners to full-scale engine combustors has been measured. The scaling behavior for sound power, spectral content and directionality has been deduced from these test results.

In one such combustor test at Boeing, inlet temperature, airflow, and fuel flow were varied independently and the parametric relation shown in figure 6 was developed. Results such as this could potentially lead to improved core noise prediction methods and indicate possible noise reduction options. A review of theories and prediction methods for combustion noise is given by Motsinger and Emmerling[7]. In terms of core noise reduction, however, there have been very few research efforts.

In 1974, core noise was isolated and substantially suppressed in a full-scale test setup at Boeing, which is shown in figure 7. The fan exhaust flow was diverted away to remove this source from consideration. Acoustic measurements were made with several nozzle configurations, including a massive muffler especially designed to attenuate the low frequency core noise. The results, shown in figure 8, indicate that core noise was suppressed over a wide frequency range by the muffler. Flightworthy, technically practicable, and economically feasible designs are under study but extensive research and development work is required before a practical design is ready for production.

A concept for reducing combustion noise generated in gas turbine engines has been evaluated at Boeing. The suppressor concept involves perforating the engine casing either over or immediately downstream of the combustor (or extending over both locations) and enclosing this area with resonator cavities. The motivation for the suppressor concept stems from the fact that in modern high-bypass ratio engines, space could be made available for a suppressor between the combustor outer wall and the fan duct as shown by the shaded area in figure 9. Experimental results were obtained for a number of flow and geometric variations using a 20 cm (8 in.) dia. single can-type research combustor. Several configurations, including the 10 cm (4 in.) deep suppressor, gave 5- to 10-dB noise reduction in the 100- to 1000-Hz frequency range when located immediately downstream of the combustor. Typical results obtained are shown in figure 10. The test results indicate that it may be possible to reduce low frequency combustion noise by liners considerably less deep than the conventional theories would predict. More details on this experiment are given in reference (8).
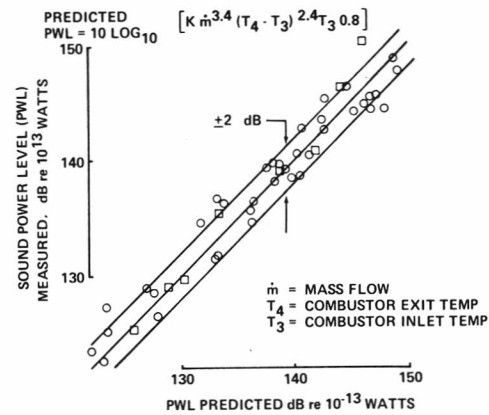
$$\text{PREDICTED PWL} = 10 \log_{10} \left[K \dot{m}^{3.4} (T_4 \cdot T_3)^{2.4} T_3^{0.8} \right]$$

$\dot{m}$ = MASS FLOW
T_4 = COMBUSTOR EXIT TEMP
T_3 = COMBUSTOR INLET TEMP

Fig. 6.   Combustion Noise Correlation — Boeing Test Results
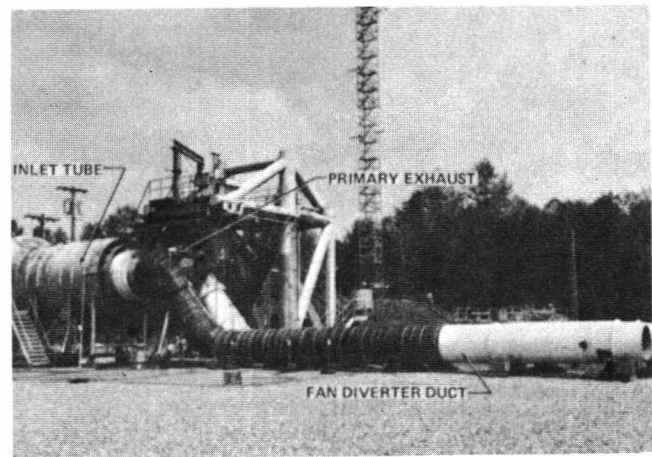

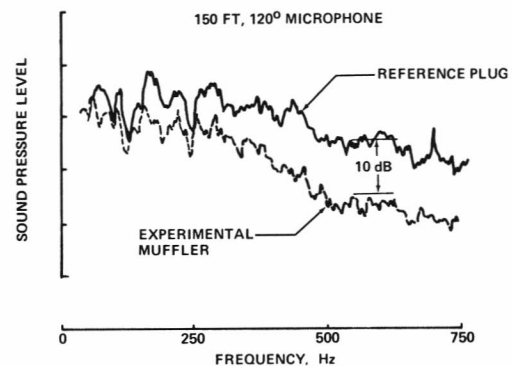*Fig. 7.   JT9D Core Noise Isolation Test Hardware*
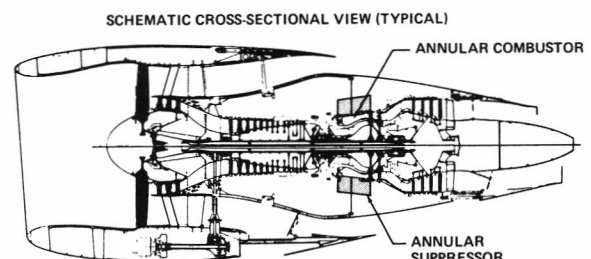

Fig. 8.   Experimental Core Noise Muffler Results

Fig. 9.   High Bypass Ratio Engine Showing Possible Location for the Combustor/Suppressor
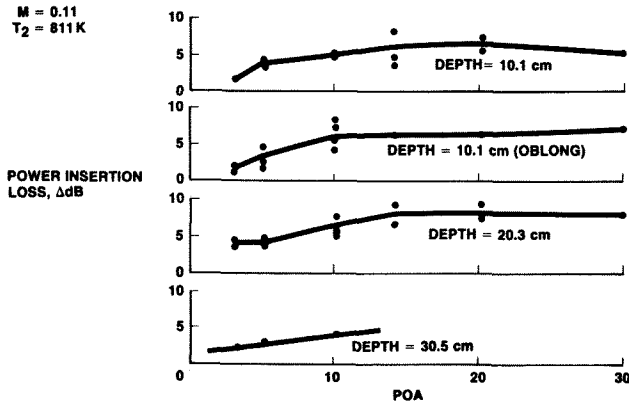
602

