

NASA
Reference
Publication
1218

July 1989

Airfoil Self-Noise and Prediction

Thomas F. Brooks,
D. Stuart Pope,
and Michael A. Marcolini

(NASA-RE-1218) AIRFOIL SELF-NOISE AND
PREDICTION (NASA Langley Research Center)
145 P CSCL 20A

N89-25673

H1/71 Unclas
0197717

NASA

**NASA
Reference
Publication
1218**

1989

**Airfoil Self-Noise
and Prediction**

Thomas F. Brooks
*Langley Research Center
Hampton, Virginia*

D. Stuart Pope
*PRC Kentron, Inc.
Aerospace Technologies Division
Hampton, Virginia*

Michael A. Marcolini
*Langley Research Center
Hampton, Virginia*



National Aeronautics and
Space Administration
Office of Management
Scientific and Technical
Information Division

Contents

Summary	1
1. Introduction	2
1.1. Noise Sources and Background	2
1.1.1. Turbulent-Boundary-Layer-Trailing-Edge (TBL-TE) Noise	2
1.1.2. Separation-Stall Noise	3
1.1.3. Laminar-Boundary-Layer-Vortex-Shedding (LBL-VS) Noise	3
1.1.4. Tip Vortex Formation Noise	4
1.1.5. Trailing-Edge-Bluntness-Vortex-Shedding Noise	4
1.2. Overview of Report	4
2. Description of Experiments	5
2.1. Models and Facility	5
2.2. Instrumentation	5
2.3. Test Conditions	5
2.4. Wind Tunnel Corrections	5
3. Boundary-Layer Parameters at the Trailing Edge	9
3.1. Scaled Data	9
3.2. Calculation Procedures	9
4. Acoustic Measurements	15
4.1. Source Identification	15
4.2. Correlation Editing and Spectral Determination	15
4.3. Self-Noise Spectra	17
5. Spectral Scaling	51
5.1. Turbulent-Boundary-Layer-Trailing-Edge Noise and Separated Flow Noise	51
5.1.1. Scaled Data	51
Zero angle of attack	51
Nonzero angle of attack	54
5.1.2. Calculation Procedures	59
5.1.3. Comparison With Data	62
5.2. Laminar-Boundary-Layer-Vortex-Shedding Noise	62
5.2.1. Scaled Data	63
5.2.2. Calculation Procedures	66
5.2.3. Comparison With Data	71
5.3. Tip Vortex Formation Noise	71
5.3.1. Calculation Procedures	71
5.3.2. Comparison With Data	73
5.4. Trailing-Edge-Bluntness-Vortex-Shedding Noise	73
5.4.1. Experiment	73

5.4.2. Scaled Data	74
5.4.3. Calculation Procedures	78
5.4.4. Comparison With Data	81
6. Comparison of Predictions With Published Results	83
6.1. Study of Schlinker and Amiet	83
6.1.1. Boundary-Layer Definition	83
6.1.2. Trailing-Edge Noise Measurements and Predictions	83
6.2. Study of Schlinker	88
6.3. Study of Fink, Schlinker, and Amiet	88
7. Conclusions	99
Appendix A—Data Processing and Spectral Determination	100
Appendix B—Noise Directivity	105
Appendix C—Application of Predictions to a Rotor Broadband Noise Test	108
Appendix D—Prediction Code	112
References	133
Tables	134

Symbols

A	spectral shape function for TBL-TE noise, eqs. (35) and (36)	\mathcal{L}	turbulence correlation scale, m
a, a_0	parameters of shape function A , eqs. (37) and (38)	ℓ	spanwise extent of tip vortex at TE, m
B	spectral shape function for separation noise, eqs. (41) and (42)	M	Mach number, U/c_o
b, b_0	parameters of shape function B , eqs. (43) and (44)	M_{\max}	maximum Mach number in tip vortex formation region
c	chord length, m	R_c	Reynolds number based on chord length, cU/ν
c_o	medium speed of sound, m/s	$R_{ij}(\tau)$	cross-correlation between microphones M_i and M_j , Pa^2
\bar{D}_h	directivity function for TE noise (high-frequency limit), eq. (B1)	r	distance from source to observer, m
\bar{D}_ℓ	directivity function for translating dipole (low-frequency limit), eq. (B2)	$S(f)$	spectrum of self-noise, Pa^2/Hz
$F(\text{St})$	universal spectral shape function, eq. (18)	$\text{St}, \text{St}_1, \bar{\text{St}}_1, \text{St}_2$	Strouhal numbers defined for TBL-TE and separation noise scaling, section 5.1.
f	frequency, Hz	St', St'_1	Strouhal numbers defined for LBL-VS noise scaling, section 5.2.
G_1	spectral shape function for LBL-VS noise, eq. (57)	St''	Strouhal number defined for tip vortex formation noise, section 5.3.
G_2	R_c -dependence for LBL-VS noise peak amplitude, eq. (58)	St'''	Strouhal number defined for TE-bluntness-vortex-shedding noise, section 5.4.
G_3	angle dependence for G_2 function, eq. (60)	t	time, s
G_4	peak level function for G_5 , eq. (74)	U	free-stream velocity, m/s
G_5	spectral shape function for TE bluntness noise, eqs. (75)–(82)	U_c	convection velocity, m/s
$G_{ij}(f)$	cross-spectrum between microphone M_i and M_j , Pa^2/Hz	x	streamwise axis, see fig. B3, m
H	tunnel height, m	y	lateral axis, m
h	TE thickness (degree of bluntness), m	z	vertical axis, m
$K, K_1, \Delta K_1, K_2$	constants, defined by eqs. (18), (47), (48), and (49)	α_{TIP}	angle of attack of airfoil tip to oncoming flow, deg
L	span, m	α'_{TIP}	corrected angle of attack of airfoil tip, eq. (66), deg
L'	sectional lift of blade, lift per unit span	α_t	airfoil angle of attack referenced to tunnel streamwise axis, deg
		α_*	effective aerodynamic angle of attack, corrected for open wind tunnel effects, deg
		δ	boundary-layer thickness, m
		δ^*	boundary-layer displacement thickness, m

Γ	tip vortex strength, m^2/s	0	for δ_0 , δ_0^* , and θ_0 is for airfoil at zero angle of attack, reference value
Θ	angle from source streamwise axis x to observer, see fig. B3, deg	1/3	1/3-octave spectral presentation
θ	boundary-layer momentum thickness, m	Abbreviations:	
ν	kinematic viscosity of medium, m^2/s	BL	boundary layer
τ	time delay, s	LBL	laminar boundary layer
Φ	angle from source lateral axis y to observer, see fig. B3, deg	LE	leading edge of airfoil blade
ϕ	cross-spectral phase angle, deg	LHS	left-hand side
Ψ	angle parameter related to surface slope at TE, section 5.4, deg	M_i	microphone number i for $i = 1$ through 9, see fig. 4
Subscripts:		OASPL	overall sound pressure level, dB
avg	average	RHS	right-hand side
e	retarded coordinate	SPL	sound pressure level, spectrum, dB (re 2×10^{-5} Pa)
p	pressure side of airfoil	TBL	turbulent boundary layer
s	suction side of airfoil	TE	trailing edge of airfoil blade
TIP	tip of blade	UTRC	United Technologies Research Center
TOT	total	VS	vortex shedding
α	angle dependent	2D	two-dimensional
		3D	three-dimensional

Summary

An overall prediction method has been developed for the self-generated noise of an airfoil blade encountering smooth flow. Prediction methods for individual self-noise mechanisms are semiempirical and are based on previous theoretical studies and the most comprehensive self-noise data set available. The specially processed data set, most of which is newly presented in this report, is from a series of aerodynamic and acoustic tests of two- and three-dimensional airfoil blade sections conducted in an anechoic wind tunnel. Five self-noise mechanisms due to specific boundary-layer phenomena have been identified and modeled: boundary-layer turbulence passing the trailing edge, separated-boundary-layer and stalled-airfoil flow, vortex shedding due to laminar-boundary-layer instabilities, vortex shedding from blunt trailing edges, and the turbulent vortex flow existing near the tips of lifting blades. The data base, with which the predictions are matched, is from seven NACA 0012 airfoil blade sections of different sizes (chord lengths from 2.5 to 61 cm) tested at wind tunnel speeds up to Mach 0.21 (Reynolds number based on chord up to 3×10^6) and at angles of attack from 0° to 25.2° . The predictions are compared successfully with published data from three self-noise studies of different airfoil shapes, which were tested up to Mach and Reynolds numbers of 0.5 and 4.6×10^6 , respectively. An application of the prediction method is reported for a large-scale-model helicopter rotor and the predictions compared well with data from a broadband noise test of the rotor, conducted in a large anechoic wind tunnel. A computer code of the methods is given for the predictions of 1/3-octave formatted spectra.

1. Introduction

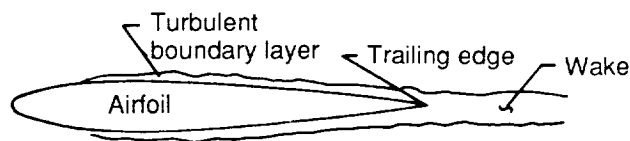
Airfoil self-noise is due to the interaction between an airfoil blade and the turbulence produced in its own boundary layer and near wake. It is the total noise produced when an airfoil encounters smooth nonturbulent inflow. Over the last decade, research has been conducted at and supported by NASA Langley Research Center to develop fundamental understanding, as well as prediction capability, of the various self-noise mechanisms. The interest has been motivated by its importance to broadband helicopter rotor, wind turbine, and airframe noises. The present paper is the cumulative result of a series of aerodynamic and acoustic wind tunnel tests of airfoil sections, which has produced a comprehensive data base. A correspondingly extensive semi-empirical scaling effort has produced predictive capability for five self-noise mechanisms.

1.1. Noise Sources and Background

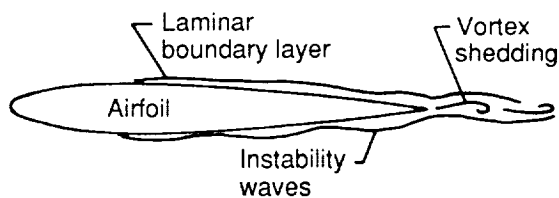
Previous research efforts (prior to 1983) for the broadband noise mechanisms are reviewed in some detail by Brooks and Schlinker (ref. 1). In figure 1, the subsonic flow conditions for five self-noise mechanisms of concern here are illustrated. At high Reynolds number R_c (based on chord length), turbulent boundary layers (TBL) develop over most of the airfoil. Noise is produced as this turbulence passes over the trailing edge (TE). At low R_c , largely laminar boundary layers (LBL) develop, whose instabilities result in vortex shedding (VS) and associated noise from the TE. For nonzero angles of attack, the flow can separate near the TE on the suction side of the airfoil to produce TE noise due to the shed turbulent vorticity. At very high angles of attack, the separated flow near the TE gives way to large-scale separation (deep stall) causing the airfoil to radiate low-frequency noise similar to that of a bluff body in flow. Another noise source is vortex shedding occurring in the small separated flow region aft of a blunt TE. The remaining source considered here is due to the formation of the tip vortex, containing highly turbulent flow, occurring near the tips of lifting blades or wings.

1.1.1. Turbulent-Boundary-Layer-Trailing-Edge (TBL-TE) Noise

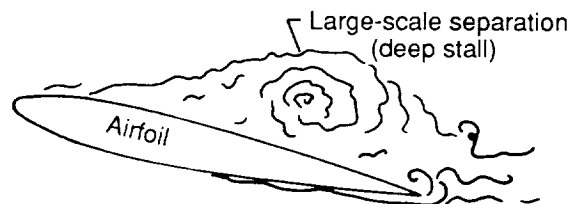
Using measured surface pressures, Brooks and Hodgson (ref. 2) demonstrated that if sufficient information is known about the TBL convecting surface pressure field passing the TE, then TBL-TE noise can be accurately predicted. Schlinker and Amiet (ref. 3) employed a generalized empirical description of surface pressure to predict measured noise. However, the lack of agreement for many cases indicated



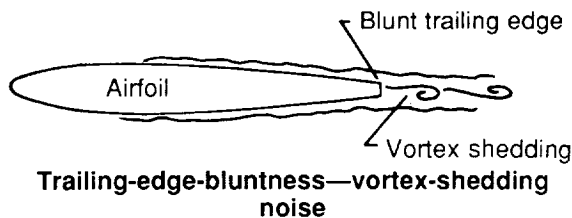
Turbulent-boundary-layer—trailing-edge noise



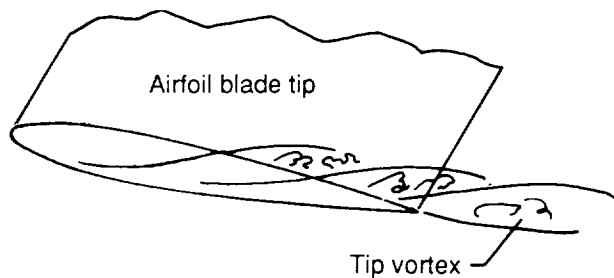
Laminar-boundary-layer—vortex-shedding noise



Separation-stall noise



Trailing-edge-bluntness—vortex-shedding noise



Tip vortex formation noise

Figure 1. Flow conditions producing airfoil blade self-noise.

a need for a more accurate pressure description than was available. Langley supported a research effort (ref. 4) to model the turbulence within boundary layers as a sum of discrete “hairpin” vortex elements. In a parallel and follow-up effort, the present authors matched measured and calculated mean boundary-layer characteristics to prescribed distributions of the discrete vortex elements so that associated surface pressure could be determined. The use of the model to predict TBL-TE noise proved disappointing because of its inability to show correct trends with angle of attack or velocity. The results showed that to successfully describe the surface pressure, the history of the turbulence must be accounted for in addition to the mean TBL characteristics. This level of turbulence modeling has not been attempted to date.

A simpler approach to the TBL-TE noise problem is based on the Ffowcs Williams and Hall (ref. 5) edge-scatter formulation. In reference 3, the noise data were normalized by employing the edge-scatter model with the mean TBL thickness δ used as a required length scale. When δ was unknown, simple flat plate theory was used to estimate δ . Spectral data initially differing by 40 dB collapsed to within 7 dB, consistent with the results of the approach discussed above using surface pressure models. The extent of agreement between data sets was largely due to the correct scaling of the velocity dependence, which is the most sensitive parameter in the scaling approach. The dependence of the overall sound pressure level on velocity to the fifth power had been verified in a number of studies. The extent to which the normalized data deviation was due to uncertainty in δ was addressed by Brooks and Marcolini (ref. 6) in a forerunner to the present report. For large R_c and small angles of attack, which matched the conditions of reference 3, the use of measured TBL thicknesses δ , displacement thicknesses δ^* , or momentum thicknesses θ in the normalization produced the same degree of deviation within the TBL-TE noise data. Subsequently, normalizations based on boundary-layer maximum shear stress measurements and, alternately, profile shape factors were also examined. Of particular concern in reference 6 was that when an array of model sizes, rather than just large models, was tested at various angles of attack, the normalized spectrum deviations increased to 10 or even 20 dB. These large deviations indicate a lack of fidelity of the spectrum normalization and any subsequent prediction methods based on curve fits. They also reinforce the conclusion from the aforementioned surface pressure modeling effort that knowledge of the mean TBL characteristics alone is insufficient to define the turbulence structure. The conditions under which the turbulence evolves were

found to be important. The normalized data appeared to be directly influenced by factors such as Reynolds number and angle of attack, which in previous analyses were assumed to be of pertinence only through their effect on TBL thickness δ (refs. 3 and 7).

Several prediction schemes for TBL-TE noise have been used previously for helicopter rotor noise (refs. 3 and 8) and for wind turbines (refs. 9 and 10). These schemes have all evolved from scaling law equations which were fitted to the normalized data of reference 3 and, thus, are limited by the same concerns of generality discussed above.

1.1.2. Separation-Stall Noise

Assessments of the separated flow noise mechanism for airfoils at moderate to high angles of attack have been very limited (ref. 1). The relative importance of airfoil stall noise was illustrated in the data of Fink and Bailey (ref. 11) in an airframe noise study. At stall, noise increased by more than 10 dB relative to TBL-TE noise, emitted at low angles of attack. Paterson et al. (ref. 12) found evidence through surface to far field cross-correlations that for mildly separated flow the dominant noise is emitted from the TE, whereas for deep stall the noise radiated from the chord as a whole. This finding is consistent with the conclusions of reference 11.

No predictive methods are known to have been developed. A successful method would have to account for the gradual introduction of separated flow noise as airfoil angle of attack is increased. Beyond limiting angles, deep stall noise would be the only major contributing source.

1.1.3. Laminar-Boundary-Layer-Vortex-Shedding (LBL-VS) Noise

When a LBL exists over most of at least one side of an airfoil, vortex shedding noise can occur. The vortex shedding is apparently coupled to acoustically excited aerodynamic feedback loops (refs. 13, 14, and 15). In references 14 and 15, the feedback loop is taken between the airfoil TE and an upstream “source” point on the surface, where Tollmien-Schlichting instability waves originate in the LBL. The resulting noise spectrum is composed of quasi-tones related to the shedding rates at the TE. The gross trend of the frequency dependence was found by Paterson et al. (ref. 16) by scaling on a Strouhal number basis with the LBL thickness at the TE being the relevant length scale. Simple flat plate LBL theory was used to determine the boundary-layer thicknesses δ in the frequency comparisons. The use of measured values of δ in reference 6 verified the general Strouhal dependence. Additionally,

for zero angle of attack, Brooks and Marcolini (ref. 6) found that overall levels of LBL-VS noise could be normalized so that the transition from LBL-VS noise to TBL-TE noise is a unique function of R_c .

There have been no LBL-VS noise prediction methods proposed, because most studies have emphasized the examination of the rather erratic frequency dependence of the individual quasi-tones in attempts to explain the basic mechanism. However, the scaling successes described above in references 6 and 16 can offer initial scaling guidance for the development of predictions in spite of the general complexity of the mechanism.

1.1.4. Tip Vortex Formation Noise

The tip noise source has been identified with the turbulence in the local separated flow associated with formation of the tip vortex (ref. 17). The flow over the blade tip consists of a vortex with a thick viscous turbulent core. The mechanism for noise production is taken to be TE noise due to the passage of the turbulence over the TE of the tip region. George and Chou (ref. 8) proposed a prediction model based on spectral data from delta wing studies (assumed to approximate the tip vortex flow of interest), mean flow studies of several tip shapes, and TE noise analysis.

Brooks and Marcolini (ref. 18) conducted an experimental study to isolate tip noise in a quantitative manner. The data were obtained by comparing sets of two- and three-dimensional test results for different model sizes, angles of attack, and tunnel flow velocities. From data scaling, a quantitative prediction method was proposed which had basic consistency with the method of reference 8.

1.1.5. Trailing-Edge-Bluntness-Vortex-Shedding Noise

Noise due to vortex shedding from blunt trailing edges was established by Brooks and Hodgson (ref. 2) to be an important airfoil self-noise source. Other studies of bluntness effects, as reviewed by Blake (ref. 19) and Brooks and Schlinker (ref. 1), were only aerodynamic in scope and dealt with TE thicknesses that were large compared with the boundary-layer

displacement thicknesses. For rotor blade and wing designs, the bluntness is likely to be small compared with boundary-layer thicknesses.

Grosveld (ref. 9) used the data of reference 2 to obtain a scaling law for TE bluntness noise. He found that the scaling model could explain the spectral behavior of high-frequency broadband noise of wind turbines. Chou and George (ref. 20) followed suit with an alternative scaling of the data of reference 2 to model the noise. For both modeling techniques neither the functional dependence of the noise on boundary-layer thickness (as compared with the TE bluntness) nor the specifics of the blunted TE shape were incorporated. A more general model is needed.

1.2. Overview of Report

The purpose of this report is to document the development of a self-noise prediction method and to verify its accuracy for a range of applications. The tests producing the data base for the scaling effort are described in section 2. In section 3, the measured boundary-layer thickness and integral parameter data, used to normalize airfoil noise data, are documented. The acoustic measurements are reported in section 4, where a special correlation editing procedure is used to extract clean self-noise spectra from data containing extraneous test rig noise. In section 5, the scaling laws are developed for the five self-noise mechanisms. For each, the data are first normalized by fundamental techniques and then examined for dependences on parameters such as Reynolds number, Mach number, and geometry. The resulting prediction methods are delineated with specific calculation procedures and results are compared with the original data base. The predictions are compared in section 6 with self-noise data from three studies reported in the literature. In appendix A, the data processing technique is detailed; in appendix B, the noise directivity functions are defined; and in appendix C, an application of the prediction methods is reported for a helicopter rotor broadband noise study. In appendix D, a computer code of the prediction method is given.

2. Description of Experiments

The details of the measurements and test facility have been reported in reference 6 for the sharp TE two-dimensional (2D) airfoil model tests, in reference 18 for corresponding three-dimensional (3D) tests, and in reference 2 for the blunt TE 2D airfoil model test. Specific information applicable to this report is presented here.

2.1. Models and Facility

The models were tested in the low-turbulence potential core of a free jet located in an anechoic chamber. The jet was provided by a vertically mounted nozzle with a rectangular exit with dimensions of 30.48×45.72 cm. The 2D sharp TE models are shown in figure 2. The models, all of 45.72-cm span, were NACA 0012 airfoils with chord lengths of 2.54, 5.08, 10.16, 15.24, 22.86, and 30.48 cm. The models were made with very sharp TE, less than 0.05 mm thick, without beveling the edge. The slope of the surface near the uncupped TE corresponded to the required 7° off the chord line. The sharp TE 3D models, shown in figure 3, all had spans of 30.48 cm and chord lengths that were the same as the five largest 2D models. The 3D models had rounded tips, defined by rotating the NACA 0012 shape about the chord line at 30.48-cm span. An NACA 0012 model of pertinence to the present paper, which is not shown here, is the blunt-TE airfoil of reference 2, with a chord length of 60.96 cm. Details of the blunt TE of this large model are given in section 5.

The cylindrical hubs, shown attached to the models, provided support and flush-mounting on the side plates of the test rig. At a geometric tunnel angle of attack α_t of 0° , the TE of all models was located 61.0 cm above the nozzle exit. The tunnel angle α_t is referenced to the undisturbed tunnel streamline direction. In figure 4, an acoustic test configuration for a 3D model is shown. A 3D setup is shown so that the model can be seen fitted to the side plate. The side plates ($152.4 \times 30.0 \times 1$ cm) were reinforced and flush mounted on the nozzle lips. For the 2D configurations, an additional side plate was used.

2.2. Instrumentation

For all of the acoustic testing, eight 1.27-cm-diameter (1/2-in.) free-field-response microphones were mounted in the plane perpendicular to the 2D model midspan. One microphone was offset from this midspan plane. In figure 4, seven of these are shown with the identification numbers indicated. Microphones M1 and M2 were perpendicular to the chord line at the TE for $\alpha_t = 0^\circ$. The other microphones shown were at radii of 122 cm from the TE, as with M1 and M2, but were positioned

30° forward (M4 and M7) and 30° aft (M5 and M8). The data acquisition and processing approaches are described in appendix A.

For the aerodynamic tests the microphones to the right in figure 4 were removed and replaced by a large three-axis computer-controlled traverse rig used to position hot-wire probes. The miniature probes included both cross-wire and single-wire configurations. In figure 5, a cross-wire probe is shown mounted on the variable-angle arm of the traverse rig. Again, for clarity, a 3D airfoil model is shown. The probes were used to survey the flow fields about the models, especially in the boundary-layer and near-wake region just downstream of the trailing edge.

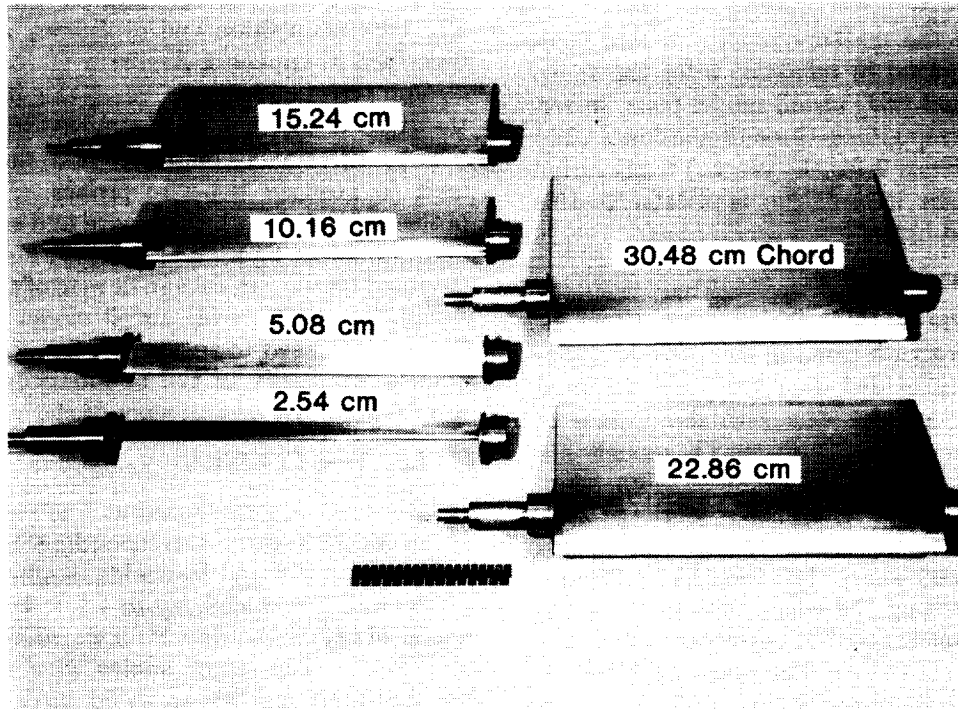
2.3. Test Conditions

The models were tested at free-stream velocities U up to 71.3 m/s, corresponding to Mach numbers up to 0.208 and Reynolds numbers, based on a 30.48-cm-chord model, up to 1.5×10^6 . The tunnel angles of attack α_t were 0° , 5.4° , 10.8° , 14.4° , 19.8° , and 25.2° . The larger angles were not attempted for the larger models to avoid large uncorrectable tunnel flow deflections. For the 22.86-cm- and 30.48-cm-chord models, α_t was limited to 19.8° and 14.4° , respectively.

For the untripped BL cases (natural BL development), the surfaces were smooth and clean. For the tripped BL cases, BL transition was achieved by a random distribution of grit in strips from the leading edge (LE) to 20 percent chord. This tripping is considered heavy because the chordwise extent of the strip produced thicker than normal BL thicknesses. It was used to establish a well-developed TBL even for the smaller models and at the same time retain geometric similarity. The commercial grit number was No. 60 (nominal particle diameter of 0.29 mm) with an application density of about 380 particles/cm². An exception was the 2.54-cm-chord airfoil which had a strip at 30 percent chord of No. 100 grit with a density of about 690 particles/cm².

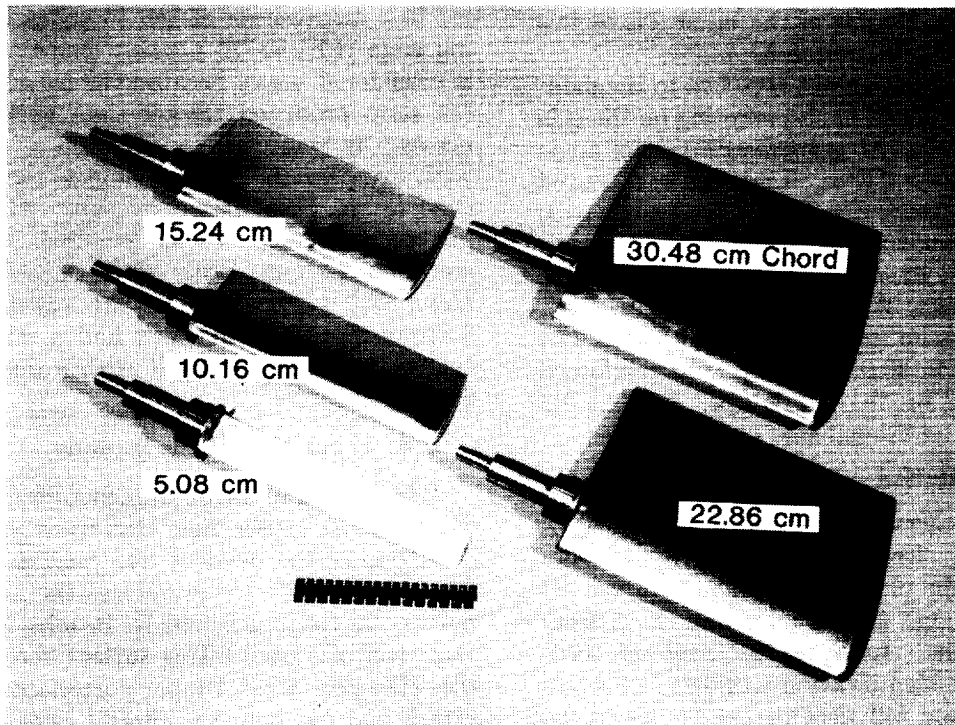
2.4. Wind Tunnel Corrections

The testing of airfoil models in a finite-size open wind tunnel causes flow curvature and downwash deflection of the incident flow that do not occur in free air. This effectively reduces the angle of attack, more so for the larger models. Brooks, Marcolini, and Pope (ref. 21) used lifting surface theory to develop the 2D open wind tunnel corrections to angle of attack and camber. Of interest here is a corrected angle of attack α_* representing the angle in free air required to give the same lift as α_t would give in the tunnel. One has from reference 21, upon ignoring



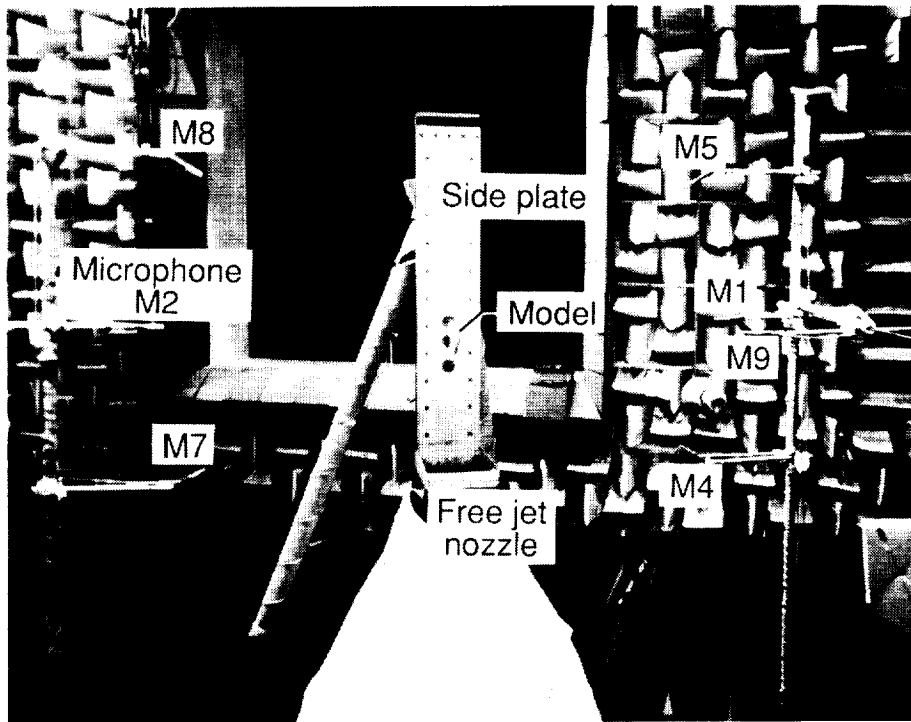
L-82-4573

Figure 2. Two-dimensional NACA 0012 airfoil blade models.



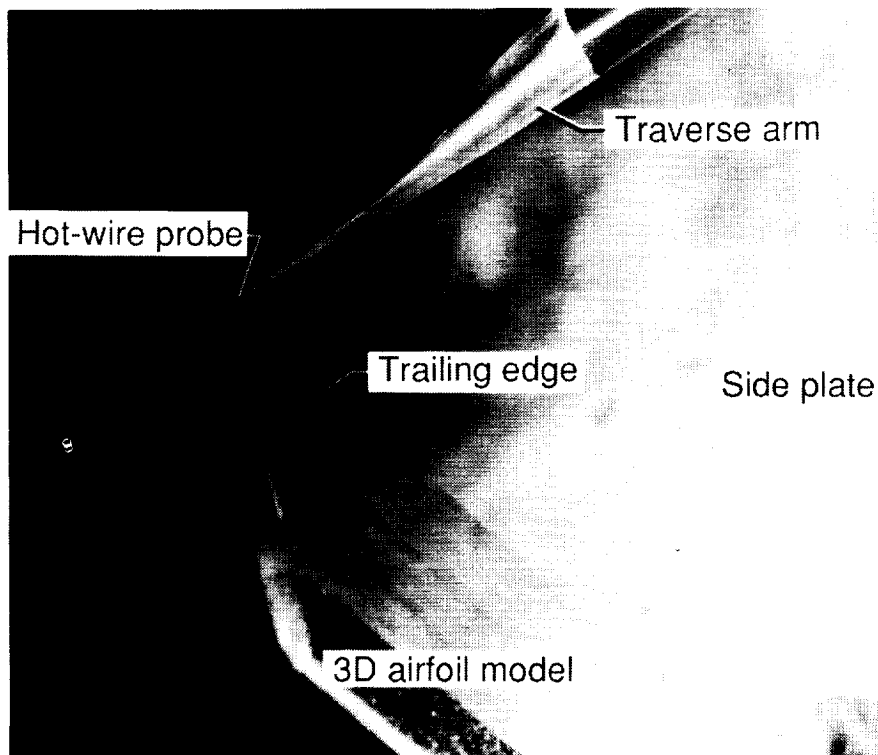
L-82-4570

Figure 3. Three-dimensional NACA 0012 airfoil blade models.



L-89-42

Figure 4. Test setup for acoustic tests of a 3D model airfoil.



L-89-43

Figure 5. Tip survey using hot-wire probe.

small camber effects,

$$\alpha_* = \alpha_t / \zeta \quad (1)$$

where

$$\zeta = (1 + 2\sigma)^2 + \sqrt{12\sigma}$$

and

$$\sigma = (\pi^2/48)(c/H)^2$$

The term c is the airfoil chord and H is the tunnel height or vertical open jet dimension for a horizontally aligned airfoil. For the present 2D configurations, α_*/α_t equals 0.88, 0.78, 0.62, 0.50, 0.37, and 0.28 for the models with chord lengths of 2.54, 5.08, 10.16, 15.24, 22.86, and 30.48 cm, respectively.

3. Boundary-Layer Parameters at the Trailing Edge

The purpose of this section is to present measured boundary-layer thicknesses from reference 21 and to document corresponding curve fit scaling equations to be employed in the normalization of the airfoil self-noise data.

The data presented are the result of hot-wire probe measurements made in the boundary-layer/near-wake region of the sharp TE of the 2D airfoil models. The probes were traversed perpendicular to the model chord lines downstream of the TE. These measurements were made at 0.64 mm from the TE for the 2.54-cm-chord airfoil and at 1.3 mm for the other airfoils. The integral BL parameters—displacement thickness δ^* and momentum thickness θ —were calculated from mean velocity profiles with the BL/near-wake thickness δ specified. The thickness δ is that distance from the airfoil surface where the mean velocity reaches 99 percent of the potential flow stream velocity. The values of δ were chosen by carefully examining the respective turbulent velocity and Reynolds stress distributions as well as the mean profiles. For all cases, the estimated accuracy of δ is within ± 5 percent for the turbulent-boundary-layer (TBL) flow and ± 10 percent for the laminar and transitional flows, whereas the error range for the integral thicknesses δ^* and θ is less (ref. 21).

3.1. Scaled Data

The thicknesses δ and integral properties δ^* and θ at the TE of the sharp TE 2D airfoil models at $\alpha_t = 0^\circ$ are given in figure 6 for both the artificially tripped and the untripped boundary-layer conditions. The subscript 0 for the thicknesses indicates that the airfoil is at zero angle of attack. The parameters are normalized by the chord length c and are given

3.2. Calculation Procedures

The boundary-layer thickness parameters at the TE of a symmetric NACA 0012 airfoil at zero angle of attack are approximated by the curve fits to the data of figure 6. The expressions for the curve fits for boundary-layer thickness δ , displacement thickness δ^* , and momentum thickness θ are, for the *heavily tripped boundary layer*,

$$\delta_0/c = 10^{[1.892 - 0.9045 \log R_c + 0.0596(\log R_c)^2]} \quad (2)$$

$$\delta_0^*/c = \begin{cases} 0.0601R_c^{-0.114} & (R_c \leq 0.3 \times 10^6) \\ 10^{[3.411 - 1.5397 \log R_c + 0.1059(\log R_c)^2]} & (R_c > 0.3 \times 10^6) \end{cases} \quad (3)$$

$$\theta_0/c = \begin{cases} 0.0723R_c^{-0.1765} & (R_c \leq 0.3 \times 10^6) \\ 10^{[0.5578 - 0.7079 \log R_c + 0.0404(\log R_c)^2]} & (R_c > 0.3 \times 10^6) \end{cases} \quad (4)$$

as a function of Reynolds number based on the chord R_c . As R_c increases, the thicknesses decrease for both the tripped and the untripped boundary layers. The tripped boundary layers are almost uniformly thicker than the corresponding untripped boundary layers. One should refer to reference 21 for details of the boundary-layer character for the cases of figure 6. In general, however, one can say that the tripped boundary layers are fully turbulent for even the lowest R_c . The untripped boundary layers are laminar or transitional at low R_c and become fully turbulent for high R_c . In figure 6, the boundary-layer thickness data are approximated by curve fits whose equations are specified in the following section.

Angle-of-attack effects on the thickness parameters are given at free-stream velocities of 71.3 and 39.6 m/s for the untripped and tripped BL airfoils in figures 7 and 8. The parameters are normalized by those measured for the corresponding cases at zero angle of attack, given in figure 6. The data are plotted against the corrected angle α_* of equation (1). The collapse of the data is much improved over that when α_t is used (ref. 21). In general, the data show that for increasing α_* (or α_t) the thicknesses increase on the suction side because of the increasing severity of the adverse pressure gradient. The converse is true for the pressure side, where the pressure gradient becomes increasingly favorable. Also included in figures 7 and 8 are curve fits to the data. For the pressure side of the airfoils, the curves are the same for the tripped and untripped cases. The suction side curves differ, reflecting differences in the angle dependence of where the TE boundary layer separates and finally stalls the airfoil.

In reference 21, the data are discussed and compared with flat plate experimental results and results from boundary-layer prediction codes.

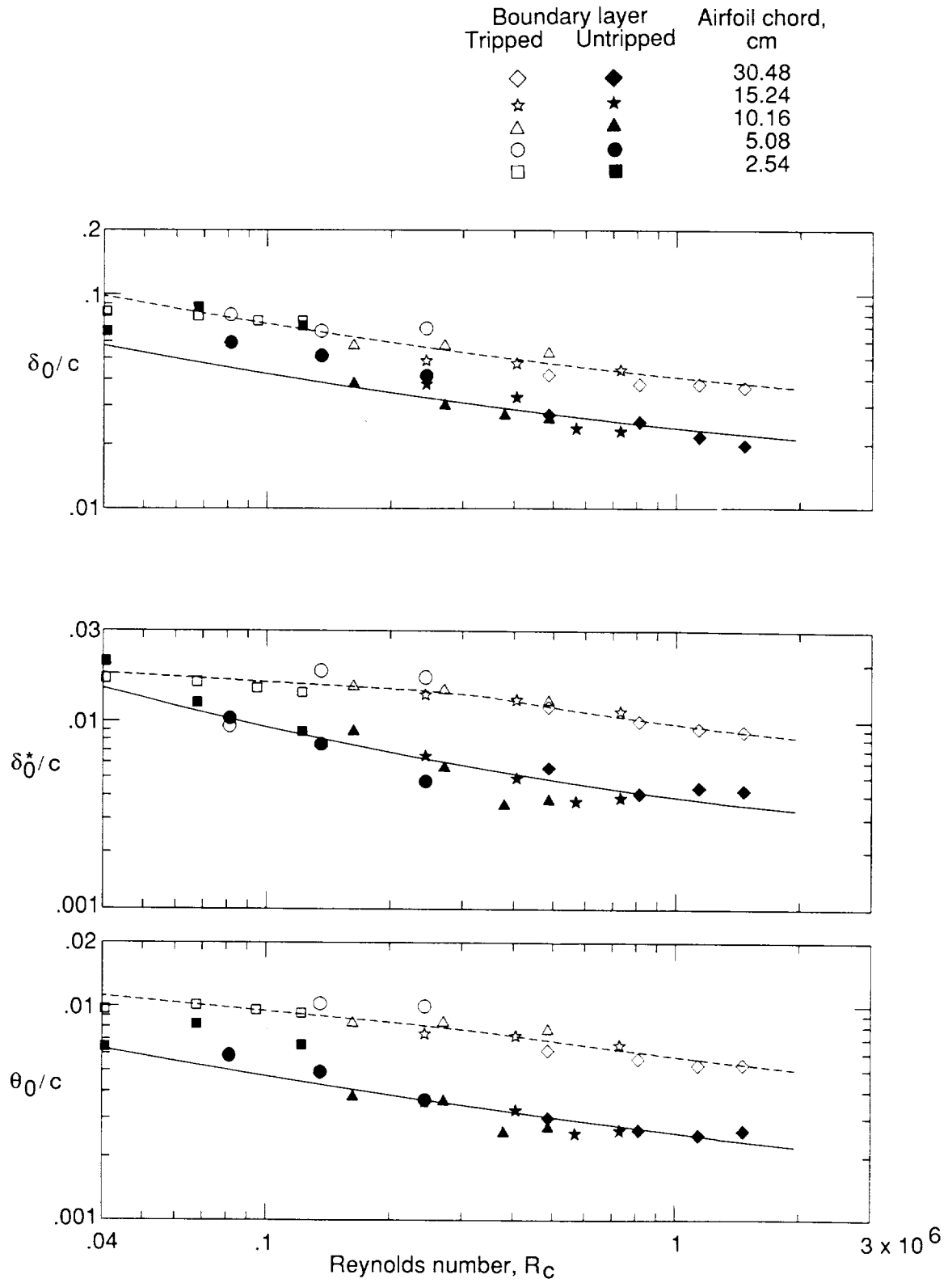
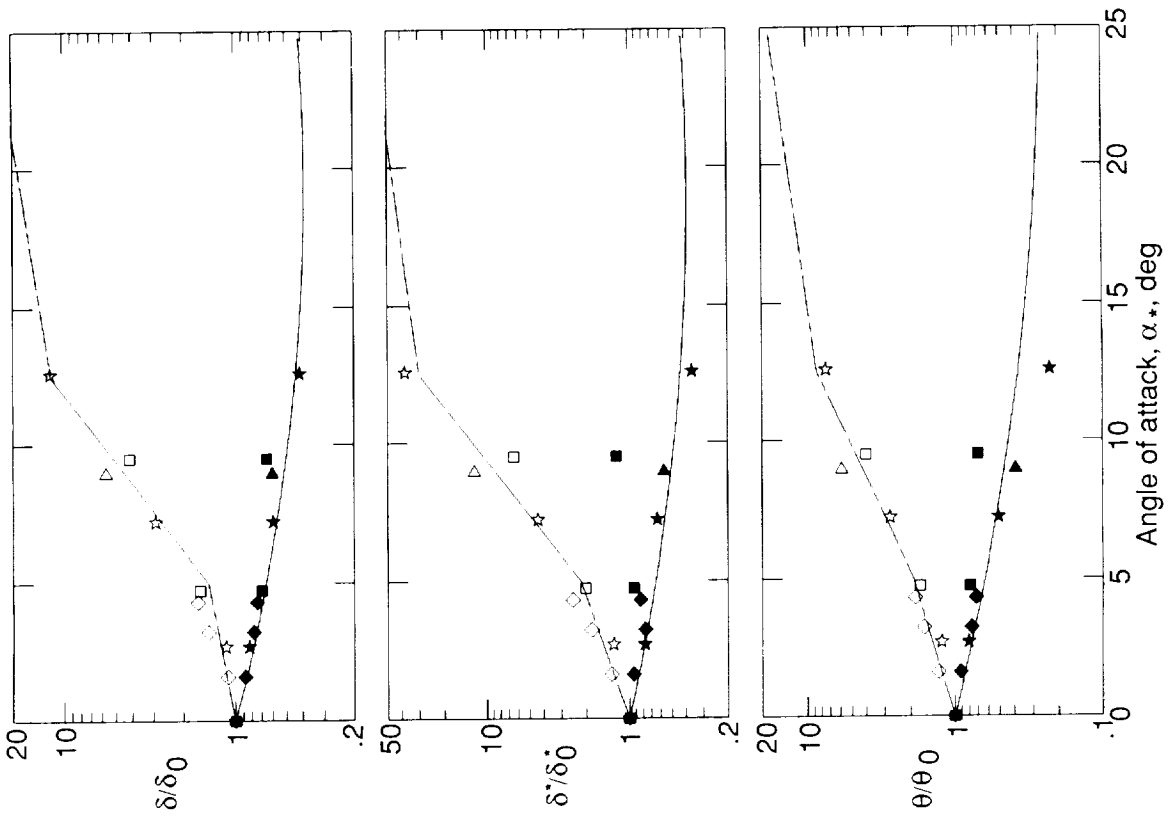
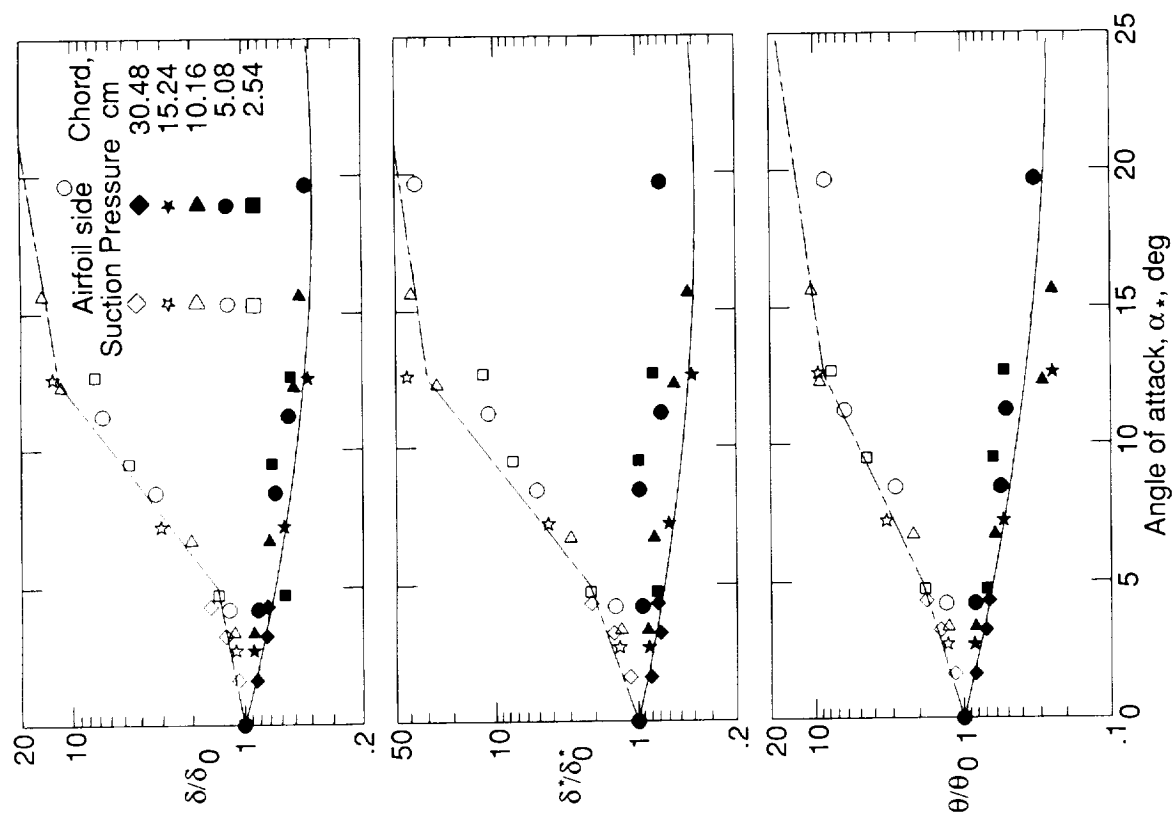


Figure 6. Boundary-layer thicknesses at the trailing edge of 2D airfoil models at angle of attack of zero. Solid lines are for untripped BL and broken lines are for tripped BL.

