NASA Contractor Report 159105

NASA-CR-159105
19800004565

Near-Field Noise Prediction for Aircraft in Cruising Flight-Methods Manual

J. G. Tibbetts

LOCKHEED-GEORGIA COMPANY
Marietta, Georgia 30063

CONTRACT NAS1-14946
AUGUST 1979

NASA
National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

NASA Contractor Report 159105

Near-Field Noise Prediction for Aircraft in Cruising Flight-Methods Manual

J. G. Tibbetts

LOCKHEED-GEORGIA COMPANY
Marietta, Georgia 30063

CONTRACT NAS1-14946
AUGUST 1979

NASA

National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

FOREWORD

This document is submitted in accordance with the requirements of NASA Contract NAS1-14946, Study of the Prediction of Cruise Noise and Laminar Flow Control Noise Criteria for Subsonic Air Transports. D. L. Lansing is the NASA Langley Contract Monitor and J. S. Gibson is the Lockheed-Georgia Project Manager.

The final technical reports of this program comprise two volumes. The first volume, CR-159104, describes the technical selection and development of cruise noise prediction and LFC noise criteria procedures. The second report (CR-159105, this volume) is a Methods Manual for applying the cruise noise prediction procedures to practical aircraft design programs.

TABLE OF CONTENTS

NEAR-FIELD NOISE PREDICTION FOR AIRCRAFT IN CRUISING FLIGHT

By J. G. Tibbetts
Lockheed-Georgia Company

SUMMARY

This document is a methods manual for noise prediction at any point on an aircraft while the aircraft is in a cruise flight regime. The methods were selected and/or developed for use in Laminar Flow Control (LFC) noise effects analyses, but can be used in any case where aircraft generated noise needs to be evaluated at a location on an aircraft while under high altitude, high speed conditions. The selection and development of the noise prediction methods are described in detail in a companion report, CR-159104.

For each noise source applicable to the LFC problem, this manual contains a detailed noise computational procedure in algorithm format, suitable for computerization. Three categories of noise sources are covered: (1) Propulsion System, (2) Airframe, and (3) LFC Suction System. In addition, procedures are given for noise modifications due to source soundproofing and the shielding effects of the aircraft structure wherever needed.

Sample cases, for each of the individual noise source procedures, are provided to familiarize the user with typical input and computed data.

1. INTRODUCTION

The application of LFC criteria for the determination of aircraft noise effects on the laminar flow regions of LFC aircraft during cruise requires a method for the prediction of the incident noise levels and frequencies at any desired point on the airplane. A primary objective of the present study then has been to select from the currently available and evolving technology those noise prediction methods which appear to best meet this need. Discussions of the noise sources considered and the selection and derivation of the noise prediction methods for those sources considered to be significant for the LFC problem are contained in the companion analysis report, CR-159104. The purpose of this volume is to present the final computational procedures of the selected prediction methods of CR-159104 as a "Methods Manual" for use in LFC airplane design and analysis programs. A primary requirement in the development of these computational procedures is that they be in algorithm form so they can be directly convertible to computer programs by users without requiring the user to be knowledgeable in the aero-acoustics field.

The development of each of the computational procedures presented herein has followed a similar pattern. What is believed to be the best available prediction method for each important noise source was selected and then developed as required for application to cruise noise predictions. Not surprisingly, no methods were found in the available literature which were directly applicable to near-field noise prediction in the cruise regime. The term "near-field" when used in this volume actually refers to the "close-in" far field, where the far field generally refers to source-to-receiver distances which are large relative to a characteristic physical dimension of the source or the sound wavelength; and, as a result, the source may be considered a "point" source. In the close-in far field, the physical dimensions of the noise generating mechanism and its orientation relative to the receiver are important and should be considered. In the case of prediction methods for propulsion (or LFC suction system) noise sources, the available methods have typically been derived for the far-field case at near sea level conditions with no forward speed or for the relatively slow speeds associated with takeoff or landing. The single exception is the procedure for jet mixing noise which is a near-field (close-in far field) noise predictor. Consequently, it has been necessary to attempt to adapt these procedures for application to the close-in far field at cruise conditions. In particular, the selected prediction methods for the turbomachinery sources (fan/compressor, turbine, and core (combustion)) were derived for far-field conditions. However, it is believed that these methods will also produce reasonable and satisfactory predictions as they stand for the close-in far field since these propulsion sources are expected to behave approximately as acoustic point sources. Consequently, the basic far-field prediction procedures for these sources have been applied to the near-field LFC case with appropriate corrections as outlined later to account for the cruise environment. In the case of jet mixing noise, the prediction procedure selected is a near-field predictor applicable to a single nozzle under static conditions. It has been modified using the recommendations of the Bolt, Beranek, and Newman contracted study described in CR-159104 for application to coaxial jets with forward speed. Finally, for the propulsion noise sources, the jet shock associated broadband noise prediction procedure selected for this study was also developed as a far-field predictor. However, since this source is a distributed source, like jet mixing noise, the far-field procedure could not be applied to the close-in far field with the same level of confidence as in the case of the turbomachinery sources without some verification of the accuracy of the procedure in the near field. Consequently, a comparison was made between results obtained with this far-field prediction procedure and some of the near-field jet/shock noise data measured by General Electric and reported in AFAPL-TR-76-78, Volume IV, July 1976. This comparison indicates reasonable agreement between the measured and predicted noise levels and spectra with the prediction procedure yielding results which appear slightly conservative (higher levels) relative to the measured data. As a result, this procedure was selected for the prediction of jet shock associated broadband noise. Another alternative would be to re-develop the procedure as a near-field predictor but this is beyond the scope of the present study. For more details and discussions of the analyses and rational employed in the selection and development of these procedures refer to CR-159104.

The available airframe noise source prediction procedures were less well defined than those for the propulsion system sources. Consequently, for the sources considered significant for this study, it has been possible to develop the evolving prediction technologies into methods which are basically more applicable to the near-field cruise case than those of the propulsion system.

This is especially desirable since both of the sources considered significant for this study (trailing-edge noise and turbulent boundary layer noise) are clearly distributed sources and not amenable to point source analyses. The development of the airframe noise prediction methods is discussed in CR-159104.

No special noise prediction methods have been selected or developed for the LFC suction systems since the procedures given for the propulsion system are applicable to the more dominant noise sources of the suction system.

As mentioned above, all of the prediction procedures have had to be modified or developed for application to high altitude, high forward speed, and/or the close-in (acoustic) far field. The following effects have been accounted for in this development of the noise prediction procedures contained in this report:

- o Forward-speed propagation effects:

  (a) emitted/perceived directivity angle } forward-speed transformation
  (b) sound propagation path length } forward-speed transformation
  (c) convective amplification
  (d) dynamic amplification

- o relative velocity/forward-speed effects on noise source strength

- o high altitude acoustics (i.e., acoustic impedance and atmospheric attenuation)

- o acoustic suppression materials (e.g., turbine/core noise suppression by wall treatment in the primary exhaust duct)

- o structural shielding (i.e., line-of-sight shielding by aircraft surfaces between source and receiver location)

Other acoustic radiation effects and acoustic phenomena which are not required or may be neglected for the LFC problem are the Doppler (frequency shift) effect on the received frequency of sound, atmospheric attenuation of sound, and surface and boundary layer effects (e.g., reflection and refraction) on incident wave fields. A Doppler correction is not required for the present case since both the source and receiver move together. Consequently, the frequency measured at the receiver location is equal to the true emission frequency. Atmospheric absorption is not included in the prediction procedures of this volume since its effect is estimated to be small and neglecting it will result in slightly conservative noise estimates. The methods used to estimate the effect of atmospheric attenuation are discussed in CR-159104. Finally, surface effects are not included in the procedures since their effect is incorporated into the LFC criteria applications method of CR-159104. Consequently, the prediction procedures of this volume are for free-field conditions.

The inputs required to utilize the prediction methods of this volume are described within each computational algorithm. All of the parameters involved are further described in the nomenclature of the following section. Each computational algorithm predicts the following parameters for any specified location in the near field (close-in far field) of the noise source:

- the 1/3 octave band sound pressure levels (dB) from 50 to 10,000 Hz
- the overall sound pressure level (dB)
- spectral (1 Hz bandwidth) sound pressure levels (dB) for each 1/3 octave band

In addition, computational procedures are provided which allow the user to include the effects of designs for acoustic sound suppression (acoustic liners and high Mach number inlets), and shielding of noise by aircraft surfaces on predicted noise levels. All predicted sound pressure levels are for free-field, lossless conditions and are referenced to 0.00002 N/m^2 (0.0002 $dynes/cm^2$, $2.9008(10)^{-9}$ PSI). To compute the total sound pressure levels (1/3 octave band or spectrum levels) of two or more noise sources it will be necessary to add the individual source levels on an energy basis (i.e., add the levels logarithmically).

The methods presented in this report are believed to represent the current state-of-the-art in close-in far-field cruise noise prediction, and to provide satisfactory noise estimates for application to LFC aircraft preliminary design studies. However, at this time, the required data is simply not available to develop a reliable estimate of the accuracy of these methods. It is clear, nonetheless, that additional testing and verification is required to improve accuracy and confidence in the procedures. The recommendations section of CR-159104 indicates some of the means by which this can be accomplished.

1.1 Nomenclature

A	primary nozzle (or single nozzle) exit area
A_j	elemental surface area
A_{ref}	reference area
AR	ratio of secondary jet exit area to primary jet exit area
BPF	blade passage frequency (Hz)
c_a	ambient speed of sound
c_o	sea level, standard day speed of sound
D	blade tip diameter of last turbine stage
De	nozzle exit equivalent diameter
D_J	primary nozzle (or single nozzle) exit diameter
D_s	diameter of secondary jet nozzle exit
f	frequency (Hz)

f_b	center frequency of the 1/3 octave band containing the blade passage or peak frequency (Hz)
f_d	acoustic suppression material design frequency (Hz)
f_i	center frequency of the i^{th} 1/3 octave band (Hz)
f_{PK}	peak frequency (Hz)
f_{ref}	reference frequency for fan broadband noise (Hz)
H	plug nozzle annulus height
H_d	duct height between opposite acoustic liner surfaces
H_J	jet exit characteristic dimension for jet shock-associated noise
H_N	Hemholtz number (fD_J/c_a)
ℓ	length of trailing-edge span
L	effective length of acoustic treatment material
M_A	airplane Mach number
M_d	mean duct Mach number
M_J	jet exit fully-expanded Mach number
M_T	fan/compressor rotor tip Mach number
M_{TR}	fan/compressor rotor tip relative Mach number
$(M_{TR})_D$	fan/compressor rotor tip design relative Mach number
$\dot{m}$	mass flowrate through the engine fan or compressor
$\dot{m}_{ref}$	reference mass flowrate, 142.9 kg/s (315 lbm/s, 9.79 slug/s)
N	fan, compressor or turbine shaft rotational speed (RPM)
N_R	number of rotor blades
N_S	number of stator vanes
NS	number of inlet acoustic splitter rings
OASPL	overall sound pressure level (dB re: 0.00002 N/m^2)
PR	low-pressure turbine pressure ratio (inlet total pressure/exit static pressure)

P_{ref}	reference acoustic pressure, 0.00002 N/m^2 (0.0002 $dynes/cm^2$, 2.9008 $(10)^{-9}$ PSI)
RSS	fan/compressor rotor-stator spacing (percent)
R_W	ratio of secondary to primary jet mass flow rates
r	distance from the noise source to desired "observer" location on the aircraft
r´	actual sound propagation path length from source to observer
r_{ref}	reference source-to-observer distance
SPL_i	sound pressure level of the i^{th} 1/3 octave band (dB re: 0.00002 N/m^2)
$SSPL_i$	spectral (1 Hz bandwidth) sound pressure level for the i^{th} 1/3 octave band center frequency (dB re: 0.00002 N/m^2)
T_J	primary jet exhaust total temperature
T_{ref}	reference jet exhaust total temperature, 555.6 °K (1000 °R)
T_S	secondary jet exhaust total temperature
T_3	combustor inlet total temperature
T_4	combustor exit total temperature
$(T_4-T_5)_{des}$	total temperature drop across the entire turbine section at the engine design point
t_a	ambient static temperature
U	freestream velocity
U_{ref}	reference freestream or rotor tip velocity
V_A	aircraft velocity
V_J	jet fully-expanded exhaust velocity
V_{ref}	reference velocity, 51.44 m/s (168.8 f/s, 100 knots. 115.1 mph)
V_S	secondary jet fully-expanded exhaust velocity
W_{ref}	reference combustor airflow, 15.9 kg/s (35 lbm/s, 1.09 slug/s)
W_3	combustor total airflow
x_j, y_j, z_j	Cartesian coordinates of receiver location relative to a surface elemental area for turbulent boundary layer noise

x	receiver location measured along jet axis from the primary jet nozzle exit plane (positive aft)
x_s	receiver location measured along jet axis from the secondary jet exit plane (positive aft)
y	receiver location measured perpendicular to the jet axis
ΔT_{stage}	total temperature rise across a fan or compressor stage (actual)
ΔT_{ref}	reference fan/compressor stage or combustor total temperature rise
δ	turbulent boundary layer thickness
δ^*	turbulent boundary layer displacement thickness
λ	wavelength
ρ_a	ambient air density
ρ_o	sea level, standard day air density
ρ_3	combustor inlet density
ϕ	angle between line from source (e.g., the center of the nozzle exit plane) to receiver and engine inlet axis or direction of motion (deg)
ϕ'	source emission angle measured with respect to the inlet axis or direction of motion (deg)

2. COMPUTATIONAL ALGORITHMS FOR PROPULSION NOISE SOURCES

2.1 Fan and Compressor Noise

The method selected for the prediction of fan and compressor noise is discussed in Section 2.3.3.1 of CR-159104. This empirical method was developed by Boeing/Ames and modified and expanded by NASA Lewis. The applications and limitations of this method adapted for the LFC cruise study and the following computational algorithm are outlined below.

Applications and Limitations:

(a) The basic empirical method predicts up to five source noise components (three inlet and two fan discharge) for free-field lossless conditions in the far field. These components consist of broadband, discrete tone, and combination tone noise.

(b) The method is applicable to 1- or 2-stage fans or the 1st stage of an axial compressor. (Note: For 2-stage fans, the noise of each stage is computed separately, using the procedure of this section, then the two noise signatures must be summed on an energy basis.)

(c) The method is based on a somewhat limited range of fan types, but is considered the best available method and consistent with the fan types expected for the 1985-1990 time frame and LFC mission.

(d) Combination tone nosie is only computed for 1st-stage fans (rotor tip relative Mach numbers greater than 1.0) and is assumed to radiate only from the engine inlet.

(e) The prediction procedure will account for the presence or absence of inlet guide vanes (IGV's) for 1st-stage fans, but assumes IGV's are present in all other cases.

(f) Inlet flow distortions associated with static or ground-roll operation are assumed to be absent.

(g) The method assumes aerodynamically clean, relatively short-duct nacelles with hard walls (no acoustic treatment). To include the effects of acoustically treated nacelles use the method of Section 4.1.

(h) The rotor-tip relative Mach numbers must be less than or equal to the design value ($M_{TR} \leq \{M_{TR}\}_D$).

(i) The radiated sound field is assumed to be axisymmetric with respect to the engine centerline.

(j) Corrections for rotor-stator spacing are applied.

(k) A tone cut-off correction is employed when computed to exist, but is applied only to the fundamental tone.

The following input is required:

ΔT_{stage} = total temperature rise across a fan or compressor stage (actual)

$\dot{m}$ = mass flow rate through the fan or compressor

N = fan or compressor shaft rotational speed (RPM)

M_{T_R} = rotor tip relative Mach number

$(M_{T_R})_D$ = rotor tip design relative Mach number

M_T = rotor tip Mach number

N_R = number of rotor blades

N_S = number of stator vanes

RSS = rotor-stator spacing (percent) (see figure 2.1.1)

ϕ_I = observer angle from the engine inlet measured at the inlet* (deg)

r_I = observer distance from the engine inlet*

ϕ_D = observer angle from the engine inlet measured at the fan discharge duct* (deg)

r_D = observer distance from the fan discharge duct*

M_A = aircraft Mach number

ρ_a = ambient density

c_a = ambient speed of sound

ρ_o = sea level, standard day density

c_o = sea level, standard day speed of sound

ΔT_{ref} = reference stage temperature rise, 44.4 °K or °C (80 °F or °R)

$\dot{m}_{ref}$ = reference mass flow rate, 142.9 kg/s (315 lbm/s, 9.79 slug/s)

r_{ref} = reference source to observer distance, 1 m (3.28 f, 39.37 in)

---

* *See figure 2.1.2*

A. <u>Compute parameters which are common to both fan inlet and discharge duct noise prediction calculations.</u>

1. Compute the correction required to "de-normalize" the predicted peak 1/3 O.B. sound pressure levels:

$$C_N = 20 \log_{10} (\Delta T_{stage}/\Delta T_{ref}) + 10 \log_{10} (\dot{m}/\dot{m}_{ref}) + 63.0 \quad \text{dB}$$

2. Compute rotor/stator spacing correction to be applied to broadband noise:

$$C_{RSS,BB} = -5 \log_{10} (RSS/300) \quad \text{dB}$$

3. Compute rotor/stator spacing correction to be applied to discrete tone noise:

$$C_{RSS,DT} = -10 \log_{10} (RSS/300) \quad \text{dB}$$

4. Compute tone cut-off factor for discrete tone noise (if $\delta < 1.05$, a cut-off correction will be applied to the fundamental tone):

$$\delta = \left| \frac{M_T}{1 - \frac{N_S}{N_R}} \right|$$

5. Compute the blade passage frequency:

$$BPF = N \cdot N_R/60 \quad \text{Hz}$$

6. Compute the correction to be applied for local acoustic impedance:

$$C_{IMPD} = 10 \log_{10} \left(\frac{\rho_a c_a}{\rho_o c_o} \right) \quad \text{dB}$$

B. <u>Compute fan or compressor inlet radiated noise.</u>

1. Compute source emission angle (ϕ_I') and actual sound propagation path length (r_I') from input values of observer angle (ϕ_I) and distance (r_I) W.R.T. the engine inlet (forward speed effect):

(a) $M_A = 0.0$

$$\phi_I' = \phi_I \quad \text{and} \quad r_I' = r_I$$

(b) $0.0 < M_A < 1.0$

- If $\phi_I = 0°$, $\phi_I' = 0°$ and $r_I' = r_I/(1-M_A)$
- If $\phi_I = 180°$, $\phi_I' = 180°$ and $r_I' = r_I/(1+M_A)$
- If $0° < \phi_I < 180°$,

$$\cot\phi_I' = \frac{1}{1-M_A^2}\left[\cot\phi_I + M_A\sqrt{1-M_A^2 + \cot^2\phi_I}\right]$$

$$r_I' = r_I \sin\phi_I/\sin\phi_I'$$

2. Compute the correction for convective amplification:

$$C_{A_I} = -40 \log_{10}(1-M_A \cos\phi_I') \quad \text{dB}$$

3. Broadband noise

(a) Calculate the peak normalized 1/3 O.B. SPL:

- For $M_{T_R} \leq 0.9$

  If $(M_{T_R})_D \leq 1.0$, $\bar{L}_{BB} = 58.5$ dB

  If $(M_{T_R})_D > 1.0$, $\bar{L}_{BB} = 58.5 + 20 \log_{10} (M_{T_R})_D$ dB

- For $M_{T_R} > 0.9$

  If $(M_{T_R})_D < 1.0$, $\bar{L}_{BB} = 57.6 - 20 \log_{10} (M_{T_R})$ dB

  If $(M_{T_R})_D \geq 1.0$, $\bar{L}_{BB} = 57.6 + 20 \log_{10} (M_{T_R})_D - 20 \log_{10} (M_{T_R})$ dB

  where $M_{T_R} \leq (M_{T_R})_D$

(b) "De-normalize" the peak level for the specific fan or compressor operating condition:

$$L_{BB,PK} = \bar{L}_{BB} + C_N \quad \text{dB}$$

(c) Correct the peak level for local acoustic impedance, rotor/stator spacing, and convective amplification:

$$L_{BB,PK} = L_{BB,PK} + C_{IMPD} + C_{RSS,BB} + C_{A_I} \quad \text{dB}$$

(d) Find the directivity correction for the inlet broadband component from table 2.1.1 (linear interpolation).

$$\Delta dB_{I,BB} = F^N(\phi_I') \quad \text{dB}$$

(e) Apply corrections to the peak 1/3 O.B. level for directivity and distance:

$$L_{BB} = L_{BB,PK} + \Delta dB_{I,BB} - 20 \log_{10} (r_I'/r_{ref}) \quad \text{dB}$$

(f) Compute the broadband 1/3 O.B. spectrum levels using the following equation:

$$(SPL_{i,BB} - L_{BB}) = 10 \log_{10} e^{-4.264[\log_{10}(f_i/f_{ref})]^2}$$

where $SPL_{i,BB} = i^{TH}$ 1/3 O.B. SPL (dB)

$f_i = i^{TH}$ 1/3 O.B. center frequency (Hz)

f_{ref} = center frequency of the 1/3 O.B. containing the frequency equal to 2.5 times the center freq. of the 1/3 O.B. containing the BPF.