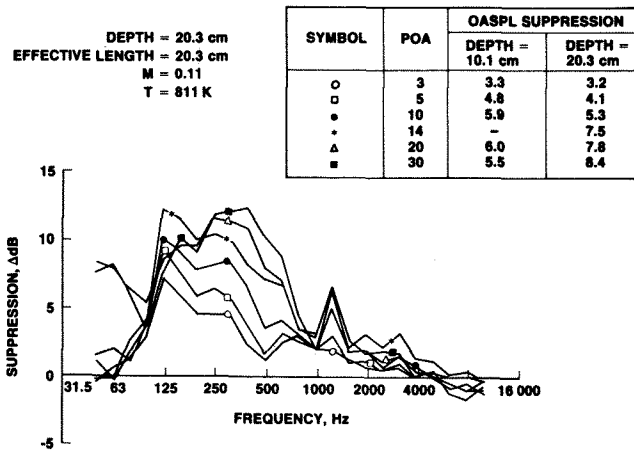Fig. 10A. Power Insertion Loss as a Function of Liner Depth and POA

DEPTH = 20.3 cm
EFFECTIVE LENGTH = 20.3 cm
M = 0.11
T = 811 K

SYMBOL	POA	OASPL SUPPRESSION	
		DEPTH = 10.1 cm	DEPTH = 20.3 cm
o	3	3.3	3.2
□	5	4.8	4.1
●	10	5.9	5.3
◆	14	–	7.5
△	20	6.0	7.8
■	30	5.5	8.4

Fig. 10B. Effects of Liner Depth and Percent Open Area

Potential Improvements

As is true for any noise problem, it is desirable to reduce the noise generated at the source instead of attempting to attenuate it with acoustic linings. This is especially true for core noise which requires relatively larger volumes for suppressors because of the low frequencies involved. Combustor suppressor tests discussed above have shown that the suppressor depths required may be much smaller than the conventional theories predict. Even so, suppressors still mean an extra weight penalty and should be considered only if there is no other choice.

It is known that different burners with the same flow and heat release rates could generate different levels of noise depending upon the combustor design. The critical parameters include fuel vaporization methods, turbulence (both level and intensity) in combustion zones, flame retention and swirl. Research conducted to design clean combustors has shown that some of them are also quieter. As a result of this, it should be possible to include noise as an additional criterion in burner design along with performance and pollution. There are some indications that increasing flame holder area would also lead to quieter burning.

The importance of other core noise sources is currently being studied. Evaluation of combustion noise attenuation, because of its passage through the turbine, is being made. Determination of the effect of engine exit geometry on noise radiation and flight effects on core noise are areas where further research is needed.

Jet Noise

Background / Progress

Jet noise is proving to be extremely difficult to control. Twenty-four years after Lighthill's paper[9], "On Sound Generated Aerodynamically," the most direct method of reducing jet noise is still to slow down the jet. Jet noise suppressor nozzles promote mixing between the high-velocity noise-producing flow and a lower-speed flow, thus effectively slowing the jet. This flow is either the bypass flow in a turbofan engine or the ambient flow for both turbofan and turbojet engines.

Steps have been taken to reduce jet noise in recent years. Engine cycles and methods of mixing the flows have been improved. An improved theoretical and empirical understanding of noise production and propagation has provided concepts such as flow profile shaping and fluid layer shields. Some of these concepts, or variations on them, will probably be developed and applied to future aircraft engines.

However, there is a basic problem. The continuing requirement for more thrust from a given aircraft engine design during its production life almost invariably results in an increase in the exhaust jet velocity, with a resulting increase in jet noise. Thus an engine originally designed to be quiet may become noisier as its thrust performance is improved. Applying a noise suppressor which reduces the thrust of the engine is a self-defeating system. Therefore noise reduction with minimum thrust penalty becomes the goal for jet noise suppression. In addition, the suppressor system must be light, reasonable cost, and require low maintenance to have a minimal effect on aircraft performance and cost of operation.

Theoretical Understanding

The pioneering work of Lighthill showed how sound is produced by fluctuating velocities present in a turbulent jet. Later refinements extended the results to high jet exhaust velocities and aircraft in flight. The application of Lighthill's theory is difficult because of the complex flow and acoustic process in a jet. Alternate approaches have also been proposed. The special case of a parallel mean flow, as treated by Lilley, considers the velocity fluctuations as in the Lighthill theory, but also treats the effect of the sheared, heated jet flow field on sound propagation inside the jet[10]. Some solutions of the Lilley equation have been published[11,12]. Early experiments have revealed large-scale disturbances in a jet and a mixing mechanism called vortex pairing. However, recent experiments by Laufer, show that the noise sources are relatively compact so that the influence of large-scale coherent disturbances on noise is small [13].

A major requirement for prediction of jet noise based on turbulence source theories is a knowledge of both the mean and turbulent flow field. Computer programs for this purpose have been written for axisymmetric jets. The Boeing program has been combined with an acoustic theory based on the Lighthill approach. The program calculates the kinetic energy and length scale of the turbulent velocity fluctuations. Empirical turbulence frequency spectra and correlation volumes were obtained from experimental data and are related by similarity arguments to the local jet flow parameters. Calculated acoustic results are shown for a simple round jet in figures 11 and 12. The spectrum shape agreement with data, in the geometric far-field, is quite good at 90°, the only angle for which the analysis is technically valid because of the neglect of jet propagation effects in the Lighthill approach. A single empirical constant, which has the same value for all cases, provides a good match between the absolute level of the theory and the level of the measured data. Acoustic results for a bypass flow are

shown in figure 13. The program has the capability to include an ambient flow. Thus it can also be used to study flight effects.