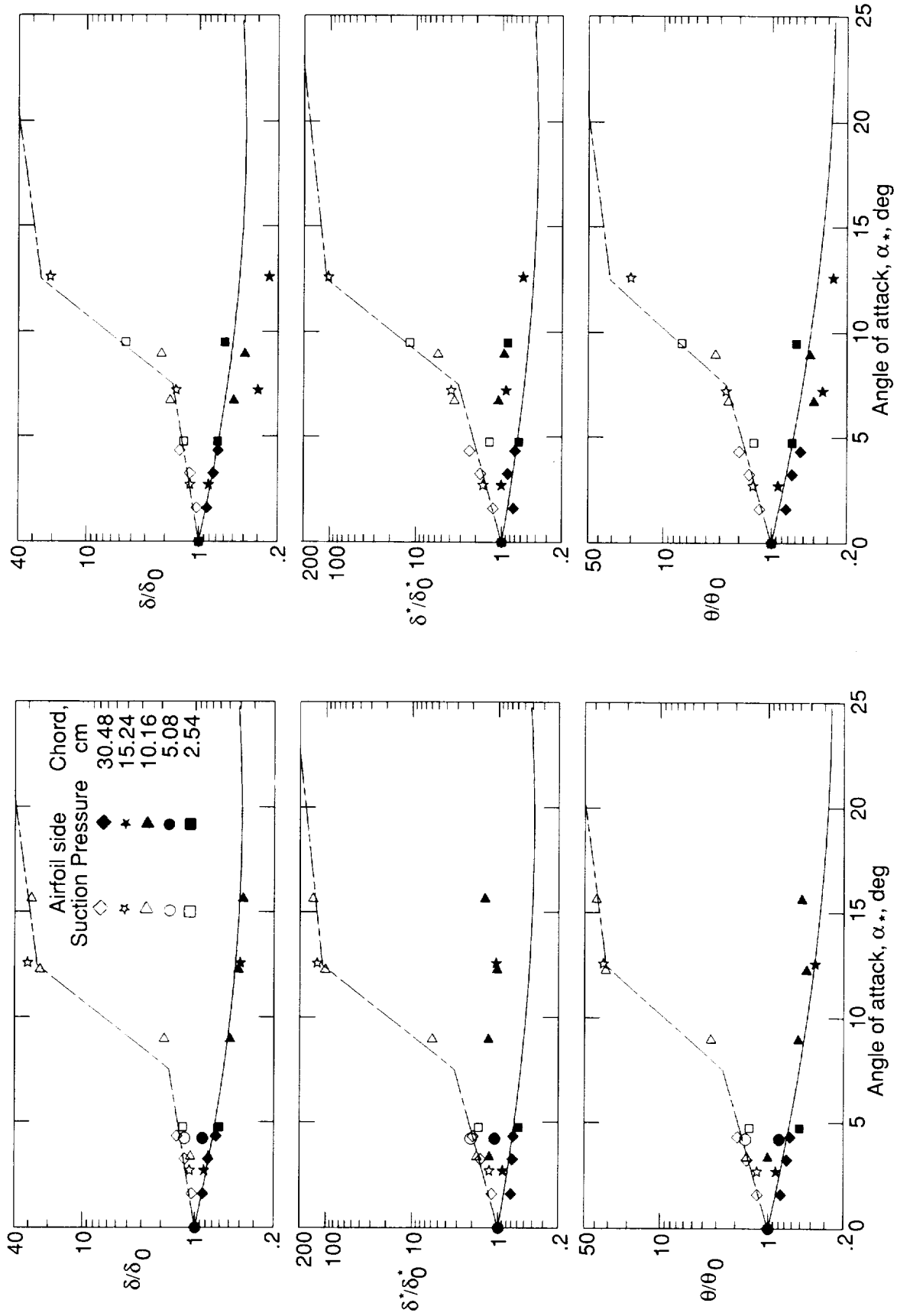


(b) $U = 39.6$ m/s.



(a) $U = 71.3$ m/s.

Figure 7. Tripped BL thicknesses at the TE versus corrected angle of attack. Solid lines are for pressure side of airfoil and broken lines are for suction side.



(a) $U = 71.3$ m/s.

(b) $U = 39.6$ m/s.

Figure 8. Untripped BL thicknesses at the TE versus corrected angle of attack. Solid lines are for pressure side of airfoil and broken lines are for suction side.

where the zero subscripts indicate zero angle of attack, zero lift on these symmetric airfoils. For the *untripped* (natural transition) *boundary layers*,

$$\delta_0/c = 10^{[1.6569 - 0.9045 \log R_c + 0.0596(\log R_c)^2]} \quad (5)$$

$$\delta_0^*/c = 10^{[3.0187 - 1.5397 \log R_c + 0.1059(\log R_c)^2]} \quad (6)$$

$$\theta_0/c = 10^{[0.2021 - 0.7079 \log R_c + 0.0404(\log R_c)^2]} \quad (7)$$

The boundary-layer thicknesses for the airfoils at nonzero angle of attack, in terms of the zero-angle-of-attack thicknesses and the corrected angles α_* , are given in figures 7 and 8. The expressions for the curve fits for the *pressure side*, for both the *tripped* and the *untripped boundary layers*, are

$$\frac{\delta_p}{\delta_0} = 10^{[-0.04175\alpha_* + 0.00106\alpha_*^2]} \quad (8)$$

$$\frac{\delta_p^*}{\delta_0^*} = 10^{[-0.0432\alpha_* + 0.00113\alpha_*^2]} \quad (9)$$

$$\frac{\theta_p}{\theta_0} = 10^{[-0.04508\alpha_* + 0.000873\alpha_*^2]} \quad (10)$$

For the suction side, the parametric behavior of the thicknesses depends on whether the boundary layers are attached, separated near the trailing edge, or separated a sufficient distance upstream to produce stall. For the *suction side for the tripped boundary layers* (fig. 7),

$$\frac{\delta_s}{\delta_0} = \begin{cases} 10^{0.0311\alpha_*} & (0^\circ \leq \alpha_* \leq 5^\circ) \\ 0.3468(10^{0.1231\alpha_*}) & (5^\circ < \alpha_* \leq 12.5^\circ) \\ 5.718(10^{0.0258\alpha_*}) & (12.5^\circ < \alpha_* \leq 25^\circ) \end{cases} \quad (11)$$

$$\frac{\delta_s^*}{\delta_0^*} = \begin{cases} 10^{0.0679\alpha_*} & (0^\circ \leq \alpha_* \leq 5^\circ) \\ 0.381(10^{0.1516\alpha_*}) & (5^\circ < \alpha_* \leq 12.5^\circ) \\ 14.296(10^{0.0258\alpha_*}) & (12.5^\circ < \alpha_* \leq 25^\circ) \end{cases} \quad (12)$$

$$\frac{\theta_s}{\theta_0} = \begin{cases} 10^{0.0559\alpha_*} & (0^\circ \leq \alpha_* \leq 5^\circ) \\ 0.6984(10^{0.0869\alpha_*}) & (5^\circ < \alpha_* \leq 12.5^\circ) \\ 4.0846(10^{0.0258\alpha_*}) & (12.5^\circ < \alpha_* \leq 25^\circ) \end{cases} \quad (13)$$

For the suction side for the untripped boundary layers (fig. 8),

$$\frac{\delta_s}{\delta_0} = \begin{cases} 10^{0.03114\alpha_*} & (0^\circ \leq \alpha_* \leq 7.5^\circ) \\ 0.0303(10^{0.2336\alpha_*}) & (7.5^\circ < \alpha_* \leq 12.5^\circ) \\ 12(10^{0.0258\alpha_*}) & (12.5^\circ < \alpha_* \leq 25^\circ) \end{cases} \quad (14)$$

$$\frac{\delta_s^*}{\delta_0^*} = \begin{cases} 10^{0.0679\alpha_*} & (0^\circ \leq \alpha_* \leq 7.5^\circ) \\ 0.0162(10^{0.3066\alpha_*}) & (7.5^\circ < \alpha_* \leq 12.5^\circ) \\ 52.42(10^{0.0258\alpha_*}) & (12.5^\circ < \alpha_* \leq 25^\circ) \end{cases} \quad (15)$$

$$\frac{\theta_s}{\theta_0} = \begin{cases} 10^{0.0559\alpha_*} & (0^\circ \leq \alpha_* \leq 7.5^\circ) \\ 0.0633(10^{0.2157\alpha_*}) & (7.5^\circ < \alpha_* \leq 12.5^\circ) \\ 14.977(10^{0.0258\alpha_*}) & (12.5^\circ < \alpha_* \leq 25^\circ) \end{cases} \quad (16)$$

4. Acoustic Measurements

The aim of the acoustic measurements was to determine spectra for self-noise from airfoils encountering smooth airflow. This task is complicated by the unavoidable presence of extraneous tunnel test rig noise. In this section, cross-correlations between microphones are examined to identify the self-noise emitted from the TE in the presence of other sources. Then, the spectra of self-noise are determined by performing Fourier transforms of cross-correlation data which have been processed and edited to eliminate the extraneous contributions. The results are presented as 1/3-octave spectra, which then form the data base from which the self-noise scaling prediction equations are developed.

4.1. Source Identification

The upper curves in figure 9 are the cross-correlations, $R_{12}(\tau) = \langle p_1(t)p_2(t + \tau) \rangle$, between the sound pressure signals p_1 and p_2 of microphones M1 and M2 identified in figure 4. Presented are cross-correlations both with and without the tripped 30.48-cm-chord airfoil mounted in the test rig. Because the microphones were on opposite sides of, and at equal distance from, the airfoil, a negative correlation peak occurs at a signal delay time τ of 0. This correlation is consistent with a broadband noise source of dipole character, whose phase is reversed on opposing sides. When the airfoil is removed, the strong negative peak disappears leaving the contribution from the test rig alone. The most coherent parts of this noise are from the lips of the nozzle and are, as with the airfoil noise, of a dipole character. The microphone time delays predicted for these sources are indicated by arrows. The predictions account for the effect of refraction of sound by the free-jet shear layer (refs. 22 and 23), as well as the geometric relationship between the microphones and the hardware and the speed of sound.

The lower curves in figure 9 are the cross-correlations, $R_{45}(\tau)$, between microphones M4 and M5. The predicted delay times again appear to correctly identify the correlation peaks associated with the noise emission locations. The peaks are positive for $R_{45}(\tau)$ because both microphones are on the same side of the dipoles' directional lobes. The noise field is dominated by TE noise. Any contribution to the noise field from the LE would appear where indicated in the figure. As is subsequently shown, there are contributions in many cases. For such cases the negative correlation peak for $R_{12}(\tau)$ would be the sum of the TE and LE correlation peaks brought together at $\tau = 0$ and inverted in sign.

In figure 10, the cross-correlations $R_{45}(\tau)$ are shown for tripped BL airfoils of various sizes. The TE noise correlation peaks are at $\tau_{TE} = -0.11$ ms

for all cases because at $\alpha_t = 0^\circ$, the TE location of all models is the same. The LE location changes with chord size, as is indicated by the change in the predicted LE noise correlation peak delay times.

For the larger airfoils in figure 10, the TE contribution dominates the noise field. As the chord length decreases, the LE noise peaks increase to become readily identifiable in the correlation. For the smallest chord the LE contribution is even somewhat more than that of the TE. Note the extraneous, but inconsequential, source of discrete low-frequency noise contributing to the 22.86-cm-chord correlation, which can be readily edited in a spectral format.

It is shown in reference 6 that the LE and TE sources are uncorrelated. The origin of LE noise appears to be inflow turbulence to the LE from the TBL of the test rig side plates. This should be the case even though the spanwise extent of this TBL is small compared with the portion of the models that encounter uniform low-turbulence flow from the nozzle. Inflow turbulence can be a very efficient noise mechanism (ref. 24); however its full efficiency can be obtained only when the LE of the model is relatively sharp compared with the scale of the turbulence. The LE noise contributions diminish for the large chord because of the proportional increase in LE radius with chord. When this radius increases to a size that is large compared with the turbulent scale in the side plate TBL, then the sectional lift fluctuations associated with inflow turbulence noise are not developed.

4.2. Correlation Editing and Spectral Determination

The cross-spectrum between microphones M1 and M2, denoted $G_{12}(f)$, is the Fourier transform of $R_{12}(\tau)$. If the contributions from the LE, nozzle lips, and any other coherent extraneous source locations were removed, $G_{12}(f)$ would equal the autospectrum of the airfoil TE self-noise, $S(f)$. Actually the relationship would be $G_{12}(f) = S(f) \exp[i(2\pi f \tau_{TE} \pm \pi)]$, where $i = \sqrt{-1}$ and τ_{TE} is the delay time of the TE correlation peak. This approach is formalized in reference 2.

In reference 6, the spectra were found from $G_{12}(f)$ determined with the models of the test rig after a point-by-point vectorial subtraction of $G_{12}(f)$ determined with the airfoil removed. This was equivalent to subtracting corresponding $R_{12}(\tau)$ results, such as those of figure 9, and then taking the Fourier transform. This resulted in "corrected" spectra which were devoid of at least a portion of the background test rig noise, primarily emitted from the nozzle lips. The spectra still were contaminated by the LE noise due to the inflow turbulence.

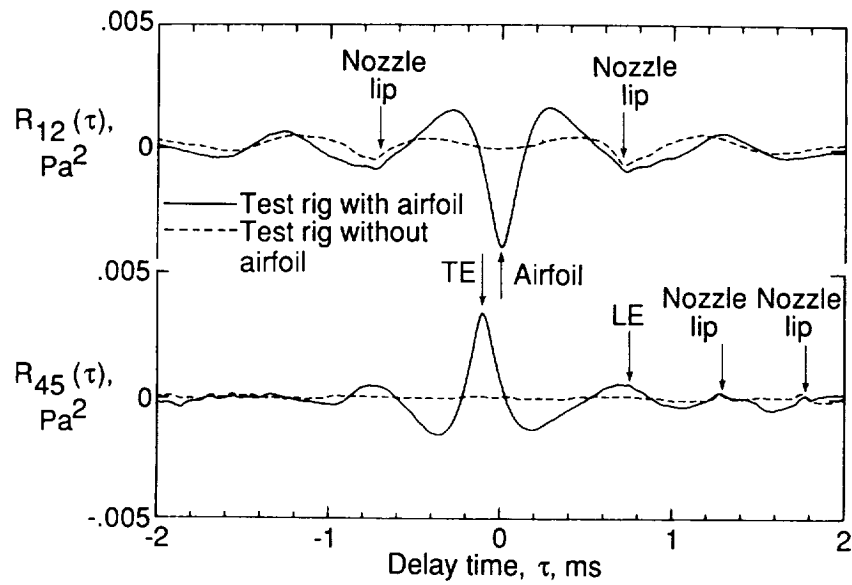


Figure 9. Cross-correlations for two microphone pairs with and without airfoil mounted in test rig. $c = 30.48$ cm; BL tripped; $\alpha_t = 0^\circ$; $U = 71.3$ m/s. Arrows indicate predicted values of τ . (From ref. 6.)

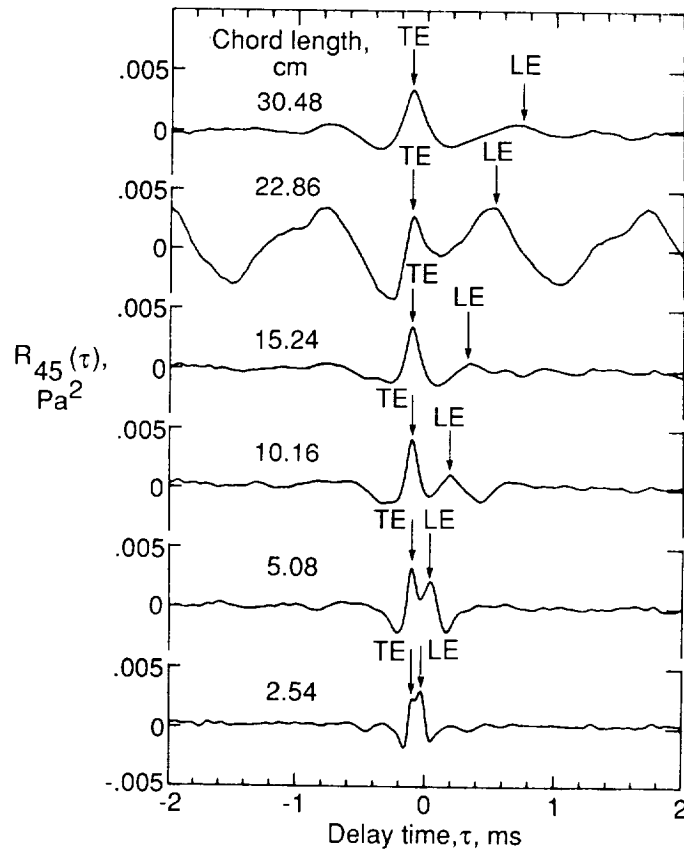


Figure 10. Cross-correlations between microphones M4 and M5 for tripped BL airfoils of various chord sizes. $U = 71.3$ m/s. Arrows indicate predicted values of τ . (From ref. 6.)

In the present paper, most spectra presented were obtained by taking the Fourier transform of microphone-pair cross-correlations which had been edited to eliminate LE noise (see details in appendix A). The microphone pairs used included M4 and M5, M4 and M8, and M4 and M2. These pairs produced correlations where the TE and LE noise peaks were generally separated and readily identifiable. Referring to figure 10 for $R_{45}(\tau)$, the approach was to employ only the left-hand side (LHS) of the TE noise peak. The LHS was “folded” about τ at the peak (τ_{TE}) to produce a nearly symmetrical correlation. Care was taken in the processing to maintain the actual shapes near the very peak, to avoid to the extent possible the artificial introduction of high-frequency noise in the resulting spectra. Cross-spectra were then determined which were equated to the spectra of TE self-noise.

The data processing was straightforward for the larger chord airfoils because the LE and TE peaks were sufficiently separated from one another that the influence of the LE did not significantly impact the TE noise correlation shapes. For many of the smaller airfoils, such as those with chord lengths of 2.54, 5.08, and 10.16 cm shown in figure 10, the closeness of the LE contribution distorted the TE noise correlation. A processing procedure was developed to effectively “separate” the TE and LE peaks to a sufficient distance from one another, within the correlation presentation, so that the correlation folding of the LHS about τ_{TE} produced a more accurate presentation of the TE noise correlation shape. The separation processing employed symmetry assumptions for the TE and LE noise correlations to allow manipulation of the correlation records. This processing represented a contamination removal method used for about one-

quarter of the spectra presented for the three smallest airfoil chord lengths. Each case was treated individually to determine whether correlation folding alone, folding after the separation processing, or not folding at all produced spectra containing the least apparent error. In appendix A, details of the editing and Fourier transform procedures, as well as the separation processing, are given.

4.3. Self-Noise Spectra

The self-noise spectra for the 2D NACA 0012 airfoil models with sharp TE are presented in a 1/3-octave format in figures 11 to 74. Figures 11 to 43 are for airfoils where the boundary layers have been tripped and figures 44 to 74 are for smooth surface airfoils where the boundary layers are untripped (natural transition). Each figure contains spectra for a model at a specific angle of attack for various tunnel speeds. Note that the spectra are truncated at upper and lower frequencies. This editing of the spectra was done because, as described in appendix A, a review of the narrow-band amplitude and phase for all cases revealed regions where extraneous noise affected the spectra in a significant way (2 dB or more). These regions were removed from the 1/3-octave presentations.

The spectra levels have been corrected for shear layer diffraction and TE noise directivity effects, as detailed in appendix B. The noise should be that for an observer positioned perpendicular to, and 1.22 m from, the TE and the model midspan. In terms of the directivity definitions of appendix B, $r_e = 1.22$ m, $\Theta_e = 90^\circ$, and $\Phi_e = 90^\circ$. In section 5 (beginning on p. 51), the character and parametric behavior of the self-noise, as well as the predictions which are compared with the data, are discussed.

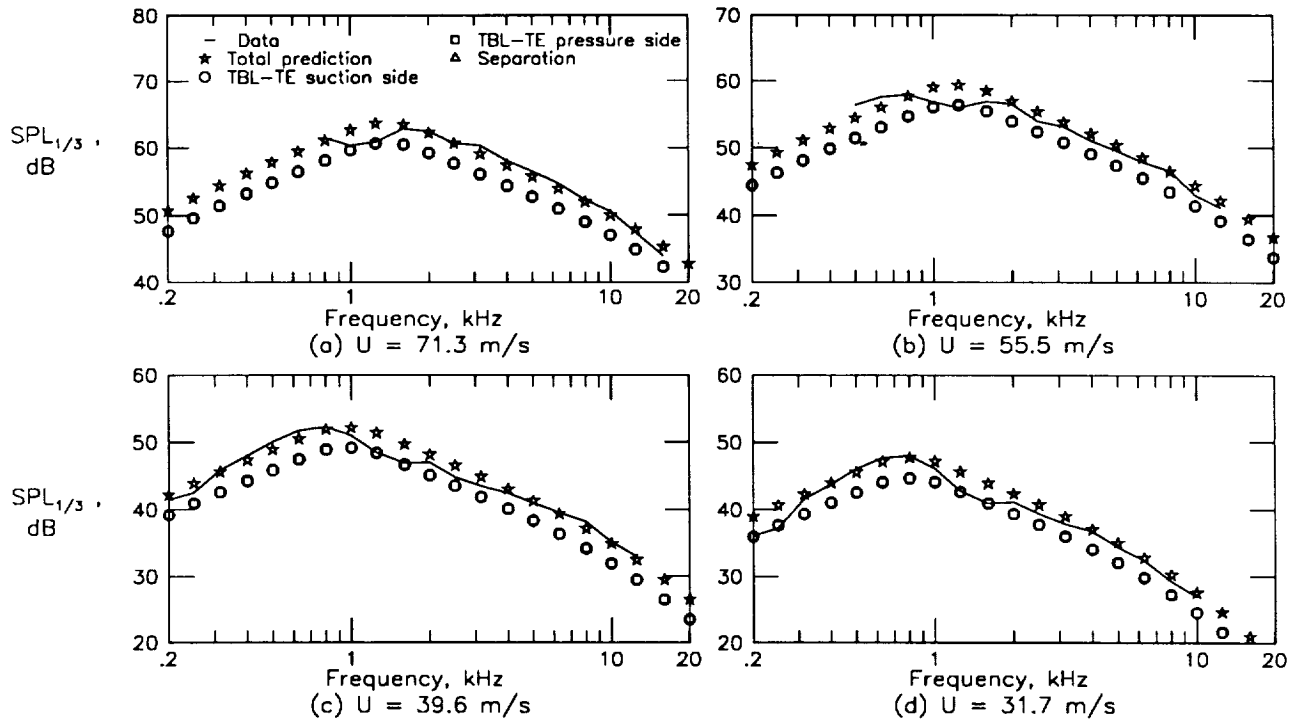


Figure 11. Self-noise spectra for 30.48-cm-chord airfoil with tripped BL at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

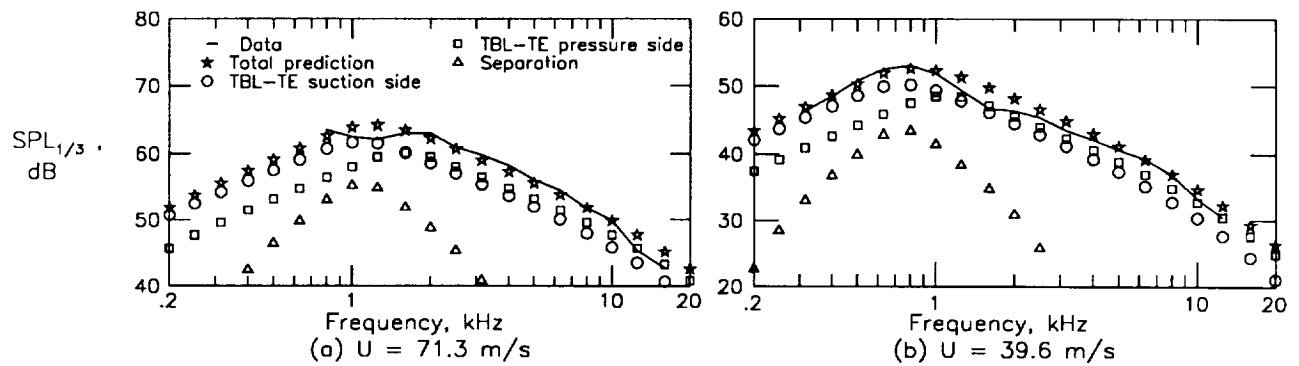


Figure 12. Self-noise spectra for 30.48-cm-chord airfoil with tripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 1.5^\circ$).

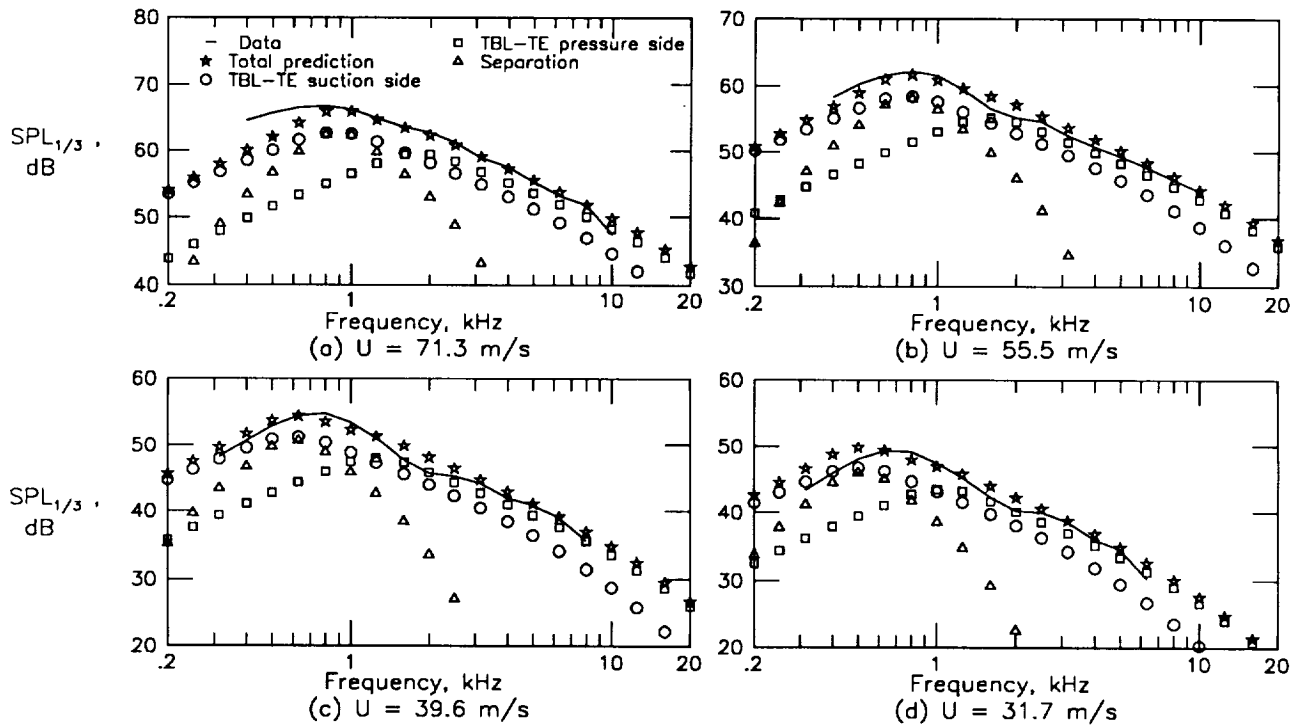


Figure 13. Self-noise spectra for 30.48-cm-chord airfoil with tripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 3.0^\circ$).

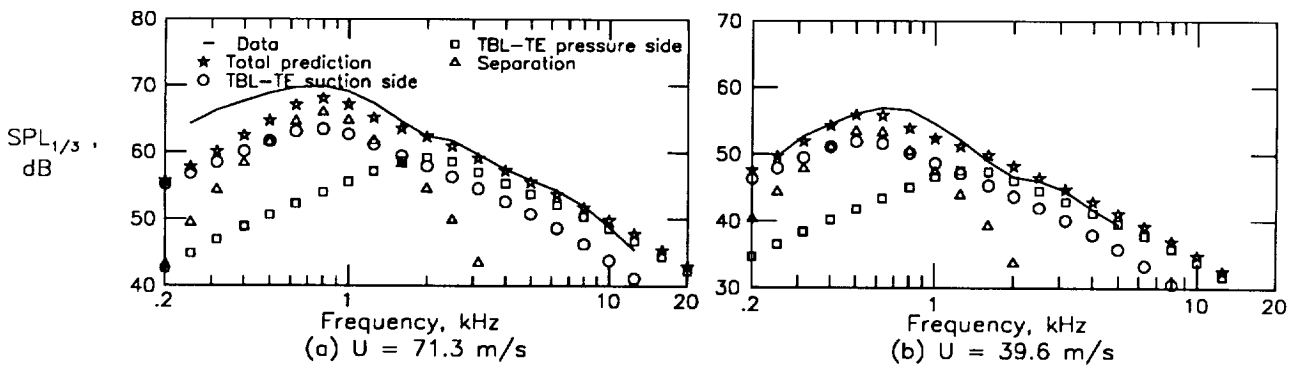


Figure 14. Self-noise spectra for 30.48-cm-chord airfoil with tripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 4.0^\circ$).

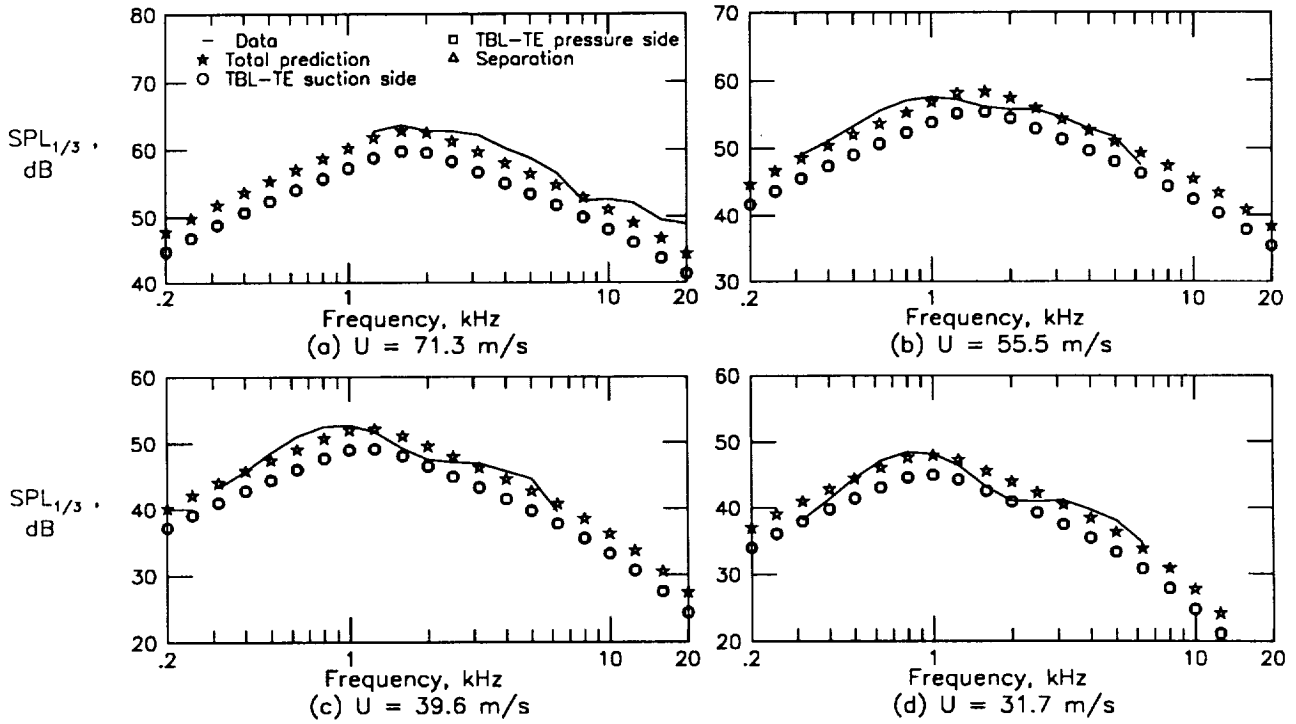


Figure 15. Self-noise spectra for 22.86-cm-chord airfoil with tripped BL at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

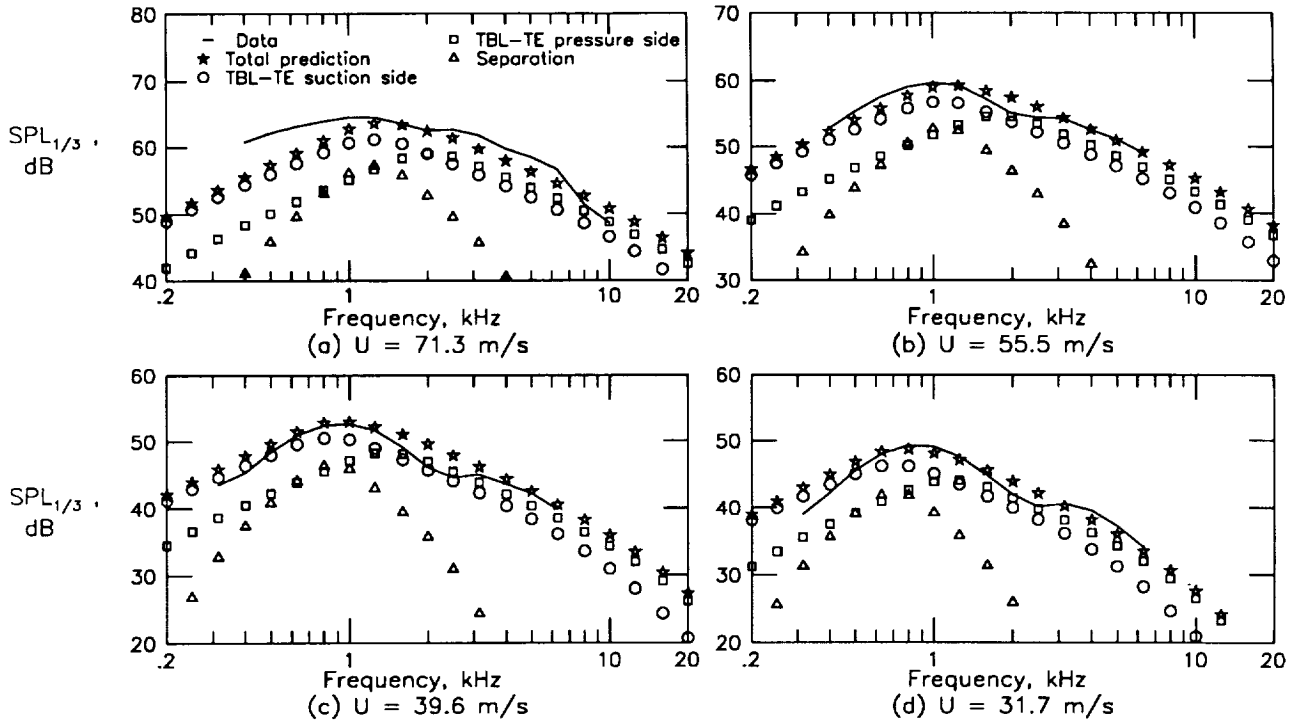


Figure 16. Self-noise spectra for 22.86-cm-chord airfoil with tripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 2.0^\circ$).

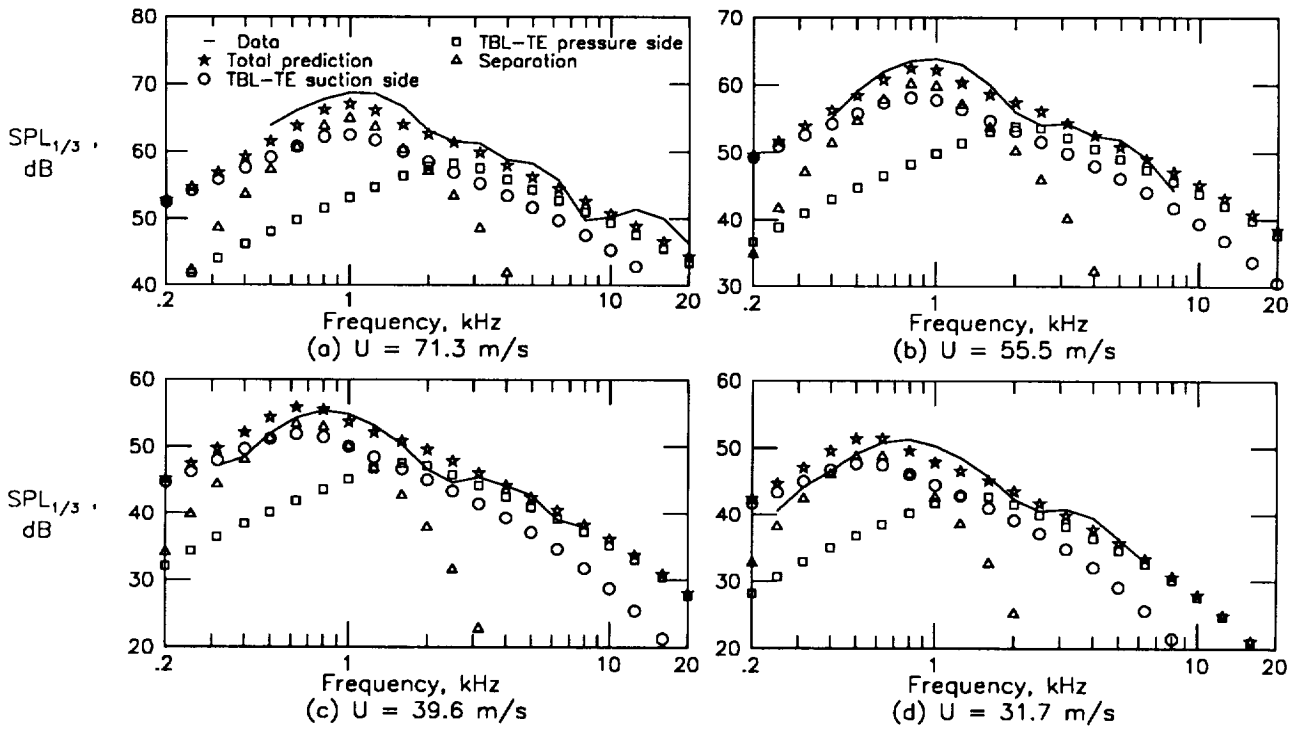


Figure 17. Self-noise spectra for 22.86-cm-chord airfoil with tripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 4.0^\circ$).

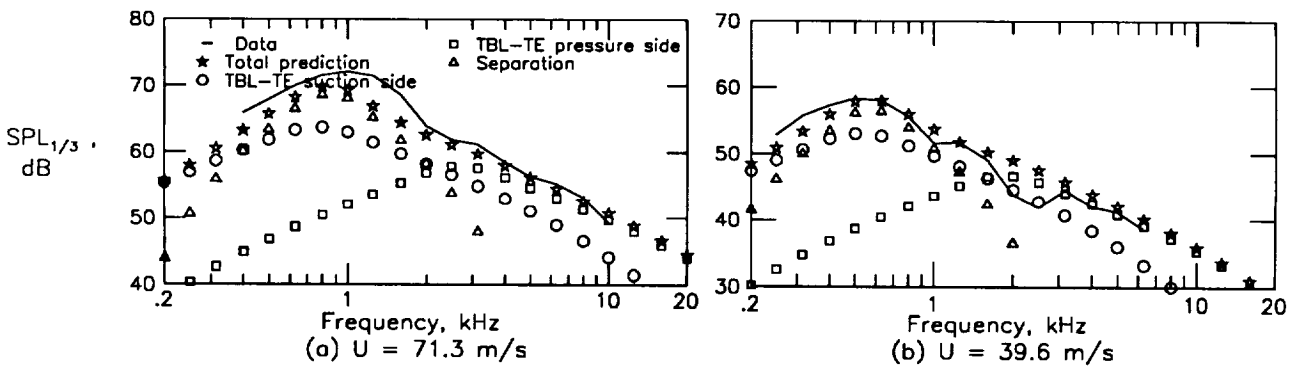


Figure 18. Self-noise spectra for 22.86-cm-chord airfoil with tripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 5.3^\circ$).

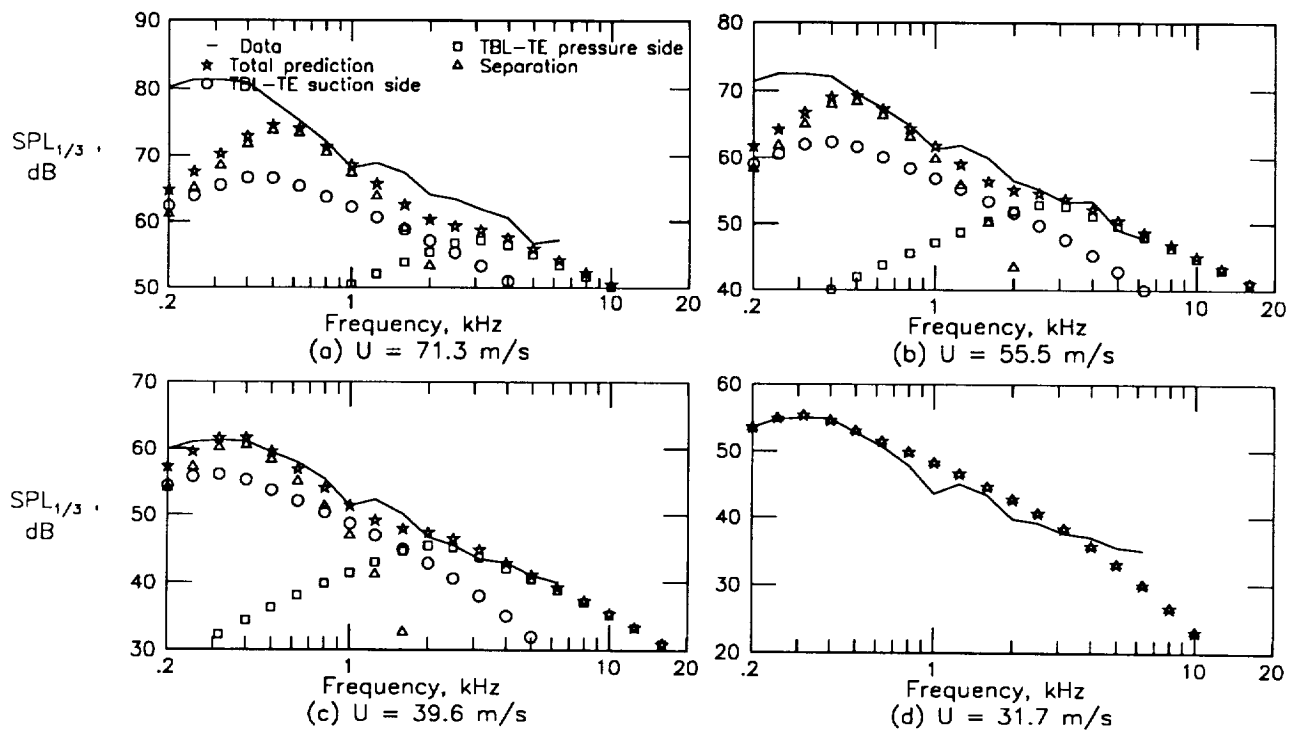


Figure 19. Self-noise spectra for 22.86-cm-chord airfoil with tripped BL at $\alpha_t = 19.8^\circ$ ($\alpha_* = 7.3^\circ$).

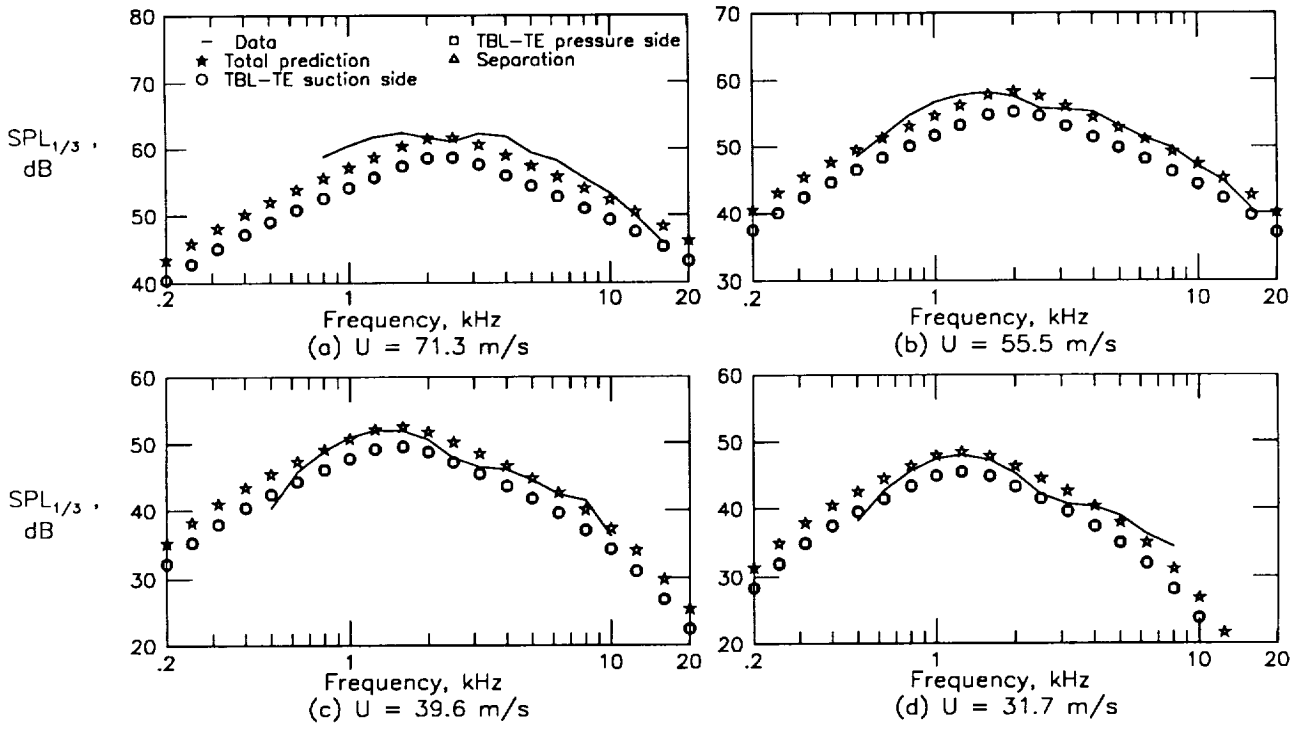


Figure 20. Self-noise spectra for 15.24-cm-chord airfoil with tripped BL at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

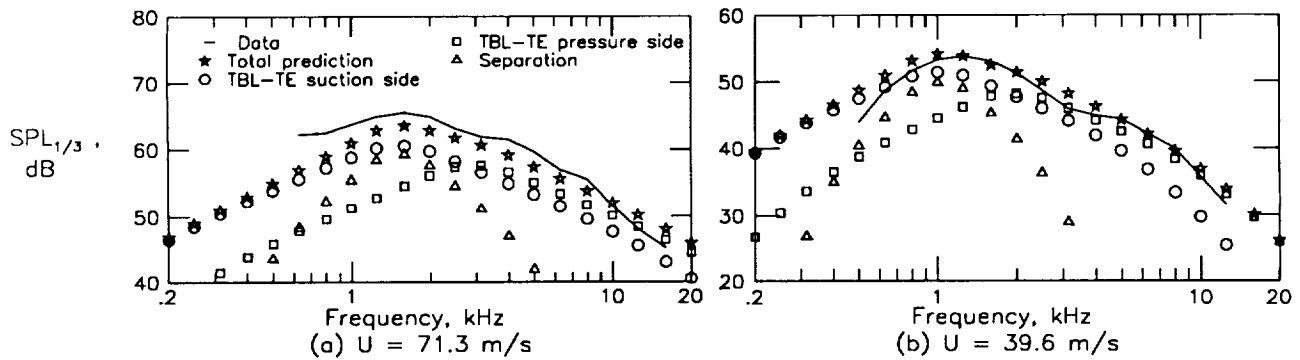


Figure 21. Self-noise spectra for 15.24-cm-chord airfoil with tripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 2.7^\circ$).

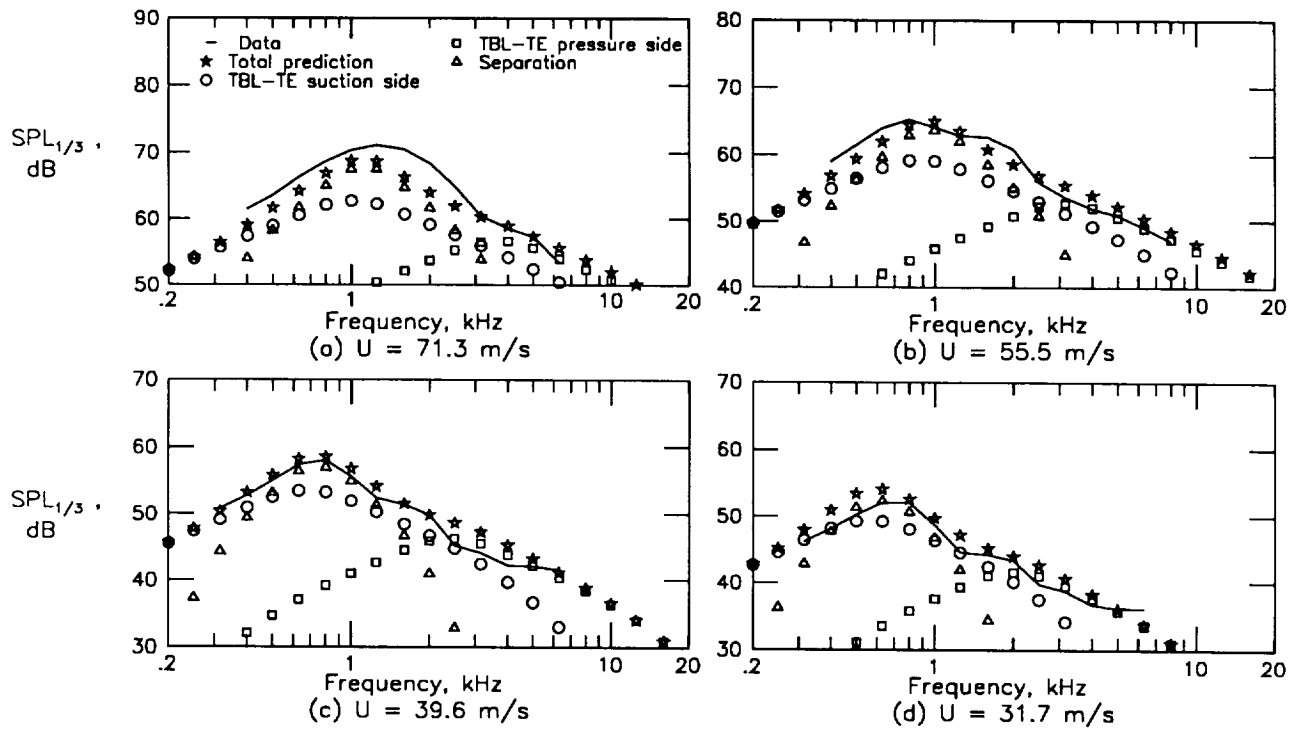


Figure 22. Self-noise spectra for 15.24-cm-chord airfoil with tripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 5.4^\circ$).