4. Discrete tone noise

(a) Calculate the peak normalized 1/3 O.B. SPL of the fundamental tone:

- For $M_{TR} \leq 0.72$

If $(M_{TR})_D \leq 1.0$, $\bar{L}_T = 60.5$ dB

If $(M_{TR})_D > 1.0$, $\bar{L}_T = 60.5 + 20 \log_{10} (M_{TR})_D$ dB

- For $M_{TR} > 0.72$

If $(M_{TR})_D < 1.0$,

$$\text{USE LESSER OF} \begin{cases} \bar{L}_T = 60.5 + 50 \log_{10} (M_{TR}/0.72) \quad \text{dB} \\ \bar{L}_T = 59.5 - 80 \log_{10} (M_{TR}) \quad \text{dB} \end{cases}$$

If $(M_{TR})_D \geq 1.0$,

$$\text{USE LESSER OF} \begin{cases} \bar{L}_T = 60.5 + 20 \log_{10} (M_{TR})_D + 50 \log_{10} (M_{TR}/0.72) \quad \text{dB} \\ \bar{L}_T = 59.5 + 80 \log_{10} \left[\frac{(M_{TR})_D}{M_{TR}}\right] \quad \text{dB} \end{cases}$$

where $M_{TR} \leq (M_{TR})_D$

(b) "De-normalize" the peak tone level for the specific operating condition:

$$L_{T,PK} = \bar{L}_T + C_N \quad \text{dB}$$

(c) Correct the peak level for local acoustic impedance, rotor/stator spacing, and convective amplification:

$$L_{T,PK} = L_{T,PK} + C_{IMPD} + C_{RSS,DT} + C_{A_I} \quad \text{dB}$$

(d) Find the directivity correction for the inlet discrete tones from table 2.1.1 (linear interpolation):

$$\Delta dB_{I,DT} = F^N(\phi_I') \quad dB$$

(e) Apply corrections to the peak tone for directivity and distance:

$$L_T = L_{T,PK} + \Delta dB_{I,DT} - 20 \log_{10} (r_I'/r_{ref}) \quad dB$$

(f) Compute the 1/3 O.B. levels of the harmonic tones:

(1) 1^{st} stage fans w/o inlet guide vanes

$$L_k = L_T + 3 - 3k \quad dB \quad k \geq 2$$

$$\text{If } \delta \geq 1.05,\ L_1 = L_T \quad dB$$

$$\text{If } \delta < 1.05,\ L_1 = L_T - 8.0 \quad dB$$

(2) All other cases

$$L_k = L_T - 3 - 3k \quad dB \quad k \geq 2$$

$$\text{If } \delta \geq 1.05,\ L_1 = L_T \quad dB$$

$$\text{If } \delta < 1.05,\ L_1 = L_T - 8.0 \quad dB$$

where k = 1,2,3,..integer multiples of the center frequency of the 1/3 O.B. containing the BPF. The level is assigned to the 1/3 O.B. containing the harmonic frequency.

5. Combination tone noise (CTN)

Note: Applies only to 1^{st}-stage fans where $M_{TR} > 1.0$

(a) Compute the normalized peak level for each of the three components of this inlet source:

(1) $f/f_b = 1/2$

$$\text{If } M_{TR} \leq 1.146,\ \bar{L}_{c,1/2} = 785.68 \log_{10} (M_{TR}) + 30.0 \quad dB$$

$$\text{If } M_{TR} > 1.146,\ \bar{L}_{c,1/2} = -49.62 \log_{10} (M_{TR}) + 79.44 \quad dB$$

(2) $f/f_b = 1/4$

$$\text{If } M_{TR} \leq 1.322,\ \bar{L}_{c,1/4} = 391.81 \log_{10} (M_{TR}) + 30.0 \quad dB$$

$$\text{If } M_{TR} > 1.322,\ \bar{L}_{c,1/4} = -50.06 \log_{10} (M_{TR}) + 83.57 \quad dB$$

(3) $f/f_b = 1/8$

If $M_{T_R} \leq 1.610$, $\bar{L}_{c,1/8} = 199.20 \log_{10} (M_{T_R}) + 30.0$ dB

If $M_{T_R} > 1.610$, $\bar{L}_{c,1/8} = -49.89 \log_{10} (M_{T_R}) + 81.52$ dB

where f = peak frequency of the combination tone noise component (1/3 O.B. center frequency).

f_b = center frequency of the 1/3 O.B. containing the BPF.

(b) "De-normalize" the peak levels of the three components of CTN:

$$L_{c,n} = \bar{L}_{c,n} + C_N \quad \text{dB} \quad n = 1/2, 1/4, 1/8$$

(c) Apply correction for inlet guide vanes, if present:

$$L_{c,n} = L_{c,n} - 5.0 \quad \text{dB} \quad n = 1/2, 1/4, 1/8$$

(d) Correct peak levels for local acoustic impedance and convective amplification.:

$$L_{c,n} = L_{c,n} + C_{IMPD} + C_{A_I} \quad \text{dB} \quad n = 1/2, 1/4, 1/8$$

(e) Find the directivity correction for CTN from table 2.1.1 (linear interpolation),

$$\Delta dB_{CTN} = F^N (\phi_I') \quad \text{dB}$$

(f) Correct the CTN components for directivity and distance:

$$L_{c,n} = L_{c,n} + \Delta dB_{CTN} - 20 \log_{10} (r_I'/r_{ref}) \quad \text{dB} \quad n = 1/2, 1/4, 1/8$$

(g) Compute the 1/3 O.B. spectrum levels for each of the 3 components of CTN using the following equations:

(1) $f/f_b = 1/2$

If $f_i/f_b \leq 0.5$, $SPL_{i,1/2} = L_{c,1/2} + 30 \log_{10} (f_i/f_b) + 9.03$

If $f_i/f_b > 0.5$, $SPL_{i,1/2} = L_{c,1/2} - 30 \log_{10} (f_i/f_b) - 9.03$

(2) $f/f_b = 1/4$

If $f_i/f_b \leq 0.25$, $SPL_{i,1/4} = L_{c,1/4} + 50 \log_{10} (f_i/f_b) + 30.1$

If $f_i/f_b > 0.25$, $SPL_{i,1/4} = L_{c,1/4} - 50 \log_{10} (f_i/f_b) - 30.1$

(3) $f/f_b = 1/8$

If $f_i/f_b \leq 0.125$, $SPL_{i,1/8} = L_{c,1/8} + 50 \log_{10} (f_i/f_b) + 45.15$

If $f_i/f_b > 0.125$, $SPL_{i,1/8} = L_{c,1/8} - 30 \log_{10} (f_i/f_b) - 27.09$

where, f_i = the i^{th} 1/3 O.B. center frequency

6. Compute the 1/3 O.B. SPL's of the total inlet radiated noise at the observer location.

   (a) $$SPL_{I,i} = 10 \log_{10} \left[10^{(SPL_{i,BB}/10)} + 10^{(SPL_{i,1/2}/10)} + 10^{(SPL_{i,1/4}/10)} + 10^{(SPL_{i,1/8}/10)}\right]$$

   where $SPL_{I,i}$ = the i^{th} 1/3 O.B. SPL of the total inlet radiated noise (dB)

   $SPL_{i,BB}$ from Step B-3(f)

   $SPL_{i,n}$ from Step B-5(g) for n = 1/2,1/4,1/8

   (b) If a harmonic of the fundamental tone occurs in the ith 1/3 O.B. correct $SPL_{I,i}$ computed above as follows:

   $$SPL_{I,i} = 10 \log_{10} \left[10^{(SPL/10)} + 10^{(L_k/10)}\right] \quad dB$$

   where SPL = $SPL_{I,i}$ from (a) above; L_k is from Step B-4(f) and k = 1,2,3....

7. Correct the inlet radiated noise spectrum for acoustic suppression if treatment is applied to the inlet duct. This correction is treated as a separate prediction module and is described in Section 4.1.

8. If appropriate, correct inlet radiated noise for structural shielding. This correction is also treated as a separate module. See Section 4.2. The source location is taken as the center of the inlet plane (see figure 2.1.2).

9. Compute the overall sound pressure level for the inlet radiated noise at the observer location for the 24 1/3 octave bands from 50 to 10,000 Hz.

   $$OASPL_I = 10 \log_{10} \sum_{i=1}^{24} 10^{(SPL_{I,i}/10)} \quad dB$$

10. Compute the spectral levels (1 Hz bandwidth) of the inlet radiated noise at the desired observer location.

    (a) $$SSPL_{I,i} = \left[SPL_{I,i}\right]_{\text{from step B-6(a)}} - 10 \log_{10} (\Delta f)_i \quad dB$$

    where $SSPL_{I,i}$ = the spectral sound pressure level (SPL) at the observer location for the i^{th} 1/3 octave band center frequency.

$(\Delta f)_i$ = the bandwidth of the i^{th} 1/3 octave band (Hz).

(b) If a harmonic of the fundamental tone occurs in the i^{th} 1/3 octave band, correct $SSPL_{I,i}$ of (a) above as follows:

$$SSPL_{I,i} = 10 \log_{10} \left[10^{(SPL/10)} + 10^{(L_k/10)} \right] \text{ dB}$$

where SPL = $SSPL_{I,i}$ from 10(a) above; L_k is from STEP B-4(f) and $k = 1,2,3,\ldots\ldots$

(c) Apply the corrections of Steps B-7 and B-8, if applicable, to the computed spectral levels of (a) and (b) above.

C. <u>Noise radiated from the fan discharge duct.</u>

1. Compute the source emission angle ($\phi_{\acute{D}}$) and the actual sound propagation path length ($r_{\acute{D}}$) from input values of observer angle (ϕ_D) and distance (r_D) W.R.T. the fan discharge duct exit (forward speed effect):

(a) $M_A = 0.0$

$$\phi_{\acute{D}} = \phi_D \quad \text{and} \quad r_{\acute{D}} = r_D$$

(b) $0.0 < M_A < 1.0$

- If $\phi_D = 0°$, $\phi_{\acute{D}} = 0°$ and $r_{\acute{D}} = r_D/(1 - M_A)$
- If $\phi_D = 180°$, $\phi_{\acute{D}} = 180°$ and $r_{\acute{D}} = r_D/(1 + M_A)$
- If $0° < \phi_D < 180°$,

$$\cot\phi_{\acute{D}} = \frac{1}{1-M_A^2} \left[\cot\phi_D + M_A \sqrt{1 - M_A^2 + \cot^2\phi_D} \right]$$

$$r_{\acute{D}} = r_D \sin\phi_D / \sin\phi_{\acute{D}}$$

2. Compute the correction for convective amplification

$$C_{A_D} = -40 \log_{10} (1 - M_A \cos\phi_{\acute{D}}) \text{ dB}$$

3. Broadband noise

(a) Compute the peak normalized 1/3 O.B. level:

- For $M_{T_R} \leq 1.0$

If $(M_{T_R})_D \leq 1.0$, $\bar{S}_{BB} = 60.0$ dB

If $(M_{T_R})_D > 1.0$, $\bar{S}_{BB} = 60.0 + 20 \log_{10} (M_{T_R})_D$ dB

- For $M_{T_R} > 1.0$, $\bar{S}_{BB} = 60.0 + 20 \log_{10} (M_{T_R})_D$

$$- 20 \log_{10} (M_{T_R}) \quad \text{dB}$$

where, $M_{T_R} \leq (M_{T_R})_D$

(b) "De-normalize" the peak level for the specific fan operating conditions:

$$S_{BB,PK} = \bar{S}_{BB} + C_N \quad \text{dB}$$

(c) Correct the peak level for local acoustic impedance, rotor/stator spacing, and convective amplification:

$$S_{BB,PK} = S_{BB,PK} + C_{IMPD} + C_{RSS,BB} + C_{A_D} \quad \text{dB}$$

(d) Apply correction to peak level for inlet guide vanes, if present:

$$S_{BB,PK} = S_{BB,PK} + 3.0 \quad \text{dB}$$

(e) Find the directivity correction for the fan discharge duct broadband component in table 2.1.1 (linear interpolation):

$$\Delta dB_{D,BB} = F^N (\phi_D') \quad \text{dB}$$

(f) Correct the peak level for directivity and distance:

$$S_{BB} = S_{BB,PK} + \Delta dB_{D,BB} - 20 \log_{10} (r_D'/r_{ref}) \quad \text{dB}$$

(g) Compute the broadband 1/3 O.B. spectrum levels using the following equation:

$$(SPL_{i,BB} - S_{BB}) = 10 \log_{10} e^{-4.264\left[\log_{10} (f_i/f_{ref})\right]^2}$$

where $SPL_{i,BB}$ = i^{th} 1/3 O.B. SPL (dB)

f_i = i^{th} 1/3 O.B. center frequency

f_{ref} = center frequency of the 1/3 O.B. containing the frequency equal to 2.5 times the center frequency of the 1/3 O.B. containing the BPF.

4. Discrete tone noise

(a) Compute the peak normalized 1/3 O.B. SPL of the fundamental tone:

- For $M_{T_R} \leq 1.0$

If $(M_{T_R})_D \leq 1.0$, $\bar{S}_T = 63.0$ dB

If $(M_{T_R})_D > 1.0$, $\bar{S}_T = 63.0 + 20 \log_{10} (M_{T_R})_D$ dB

- For $M_{T_R} > 1.0$, $\bar{S}_T = 63.0 + 20 \log_{10} (M_{T_R})_D - 20 \log_{10} (M_{T_R})$ dB

  where $M_{T_R} \leq (M_{T_R})_D$

(b) "De-normalize" the peak tone level for the specific fan operating condition:

$$S_{T,PK} = \bar{S}_T + C_N \quad \text{dB}$$

(c) Correct the peak level for local acoustic impedance, rotor/stator spacing, and convective amplification:

$$S_{T,PK} = S_{T,PK} + C_{IMPD} + C_{RSS,DT} + C_{A_D} \quad \text{dB}$$

(d) Correct the peak level for inlet guide vanes, if present:

$$S_{T,PK} = S_{T,PK} + 6.0 \quad \text{dB}$$

(e) Find the directivity correction for the fan discharge duct discrete tones from table 2.1.1 (linear interpolation):

$$\Delta dB_{D,DT} = F^N (\phi_{\acute{D}}) \quad \text{dB}$$

(f) Apply corrections to the peak tone for directivity and distance:

$$S_T = S_{T,PK} + \Delta dB_{D,DT} - 20 \log_{10} (r_{\acute{D}}/r_{ref}) \quad \text{dB}$$

(g) Compute the 1/3 O.B. levels of the harmonic tones:

(1) 1^{st}-stage fans w/o inlet guide vanes

$S_k = S_T + 3 - 3k$ dB $\quad k \geq 2$

If $\delta \geq 1.05$, $S_1 = S_T$ dB

If $\delta < 1.05$, $S_1 = S_T - 8.0$ dB

(2) All other cases

$S_k = S_T - 3 - 3k$ dB $\quad k \geq 2$

If $\delta \geq 1.05$, $S_1 = S_T$ dB

If $\delta < 1.05$, $S_1 = S_T - 8.0$ dB

where $k = 1,2,3,\ldots$ integer multiples of the center frequency of the 1/3 O.B. containing the BPF. The level is assigned to the 1/3 O.B. containing the harmonic frequency.

5. Compute the 1/3 O.B. SPL's of the total noise radiated from the fan discharge duct

$$SPL_{D,i} = SPL_{i,BB} \quad \text{dB}$$

If a harmonic of the fundamental tone occurs in the i^{th} 1/3 O.B., compute the total SPL as follows:

$$SPL_{D,i} = 10 \log_{10} \left[10^{(SPL_{i,BB}/10)} + 10^{(S_k/10)}\right] \quad dB$$

where, $SPL_{D,i}$ = the i^{th} 1/3 O.B. SPL of the total fan discharge noise

$SPL_{i,BB}$ from Step C-3(g)

S_k from Step C-4(g)

and $k = 1,2,3....$

6. Correct the aft-radiated fan noise spectrum for acoustic suppression if treatment is applied to the fan discharge duct. This correction is treated as a separate prediction module and is described in Section 4.1.

7. If appropriate, correct the aft-radiated fan noise for structural shielding. This correction is also treated as a separate module. See Section 4.2. The source location is taken to be the center of the exhaust duct exit plane. (See figure 2.1.2)

8. Compute the overall sound pressure level for the aft-radiated fan noise at the observer location for the 24 1/3 octave bands from 50 to 10,000 Hz.

$$OASPL_D = 10 \log_{10} \sum_{i=1}^{24} 10^{(SPL_{D,i}/10)} \quad dB$$

9. Compute the spectral levels (1 Hz bandwidth) of the aft-radiated fan noise at the desired observer location.

   (a) $SSPL_{D,i} = \left[SPL_{i,BB}\right]_{\text{From Step C-3(g)}} - 10 \log_{10} (\Delta f)_i \quad dB$

   where, $SSPL_{D,i}$ = the spectral sound pressure level (SPL) at the observer location for the i^{th} 1/3 octave band center frequency (dB).

   $(\Delta f)_i$ = the bandwidth of the ith 1/3 octave band (Hz).

   (b) If a harmonic of the fundamental tone occurs in the i^{th} 1/3 octave band, correct $SSPL_{D,i}$ of (a) above as follows:

   $$SSPL_{D,i} = 10 \log_{10} \left[10^{(SPL/10)} + 10^{(S_k/10)}\right] \quad dB$$

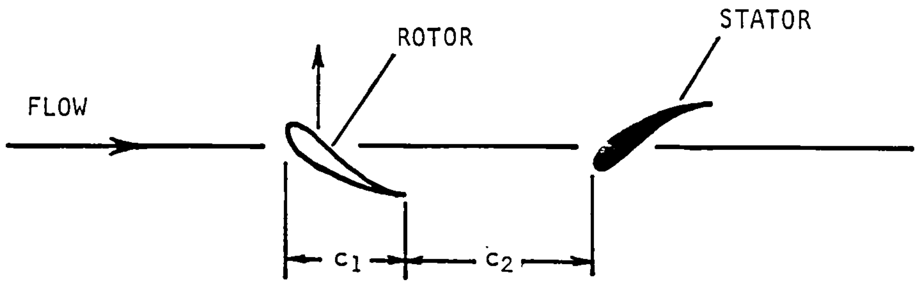
   where SPL = $SSPL_{D,i}$ from 9(a) above S_k is from Step C-4(g) and

$k = 1,2,3.....$

(c) Apply the corrections of Steps C-6 and C-7, if applicable, to the computed spectral levels of (a) and (b) above.