Future theoretical developments at Boeing will apply numerical methods of solution of the Lilley equation to the present Boeing flow/noise program, and also study three-dimensional flow effects on noise production and propagation.
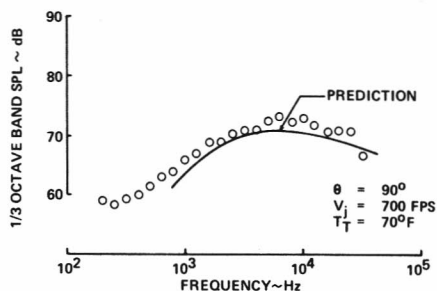

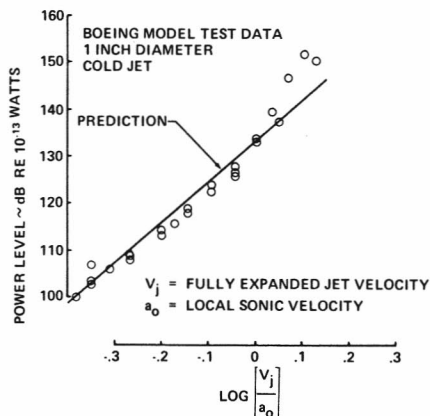**Fig. 11.** *Spectral Comparison of Jet Prediction Method with Cold 1-Inch Diameter Model Data*

Fig. 12. *Comparison of Jet Prediction Method with Cold Model Data*
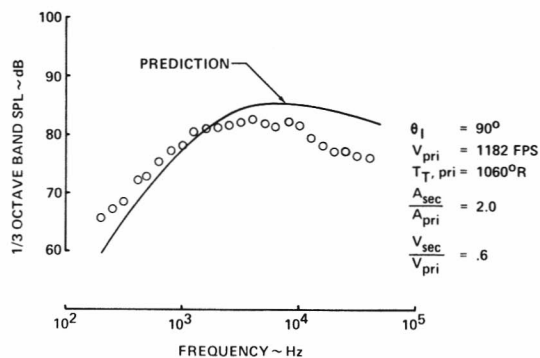
Fig. 13. *Spectral Comparison of Bypass Jet Prediction with Model Data (Hot Primary)*

Ground-to-Flight Effects

The effects of flight on the jet noise characteristics of current and future aircraft engines are an important technology area. The need for accurate flight effect predictions will increase as noise certification requirements become more stringent. Engine cycle variations. installation modifications, and noise suppression devices that are designed to reduce jet noise have been typically developed and evaluated using static-noise test facilities. The poten-

tial improvements must be effective during flight. This requires a precise knowledge of jet noise flight effects or a direct measurement in a simulated flight environment.

Flight tests provide the ultimate demonstration of noise-reduction systems. The use of flight-test during development phases and for detailed noise research has severe limitations in terms of experiment control, accuracy, timing and cost. Consequently, industry and government agencies are developing various alternates to flight-test, such as closed wind tunnels, free jet tunnels, ground based vehicles, and rotating-arm facilities to study flight effects in a controlled and cost-effective manner.

The free jet wind tunnel is the most widespread flight simulation experimental technique in use. Model jets are surrounded by a substantially larger diameter free stream to simulate the flight velocity. Noise measurements are taken outside the surrounding jet. The propagation of the flight jet noise through the shear layer between the free jet and the ambient air requires corrections to the measured noise data. These corrections are currently being developed by a number of researchers. A consensus on the proper correction technique is required before the free jet flight simulation becomes an accepted tool.

Closed wind tunnels, where noise measurements are made within the free stream, are also being developed as a flight simulation technique. Boeing has conducted model tests in a 9 by 9 ft. facility and recently completed a NASA contracted test program in the NASA-Ames 40 by 80 ft. wind tunnel using a JT8D-17 turbofan engine[14]. The engine was tested in three configurations: a quiet nacelle with a 20-lobe ejector/suppressor nozzle (figure 14), an internal mixer, and a baseline conical nozzle configuration. Reverberation, tunnel background noise, and measurements made close to engine noise sources represent the biggest problems for this test technique. Special test and analysis procedures were developed by Boeing to solve these problems. Excellent agreement was obtained between 40 by 80 results and 727/JT8D flight-test results for the quiet nacelle and baseline configurations (figures 15 and 16). These results indicate that the closed wind tunnel is a viable technique for simulating flight noise.

Ground based vehicles include the use of aircraft in taxi mode, tracked vehicles such as the Bertin Aerotrain, and high-speed automobiles as being developed by NASA-Langley. Test hardware costs are reduced and experiment control is improved relative to flight test. The biggest concerns involve flexibility of the vehicle test engine to represent the particular cycle of interest and the relatively short data samples that are obtainable as the
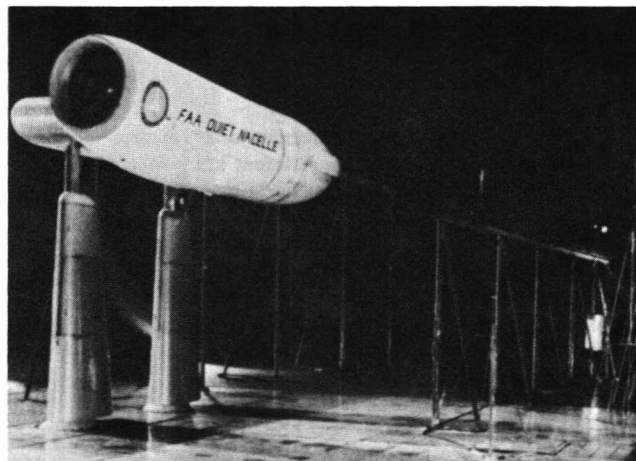