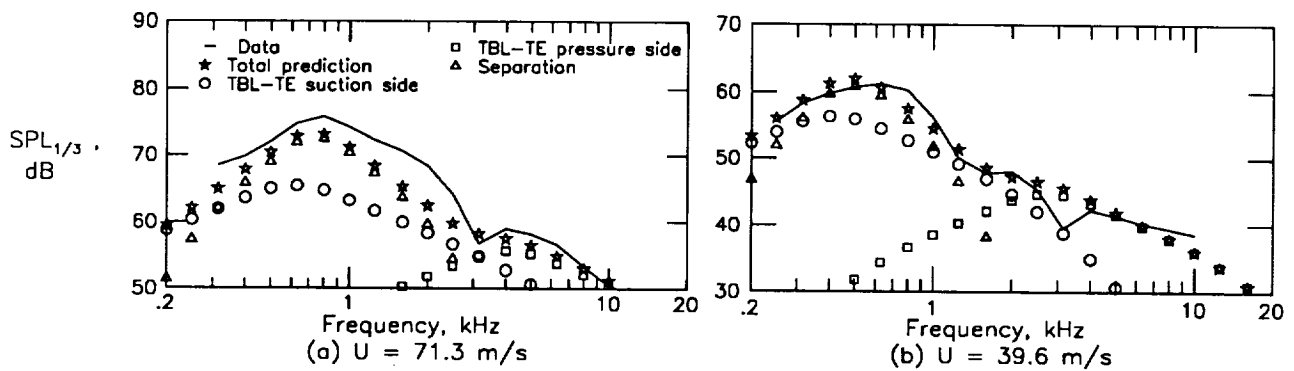


Figure 23. Self-noise spectra for 15.24-cm-chord airfoil with tripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 7.2^\circ$).

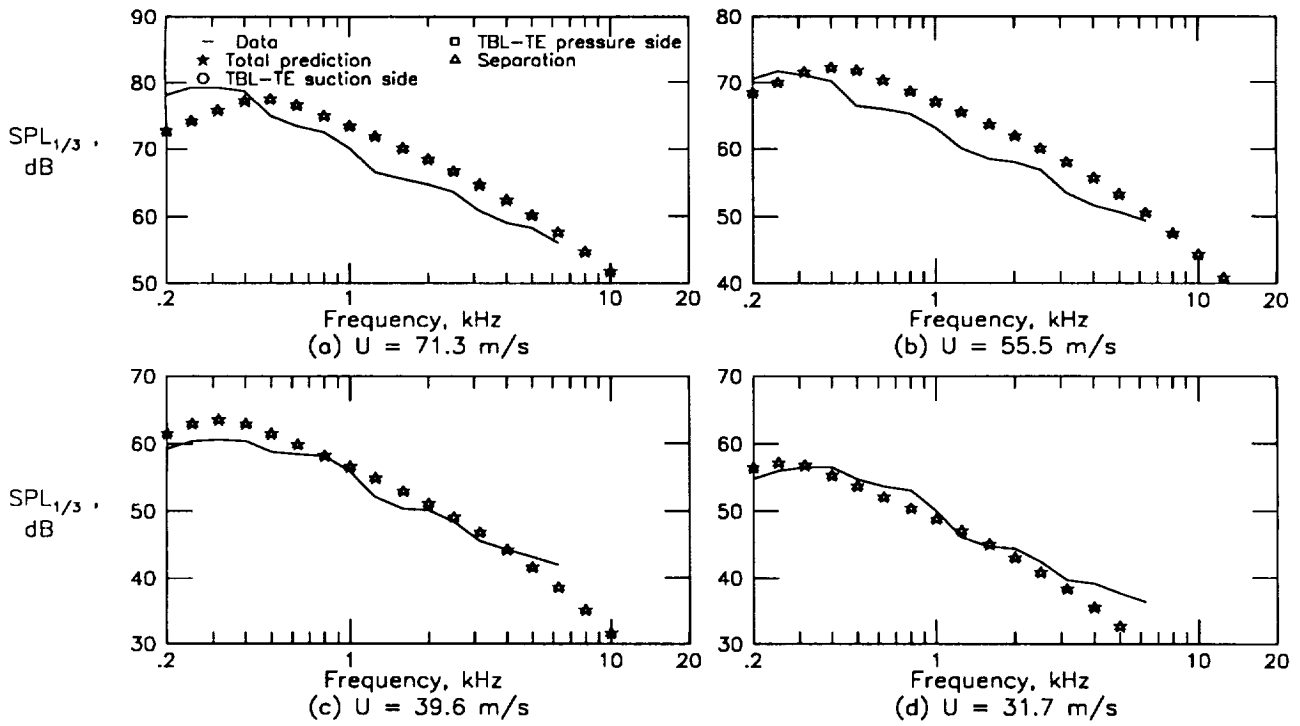


Figure 24. Self-noise spectra for 15.24-cm-chord airfoil with tripped BL at $\alpha_t = 19.8^\circ$ ($\alpha_* = 9.9^\circ$).

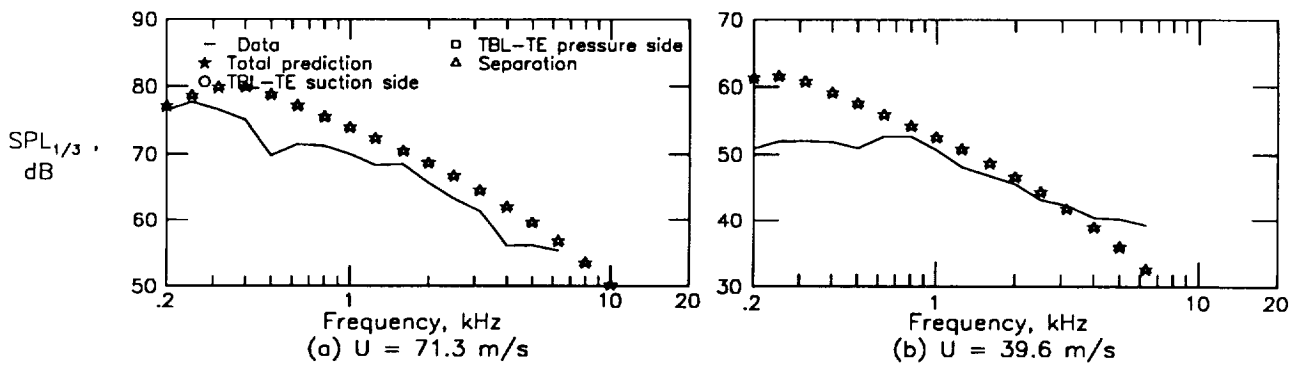


Figure 25. Self-noise spectra for 15.24-cm-chord airfoil with tripped BL at $\alpha_t = 25.2^\circ$ ($\alpha_* = 12.6^\circ$).

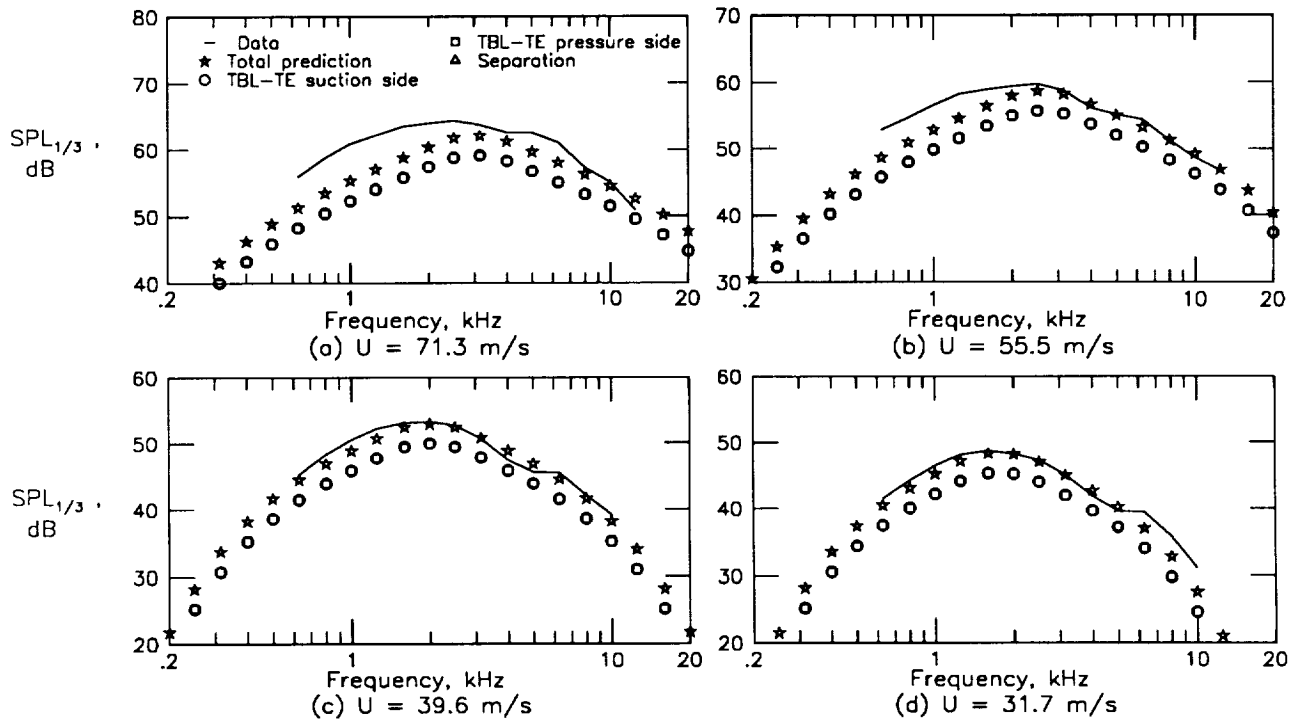


Figure 26. Self-noise spectra for 10.16-cm-chord airfoil with tripped BL at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

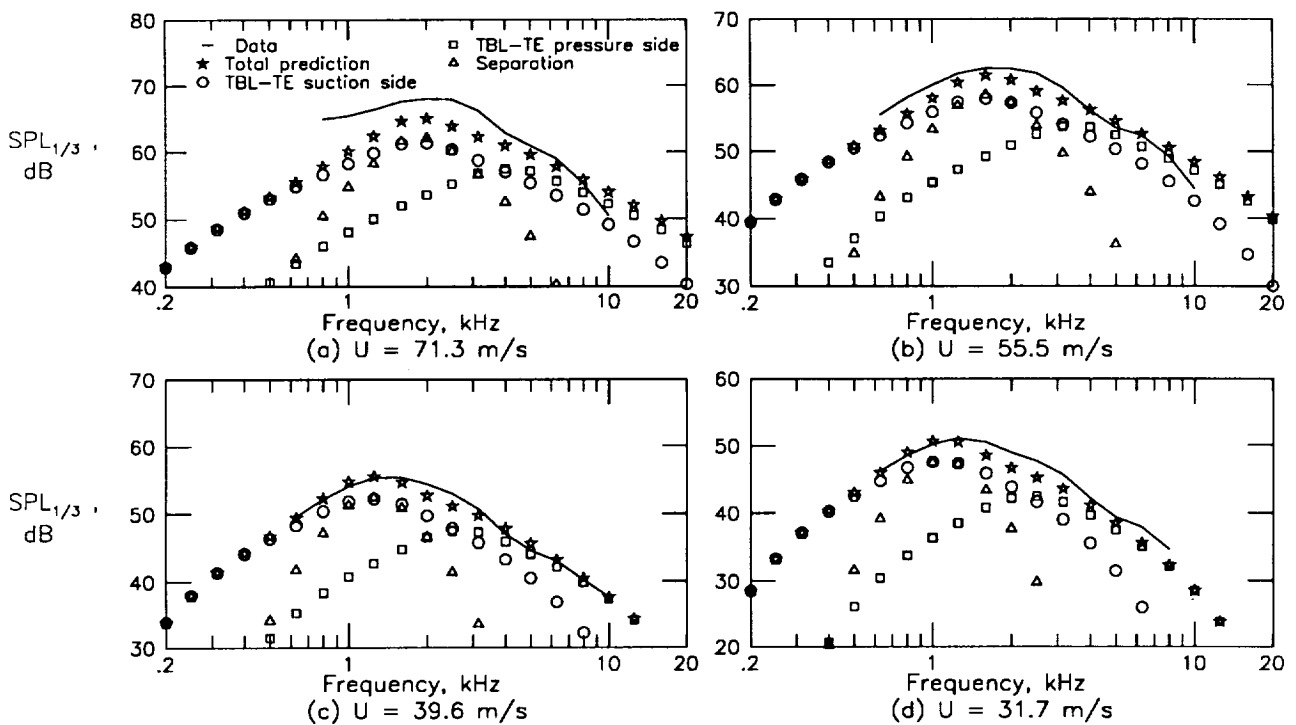


Figure 27. Self-noise spectra for 10.16-cm-chord airfoil with tripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 3.3^\circ$).

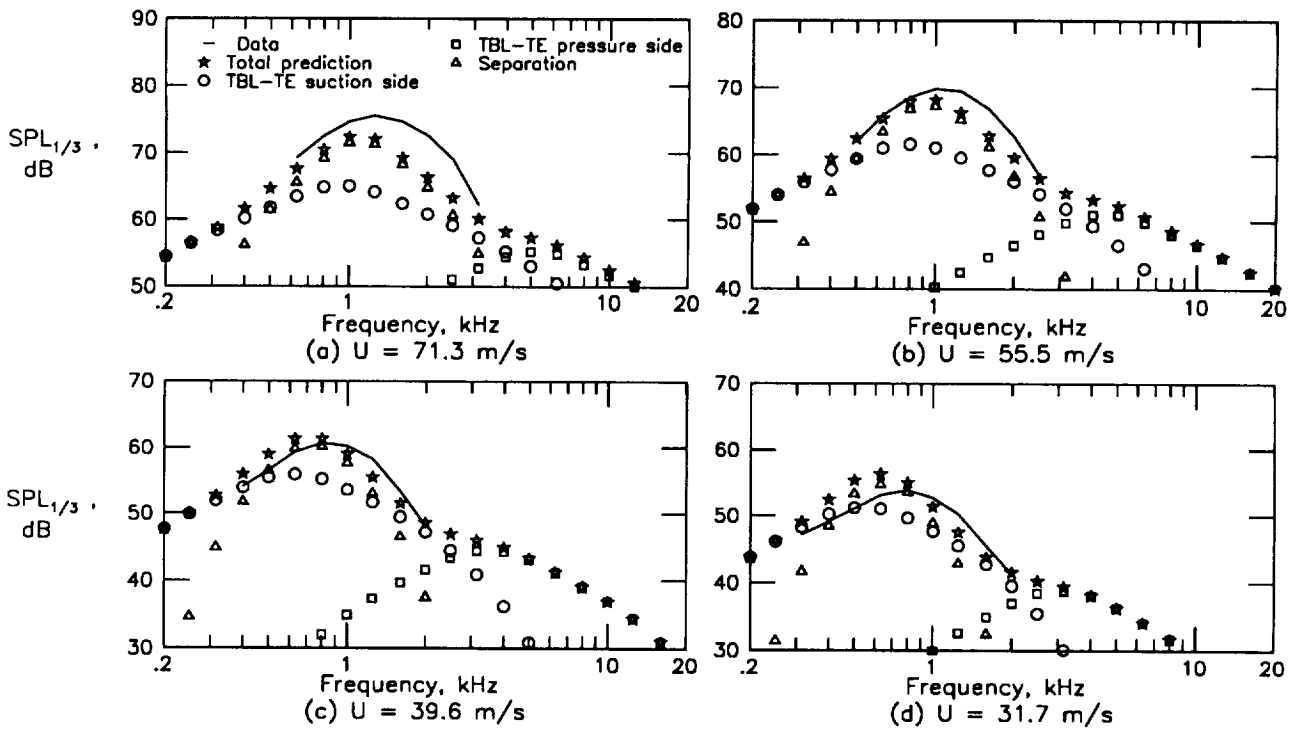


Figure 28. Self-noise spectra for 10.16-cm-chord airfoil with tripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 6.7^\circ$).

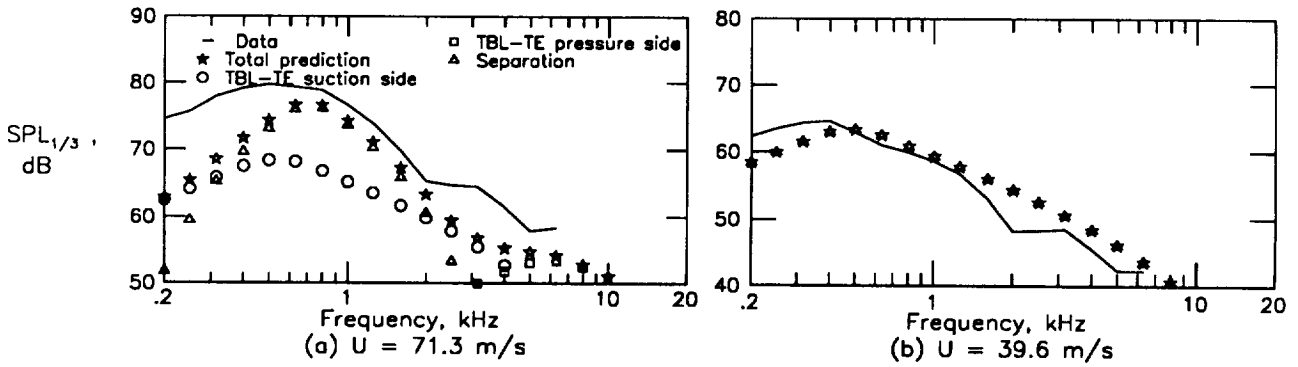


Figure 29. Self-noise spectra for 10.16-cm-chord airfoil with tripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 8.9^\circ$).

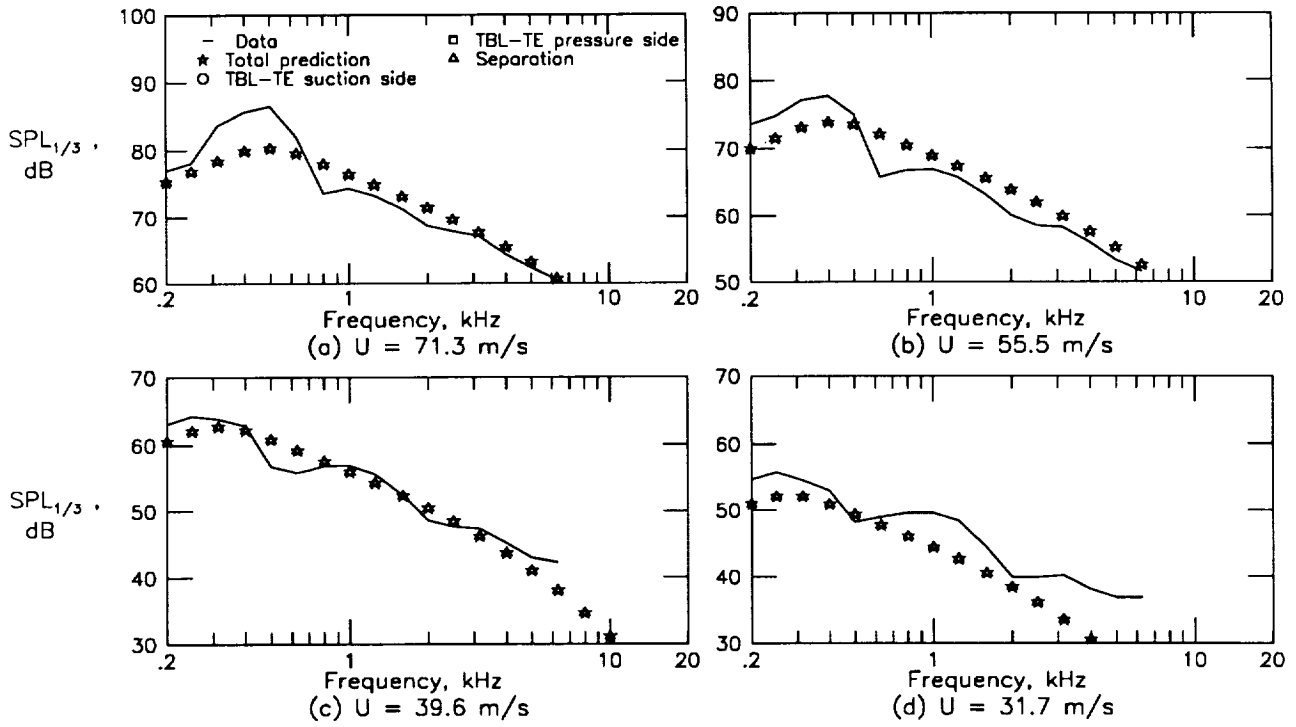


Figure 30. Self-noise spectra for 10.16-cm-chord airfoil with tripped BL at $\alpha_t = 19.8^\circ$ ($\alpha_* = 12.3^\circ$).

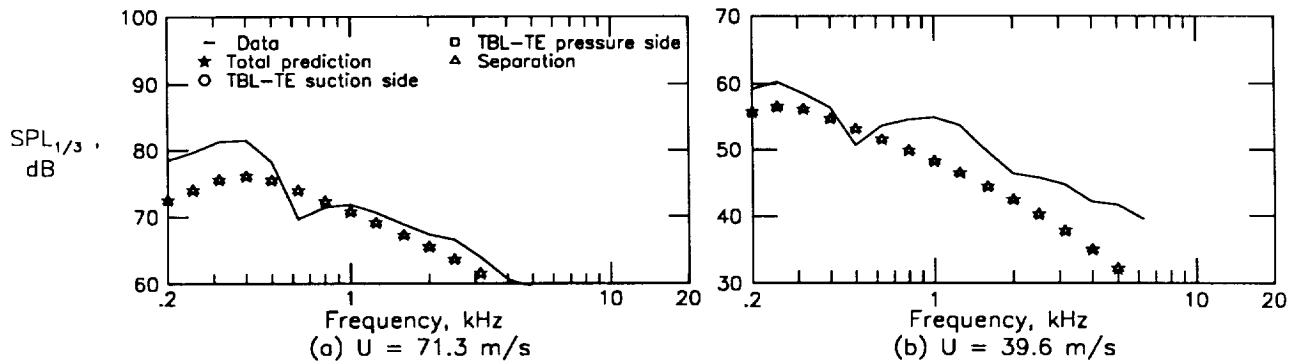


Figure 31. Self-noise spectra for 10.16-cm-chord airfoil with tripped BL at $\alpha_t = 25.2^\circ$ ($\alpha_* = 15.6^\circ$).

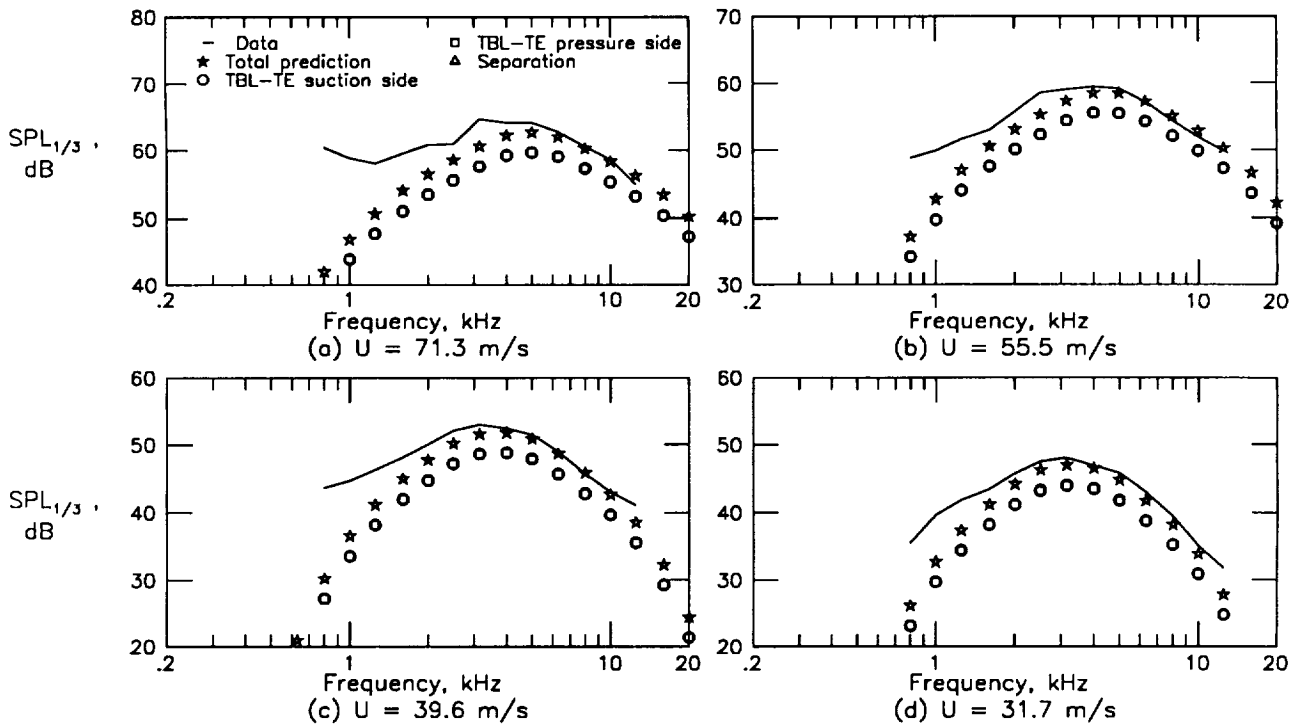


Figure 32. Self-noise spectra for 5.08-cm-chord airfoil with tripped BL at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

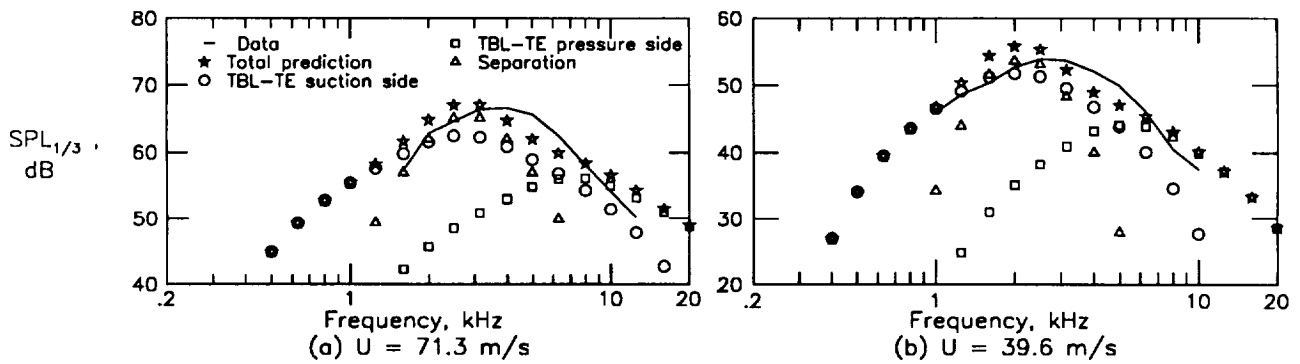


Figure 33. Self-noise spectra for 5.08-cm-chord airfoil with tripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 4.2^\circ$).

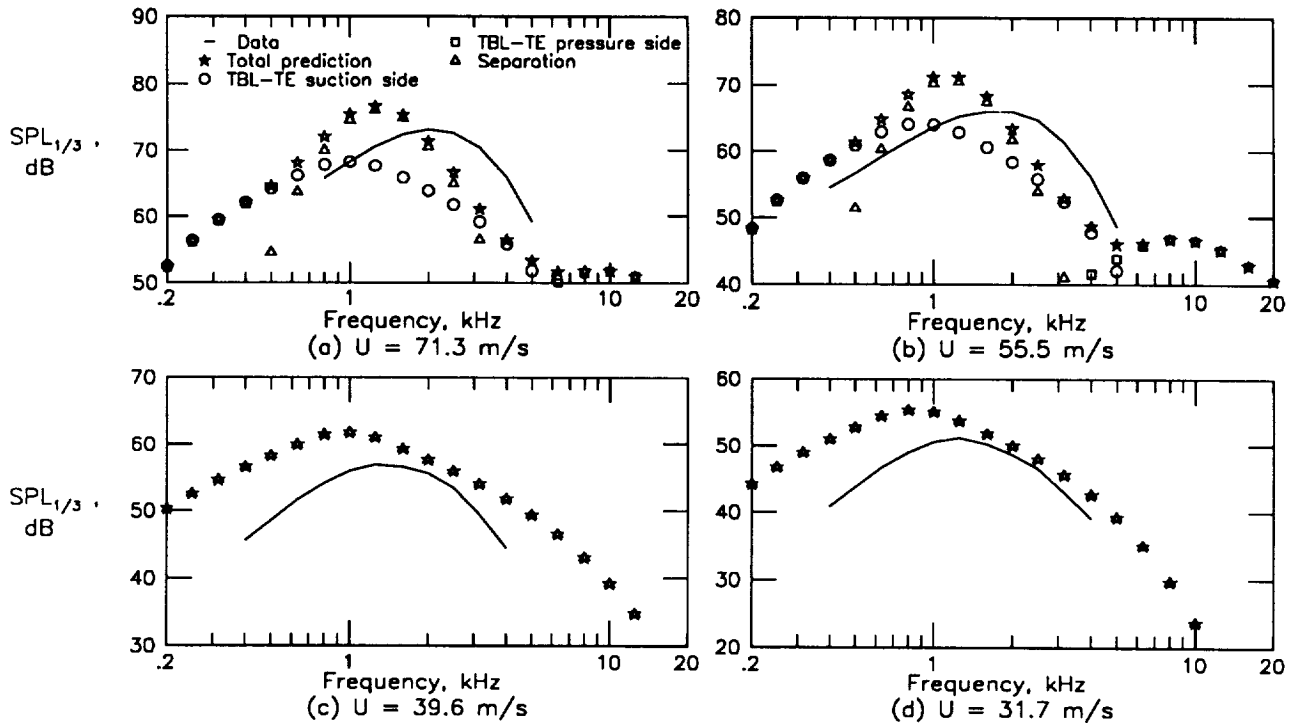


Figure 34. Self-noise spectra for 5.08-cm-chord airfoil with tripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 8.4^\circ$).

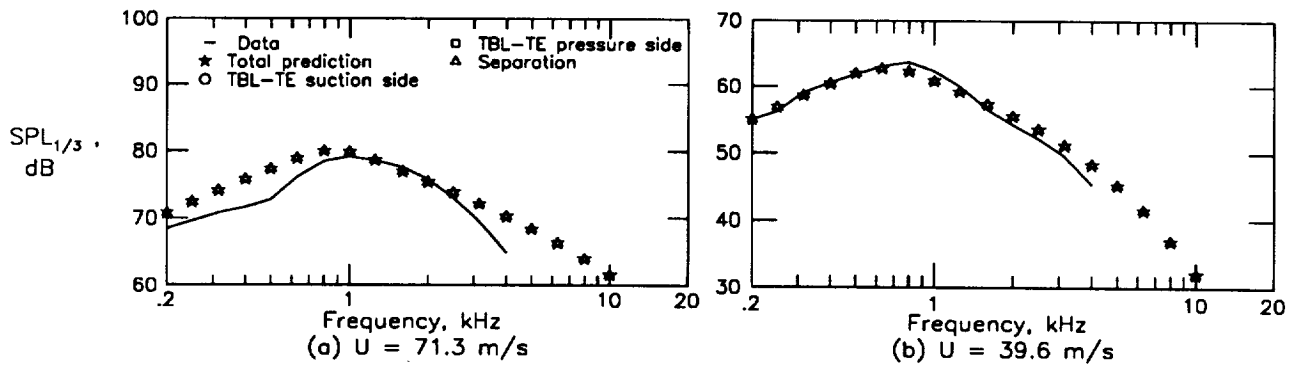


Figure 35. Self-noise spectra for 5.08-cm-chord airfoil with tripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 11.2^\circ$).

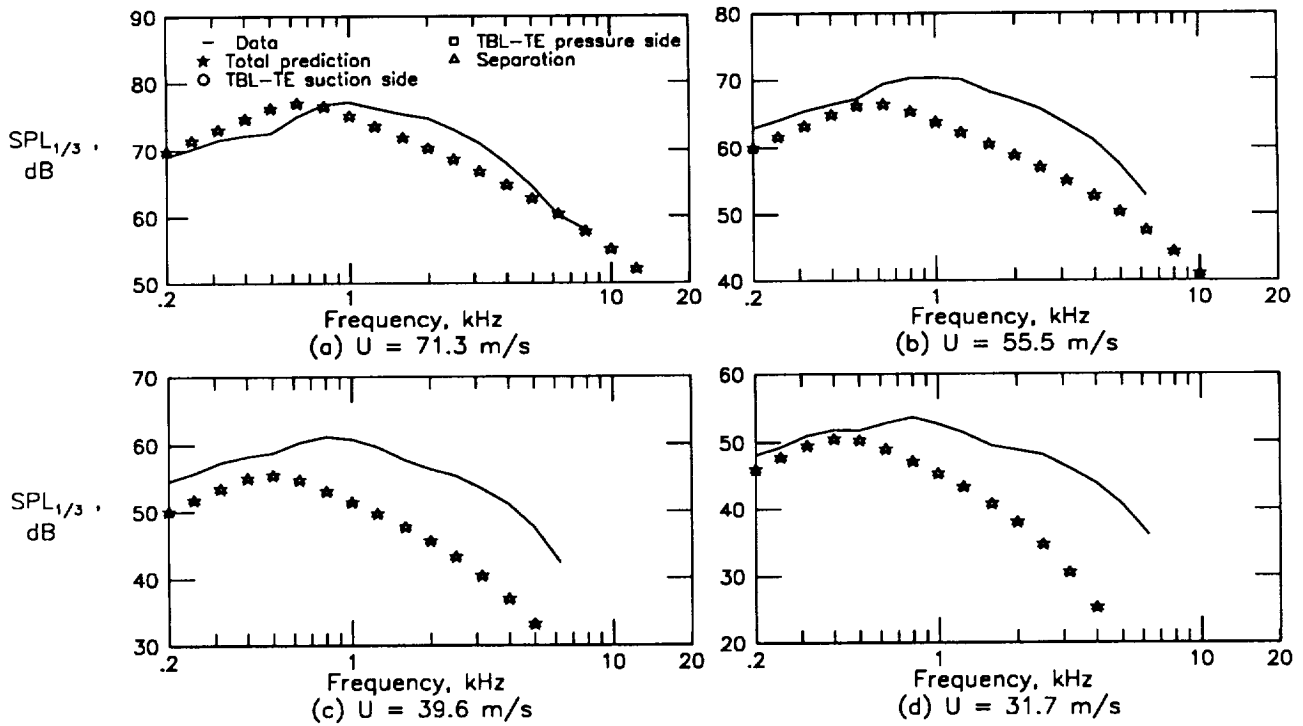


Figure 36. Self-noise spectra for 5.08-cm-chord airfoil with tripped BL at $\alpha_t = 19.8^\circ$ ($\alpha_* = 15.4^\circ$).

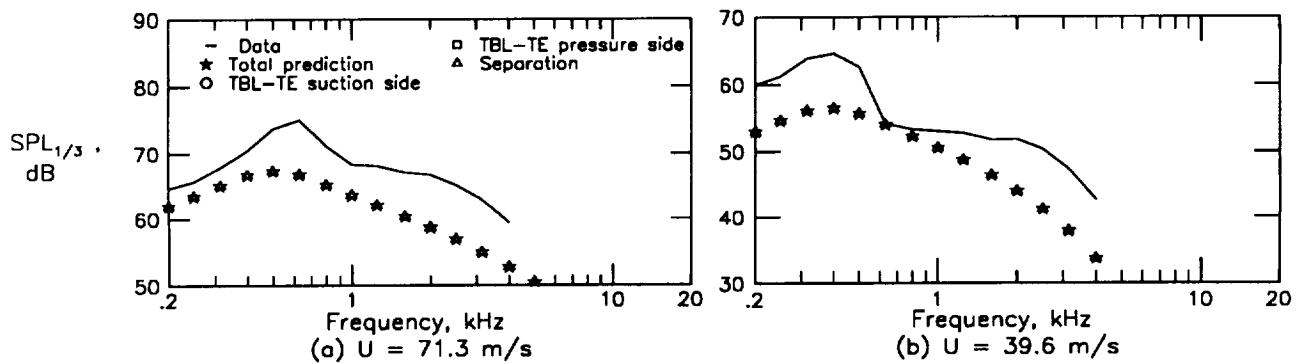


Figure 37. Self-noise spectra for 5.08-cm-chord airfoil with tripped BL at $\alpha_t = 25.2^\circ$ ($\alpha_* = 19.7^\circ$).

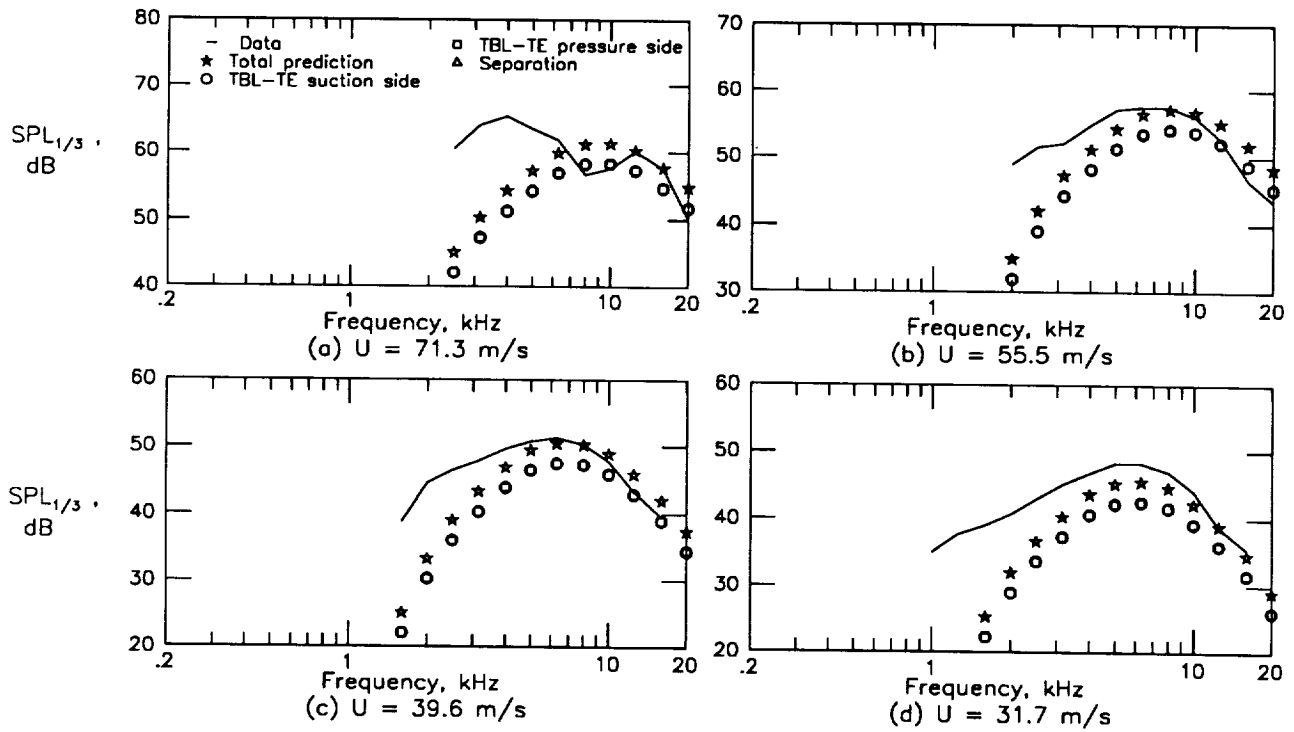


Figure 38. Self-noise spectra for 2.54-cm-chord airfoil with tripped BL at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

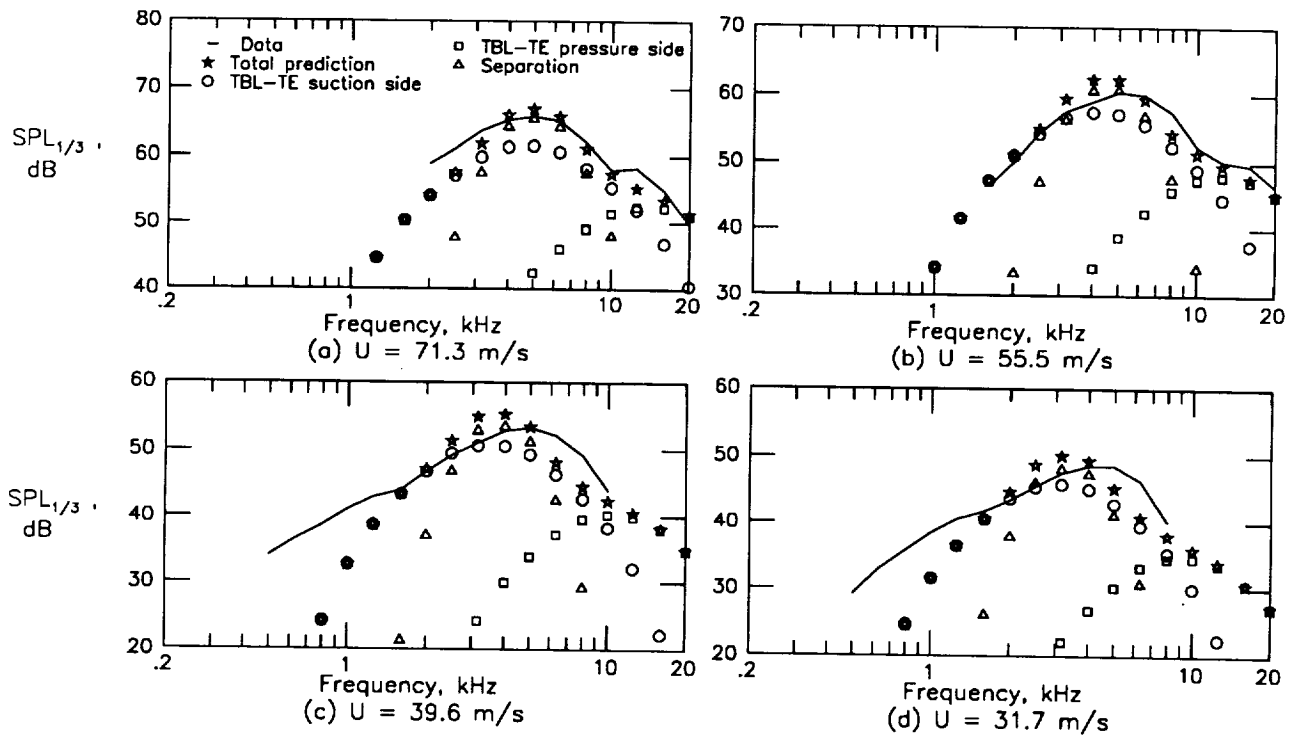


Figure 39. Self-noise spectra for 2.54-cm-chord airfoil with tripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 4.8^\circ$).

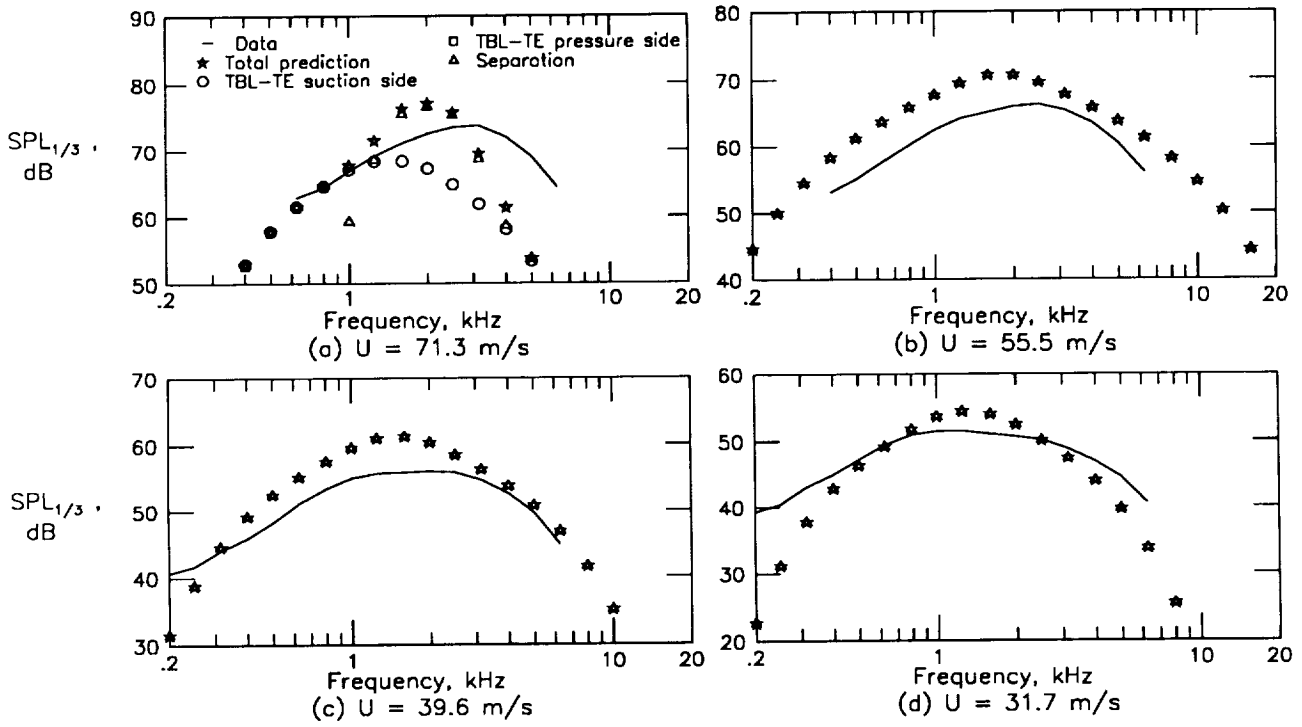


Figure 40. Self-noise spectra for 2.54-cm-chord airfoil with tripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 9.5^\circ$).

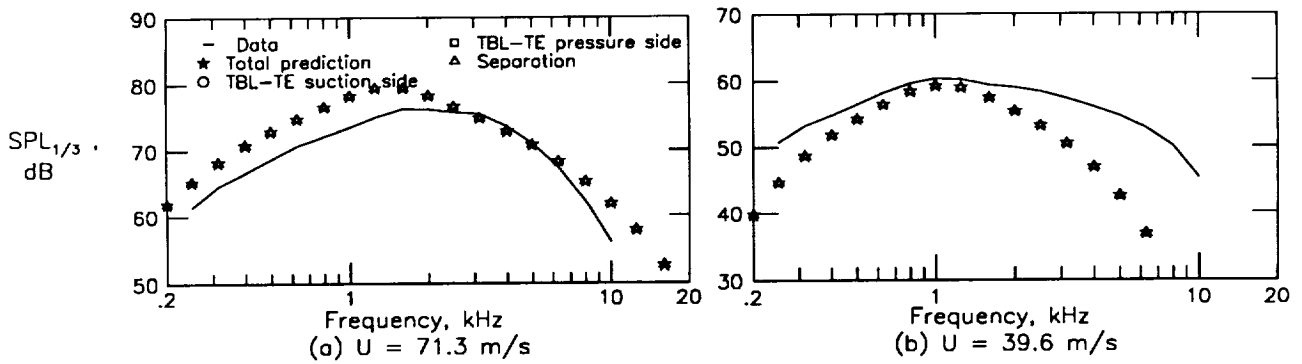


Figure 41. Self-noise spectra for 2.54-cm-chord airfoil with tripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 12.7^\circ$).