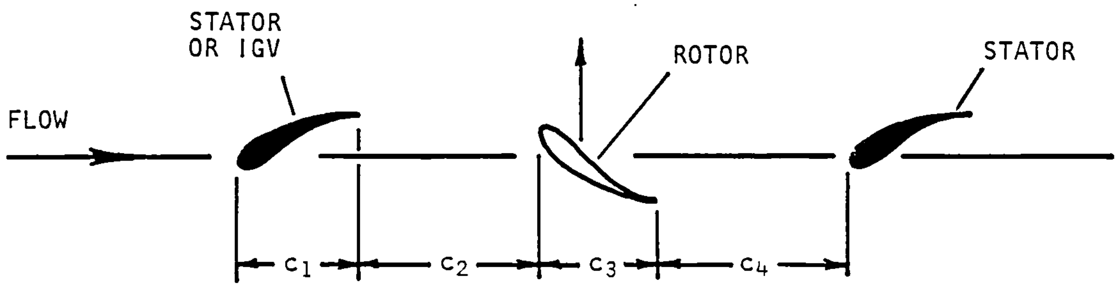
Table 2.1.1 Fan Noise Directivity

ΔdB CORRECTIONS					
Angle From Inlet (Deg) ϕ'	Broadband		Discrete Tones		CTN
	Inlet	Discharge	Inlet	Discharge	
0	-2.2	-41.7	-2.9	-38.8	-9.5
10	-1.0	-37.4	-1.5	-34.8	-8.5
20	0	-33.1	0	-30.8	-7.0
30	0	-28.8	0	-26.8	-5.0
40	0	-24.3	0	-22.8	-2.0
50	-2.0	-20.1	-1.2	-18.9	0
60	-4.5	-15.8	-3.5	-15.0	0
70	-7.5	-11.5	-6.8	-11.0	-3.5
80	-11.0	-8.0	-10.5	-8.0	-7.5
90	-15.0	-5.0	-14.5	-5.0	-9.0
100	-19.5	-2.7	-19.0	-3.0	-9.5
110	-25.0	-1.2	-23.3	-1.0	-10.0
120	-30.6	-0.3	-27.8	0	-10.5
130	-36.3	0	-32.4	0	-11.0
140	-42.1	-2.0	-36.9	-2.0	-11.5
150	-47.6	-6.0	-41.5	-5.5	-12.0
160	-53.3	-10.0	-46.0	-9.0	-12.5
170	-58.8	-15.0	-50.4	-13.0	-13.0
180	-64.6	-20.0	-55.0	-18.0	-13.5



RSS = $(c_2/c_1) \times 100$

(a) GEOMETRY FOR 1^{ST}-STAGE FANS WITHOUT INLET GUIDE VANES.



RSS = MINIMUM OF: $(c_2/c_1) \times 100$ OR $(c_4/c_3) \times 100$

(b) GEOMETRY FOR 1^{ST}-STAGE FANS WITH INLET GUIDE VANES (IGV), 2^{ND}-STAGE FANS, OR 1^{ST}-STAGE OF COMPRESSORS

Figure 2.1.1 Fan or compressor geometry.

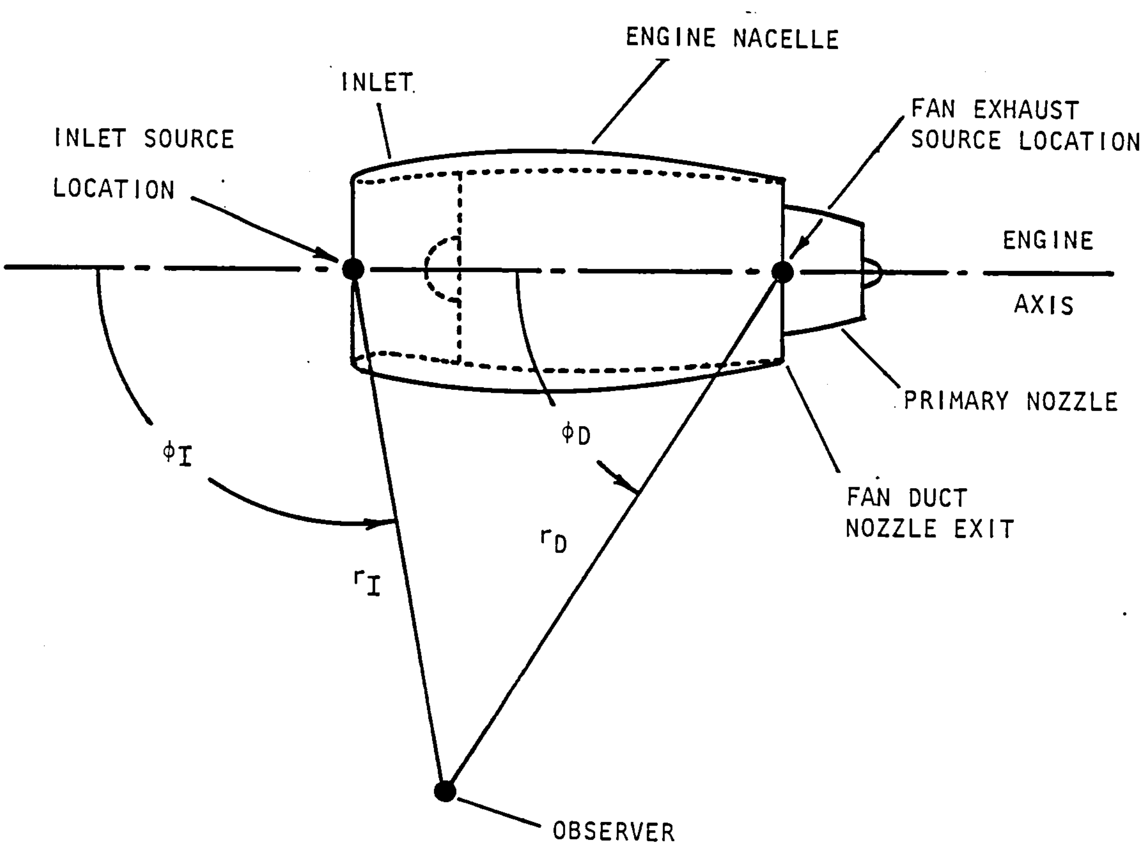


Figure 2.1.2 Description of input for observer location.

2.2 Turbine Noise

Section 2.3.3.2 of CR-159104 contains a discussion of the selection of the turbine noise prediction procedure. The selected procedure adapted here for LFC studies was developed by General Electric under contract to the FAA. The applications and/or limitations of the empirical method are tabulated below.

Applications and Limitations:

(a) The basic prediction procedure selected is a semi-empirical methodology verified for static, sea level conditions to predict turbine noise in the far field. For free-field estimates, the procedure is expected to run about 1.5 dB on the high side as a result of the test data used in developing the method.

(b) The turbine noise is predicted to consist of two components (1) a broadband spectrum, and (2) a discrete tone occurring at the blade passage frequency of the last turbine stage.

(c) The last turbine stage is assumed to be the dominant stage.

(d) The method is considered to be applicable only to engines whose low-pressure turbine consists of three or more stages (this is expected to be the case for engines of interest in the 1985 to 1990 time frame).

(e) The equation presented here for the overall sound pressure level is the equation of the reference method corrected to lossless conditions to simplify the procedure. This was done assuming a turbine blade passage frequency of near 10,000 Hz. This will result in slightly conservative (higher) OASPL's at lower frequencies.

(f) The "hay stacking" effect or the redistribution of turbine radiated discrete tones over adjacent frequencies by interaction with the turbulent jet flow is not included in the present procedure.

TURBINE NOISE COMPUTATIONAL ALGORITHM

The following engine and flight parameters are required:

PR = low-pressure turbine pressure ratio, (inlet total pressure/exit static pressure)$_{ideal}$

N = low-pressure turbine rotational speed (RPM)

D = blade tip diameter of last turbine stage

A = primary nozzle exit area

N_R = number of blades in last turbine stage

M_A = aircraft Mach number

ρ_a = ambient density

c_a = ambient speed of sound

ρ_o = sea level, standard day density

c_o = sea level, standard day speed of sound

r = observer distance from center of primary nozzle exit plane

ϕ = angle between engine inlet axis and line from observer to center of the primary nozzle exit plane (deg.)

U_{ref} = reference tip speed, 340.3 m/s (1116.4 f/s)

A_{ref} = reference nozzle area, 0.0929 m^2(1.0 f^2, 144 in^2)

r_{ref} = reference distance, 70.4 m (230.9 f)

Step 1 Compute the turbine tip speed, U_T

$$U_T = (D/2)\ (\frac{2\pi N}{60})$$

Step 2 Compute the total (lossless) OASPL (broadband + discrete) at 120° from the engine inlet on a 61 m sideline.

$$OASPL_{PK} = 40\ \log_{10}\left[1 - (\frac{1}{PR})^{0.286}\right] -20\ \log_{10}\ (U_T/U_{ref})$$

$$+\ 10\ \log_{10}(A/A_{ref}) + 109.\ dB$$

Step 3 Correct the OASPL of Step 2 for local impedance.

$$OASPL_{PK} = OASPL_{PK} + 10\ \log_{10}\left[\frac{\rho_a c_a}{\rho_o c_o}\right]\ dB$$

Step 4 Compute the discrete frequency 1/3 O.B. SPL (lossless) at 120° from the engine inlet on a 61 m sideline.

$$SPL_{PK} = OASPL_{PK} - 5.0\ dB$$

Step 5 Find the source emission angle, ϕ', and actual sound path length, r', corresponding to the desired observer location given by r and ϕ.

($M_A < 1.0$). (Forward Speed Effect)

(a) $M_A = 0.0$

$$\phi' = \phi \text{ and } r' = r$$

(b) $0.0 < M_A < 1.0$

- If $\phi = 0°$, $\phi' = 0°$ and $r' = r/(1-M_A)$
- If $\phi = 180°$, $\phi' = 180°$ and $r' = r/(1+M_A)$
- If $0° < \phi < 180°$,

$$\phi' = \cot^{-1}\left[\frac{1}{1-M_A^2}\{\cot\phi + M_A\sqrt{1-M_A^2+\cot^2\phi}\}\right] \text{ deg.}$$

$$r' = r \sin\phi/\sin\phi'$$

Step 6 Correct the total $OASPL_{PK}$ of Step 3 for directivity angle (ϕ') (using Table 2.2.1 with linear interpolation) and distance (r'):

$$OASPL = OASPL_{PK} + (OASPL - OASPL_{PK})_{Table\ 2.2.1} - 20\log_{10}(r'/r_{ref})\ dB$$

Step 7 Correct the tone SPL_{PK} of Step 4 for the directivity angle (ϕ') (using Table 2.2.2 with linear interpolation) and distance (r'):

$$SPL_D = SPL_{PK} + (SPL - SPL_{PK})_{TABLE\ 2.2.2} - 20\log_{10}(r'/r_{ref})\ dB$$

Step 8 For $M_A > 0.0$, apply convective amplification correction to the total OASPL and tone SPL of Steps 6 and 7, respectively.

$$OASPL = OASPL - 40\log_{10}(1-M_A\cos\phi')\ dB$$

$$SPL_D = SPL_D - 40\log_{10}(1-M_A\cos\phi')\ dB$$

Step 9 Compute the broadband OASPL (lossless) at the desired location by subtracting, logarithmically, the tone SPL from the total OASPL.

$$OASPL_{BB} = 10\log_{10}\left[10^{(OASPL/10)} - 10^{(SPL_D/10)}\right]\ dB$$

Step 10 Compute the blade passage frequency.

$$BPF = N_R(N)/60\ \ Hz$$

Step 11 Find the 1/3 O.B. containing the blade passage frequency.

f_b = 1/3 O.B. center frequency (Hz) of the 1/3 O.B. containing the BPF

Step 12 Compute the broadband spectrum (lossless) using Table 2.2.3 with linear interpolation.

$$SPL_i = OASPL_{BB} + \underbrace{(SPL - SPL_{PK})_i}_{\text{TABLE 2.2.3}} - 7.3 \text{ dB}$$

where SPL_i = the sound pressure level in the i^{th} 1/3 octave band.

Step 13 If the BPF of Step 10 falls within the range of the 24 1/3 O.B.'s from 50 to 10K Hz, add the tone SPL of Step 8 to the broadband SPL of the corresponding 1/3 O.B. to obtain the total spectrum shape (discrete + broadband). Otherwise, go to Step 14.

$$SPL_{\substack{1/3\ O.B.\\ \text{Peak Freq.}}} = 10 \log_{10} \left[10^{(SPL_{PK}/10)} + 10^{(SPL_D/10)} \right]$$

where SPL_{PK} = peak SPL of the broadband spectrum from Step 12.

Step 14 Apply corrections to the spectrum of Steps 12 and 13 for acoustic suppression if treatment is applied to the primary exhaust duct. This correction is treated as a separate noise prediction module in section 4.1.

Step 15 If applicable, correct the turbine radiated noise for structural shielding. This correction is treated as a separate correction and is described in section 4.2. The source location for shielding estimates is taken as the center of the primary nozzle exit plane.

Step 16 Compute the overall sound pressure level of the turbine noise at the observer location for the 24 1/3 octave bands from 50 to 10,000 Hz.

$$OASPL = 10 \log_{10} \sum_{i=1}^{24} 10^{(SPL_i/10)} \text{ dB}$$

where, SPL_i is from Steps 12 and 13.

Step 17 Compute the turbine noise spectral levels (1 Hz bandwidth) as follows:

(a) 1/3 octave band containing the blade passage frequency

$$SSPL_{PK} = 10 \log_{10} \left[10^{(SPL_{PK}-K)/10} + 10^{SPL_D/10} \right] \text{ dB}$$

where, $SSPL_{PK}$ = spectral sound pressure level (SPL) of the peak frequency, f_b(Hz), of Step 11(dB).

SPL_{PK} = same as in Step 13.

K = $10 \log_{10} (\Delta f)_{PK}$ and $(\Delta f)_{PK}$ = bandwidth (Hz) of the 1/3 O.B. containing the peak frequency.

SPL_D = SPL of the discrete tone (Step 8).

(b) All other 1/3 octave bands

$$SSPL_i = SPL_i - 10 \log_{10} (\Delta f)_i \text{ dB}$$

where, $SSPL_i$ = the spectral sound pressure level (SPL)of the i^{th} 1/3 octave band (dB).

SPL_i = 1/3 O.B. SPL of the i^{th} 1/3 octave band from Step 12 (dB).

$(\Delta f)_i$ = bandwidth of the i^{th} 1/3 O.B. (Hz).

TABLE 2.2.1.— TURBINE OASPL DIRECTIVITY

ANGLE FROM ENGINE INLET ϕ' (DEG)	$OASPL - OASPL_{PK}$ (dB)
0	-13.8
92	- 4.5
120	0.0
130	- 1.3
140	- 2.8
160	- 5.4
180	- 6.6

TABLE 2.2.2.— TURBINE TONE DIRECTIVITY

ANGLE FROM ENGINE INLET ϕ' (DEG)	$SPL - SPL_{PK}$ (dB)
0	-25.6
90	- 9.5
111	- 2.5
120	0.0
130	- 4.3
140	- 6.5
150	-11.8
180	-19.5

TABLE 2.2.3.— TURBINE BROADBAND NOISE SPECTRUM

f/f_b	$SPL-SPL_{PK}$
.001	-60.0
.005	-41.0
.01	-36.0
.02	-29.5
.03	-26.0
.05	-21.5
.10	-14.5
.15	-10.9
.20	- 8.3
.25	- 6.6
.30	- 5.4
.35	- 4.4
.40	- 3.7
.45	- 3.0
.50	- 2.5
.60	- 1.7
.70	- 1.0
.80	- 0.5
.90	- 0.25
1.00	0.00
1.10	- 0.2
1.20	- 0.4
1.30	- 0.6
1.40	- 1.0
1.50	- 1.3
1.60	- 1.7
1.70	- 2.2
1.80	- 2.8
1.90	- 3.5
2.00	- 4.5

f = 1/3 O.B. CENTER FREQUENCY

f_b = PEAK FREQUENCY FROM STEP 11.

2.3 Core (Combustion) Noise

The engine core noise is predicted by an empirical procedure which predicts combustion noise and includes turbine transmission loss effects. The basic method developed by General Electric under contract to the FAA was adapted here for the near-field cruise condition. For a discussion of the selection of a core noise procedure, see Section 2.3.3.3 of CR-159104. Applications and limitations of the following computational algorithm are itemized as follows:

(a) The basic method is empirical and developed for static, sea level conditions to predict core (combustion) noise in the far field under free-field conditions.

(b) The peak frequency is assumed to be 400 Hz. The test data indicate this may vary by plus or minus one 1/3-octave band. Furthermore, there is some indication that low-emission combustors may peak at higher frequencies (630 to 1000 Hz).

(c) The method is applicable to turbojets, turbofans, or turboprops.

CORE NOISE COMPUTATIONAL ALGORITHM

The following input parameters are required:

W_3 = combustor total airflow

T_4 = combustor exit total temperature

T_3 = combustor inlet total temperature

$(T_4-T_5)_{des}$ = total temperature drop across the entire high and low pressure turbine section at the engine design point

ρ_3 = combustor inlet density

ρ_a = ambient density

ρ_0 = sea level, standard day density

c_a = ambient speed of sound

c_o = sea level, standard day speed of sound

r = distance from center of primary nozzle exit plane to observer

ϕ = angle between the engine inlet axis and the line defined by r above (deg.)

M_A = aircraft Mach number

W_{ref} = reference combustor airflow, 15.9 kg/s (35 lbm/s, 1.09 slug/s)

ΔT_{ref} = reference combustor temperature rise, 777.8 °K or °C (1400 °F or °R)

r_{ref} = reference source to observer distance, 70.4 m (230.9 f)

Step 1 Find the overall sound power level from the following equation:

$$OAPWL = 10 \log_{10} [(W_3/W_{ref})(\frac{T_4-T_3}{\Delta T_{ref}})^2(\rho_3/\rho_0)^2] -40 \log_{10}\left[\frac{(T_4-T_5)_{des}}{\Delta T_{ref}}\right]$$

$$+ 123.5 \text{ dB re: } 10^{-13} W$$

Step 2 Compute the 1/3 octave band power spectrum at approximately 120° from the engine inlet using Table 2.3.1.

$$PWL_{i,120°} = PWL_{400 \text{ Hz}} + DELPWL$$

where, $PWL_{i,120°}$ = the i^{th} 1/3 octave band sound power level (dB)

$PWL_{400 \text{ Hz}}$ = OAPWL -6.8 dB (the peak 1/3 O.B. sound power level)

DELPWL = ΔdB correction from Table 2.3.1.

Step 3 Calculate the 1/3 octave band sound pressure level spectrum from 50 to 10,000 Hz for free-field, lossless conditions at the reference distance corrected for local acoustic impedance (approx. 120° from the engine inlet).

$$SPL_{i,120°} = PWL_{i,120°} + 10 \log_{10}(\rho_a c_a/\rho_o c_o) -57.8 \text{ dB}$$

where, SPL_i and PWL_i = the i^{th} 1/3 O.B. sound pressure level and sound power level, respectively.

Step 4 Computh the source emission angle (ϕ') and the sound propagation path length (r') for the desired observer location (forward speed effect).