Fig. 14. *FAA Quiet Nacelle*

vehicle passes through the measurement zone. Measurement and analysis procedures are being developed at Boeing to improve the accuracy of the simulated flight noise. Tests utilizing these procedures with an F-86 aircraft in flight and taxiing have shown good agreement.
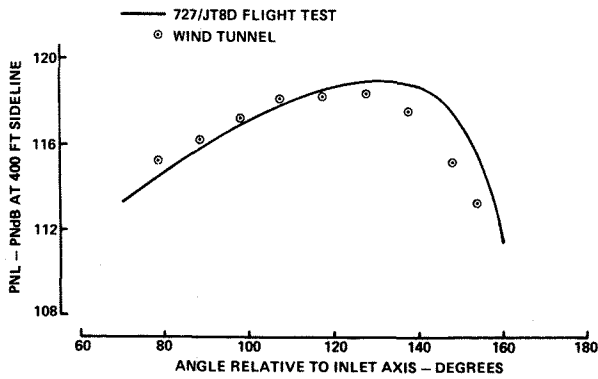

Fig. 15. NASA-AMES 40 by 80 ft. Wind Tunnel Test JT8D Engine with Baseline Nozzle
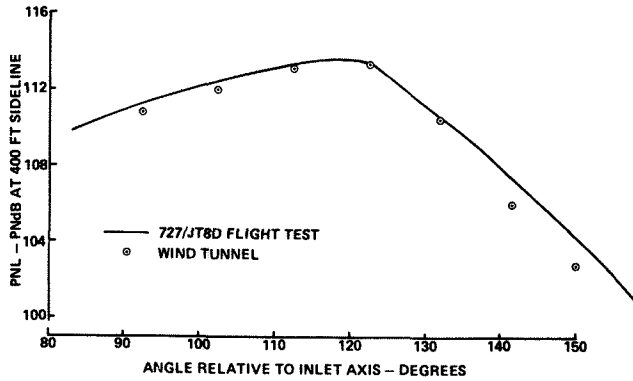

Fig. 16. NASA-AMES 40 by 80 ft. Wind Tunnel Test JT8D Engine with 20 Lobe Ejector/Suppressor

Rolls-Royce is developing a rotating-arm facility to study model jet noise flight effects. The model nozzle is mounted on the tip of the rotating arm where relatively high forward speeds may be obtained. Operating costs are relatively low and a variety of nozzles can be tested. There are problems associated with centrifugal effects and the complicated analysis procedures required to achieve reasonably accurate flight noise from the measured transient signal. Recent test results from the rotor rig facility indicate that significant technique improvements have been made.

Potential Improvements

New developments in jet noise fundamentals and analysis techniques may provide for efficient reduction of jet noise levels. The current problem is to obtain modest noise reductions with low performance losses and minimum weight. Three areas of potential reductions in jet noise are: (1) shaping of the jet velocity profile for minimum noise, (2) using internal mixer nozzles to reduce exit velocities and (3) redirecting the noise with a fluid-layer shield.

Recent analytical studies of the effect of velocity profile shape on jet noise for a high-bypass engine (figure 17) indicate that minimum noise is obtained using a fully mixed or an inverted profile. The inverted profile is obtained by interchanging the primary and secondary streams. A duct-burning turbofan engine has a natural inverted profile shape[15]. The trends indicated by
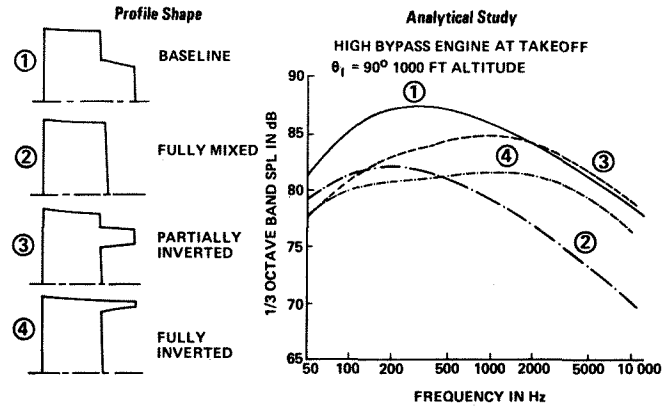

Fig. 17. Jet Noise Reduction with Flow Profile Shaping

the analytical results have been confirmed by recent model-scale experimental results, shown in figure 18. On a spectral basis, the low-frequency noise is minimized with an inverted profile, but at the expense of an increase in high-frequency noise. As shown in Figure 19, the partially-inverted velocity profile yields the lowest PNL for the conditions shown.
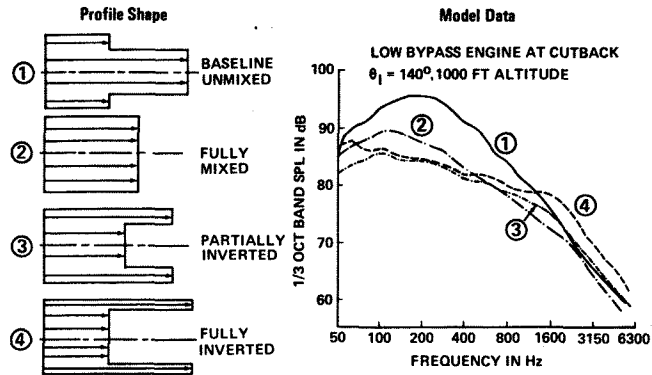

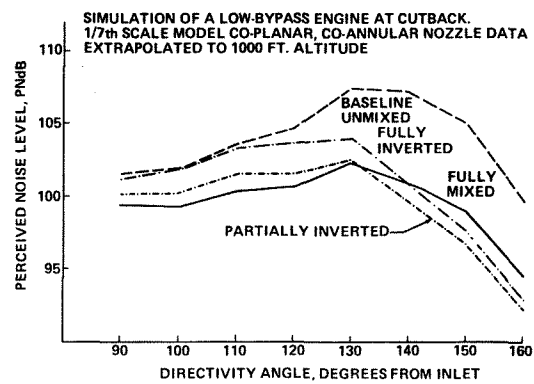Fig. 18. Jet Noise Reduction with Flow Profile Shaping