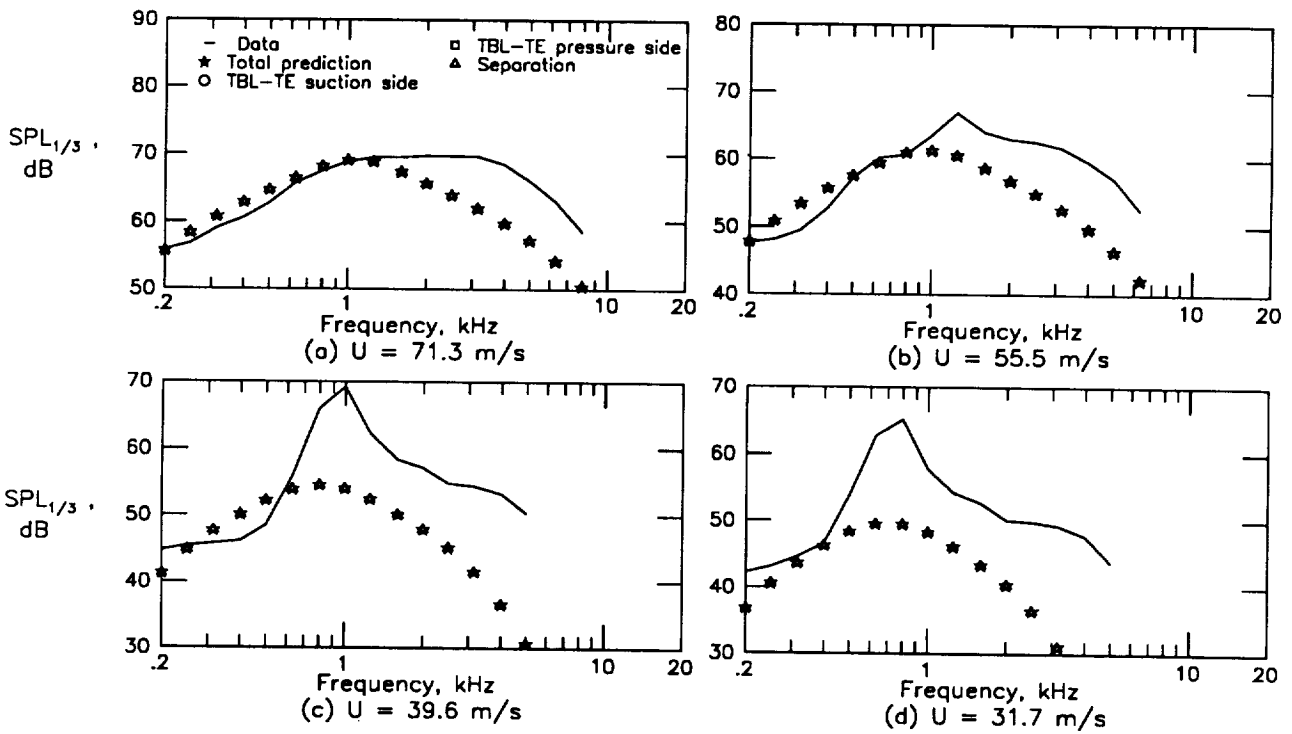


Figure 42. Self-noise spectra for 2.54-cm-chord airfoil with tripped BL at $\alpha_t = 19.8^\circ$ ($\alpha_* = 17.4^\circ$).

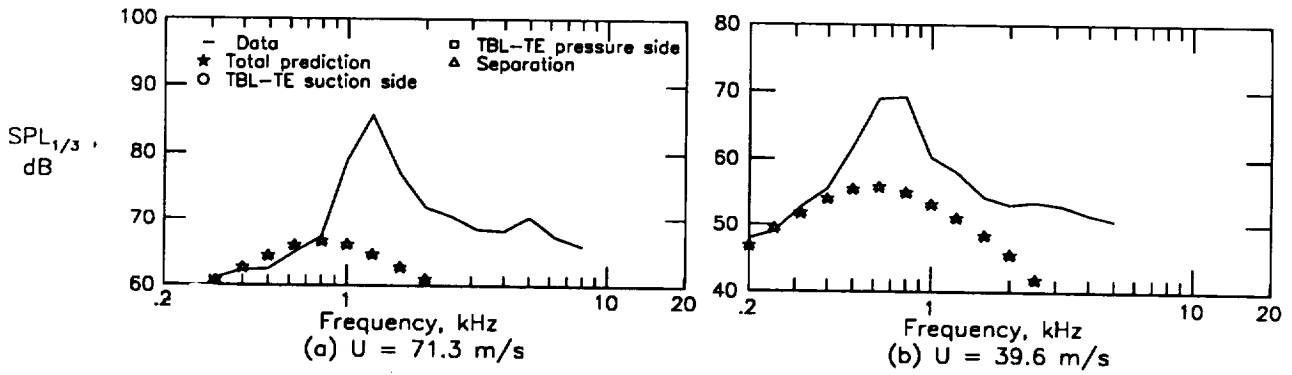


Figure 43. Self-noise spectra for 2.54-cm-chord airfoil with tripped BL at $\alpha_t = 25.2^\circ$ ($\alpha_* = 22.2^\circ$).

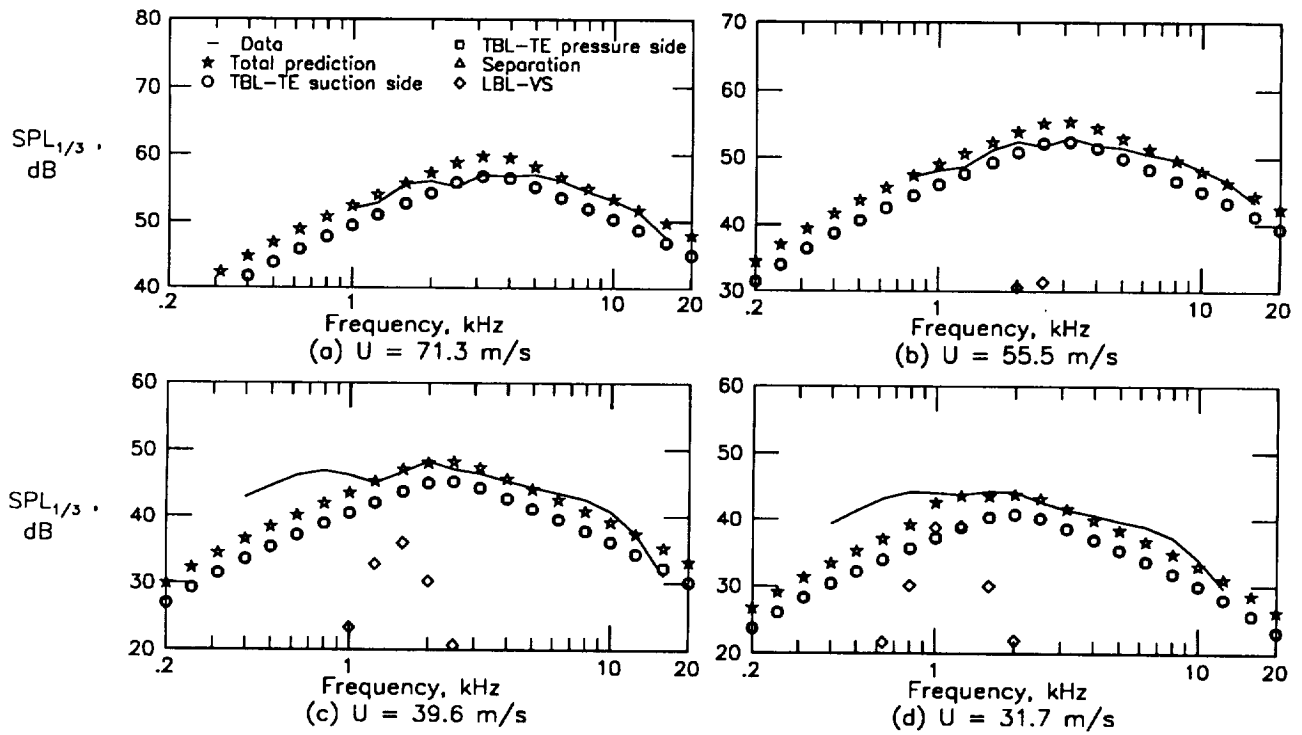


Figure 44. Self-noise spectra for 30.48-cm-chord airfoil with untripped BL (natural transition) at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

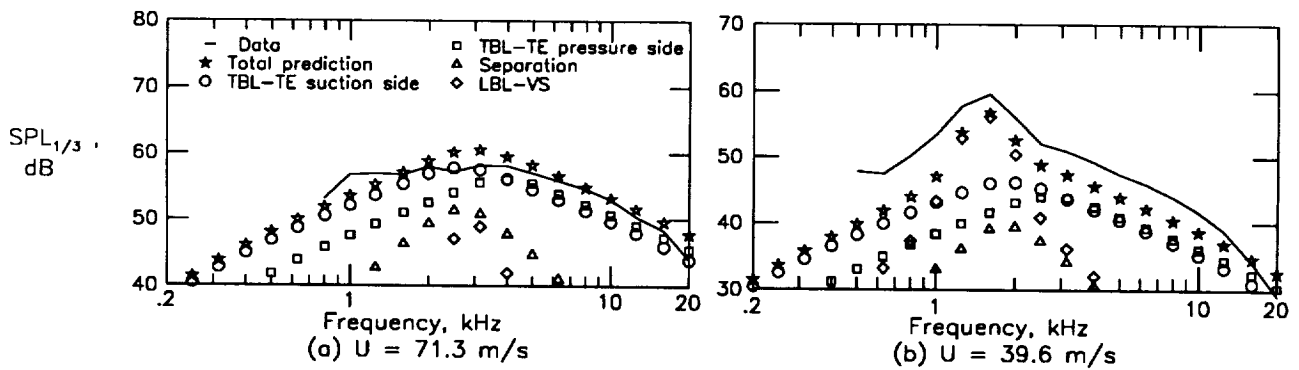


Figure 45. Self-noise spectra for 30.48-cm-chord airfoil with untripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 1.5^\circ$).

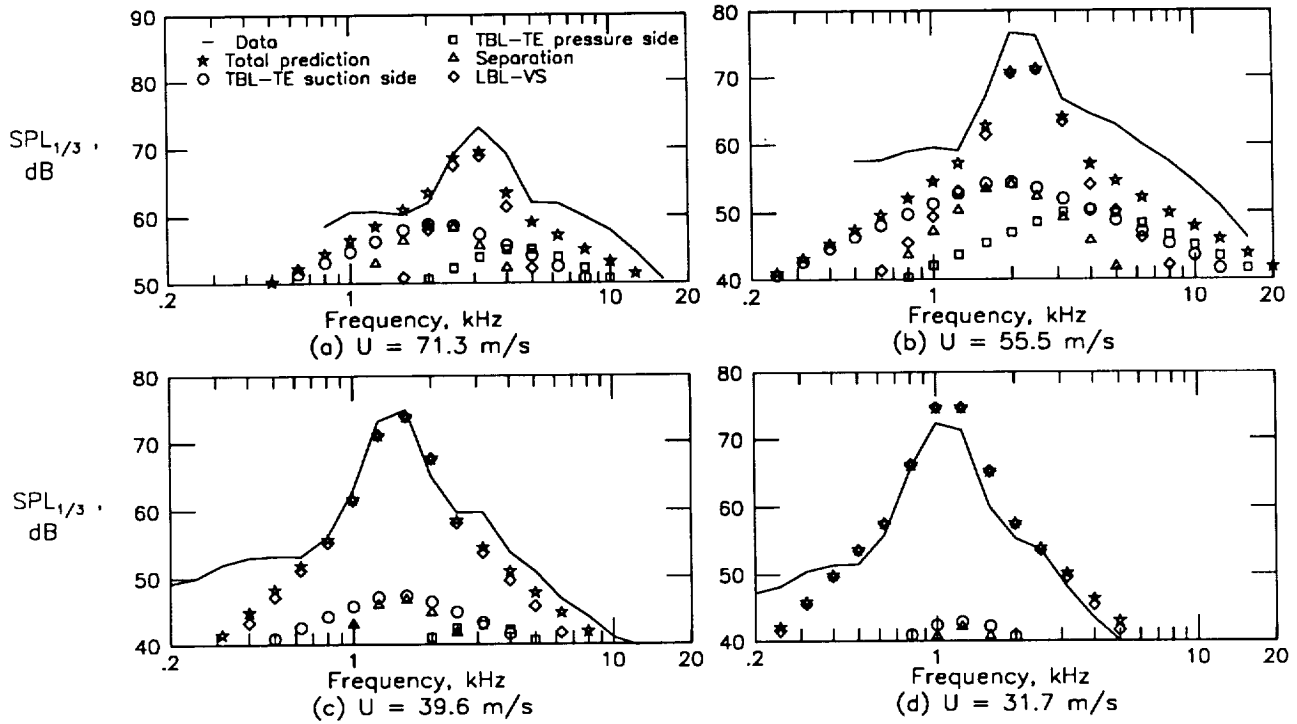


Figure 46. Self-noise spectra for 30.48-cm-chord airfoil with untripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 3.0^\circ$).

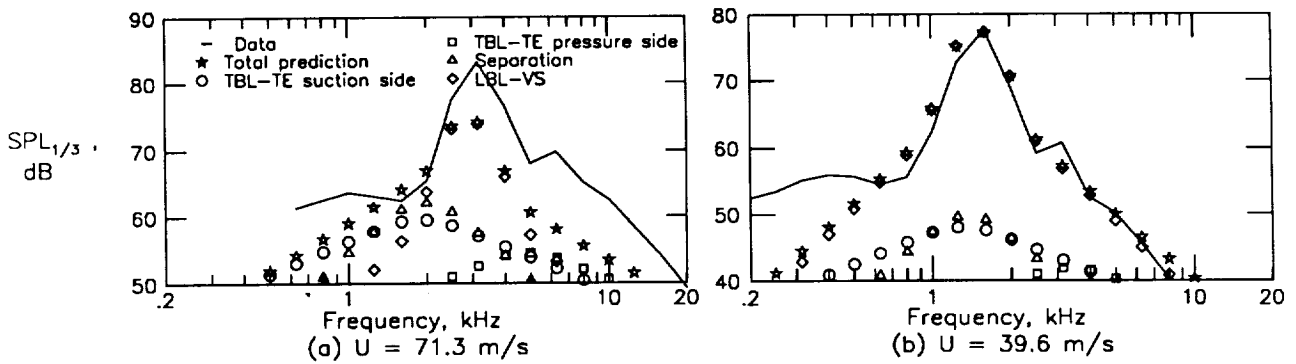


Figure 47. Self-noise spectra for 30.48-cm-chord airfoil with untripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 4.0^\circ$).

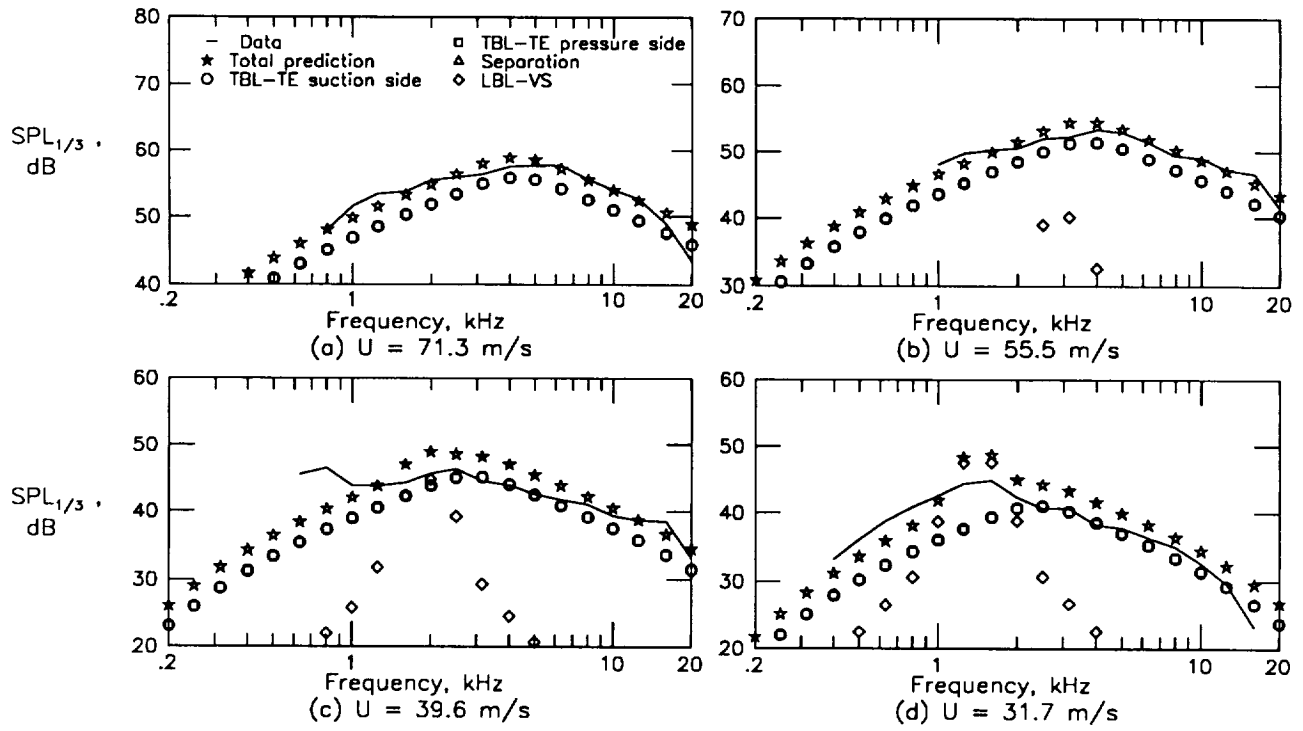


Figure 48. Self-noise spectra for 22.86-cm-chord airfoil with untripped BL at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

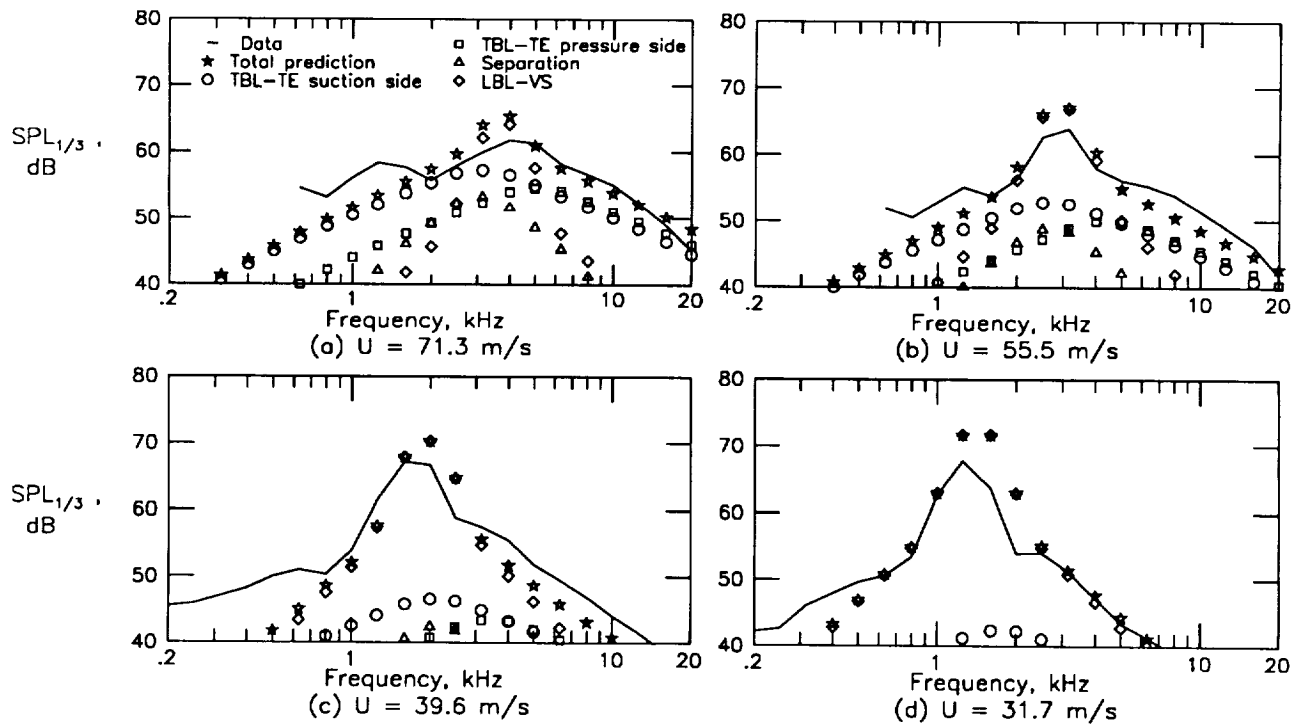


Figure 49. Self-noise spectra for 22.86-cm-chord airfoil with untripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 2.0^\circ$).

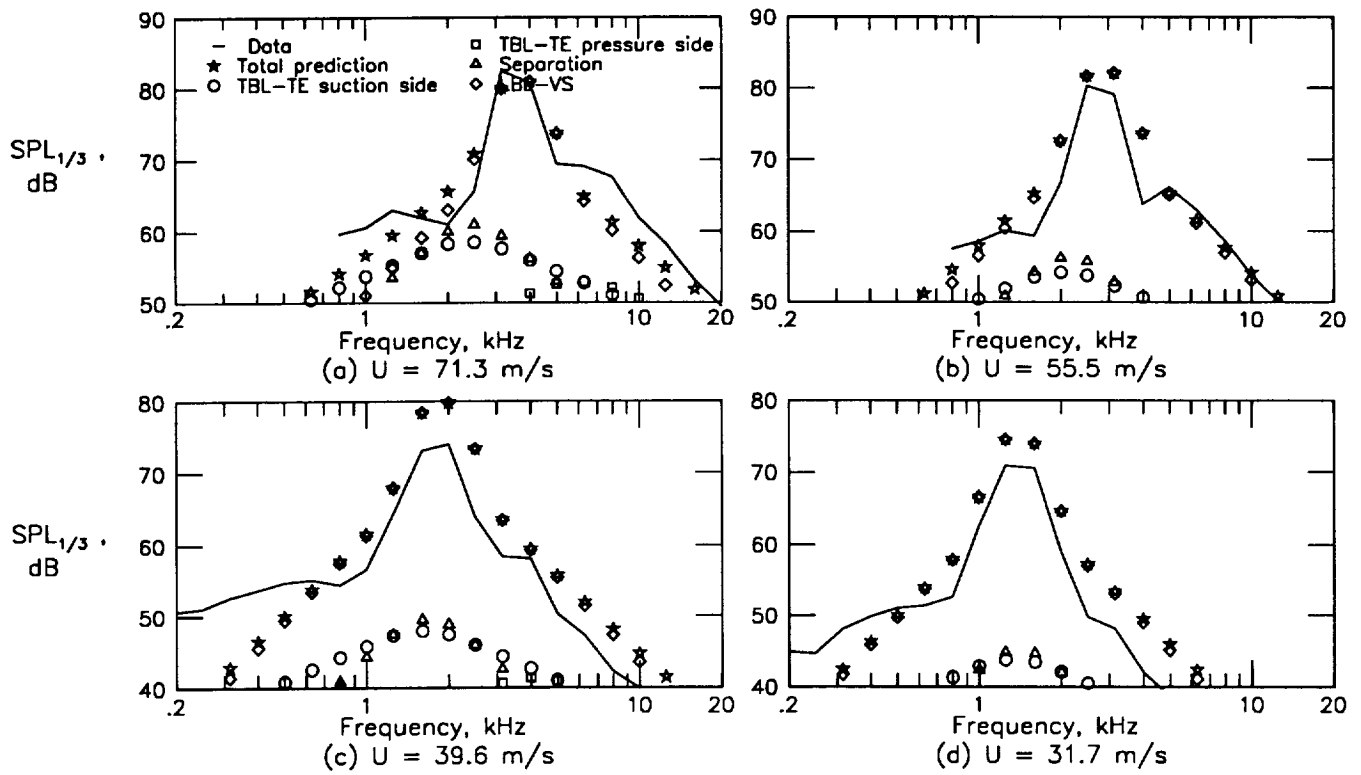


Figure 50. Self-noise spectra for 22.86-cm-chord airfoil with untripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 4.0^\circ$).

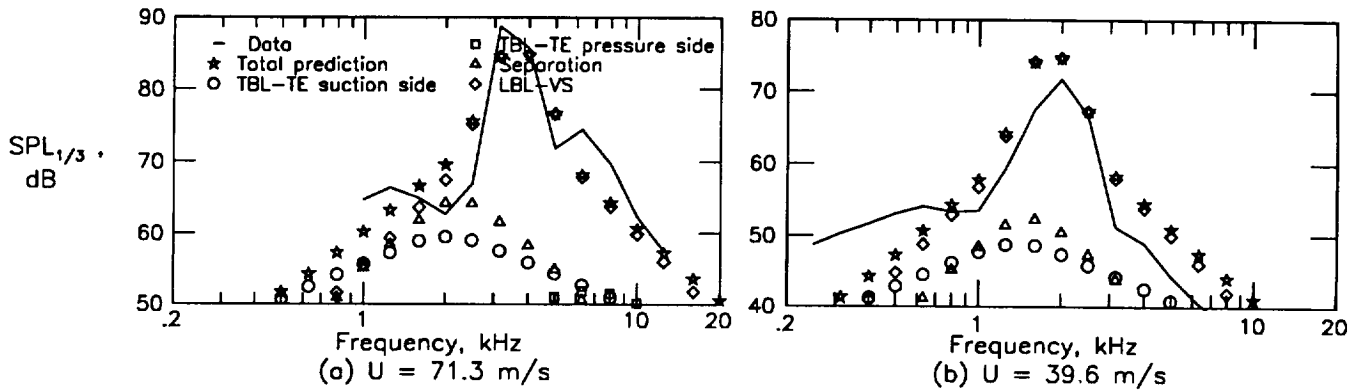


Figure 51. Self-noise spectra for 22.86-cm-chord airfoil with untripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 5.3^\circ$).

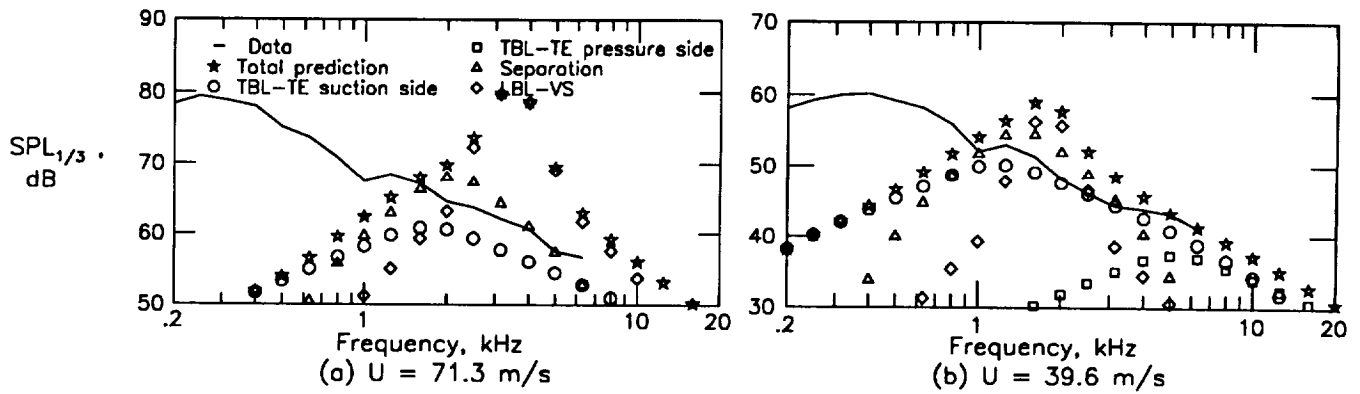


Figure 52. Self-noise spectra for 22.86-cm-chord airfoil with untripped BL at $\alpha_t = 19.8^\circ$ ($\alpha_* = 7.3^\circ$).

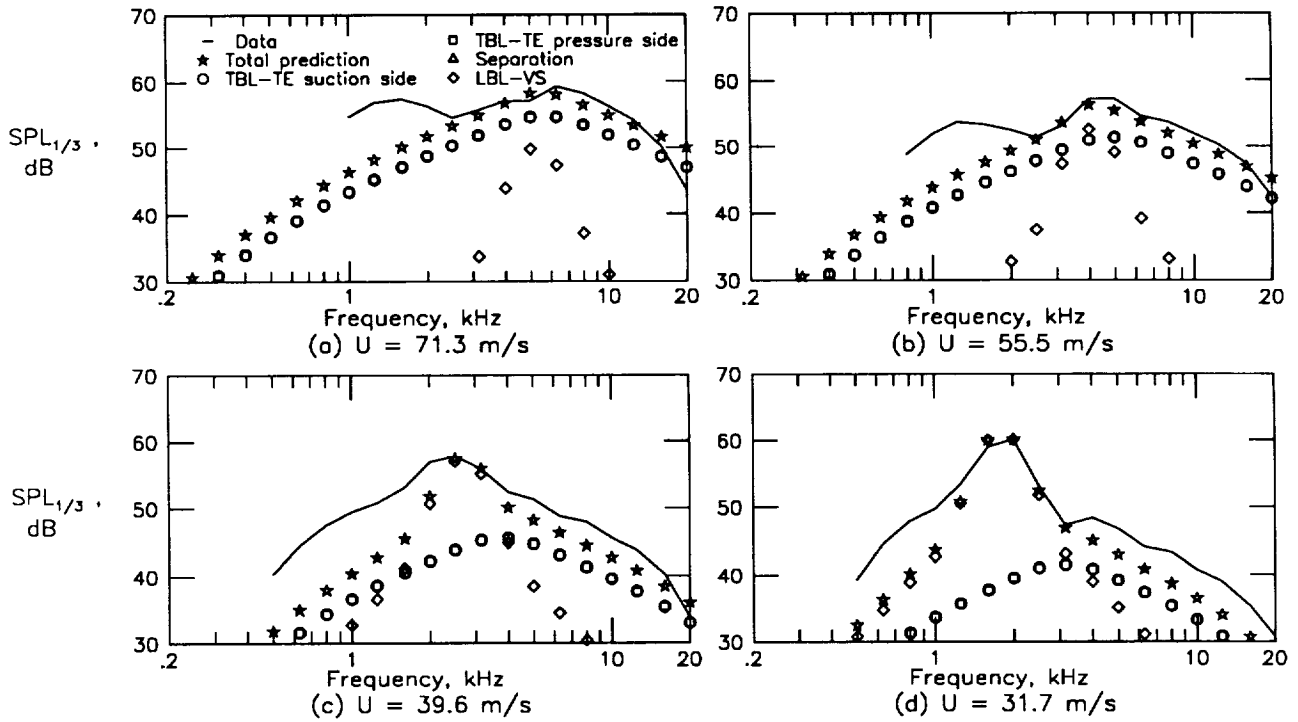


Figure 53. Self-noise spectra for 15.24-cm-chord airfoil with untripped BL at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

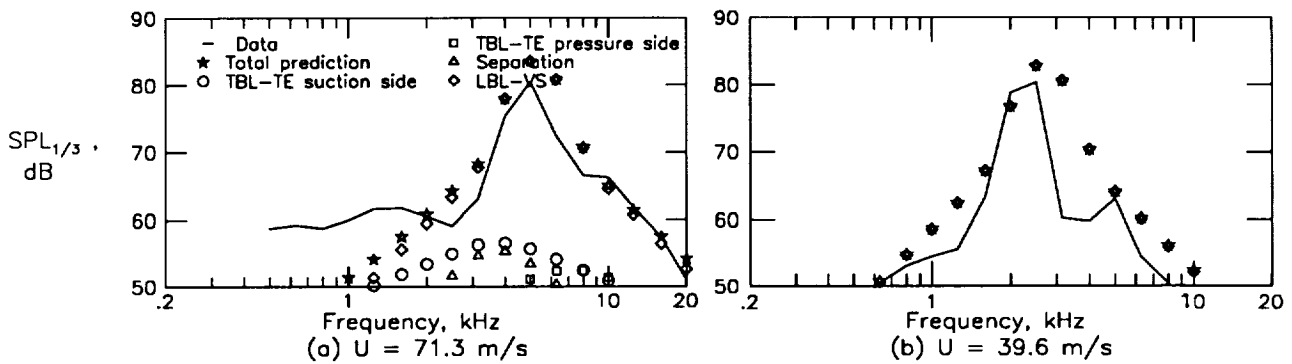


Figure 54. Self-noise spectra for 15.24-cm-chord airfoil with untripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 2.7^\circ$).

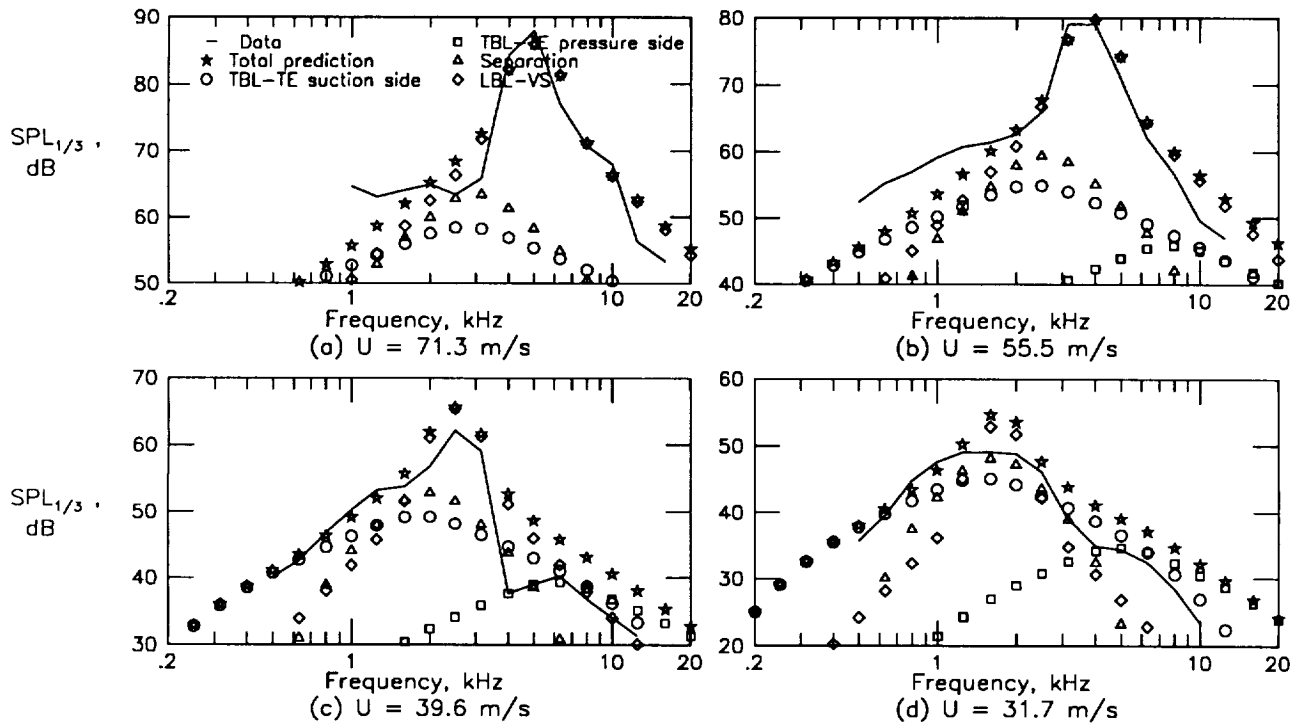


Figure 55. Self-noise spectra for 15.24-cm-chord airfoil with untripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 5.4^\circ$).

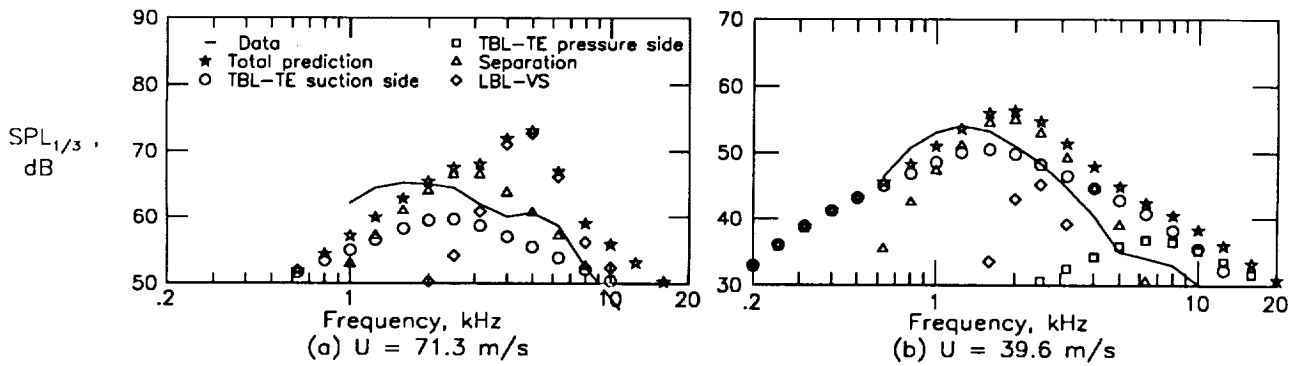


Figure 56. Self-noise spectra for 15.24-cm-chord airfoil with untripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 7.2^\circ$).

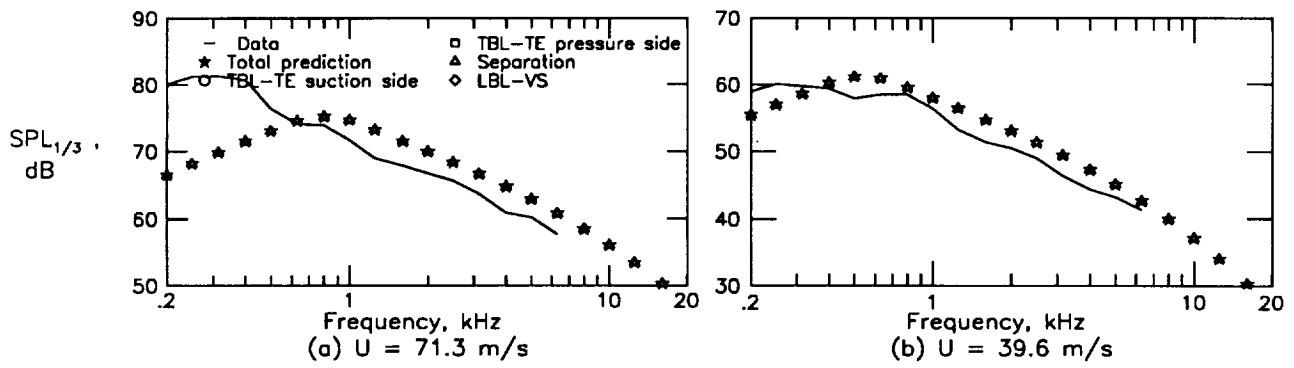


Figure 57. Self-noise spectra for 15.24-cm-chord airfoil with untripped BL at $\alpha_t = 19.8^\circ$ ($\alpha_* = 9.9^\circ$).

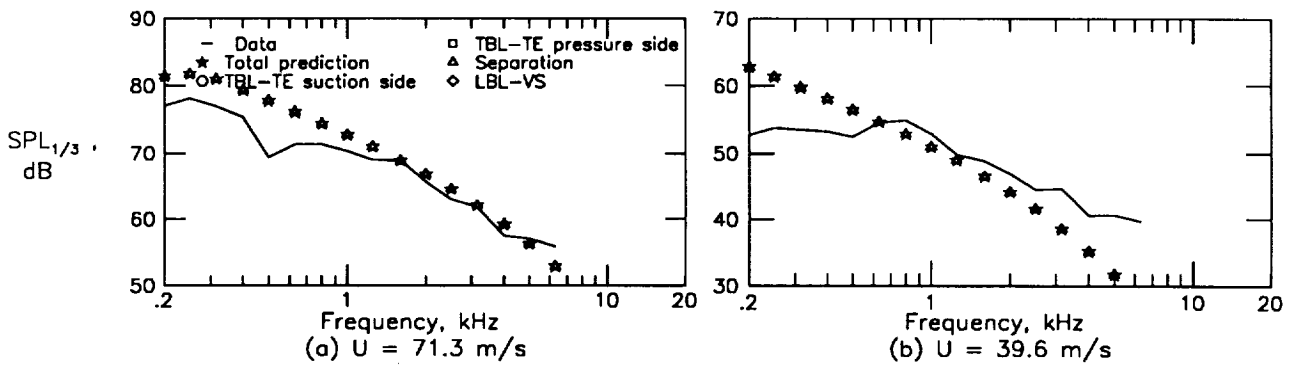


Figure 58. Self-noise spectra for 15.24-cm-chord airfoil with untripped BL at $\alpha_t = 25.2^\circ$ ($\alpha_* = 12.6^\circ$).

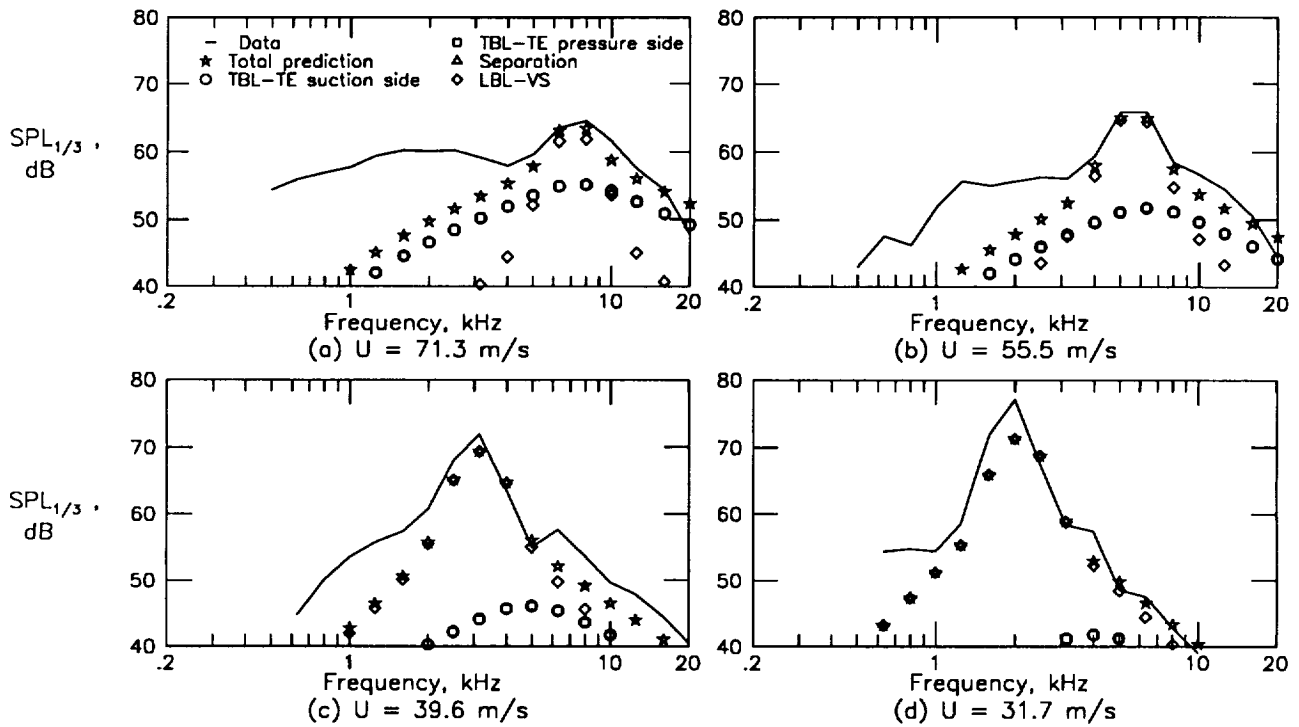


Figure 59. Self-noise spectra for 10.16-cm-chord airfoil with untripped BL at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

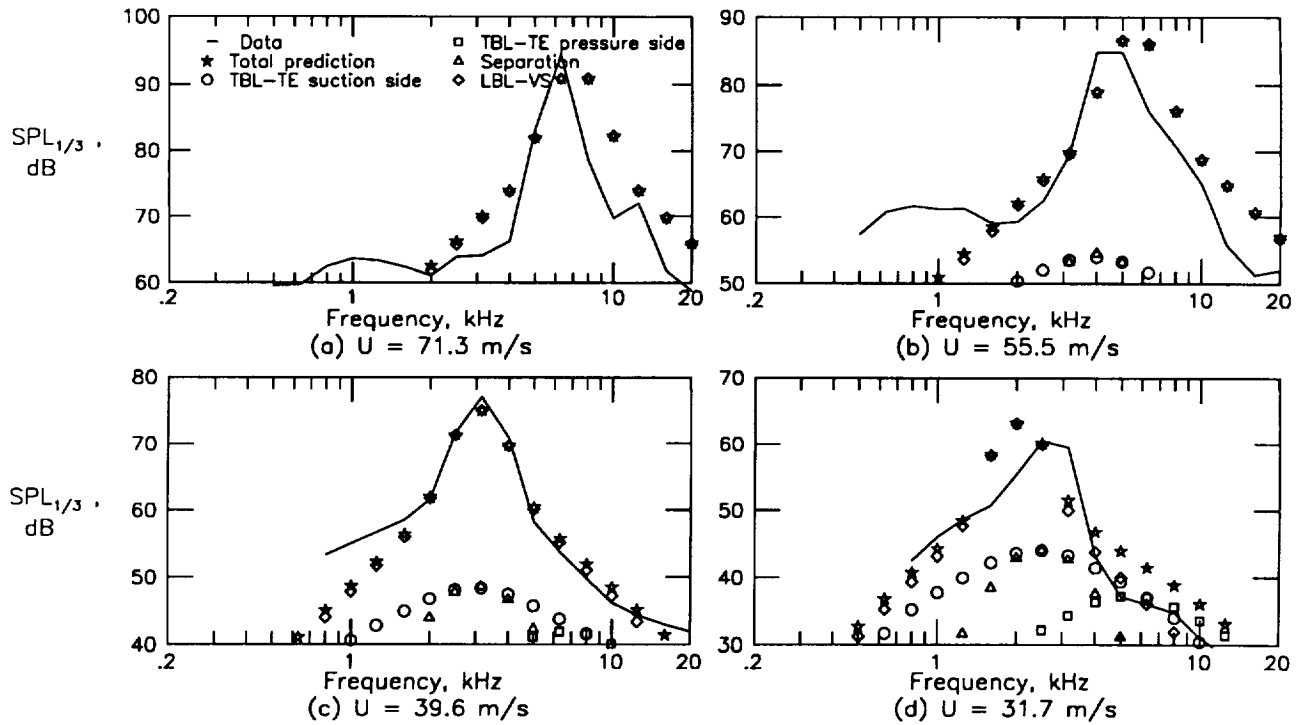


Figure 60. Self-noise spectra for 10.16-cm-chord airfoil with untripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 3.3^\circ$).

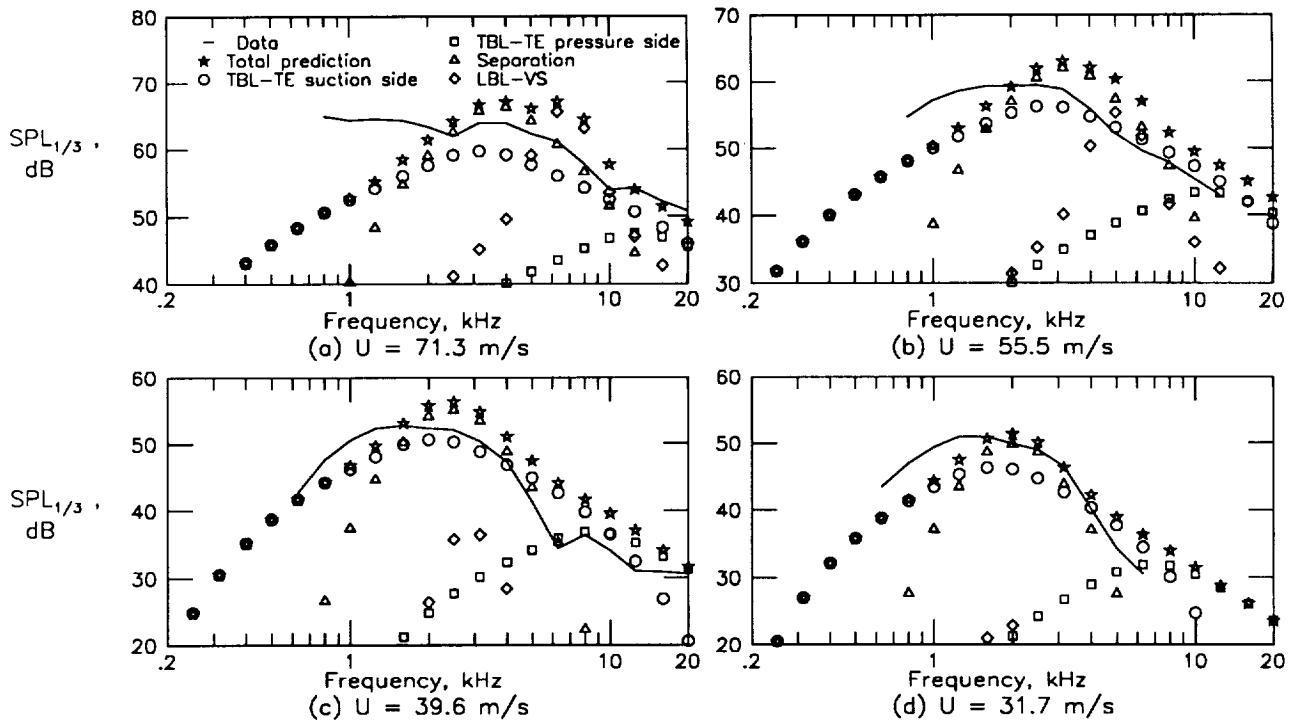


Figure 61. Self-noise spectra for 10.16-cm-chord airfoil with untripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 6.7^\circ$).