(a) $M_A = 0.0$

$$\phi' = \phi \text{ and } r' = r$$

(b) $0.0 < M_A < 1.0$

- If $\phi = 0°$, $\phi' = 0°$ and $r' = r/(1-M_A)$
- If $\phi = 180°$, $\phi' = 180°$ and $r' = r/(1+M_A)$

o If $0° < \phi < 180°$

$$\phi' = \cot^{-1}\left[\frac{1}{1-M_A^2}\{\cot\phi + M_A\sqrt{1-M_A^2+\cot^2\phi}\}\right] \text{ deg.}$$

$$r' = r \sin\phi/\sin\phi'$$

Step 5 Apply correction for directivity angle, distance, and convective amplification.

(a) Dual-flow engine exhaust configurations (e.g., 3/4-duct turbofan).

$$SPL_i = SPL_{i,120°} + DFDI - 20\log_{10}(r'/r_{ref}) - 40\log_{10}(1-M_A\cos\phi')\text{dB}$$

where, SPL_i = the i^{th} 1/3 O.B. sound pressure level at the desired location (dB)

DFDI = directivity index from Table 2.3.2 as a function of ϕ' (linear interpolation).

(b) Single-flow exhaust configurations (e.g., mixed-flow exhaust).

$$SPL_i = SPL_{i,120°} + SFDI - 20\log_{10}(r'/ref) - 40\log_{10}(1-M_A\cos\phi')\text{dB}$$

where, SFDI = directivity index from Table 2.3.3 as a function of ϕ' (linear interpolation).

Step 6 Apply corrections to the core noise spectrum for acoustic suppression if treatment is applied to the primary duct. This correction is treated as a separate prediction module. See section 4.1.

Step 7 1/3 octave band corrections for structural shielding, if applicable, are applied using the method of section 4.2. The source location is taken as the center of the primary nozzle exit plane.

Step 8 Compute the OASPL at the observer location for the 24 1/3 octave bands from 50 to 10,000 Hz.

$$OASPL = 10\log_{10}\sum_{i=1}^{24} 10^{(SPL_i/10)} \quad \text{dB}$$

Step 9 Compute the spectral levels (1 Hz bandwidth) of the core noise component as follows:

$$SSPL_i = SPL_i - 10\log_{10}(\Delta f)_i \quad \text{dB}$$

where, $SSPL_i$ = the spectral SPL at the observer location for the i^{th} 1/3 O.B.

$(\Delta f)_i$ = the bandwidth of the i^{th} 1/3 octave band (Hz).

TABLE 2.3.1.— ENGINE CORE NOISE SPECTRUM SHAPE

1/3 O.B. Frequency, Hz	PWL - Peak Frequency PWL, dB (DELPWL)
50	-24.0
63	-20.0
80	-16.0
100	-13.0
125	-10.0
160	- 7.0
200	- 4.5
250	- 2.5
315	- 1.0
400	0
500	- 1.0
630	- 2.5
800	- 4.5
1000	- 7.0
1250	-10.0
1600	-13.0
2000	-16.0
2500	-20.0
3150	-24.0
4000	-27.5
5000	-31.5
6300	-36.0
8000	-40.0
10000	-45.0

TABLE 2.3.2.— DIRECTIVITY INDICES FOR DUAL-FLOW EXHAUST SYSTEMS, DUAL-FLOW DIRECTIVITY INDEX, DFDI

1/3 O.B. Frequency, Hz	Angle from engine inlet (ϕ'), Degrees 40	50	60	70	80	90	100	110	120	130	140
50	-4.0	-3.8	-3.2	-3.0	-2.7	-2.0	-0.8	+0.8	+3.0	+5.0	+7.0
63	-4.0	-3.8	-3.2	-3.0	-2.7	-2.0	-0.8	+0.8	+3.0	+5.0	+7.0
80	-4.0	-3.8	-3.2	-3.0	-2.7	-2.0	-0.8	+0.8	+3.0	+5.0	+7.0
100	-4.0	-3.8	-3.2	-3.0	-2.7	-2.0	-0.8	+0.8	+3.0	+5.0	+7.0
125	-6.5	-5.8	-5.0	-4.0	-3.0	-1.5	0	+1.8	+4.0	+5.0	+5.0
160	-6.5	-5.8	-4.5	-3.5	-2.5	-1.5	0	+1.5	+3.5	+4.8	+6.0
200	-6.5	-5.8	-5.0	-4.5	-4.0	-3.0	-1.8	+1.0	+3.5	+5.5	+6.5
250-10K	-10.0	-8.5	-6.5	-4.5	-2.5	-0.5	+1.0	+2.5	+5.0	+4.5	+3.5

TABLE 2.3.3.— DIRECTIVITY INDICES FOR SINGLE-FLOW EXHAUST SYSTEMS, SINGLE-FLOW DIRECTIVITY INDEX, SFDI

Angle From Engine Inlet (ϕ'), Degrees	SFDI
10	-8.0
20	-7.5
30	-7.0
40	-6.5
50	-6.0
60	-5.3
70	-4.6
80	-4.0
90	-1.7
100	0.7
110	3.0
120	5.0
130	3.5
140	1.2
150	-1.8
160	-5.0

2.4 Jet Mixing Noise

The jet mixing noise is the only propulsion system noise component for which a basic near-field prediction procedure could be found. This semi-empirical procedure was developed at Lockheed-Georgia and is discussed in section 2.3.3.5 of CR-159104. The applications and limitations are indicated as follows:

(a) The semi-empirical method predicts near-field jet mixing noise sound pressure levels in three adjacent octave bands and an overall sound pressure level. The method is applicable to a single jet at sea level static conditions.

(b) The present analysis assumes no doppler frequency shift for jet mixing noise with forward motion. See the reference documents given in section 2.3.3.5 of CR-159104 for further discussion.

(c) The methods and assumptions involved in adapting the basic procedure to coaxial jets with forward motion are described in the reference documents referred to in (b) above.

NEAR-FIELD JET MIXING NOISE COMPUTATIONAL ALGORITHM

Input required:

V_J = primary jet exhaust velocity

T_J = primary jet exhaust total temperature (absolute)

t_a = ambient temperature (absolute)

M_J = primary jet exhaust Mach number

D_J = primary jet exit diameter

M_A = aircraft Mach number

V_A = aircraft velocity

x = receiver location along jet axis measured from primary jet exit plane (positive aft)

y = receiver location measured perpendicular to jet axis

ρ_a = ambient density

c_a = ambient speed of sound

ρ_o = sea level, standard day density

c_o = sea level, standard day speed of sound

T_{ref} = reference exhaust total temperature, 555.6° K (1000° R)

P_{ref} = reference acoustic pressure, 0.00002 N/m^2 (0.0002 $dynes/cm^2$, $2.9008(10)^{-9}$ PSI)

For plug nozzles:

H = annulus height

For coaxial jets:

V_S = secondary jet exhaust velocity

T_S = secondary jet exhaust total temperature (absolute)

AR = ratio of secondary jet exit area to primary jet exit area

R_w = ratio of secondary jet mass flowrate to the primary jet mass flowrate

D_S = diameter of secondary jet nozzle exit

x_S = receiver location along jet axis measured from the secondary nozzle exit plane (positive aft).

Step 1 Compute the primary jet effective Mach number (M):

(a) Single circular jet (or mixed-flow exhaust) at zero forward speed or coaxial jet:

$$M = M_J$$

or

(b) Single circular jet (or mixed-flow exhaust) with forward speed

$$M = M_J \left[1 - \frac{V_A}{V_J}\right]^{0.75}$$

Step 2 (Plug nozzles only) Compute equivalent diameter and R_d parameter for plug nozzles (see figure 2.4.1).

$$D_e = 2\sqrt{H(D_J-H)}$$

$$R_d = 2(H/D_e)$$

Step 3 Compute reduced *nozzle*-to-receiver distances (see figure 2.4.2):

$$\bar{x} = x/D_e$$

$$\bar{y} = y/D_e$$

where, D_e = jet diameter (D_J) or D_e from Step 2 if plug nozzle

Step 4 Compute reduced *source*-to-receiver coordinates in polar and cartesian form for each of the three octave bands of the prediction method plus the overall. Obtain the required coefficients, A(I,J) for I = 1, 2, 3 and 4 from Table 2.4.2. See also figure 2.4.2.

$$\bar{x}_o(I) = A(I,13)M^{A(I,14)}\ (T_J/T_{ref})^{A(I,15)} + A(I,16)M$$

$$\bar{y}_o(I) = 0.5 + 0.132\ \bar{x}_o(I)$$

Case 1 $\bar{y} > \bar{y}_o(I)$:

$$\theta_s(I) = \tan^{-1}\left(\frac{\bar{y} - \bar{y}_o(I)}{\bar{x} - \bar{x}_o(I)}\right) \text{ deg} \qquad \text{NOTE: } \theta_s(I) \text{ must be} \geq 7.5°$$

$$\bar{r}_s(I) = \sqrt{(\bar{x} - \bar{x}_o(I))^2 + (\bar{y} - \bar{y}_o(I))^2}$$

Case 2 $\bar{y} \leq \bar{y}_o(I)$ and $\bar{x} < 0$ (positive aft)

$$\text{SET} = \theta_s(I) = 180°$$

$$\bar{r}_s(I) = \sqrt{(\bar{x} - \bar{x}_o(I))^2 + \bar{y}^2}$$

Step 5 (Forward speed only) Compute the source emission angle W.R.T. the jet exhaust axis, and the actual reduced source-to-receiver distance.

(a) $M_A = 0.0$

$$\theta_s'(I) = \theta_s(I)$$

$$\bar{r}_s'(I) = \bar{r}_s(I)$$

(b) $0.0 < M_A < 1.0$

If $\bar{y} \leq \bar{y}_o(I)$ and $\bar{x} < 0$,

$$\theta_s'(I) = 180^\circ \text{ and } \bar{r}_s'(I) = \bar{r}_s(I)/(1.0 - M_A)$$

If $\bar{y} > \bar{y}_o(I)$ and $\theta_s(I) \geq 7.5^\circ$

$$\theta_s'(I) = \cot^{-1}\left[\frac{1}{1-M_A^2}\left(\cot\theta_s(I) - M_A\sqrt{1-M_A^2+\cot^2\theta_s(I)}\right)\right] \text{deg}$$

$$\bar{r}_s'(I) = \frac{\bar{y} - \bar{y}_o(I)}{\sin\theta_s'(I)}$$

Step 6 Compute parameters for basic prediction equation which are not frequency dependent.

$$C_1 = M^{2.34}$$

$$C_2 = 10.65\ (T_J/t_a)^{0.93}$$

$$C_3 = -15.18\ (T_J/t_a)^{1.11}\ M^{0.89}$$

$$C_6 = 17.5\ (T_J/3T_{ref})^{0.89(M^2-1)}$$

$$C_7 = 0.41\ (T_J/3.6T_{ref})^{0.566(M^2-1)}$$

Step 7 Compute root-mean-square acoustic pressure corrected for dynamic amplification for each of the three octave bands and the overall. Use coefficients, A(I,J) for I = 1, 2, 3 and 4 from Table 2.4.2.

$$\alpha^2 = A(I,1)M^{A(I,2)}(T_J/T_{ref})^{A(I,3)} + A(I,4)M$$

$$C_4 = A(I,5)M^{A(I,6)}(T_J/T_{ref})^{A(I,7)} + A(I,8)M$$

$$C_5 = A(I,9)M^{A(I,10)}(T_J/T_{ref})^{A(I,11)} + A(I,12)M$$

$$C_9 = \left[\frac{(1 + \alpha^2 M^2)}{\alpha^2 M^2 + \left(1 - \frac{M\cos\theta}{1 + C_6 e^{-C_7 r}}\right)^2}\right]^{5/2}$$

$$C_{10} = 1 + \frac{C_4 e^{-C_5\theta}}{1 + C_6 e^{-C_7 r/4}}$$

$$\bar{p}^2(I)/P^2_{ref} = \left(\frac{1}{C_{10}}\right)\left[A(I,17)C_9(T_J/T_{ref})^{1\cdot 54}M^4(1+\cos^4\theta)\right]\left(\frac{C_1}{r^2}+\frac{C_2}{r^4}+\frac{C_3}{r^6}\right)$$

$$\Big/ (1 \mp M_A \cos\theta)$$

where, $r = \bar{r}_s\ (I)$ If $M_A = 0.0$, or $\bar{r}_s'(I)$ If $M_A > 0.0$

$\theta = \theta_s\ (I)$ If $M_A = 0.0$, or $\theta_s'(I)$ If $M_A > 0.0$ (radians)

Step 8 Compute frequencies and SPL's for the three octave bands and the overall ($I = 1,2,3,4$):

(a) Use these equations for primary nozzles without plugs, or for secondary jets, if Steps 11, 12 and 13 have been computed:

$$LP(I) = 10\ \log_{10}\left[\bar{p}^2(I)/P^2_{ref}\right]\ \text{dB}$$

$$F(I) = H_N(I)c_a/D_J \quad \text{(Hz)}$$

(b) Use these equations for primary jets with plug nozzles

$$LP(I) = 10\ \log_{10}\left[\bar{p}^2(I)/P^2_{ref}\right] + 3\ \log_{10}\ (0.10 + 2H/D_e) \quad \text{dB}$$

$$F(I) = H_N(I)c_a/(D_e R_d^{\,0\cdot 4}) \quad \text{(Hz)}$$

where $H_N(I)$ = Hemholtz number from Table 2.4.1

D_e = equivalent diameter of plug nozzle (Step 2)

R_d = Strouhal number correction parameter for plug nozzles (Step 2)

Step 9 (For coaxial jets only) Compute correction to primary jet SPL's to account for presence of the secondary jet. ($I = 1,2,3,4$)

(a) Find M_c from Table 2.4.3 (linear interpolation)

(b) $\Delta L_p(I) = 10\ M_c\ \log_{10}(1 - \frac{V_S}{V_J})$

(c) $L_p(I) = L_p(I)_{STEP\ 8} + \Delta L_p(I)$

Step 10 Use Table 2.4.4 to expand the frequency range. Compute SPL's in octave bands to either side of the three bands of the prediction method to cover the range from 50 to 10,000 Hz.

(a) For frequencies less than F(2) (see Step 8) where F(2) > 50 Hz.

(1) Compute octave band center frequencies to cover range from F(2) to 50 Hz.

$$F_{A(N)} = F(2)/N$$

where $N = 1,2,4,8,\ldots$ etc. until $F_{A(N)} \leq 50$.

(2) Compute Strouhal numbers for above frequencies.

$$S_{A(N)} = \frac{0.221\ c_a}{V_J} \frac{F_{A(N)}}{F(2)} \qquad N = 1,2,4,8,\ldots$$

(3) From Table 2.4.4, find reduced sound pressure level, $L_{A(N)}$, for each $S_{A(N)}$ and then compute an SPL for each new octave band below F(2). For computer purposes, interpolation in x/D_J, and $\log_{10}(S)$ is recommended.

$L_{A(N)} = F^N[S_{A(N)}]$ from Table 2.4.4, then

$$L_{P_{A(N)}} = L_p(2) - L_{A(N)} + L_{A(1)} \qquad N = 2,4,8,\ldots$$

(b) For frequencies above F(4) (see Step 8) where F(4) < 10,000 Hz.

(1) Compute octave band frequencies to cover range from F(4) to 10,000 Hz.

$F_{B(N)} = N \cdot F(4)$ where $N = 1,2,4,8,\ldots$ etc. until

$F_{B(N)} \geq 10{,}000$ Hz.

(2) Compute Strouhal numbers for above frequencies.

$$S_{B(N)} = \frac{0.884\ c_a}{V_J} \frac{F_{B(N)}}{F(4)} \qquad N = 1,2,4,8,\ldots$$

(3) From Table 2.4.4, find reduced SPL, $L_{B(N)}$, for each $S_{B(N)}$ and then compute an SPL for each new octave band above F(4). For computer purposes interpolation in x/D_J and $\log_{10}(S)$ is recommended.

$L_{B(N)} = F^N[S_{B(N)}]$ from Table 2.4.4, then

$$L_{P_{B(N)}} = L_p(4) - L_{B(N)} + L_{B(1)} \qquad N = 2,4,8,\ldots$$

where, $L_p(2)$ and $L_p(4)$ are obtained from Step 8 or 9.

F(2) and F(4) are obtained from Step 8.

NOTE: *Skip to Step 14, for single jets.*

Step 11 (For coaxial jets) Compute effective secondary jet flow parameters.

$$T_E = (T_J + R_W T_S)/(1 + R_W)$$

$$V_E = (V_J + R_W V_S)/(1 + R_W)$$

$$M_E = \frac{V_E}{20.04 \sqrt{T_E - (4.978(10)^{-4})V_E^2}}$$

where V_E is in m/s and T_E is in °K

for forward speed, set $M_E = M_E(1 - \frac{V_A}{V_E})^{0.75}$

Step 12 (For coaxial jets) Compute effective reduced nozzle-to-observer distances for secondary flow.

$$\bar{x} = x_s/D_s$$

$$\bar{y} = y/D_s$$

Step 13 (For coaxial jets) To obtain the octave band spectrum for the secondary jet repeat Steps 4 through 8 and Step 10 using following parameters:

Substitute M_E for M, T_E for T_J, V_E for V_J, and D_s for D_J.

Use $\bar{x}$ and $\bar{y}$ from Step 12.

Step 14 We now have an octave band spectrum for a single jet or in the case of a coaxial jet, a primary jet spectrum and a secondary jet spectrum covering the frequency range from 50 to 10,000 Hz. Use interpolation to obtain octave band SPL's at the 24 1/3 O.B. frequencies from 50 to 10,000 Hz; interpolation in $\log_{10}(f)$ is recommended.

Step 15 Convert octave band levels at 1/3 O.B. center frequencies to 1/3 O.B. levels.

$$(1/3 \text{ O.B. SPL})_{f_i} = (\text{O.B. SPL})_{f_i} - 4.85 \text{ dB}$$

where, f_i = 1/3 O.B. center frequency, i = 1 to 24.

Step 16 (Coaxial jets) To obtain the total jet spectrum add, logarithmically, the primary and secondary jet 1/3 octave band levels at each of the 24 1/3 O.B. center frequencies.

$$SPL_i = \text{Total } 1/3 \text{ O.B. SPL @ } f_i = 10 \log_{10}\left(10^{(SPL_1/10)} + 10^{(SPL_2/10)}\right)$$

where, f_i = the i^{th} 1/3 O.B. center frequency

SPL_1 = primary jet 1/3 O.B. SPL @ f_i

SPL_2 = secondary jet 1/3 O.B. SPL @ f_i

Step 17 Correct the total jet mixing noise 1/3 octave band sound pressure levels for local acoustic impedance.

$$SPL_i = SPL_i + 10 \log_{10}\left(\frac{\rho_a c_a}{\rho_o c_o}\right) \text{ dB} \qquad i = 1 \text{ to } 24$$

Step 18 Structural shielding effects, where appropriate, are estimated using the jet noise shielding procedure of section 4.2.

Step 19 Compute the jet mixing noise overall sound pressure level (OASPL) at the observer location.

$$OASPL = 10 \log_{10} \sum_{i=1}^{24} 10^{(SPL_i/10)} \quad \text{dB}$$

Step 20 Compute the spectral levels (1 Hz bandwidth) of the jet mixing noise for each 1/3 octave band by applying a bandwidth correction as follows:

$$SSPL_i = SPL_i - 10 \log_{10} (\Delta f)_i \quad \text{dB}$$

where, $SSPL_i$ = the spectral SPL corresponding to the i^{th} 1/3 octave band

$(\Delta f)_i$ = the bandwidth of the i^{th} 1/3 O.B. (Hz).