Fig. 19. PNL Directivity Patterns

Wind tunnel investigations of a full-scale JT8D-17 with internal mixer indicated a static suppression of 2.6 PNdB and a suppression in flight of 3.4 PNdB as presented in figure 20[14]. These particular data show that forward speed improves mixer nozzle suppression.
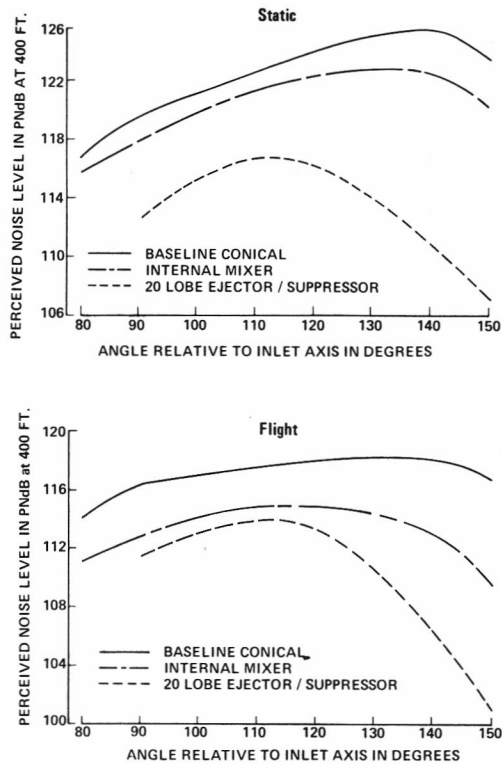
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Fig. 20. Static/Flight PNL Directivity Comparison NASA-AMES 40 by 80 ft. Wind Tunnel Test of a JT8D Engine

Mixing the primary and fan streams before discharge through the common nozzle can theoretically result in higher thrust and lower SFC. However, in actual installations, mechanical devices such as lobe mixer nozzles have losses which tend to balance the performance gains obtained through mixing the streams.

The fluid-layer shield, shown schematically in figure 21 could reduce engine internal noise and jet noise. The shield reduces noise by redirecting the noise through the use of an impedance change produced by the fluid layer. A theory has been developed to predict the practical case of the open top, semiannular fluid-layer shield. The design of practical installations, and model-scale verification are the next series of tasks in the shield development.
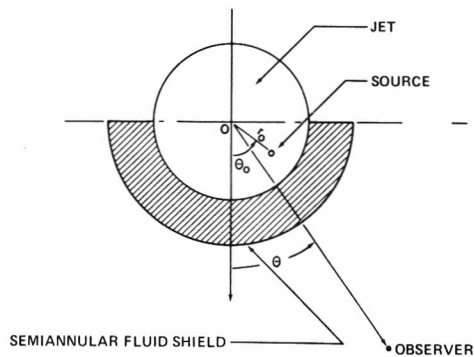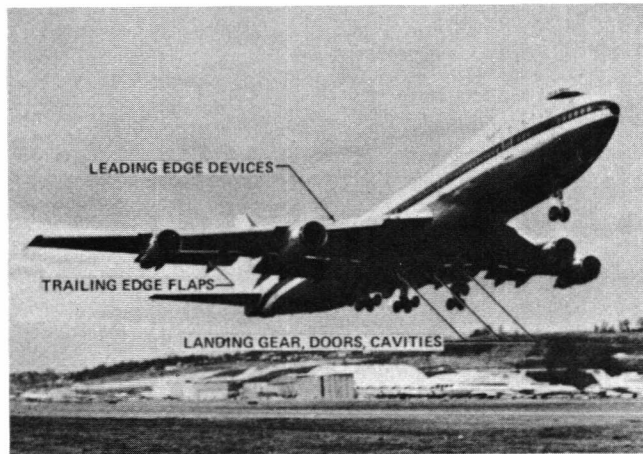

Fig. 21. Fluid Layer Shield

Airframe Noise

Background

Airframe noise is defined as the noise generated by the turbulent air flowing over the basic airframe surfaces including wings, leading and trailing edge high-lift devices, landing gear, doors and cavities (figure 22). Measurements of airframe noise with the engines either at idle or shut down have been made by various manufacturers and governmental agencies and have provided a general indication of the variation of airframe noise with major airplane configuration and performance parameters. From these measurements a number of empirical correlations have been derived. Two important findings are:

1.   Typical airframe noise levels during landing approach for the current commercial transport fleet lie about 6 to 10 EPNdB below FAR 36 limits as shown in figure 23. Thus, as propulsion system noise levels are lowered, there will be a point of diminishing returns in total airplane noise reduction as the "airframe noise floor" is approached.

2.   Several airframe components, such as leading-edge flaps, trailing-edge flaps and landing gear may be producing roughly the same level of noise. In order to achieve any meaningful noise reductions, it will be necessary to reduce all of these sources to varying degrees.