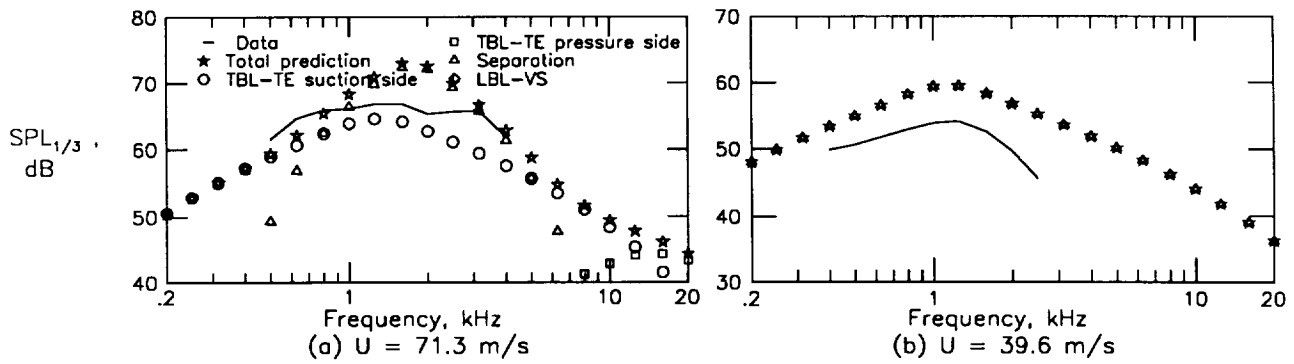


Figure 62. Self-noise spectra for 10.16-cm-chord airfoil with untripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 8.9^\circ$).

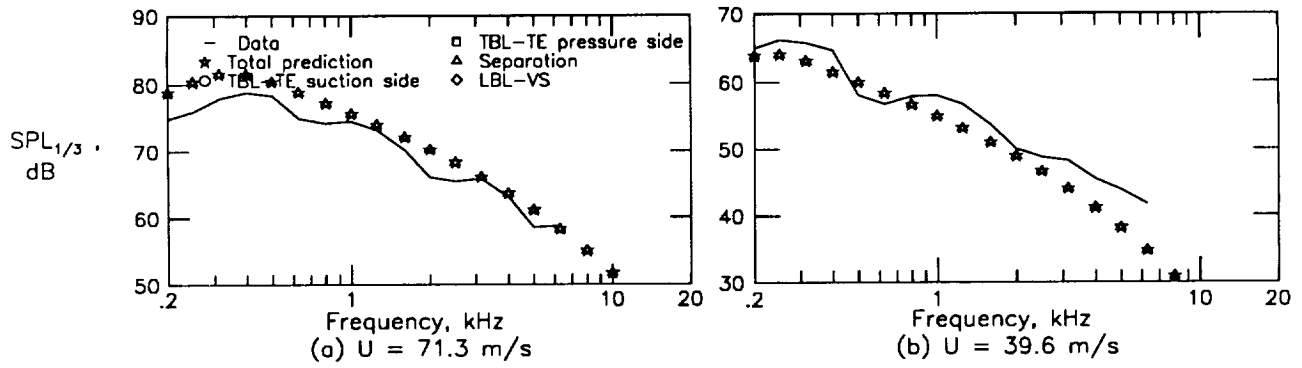


Figure 63. Self-noise spectra for 10.16-cm-chord airfoil with untripped BL at $\alpha_t = 19.8^\circ$ ($\alpha_* = 12.3^\circ$).

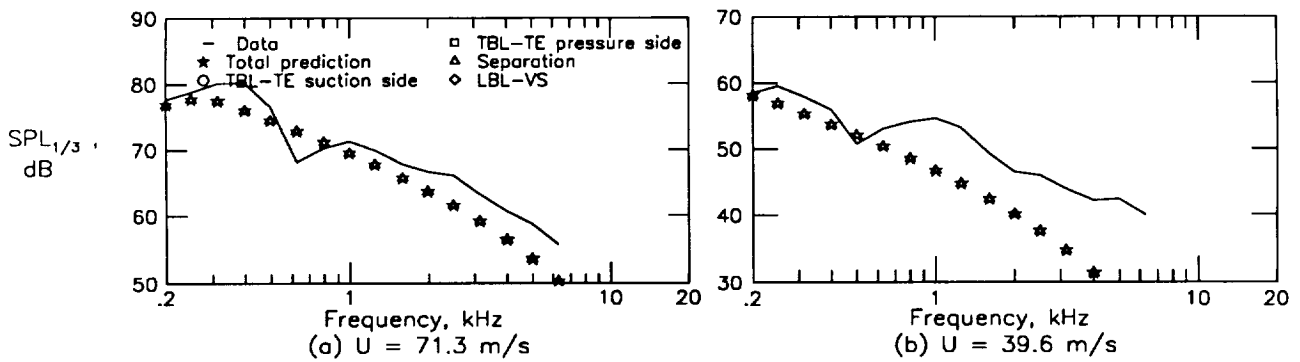


Figure 64. Self-noise spectra for 10.16-cm-chord airfoil with untripped BL at $\alpha_t = 25.2^\circ$ ($\alpha_* = 15.6^\circ$).

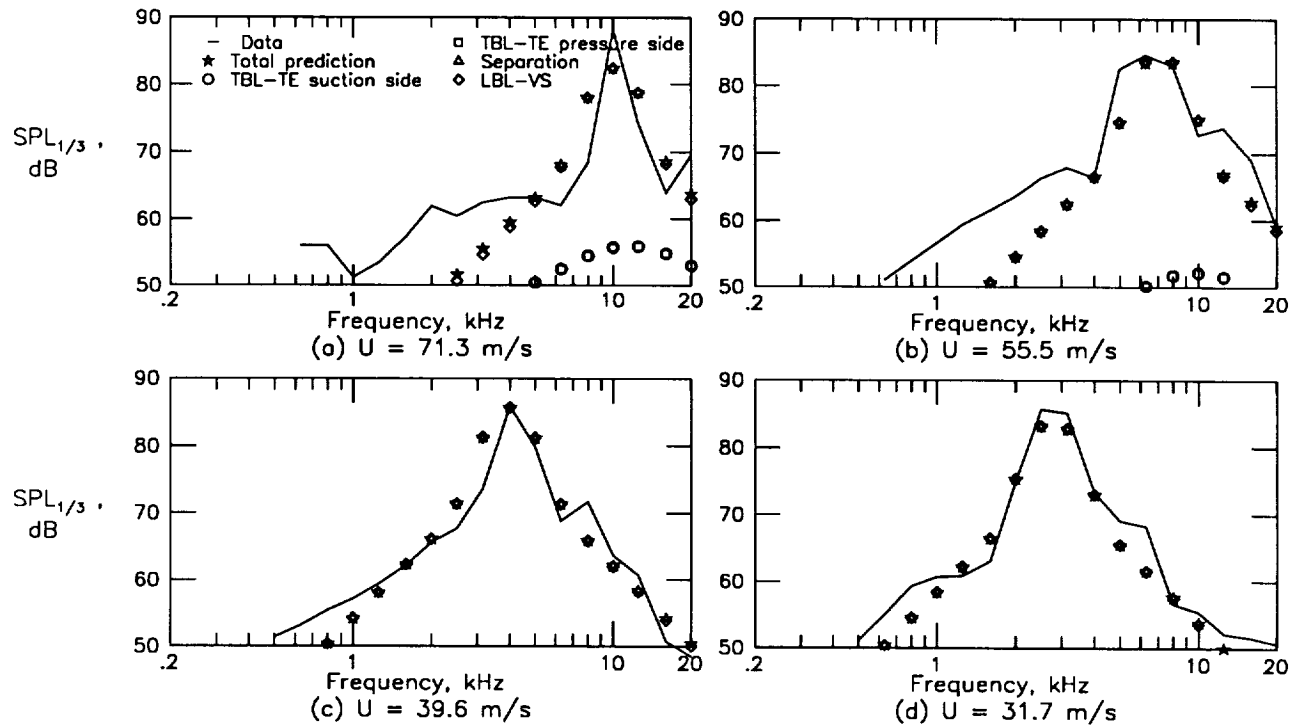


Figure 65. Self-noise spectra for 5.08-cm-chord airfoil with untripped BL at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

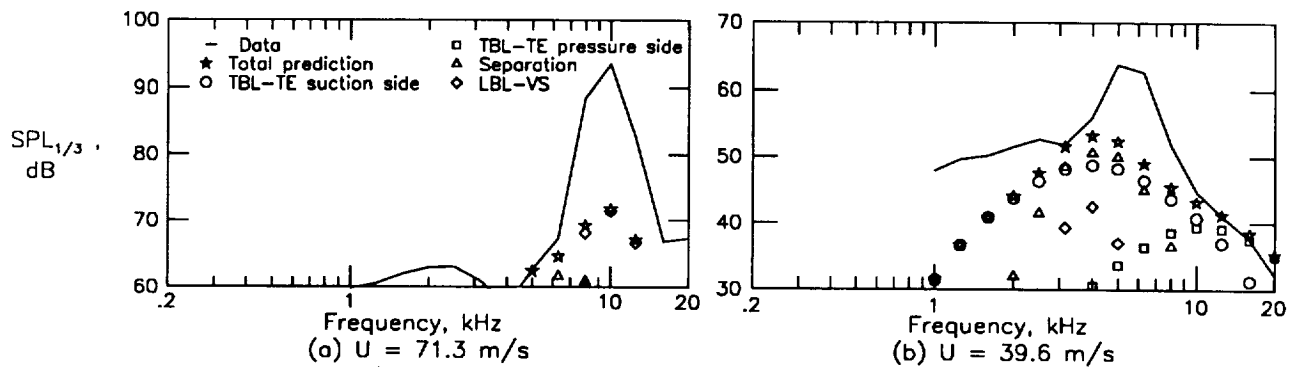


Figure 66. Self-noise spectra for 5.08-cm-chord airfoil with untripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 4.2^\circ$).

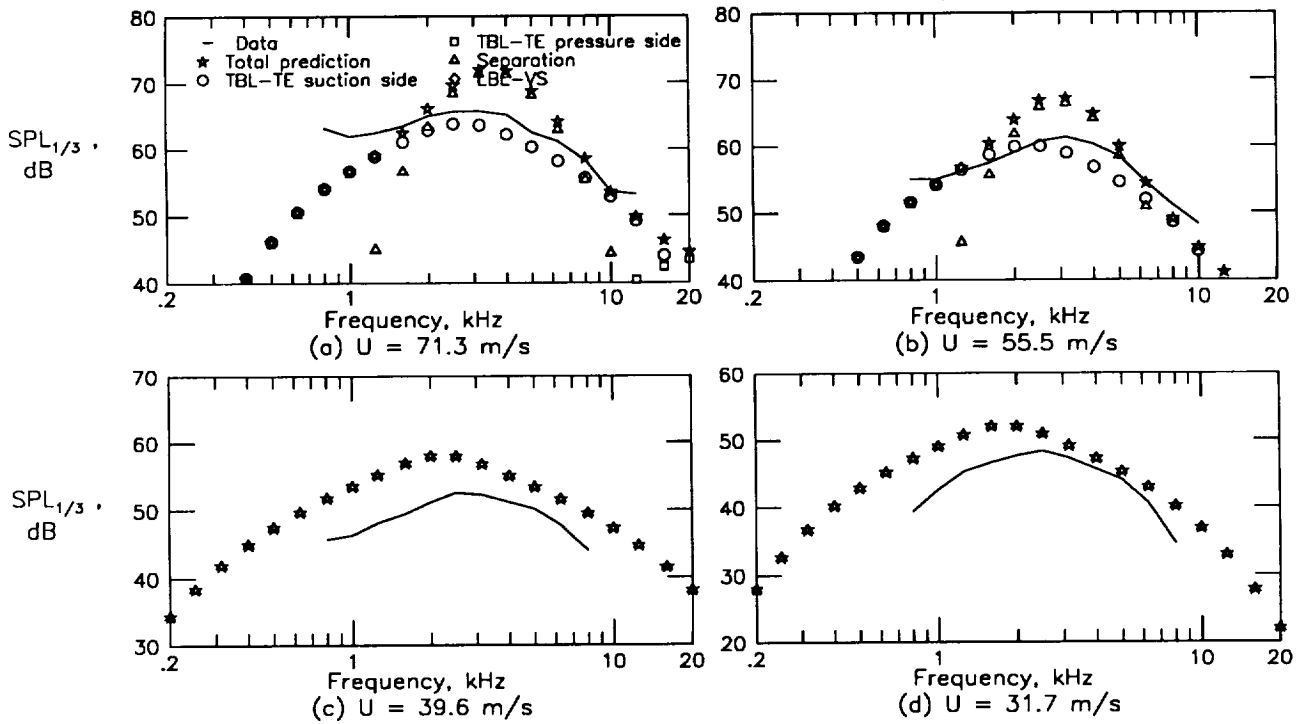


Figure 67. Self-noise spectra for 5.08-cm-chord airfoil with untripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 8.4^\circ$).

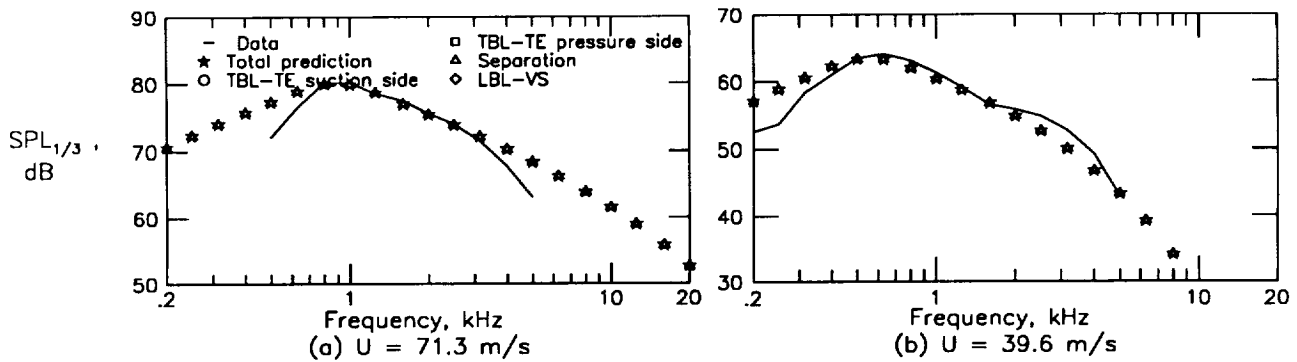


Figure 68. Self-noise spectra for 5.08-cm-chord airfoil with untripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 11.2^\circ$).

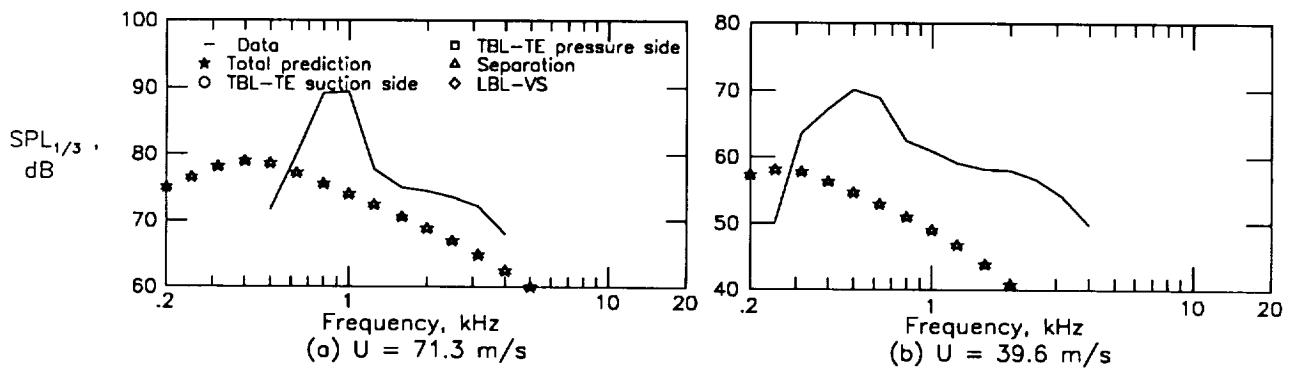


Figure 69. Self-noise spectra for 5.08-cm-chord airfoil with untripped BL at $\alpha_t = 19.8^\circ$ ($\alpha_* = 15.4^\circ$).

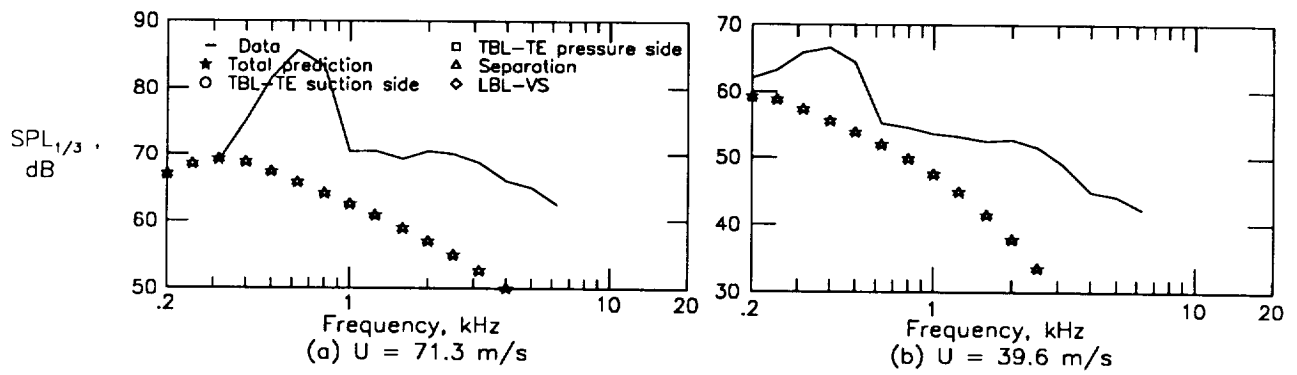


Figure 70. Self-noise spectra for 5.08-cm-chord airfoil with untripped BL at $\alpha_t = 25.2^\circ$ ($\alpha_* = 19.6^\circ$).

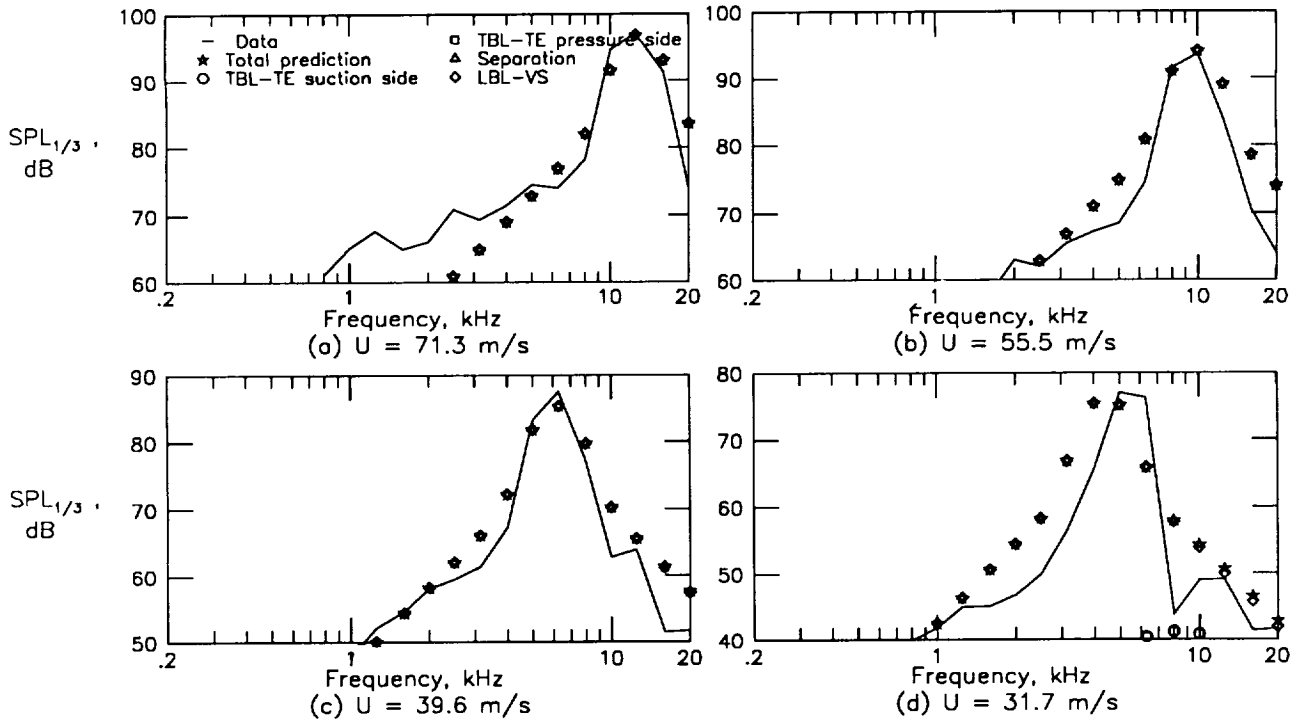


Figure 71. Self-noise spectra for 2.54-cm-chord airfoil with untripped BL at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$).

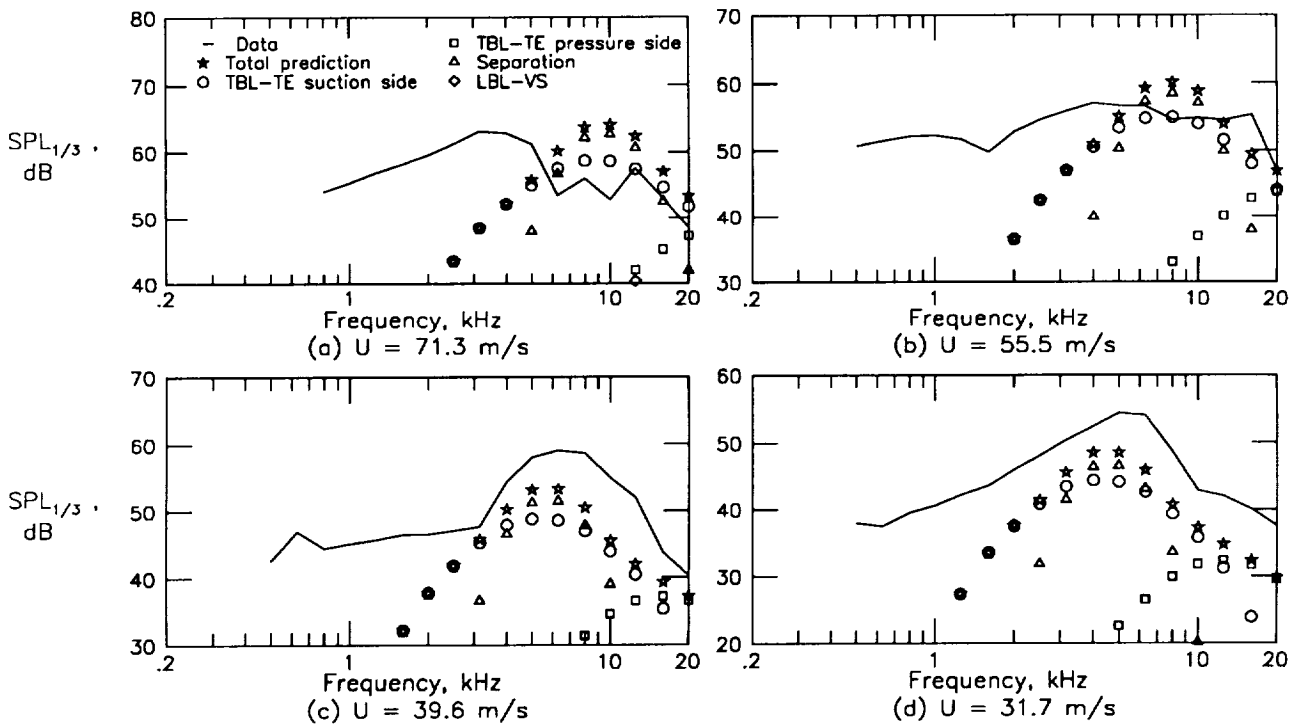


Figure 72. Self-noise spectra for 2.54-cm-chord airfoil with untripped BL at $\alpha_t = 5.4^\circ$ ($\alpha_* = 4.8^\circ$).

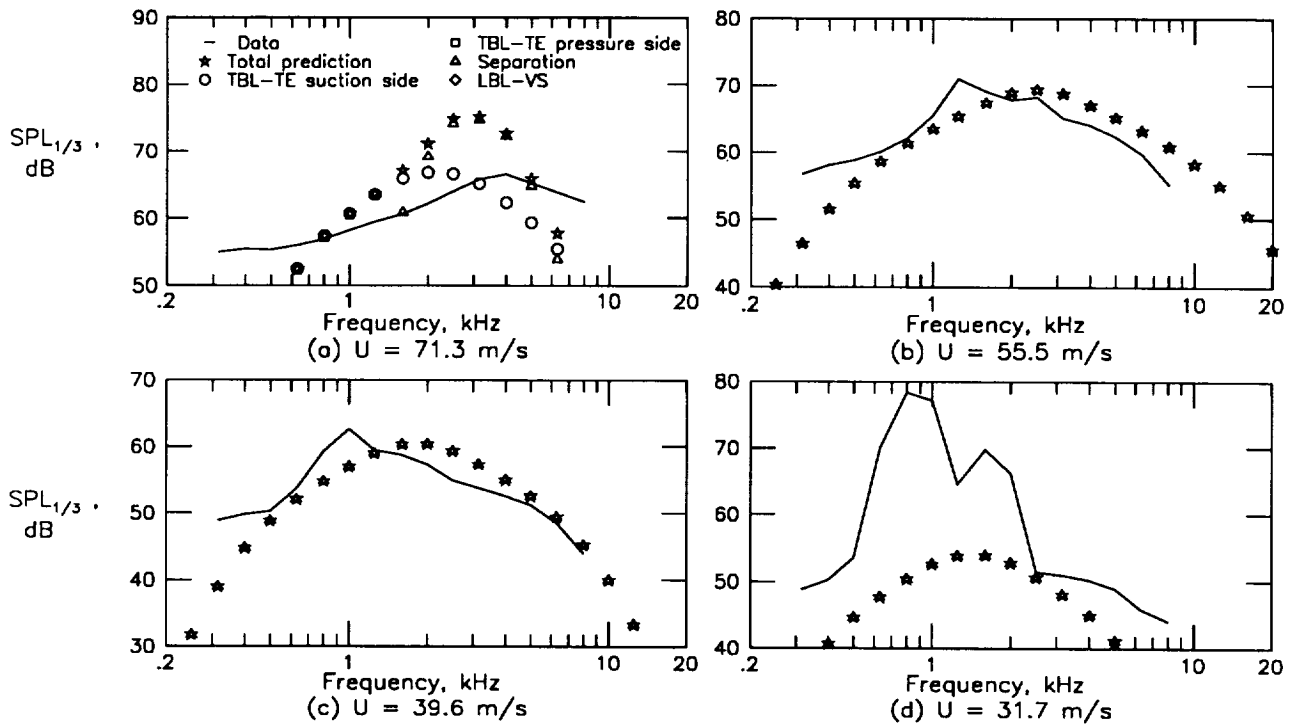


Figure 73. Self-noise spectra for 2.54-cm-chord airfoil with untripped BL at $\alpha_t = 10.8^\circ$ ($\alpha_* = 9.5^\circ$).

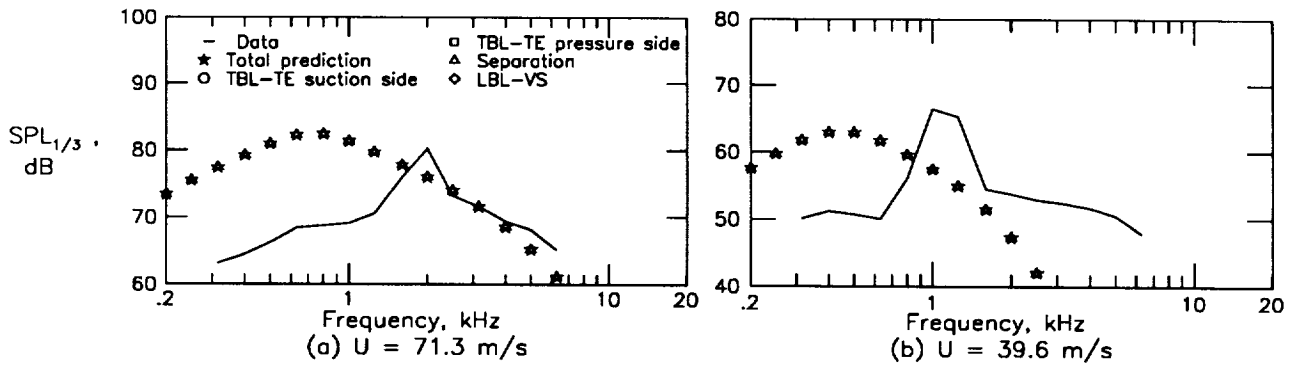


Figure 74. Self-noise spectra for 2.54-cm-chord airfoil with untripped BL at $\alpha_t = 14.4^\circ$ ($\alpha_* = 12.7^\circ$).

5. Spectral Scaling

In this section, the scaling laws are developed for the five self-noise mechanisms. The spectra of figures 11 to 74 form the basis of the scaling for three of the mechanisms: turbulent-boundary-layer-trailing-edge (TBL-TE) noise and separation noise were scaled from the tripped boundary-layer cases, and laminar-boundary-layer-vortex-shedding (LBL-VS) noise was scaled from the untripped cases. For the tip vortex formation noise mechanism, both the data and the scaling approach are obtained from reference 18. Finally, for TE-bluntness-vortex-shedding noise, spectral data from the study of reference 2, as well as previously unpublished data from that study, form the basis of scaling analysis.

5.1. Turbulent-Boundary-Layer-Trailing-Edge Noise and Separated Flow Noise

What has become traditional TE noise scaling is based on the analysis of Ffowcs Williams and Hall (ref. 5). For the problem of turbulence convecting at low subsonic velocity U_c above a large plate and past the trailing edge into the wake, the primary result is

$$\langle p^2 \rangle \propto \rho_0^2 v'^2 \frac{U_c^3}{c_0} \left(\frac{L\mathcal{L}}{r^2} \right) \bar{D} \quad (17)$$

where $\langle p^2 \rangle$ is the mean-square sound pressure at the observer located a distance r from the edge. The medium density is ρ_0 , v'^2 is the mean-square turbulence velocity, c_0 is the speed of sound, L is the spanwise extent wetted by the flow, and \mathcal{L} is a characteristic turbulence correlation scale. The directivity factor \bar{D} equals 1 for observers normal to the surface from the TE. The usual assumptions for boundary-layer flow are that $v' \propto U_c \propto U$ and $\mathcal{L} \propto \delta$ or δ^* , where δ and δ^* are, respectively, the boundary-layer thickness and displacement thickness. Fink (ref. 25), when normalizing airframe noise data where TBL-TE noise was believed to be dominant, assumed a universal spectrum shape $F(\text{St})$ for the noise, where St is the Strouhal number $f\delta/U$. The shape $F(\text{St})$ depended only on the ratio of St to its peak value St_{peak} . This gave the following normalization form for the 1/3-octave sound pressure level spectral presentation:

$$\text{SPL}_{1/3} - 10 \log \left[\left(\frac{U}{100} \right)^5 \frac{\delta L}{r^2} \right] = F(\text{St}) + K \quad (18)$$

with $\text{SPL}_{1/3} = \text{OASPL} + F(\text{St})$ and where K is an empirical constant which was determined when the velocity U is given in units of knots.

As mentioned in section 1, some of the airfoil self-noise spectral data of the present report were presented, in uncorrected form, in reference 6, and normalized in the manner of equation (18) using measured values of δ . It was found that, contrary to what was previously assumed (e.g., refs. 25 and 3), the normalized levels, spectral shape, and Strouhal number were not independent of airfoil size, airfoil angle of attack, and free-stream velocity. However, the limited scope of the paper, as well as the uncertainty caused by the aforementioned extraneous noise contamination of the uncorrected spectra, prevented a clear definition of the functional dependences. The corrected spectra of the present report are used to determine the parametric dependences and to account for these in the spectral scaling.

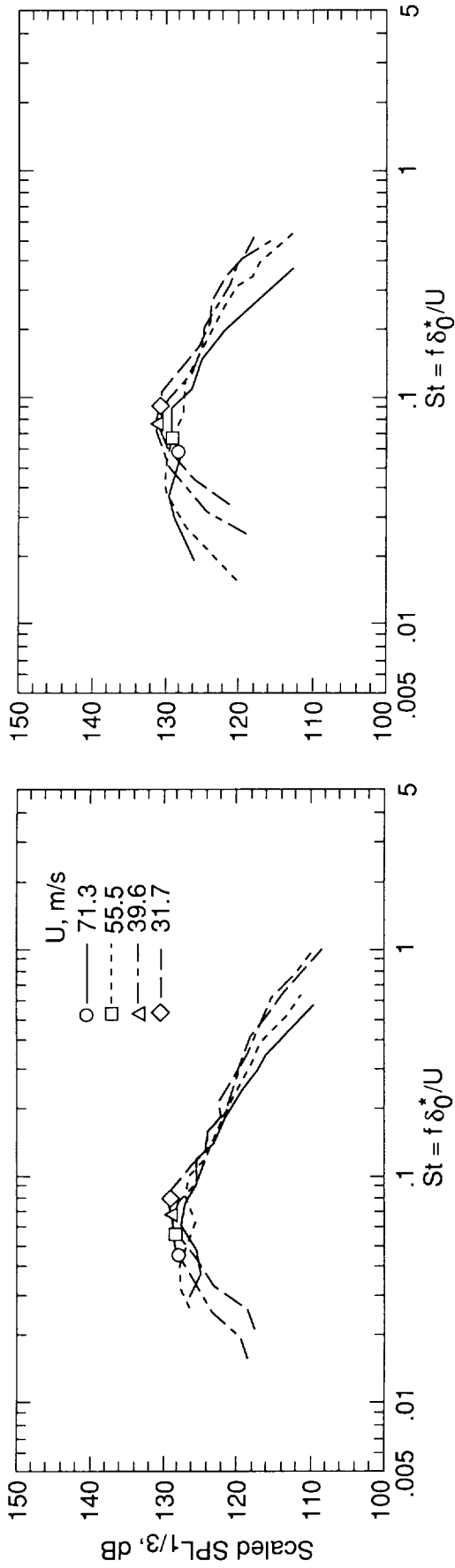
5.1.1. Scaled Data

Zero angle of attack. In figure 75, 1/3-octave spectra for four airfoil sizes, each at four tunnel speeds, are scaled. The spectra are obtained from figures 11, 20, 26, and 32. The angle of attack is zero and the boundary layers are tripped. The form of the normalization is

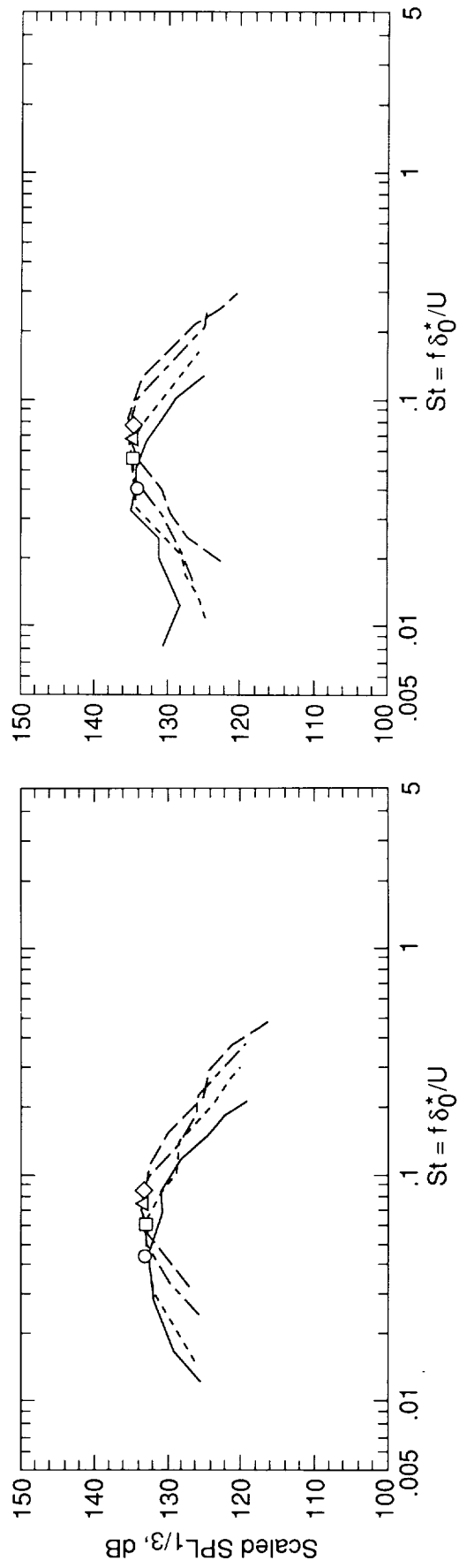
$$\text{Scaled SPL}_{1/3} = \text{SPL}_{1/3} - 10 \log \left(M^5 \frac{\delta_0^* L}{r_e^2} \right) \quad (19)$$

where Mach number replaces the velocity in knots, δ_0^* replaces δ , and r_e replaces r . The retarded observer distance r_e equals here the measured value, 122 cm (see appendix B). For the right side of equation (19) to be accurately expressible by the form $F(\text{St}) + K$ of equation (18), the scaled spectra of figure 75 should be identical to one another for all cases. However, the peak Strouhal number, spectral shape, and scaled level vary significantly.

For each spectrum in figure 75, a symbol indicates the approximate spectral peak location. The peak locations were based on gross spectral shapes and trends rather than specific peak maximums. The peak Strouhal number, $\text{St}_{\text{peak}} = (f\delta^*/U)_{\text{peak}}$, and scaled levels corresponding to these peak locations are shown in figures 76 and 77, respectively, as a function of Reynolds number R_c . These data are also presented in table 1 (at the back of this report). Included in the figures are the other cases for tripped BL airfoils of different chord lengths. Also included are data at nonzero angle of attack for subsequent discussion. The displacement thicknesses for the suction side, δ_s^* , are used for these normalizations. In figure 76, St_{peak} for zero angle of attack (solid symbols) shows no clear R_c -dependence, but a Mach number dependence is apparent. The horizontal lines through the data correspond to the function



(a) 30.48-cm-chord airfoil.



(b) 15.24-cm-chord airfoil.

(c) 10.16-cm-chord airfoil.

(d) 5.08-cm-chord airfoil.

Figure 75. Scaled 1/3-octave spectra of tripped BL airfoils at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$). Symbols indicate approximate spectral peak locations.

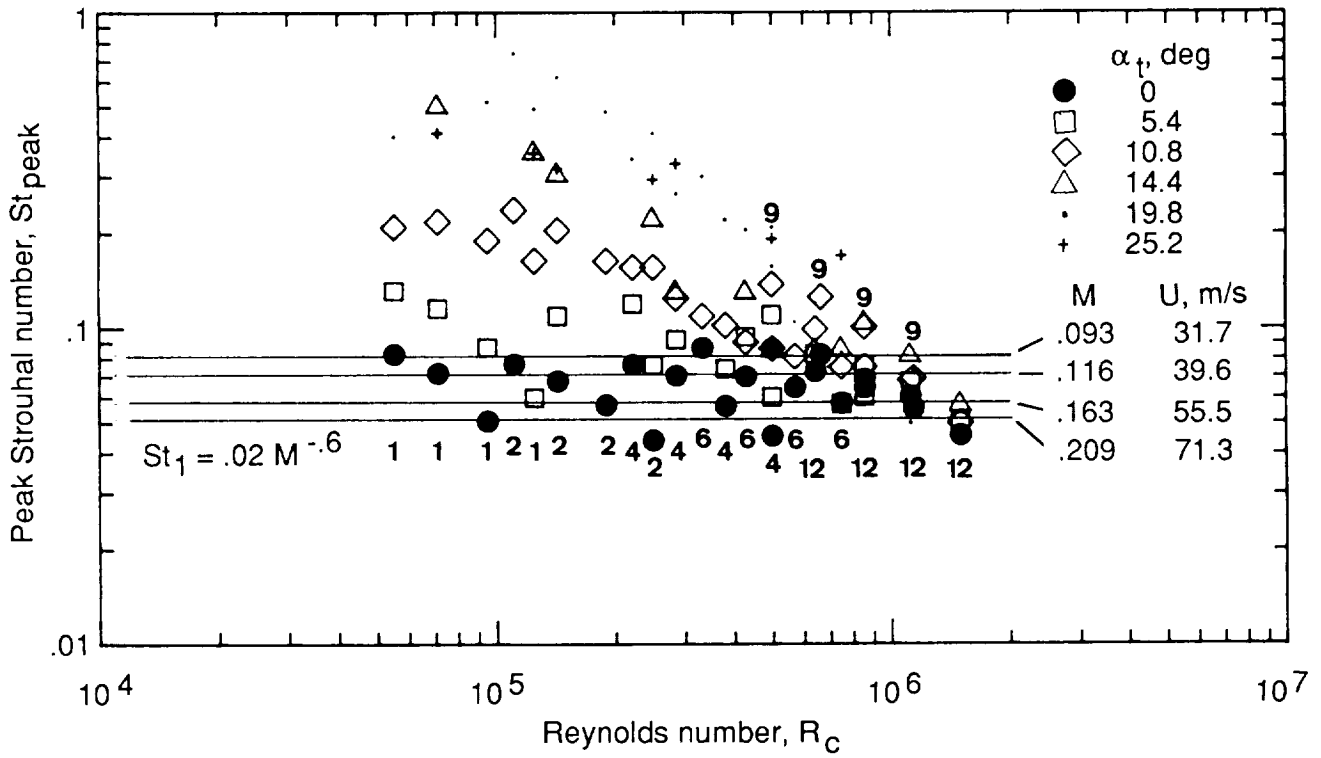


Figure 76. Peak Strouhal number for TBL-TE noise versus Reynolds number. Numbers aligned with data are chord sizes in inches (for brevity of notation).

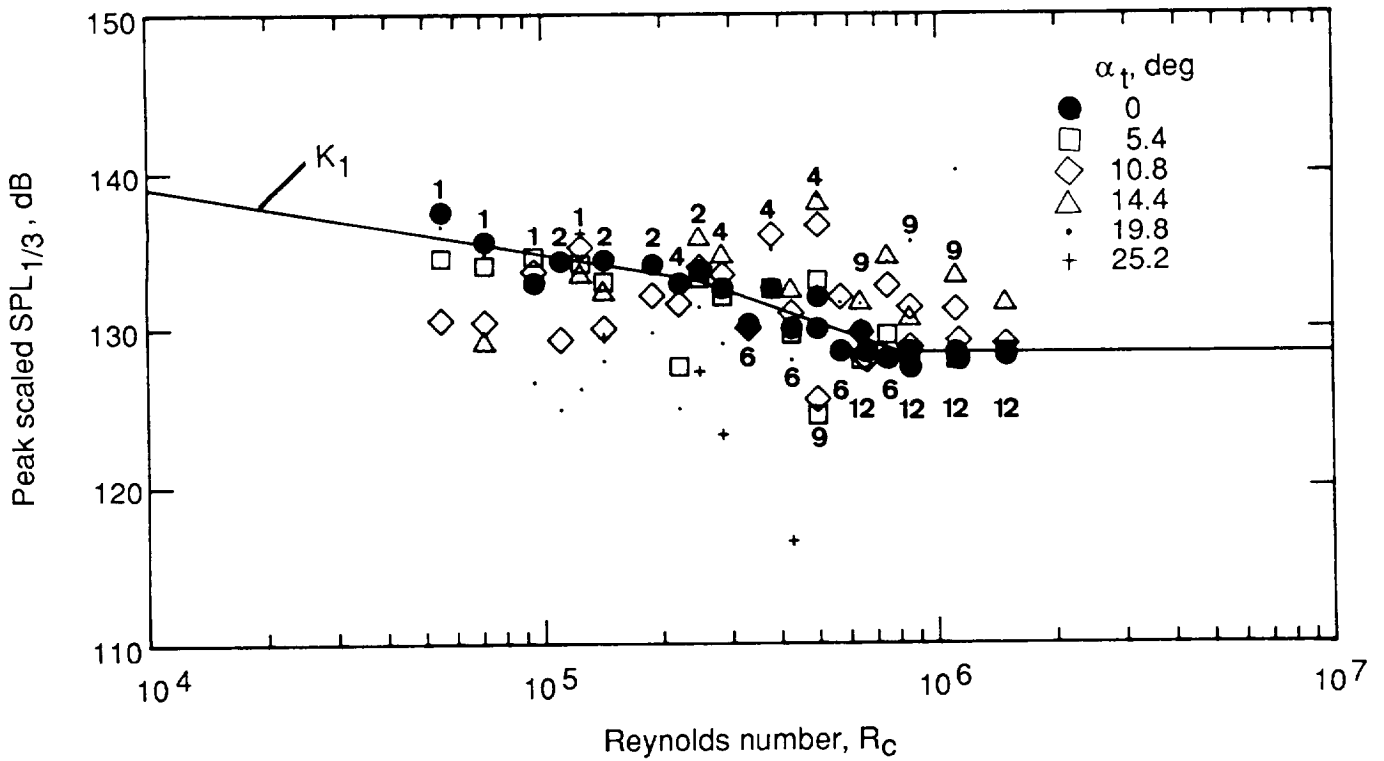


Figure 77. Peak scaled level for TBL-TE noise versus Reynolds number. Numbers aligned with data are chord sizes in inches.

$St_1 = 0.02M^{-0.6}$ for the presented values of Mach number and is taken to approximate the behavior of St_{peak} . For the scaled levels in figure 77, a continuous function, designated as K_1 , that is comprised of R_c -dependent segmented lines is drawn to approximate the zero-angle-of-attack data. Other choices for a function to approximate these data are possible but the one shown, which is chosen to be constant for high R_c , was found to be compatible with higher Reynolds number data obtained from other studies, as is shown subsequently. Note that the behavior of K_1 at very low R_c is at most academic because of the lack of importance of this TBL-TE noise mechanism in this range.

In figure 78(a), a shape function denoted by A is proposed as representative of the 1/3-octave spectral shape of the TBL-TE noise mechanism. (Fig. 78(b) presents a corresponding shape function for separated flow noise.) The spectrum A is a function of the ratio St/St_{peak} that is symmetric about $St/St_{\text{peak}} = 1.0$. The spectral width or broadness depends on R_c . Two extremes in A are shown corresponding to so-called maximum and minimum Reynolds numbers. Intermediate values of R_c require interpolation. As seen in figure 75, the larger chords have the broadest TBL-TE spectra. The spectrum A was matched to these and the other chord lengths. The specific details of A and the other functions are given in the calculation procedures section (5.1.2.).

One of the key results of reference 2 is that each side of an airfoil with well-developed boundary layers produces TBL-TE noise independently of the other side. This is not in conflict with our scaling approach for the symmetric airfoil at zero angle of attack. Consistency of this with equation (19) merely requires a level adjustment (-3 dB) of the scaling equations to account for the equal contributions of the two sides to the total spectrum. For the pressure and suction sides, $i = p$ or s ,

$$\begin{aligned} \text{Scaled SPL}_i &= \text{SPL}_i - 10 \log \left(M^5 \frac{\delta_i^* L}{r_e^2} \right) \\ &= A \left(\frac{St_i}{St_1} \right) + (K_1 - 3) \end{aligned} \quad (20)$$

where $St_i = (f\delta_i^*/U)$. The total TBL-TE noise for zero angle of attack then is

$$\text{SPL}_{\text{TBL-TE}} = 10 \log \left(10^{\text{SPL}_s/10} + 10^{\text{SPL}_p/10} \right) \quad (21)$$

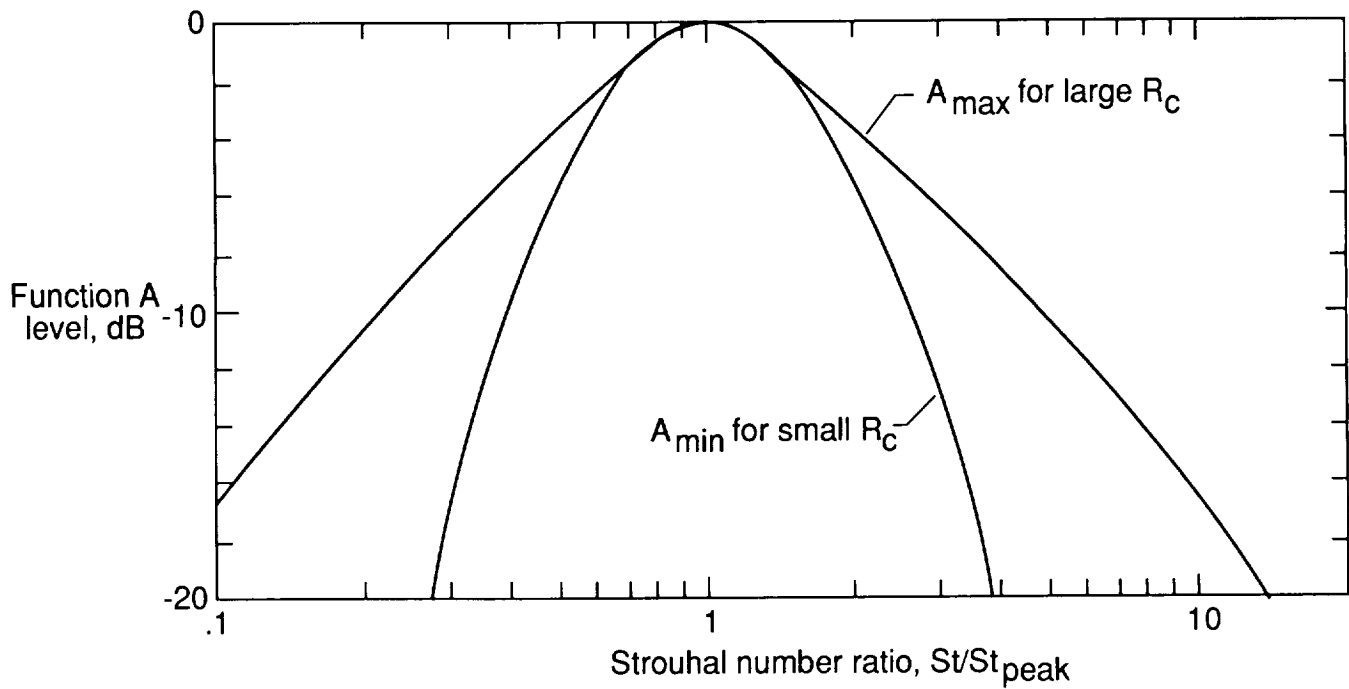
where a 1/3-octave presentation for spectra is understood.