TABLE 2.4.1

IDENTIFICATION OF FREQUENCY BANDS FOR JET NOISE ESTIMATION

		Hemholtz Number $H_N = fD/c_a$	
Index I	Freq. Band	Band Center	Range
1	Overall	1.25	0.078 to 20
2	Octave	0.221	0.156 to 0.312
3	Octave	0.442	0.312 to 0.625
4	Octave	0.884	0.625 to 1.25

TABLE 2.4.2

COEFFICIENTS A(I,J) FOR JET NOISE PARAMETERS

	Frequency Band I			
Coefficient Number J	1 OVERALL	2 OCTAVE	3 OCTAVE	4 OCTAVE
1	0.8239	0.4176	0.32555	0.74038
2	-1.24	-1.7269	0.2482	-3.4538
3	-2.3019	-1.3226	-4.0206	-4.7181
4	1.4745	1.3507	2.6807	3.2662
5	45.759	18.932	202.88	331.55
6	-2.1086	-3.2199	3.8424	0.53294
7	3.3692	1.3276	10.520	9.5983
8	-2.3864	-0.1765	-8.0248	-6.829
9	12.808	8.4045	10.491	10.429
10	-1.8167	-1.5494	-0.019983	0.36732
11	-1.7894	-0.16571	0.73551	0.89132
12	1.1641	0.22007	-0.76495	-0.87600
13	4.512	5.2846	4.5428	2.6963
14	-0.028729	1.3015	0.20763	1.8687
15	0.789	0.85116	0.27691	3.0174
16	-0.51772	-0.60053	0.036298	-2.1329
17	2.293×10^{14}	1.993×10^{13}	6.086×10^{13}	4.794×10^{13}

TABLE 2.4.3

CORRECTION FACTOR (M_c) FOR THE PRIMARY JET NOISE OF COAXIAL JETS

$\log_{10}\left[\frac{f(I)D_e}{V_J}\right]$	M_c AREA RATIO (AR) 1	3	6	10	15
-1.15	2.00	0.85	0.00	0.00	0.95
-1.00	1.22	0.80	0.66	0.74	1.60
-0.8	0.46	1.02	1.45	1.66	2.63
-0.6	0.74	1.34	2.16	2.46	3.60
-0.4	1.06	1.68	2.82	3.15	4.32
-0.2	1.41	2.07	3.47	3.74	4.77
0.0	1.78	2.49	4.07	4.23	4.99
0.2	2.16	2.93	4.62	4.62	5.02
0.4	2.55	3.38	4.97	4.91	5.03
0.6	2.95	3.85	5.04	5.04	5.04
0.8	3.36	4.31	5.06	5.06	5.06
1.0	3.80	4.72	5.08	5.08	5.08
1.2	4.24	5.04	5.10	5.10	5.10
1.4	4.70	5.10	5.10	5.10	5.10
1.6	5.00	5.10	5.10	5.10	5.10
1.8	5.10	5.10	5.10	5.10	5.10
2.0	5.10	5.10	5.10	5.10	5.10

$f(I)$ = Primary jet frequencies from Step 8, I = 1,2,3,4

D_e = D or $D_e R_d^{0.4}$ for primary nozzles without or with plugs, respectively.

TABLE 2.4.4

REDUCED SPECTRA SHAPES FOR NEAR-FIELD JET NOISE

$Y/D \leq 30$

REDUCED SOUND PRESSURE LEVEL, $L_{A(N)}$ OR $L_{B(N)}$ (dB)

Strouhal No.	X/D_J			
$S_{A(N)}$ or $S_{B(N)}$	0 → 5*	10	20	30
0.04	23.8	19.9	16.5	13.6
0.05	21.9	18.1	14.9	11.4
0.06	20.5	16.8	13.3	9.9
0.07	19.2	15.6	12.3	8.6
0.08	18.2	14.6	11.4	7.6
0.09	17.3	13.8	10.6	6.8
0.10	16.6	13.0	9.9	6.2
0.12	15.3	11.9	8.9	5.2
0.16	13.4	10.0	7.3	4.1
0.20	11.9	8.6	6.2	3.6
0.25	10.4	7.4	5.3	3.6
0.30	9.2	6.5	4.8	3.8
0.40	7.5	5.3	4.1	4.7
0.50	6.2	4.3	3.8	5.8
0.60	5.2	3.8	3.8	6.8
0.70	4.5	3.2	4.0	7.6
0.80	3.8	3.0	4.2	8.5
0.90	3.4	2.8	4.6	9.2
1.0	3.0	2.6	5.0	10.0
1.2	2.6	2.7	5.6	11.3
1.6	2.2	3.2	6.9	13.6
2.0	2.2	3.8	8.0	15.5
2.5	2.6	4.7	9.3	17.4
3.0	2.9	5.4	10.4	19.2
4.0	3.6	6.8	12.4	21.9
5.0	4.2	7.9	14.0	24.1
6.0	4.9	9.0	15.5	26.0
7.0	5.5	10.0	16.6	27.5
8.0	6.0	10.9	17.7	28.9
9.0	6.6	11.7	18.7	30.1
10.0	7.1	12.4	19.5	31.4

*These values are assumed to apply to the forward quadrant up to $X/D \geq -30$.

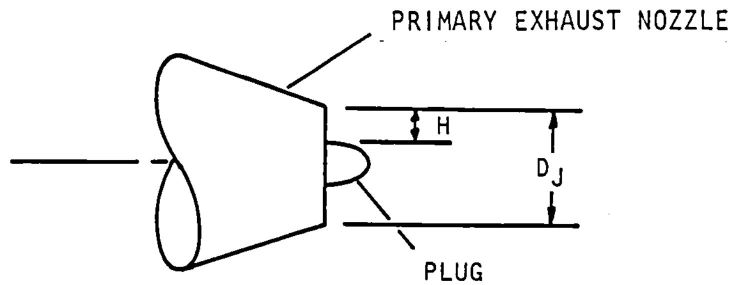


Figure 2.4.1 Primary exhaust plug nozzle geometry.

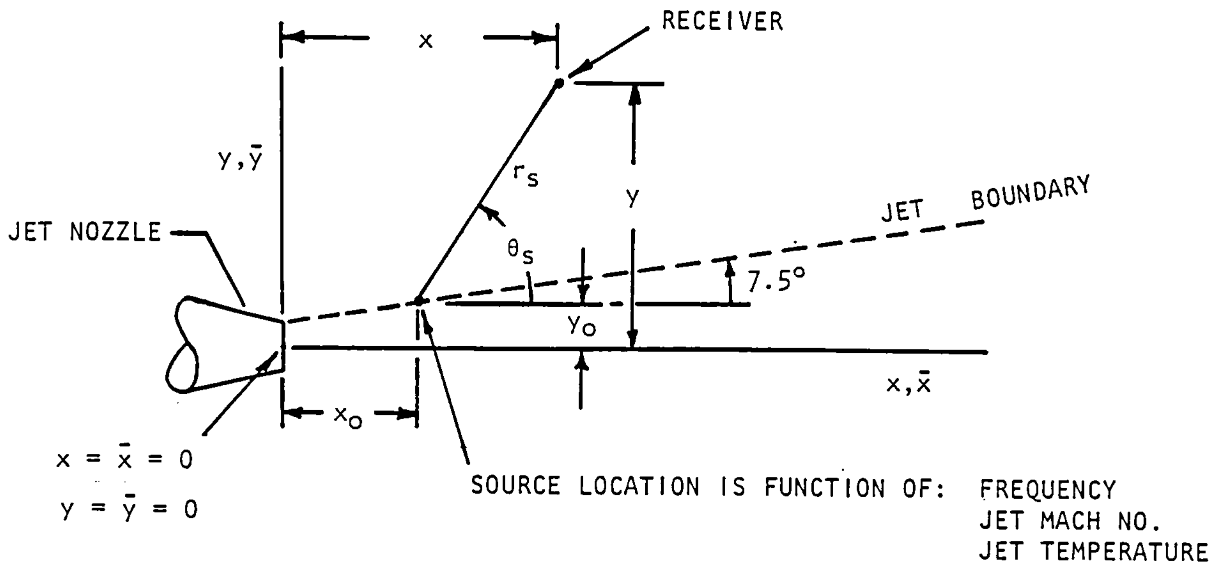


Figure 2.4.2 Source/receiver geometry for jet mixing noise.

2.5 Jet Shock Associated Broadband Noise

The selected prediction procedure is discussed in section 2.3.3.5 of CR-159104. The method was originally derived by Harper-Bourne and Fisher and later modified slightly by Lockheed-Georgia. This method predicts the broadband noise associated with expansion shock waves in the jet exhaust of nozzles operating at supercritical pressure ratios. The applications of the procedure are outlined as follows:

(a) By replacing jet diameter with a characteristic jet dimension, the method is believed to be applicable to secondary jets (an annular nozzle exit) as well as primary or single circular jets.

(b) The method is only applicable to source emission angles (ϕ''s) of 150° or less with respect to the engine inlet.

(c) The method as applied here assumes that the ratio of jet exit static temperature to ambient temperature is equal to or greater than 0.9.

(d) The present procedure is only applicable to modified shock Strouhal numbers, S_i (see Step 5(a) of the algorithm), of 0.2 to 70.0 inclusive.

SHOCK NOISE COMPUTATIONAL ALGORITHM

Input required:

V_J = jet exit velocity

M_J = jet exit Mach number

D_e = equivalent diameter of the nozzle exit.
$D_e = 2\sqrt{A/\pi}$ where A = nozzle exit area.

H_J = jet exit characteristic dimension
for single circular jets or primary jets, H_J = nozzle equivalent diameter
for a separate-flow fan nozzle, H_J = annulus height

M_A = airplane Mach number

c_a = ambient speed of sound

ρ_a = ambient density

c_o = sea level, standard day speed of sound

ρ_o = sea level, standard day density

r = distance from nozzle exit to observer

ϕ = observer angle from the engine inlet measured at the nozzle exit (deg)

Step 1 Compute the following shock parameters:

$$\beta = \sqrt{M_J^2 - 1.0}$$

$$L_o = K_o H_J \beta$$

$$L_1 = K_1 H_J \beta$$

$$M_c = C \cdot (V_J/c_a)$$

$$V_c = C \cdot V_J$$

where C = 0.70 (eddy convection velocity constant)

K_0 = 1.10 (average shock spacing constant)

K_1 = 1.31 (first shock spacing constant)

Step 2 Compute source emission angle (ϕ') and actual sound propagation path length (r') from actual observer angle and distance from the nozzle (forward speed effect) where $M_A < 1.0$.

(a) $M_A = 0.0$

$$\phi' = \phi \text{ and } r' = r$$

(b) $0.0 < M_A < 1.0$

- If $\phi = 0°$, $\phi' = 0°$ and $r' = r/(1-M_A)$
- If $\phi = 180°$, $\phi' = 180°$ and $r' = r/(1+M_A)$
- If $0° < \phi < 180°$,

$$\phi' = \cot^{-1}\left[\frac{1}{1-M_A^2}\left(\cot\phi + M_A\sqrt{1-M_A^2 + \cot^2\phi}\right)\right] \text{ deg}$$

$$r' = r \sin\phi/\sin\phi'$$

Step 3 Compute the following two terms of the shock noise prediction equation:

$W_1 = 2\pi B_c H_J \beta / c_a$ where B_c = 0.2316 (bandwidth constant)

and, If $\beta \leq 1.0$, $ANSI = 40 \log_{10} \beta - 20 \log_{10} (r'/D_e)$ (dB)

If $\beta > 1.0$, $ANSI = 20 \log_{10} \beta - 20 \log_{10} (r'/D_e)$ (dB)

Step 4 Compute the following parameter as a function of angle (ϕ').

$$D_F = 1.0 - M_c \cos(180 - \phi')$$

$$W_2 = L_1 D_F/V_c$$

Step 5 Compute the 1/3 octave band SPL's at each frequency from 50 to 10,000 Hz as follows:

(a) $S_i = 2\pi f_i L_o/c_a$

where, f_i = the i^{th} 1/3 O.B. center frequency from 50 to 10,000 Hz.

(b) From Table 2.5.1, find H_i as a function of S_i. (linear interpolation in $\log_{10}(S_i)$ and H_i if required)

(c) Find C_i from Table 2.5.1 as a function of S_i. (linear interpolation in $\log_{10}(S_i)$ and C_i if required)

(d) Compute the following summation:

$$SUM = \sum_{n=1}^{NS-1} \left[C_i n^2 \sum_{S=1}^{NS-n} \{\cos(W_c Q_{ns}) \sin(\frac{B_c W_c Q_{ns}}{2.0})/Q_{ns}\} \right]$$

where, $Q_{ns} = nW_2 \left[1.0 - 0.06\{(S-1) + (\frac{n+1}{2})\} \right]$

$W_c = 2\pi f_i$

NS = 8 (number of shocks)

B_c = 0.2316

(e) Compute the following parameters:

$$ANS2_i = 10 \log_{10} \left| 1.0 + \frac{4(SUM)}{NS \cdot B_c \cdot W_c} \right|$$

$$ANS3_i = 10 \log_{10} (W_1 f_i)$$

(f) Finally, compute the 1/3 octave band sound pressure level (free-field and lossless).

$$SPL_i = H_i + ANS1 + ANS2_i + ANS3_i$$

(g) Correct SPL_i for local impedance, convective amplification, and dynamic amplification.

$$SPL_i = SPL_i + 10 \log_{10} \left(\frac{\rho_a c_a}{\rho_o c_o}\right) - 40 \log_{10} (1 - M_A \cos \phi') - 10 \log_{10} (1 - M_A \cos \phi') \quad dB$$

NOTE: *Repeat the above procedure (Steps (a) through (g)) for each 1/3 octave band center frequency, f_i (Hz).*

Step 6 To estimate structural shielding effects on shock noise, use the method of section 4.2. For this calculation, the shock noise is assumed to act as a point source located on the jet centerline at distance L_1 downstream of the nozzle exit.

Step 7 Compute the overall sound pressure level for the 24 1/3 octave bands from 50 to 10,000 Hz.

$$OASPL = 10 \log_{10} \sum_{i=1}^{24} 10^{(SPL_i/10)} \quad dB$$

Step 8 Compute the spectral SPL's corresponding to the 24 1/3 octave bands as follows:

$$SSPL_i = SPL_i - 10 \log_{10} (\Delta f)_i \quad dB$$

where, $SSPL_i$ = the spectral (1 Hz bandwidth) SPL for the i^{th} 1/3 O.B.

$(\Delta f)_i$ = the bandwidth of the i^{th} 1/3 O.B.

TABLE 2.5.1

MASTER SPECTRA

S_i	H_i	C_i
0.2	116.0	0.70
0.3	121.6	0.71
0.4	125.5	0.71
0.7	132.5	0.72
1.0	137.7	0.73
1.5	142.7	0.74
2.0	145.7	0.74
3.0	148.5	0.71
3.5	149.1	0.69
4.0	149.2	0.67
4.5	149.1	0.64
5.0	148.8	0.62
6.0	147.9	0.58
7.0	146.7	0.54
8.0	145.7	0.50
10.0	143.7	0.45
20.0	137.4	0.28
40.0	130.5	0.12
68.0	125.4	0.02
70.0	125.2	0.02

2.6 Jet Shock Discrete Tone (Screech) Noise

The method presented here for the prediction of shock screech noise is taken and adapted from various sources in the literature. It is applicable to the acoustic near and far fields. The details of the development of this procedure are discussed in section 2.3.3.5 of CR-159104. The most important aspect of this noise source is that it does not always appear to be present and no criteria are available to determine the presence (or absence) of shock screech. To be on the conservative side it is recommended that this source normally be included in LFC aircraft cruise noise predictions where jet nozzle pressure ratios are 2.0 or more unless it is otherwise known that shocks or shock screech will not be present.

Applications and/or limitations of this procedure are as follows:

(a) The sound pressure levels predicted are considered as upper bound levels.

(b) Noise levels are predicted for only the fundamental and second harmonic discrete frequencies.

(c) The noise source is considered to be a point source on the jet exhaust centerline.

(d) Cruise altitude and Mach number are assumed to not effect the jet shock flow structure.

(e) For separate flow engines, this prediction procedure is applicable to either or both of the exhaust flows.

SHOCK SCREECH NOISE COMPUTATIONAL ALGORITHM

Input required:

V_J = jet exhaust velocity

NPR = Jet nozzle pressure ratio

M_A = airplane Mach number

H_J = jet nozzle exit characteristic dimension
for primary jets, H_J = nozzle exit equivalent diameter
for fan jet flows, H_J = nozzle exit annulus height

D_e = equivalent diameter of the nozzle exit

A = nozzle exit area

x = receiver location measured along jet axis from the nozzle exit plane (positive aft)

y = receiver location measured perpendicular to the jet axis

c_a = ambient speed of sound

ρ_a = ambient density

c_o = sea level, standard day speed of sound

ρ_o = sea level, standard day density

A_{ref} = reference nozzle exit area, $2.60(10)^{-3}$ m^2 (0.028 f^2, 4.03 in^2)

<u>Step 1</u> Compute the noise source location.

$$x_s = 3.5\ H_J \sqrt{NPR-1.893}$$

$$\phi_s = \tan^{-1}\left(\frac{y}{x_s - x}\right) \text{ deg}$$

$$r_s = \sqrt{y^2 + (x_s - x)^2}$$

NOTE: *The source is assumed to be located on the jet centerline at a distance x_s downstream of the appropriate nozzle exit, r_s and ϕ_s are the polar coordinates of the receiver location with the origin at the source location and the angle ϕ_s measured from the inlet (jet) axis.*

<u>Step 2</u> Compute the source emission angle (ϕ_s') and actual sound propagation path length (r_s') from the observer angle and distance from the nozzle (forward speed effect) where $M_A < 1.0$.

(a) $M_A = 0.0$

$\phi_s' = \phi_s$ and $r_s' = r_s$

(b) $0.0 < M_A < 1.0$

- If $\phi_s = 0°$, $\phi_s' = 0°$ and $r_s' = r_s/(1-M_A)$
- If $\phi_s = 180°$, $\phi_s' = 180°$ and $r_s' = r_s/(1+M_A)$
- If $0° < \phi_s < 180°$

$$\phi_s' = \cot^{-1}\left[\frac{1}{1-M_A^2}\left(\cot\phi_s + M_A\sqrt{1-M_A^2+\cot^2\phi_s}\right)\right] \text{ deg}$$

$$r_s' = r_s \sin\phi_s / \sin\phi_s'$$

Step 3 Compute reference sound pressure levels for the fundamental and second harmonic discrete tones using the data of Table 2.6.1.

$$SPL_{1,ref} = SPLREF1_{\text{Table 2.6.1}} + 10 \log_{10} (A/A_{ref}) \quad \text{dB}$$

$$SPL_{2,ref} = SPLREF2_{\text{Table 2.6.1}} + 10 \log_{10} (A/A_{ref}) \quad \text{dB}$$

Step 4 Compute directivity and distance corrections.

$$\Delta SPL_1 = 20 \log_{10} \left(\cos^2 \frac{\phi_s'}{2} + \frac{1}{2} \sin^2 \frac{\phi_s'}{2}\right)$$

$$- 20 \log_{10} (r_s'/r_{s,ref}) + 2.5 \quad \text{dB}$$

$$\Delta SPL_2 = 20 \log_{10} (\sin \phi_s') - 20 \log_{10} (r_s'/r_{s,ref}) \quad \text{dB}$$

where $r_{s,ref} = 4 D_e$

Step 5 Compute fundamental and second harmonic sound pressure levels corrected for nozzle area, directivity, and distance.

$$SPL_1 = SPL_{1,ref} + \Delta SPL_1 \quad \text{dB}$$

$$SPL_2 = SPL_{2,ref} + \Delta SPL_2 \quad \text{dB}$$

Step 6 Apply corrections for convective amplification and local acoustic impedance.

$$\Delta SPL = 10 \log_{10} \left(\frac{\rho_a c_a}{\rho_o c_o}\right) - 40 \log_{10} (1 - M_A \cos \phi_s') \quad \text{dB}$$

then, $SPL_1 = SPL_1 + \Delta SPL$ dB

$SPL_2 = SPL_2 + \Delta SPL$ dB

Step 7 Compute the fundamental and second harmonic frequencies of the shock screech tones.

$$f_1 = \frac{c_a[M_A + 0.625 (M_{Ja}-M_A)](1-M_A)(1+0.625 M_{Ja})}{1.25 \sqrt{NPR - 1.893} [1 + 0.625 (M_{Ja}-M_A)] M_{Ja} H_J} \quad \text{Hz}$$

$$f_2 = 2 f_1 \quad \text{Hz}$$

where, f_1 = fundamental frequency (Hz)

f_2 = second harmonic frequency (Hz)

$M_{Ja} = V_J/c_a$

NOTE: *Normally, for computer purposes, such as running various component noise sources, the sound pressure levels of the two screech tones are assigned to the 1/3 octave bands containing f_1 and f_2.*

Step 8 The predicted spectrum of shock screech consists of the tones of Step 6 which occur at the discrete frequencies computed in Step 7. Consequently, a bandwidth correction to convert 1/3 octave band SPL's to spectrum levels is not appropriate, i.e. the computed spectrum levels (1 Hz bandwidth) are also 1/3 O.B. levels for the 1/3 O.B.'s containing f_1 and f_2.