Fig. 22.   Major Airframe Noise Source Locations
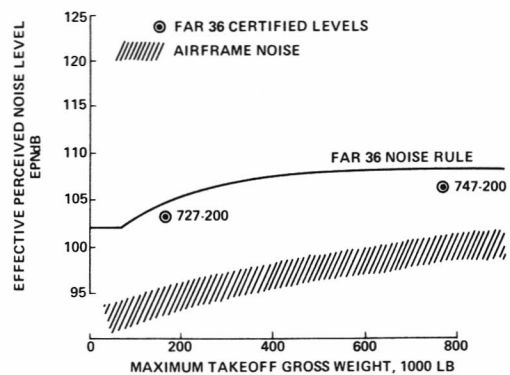

Fig. 23.   Flight Test Results, Airframe Noise (Approach, 1.0 NMI)

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Theoretical Understanding

Curle's extension of Lighthill's theory of aerodynamic sound to include the presence of solid surfaces in a turbulent flow forms the basis for understanding how an airframe generates noise [9,16]. According to Curle's theory, the fluctuating force exerted on the fluid by the surface is equivalent to an acoustic dipole distribution. Curle's theory has been verified for an airfoil of dimension smaller than a typical acoustic wavelength[17 18]. In this case the airfoil, as a generator of surface noise, may be represented by a single acoustic dipole. For airframe noise, the more important case is that of an airfoil of dimension much greater than any acoustic wavelength of interest. Powell has shown that in the case of an infinitely large plane and rigid surface, the acoustic dipole distribution does not itself generate surface noise but, rather, accounts for reflection of the turbulent-flow-produced sound waves [19]. Since the direct radiated sound from a turbulent boundary layer is low level, the application of Powell's reflection principle to airframe noise indicates that the sound generated by turbulent boundary layers on fuselage and wing surfaces will be negligible.

But airframes are not infinitely large, and edges, primarily wing/flap system trailing edges, cause an edge noise, defined by Powell as the acoustic radiation resulting from pressure differentials existing across a surface close to an edge [19]. Ffowcs-Williams and Hall showed that a turbulent flow near an edge is a much more efficient radiator of sound than if no edge were present [20]. Hence, turbulent flow over edges, as it occurs in a wing/flap system, is considered a dominant source of airframe noise.

The effects of solid boundaries on the aerodynamic sound problem have been studied by numerous investigators. A principal result of the theoretical investigations shows how the radiated far-field acoustic intensity depends on the cross-correlation. At Boeing, experiments have been conceived and carried out to measure surface pressure cross-correlations in the trailing-edge region of an airfoil. A semiempirical model of surface pressure correlation for a jet blowing over an edge provided a basis to successfully develop a prediction method, for engine exhaust jet/trailing edge interaction noise applicable to an engine-over-the-wing airplane configuration. A simple dipole model of airframe noise was used to estimate a variety of cases ranging from an airfoil in a model jet to 707 flap-generated noise. A dipole model was also used to demonstrate the suppression effect of sweep on trailing-edge noise, and experiments on sawtooth edges confirmed the suppression concept [21].

Empirical Estimation Methods

Empirical equations for airframe noise levels have been determined by Boeing and several other investigators using available flight-test data. The equations generally are of the form:

$$SPL = 10 \log \left[\frac{V^a \, S \, C_{dp}}{h^2} \right] + K$$

where V = velocity, S = reference wing area, C_{dp} = parasitic drag coefficient, and h = altitude. Typical values used for a are 5 to 6.

Design for low airframe noise will ultimately require relating the aerodynamics of lifting airfoil elements to noise generation. Development of such analytical estimation procedures is under way at Boeing.

Experimental Techniques

Boeing has performed detailed studies of facilities, instrumentation, and test techniques for application in airframe noise research. Substantial capabilities in airframe noise testing have been developed by studies of wind tunnels and small free jet facilities and development of acoustic measurement equipment and techniques. These are described in the following paragraphs.

A special microphone probe was designed and tested for the purpose of measuring airframe noise in wind tunnels. It was designed to minimize the noise caused by turbulent flow over the probe. The capability was also developed and applied for measuring airframe noise in a test where the facility noise contaminated conventional measurements.

Acoustic surveys were conducted of wind tunnels at Boeing, NASA, Naval Ship Research and Development Center, and other private installations. It was concluded that conventional wind tunnels generated self-noise levels which precluded model airframe noise testing. However, a study conducted concerning the practicality of testing large-scale landing gear models for airframe noise in conventional wind tunnels indicated that such tests were feasible. In these studies, noise spectra of the 747 gear measured in flight-tests were extrapolated to test scales and superimposed onto tunnel noise floor spectra determined from the survey mentioned previously. Results showed that a good signal to noise ratio may be expected in those tests and that relative noise levels of large-scale gear components can be measured in quieter conventional wind tunnels.

Some practical limitations to small-scale airframe noise testing in a free jet were determined in evaluations of a test conducted at Boeing in a large anechoic test chamber. The analysis is summarized in figure 24 where noise levels of three basic test elements are plotted against free jet velocity. The elements are upstream line noise and jet noise, which collectively set the noise floor, and the subject edge noise. The edge noise shown is representative of a flat plate, but the upstream noise will vary widely with different facilities. The velocity dependence of upstream noise generally ranges from V^1 to V^4, and will nearly always be less sensitive to exhaust velocity than the edge noise. The results are typical of a free jet of diameter d, test edge span d/2, chord of 3d, and distance to microphone of 10d. The analysis shows that the optimum test window between the jet noise-dominated case and the upstream noise-dominated case can be expected at or below 200 ft/sec. This is based on optimization of a broad band signal-to-noise ratio. A large free jet would generate the jet noise floor at low frequencies, with much lower noise levels in the frequency range of interest. In that case, directional microphones could discriminate between the noise coming from the model and that coming from elsewhere along the jet or the facility. Thus, it is indicated that the size of the free jet may determine the limiting signal-to-noise ratio capability.
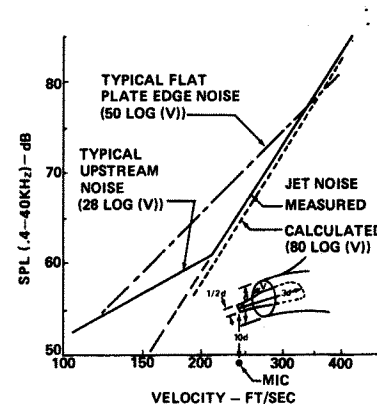

Fig. 24. Typical Noise Floor and Edge Noise Levels in Small Scale Free Jets

A program was conducted jointly with NASA-Langley in which a 0.03 scale model of the 747 was tested in the quiet flow facility of the Naval Ship Research and Development Center as shown in figure 25. This tunnel is an open-throat test facility with very low turbulence, surrounded by an anechoic chamber. A directional microphone was mounted out of the flow.