Nonzero angle of attack. In figure 79, scaled noise spectra are presented for the same tripped BL airfoil

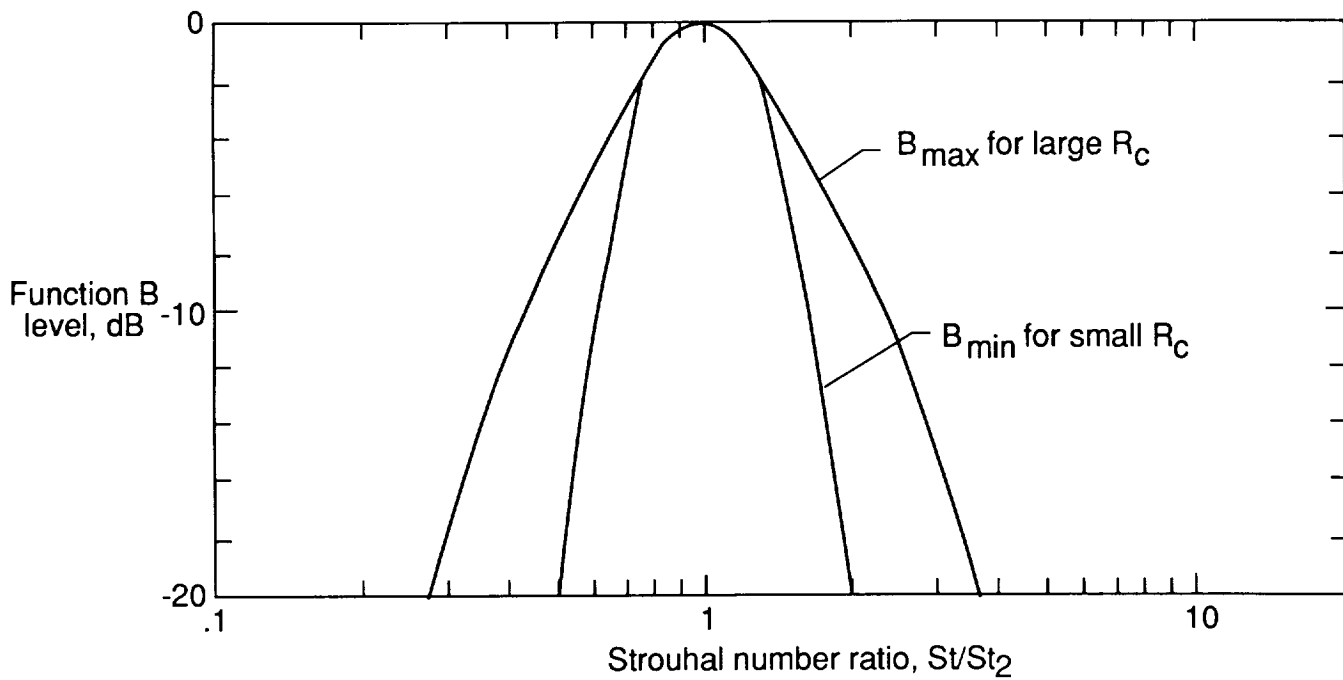
models as in figure 75, but here the angle of attack is varied while holding tunnel velocity constant at $U = 71.3$ m/s. The tunnel angles of attack α_t are given along with the effective angles α_* . The level normalization approach and Strouhal scaling are the same as in figure 75 except that here the displacement thickness of the suction side of the airfoil δ_s^* is used. For increasing α_* the peak Strouhal number and level increase and the spectra become sharper at the peaks. Beyond limiting values of α_* , roughly corresponding to stall, substantial changes occur to the scaled spectra.

If equations (20) and (21) were used to predict the spectra in figure 79 and the predictions scaled accordingly, one would find for increasing angle of attack that peak Strouhal number would remain constant, peak level would decrease, and the spectral shape would become broader at the peak. This is because the suction side contribution would remain dominant and that of the pressure side would shift to higher frequencies at reduced levels. These trends, of course, are virtually opposite to those observed. The approach that is now taken is to postulate at nonzero angles of attack an additional contribution to the spectrum that controls the spectral peak. To justify this, one could hypothesize that the spectrum is the total from attached TBL contributions, as formulated in equations (20) and (21), and a contribution from a separated portion of the TBL on the suction side. The modeling approach, however, is not without conflict at the low Reynolds numbers, as is discussed subsequently. Model details are developed below, after establishing the Strouhal and level scaling behavior for the angle cases.

In figure 79, for each spectrum, symbols indicate the approximate peak Strouhal locations. As in figure 75, the locations of the peaks were based on gross trends and shapes of the spectra rather than precise peaks. These values of St_{peak} are included in figure 76 for the various chords, speeds, and angles of attack, along with the zero angle values previously discussed. Again little direct R_c -dependence is noted for St_{peak} . The basic trends observed can be explained by velocity and angle dependence. The values of St_{peak} are plotted versus corrected angle of attack α_* in figure 80. For reference, the chord lengths (in units of inches for presentation convenience) are given. Through the data are drawn data-fit lines designated as St_2 , corresponding to two velocity values. At $\alpha_* = 0^\circ$, St_2 becomes the function St_1 of figure 76. In the hand-fitting procedure to determine St_2 , some preference was given to the higher speed cases. This preference is discussed subsequently with regard to Strouhal peak level scaling. As for the substantial

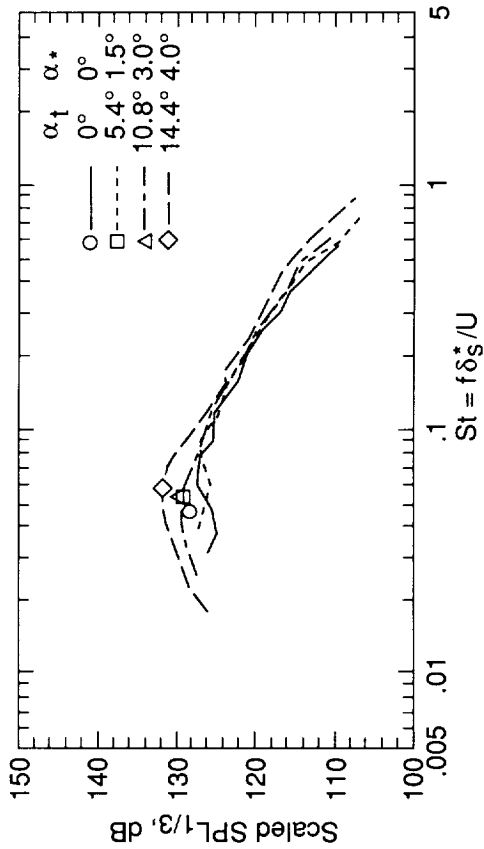


(a) Function A for TBL-TE noise, equations (35) to (40).

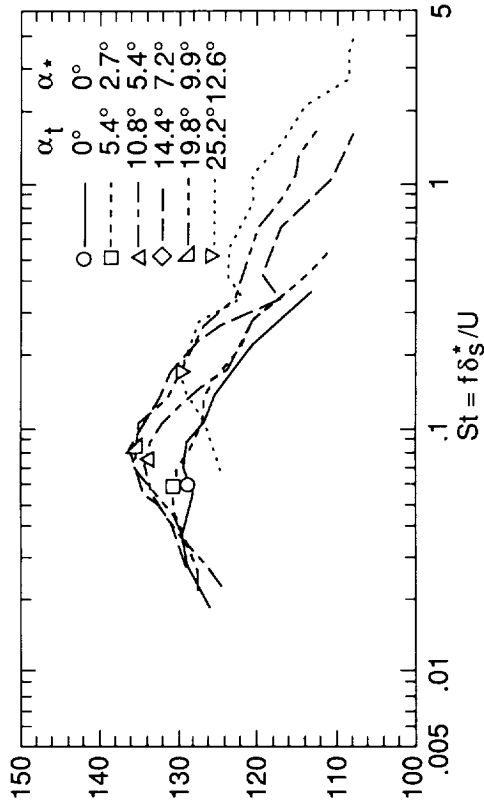


(b) Function B for separated flow noise, equations (41) to (46).

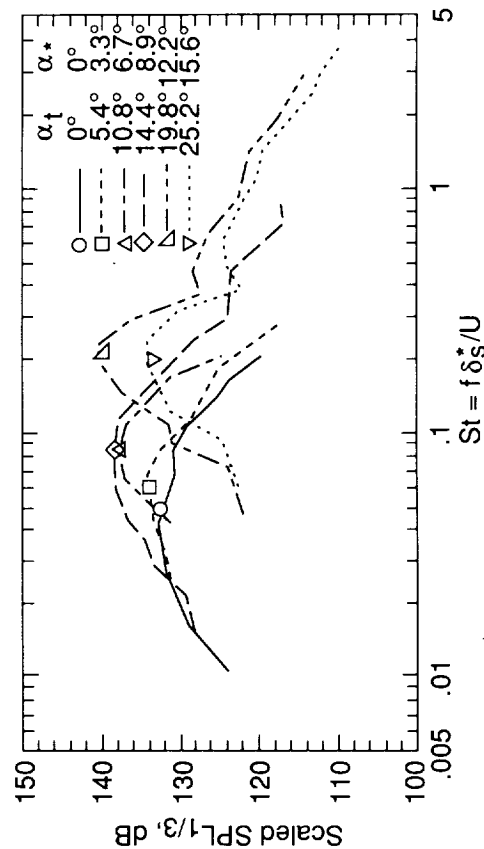
Figure 78. One-third-octave spectral shapes as functions of Strouhal and Reynolds numbers.



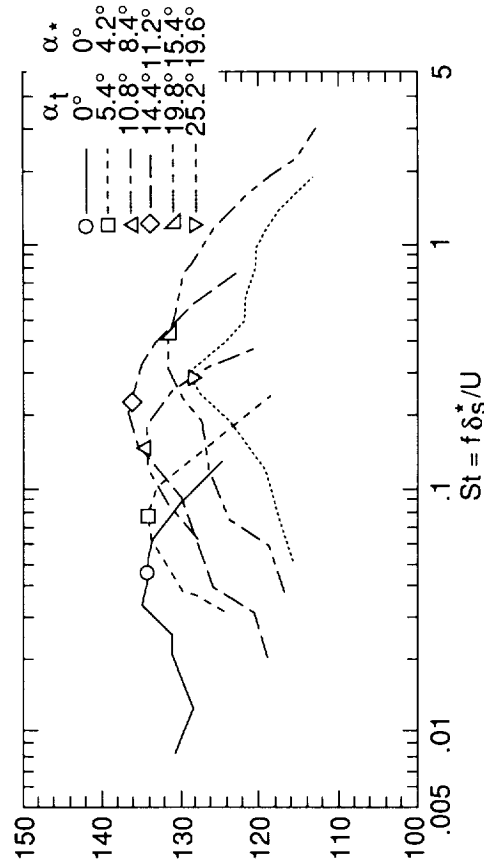
(a) 30.48-cm-chord airfoil.



(b) 15.24-cm-chord airfoil.



(c) 10.16-cm-chord airfoil.



(d) 5.08-cm-chord airfoil.

Figure 79. Scaled 1/3-octave spectra for tripped BL airfoils at various angles of attack. $U = 71.3$ m/s. Symbols indicate approximate spectral peak locations.

data scatter of figure 80, some comments are warranted. It was found that if one used the actual measured values of δ_s^* (where available) in the Strouhal scaling, one would have a similar degree of scatter to that shown in figure 80, where scaled values of δ_s^* (eq. (12)) were used. Also if untripped BL airfoil results were plotted, for those limited number of cases where the LBL-VS shedding source is not apparent in the spectra, the scatter and trend would be about the same as those shown in figure 80. Other deviations of the data from the St_2 lines occur at mid to high angles of attack, where the low-frequency parts of the spectra were limited by the experimental high-pass filtering and thus values of St_{peak} were inaccurately large. The behavior of St_2 seen in figure 80 at the higher angles of attack (where the horizontal lines are placed lower than the data) was chosen to approximately correct this bias.

The scaled levels corresponding to spectral peaks chosen in figure 79 are shown in figure 77 with the other cases. The previously indicated conflict within the data base for the proposed modeling approach, which hypothesizes contributions from two attached TBL's and an angle-dependent separation-related portion, is seen in figure 77. Peak levels for the two smallest chord lengths, except at the highest speeds, significantly decrease as the angle of attack increases from zero. This is incompatible with the modeling approach. A choice is made to ignore the conflicting low Reynolds number data in the model development. While admitting that the inclusion of the low Reynolds number behavior would conceptually be desirable for completeness of the modeling, the exclusion is believed justifiable because of the greater interest in higher Reynolds number conditions. The TBL-TE noise mechanism is not considered important for low Reynolds numbers. Even if this were not the case, it is not certain that the present test flow conditions with heavy leading-edge tripping for airfoils at nonzero angles of attack properly represent the mechanism, especially for higher angles where relaminarization of the pressure-side boundary layer is possible. Regardless, the results of the scaling are compared subsequently with the spectra of all the data to allow a direct assessment of the effect of modeling choices.

The scaled levels of figure 77 for chord lengths of 10.16, 15.24, 22.86, and 30.48 cm are plotted in figure 81 versus α_* . If the portion of these levels that cannot be accounted for by the modeling of equations (20) and (21) can be extracted, this portion would be designated as the separated flow noise contribution. Calculations were performed by taking into account that the Strouhal dependence of A in equation (20) would follow St_1 of figure 76

rather than St_2 of figure 80, which applies to that portion extracted. The extracted levels are given in figure 82. These extracted levels are normalized by subtracting the zero-angle-of-attack function of figure 77 (K_1) for the particular chord lengths and speeds. Although substantial scatter is present, a basic trend of increasing importance for increasing angle and speed is seen. Drawn through the data is a function designated as $K_2 - K_1$ which represents a partially observed, partially postulated dependence on velocity and angle of attack. The assigned spectral shape for this additive source is function B , which is given in figure 78(b) and is defined in a manner similar to function A of figure 78(a) to have a width which is dependent on chord Reynolds number.

The resulting scaling model for the angle-dependent noise SPL_α is

$$\begin{aligned} \text{Scaled } SPL_\alpha &= SPL_\alpha - 10 \log \left(M^5 \frac{\delta_s^* L}{r_e^2} \right) \\ &= B \left(\frac{St_s}{St_2} \right) + K_2 \end{aligned} \quad (22)$$

where this represents the separated-boundary-layer noise contribution to the total noise. The total TBL-TE and separation noise is then

$$\begin{aligned} SPL_{TOT} &= 10 \log \left(10^{SPL_\alpha/10} + 10^{SPL_s/10} \right. \\ &\quad \left. + 10^{SPL_p/10} \right) \end{aligned} \quad (23)$$

During development of the scaling procedures, equations (20), (22), and (23) were compared with spectra for all tripped BL airfoils and with spectra for the untripped BL airfoils for which TBL-TE noise appeared to significantly contribute. Analyses of comparisons resulted in optimization of curves A and B, as well as development of the specific calculation procedures. The analysis found that better results are obtained when the Strouhal dependency of the suction-side spectrum SPL_s is $(St_1 + St_2)/2$ rather than St_1 . It was found that for better SPL agreement, one should make an adjustment in pressure-side level SPL_p (defined as ΔK_1 in the following section) as a function of angle of attack and Reynolds number based on the displacement thickness δ_p^* . This adjustment diminishes the pressure-side contribution for increasing angle and decreasing velocity. Also it was found that the drastic spectral shape changes that occur at sufficiently high angles of attack, near stall, are roughly simulated by a calculation procedure change. At the value of α_* corresponding to the peak of the appropriate K_2 curve, the spectral

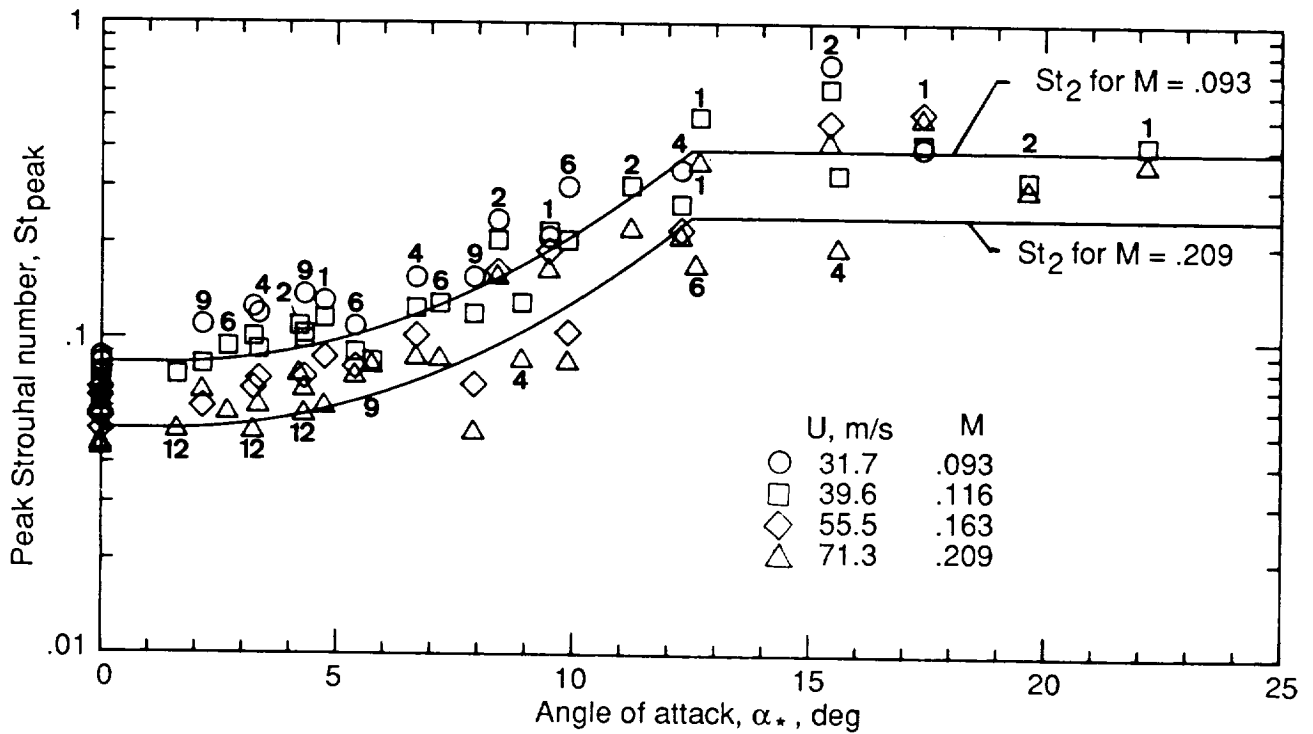


Figure 80. Peak Strouhal number for TBL-TE noise versus angle of attack. Data from figure 76. Numbers aligned with data are chord sizes in inches.

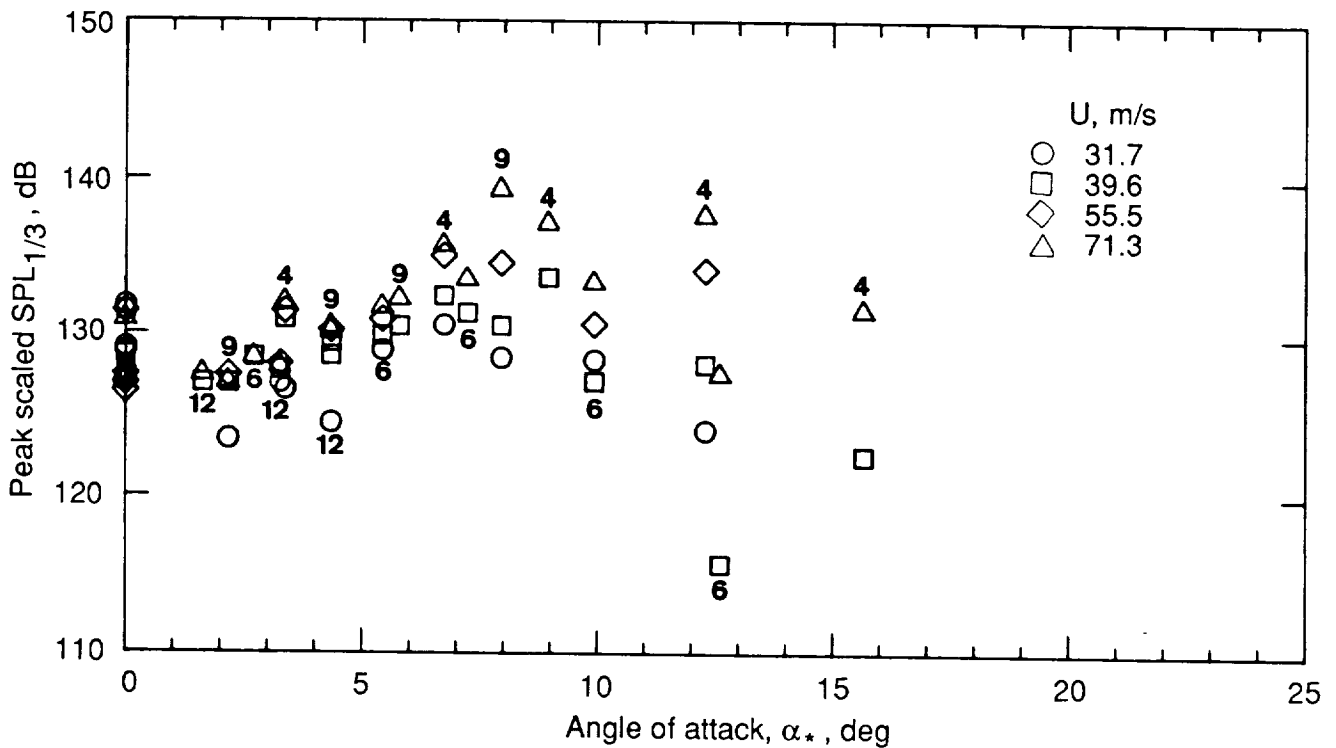


Figure 81. Peak scaled level for TBL-TE versus angle of attack. Data from figure 77. Numbers aligned with data are chord sizes in inches.

contributions SPL_s and SPL_p in equation (23) are eliminated and the B curve of equation (22) is replaced by an A curve corresponding to a value of R_c which is three times the actual value.

The calculation procedures are specified in the next section followed by comparison with the spectral data base.

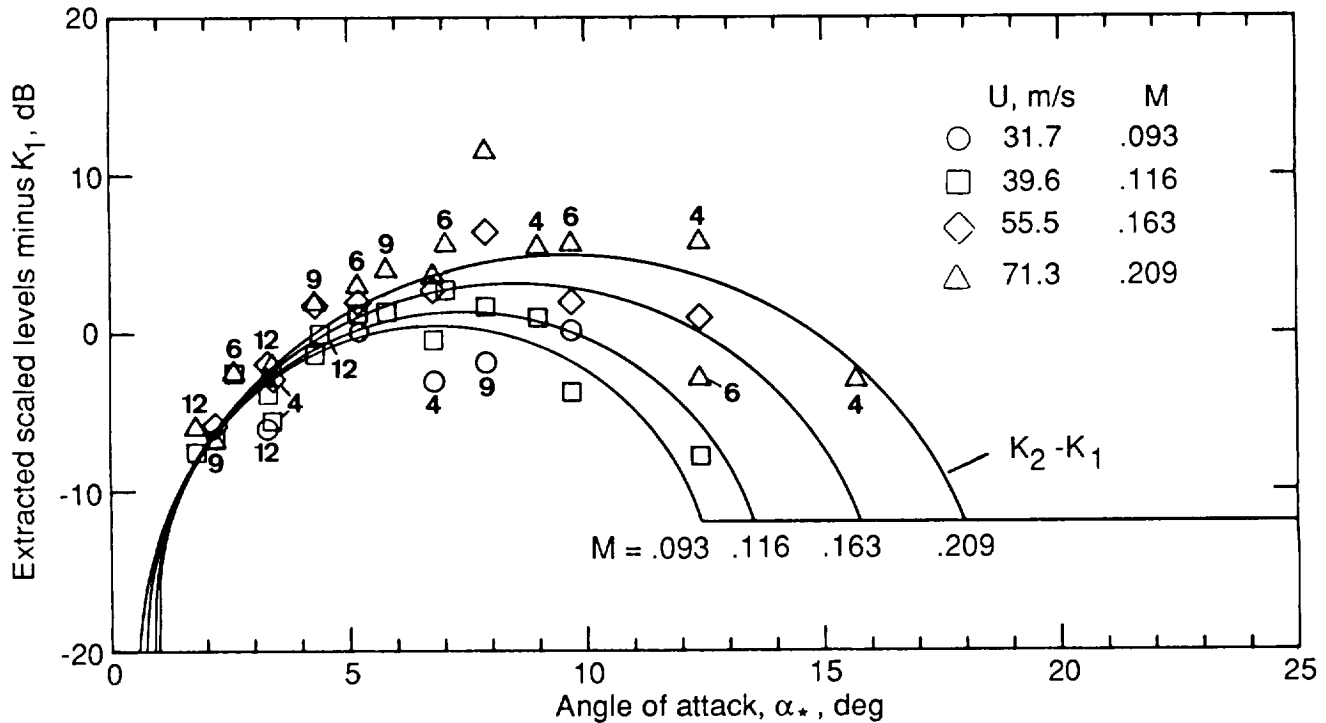


Figure 82. Angle-dependent scaled noise levels as referenced to zero angle of attack, TBL-TE noise model. Numbers aligned with data are chord sizes in inches.

5.1.2. Calculation Procedures

The total TBL-TE and separation noise spectrum in a 1/3-octave presentation is predicted by

$$SPL_{TOT} = 10 \log \left(10^{SPL_{\alpha}/10} + 10^{SPL_s/10} + 10^{SPL_p/10} \right) \quad (24)$$

where

$$SPL_p = 10 \log \left(\frac{\delta_p^* M^5 L \bar{D}_h}{r_e^2} \right) + A \left(\frac{St_p}{St_1} \right) + (K_1 - 3) + \Delta K_1 \quad (25)$$

$$SPL_s = 10 \log \left(\frac{\delta_s^* M^5 L \bar{D}_h}{r_e^2} \right) + A \left(\frac{St_s}{St_1} \right) + (K_1 - 3) \quad (26)$$

and

$$SPL_{\alpha} = 10 \log \left(\frac{\delta_s^* M^5 L \bar{D}_h}{r_e^2} \right) + B \left(\frac{St_s}{St_2} \right) + K_2 \quad (27)$$

for angles of attack up to $(\alpha_*)_0$, an angle to be defined later in this section. At angles above $(\alpha_*)_0$,

$$SPL_p = -\infty \quad (28)$$

$$SPL_s = -\infty \quad (29)$$

and

$$\text{SPL}_\alpha = 10 \log \left(\frac{\delta_s^* M^5 L \bar{D}_\ell}{r_e^2} \right) + A' \left(\frac{\text{St}_s}{\text{St}_2} \right) + K_2 \quad (30)$$

where A' is the curve A but for a value of R_c which is three times the actual value. The directivity functions \bar{D}_h and \bar{D}_ℓ are given in appendix B by equations (B1) and (B2), respectively.

The Strouhal definitions are (see figs. 76 and 80)

$$\text{St}_p = \frac{f \delta_p^*}{U} \quad \text{St}_s = \frac{f \delta_s^*}{U} \quad (31)$$

$$\text{St}_1 = 0.02 M^{-0.6} \quad (32)$$

$$\bar{\text{St}}_1 = \frac{\text{St}_1 + \text{St}_2}{2} \quad (33)$$

and

$$\text{St}_2 = \text{St}_1 \times \begin{cases} 1 & (\alpha_* < 1.33^\circ) \\ 10^{0.0054(\alpha_* - 1.33)^2} & (1.33^\circ \leq \alpha_* \leq 12.5^\circ) \\ 4.72 & (12.5^\circ < \alpha_*) \end{cases} \quad (34)$$

For the spectral shape function definitions, we first consider the function A of figure 78(a). As discussed, the function A for a particular Reynolds number R_c is obtained from an interpolation of the curves A_{\max} and A_{\min} , corresponding to chosen values, $(R_c)_{\max}$ and $(R_c)_{\min}$. The two curves are defined as

$$A_{\min}(a) = \begin{cases} \sqrt{67.552 - 886.788a^2} - 8.219 & (a < 0.204) \\ -32.665a + 3.981 & (0.204 \leq a \leq 0.244) \\ -142.795a^3 + 103.656a^2 - 57.757a + 6.006 & (0.244 < a) \end{cases} \quad (35)$$

and

$$A_{\max}(a) = \begin{cases} \sqrt{67.552 - 886.788a^2} - 8.219 & (a < 0.13) \\ -15.901a + 1.098 & (0.13 \leq a \leq 0.321) \\ -4.669a^3 + 3.491a^2 - 16.699a + 1.149 & (0.321 < a) \end{cases} \quad (36)$$

where a is the absolute value of the logarithm of the ratio of Strouhal number, $\text{St} = \text{St}_p$ or St_s , to the peak Strouhal number, $\text{St}_{\text{peak}} = \text{St}_1, \bar{\text{St}}_1$, or St_2 :

$$a = |\log(\text{St}/\text{St}_{\text{peak}})| \quad (37)$$

The absolute value is used because the spectral shape is modeled to be symmetric about $a = 0$.

The interpolative procedure includes defining a value, $a_0(R_c)$, at which the spectrum has a value of -20 dB. This -20 dB corresponds to a horizontal axis intercept in figure 78(a) for an interpolated curve. The function

$a_0(R_c)$ is given by

$$a_0(R_c) = \begin{cases} 0.57 & (R_c < 9.52 \times 10^4) \\ (-9.57 \times 10^{-13})(R_c - 8.57 \times 10^5)^2 + 1.13 & (9.52 \times 10^4 \leq R_c \leq 8.57 \times 10^5) \\ 1.13 & (8.57 \times 10^5 < R_c) \end{cases} \quad (38)$$

An interpolation factor $A_R(a_0)$ is determined from

$$A_R(a_0) = \frac{-20 - A_{\min}(a_0)}{A_{\max}(a_0) - A_{\min}(a_0)} \quad (39)$$

where $A_{\min}(a_0)$ and $A_{\max}(a_0)$ are the A_{\max} and A_{\min} spectra evaluated at a_0 . The spectrum shape A can now be evaluated for any frequency by computing the Strouhal number St and the corresponding a and using the interpolation factor. The result for use in equations (25), (26), and (30) is

$$A(a) = A_{\min}(a) + A_R(a_0) [A_{\max}(a) - A_{\min}(a)] \quad (40)$$

The function B in equation (27) and shown plotted in figure 78(b) is calculated in a manner similar to function A above. The two curves B_{\max} and B_{\min} , through which B is obtained from interpolation, are

$$B_{\min}(b) = \begin{cases} \sqrt{16.888 - 886.788b^2} - 4.109 & (b < 0.13) \\ -83.607b + 8.138 & (0.13 \leq b \leq 0.145) \\ -817.810b^3 + 355.210b^2 - 135.024b + 10.619 & (0.145 < b) \end{cases} \quad (41)$$

and

$$B_{\max}(b) = \begin{cases} \sqrt{16.888 - 886.788b^2} - 4.109 & (b < 0.10) \\ -31.330b + 1.854 & (0.10 \leq b \leq 0.187) \\ -80.541b^3 + 44.174b^2 - 39.381b + 2.344 & (0.187 < b) \end{cases} \quad (42)$$

where

$$b = |\log(St_s/St_2)| \quad (43)$$

The spectral shape B for intermediate values of R_c have horizontal axis intercepts at -20 dB in figure 78(b) for values of b of

$$b_0(R_c) = \begin{cases} 0.30 & (R_c < 9.52 \times 10^4) \\ (-4.48 \times 10^{-13})(R_c - 8.57 \times 10^5)^2 + 0.56 & (9.52 \times 10^4 \leq R_c \leq 8.57 \times 10^5) \\ 0.56 & (8.57 \times 10^5 < R_c) \end{cases} \quad (44)$$

The interpolation factor $B_R(b_0)$ is defined as

$$B_R(b_0) = \frac{-20 - B_{\min}(b_0)}{B_{\max}(b_0) - B_{\min}(b_0)} \quad (45)$$

and thus the result for use in equation (27) is

$$B(b) = B_{\min}(b) + B_R(b_0) [B_{\max}(b) - B_{\min}(b)] \quad (46)$$

The amplitude function K_1 in equations (25) and (26) is plotted in figure 77 and is given by

$$K_1 = \begin{cases} -4.31 \log(R_c) + 156.3 & (R_c < 2.47 \times 10^5) \\ -9.0 \log(R_c) + 181.6 & (2.47 \times 10^5 \leq R_c \leq 8.0 \times 10^5) \\ 128.5 & (8.0 \times 10^5 < R_c) \end{cases} \quad (47)$$

The level adjustment previously mentioned for the pressure-side contribution for nonzero angles of attack appears as ΔK_1 in equation (25). This is given by

$$\Delta K_1 = \begin{cases} \alpha_* [1.43 \log(R_{\delta_p^*}) - 5.29] & (R_{\delta_p^*} \leq 5000) \\ 0 & (5000 < R_{\delta_p^*}) \end{cases} \quad (48)$$

where $R_{\delta_p^*}$ is the Reynolds number based on pressure-side displacement thickness.

The amplitude function K_2 of equations (27) and (30) is plotted for some values of M in figure 82 and is given as

$$K_2 = K_1 + \begin{cases} -1000 & (\alpha_* < \gamma_0 - \gamma) \\ \sqrt{\beta^2 - (\beta/\gamma)^2(\alpha_* - \gamma_0)^2} + \beta_0 & (\gamma_0 - \gamma \leq \alpha_* \leq \gamma_0 + \gamma) \\ -12 & (\gamma_0 + \gamma < \alpha_*) \end{cases} \quad (49)$$

where

$$\left. \begin{aligned} \gamma &= 27.094M + 3.31 & \gamma_0 &= 23.43M + 4.651 \\ \beta &= 72.65M + 10.74 & \beta_0 &= -34.19M - 13.82 \end{aligned} \right\} \quad (50)$$

The angle definitions above are in units of degrees and are taken as positive in sign. The K_2 definition above is valid for all values of α_* , even when the calculation of the total noise in equation (24) switches from the use of equations (25), (26), and (27) for assumed attached TBL flow to equations (28), (29), and (30) for a supposedly stalled flow condition. The angle where the switch occurs, specified previously as $(\alpha_*)_0$, is taken to be equal to the peak of the K_2 function defined by γ_0 in equation (50) or whenever α_* exceeds 12.5° , whichever is first.

5.1.3. Comparison With Data

The scaling predictions of TBL-TE and separation noise are compared with the noise data in figures 11 to 43 for the tripped BL airfoils. The calculations used the appropriate values of δ^* from section 3 and the directivity functions from appendix B (where $r_e = 1.22$ m, $\Theta_e = 90^\circ$, and $\Phi_e = 90^\circ$). The total self-noise is given as well as the individual noise components of TBL-TE noise from the suction and pressure sides and separation noise. The predictions follow the shapes and levels of the data, especially for the larger airfoils and the lower angles of attack where the scaling accuracy was most emphasized. Predictions of TBL-TE and separation noise are also shown for the untripped BL airfoils in figures 44

to 74. For the many untripped cases where these sources are predicted to be dominant, the agreement is generally good. Even where the LBL-VS noise dominates, the TBL-TE and separation contributions help with the overall spectral agreement.

5.2. Laminar-Boundary-Layer-Vortex-Shedding Noise

As previously described in section 1, laminar-boundary-layer instabilities couple with acoustic feedback to produce quasi-tonal noise. In contrast to TBL-TE noise, there are no LBL-VS noise scaling methods established in the literature because of the erratic behavior of the multiple tones in the narrow-band spectra and the general complexity of the

mechanism. Two key results from the literature which provide initial scaling guidance are (1) that the gross trend of the frequency dependence was found to scale on a Strouhal basis, with the relevant length scale being the laminar-boundary-layer thickness at the airfoil trailing edge (ref. 16), and (2) that on the basis of the limited data from the data base of the present paper as reported in reference (6), overall levels tended to coalesce to a unique function of R_c when normalized in the fashion of TBL-TE noise.

The scaling approach taken herein is similar to that taken for TBL-TE noise in the last section in that a universal spectral shape and Strouhal dependence is modeled in terms of boundary-layer parameters, Mach number, angle of attack, and Reynolds number. The use of 1/3-octave spectra, rather than narrow band, permits such an approach because the broad spectral bands overlap the tonal frequency spacing to give smoother and generally single-peaked spectra.

5.2.1. Scaled Data

Scaled 1/3-octave sound pressure level spectra for four airfoil sizes, each at four tunnel speeds, are presented in figure 83 from figures 44, 53, 59, and 65. The angle of attack for all is zero and the boundary layers are untripped. The normalization employs

$$\text{Scaled SPL}_{1/3} = \text{SPL}_{1/3} - 10 \log \left(M^5 \frac{\delta_p L}{r_e^2} \right) \quad (51)$$

for level scaling and

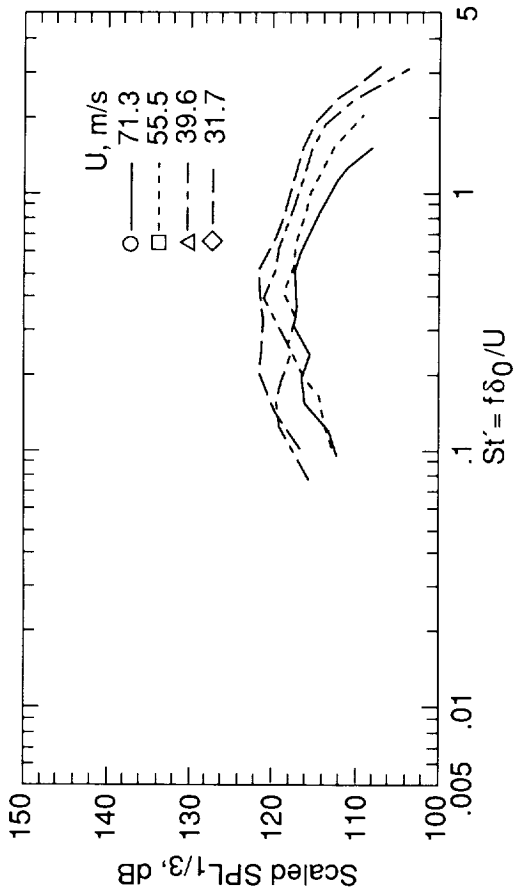
$$\text{St}' = \frac{f \delta_p}{U} \quad (52)$$

for Strouhal frequency scaling. For the symmetric airfoils at zero angle of attack, $\delta_p = \delta_s = \delta_0$. The scaling approach differs from the TBL-TE noise scaling because of the use of δ_p , the boundary-layer thickness on the pressure side of the airfoil, rather than δ_s^* , the boundary-layer displacement thickness on the suction side. The use of δ_p as the pertinent length scale follows from reference 16 and was found to give seemingly better results in initial scaling of the present data base than δ_p^* and by far better than c , δ_s , or δ_s^* for angles of attack other than zero.

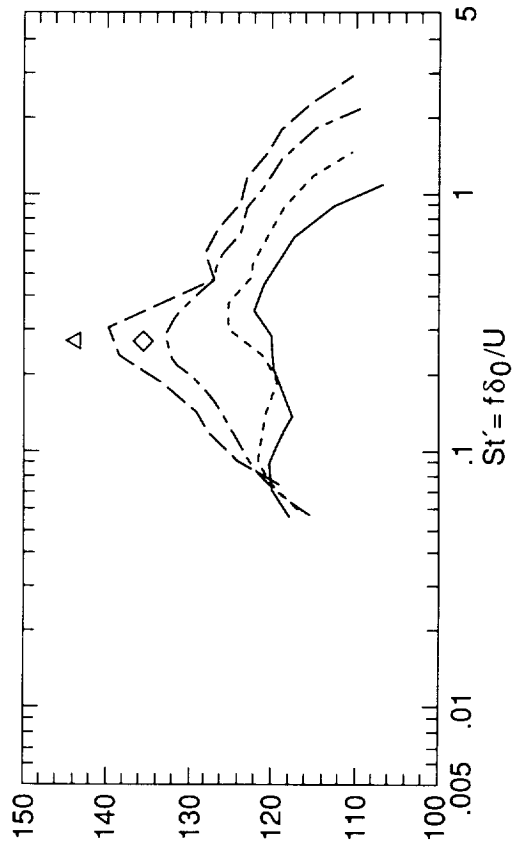
In figure 83(a) for the large 30.48-cm-chord airfoil, the spectra appear to be of smooth broad hump shapes. There is no apparent contribution to the spectra from LBL-VS noise which is peaked in character. The boundary layers are fully turbulent in

the vicinity of the trailing edge at all four tunnel speeds (ref. 21), so no laminar vortex shedding is established. The noise produced is assumed to be all TBL-TE noise. In figure 83(b) for the 15.24-cm-chord airfoil, the broad spectral shapes are changed by the addition of a peak when the flow velocity is diminished. The peak levels increase with decreasing velocity. Although the boundary layer is turbulent at the trailing edge at all velocities shown, laminar flow exists over larger portions of the airfoil at the lower velocities. As mentioned for the LBL-VS noise mechanism, any spectral peaks containing a number of tonal contributions should scale with Strouhal numbers based on boundary-layer thickness. This is the case in figure 83(b) with $\text{St}' \approx 0.27$. For the shorter 10.16-cm-chord airfoil, in figure 83(c), the LBL-VS noise peaks become even more dominant for decreasing velocity. Note also the changing Strouhal dependence, not noted in previous studies. The shorter 5.08-cm-chord airfoil, in figure 83(d), has even more pronounced level and Strouhal dependence with velocity variations.

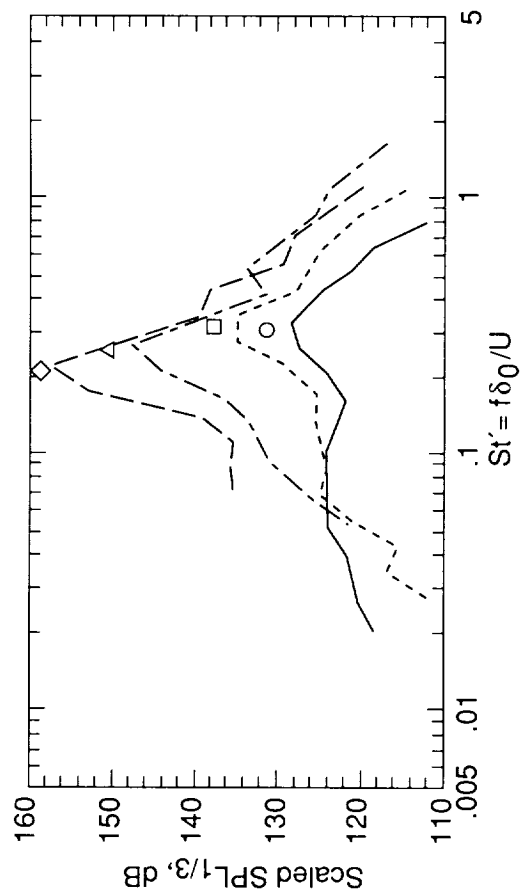
Whereas figure 83 shows the dependence of LBL-VS noise on velocity for the various airfoil sizes at zero angle of attack, figure 84 shows the effect of angle of attack α_* of the airfoils at a velocity of 71.3 m/s. The spectra for the 30.48-cm-chord airfoil, shown in figure 84(a), change from being dominated by TBL-TE noise, for $\alpha_* = 0^\circ$, to being dominated by LBL-VS noise, for $\alpha_* = 4.0^\circ$. So even with a large Reynolds number ($R_c = 1.52 \times 10^6$), LBL-VS noise occurs. With increasing α_* , the boundary layer on the pressure side becomes more laminar over a sufficiently large portion of the chord to result in increased shedding and corresponding noise. For the 15.24-cm-chord airfoil ($R_c = 7.58 \times 10^5$), shown in figure 84(b), the LBL-VS noise increases with α_* until a certain value is reached where it diminishes. At $\alpha_* = 7.2^\circ$, no apparent shedding noise is shown. At $\alpha_* = 9.9^\circ$, the noise changes appreciably to that for stalled flow as discussed in the last section. The use of δ_p as the characteristic length scale apparently results in a proper Strouhal scaling for the shedding noise peaks; but, as expected, the spectra for $\alpha_* = 0^\circ$, 7.2° , and 9.9° , which are dominated by TBL-TE and separated flow noise, diverge in this normalized format. A similar angle-dependent behavior where spectra do not coalesce is seen for the 10.16-cm-chord airfoil, in figure 84(c), where LBL-VS noise is apparent at $\alpha_* = 0^\circ$ and 3.3° but not at the higher angles. For the 5.08-cm-chord model, figure 84(d) shows large-amplitude LBL-VS noise at $\alpha_* = 0^\circ$ and 4.2° .



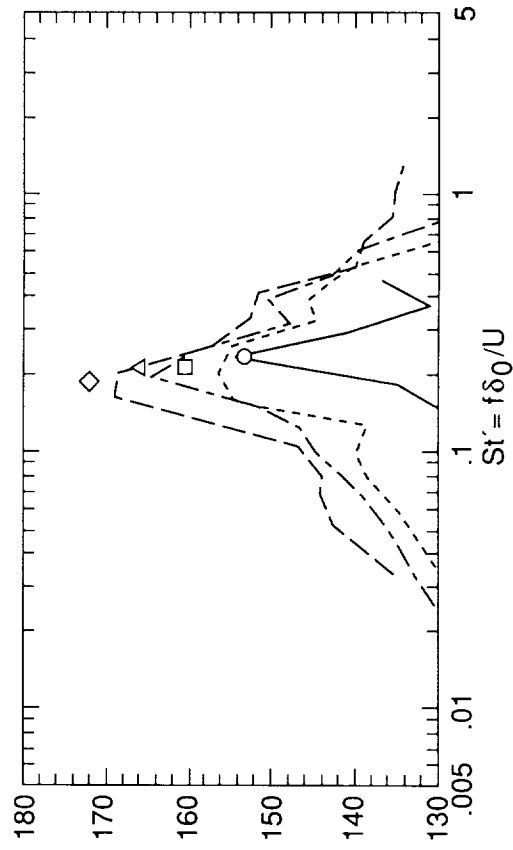
(a) 30.48-cm-chord airfoil.



(b) 15.24-cm-chord airfoil.

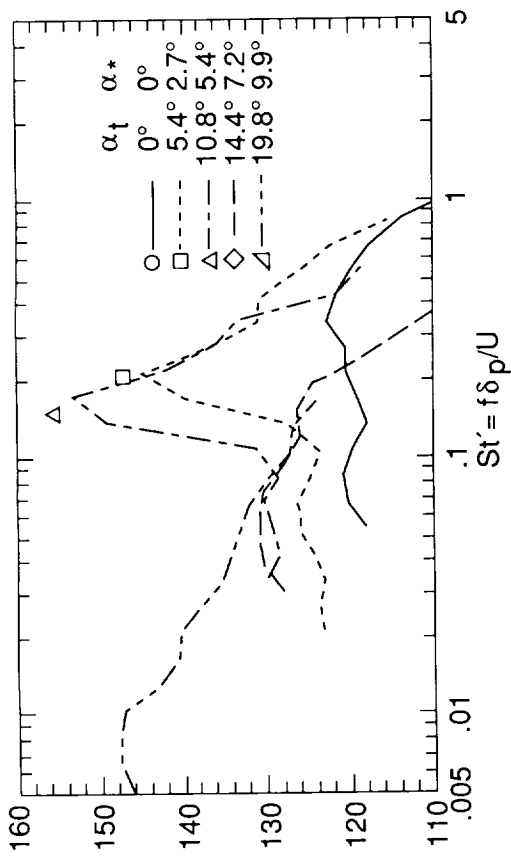


(c) 10.16-cm-chord airfoil.