Step 9 To estimate the effects of structural shielding on shock screech noise use the method of section 4.2. For this calculation, use the source location as computed in Step 1, i.e. on the jet centerline at a distance x_s downstream of the appropriate nozzle exit plane.

TABLE 2.6.1

SHOCK SCREECH REFERENCE SOUND PRESSURE LEVELS

AT $r_s = 4D_e$ AND $\phi_s = 90°$

NOZZLE EXIT PRESSURE RATIO (NPR)	FUNDAMENTAL SPLREF1, dB	SECOND HARMONIC SPLREF2, dB
2.0	110.0	110.0
2.5	127.0	124.0
3.0	136.0	130.0
3.5	141.0	133.0
4.0	143.5	134.5
4.5	144.5	135.0
5.0	144.5	134.5
6.0	143.0	132.0
7.0	140.5	128.0

3. COMPUTATIONAL ALGORITHMS FOR AIRFRAME NOISE SOURCES

3.1 TRAILING-EDGE NOISE

The selection of a method for the prediction of trailing-edge noise is discussed in section 2.3.4.3 of CR-159104. The method selected was developed as an alternate procedure to the NASA ANOPP method and modified for application to the close-in far field by M. R. Fink.

The following applications and limitations should be noted for this procedure:

(a) The method is applicable to "turbulent" and laminar-flow-controlled surfaces as long as the boundary layer at the trailing edge consists of fully developed turbulent flow.

(b) In its present form, the following computational algorithm is only applicable to a receiver located in a plane perpendicular to the trailing edge span at the mid-point of the span.

TRAILING-EDGE NOISE COMPUTATIONAL ALGORITHM

Input required: (See figure 3.1.1)

M_A = aircraft Mach number

V_A = aircraft velocity

ρ_a = ambient density

c_a = ambient speed of sound

ρ_o = sea level, standard day density

c_o = sea level, standard day speed of sound

ϕ = observer angle measured from forward of the trailing edge in a plane perpendicular to the T.E. (deg)

r = distance from observer to mid-point of T.E.

ℓ = length of T.E. span being considered

δ = turbulent boundary layer thickness at the trailing edge. If this input is not available or desired, Step 1 may be used to calculate δ based on flat plate flow.

V_{ref} = reference velocity, 51.44 m/s (168.8 f/s, 100 knots, 115.1 MPH)

<u>Step 1</u> (If δ not input) Compute boundary layer thickness at the trailing edge.

$$\delta = 0.376\,\bar{c}\left(\frac{V_A \bar{c}\,\rho_a}{\mu_a}\right)^{-1/5}$$

where, μ_a = ambient viscosity

$\bar{c}$ = mean aerodynamic chord of airfoil section of interest

<u>Step 2</u> Compute the source (trailing edge mid-point) emission angle (ϕ') and propagation path length (r') corresponding to the desired observer location (forward speed effect where $M_A < 1.0$).

(a) $M_A = 0.0$

$$\phi' = \phi$$

$$r' = r$$

(b) $M_A > 0.0,\ 0° < \phi < 180°$

$$\phi' = \cot^{-1}\left[\frac{1}{1-M_A^2}\left(\cot\phi + M_A\sqrt{1-M_A^2+\cot^2\phi}\right)\right] \quad \text{deg}$$

$$r' = r\,\sin\phi/\sin\phi'$$

(c) $M_A > 0.0,\ \phi = 0°$

$$\phi' = 0°$$

$$r' = r/(1-M_A)$$

(d) $M_A > 0.0,\ \phi = 180°$

$$\phi' = 180°$$

$$r' = r/(1+M_A)$$

Step 3 Compute the angle subtended from equivalent observer location given by ϕ' and r' to ends of the trailing-edge span.

$$\beta' = 2\tan^{-1}\left(\frac{\ell}{2r'}\right) \quad \text{deg}$$

Step 4 Compute the uncorrected overall sound pressure level at the observer location.

$$\text{OASPL} = 50\log_{10}\left(\frac{V_A}{V_{ref}}\right) + 10\log_{10}(\delta\beta'/r') + 10\log_{10}[\cos(\phi'/2)]^2 + 80.7 \quad \text{dB}$$

Step 5 Correct the OASPL for local acoustic impedance and convective amplification.

$$\text{OASPL} = \text{OASPL} + 10\log_{10}\left(\frac{\rho_a c_a}{\rho_o c_o}\right) - 40\log_{10}(1-M_A\cos\phi') \quad \text{dB}$$

Step 6 Compute the peak frequency.

$$f_{PK} = 0.1\,(V_A)/\delta \quad \text{Hz}$$

Step 7 Compute the 1/3 octave band sound pressure levels from 50 to 10,000 Hz using the following equation:

$$SPL_i = OASPL + 10 \log_{10} \left[0.613 \left(\frac{f_i}{f_b} \right)^4 \left| \left(\frac{f_i}{f_b} \right)^{1.5} + 0.5 \right|^{-4} \right] \quad dB$$

where SPL_i and f_i = the i^{th} 1/3 octave band sound pressure level and center frequency (Hz), respectively

f_b = the center frequency of the 1/3 O.B. containing the peak frequency (Hz)

Step 8 Shielding effects are taken to be negligible for trailing-edge noise as a result of this sources location, orientation, and directivity pattern.

Step 9 Compute the overall sound pressure level (OASPL) at the observer location for the 24 1/3 octave bands from 50 to 10,000 Hz as follows:

$$OASPL = 10 \log_{10} \sum_{i=1}^{24} 10^{(SPL_i/10)} \quad dB$$

Step 10 Compute the spectral level (1 Hz bandwidth) for each of the 1/3 O.B. center frequencies as follows:

$$SSPL_i = SPL_i - 10 \log_{10} (\Delta f)_i$$

where $SSPL_i$ = 1 Hz bandwidth spectral level (dB)

$(\Delta f)_i$ = bandwidth of the i^{th} 1/3 O.B. (Hz)

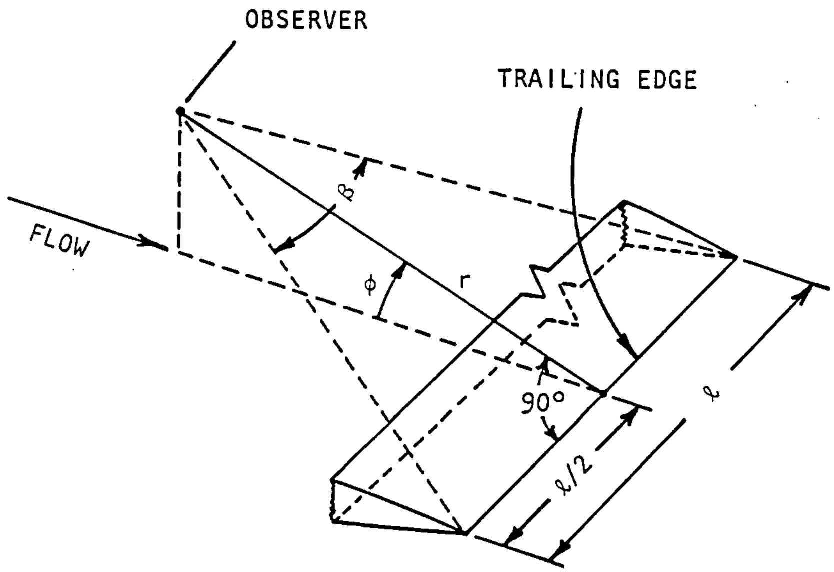


Figure 3.1.1 Source/observer geometry for trailing-edge noise.

3.2 Turbulent Boundary Layer Noise

The details of the selection and development of a prediction procedure for turbulent boundary layer noise are given in section 2.3.4.2 of CR-159104. This procedure was developed from the literature for application to the present LFC problem by Lockheed-Georgia.

The implementation of this method as described in the algorithm will normally involve approximating the surface of interest (say the side of a fuselage) by one or more flat surfaces subdivided into several elemental areas. Following the recommended procedure, the total noise from this source is estimated by summing, on an energy basis, the noise contribution from each of these sub-areas. In the near or close-in far field, the total radiated noise levels computed for a given surface will be dependent upon the number (and orientation) of these elemental areas. Unfortunately, no guidance is presently available as to a selection criteria which relates the number and orientation of the representative elemental areas to the accuracy of the prediction procedure. Note, however, that for near-field estimates the more subdividions made the more accurate will be the procedure (maximum dimension of the elemental area small compared to source-to-receiver distance), although fewer "segments" may give a conservative (high) estimate. For far-field estimates, one representative segment may be sufficient.

Input required: (See figure 3.2.1)

A_j = surface area of an element of the airframe surface under consideration

U = free-stream velocity (normally $U = V_A = c_a M_A$)

x_j, y_j, z_j = coordinates of observer location relative to a surface elemental area A_j

M_A = aircraft Mach number

ρ_a = ambient density

c_a = ambient speed of sound

ρ_o = sea level, standard day density

c_o = sea level, standard day speed of sound

δ^*_j = turbulent boundary layer displacement thickness at A_j (or may be computed as in Step 4)

U_{ref} = reference velocity, 236 m/s (774.3 f/s, 458.7 knots, 527.9 MPH)

A_{ref} = reference area, 0.0929 m^2(1.0 f^2, 144.0 in^2)

r_{ref} = reference distance, 1.0 m (3.28 ft, 39.37 in)

<u>Step 1</u> Compute the source emission angle relative to a normal to the surface (γ'_j) and the sound path length (r'_j) to the desired observer location for each elemental area, A_j, (M_A<1.0).

(a) $$\phi_j = \tan^{-1}\left(\frac{\sqrt{y_j^2 + z_j^2}}{x_j}\right) \text{ deg}$$

where $\phi_j = 90°$ if $x_j = 0$, and $0° \le \phi_j \le 180°$

(b) $$r_j = \sqrt{x_j^2 + y_j^2 + z_j^2}$$

(c) Forward speed transformation $r\phi \to r'\phi'$:

(1) If $M_A = 0.0$

$$\phi'_j = \phi_j$$

$$r'_j = r_j$$

(2) If $M_A > 0.0$, and $0° < \phi_j < 180°$

$$\phi'_j = \cot^{-1}\left[\frac{1}{1-M_A{}^2}\left(\cot\phi_j + M_A\sqrt{1-M_A{}^2 + \cot^2\phi_j}\right)\right] \text{ deg}$$

$$r'_j = r_j \sin\phi_j / \sin\phi'_j$$

(3) If $\phi_j = 0°$

$$\phi'_j = 0°$$

$$r'_j = r_j/(1-M_A)$$

(4) If $\phi_j = 180°$

$$\phi'_j = 180°$$

$$r'_j = r_j/(1+M_A)$$

(d) $\gamma'_j = \cos^{-1}(y_j/r'_j)$ deg where $0° \leq \gamma'_j \leq 90°$

where ϕ_j and γ_j are the angles to the observer location measured from the plus X axis and a normal to A_j, respectively. (See figure 3.2.1) ϕ'_j and γ'_j are the source emission angles corresponding to ϕ_j and γ'_j above.

Step 2 Compute the overall sound pressure level corrected for convective amplification and dynamic amplification for each elemental area, A_j, as follows:

$$OASPL_j = 60 \log_{10}(U/U_{ref}) + 10 \log_{10}(A_j/A_{ref}) - 20 \log_{10}(r'_j/r_{ref})$$

$$+ 20 \log_{10} \cos\gamma'_j - 30 \log_{10}(1-(0.18M_A)\cos\phi'_j)$$

$$- 10 \log_{10}(1-M_A \cos\phi'_j) + 91.1 \text{ dB}$$

NOTE: If $\gamma'_j = 90°$, set $OASPL_j = 0.0$

Step 3 Correct each $OASPL_j$ of Step 2 for local acoustic impedance.

$$OASPL_j = OASPL_j + 10 \log_{10} (\rho_a c_a / \rho_o c_o) \quad dB$$

Step 4 Compute the peak 1/3 octave band level frequency of the radiated sound for each area, A_j.

$$f_{PK_j} = 0.01102 \, (U/\delta_j^*) \quad Hz$$

NOTE: If δ_j^* is not otherwise available, it may be estimated using the following equation for flat plate flow:

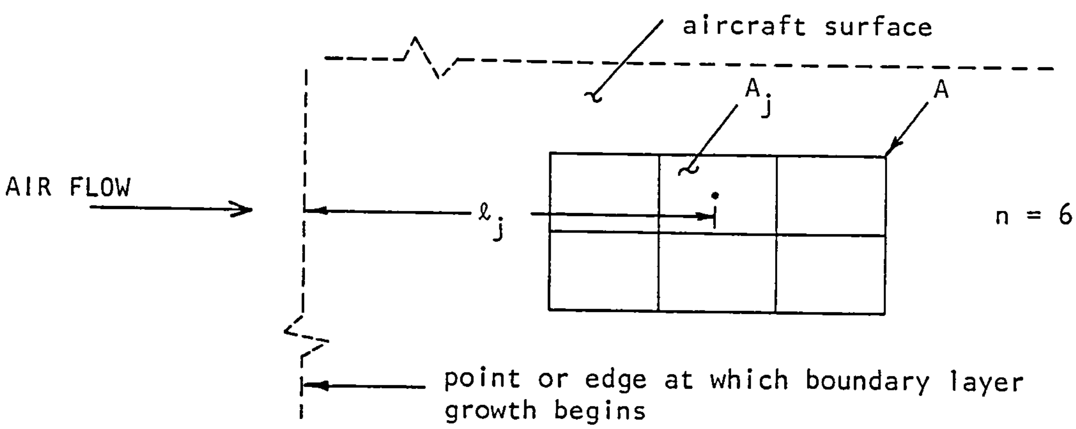
$$\delta_j = 0.376 \, \ell_j \left(\frac{\rho_a U \ell_j}{\mu_a} \right)^{-1/5}$$

Then, $\delta_j^* = \delta_j / 8.0$

where, δ_j = the turbulent boundary layer thickness

ℓ_j = distance from leading edge (or point where boundary layer growth begins)

μ_a = ambient viscosity



Note: Total scrubbed area being considered $= A = \sum_{j=1}^{n} A_j$

where n = number of elemental areas (A_j) into which the total scrubbed area being considered is subdivided.

Step 5 Compute the 1/3 octave band levels for each area, A_j, as follows:

$$SPL_{N_j} = SPL_{PEAK,j} = OASPL_j - 7.0$$

$$SPL_{N_j-1} = SPL_{N_j} - 1.0$$

$$SPL_{N_j-2} = SPL_{N_j} - 2.0$$

For $i < N_j - 2$, $SPL_i = SPL_{N_j} - 2 - (N_j-2-i)(2.7)$

For $i > N_j$, $SPL_i = SPL_{N_j} + (N_j-i)(2.2)$

where SPL_i = the i^{th} 1/3 O.B. SPL for area A_j (dB).

$SPL_{N_j} = SPL_{PEAK,j}$ = the peak 1/3 O.B. SPL for area A_j (dB).

i = the i^{th} 1/3 octave band for the 24 1/3 octave bands from 50 to 10,000 Hz.

N_j = the value of i for the 1/3 O.B. containing the peak frequency, f_{PK_j}.

Step 6 If appropriate, apply shielding correlations to each spectrum of Step 5. This procedure is described in section 4.2. The source location is taken as the center of the area, A_j.

Step 7 Compute the total spectrum levels by summing the spectra, logarithmically, of Step 5 or 6 as follows:

$$SPL_i = 10 \log_{10} \sum_{j=1}^{n} 10^{(SPL_{j,i}/10)} \quad dB$$

where n = number of elemental areas, A_j

$SPL_{j,i} = SPL_i$ from Step 5 or 6 for Area A_j

Step 8 Compute the total overall sound pressure level

$$OASPL = 10 \log_{10} \sum_{i=1}^{24} 10^{(SPL_i/10)} \quad dB$$

Step 9 Compute the spectral level (1 Hz bandwidth) for each of the 1/3 O.B. center frequencies as follows:

$$SSPL_i = SPL_i - 10 \log_{10} (\Delta f)_i$$

where $SSPL_i$ = 1 Hz bandwidth spectral level (dB)

$(\Delta f)_i$ = bandwidth of the i^{th} 1/3 O.B. (Hz)

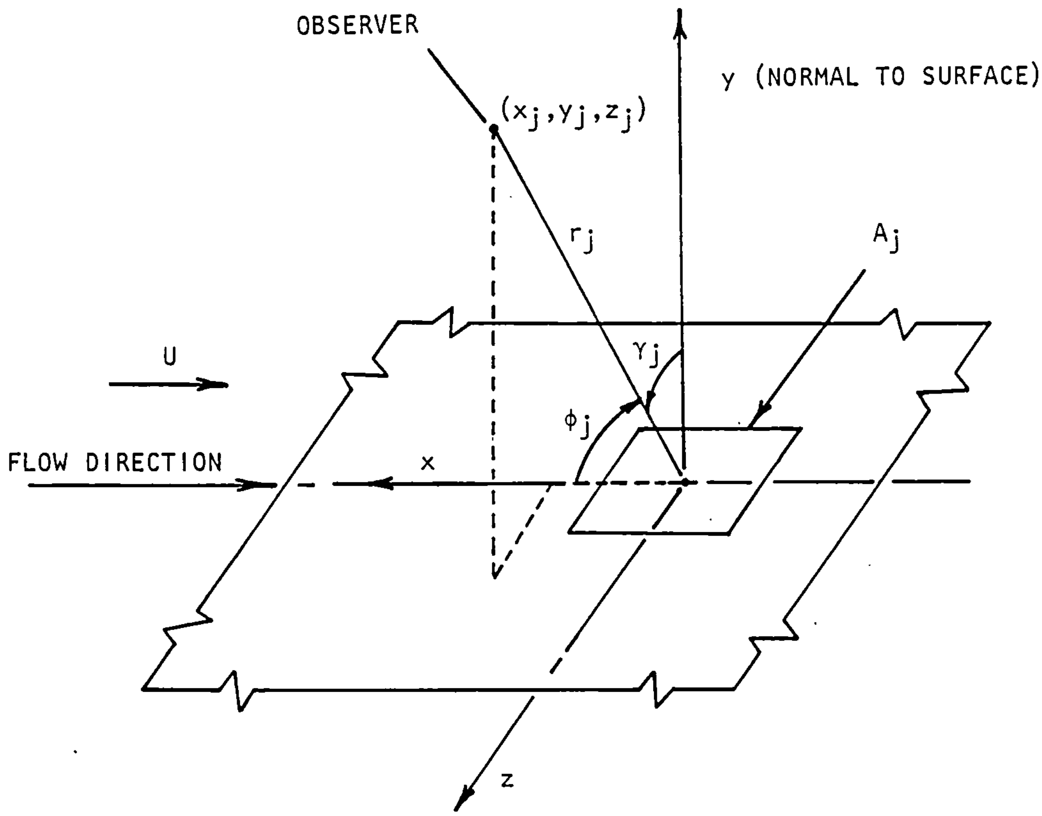


Figure 3.2.1 Geometry for turbulent boundary layer noise.