The test included scaled details of the leading-edge flaps, the main triple-slotted flaps, cavities associated with these flaps, landing gear main structure and wheels, gear cavities, and cavity doors. The finer details of the landing gear structure (cables, wires, and minor struts) were not included.
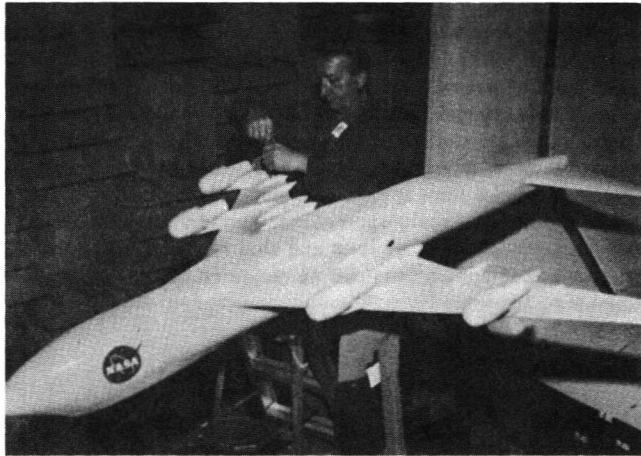
Fig. 25. 747 Test in the NASA-Langley Quiet Flow Facility

Figure 26 presents the noise source distributions in the test area of the facility without the model installed and with the model installed using different flap configurations. Levels of the 1/3-octave band centered at 16-KHz received by a parabolic dish-microphone system during an axial scan of the tunnel test area are also shown. High levels emanated from the tunnel collector which, along with other factors such as the change in these collector levels due to the influence of the model on tunnel flow, precluded accurate determination of model noise by conventional methods (i.e., with nondirectional microphones). Therefore, model noise was determined with the assistance of the directional microphone system.

Measurements of "overhead" model noise were compared to overhead flight-test spectra for ten flight configurations from Boeing flight tests of the 747. These data were normalized using the simplest possible relationships that are frequently included in aerodynamic noise formulations, i.e., $SPL \propto 10 \log (V^5 \cdot S^2 \cdot R^{-2})$, and frequency $\propto \frac{V}{S}$, where V is free stream velocity, S is scale, (S^2 is proportional to wing area), and R is distance from subject to receiver. The scaling relationship used produced excellent agreement between the model- and full-scale noise levels.

Figure 27 shows that landing gear noise was not well reproduced in the model tests. This strongly suggests that the smaller elements of the gear structure such as tubes, wires, small construction details, etc. of the full-scale airplane were responsible for the higher frequency noise components.

In-flow measurements are shown in figure 28 in comparison with far-field measurements of model airframe noise. The discrepancy in results at the high frequency end of the spectra is probably due to the sensitivity of the acoustic response of the probe to contamination of the microphone screen material. This is a handicap common to this type of measurement.
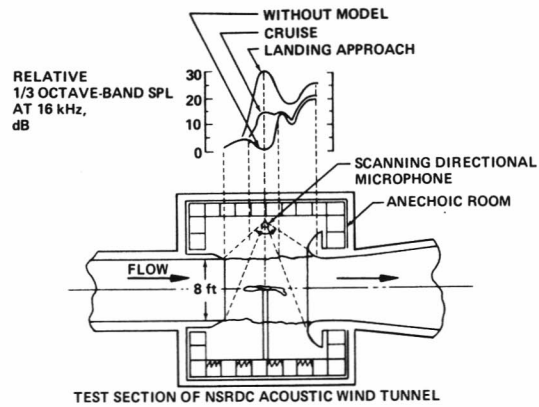
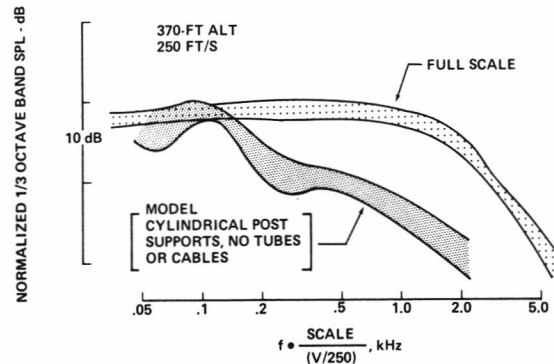Fig. 26. Spatial Resolution of Model Noise Using Directional Microphone

Fig. 27. Normalized — 0.03-Scale — Model and Full-Scale 747 Landing Gear Noise
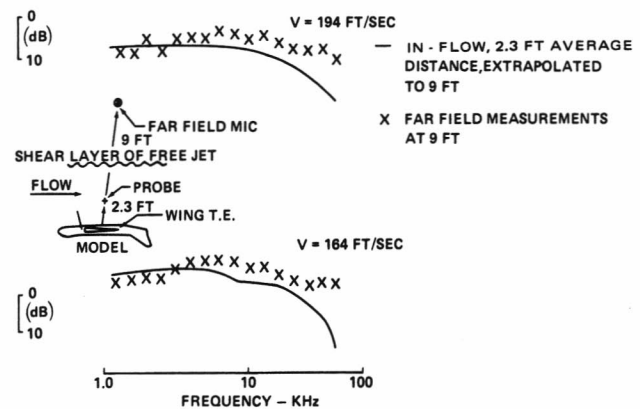
Fig. 28. Flow Measurements in Geometric Near Field

Potential Improvements

Tests were conducted at Boeing to determine the aerodynamic performance of high-lift devices with porous and sawtooth trailing edge treatments. The flap gaps were adjusted to optimize performance for each treatment. Substantial performance penalties resulted from these particular applications of edge treatments; however, there is still the possibility that edge-treated flap and wing sections with low aerodynamic performance penalties can be developed. It is apparent that the optimization is likely to require more than attaching edge treatment to an existing high-lift system.