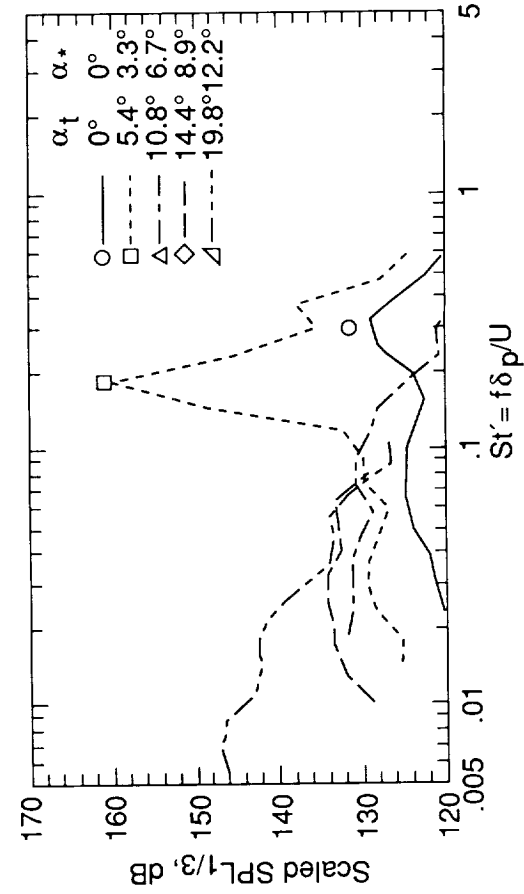


(d) 5.08-cm-chord airfoil.

Figure 83. Scaled 1/3-octave spectra for untripped BL airfoils at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$). Symbols indicate approximate LBL VS reference peak location.

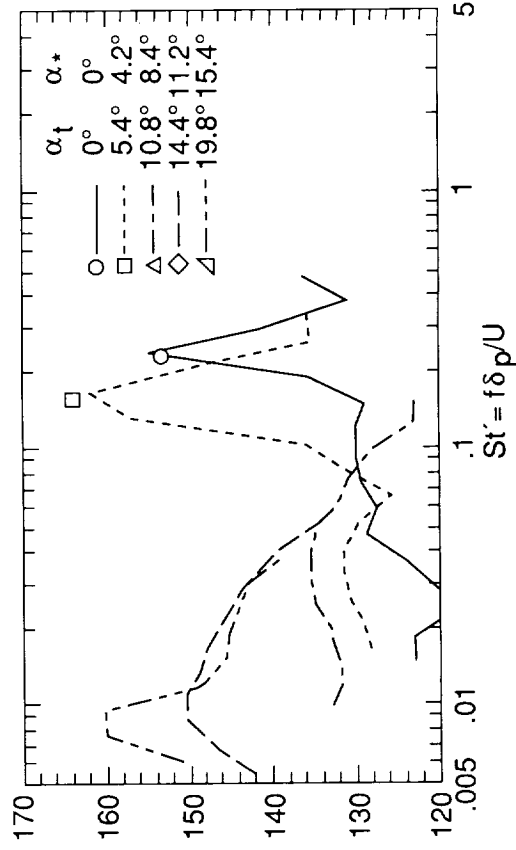
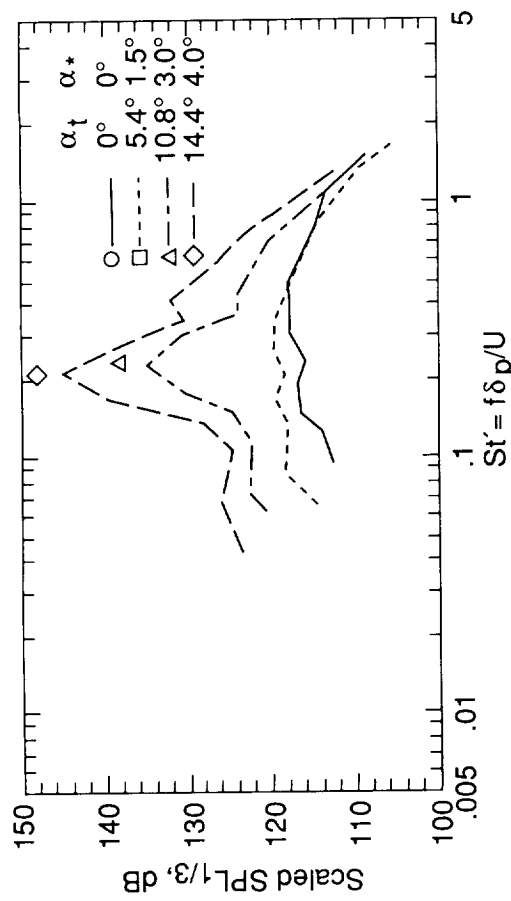


(a) 30.48-cm-chord airfoil.



(b) 15.24-cm-chord airfoil.

(c) 10.16-cm-chord airfoil.



(d) 5.08-cm-chord airfoil.

Figure 84. Scaled $1/3$ -octave spectra for untripped BL airfoils at various angles of attack. $U = 71.3$ m/s. Symbols indicate approximate LBL-VS reference peak locations.

The LBL-VS noise portions of the spectra (figs. 83 and 84) are rather invariant with respect to spectral shape. Based on this observation, a function G_1 (shown in fig. 85) was chosen as a shape to represent the LBL-VS contribution to the self-noise 1/3-octave spectra for all cases. The level of G_1 at $St' = St'_{\text{peak}}$ has a value of -3.5 dB. The reference level of 0 dB is the integrated total of G_1 . To permit an orderly study of the Reynolds number and angle dependences of the spectral data, the shape G_1 was matched to the individual spectra to obtain reference overall peak levels and Strouhal numbers. Emphasis was placed on matching the global spectral shape of G_1 to the data rather than matching 1/3-octave band peak or overall levels. Reference peak locations are indicated by the symbols in figures 83 and 84.

In figure 86, the chosen values of St'_{peak} are plotted versus the Reynolds number R_c for the 42 cases where LBL-VS noise is prominent. The values are also given in table 2 (at the back of this report) along with the effective angles of attack α_* corresponding to α_t . For $\alpha_* = 0$, St'_{peak} is approximately constant at low R_c and increases with R_c in the mid-range of R_c shown. The values of St'_{peak} are lower for nonzero angles of attack. A function St'_1 is drawn to approximate the data of zero angle of attack. A constant value for St'_1 is chosen for high R_c , where no zero-angle-of-attack data are present, because the value permits a simple angle dependence definition for St'_{peak} . In figure 87, St'_{peak} is normalized by St'_1 and plotted versus α_* . For each of the six airfoils, the line described by $10^{-0.04\alpha_*}$ approximates the angle dependence.

5.2.2. Calculation Procedures

The LBL-VS noise spectrum in a 1/3-octave presentation is predicted by

$$\text{SPL}_{\text{LBL-VS}} = 10 \log \left(\frac{\delta_p M^5 L \bar{D}_h}{r_e^2} \right) + G_1 \left(\frac{St'}{St'_{\text{peak}}} \right) + G_2 \left[\frac{R_c}{(R_c)_0} \right] + G_3(\alpha_*) \quad (53)$$

The Strouhal definitions are (see figs. 86 and 87)

$$St' = \frac{f \delta_p}{U} \quad (54)$$

$$St'_1 = \begin{cases} 0.18 & (R_c \leq 1.3 \times 10^5) \\ 0.001756 R_c^{0.3931} & (1.3 \times 10^5 < R_c \leq 4.0 \times 10^5) \\ 0.28 & (4.0 \times 10^5 < R_c) \end{cases} \quad (55)$$

The reference peak scaled levels which correspond to St'_{peak} in figure 86 are plotted versus R_c in figure 88. To show general trends more clearly, the symbols are replaced by the value of α_* , rounded off to the nearest whole degree (see table 2 for more exact values). In this format it is seen that for each α_* the scaled levels tend to increase, peak, and decrease as R_c increases. For the larger angles of attack, the peak levels are lower and the corresponding values of R_c are larger. Superimposed on the data are curves of identical shape, called here "level shape curves," which are positioned in a monotonically decreasing fashion to approximately correspond to the data trends with angle variation. The angles indicated for each curve position should not necessarily match the angle values listed for the data because the data values are rounded off in the figure, as mentioned. The intent is to use the curves, with their functional relationship to α_* and R_c shown in figure 88, to represent the amplitude definition of LBL-VS noise. In the following calculation procedures section, a function G_2 specifies the curve shape, G_3 is the angle dependence for the level of the G_2 curve, and a reference $(R_c)_0$ value is defined as a function of angle to specify the Reynolds number dependence. The success of the functions in normalizing the data is shown in figure 89 where peak scaled 1/3-octave level minus G_3 is compared with the function G_2 . In this format the individual angle numbers should ideally match the G_2 curve. Although the agreement shown is certainly not complete, it is regarded here as acceptable. Note that much better curve fits to the data would be possible if a requirement for monotonic functional behavior had not been imposed on G_3 and $(R_c)_0$.

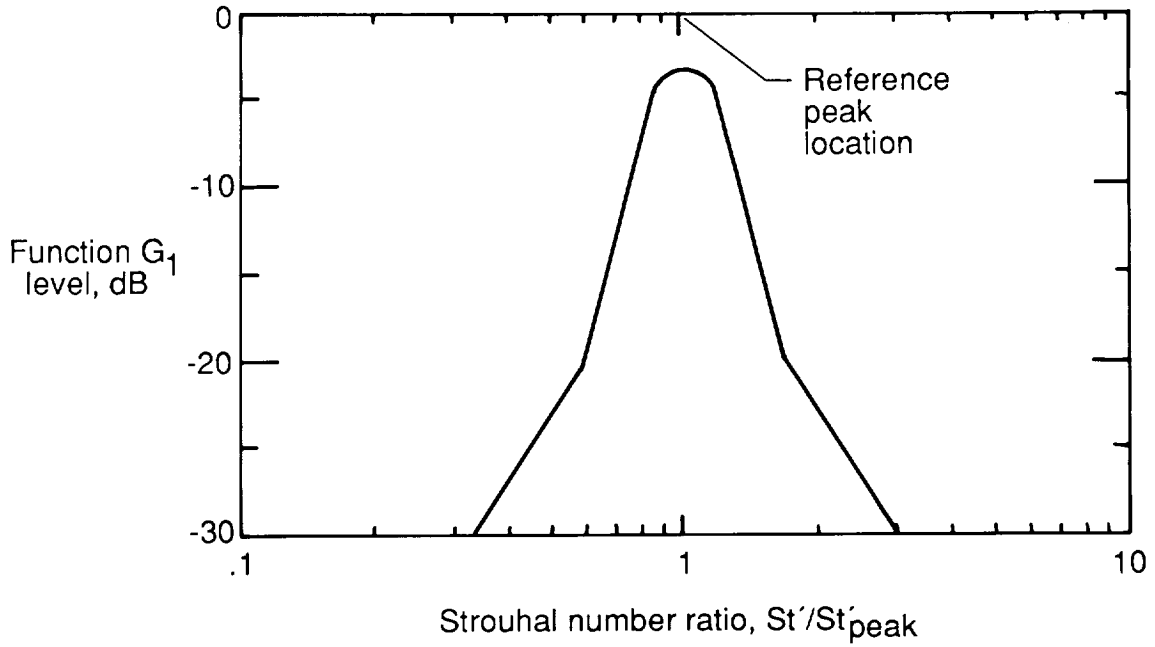


Figure 85. One-third-octave spectral shape function G_1 for LBL-VS noise, equation (57).

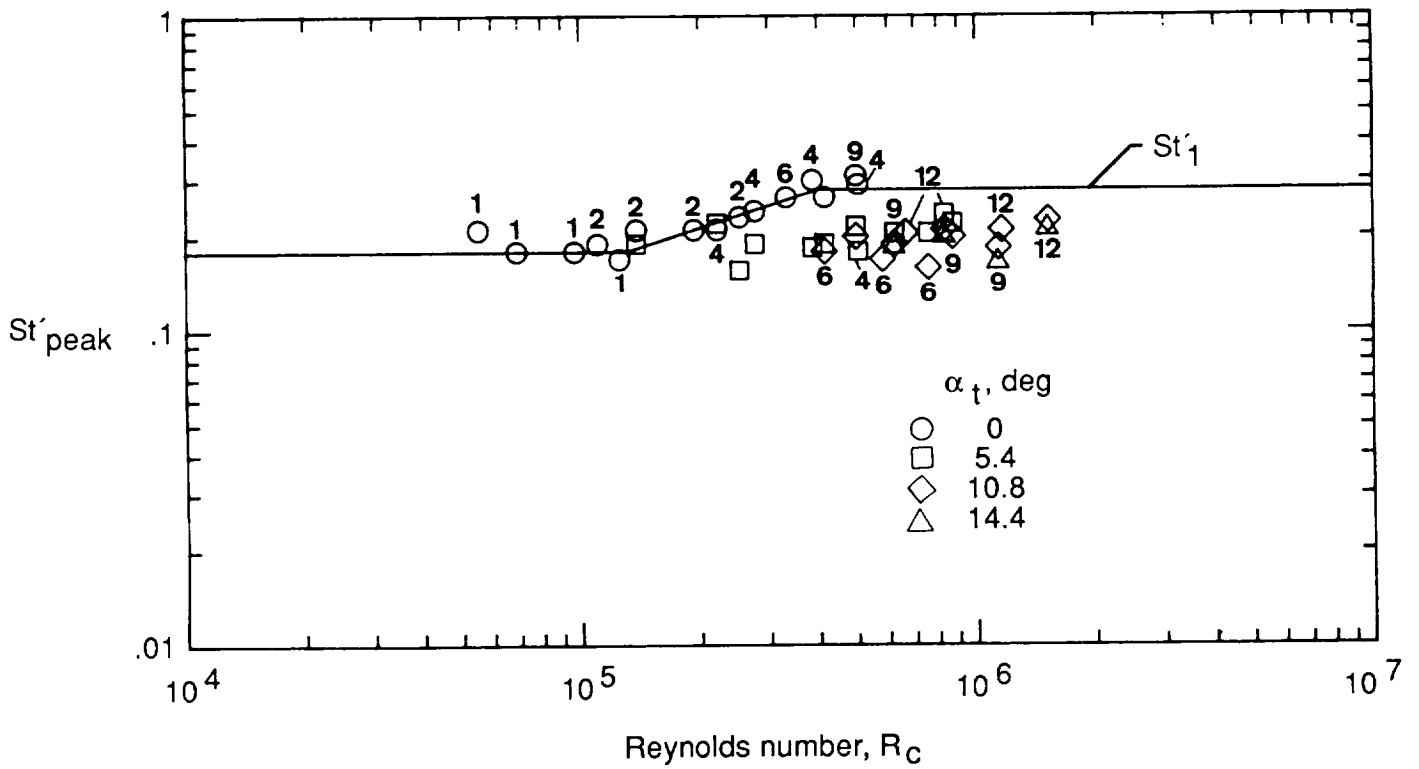


Figure 86. Peak Strouhal number for LBL-VS noise versus Reynolds number. Numbers aligned with data are chord sizes in inches.

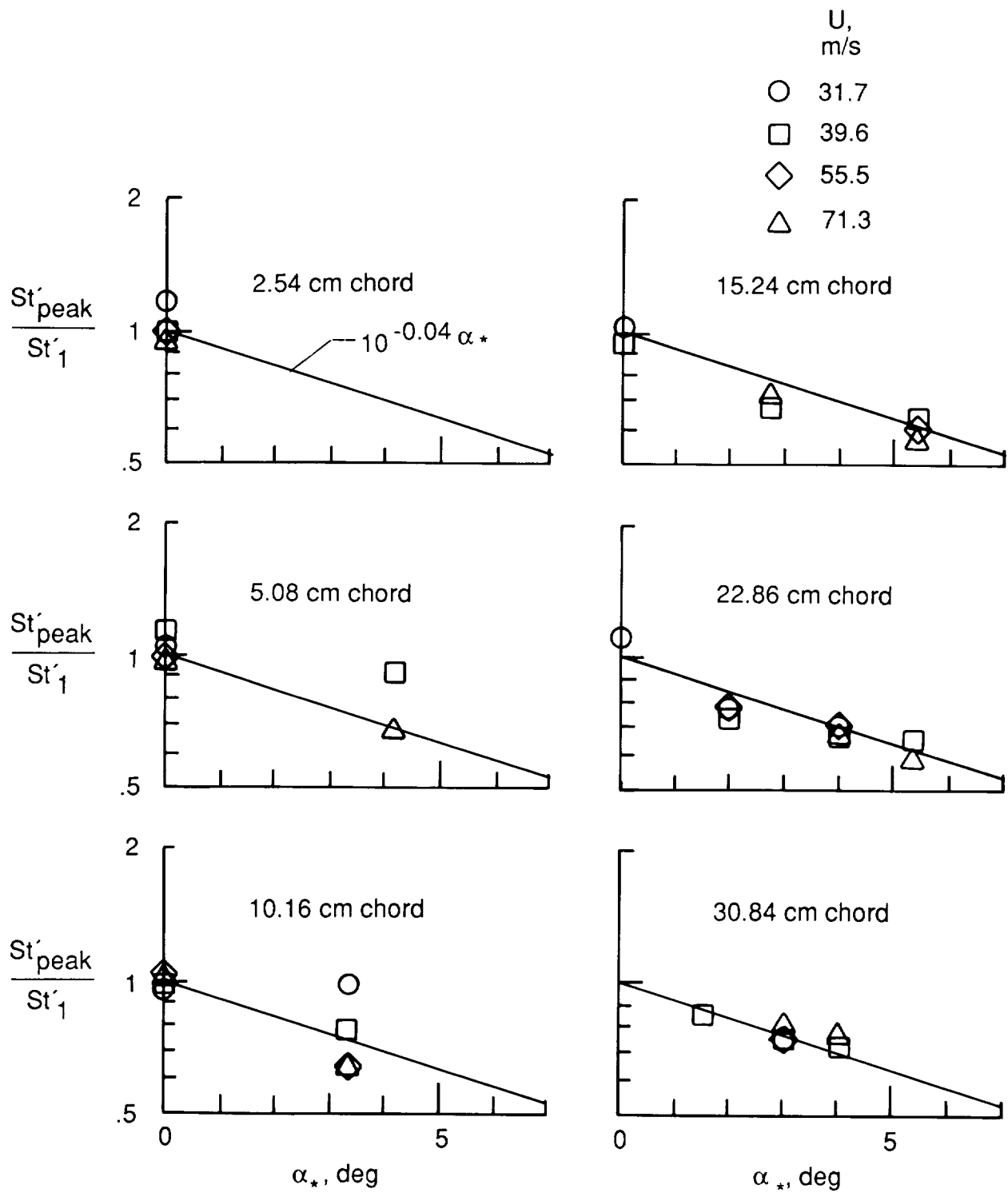


Figure 87. Peak Strouhal number for LBL-VS noise versus angle of attack. Data from figure 86.

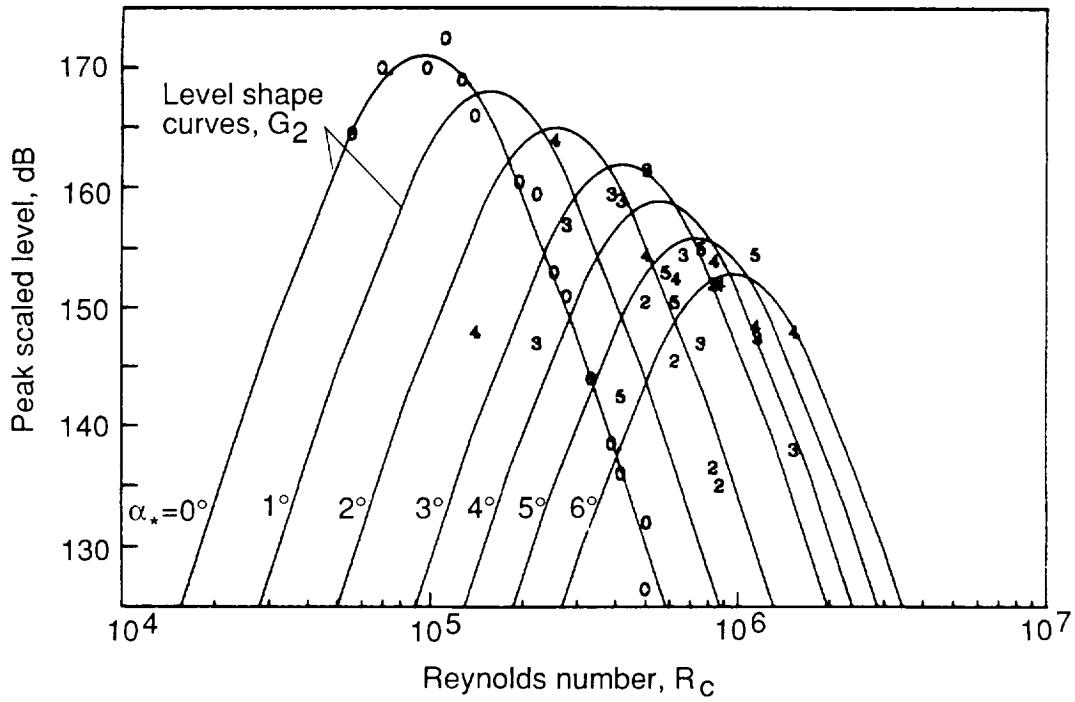


Figure 88. Peak scaled levels for LBL-VS noise versus Reynolds number. Data symbols are values of α_* , rounded off to nearest degree.

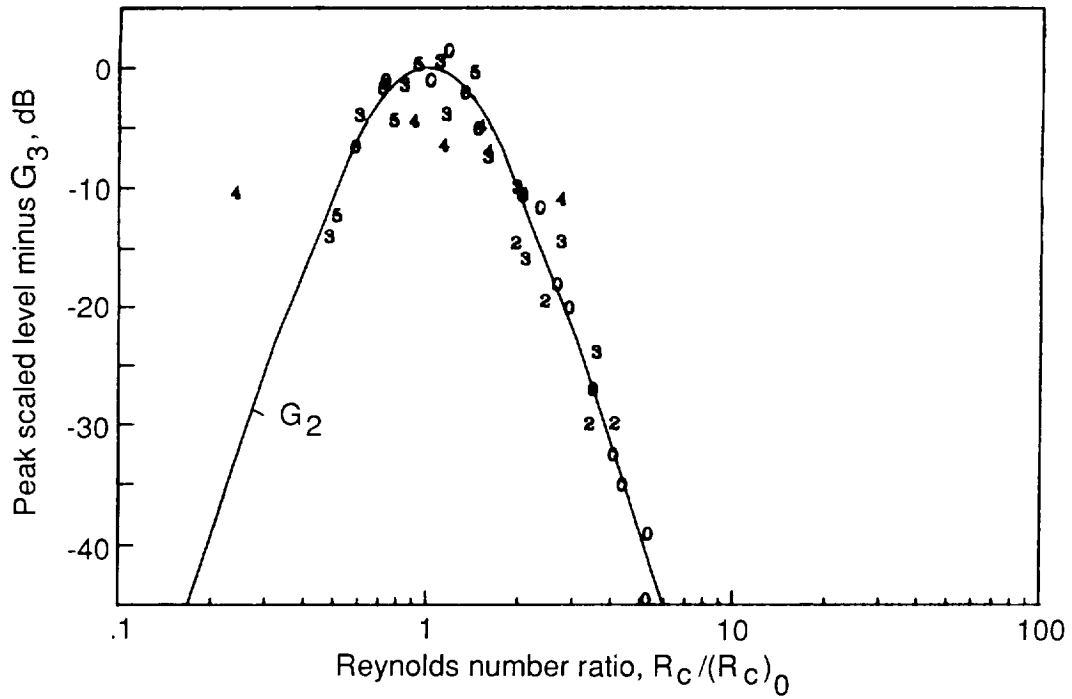


Figure 89. Normalization of LBL-VS noise peak scaled levels by functions G_2 , equations (58) and (59), and G_3 , equation (60). Data from figure 88. Data symbols are values of α_* , rounded off to nearest degree.

and

$$St'_{\text{peak}} = St'_1 \times 10^{-0.04\alpha_*} \quad (56)$$

The directivity function \bar{D}_h is given by equation (B1) in appendix B. The function G_1 defines the spectral shape, in terms of the ratio of Strouhal number to its peak, as (see fig. 85)

$$G_1(e) = \begin{cases} 39.8 \log(e) - 11.12 & (e \leq 0.5974) \\ 98.409 \log(e) + 2.0 & (0.5974 < e \leq 0.8545) \\ -5.076 + \sqrt{2.484 - 506.25[\log(e)]^2} & (0.8545 < e \leq 1.17) \\ -98.409 \log(e) + 2.0 & (1.17 < e \leq 1.674) \\ -39.8 \log(e) - 11.12 & (1.674 < e) \end{cases} \quad (57)$$

where $e = St'/St'_{\text{peak}}$. The peak scaled level shape curve G_2 depends on Reynolds number and angle and is (see figs. 88 and 89)

$$G_2(d) = \begin{cases} 77.852 \log(d) + 15.328 & (d \leq 0.3237) \\ 65.188 \log(d) + 9.125 & (0.3237 < d \leq 0.5689) \\ -114.052[\log(d)]^2 & (0.5689 < d \leq 1.7579) \\ -65.188 \log(d) + 9.125 & (1.7579 < d \leq 3.0889) \\ -77.852 \log(d) + 15.328 & (3.0889 < d) \end{cases} \quad (58)$$

where $d = R_c/(R_c)_0$ and the reference Reynolds number is

$$(R_c)_0 = \begin{cases} 10^{0.215\alpha_*+4.978} & (\alpha_* \leq 3.0) \\ 10^{0.120\alpha_*+5.263} & (3.0 < \alpha_*) \end{cases} \quad (59)$$

The angle-dependent level for the shape curve is

$$G_3(\alpha_*) = 171.04 - 3.03\alpha_* \quad (60)$$

5.2.3. Comparison With Data

The spectral predictions from the above equations are compared with the untripped BL airfoil noise data in figures 44 to 74. The great sensitivity of this mechanism to angle and velocity change can be clearly seen. In many respects the prediction agreement in shape, level, and actual occurrence of LBL-VS noise is good. Also as indicated in the last section, the combined contributions of LBL-VS, TBL-TE, and separation noise are important to the total predictions for this untripped BL airfoil data.

5.3. Tip Vortex Formation Noise

The prediction method proposed in this section for tip vortex formation noise is that developed by Brooks and Marcolini (ref. 18). The study isolated this high-frequency broadband self-noise by comparing aerodynamic and acoustic test results of both two-dimensional (2D) and three-dimensional (3D)

airfoil models shown in figures 2 and 3, respectively. The premise of the tip noise determination method was that 3D models produce both tip noise and TBL-TE noise, while the 2D models produce only the latter. The study produced a prediction method in general agreement with the physical model of the mechanism first proposed by George, Najjar, and Kim (ref. 17). The noise is associated with the turbulence in the locally separated flow region at the tip of a lifting blade, where the tip vortex is formed. The flow field is illustrated in figure 90 for an airfoil blade tip at an angle of attack α_{TIP} to the flow of velocity U . The flow over the blade tip consists of a vortex of strength Γ with a thick viscous core whose spanwise extent at the TE is ℓ . The recirculating flow within the core is highly turbulent. The mechanism of noise production is taken to be TE noise due to the passage of this turbulence over the edge and into the wake.

5.3.1. Calculation Procedures

The tip vortex formation noise spectrum in a 1/3-octave presentation is predicted by

$$\text{SPL}_{\text{TIP}} = 10 \log \left(\frac{M^2 M_{\text{max}}^3 \ell^2 \bar{D}_h}{r_e^2} \right) - 30.5(\log \text{St}'' + 0.3)^2 + 126 \quad (61)$$

The Strouhal number is

$$\text{St}'' = \frac{f \ell}{U_{\text{max}}} \quad (62)$$

The directivity function \bar{D}_h is given by equation (B1) in appendix B. The second term on the right side of equation (61), which gives the frequency dependence, is a parabolic fit about a peak Strouhal number of 0.5. The spanwise extent at the TE of the separation due to the tip vortex is, for the tested rounded tip,

$$\ell/c \approx 0.008 \alpha_{\text{TIP}} \quad (63)$$

where c is the chord length and α_{TIP} (see discussion below) is the angle of attack of the tip region to the oncoming flow. The maximum Mach number M_{max} of the flow within or about the separated flow region at the trailing edge is

$$M_{\text{max}}/M \approx (1 + 0.036 \alpha_{\text{TIP}}) \quad (64)$$

where M is the Mach number of the oncoming flow to the airfoil tip region. The velocity corresponding to M_{max} is

$$U_{\text{max}} = c_o M_{\text{max}} \quad (65)$$

Note that in the use of equations (63) and (64) to determine ℓ and M_{max} , α_{TIP} is correctly regarded as the actual angle of attack of the tip to the oncoming flow when the blade under consideration has a large aspect ratio (large span), is untwisted, and encounters uniform flow over its span. This is the reference case in reference 18. When the tip loading characteristics differ from those for the reference case, such as for some rotor and propeller blades, α_{TIP} must be redefined according to computed sectional loading. The redefined α'_{TIP} is

$$\alpha'_{\text{TIP}} = \left[\left(\frac{\partial L'/\partial y}{(\partial L'/\partial y)_{\text{ref}}} \right)_{y \rightarrow \text{TIP}} \right] \alpha_{\text{TIP}} \quad (66)$$

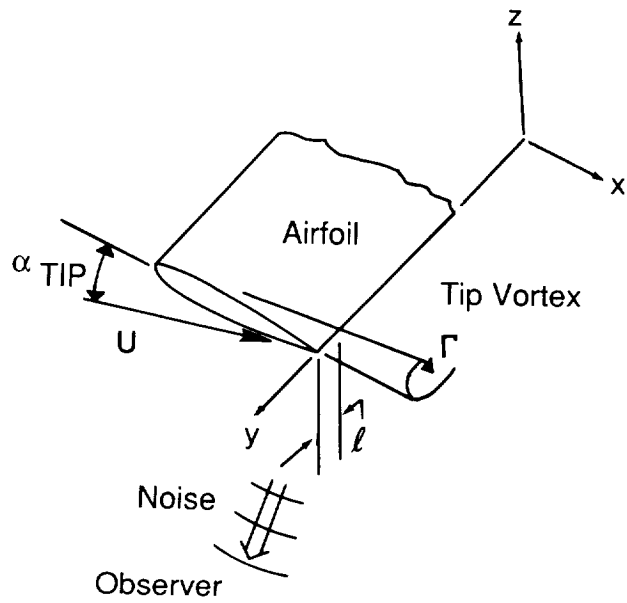


Figure 90. Formation of tip vortex.

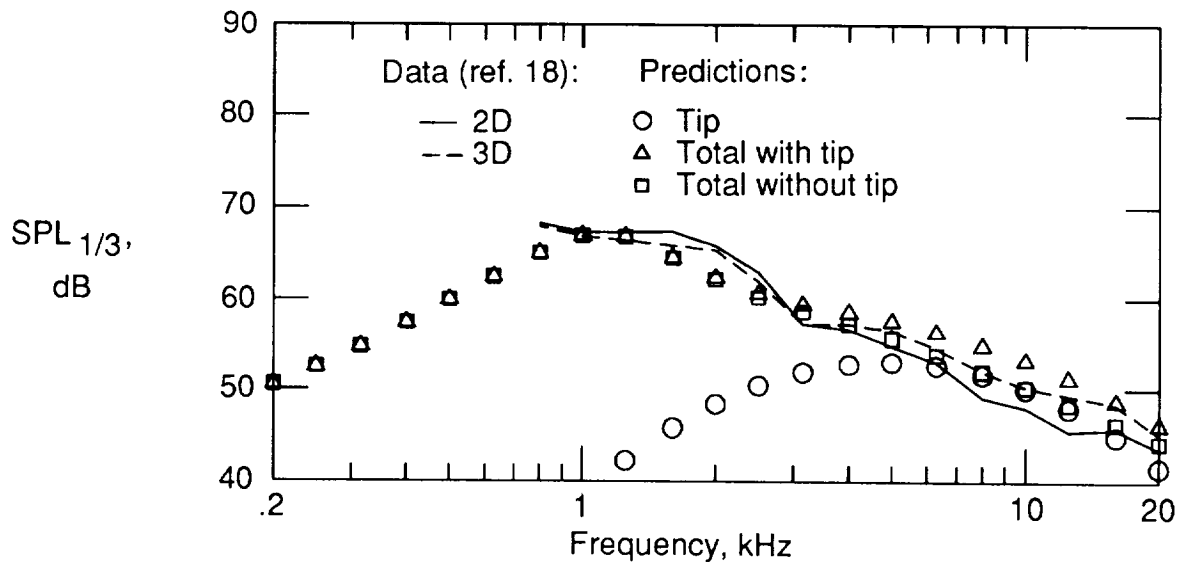


Figure 91. Noise spectra of a 3D 15.24-cm-chord airfoil with a span of 30.48 cm, and that of a 2D airfoil section where levels have been adjusted to match the same span. $U = 71.3$ m/s, $\alpha_t = 10.8^\circ$.

where α_{TIP} is the geometric angle and L' is the lift per unit span at the spanwise position y . The sectional lift slope $\partial L'/\partial y$ near the tip is taken to be proportional to the tip vortex strength Γ (of fig. 90). When tip loading is found to be high, the predicted tip noise levels increase. The use of α'_{TIP} rather than α_{TIP} in equations (63) and (64) generalizes the solution for arbitrary aspect ratios, blade twist, and spanwise flow variations. Reference 18 contains examples which provide guidance in the evaluation of equation (66) for aspect ratio, as well as tunnel testing effects.

The tip noise prediction equations are based on data from airfoils with rounded blade tips. Of interest is a flat (square-off or cut-off) tip geometry which reference 8 considered, along with rounded tips, in calculations employing limited tip flow measurements reported in the literature. The different tip geometries required a different definition of the separated flow region size ℓ . In applying the prediction equations of the present paper for flat tips, it does not appear appropriate to use the definition of reference 8. The constants in equation (61) reflect the definition of ℓ in equation (63). The measurements did not confirm the definition of ℓ for rounded tips proposed by reference 8. For consistency, the following definition for ℓ is proposed for flat tips for the present prediction equations:

$$\ell/c = \begin{cases} 0.0230 + 0.0169\alpha'_{\text{TIP}} & (0^\circ \leq \alpha'_{\text{TIP}} \leq 2^\circ) \\ 0.0378 + 0.0095\alpha'_{\text{TIP}} & (2^\circ < \alpha'_{\text{TIP}}) \end{cases} \quad (67)$$

This definition of ℓ approximately accounts for differences between the definition of reference 8 and that of equation (63) for rounded tips. There is at present no experimental confirmation of equation (67).

5.3.2. Comparison With Data

Noise data from reference 18 (fig. 7) are presented in figure 91 along with predictions of tip noise and the combined contributions of TBL-TE and separation noise. The rounded tip 3D model has a chord of 15.24 cm and a span of 30.5 cm. The corresponding 2D model has a span of 45.7 cm so its noise spectrum levels in the figure were adjusted downward by 1.8 dB (based on a $10 \log(L)$ dependency) to obtain that expected for a 30.5-cm span. The difference between the 2D and 3D spectra should be that due to tip noise. The predictions in figure 91 for TBL-TE and separation noise, which employed the angle $\alpha_* = 0.5(10.8^\circ)$ to account for the wind tunnel correction, should ideally match the 2D model spectrum. The tip noise prediction adds to the prediction to obtain a total which should match the 3D model spectrum. The tip noise prediction involved the use of equation (66) because of the finite extent of the span as well as open wind tunnel influences. Based on the lift distributions presented in reference 18, the tip angle becomes $\alpha'_{\text{TIP}} = 0.71(10.8^\circ)$. While a slight overprediction at higher frequencies is seen in figure 91 for this particular example, the differences between levels with and without tip noise are the same for both data and prediction. The comparison shows consistency and compatibility not only with the data but also between the self-noise prediction methods.

5.4. Trailing-Edge-Bluntness-Vortex-Shedding Noise

In this section, the experiment of reference 2 is briefly described, published and previously unpublished TE bluntness noise data from the study are presented, and a prediction method is developed.

5.4.1. Experiment

The Brooks-Hodgson experiment (ref. 2) employed an experimental arrangement similar to that reported in section 2 of the present paper with respect to hardware and acoustic measurement. However, in reference 2, the model airfoil tested was large with a 60.96-cm chord length. When BL tripping was used, 2.0-cm-wide strips of No. 40 grit were applied at 15 percent of the chord. Rather than the TE being sharp, the model TE thickness, or bluntness, was $h = 2.5$ mm. Figure 92 shows the TE region of the airfoil. The TE geometry was rounded at the two edges and flat between the rounded edge portions, which each comprised about one-third of the 2.5-mm thickness. The thickness h was varied, with edges of similar geometry, by alternately attaching extensions on the edge, as illustrated in figure 92(a). Also tested were sharp-edge ($h = 0$) plate extensions 15.24 and 30.48 cm long, as shown in figure 92(b). Another sharp-edge extension (not shown) was a 2.54-cm-long "flap" extension placed at 17.5° off the chord mean axis at the trailing edge. In addition, blunt plate extensions were tested which were 15.24 cm long with

$h = 2.5$ and 4.8 mm and 30.48 cm long with $h = 4.8$ mm. These extensions with rounded TE corners are shown in figure 92(c). Tape, 0.08 mm thick, was used to provide a smooth surface transition from the airfoil to the extensions.

Presented in figure 93, from reference 2, are power spectral noise data of the airfoil at four flow velocities. The airfoil is at zero angle of attack and the boundary layers are tripped. The microphone observer position is $r_e = 1.22$ m and $\Theta_e = 90^\circ$ with respect to the model trailing edge. For two speeds, the spectra are given for the four TE thicknesses of figure 92(a). The spectral results for the sharp, $h = 0$, TE cases should be all due to TBL-TE noise. The bluntness contributes additively at high frequencies to the spectrum levels. The values given for h/δ^* in figure 93 differ slightly from those specified in reference 2 because δ^* here is calculated from the BL thickness scaling equations of the present paper. Data are presented in reference 2 for the sharp geometries of figure 92(b), as well as the mentioned 17.5° sharp flap extension. These geometries give essentially the same spectra as the sharp extension of figure 92(a). This demonstrates that TBL-TE noise is rather invariant with regard to geometry changes in the edge region, as long as the TE is sharp and the boundary layers are substantially the same.

Trailing-edge bluntness noise spectra in a smoothed 1/3-octave format are presented in figure 94 for the edge geometries of figures 92(a) and 92(c). These spectra are the result of a spectral subtraction process between the total spectra and the corresponding sharp TE spectra and should thus represent the bluntness contribution only. With the exception of the eight spectra also represented in figure 93, the data have not been previously published. The indicated values of h/δ^* for the extensions are based on calculations of δ^* for the TE of the airfoil without the extensions. This is justified by indications that the boundary layers did not substantially change over the zero pressure gradient extension plates due to the influence of the upstream adverse pressure gradient (ref. 2). The spectrum for the airfoil with $h = 2.5$ mm and $h/\delta^* = 1.15$ in figure 94 is for naturally transitional boundary layers; all others are for tripped boundary layers.

5.4.2. Scaled Data

The spectra of figure 94, as well as limited frequency data of Blake (ref. 19), form the foundation of the scaling approach. As with the scaling approach for TBL-TE and LBL-VS noise, the level, frequency, and spectral shape are modeled as functions of flow and geometric parameters. For the level and frequency definition, we chose the peak of the

spectral humps as the reference. The peak value of Strouhal number, defined as

$$St_{\text{peak}}''' = \frac{f_{\text{peak}} h}{U} \quad (68)$$

is plotted versus the thickness ratio h/δ^* in figure 95. The Strouhal numbers increase with increases in thickness ratio. The Strouhal numbers for the plate extensions of figure 92(c) are uniformly higher, for the same thickness ratios, than for the edge extensions of figure 92(a). Also shown are two results obtained from Blake (ref. 19). Blake presents surface pressure data for a large array of plate edge geometries—all for very large values of h/δ^* (with the exception of the ref. 2 data reported and the one case shown in fig. 95 at $h/\delta^* = 5.19$). Blake, for most data, employed Strouhal relationships which depend on special wake stream thicknesses, and convection velocities not available without measurements. From Blake, however, it is obvious that different TE geometries have different frequency dependences, consistent with the result of figure 95 that Strouhal numbers for the flat plate extension and the airfoil TE geometries differ. The primary difference between the geometries is that the NACA 0012 airfoil has a beveled or sloping surface upstream of the trailing edge with a solid angle Ψ of 14° and the flat plate has $\Psi = 0^\circ$. The result shown from Blake in figure 95 at $h/\delta^* = 5.19$ is for a plate with $\Psi = 12.5^\circ$ and nonrounded TE corners. In figure 95, parallel curves are fitted to the data. The curves, designated with values of Ψ , are defined on the basis of a match point at $h/\delta^* = 20$ for $\Psi = 0^\circ$. From Blake's scaling for a thick flat plate (h/δ^* large) with nonrounded TE corners, one can determine that $fh/U = 0.21$ at $h/\delta^* = 20$. The curve for $\Psi = 14^\circ$ intercepts Blake's $\Psi = 12.5^\circ$ result, but this is deemed an acceptable deviation from the curve fit. For scaling purposes, values of St_{peak}''' for Ψ values other than 0° and 14° could be determined by linear interpolation as described in the calculation procedure section to follow.

For amplitude scaling, the peak values of the 1/3-octave spectra of figure 94 were normalized as

$$\text{Scaled peak SPL}_{1/3} = \text{Peak SPL}_{1/3} - 10 \log \left(\frac{M^{5.5} h L}{r_e^2} \right) \quad (69)$$

The 5.5 power for Mach number dependence was determined to give better overall scaling success than either a 5 or 6 power. Figure 96 shows the scaled levels plotted versus the thickness ratio h/δ^* . As in figure 95 for the Strouhal dependency, the scaled levels are uniformly higher for the plates than for the

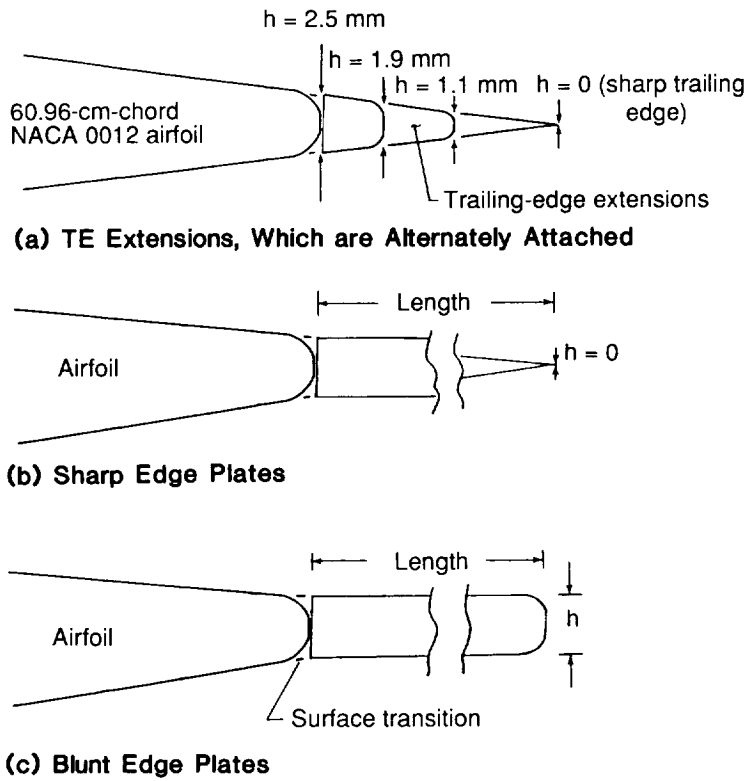


Figure 92. Illustration of trailing-edge extensions and plates. Smooth surface transition is provided for all geometries.

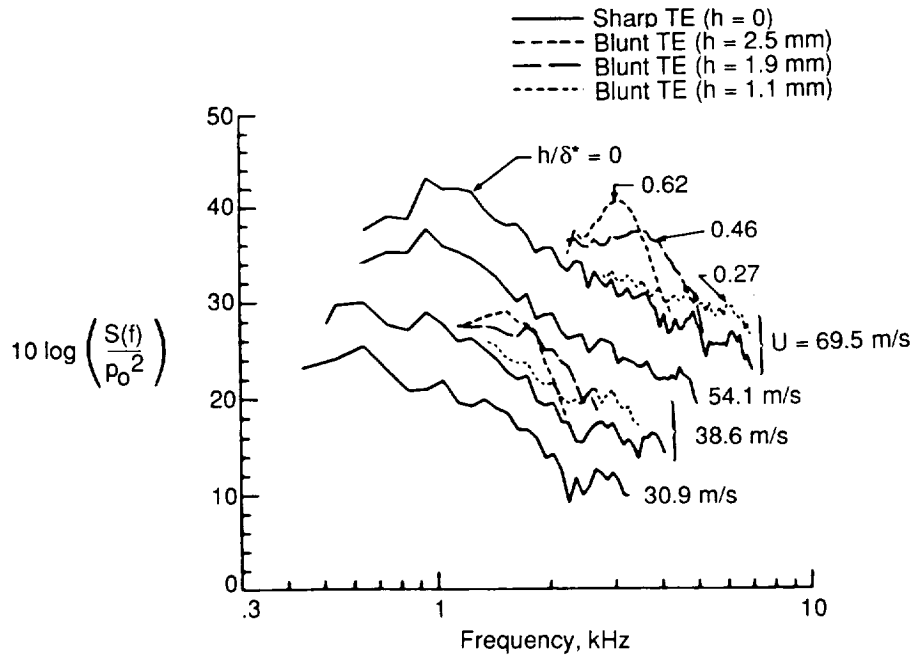


Figure 93. Spectral density for TE noise for 60.96-cm-chord airfoil with various degrees of TE bluntness. Tripped BL; $\alpha_t = 0^\circ$; $\Theta_e = 90^\circ$. Level referenced to 1-Hz bandwidth. Data from reference 2.

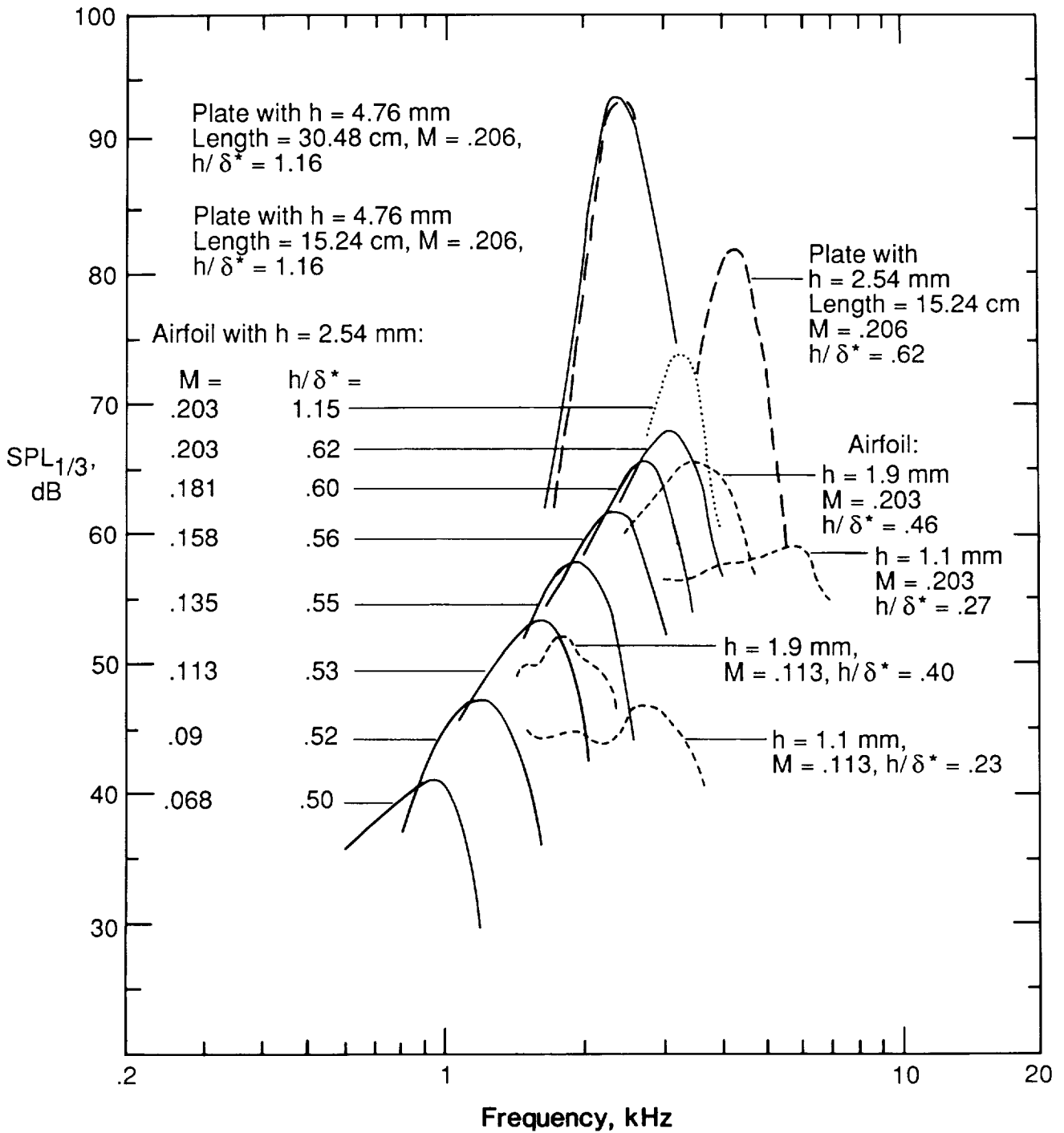


Figure 94. TE-bluntness-vortex-shedding noise, extracted from data of figure 93, data for untripped BL, and data with plate extensions (fig. 92(c)) attached.