4. COMPUTATIONAL ALGORITHMS FOR ACOUSTIC RADIATION MODIFIERS

The computational procedures of this section are used to compute corrections to the predicted turbomachinery (fan/compressor, turbine, and core) noise levels of section 2.0 whenever it is necessary to account for the effects of acoustic suppression materials and to compute corrections to all the procedures of sections 2.0 and 3.0 to account for shielding of radiated noise by aircraft surfaces. The procedure for the prediction of acoustic liner attenuation effects was developed by Lockheed-Georgia, while the general procedure for estimating shielding effects has been developed by Lockheed from the method suggested by Z. Maekawa. See CR-159104 for a discussion of these methods. Note, however, that since the airframe noise sources of section 3.0 and shock noise of section 2.5 are distributed sources (i.e. do not radiate from a single point or localized area), the procedure for shielding effect corrections will be less accurate than when applied to turbomachinery sources (i.e. sources which more nearly approximate point sources.) A separate procedure for estimating shielding effects on jet mixing noise, developed by Boeing, is provided. This empirical method accounts for the distributed source characteristic of jet mixing noise.

4.1 Acoustic Suppression Effects

This section is designed to be used in conjunction with the turbomachinery noise source computational algorithms (fan/compressor, turbine, and core) of section 2.0 to include the effects of acoustic suppression materials in the noise prediction procedure. The method given here is applicable to inlet radiated fan or compressor noise and aft-radiated fan, turbine and core noise. See figure 4.1.1. The effect of high Mach number inlets on inlet radiated noise may also be included. The approach given here is to provide a procedure for the computation of corrections to be applied to the computed 1/3 octave band levels of the associated engine noise source. The method is set-up to compute the attenuation spectrum without specifying acoustic liner detailed parameters, which are assumed to be chosen for maximum attenuation at the liner design condition. The method is conservative in that it assumes a current technology single-layer liner design, i.e. a relatively narrow attenuation bandwidth. See section 2.3.3 of CR-159104 for further discussion.

COMPUTATIONAL ALGORITHM FOR ACOUSTIC LINER ATTENUATION

The following input is required:

L the effective length of treatment in the duct (with allowances made for fastening strips, etc.)

H_d the duct height between opposite liner faces

λ the wavelength of sound in the duct at the liner design frequency ($\lambda = c/f_d = 20.044 \sqrt{t_d}/f_d$ (m) where t_d = duct flow static temperature (°K), and c = speed of sound in the duct)

M_d the mean Mach number of the flow in the duct under consideration

ϕ' the source emission angle W.R.T. the inlet axis for the desired observer location (deg.)

f_d the liner design frequency (Hz)

NS the number of inlet splitter rings

Step 1 Calculate L/H and determine the peak attenuation at M = 0 and H/λ = 1.

$$\Delta dB_{peak} = 10\left(\frac{L}{H_d}\right)^{0.7} \quad dB$$

Step 2 Determine the effect of H/λ from:

$$F_1 = \Delta dB/\Delta dB_{H/\lambda=1} = (H_d/\lambda)^{-0.6}$$

Step 3 Determine the effect of duct Mach number from:

$$F_2 = \Delta dB/\Delta dB_{M=0} = 1 - \{\frac{M_d}{2}(2-(H_d/\lambda))\}$$

Step 4 Compute the correction for observer location from:

a) Exhaust duct

$$\text{for } \phi' \leq 130° \quad F_3 = \Delta dB/\Delta dB_{Max} = \frac{\phi'}{130}$$

$$\phi' > 130° \quad F_3 = \Delta dB/\Delta dB_{Max} = \frac{205-\phi'}{75}$$

b) Inlet

$$\text{for } \phi' \leq 60° \quad F_3 = \Delta dB/\Delta dB_{Max} = \left(\frac{4 - NS}{4}\right)\left(\frac{\phi'}{100}\right) + 0.15\ NS + 0.40$$

$$\phi' > 60° \quad F_3 = \Delta dB/\Delta dB_{Max} = \frac{140-\phi'}{80}$$

Step 5 Compute the attenuation at the design frequency corrected for H/λ, duct Mach number, and emission angle.

$$\Delta dB_{f_d} = \Delta dB_{peak} \times F_1 \times F_2 \times F_3$$

Step 6 Determine attenuation spectrum, as a function of frequency f_i from:

$$\Delta dB_{f_i}/\Delta dB_{f_d} = \exp\left\{-\left[\frac{\left|\log_{10}(f_i/f_d)\right|}{0.35}\right]^{1.3}\right\}$$

where f_i = the i^{th} 1/3 O.B. center frequency from 50 to 10,000 Hz.

Step 7 To obtain the applicable source spectrum corrected for acoustic treatment attenuation, apply the dB corrections of Step 6 to the uncorrected spectrum computed from the appropriate propulsion noise source algorithm as follows:

$$(SPL_i)_{\text{corrected}} = (SPL_i)_{\text{from propulsion source algorithm}} - \Delta dB_{f_i}\ \text{STEP 6}$$

where i represents the i^{th} 1/3 octave band.

HIGH MACH NUMBER INLETS

If it is desired to estimate the effects of high Mach number inlets on inlet radiated noise, proceed as follows:

Step A Compute the non-directional correction which is independent of frequency as follows:

$$\Delta dB = 216\ (M_I - 0.5)^{2.5}\ \text{dB}$$

where M_I = the inlet throat Mach number.

Step B Apply the correction of Step A across the uncorrected 1/3 octave band fan noise spectrum.

$$(SPL_i)_{FAN,Corrected} = (SPL_i)_{FAN,Uncorrected} - \Delta dB_{STEP\ A}$$

where i represents the ith 1/3 octave band.

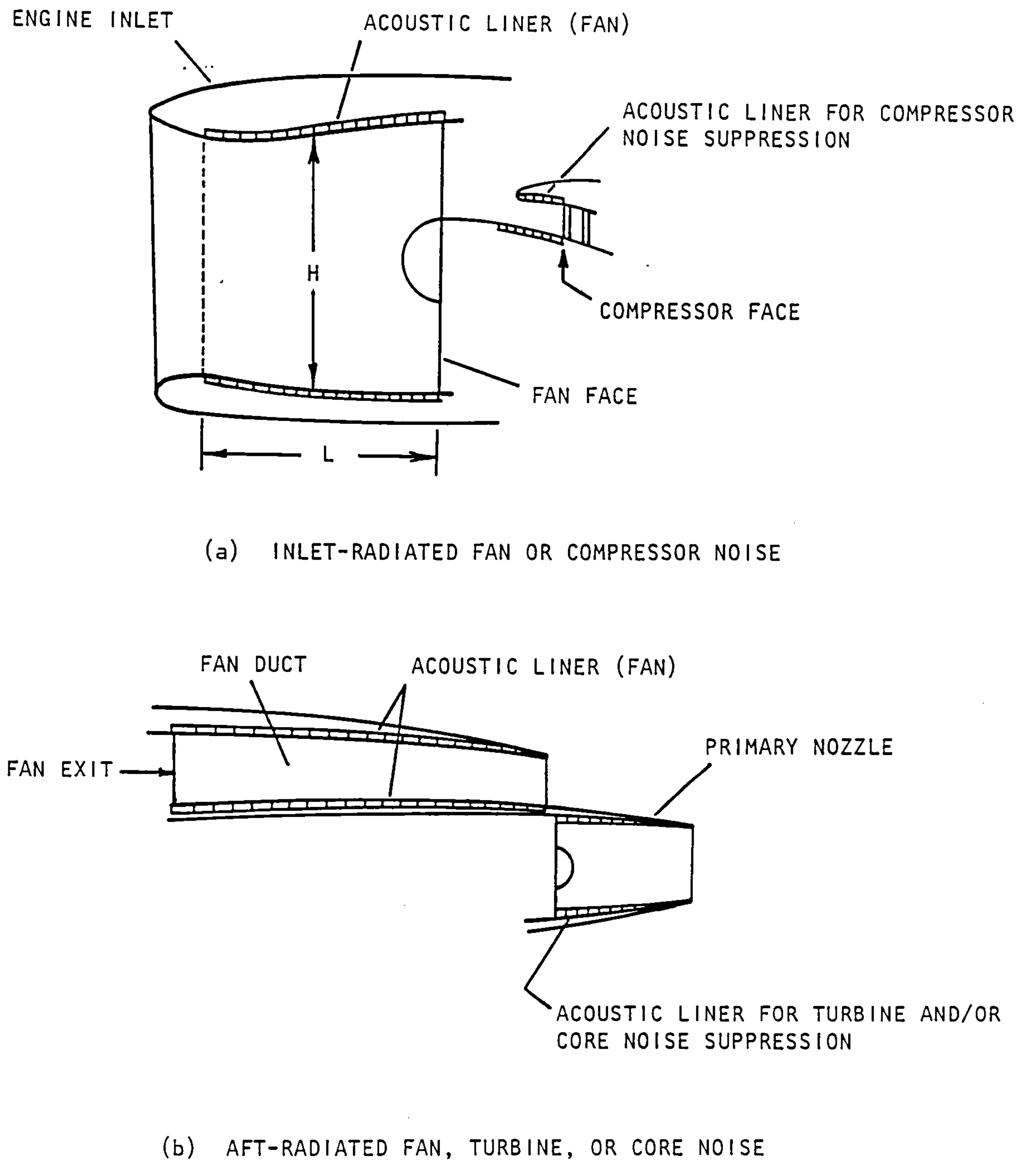


Figure 4.1.1 Typical acoustic liner installations.

4.2 Structural Shielding

The two procedures of this section are provided for use where it is desired to estimate the effect on radiated noise of direct line-of-sight shielding by aircraft surfaces. The first method is applied to all engine sources except jet mixing noise and treats these as point sources. This point source treatment is considered to be valid for fan/compressor, turbine, and core noise and to provide a reasonable approximation in the case of shock noise. The broadband shock noise source is approximated as a point source located at the first shock location downstream of the nozzle exit since there appears to be no method available to handle its distributed source character. The data available on shielding effects on jet mixing noise (also a distributed source) indicate this point source approximation can be expected to yield reasonable and conservative results. The second method is an empirical procedure for the estimation of shielding effects on jet mixing noise. For further discussion on these procedures, see section 2.4.3 of CR-159104.

It is anticipated that estimates of shielding effects on trailing-edge radiated noise will not normally be required. This results from the location and directivity of the trailing-edge source. However, the first method described above can be employed to estimate a shielding effect if desired, using the center of the appropriate span as the source location. Similarly, it is believed that the contribution of shielded turbulent boundary layer noise at most points of interest on the aircraft may be neglected. However, if shielding effects are desired to be estimated, the approach of the first method noted above may be used; with each elemental area, A_i (see section 3.2), treated as a point source. In this case, the shielding procedure will need to be included in the computational procedure of section 3.2 rather than used separately as a single final correction.

COMPUTATIONAL ALGORITHM FOR SHIELDING EFFECTS

This computational procedure is divided into two parts. The first method applies to noise components which are to be considered a point source (all sources except jet mixing noise). The second method applies to shielding of jet mixing noise.

(a) Use this procedure for all sources except jet mixing noise. The parameters required for this computation are defined in figure 4.2.1 for two shielding cases.

<u>Case 1</u> - Receiver located at some distance removed from the shielding barrier. Compute the total shielded 1/3 octave band sound pressure levels as follows:

<u>Step 1</u> Compute a correction for each 1/3 O.B. SPL for each edge of the shielding surface around which sound may be diffracted. Normally, for wing or fuselage shielding, two edges will be considered.

$$\Delta SPL_{i,j} = 10 \log_{10} \left(\frac{2 f_i z_j}{c_a} \right) + 10 \quad \text{dB}$$

If $\Delta SPL_{i,j} > 25$, Set $\Delta SPL_{i,j} = 25$

where $\Delta SPL_{i,j}$ = the ΔdB correction computed for edge j to be applied to the i^{th} 1/3 O.B. predicted (unshielded) SPL.

f_i = the i^{th} 1/3 O.B. center frequency (Hz)

z_j = the path length difference between a direct line from source to receiver and actual path length around edge j. (see figure 4.2.1)

c_a = ambient speed of sound

<u>Step 2</u> Compute the shielded spectrum associated with each 'edge' being considered.

$$SPL_{i,j} = SPL_i - \Delta SPL_{i,j} \quad \text{dB}$$

$SPL_{i,j}$ = shielded SPL for the i^{th} 1/3 O.B. associated with the j^{th} edge.

SPL_i = the predicted unshielded 1/3 O.B. SPL at the observer location (dB)

$\Delta SPL_{i,j}$ = correction from Step 1

Step 3 Compute the total shielded spectrum at the observer location by summing the contributions from each 'edge' involved.

$$SPL_{i,shielded} = 10 \log_{10} \sum_{j=1}^{n} 10^{(SPL_{i,j}/10)} \quad dB$$

where $SPL_{i,j}$ = 1/3 O.B. SPL from Step 2.

n = number of edges around which sound is considered to be diffracted to the observer location.

i = the i^{th} 1/3 O.B.

Case 2 - For a shielded receiver located at the surface of the barrier (figure 4.2.1), compute the estimated shielded spectrum using Table 4.2.1 as follows:

$$SPL_{i,shielded} = SPL_{i,unshielded} - \Delta dB_i \quad dB$$

where ΔdB_i = the dB correction to be applied to the i^{th} 1/3 O.B. SPL.

(b) Procedure for jet mixing noise shielding estimates.

Figure 4.2.2 defines the geometry used for this procedure. It is assumed that the only sound path to the observer is aft of the nozzle exit. Other input parameters are as follows.

V_J = jet exhaust velocity

c_a = ambient speed of sound

g = gravitational acceleration (9.81 m/s^2 or 32.2 f/s^2)

Step 1 For each 1/3 octave band center frequency, compute the following parameter (Z_i):

$$Z_i = \left\{\frac{1}{1 + 0.033 \left(\frac{\theta}{180 - \theta}\right)^4}\right\} \left\{\frac{f_i \; L c_a^2}{g D V_J (10)^6}\right\} \left\{1 + 4.5 (10)^9 \left(\frac{gD}{c_a^2}\right)^2\right\}$$

where f_i = the i^{th} 1/3 octave band center frequency (Hz).

Step 2 Compute the ΔSPL to be applied to the unshielded 1/3 octave band noise levels as follows:

$$\Delta SPL_i = 10 \log_{10} (1 + 0.6\, Z_i) \quad dB$$

where ΔSPL_i = the shielding correction to be subtracted from the i^{th} 1/3 octave band unshielded jet noise sound pressure level.

<u>Step 3</u> Apply the correction of Step 2 to obtain the estimated shielded 1/3 O.B. spectrum at the observer location.

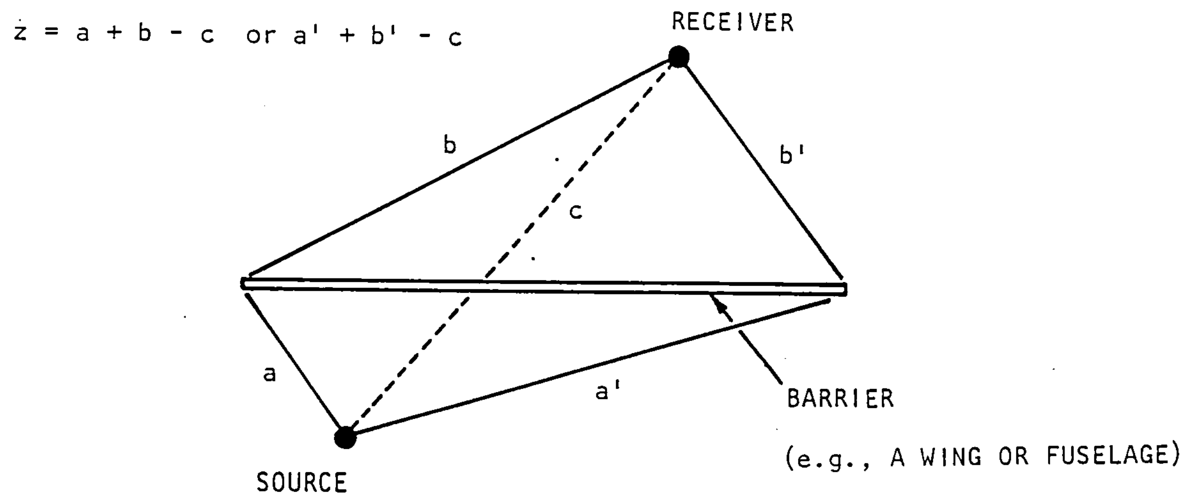
$$SPL_{i,shielded} = SPL_{i,unshielded} - \Delta SPL_i \quad dB$$

where $SPL_{i,unshielded}$ = the i^{th} 1/3 O.B. SPL predicted for the observer location without shielding (dB).

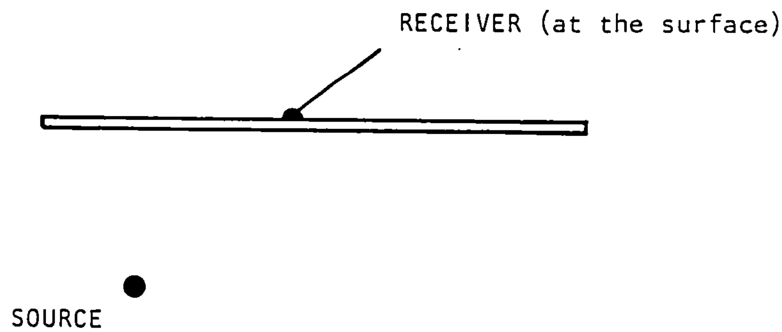
TABLE 4.2.1

ESTIMATED 1/3 Octave Band Near-Field Shielding Reductions

1/3 O.B. CENTER FREQUENCY (Hz)	SHIELDING ΔdB
50	5.7
63	6.7
80	7.8
100	8.7
125	9.7
160	10.8
200	11.7
250	12.7
315	13.7
400	14.8
500	15.7
630	16.7
800	17.8
1000	18.7
1250	19.7
1600	20.8
2000	21.7
2500	22.7
3150	23.7
4000	24.8
5000	25.0
6300	25.0
8000	25.0
10000	25.0



CASE 1



CASE 2

Figure 4.2.1 Shielding geometry for point noise sources (turbomachinery noise).