A substantial portion, perhaps even the dominant contribution, of airframe noise results from the nonaerodynamically-faired aircraft parts extended and/or opened during landing approach. These sources may be eliminated by fairing all surfaces extended into airflow, and closing up openings to the storage cavities of flaps and gear. The total benefit to airframe noise reductions has not yet been confirmed. Unfortunately, the structural and mechanical complexities required to accomplish this will increase the cost and weight of the airplane. Definitions of benefits and costs are not yet available.

Acoustic Linings

Background

Sound absorbing liners to reduce fan, compressor and turbine noise have been incorporated into the nacelles of a number of aircraft engines during the past decade, and the ability to design more effective liners has been constantly advanced as a result of theoretical studies, engine and flow duct test programs and fabrication developments. Figure 29 illustrates the trend of improved lining attenuation effectiveness.
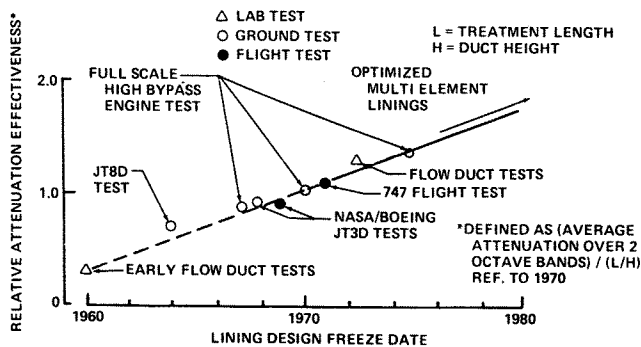

Fig. 29. Projected Improvement of Lining Performance

Theoretical Understanding

The theory of acoustic wave propagation and attenuation in ducts of constant geometry and simple cross sections is well understood. Numerical optimization methods and accompanying impedance models are available for design and analysis of liners for use in ducts of rectangular, circular and annular geometry. A thorough review of lining analysis methods used in the aircraft industry has been given by Nayfeh, et al [22].

Basically, the theory is developed by perturbing the fluid mechanic equations for conservation of mass, momentum and energy and an equation of state, subtracting the equations for the mean quantities, and neglecting nonlinear acoustic quantities. This results in a series of equations which is solved subject to given boundry conditions. The most widely used method of solution results in an eigenfunction expansion such that the sound field is visualized as being composed of a superposition of waves propagating at various angles of the duct axis. Each of these waves is called a mode and this method of analysis is termed the modal theory of sound propogation. The fundemental aspects of the modal theory have been varified expermentally. By far, the single greatest problem is the specification of the relative distribution of the sound pressure among its constitutent modes. A 'satisfactory measurement system to diagnose a large number of simultaneously propagating modes has yet to be developed.

Experimental Techniques

Experimental methods used to develop acoustically treated nacelles for aircraft engines are required to evaluate (1) the attenuation spectrum of a given liner in a selected duct; (2) the impedance of a given liner as a function of frequency and environmental conditions such as flow velocity, temperature, pressure and the imposed sound field and (3) the relative amplitudes of the modes that are simultaneously propagating at a selected frequency.

A device called a flow duct, which is a simple straight duct of either rectangular or circular geometry joining two reverberation chambers for sound field measurement,and including a test section for installation of a liner panel, is usually used to measure liner attenuation. The flow duct is usually capable of simulating an aircraft engine inlet or exhaust duct by transmitting sound against or with the mean flow .

Measurement of the acoustic impedance of a material can be done simply in a standing wave impedance tube which can be obtained commercially. However, measurement of the material impedance when a flowing environment is super-imposed is very difficult. Unfortunately it is this difficult measurement that is of importance to design of engine nacelles. One method which used waveguide principles and has met with some degree of success is shown schematically in figure 30. This facility is part of The Boeing Company acoustic facilities and is located in Wichita, Kansas [24].
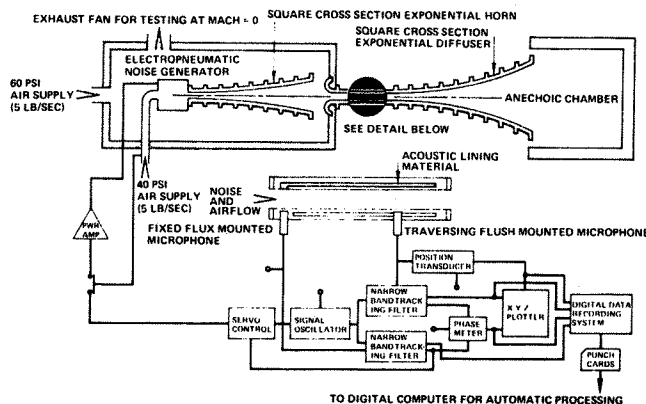

Fig. 30. Boeing Wichita Grazing Flow Impedance Test Section

Work to develop a viable method for measurement of modal distribution has been proceeding for about five years. Some simple mode fields have been investigated successfully, but a working system for use on a real engine is not yet available.

Potential Improvements

Additional significant improvements in liner attenuation performance relative to the current state-of-the-art, will be difficult to attain. The primary reason for this is that most of the essential physics is incorporated in current optimization methods. Complex liner configurations which provide a wider bandwidth of attenuation are being developed and will probably provide the next step of improvement.

In the long term, "active" liners may be developed which have impedance properties that stay near the duct optimum values for attenuation or generate an interference sound field to cause cancellation of the pure tone components of the fan-noise spectrum.

Conclusions

Further modest aircraft noise reductions will be possible in the next two decades but it will be necessary to attack each of the basic noise components simultaneously in order to obtain overall meaningful results. Much work directed towards a basic understanding of each noise-generating mechanism will be required, along with continued improvement in experimental techniques. A substantial and continued effort on the part of government, industry and the universities will be essential.

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