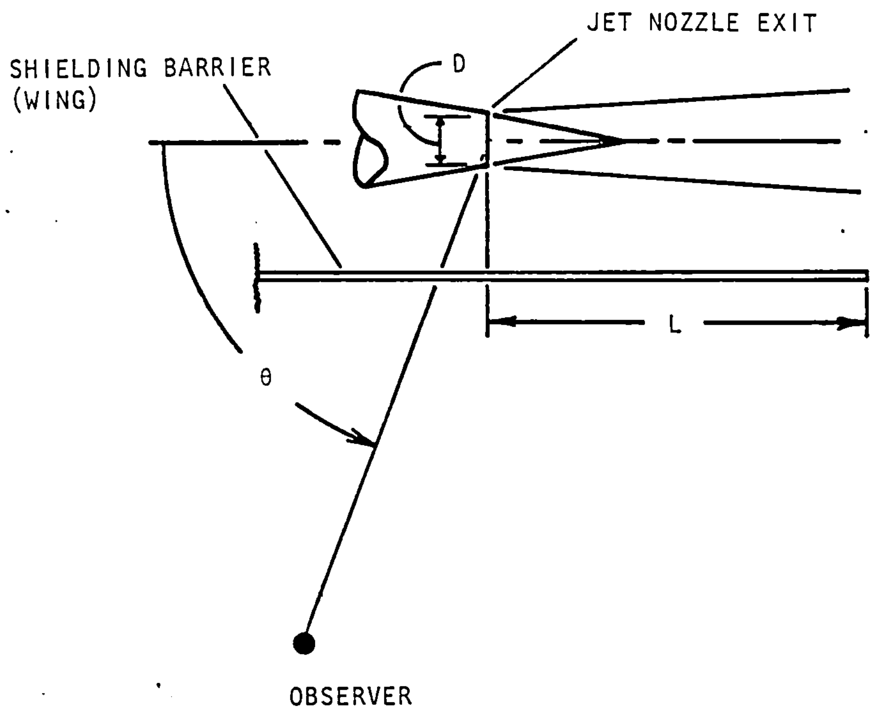


Figure 4.2.2 Geometry for jet mixing noise shielding.

5.0 COMPUTATION PROCEDURES FOR LFC SUCTION SYSTEMS

No separate noise prediction procedures are provided or recommended for the LFC suction systems since the procedures given for the aircraft propulsion system are also applicable to this system. Consequently, the procedures of section 2.1 through 2.6 should be used when it is desired to predict the compressor, turbine, combustion, jet mixing, or jet shock noise associated with the LFC suction system.

6.0 SAMPLE RESULTS FROM CHECKOUT CASES

This section presents some typical inputs and results obtained from computer runs made to checkout the computational algorithms presented previously. These results were checked against hand calculations or other available data. All check cases are for a single engine at cruise with no structural shielding. Table 6.1 presents the results of a check case for the fan/compressor noise prediction procedure. This case is for a single engine with an unsuppressed fan with no high Mach inlet effects included. Tables 6.2 and 6.3 present the results of one of the check cases run for turbine and core noise prediction checkout, respectively. Table 6.3 includes the predicted perceived wavelengths (at the receiver location) for the core noise. A jet mixing noise case is presented in Table 6.4. This case is for a turbofan engine with a separate flow exhaust nozzle with no plug in the primary nozzle. The receiver location is in the plane of the nozzle exits. Table 6.5 presents the jet shock associated broadband noise predicted for the input shown. In this case, the shock noise is generated by the secondary (fan) exhaust; no shocks are present in the primary flow.

Sample cases have also been run to checkout the computational procedures for the airframe noise sources. A check case for the trailing-edge noise prediction is shown in Table 6.6. This case represents a point at approximately 45 percent chord and 8 percent span on an LFC configuration having a wing span of about 73 meters (240 ft). Table 6.7 presents the results of one check case for the turbulent boundary layer noise prediction. A description of the input and schematic of the geometry are given in Table 6.7(a). The receiver location is on the underside of the wing (high-wing configuration) at about 3 meters (9.8 ft) spanwise from the side of the fuselage. The fuselage side is the only airframe surface that is expected to radiate significant turbulent boundary layer noise to the receiver location. This surface is represented by a flat plate 20 meters (65.6 ft) long by 3 meters (9.8 ft) wide. By inspection of the predicted noise levels for each of the 5 sub-areas used in this problem, it can be determined that noise contributions from locations forward or aft of the area selected are negligible.

TABLE 6.1

FAN NOISE PREDICTION

Input used:

ΔT_{STAGE}	= 81°R	ϕ_I	= 90°
$\dot{m}$	= 314.6 lbm/s	r_I	= 50 ft
N	= 4290 RPM	ϕ_D	= 90°
M_{T_R}	= 0.70	r_D	= 50 ft
$(M_{T_R})_D$	= 1.30	M_A	= 0.80
M_{T_A}	= 0.73	ρ_a	= 0.01883 lbm/f³
N_R	= 42	c_a	= 968 f/s
N_S	= 92	ρ_o	= 0.0765 lbm/f³
R_{SS}	= 380	c_o	= 1116 f/s

1/3 O.B. CENTER FREQUENCY (Hz)	PREDICTED 1/3 O.B. SOUND PRESSURE LEVELS (dB) INLET DUCT	DISCHARGE DUCT	TOTAL FAN	TOTAL FAN SPECTRUM LEVELS* (dB)
50	16.8	0.0	16.8	7.9
63	24.5	0.2	24.5	13.3
80	32.4	8.1	32.4	19.8
100	39.3	15.1	39.4	25.7
125	46.0	21.8	46.0	31.4
160	53.0	28.8	53.0	37.3
200	58.9	34.7	58.9	42.2
250	64.5	40.3	64.5	46.9
315	69.9	45.7	69.9	51.3
400	75.1	50.9	75.1	55.4
500	79.6	55.4	79.6	58.9
630	83.9	59.6	83.9	62.2
800	87.9	63.7	87.9	65.2
1000	91.3	67.1	91.3	67.7
1250	94.4	70.2	94.4	69.8
1600	97.4	73.2	97.4	71.7
2000	99.7	75.5	99.7	73.1
2500	101.7	77.5	101.7	74.1
3150	105.0	81.8	105.0	99.9
4000	104.7	80.5	104.8	75.1
5000	105.6	81.4	105.7	75.0
6300	108.6	85.7	108.6	104.9
8000	106.4	82.2	106.4	73.8
10000	107.6	84.3	107.6	101.9

*1 Hz bandwidth
overall sound pressure level = 114.9 dB

TABLE 6.2

TURBINE NOISE PREDICTION

Input used:

$PR = 8.48$

$N = 4290$ RPM

$D = 3.15$ ft

$A = 470$ in^2

$N_R = 127$

$M_A = 0.80$

$\rho_a = 0.01883$ lbm/f^3

$c_a = 968$ f/s

$\rho_o = 0.0765$ lbm/f^3

$c_o = 1116$ f/s

$r = 50$ ft

$\phi = 90°$

1/3 O.B. Center Frequency (Hz)	1/3 O.B. SPL (dB)	Spectrum Level* (dB)
50	64.9	54.3
63	66.2	54.6
80	67.9	55.2
100	69.9	56.3
125	71.5	56.9
160	73.8	58.1
200	76.4	59.7
250	78.2	60.5
315	80.2	61.6
400	82.2	62.5
500	84.4	63.8
630	86.2	64.6
800	88.6	65.9
1000	91.4	67.8
1250	93.2	68.6
1600	95.5	69.8
2000	97.6	70.9
2500	99.3	71.7
3150	100.8	72.2
4000	102.2	72.5
5000	103.4	72.8
6300	104.4	72.8
8000	105.4	72.7
10000	107.0	100.5

*1 Hz bandwidth
overall sound pressure level = 112.7 dB

TABLE 6.3

CORE NOISE PREDICTION

Input used:

W_3	= 29.5 lbm/s	ρ_o	= 0.0765 lbm/f^3
T_4	= 2355°F	c_a	= 968 f/s
T_3	= 972°F	c_o	= 1116 f/s
$(T_4-T_5)_{des}$	= 1529°F	M_A	= 0.80
ρ_3	= 0.3647 lbm/f^3	r	= 50 ft
ρ_a	= 0.01883 lbm/f^3	ϕ	= 90°

1/3 O.B. CENTER FREQUENCY (Hz)	1/3 O.B. SPL (dB)	SPECTRUM LEVEL* (dB)	PERCEIVED ** WAVELENGTH (1/ft)
50	62.0	51.3	0.54
63	66.0	54.3	0.68
80	70.0	57.3	0.87
100	73.0	59.3	1.08
125	73.3	58.7	1.35
160	76.3	60.6	1.73
200	78.8	62.2	2.16
250	77.1	59.5	2.70
315	78.6	60.0	3.41
400	79.6	59.9	4.33
500	78.6	57.9	5.41
630	77.1	55.4	6.81
800	75.1	52.4	8.65
1000	72.6	48.9	10.81
1250	69.6	45.0	13.52
1600	66.6	40.9	17.30
2000	63.6	36.9	21.63
2500	59.6	32.0	27.03
3150	55.6	27.0	34.06
4000	52.1	22.4	43.25
5000	48.1	17.4	54.06
6300	43.6	11.9	68.12
8000	39.6	6.9	86.50
10000	34.6	0.9	108.13

overall sound pressure level = 87.6 dB

* 1 Hz bandwidth

** measured in y direction, i.e. at ϕ = 90°, x component = 0.0

TABLE 6.4

JET MIXING NOISE PREDICTION

Input used:

D_J = 0.622 m	M_A = 0.80
V_J = 470 m/s	V_A = 236 m/s
M_J = 1.0	x = 0.0
T_J = 717°K	y = 7.62 m
ρ_a = 0.30156 kg/m³	x_s = 0.0
c_a = 295 m/s	R_W = 8.1
ρ_o = 1.225 kg/m³	D_S = 1.359 m
c_o = 340 m/s	V_S = 361 m/s
t_a = 217°K	T_S = 289°K
	AR = 3.8

1/3 O.B. CENTER FREQUENCY (Hz)	1/3 O.B. SOUND PRESSURE LEVELS (dB) PRIMARY JET	SECONDARY JET	TOTAL JET	TOTAL JET SPECTRUM LEVELS* (dB)
50	76.5	81.5	82.7	72.1
63	78.2	81.8	83.4	71.7
80	80.0	82.0	84.1	71.5
100	81.6	82.9	85.3	71.7
125	82.0	86.5	87.8	73.2
160	82.1	90.4	91.0	75.3
200	82.1	93.5	93.8	77.1
250	81.6	94.2	94.4	76.8
315	80.8	94.9	95.1	76.5
400	80.0	95.5	95.6	76.0
500	80.6	95.4	95.5	74.9
630	81.6	95.3	95.4	73.8
800	82.5	95.0	95.3	72.6
1000	82.8	94.4	94.7	71.1
1250	83.0	93.8	94.2	69.6
1600	83.1	93.1	93.5	67.8
2000	82.8	92.2	92.6	66.0
2500	82.3	91.2	91.7	64.1
3150	81.8	90.2	90.8	62.2
4000	81.0	89.2	89.8	60.1
5000	80.1	88.2	88.8	58.2
6300	79.2	87.2	87.8	56.2
8000	78.2	86.2	86.8	54.1
10000	77.2	85.2	85.8	52.2

*1 Hz bandwidth
overall sound pressure level = 106.1 dB

TABLE 6.5

JET SHOCK ASSOCIATED BROADBAND NOISE PREDICTION

Input used:

V_J = 401 m/s

M_J = 1.223

D_e = 1.21 m

H_J = 0.3305 m

M_A = 0.80

c_a = 295 m/s

ρ_a = 0.30156 kg/m^3

c_o = 340 m/s

ρ_o = 1.225 kg/m^3

r = 3.5 m

ϕ = 90°

Note: This input applies to the secondary exhaust flow where the annulus height = 0.3305 m (H_J). There are no shocks in the primary flow.

1/3 O.B. CENTER FREQUENCY (Hz)	1/3 O.B. SPL (dB)	SPECTRUM LEVEL* (dB)
50	102.7	92.0
63	106.5	94.9
80	110.2	97.6
100	113.4	99.8
125	116.2	101.6
160	118.6	102.9
200	120.0	103.3
250	120.8	103.2
315	122.5	103.9
400	128.0	108.3
500.	136.6	115.9
630	142.7	121.0
800	143.4	120.8
1000	142.0	118.4
1250	143.1	118.5
1600	142.9	117.2
2000	141.4	114.8
2500	140.8	113.1
3150	139.6	111.0
4000	138.5	108.8
5000	137.3	106.7
6300	136.0	104.4
8000	134.8	102.1
10000	133.5	99.9

*1 Hz bandwidth
overall sound pressure level = 152 dB

TABLE 6.6

TRAILING-EDGE NOISE PREDICTION

Input used:

$M_A = 0.80$	$\rho_a = 0.30156$ kg/m^3
$V_A = 236$ m/s	$c_a = 295$ m/s
$\ell = 6.0$ m	$\rho_o = 1.225$ kg/m^3
$r = 5.0$ m	$c_o = 340$ m/s
$\phi = 0°$	$\mu_a = 1.4217\times10^{-5}$ N·s/m^2
	$\bar{c} = 9.1$ m

1/3 O.B. CENTER FREQUENCY (Hz)	1/3 O.B. SPL (dB)	SPECTRUM LEVEL* (dB)
50	101.5	90.9
63	104.5	92.9
80	107.2	94.5
100	109.3	95.7
125	111.0	96.4
160	112.4	96.7
200	113.0	96.4
250	→ 113.3	95.6
315	113.0	94.4
400	112.4	92.7
500	111.5	90.8
630	110.2	88.6
800	108.8	86.1
1000	107.2	83.6
1250	105.6	81.0
1600	103.7	78.0
2000	101.9	75.2
2500	100.0	72.4
3150	98.1	69.5
4000	96.1	66.4
5000	94.2	63.6
6300	92.2	60.6
8000	90.2	57.5
10000	88.2	54.6

* 1 Hz bandwidth
overall sound pressure level = 122.4 dB

TABLE 6.7

TURBULENT BOUNDARY LAYER NOISE PREDICTION

(a) Description of Geometry and Program Input

	j=1	j=2	j=3	j=4	j=5
A_j (m^2)	12.0	12.0	12.0	12.0	12.0
x_j (m)	-8.0	-4.0	0.0	4.0	8.0
y_j (m)	3.0	3.0	3.0	3.0	3.0
z_j (m)	1.5	1.5	1.5	1.5	1.5
ℓ_j (m)	23.0	27.0	31.0	35.0	39.0

$U = 236$ m/s

$M_A = 0.80$

$\rho_a = 0.30156$ kg/m^3

$c_a = 295$ m/s

$\rho_o = 1.225$ kg/m^3

$c_o = 340$ m/s

$\mu_a = 1.4217 \times 10^{-5}$ N$\cdot$s/m^2

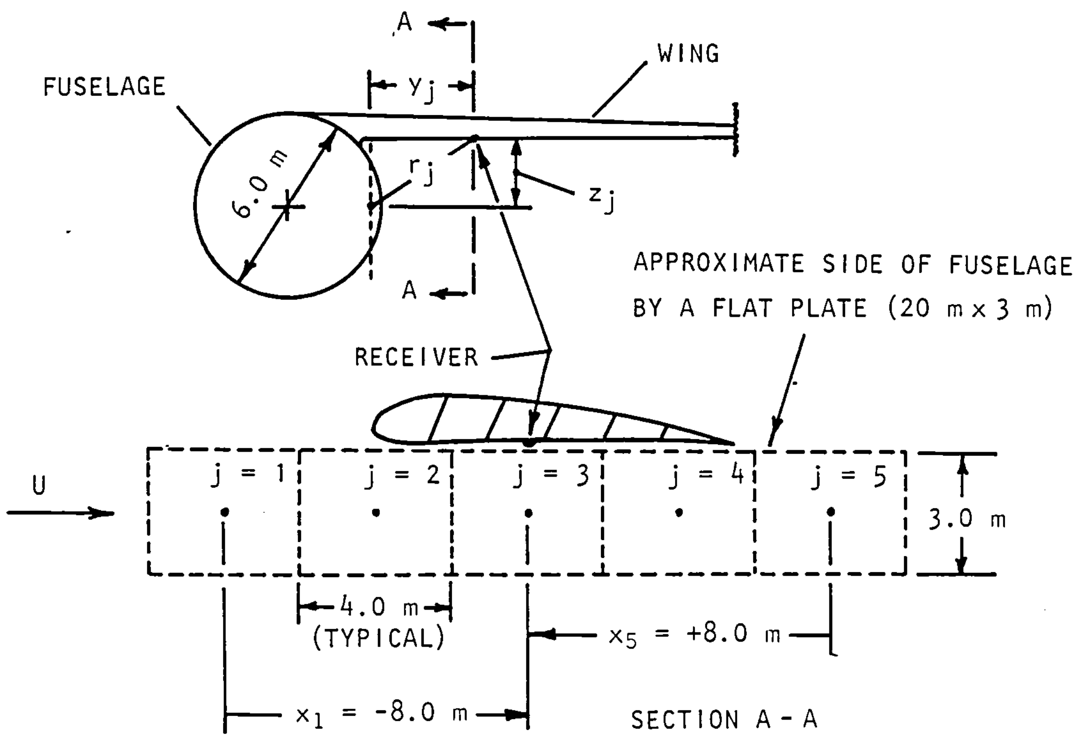


TABLE 6.7 — Continued

TURBULENT BOUNDARY LAYER NOISE PREDICTION

(b) Predicted Noise Levels

1/3 O.B. CENTER FREQUENCY (Hz)	1/3 O.B. SPL (dB)	SPECTRUM LEVEL* (dB)
50	85.5	74.9
63	86.6	75.0
80	87.6	74.9
100	85.7	72.0
125	83.5	68.8
160	81.3	65.6
200	79.1	62.4
250	76.9	59.2
315	74.7	56.0
400	72.5	52.8
500	70.3	49.6
630	68.1	46.4
800	65.9	43.2
1000	63.7	40.0
1250	61.5	36.8
1600	59.3	33.6
2000	57.1	30.4
2500	54.9	27.2
3150	52.7	24.0
4000	50.5	20.8
5000	48.3	17.6
6300	46.1	14.4
8000	43.9	11.2
10000	41.7	8.0

*1 Hz bandwidth
overall sound pressure level = 93.6 dB

1. Report No. NASA CR-159105	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle NEAR-FIELD NOISE PREDICTION FOR AIRCRAFT IN CRUISING FLIGHT — METHODS MANUAL		5. Report Date AUGUST 1979
		6. Performing Organization Code
7. Author(s) J. G. TIBBETTS		8. Performing Organization Report No. LG78ER0219
		10. Work Unit No.
9. Performing Organization Name and Address LOCKHEED-GEORGIA COMPANY MARIETTA, GEORGIA 30063		11. Contract or Grant No. NAS1-14946
		13. Type of Report and Period Covered FINAL REPORT May 1977 - June 1978
12. Sponsoring Agency Name and Address NATIONAL AERONAUTICS AND SPACE ADMINISTRATION WASHINGTON, D. C. 20546		14. Sponsoring Agency Code

15. Supplementary Notes

CONTRACT MONITOR — D. L. LANSING, NASA, LANGLEY RESEARCH CENTER HAMPTON, VIRGINIA 23665
PROGRAM MANAGER — J. S. GIBSON, LOCKHEED-GEORGIA COMPANY, MARIETTA, GEORGIA 30063

16. Abstract

A methods manual for the prediction of near-field noise at cruise has been formulated from existing technology. The procedures presented are considered to be the best available from current and evolving technology. Discussions of the methods and their selection are presented in Companion Vol., CR-159104. Computational algorithms for the prediction of near-field noise at cruise are presented for each aircraft noise source considered significant (particularly for the application of laminar flow control criteria).

17. Key Words (Suggested by Author(s)) AIRCRAFT NOISE, NEAR-FIELD NOISE, AIRCRAFT NOISE PREDICTION, PROPULSION SYSTEM NOISE, HIGH FORWARD SPEED EFFECTS, AIRFRAME NOISE	18. Distribution Statement UNLIMITED

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
UNCLASSIFIED	UNCLASSIFIED	93	

* For sale by the National Technical Information Service, Springfield, Virginia 22161