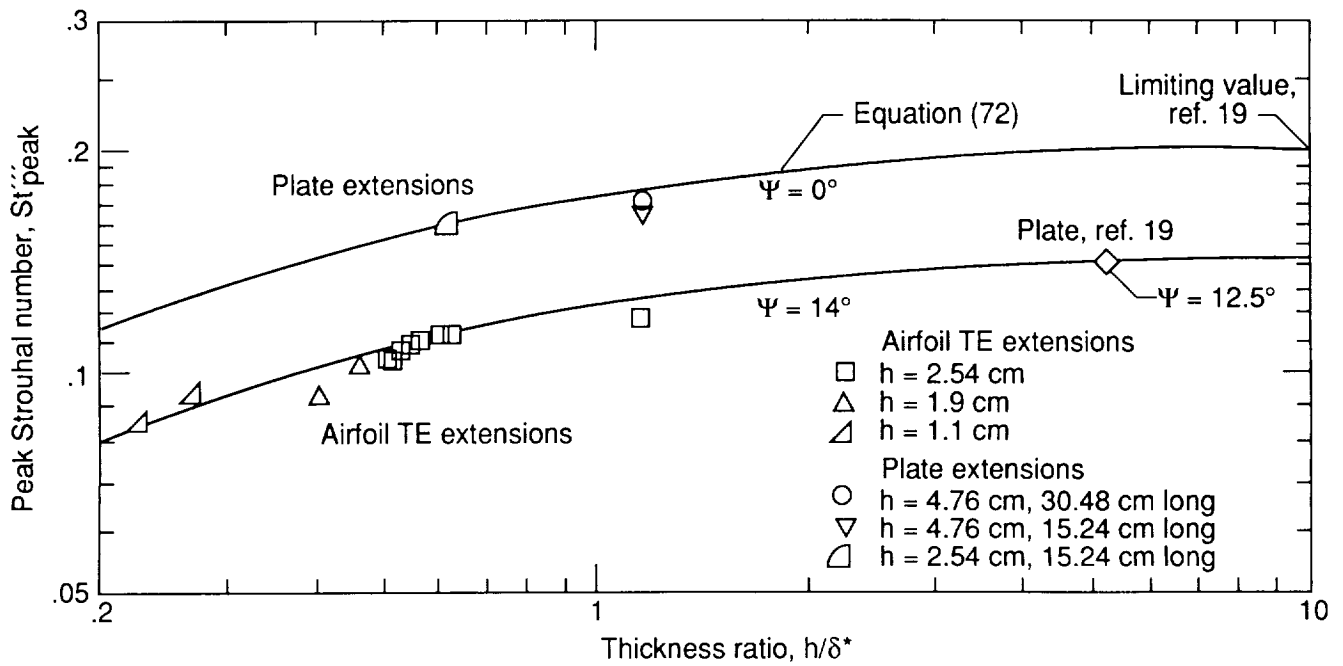


Figure 95. Peak Strouhal number for bluntness noise versus thickness ratio h/δ^* determined from figure 94 and Blake (ref. 19).

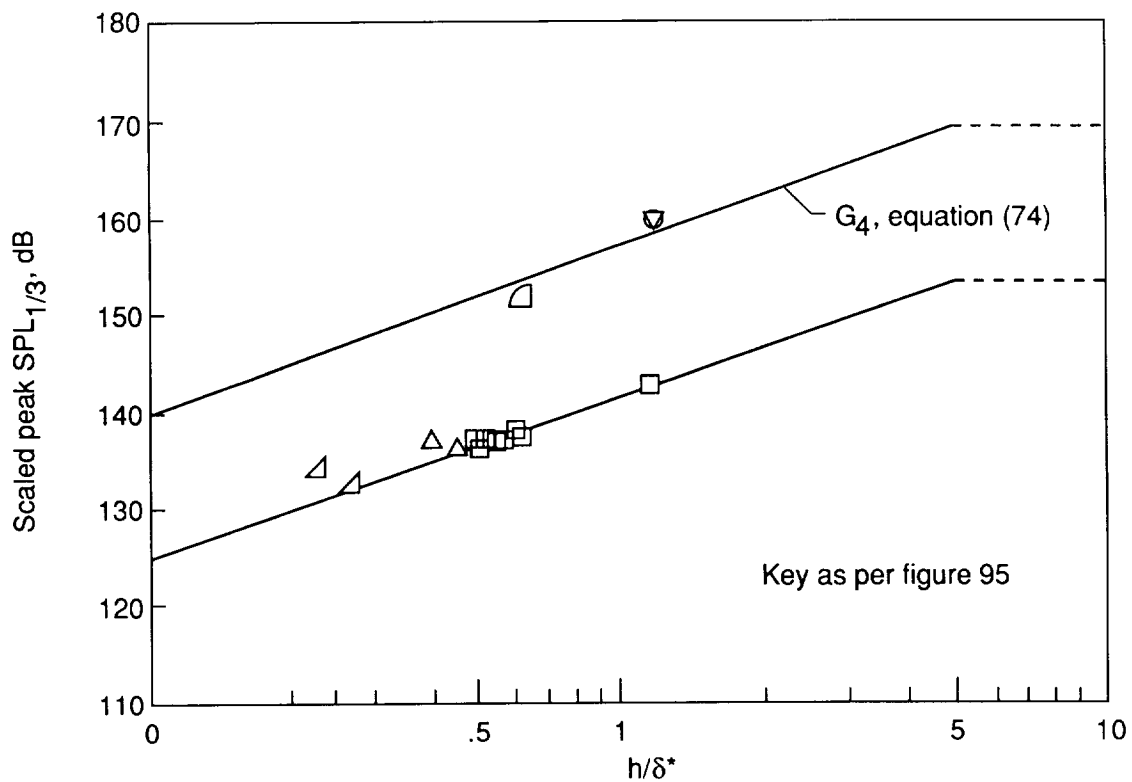


Figure 96. Peak scaled levels for bluntness noise versus thickness ratio h/δ^* determined from figure 94.

edge extensions for the same thickness ratios. The levels increase with increasing thickness ratios. The edge extension data for the two smaller thicknesses of $h = 1.1$ and 1.9 mm at $M = 0.113$ deviate most from a straight line trend. Because of signal-to-noise concerns in the specification of these points, these data have the least confidence in the figure and are thus ignored in the specification of a curve fit. However, the accuracy of the resultant scaling equations in predicting these data is subsequently examined. The curve fits, designated as $G_4(h/\delta^*, \Psi)$, shown for the data are straight lines which are chosen to level off at $h/\delta^* = 5$. The curve fit behavior at high h/δ^* is admittedly rather arbitrary, but there are no noise data available for guidance, unlike in the above Strouhal scaling where some frequency data from Blake are used. Fortunately, in practice, the likely values of h/δ^* to be found for rotor blades and

wings should be in the range where data are present and scaling confidence is greatest.

Given the specification of the functions St'''_{peak} and G_4 , a definition of the spectral shape completes the scaling. Spectral curve fits for the data of figure 94 are shown for the airfoil TE extensions, $\Psi = 14^\circ$, and for the plate extensions, $\Psi = 0^\circ$, in figures 97(a) and 97(b), respectively. The shapes reflect the observations that the spectra are sharper for the plates for the same h/δ^* , and the spectra widen in the lower frequencies for decreased h/δ^* values for both the plates and the edge extensions. The spectral curve fit is specified as the function $G_5(h/\delta^*, \Psi)$ whose peak level is 0 dB and whose shape is defined in terms of $St'''/St'''_{\text{peak}}$. The specification of G_5 for in-between values of Ψ would be an interpolation between the limiting cases shown in figures 97(a) and 97(b).

5.4.3. Calculation Procedures

The TE bluntness noise spectrum in a 1/3-octave presentation is predicted by

$$\text{SPL}_{\text{BLUNT}} = 10 \log \left(\frac{hM^{5.5}L\bar{D}_h}{r_e^2} \right) + G_4 \left(\frac{h}{\delta_{\text{avg}}^*}, \Psi \right) + G_5 \left(\frac{h}{\delta_{\text{avg}}^*}, \Psi, \frac{St'''}{St'''_{\text{peak}}} \right) \quad (70)$$

The directivity function \bar{D}_h is given by equation (B1) in appendix B. The Strouhal definitions are (see fig. 95)

$$St''' = \frac{fh}{U} \quad (71)$$

and

$$St'''_{\text{peak}} = \begin{cases} \frac{0.212 - 0.0045\Psi}{1 + 0.235 \left(h/\delta_{\text{avg}}^* \right)^{-1} - 0.0132 \left(h/\delta_{\text{avg}}^* \right)^{-2}} & (0.2 \leq h/\delta_{\text{avg}}^*) \\ 0.1(h/\delta_{\text{avg}}^*) + 0.095 - 0.00243\Psi & (h/\delta_{\text{avg}}^* < 0.2) \end{cases} \quad (72)$$

The h/δ_{avg}^* term is the ratio of TE thickness (degree of bluntness) h to the average boundary-layer displacement thickness δ_{avg}^* , where

$$\delta_{\text{avg}}^* = \frac{\delta_p^* + \delta_s^*}{2} \quad (73)$$

The angle Ψ is the solid angle, in degrees, between the sloping surfaces upstream of the trailing edge. For an edge on a flat plate $\Psi = 0^\circ$, whereas $\Psi = 14^\circ$ for an NACA 0012 airfoil. The determination for this parameter for other TE geometries is discussed in section 6 and appendix C.

The peak level of the spectrum is determined from the function G_4 (see fig. 96) where

$$G_4 \left(\frac{h}{\delta_{\text{avg}}^*}, \Psi \right) = \begin{cases} 17.5 \log \left(h/\delta_{\text{avg}}^* \right) + 157.5 - 1.114\Psi & (h/\delta_{\text{avg}}^* \leq 5) \\ 169.7 - 1.114\Psi & (5 < h/\delta_{\text{avg}}^*) \end{cases} \quad (74)$$

The shape of the spectrum is defined by the function G_5 (see figs. 97(a) and 97(b)) where the calculation procedure involves an interpolation between the spectra for $\Psi = 0^\circ$ and 14° as follows:

$$G_5 \left(\frac{h}{\delta_{\text{avg}}^*}, \Psi, \frac{St'''}{St'''_{\text{peak}}} \right) = (G_5)_{\Psi=0^\circ} + 0.0714\Psi [(G_5)_{\Psi=14^\circ} - (G_5)_{\Psi=0^\circ}] \quad (75)$$

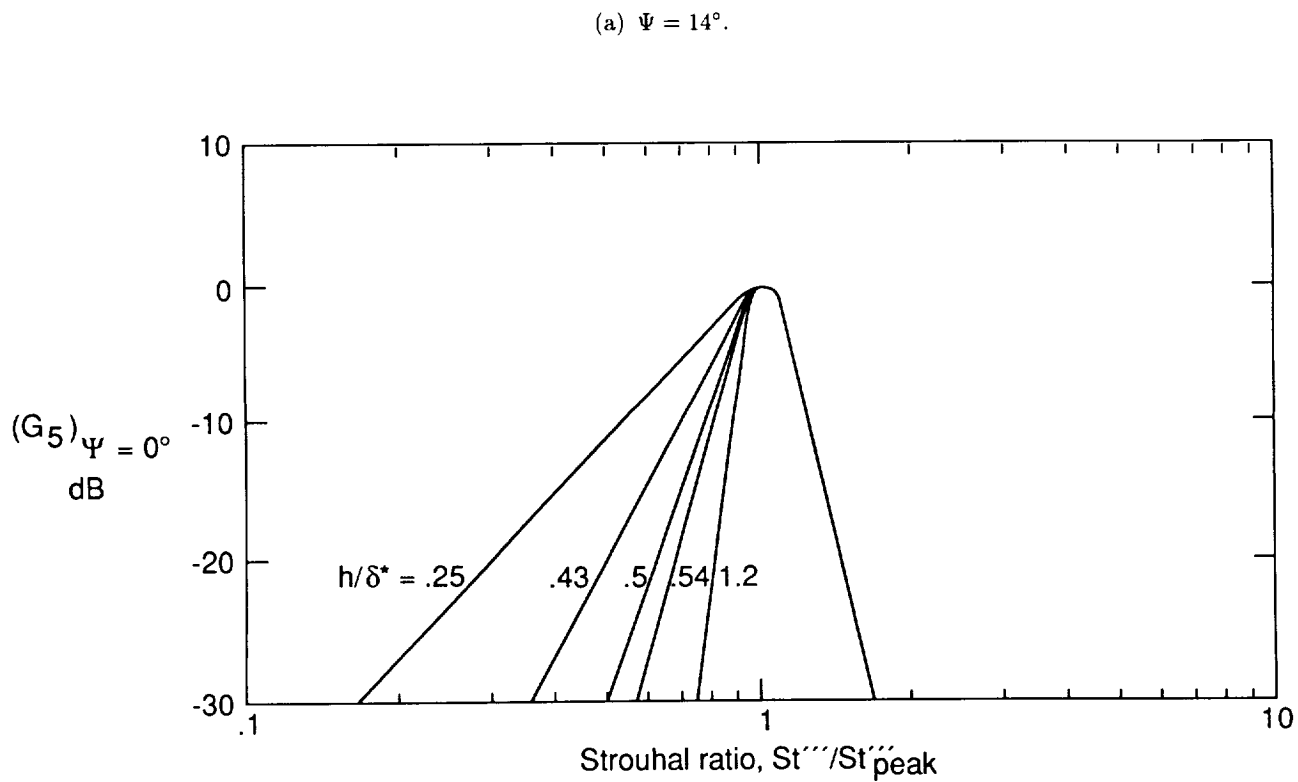
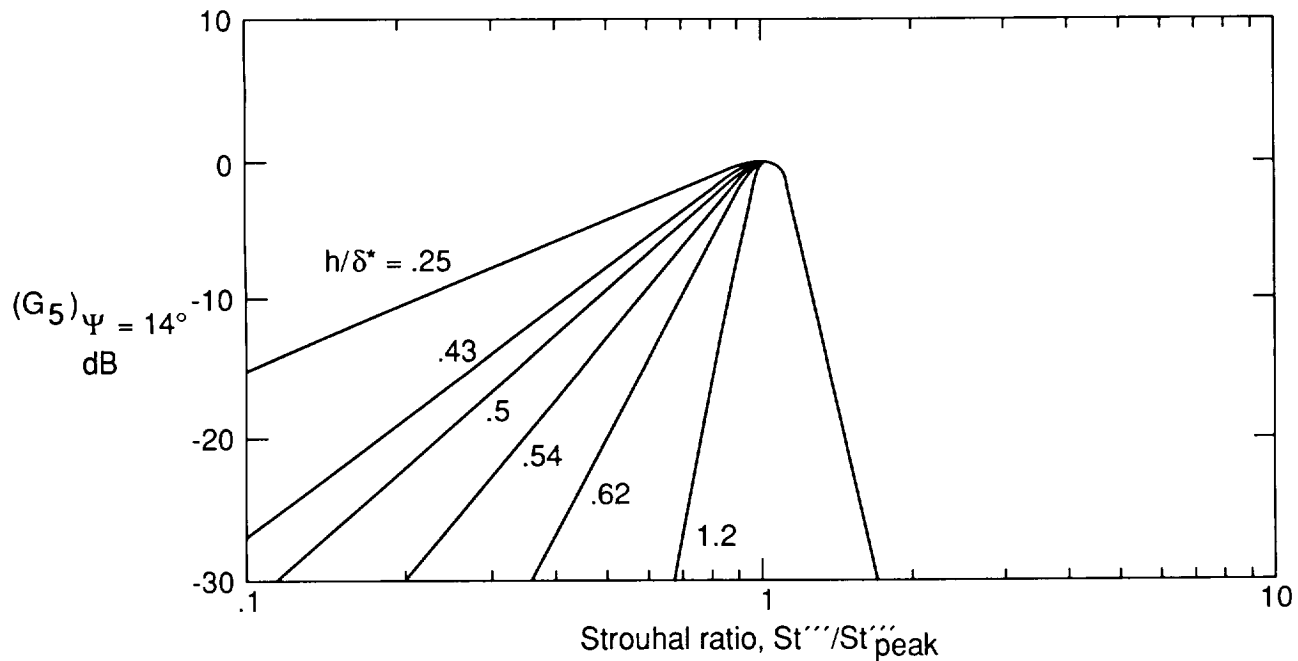


Figure 97. Spectral shape functions for TE bluntness noise.

where

$$(G_5)_{\Psi=14^\circ} = \begin{cases} m\eta + k & (\eta < \eta_0) \\ 2.5\sqrt{1 - (\eta/\mu)^2} - 2.5 & (\eta_0 \leq \eta < 0) \\ \sqrt{1.5625 - 1194.99\eta^2} - 1.25 & (0 \leq \eta < 0.03616) \\ -155.543\eta + 4.375 & (0.03616 \leq \eta) \end{cases} \quad (76)$$

$$\eta = \log(\text{St}''' / \text{St}_{\text{peak}}''') \quad (77)$$

$$\mu = \begin{cases} 0.1221 & (h/\delta_{\text{avg}}^* < 0.25) \\ -0.2175(h/\delta_{\text{avg}}^*) + 0.1755 & (0.25 \leq h/\delta_{\text{avg}}^* < 0.62) \\ -0.0308(h/\delta_{\text{avg}}^*) + 0.0596 & (0.62 \leq h/\delta_{\text{avg}}^* < 1.15) \\ 0.0242 & (1.15 \leq h/\delta_{\text{avg}}^*) \end{cases} \quad (78)$$

$$m = \begin{cases} 0 & (h/\delta_{\text{avg}}^* \leq 0.02) \\ 68.724(h/\delta_{\text{avg}}^*) - 1.35 & (0.02 < h/\delta_{\text{avg}}^* \leq 0.5) \\ 308.475(h/\delta_{\text{avg}}^*) - 121.23 & (0.5 < h/\delta_{\text{avg}}^* \leq 0.62) \\ 224.811(h/\delta_{\text{avg}}^*) - 69.35 & (0.62 < h/\delta_{\text{avg}}^* \leq 1.15) \\ 1583.28(h/\delta_{\text{avg}}^*) - 1631.59 & (1.15 < h/\delta_{\text{avg}}^* < 1.2) \\ 268.344 & (1.2 < h/\delta_{\text{avg}}^*) \end{cases} \quad (79)$$

$$\eta_0 = -\sqrt{\frac{m^2\mu^4}{6.25 + m^2\mu^2}} \quad (80)$$

and

$$k = 2.5\sqrt{1 - \left(\frac{\eta_0}{\mu}\right)^2} - 2.5 - m\eta_0 \quad (81)$$

The spectrum $(G_5)_{\Psi=0^\circ}$ is obtained by computing equations (76) through (81), as one would for $(G_5)_{\Psi=14^\circ}$, but replacing $(h/\delta_{\text{avg}}^*)$ by $(h/\delta_{\text{avg}}^*)'$ where

$$\left(\frac{h}{\delta_{\text{avg}}^*}\right)' = 6.724\left(\frac{h}{\delta_{\text{avg}}^*}\right)^2 - 4.019\left(\frac{h}{\delta_{\text{avg}}^*}\right) + 1.107 \quad (82)$$

5.4.4. Comparison With Data

Noise spectra for the airfoil with different TE thicknesses (geometry of fig. 92(a)) are presented for the flow Mach numbers of $M = 0.21$ and 0.12 in figures 98 and 99, respectively. The data were obtained by digitizing the spectra of figure 93 and converting these to 1/3-octave levels. The prediction curves shown are those of TBL-TE and bluntiness noise sources. For the sharp TE of figures 98(a)

and 99(a), there is no bluntiness contribution. Overprediction is seen for the TBL-TE noise at the lowest frequencies and some underprediction is apparent in the higher frequencies for the highest flow speed. For the nonzero TE thicknesses the bluntiness noise contributes to the total spectra at high frequencies and renders good comparisons with the data. Good agreement is found even for the aforementioned smaller thickness cases at low Mach number (figs. 99(c) and 99(d)).

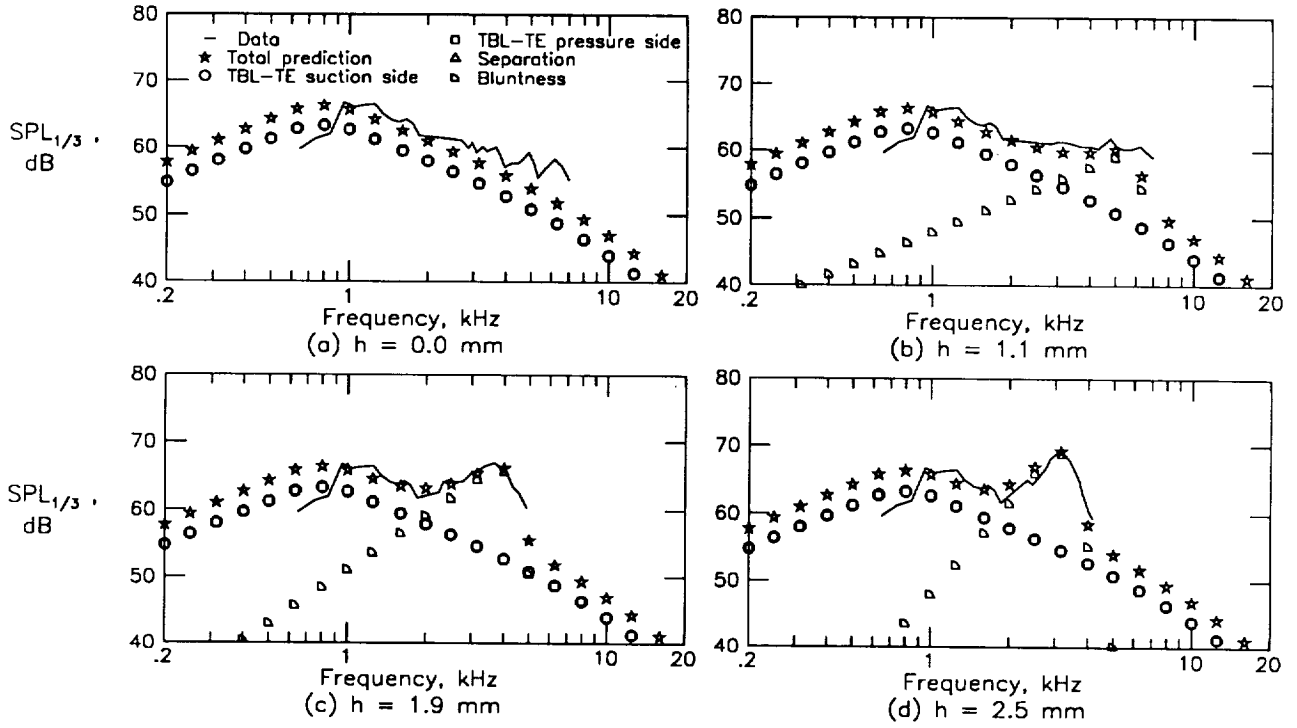


Figure 98. One-third-octave presentation of spectra of figure 93 at $U = 69.5$ m/s with predictions for various degrees of bluntness.

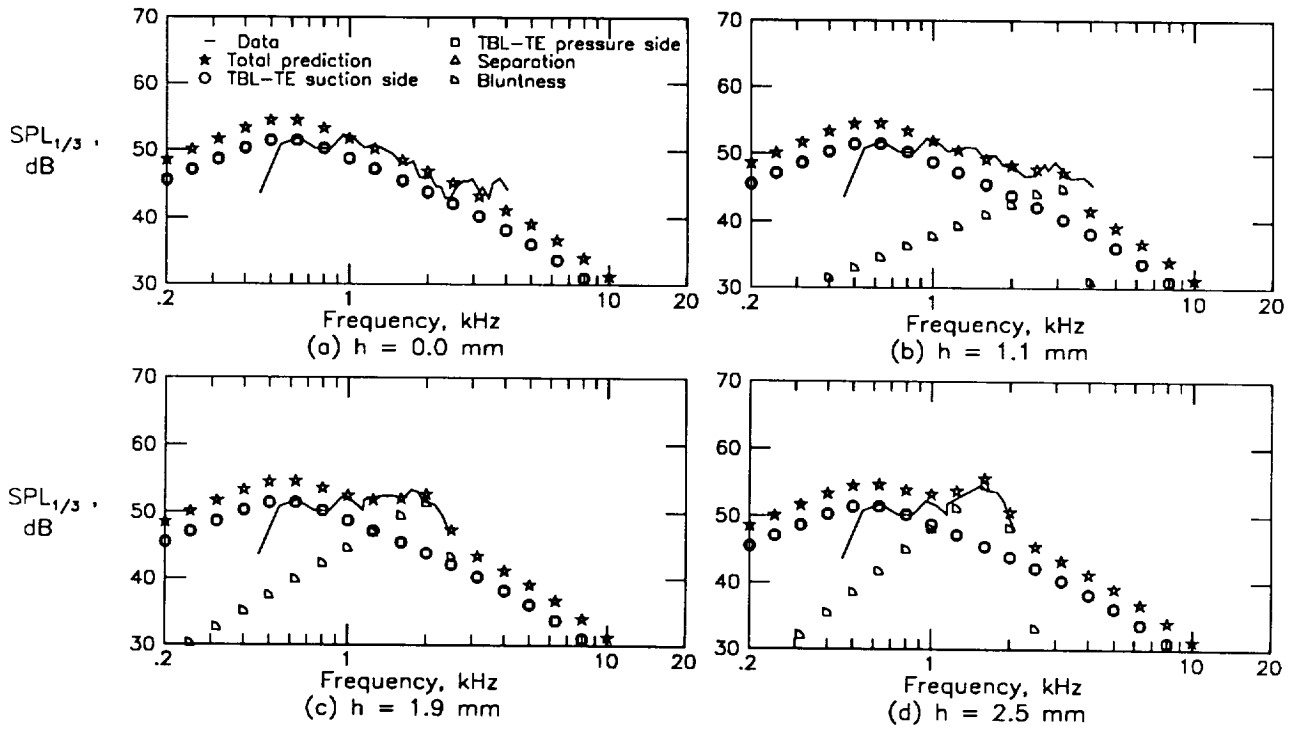


Figure 99. One-third-octave presentation of spectra of figure 93 at $U = 38.6$ m/s with predictions for various degrees of bluntness.

6. Comparison of Predictions With Published Results

The scaling law predictions are compared in this section with data from self-noise studies of airfoil sections performed at the United Technologies Research Center (UTRC).

6.1. Study of Schlinker and Amiet

Schlinker and Amiet (ref. 3) conducted tests in the UTRC Acoustic Research Tunnel to study TBL-TE noise from a cambered helicopter blade section. The cross section of the 40.6-cm-chord and 53.3-cm-span model is shown in figure 100. As in the present NASA Langley studies, the model was mounted on sidewalls and spanned the width of the open tunnel jet, so that the flow across the model was two-dimensional. The nozzle providing the flow had a rectangular exit of dimensions of 29 cm \times 53.3 cm. To isolate the TBL-TE noise from facility background noise, a directional microphone system was used. The experimental configuration, illustrating the shear layer refraction effect on the TE noise received by the directional microphone, is shown in figure 101. The Mach numbers tested ranged from 0.1 to 0.5 and the tunnel angle of attack α_t varied from -0.4° (zero lift for this cambered airfoil) to 12° .

6.1.1. Boundary-Layer Definition

Because only TBL-TE noise measurements were desired, the boundary layers were tripped by applying thin serrated aluminum tape at the blade locations indicated in figure 100. The tape thickness was on the order of the BL displacement thickness at the points of application, providing minimum surface protrusion to avoid unnaturally large TBL thicknesses downstream. This "light" trip is in contrast to the present study where the trips were "heavy" for reasons discussed.

Hot-wire measurements were made in the boundary-layer/near-wake region at the TE of the model. In figure 102, measured BL thicknesses are plotted versus Mach number for various tunnel angles of attack α_t . These data are from figure 17 of reference 3. At zero lift, $\alpha_t = -0.4^\circ$, in figure 102(a), the BL thicknesses δ_0 on the pressure and suction sides are approximately the same. This should be expected since they developed under approximately the same adverse pressure gradient. Included in figure 102(a) are corresponding values of BL displacement thicknesses, which were calculated by the present authors from velocity profiles presented in reference 3 (δ^* was not a quantity of interest in ref. 3). In figures 102(b) and 102(c), δ/c values are shown for $\alpha_t = 7.6^\circ$ and 12° , respectively. Comparing figures 102(a), 102(b),

and 102(c), one can see that as angle of attack increases, δ_s increases and δ_p decreases.

These measurements are compared with the thickness scaling equations of the present paper. First, equations (2) and (3) are used to calculate the BL thickness ratio δ_0/c and displacement thickness ratio δ_0^*/c . To make the calculations agree with the data of figure 102(a), all calculated values of δ_0/c and δ_0^*/c were multiplied by a factor 0.6. This factor is taken to be the adjustment in equations (2) and (3) needed to make them appropriate for the "light" trip of reference 3. Next, the corrected angles of attack are determined by (1) adding 0.4° to α_t so that the tunnel angle is referenced to the zero-lift case and (2) using equation (1), with $c = 40.6$ cm and $H = 79$ cm, to obtain $\alpha_* = 0^\circ, 3.9^\circ,$ and 6.1° for $\alpha_t = -0.4^\circ, 7.6^\circ,$ and 12° , respectively. These values of α_* are now used in equations (8) and (11) to obtain δ_p/δ_0 and δ_s/δ_0 , respectively. The resultant values of δ_s/c and δ_p/c are compared with the data in figures 102(b) and 102(c).

6.1.2. Trailing-Edge Noise Measurements and Predictions

Trailing-edge noise spectra in a 1/3-octave presentation are given in figure 103 for the airfoil at $\alpha_t = -0.4^\circ$ with Mach number ranging from $M = 0.1$ to 0.5. The data were obtained by the directional microphone system at differing orientations to the airfoil. Shear layer corrections and directional microphone gain adjustments were made so that the data shown represent the noise radiated from a unit length of $L = 0.3048$ m of the TE span, at an observer distance of $r_e = 3$ m, and an observer angle Θ_e which is specified in the figure. Figures 104 and 105 contain spectra for the airfoil at $\alpha_t = 7.6^\circ$ and 12° , respectively.

The TBL-TE and separation noise spectra were predicted using the calculation procedures of the present paper. The values of α_* , δ_s^* , and δ_p^* used were calculated as described in the previous section. Because of the BL trips and the 2D flow, no LBL-VS or tip noise calculations were made. In performing the calculations for TE bluntness noise, one has to assign values of the TE thickness h and the TE flow angle parameter Ψ . The thickness was indicated in reference 3 to be $h = 0.38$ mm but the shape of this small TE region was not given. A value of $\Psi = 17^\circ$ has been used in the prediction because it gives reasonable prediction-data comparisons.

In figures 103 to 105, the predictions are compared with the measurements. As in the presentation of figures 11 to 74, the individual noise contributions are shown, along with the total summed spectra. The prediction-data comparisons are good, especially

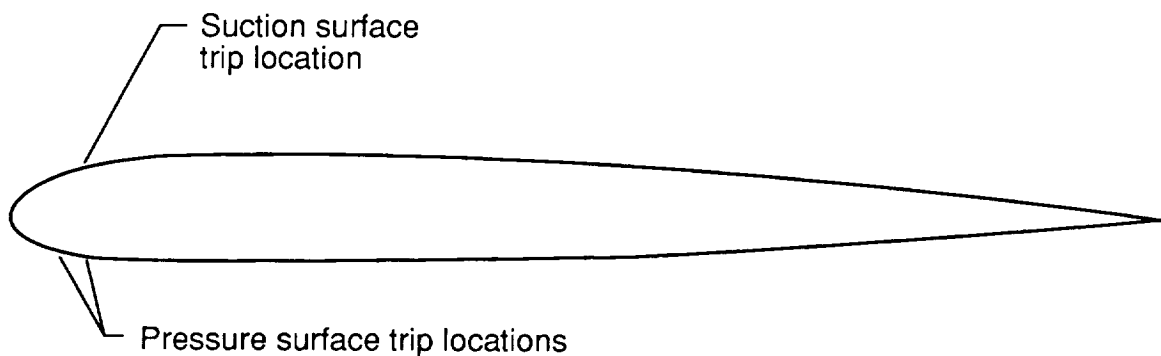


Figure 100. Cross section of Sikorsky rotor blade (ref. 3). Span is 53.3 cm and chord length is 40.6 cm.

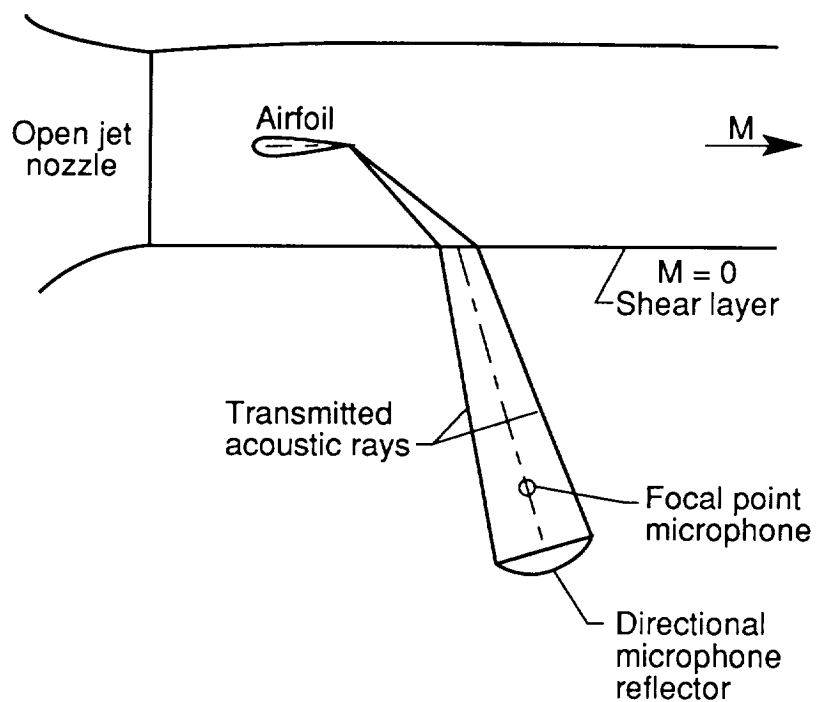


Figure 101. UTRC experimental configuration of reference 3, showing the effect of tunnel flow and shear layer refraction on the directional microphone alignment.

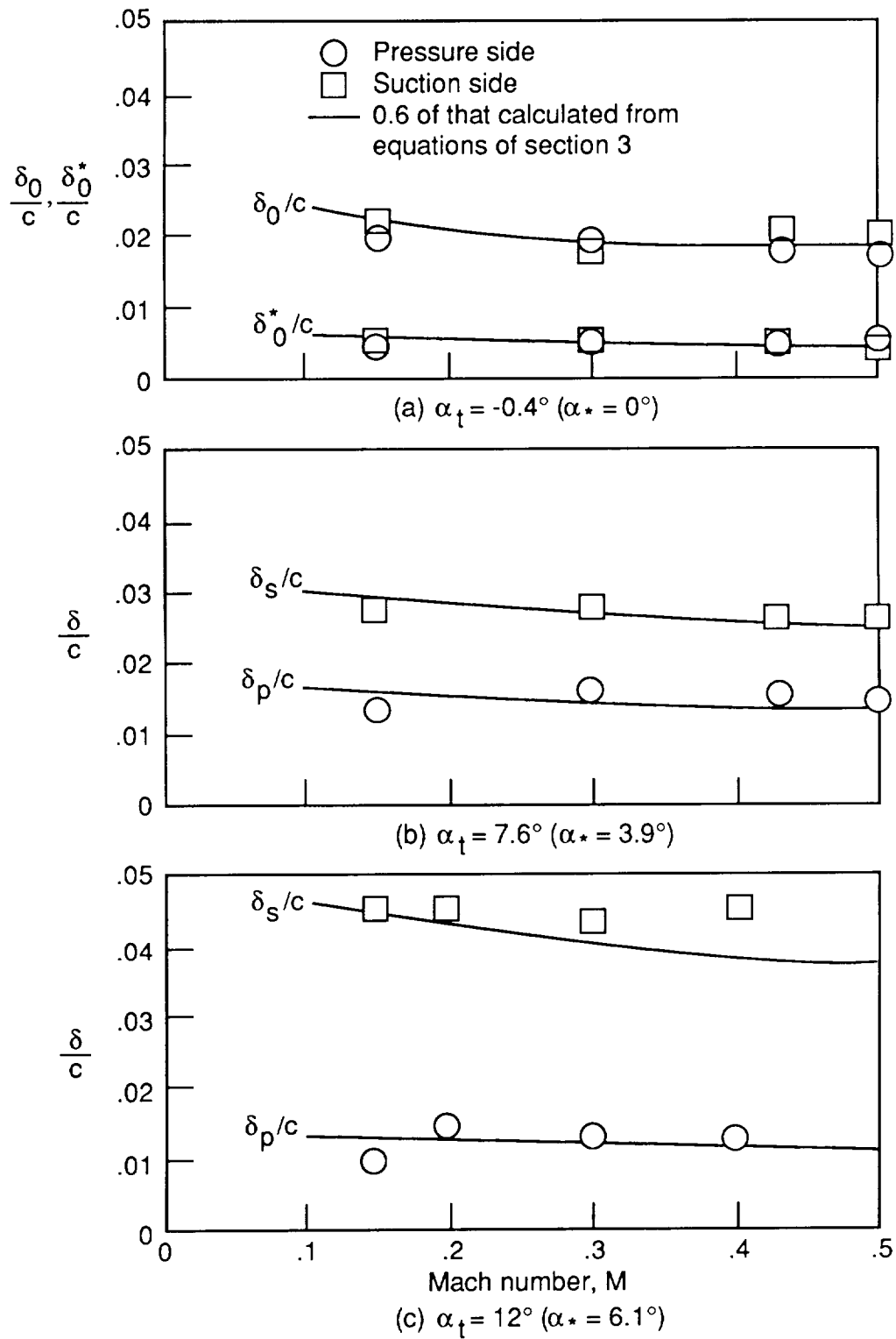


Figure 102. Measured boundary-layer thickness at the TE of the Sikorsky airfoil (ref. 3). Comparison is made with scaling equation results of present paper, multiplied by 0.6 to account for light trip condition.

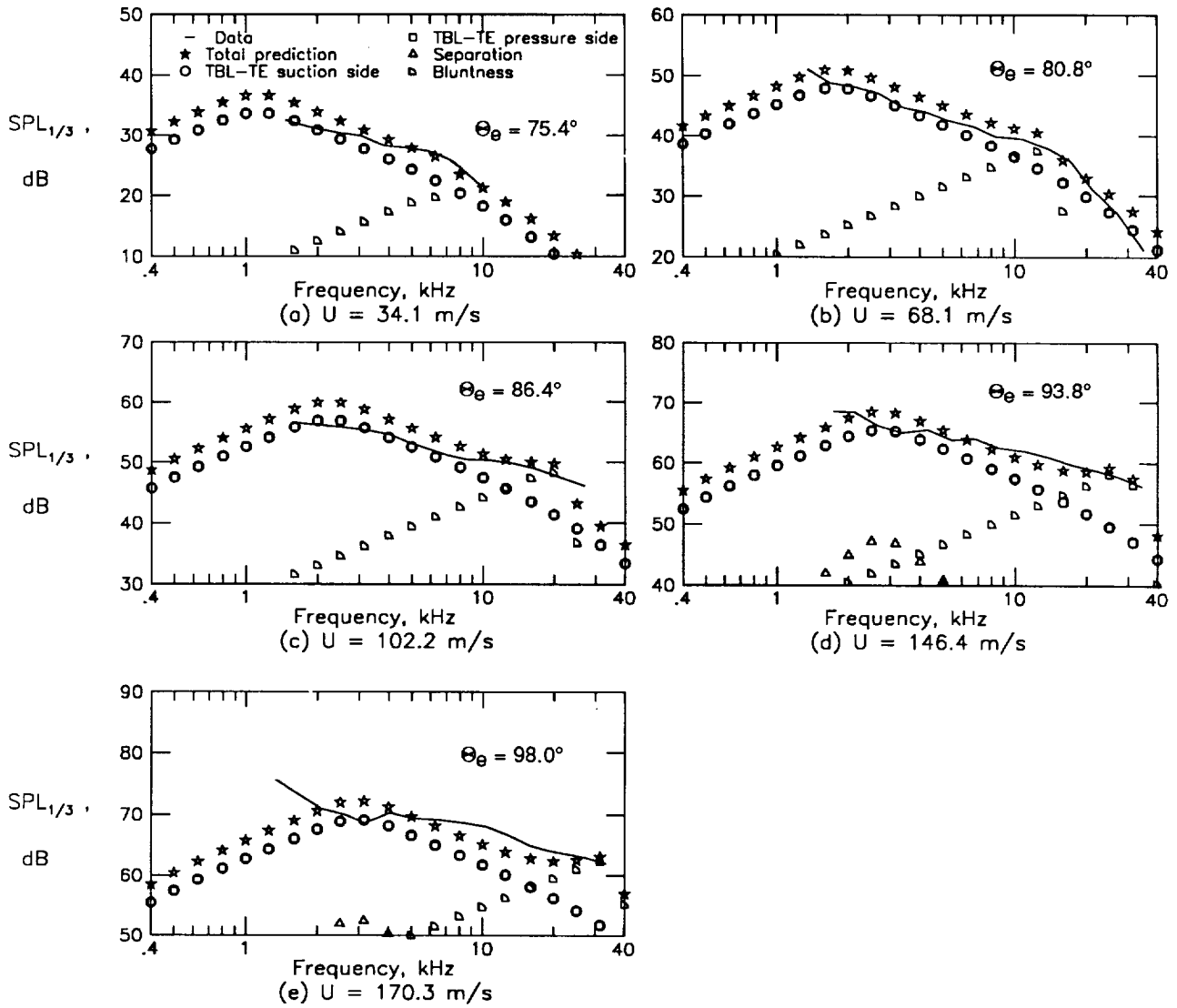


Figure 103. Noise spectra for Sikorsky airfoil at $\alpha_t = -0.4^\circ$ ($\alpha_* = 0^\circ$) from reference 3 compared with prediction of present paper.

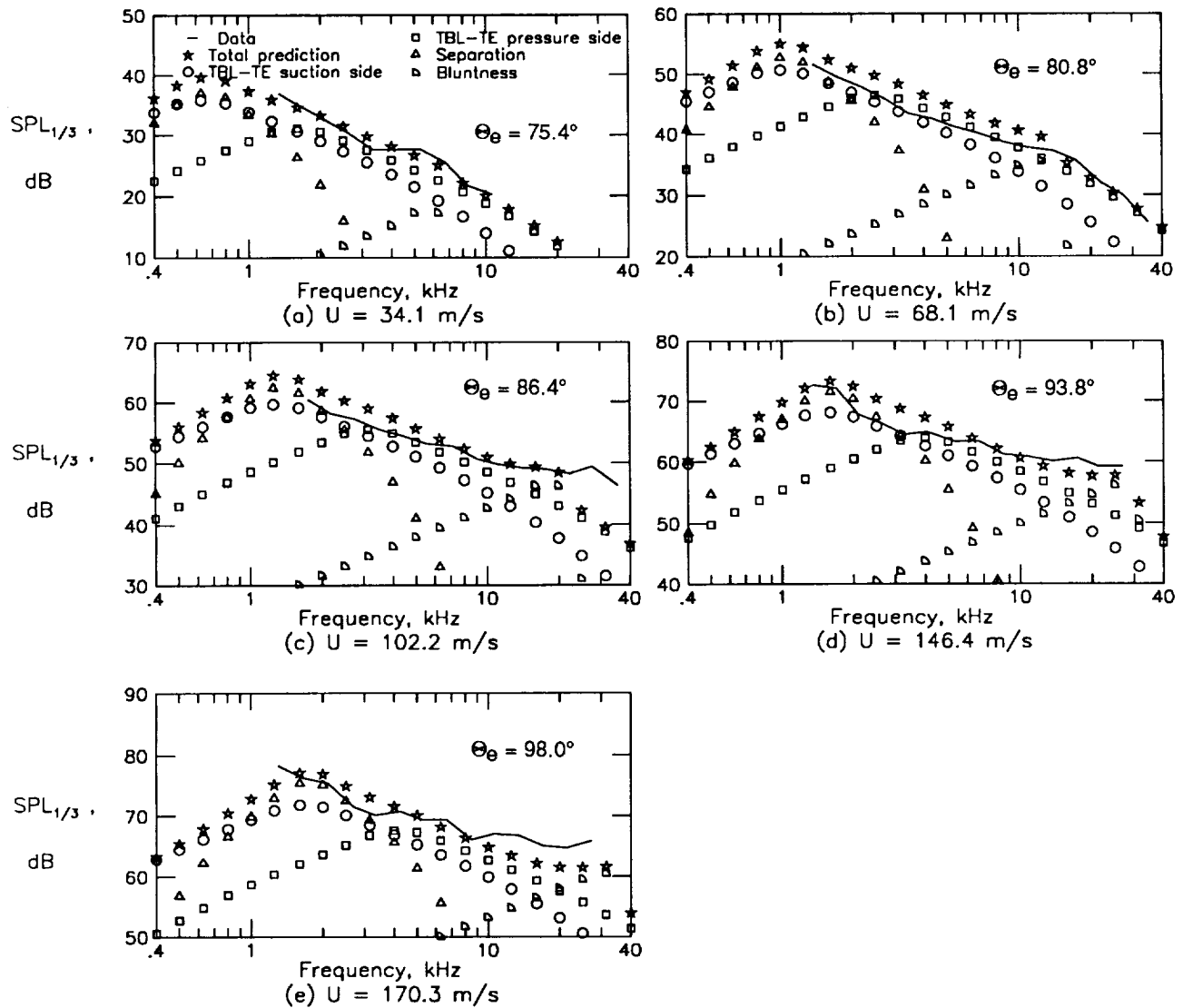


Figure 104. Noise spectra for Sikorsky airfoil at $\alpha_t = 7.6^\circ$ ($\alpha_* = 3.9^\circ$) from reference 3 compared with prediction of present paper.

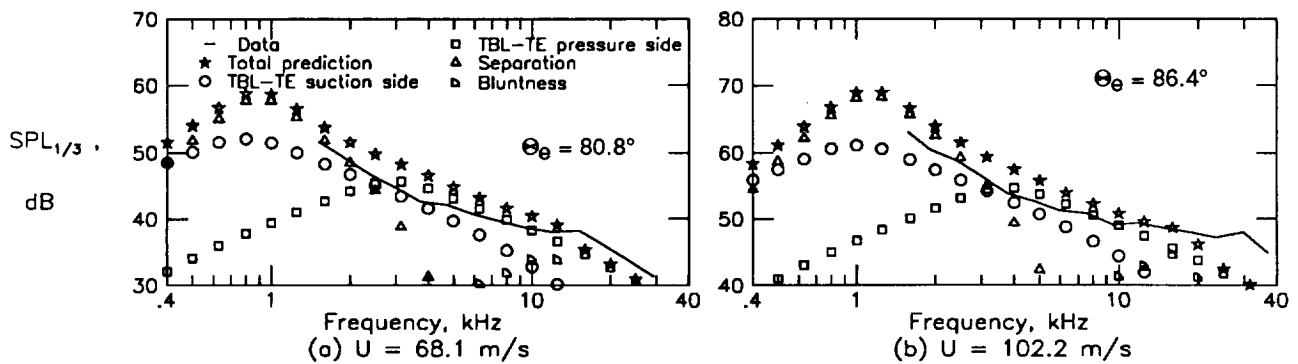


Figure 105. Noise spectra for Sikorsky airfoil at $\alpha_t = 12^\circ$ ($\alpha_* = 6.1^\circ$) from reference 3 compared with prediction of present paper.

considering that the predictions are based empirically on a different airfoil section and that the noise measurement methods were quite different. There does appear to be a mild overprediction of the TBL-TE noise, although not consistently so. The extent of agreement in the spectra where the TE bluntness noise contributes is substantially due to the aforementioned choice of $\Psi = 17^\circ$ (the previously used $\Psi = 14^\circ$ would result in a contribution about 3 dB higher than that shown).

6.2. Study of Schlinker

The tests of Schlinker (ref. 26) were similar in design to that of reference 3, whose measurement configuration is shown in figure 101. The 2D airfoil model, however, was an NACA 0012 section (as in the present study) with a chord length of $c = 22.9$ cm. Again, the aim of the tests was to measure TBL-TE and not LBL-VS noise. However, no BL trip was used at zero angle of attack because no LBL-VS noise was identified (except for the lowest speed tested and those data were not presented). At $\alpha_t = 6^\circ$, the LBL-VS noise was pervasive so a trip was placed on the pressure side at 30 percent of the chord to eliminate the LBL-VS noise.

The TE noise spectra at various tunnel velocities are shown in figures 106 and 107 for the airfoil at $\alpha_t = 0^\circ$ and 6° , respectively. The data were processed so that the levels shown are for the full airfoil span of $L = 53.3$ cm and an omnidirectional observer positioned at $r_e = 2.81$ m and $\Theta_e = 90^\circ$. For this airfoil, the corrected angles of attack, using equation (1) with $c = 22.9$ cm and $H = 79$ cm, are $\alpha_* = 0^\circ$ and 3.9° for $\alpha_t = 0^\circ$ and 6° , respectively. The predictions shown in figure 106 for zero angle of attack are for TBL-TE, LBL-VS, and TE bluntness noise. The values of δ_0 and δ_0^* used in the predictions were obtained from equations (5) and (6), for an untripped BL airfoil. The predictions shown in figure 107 for $\alpha_t = 6^\circ$ are for only TBL-TE, separation, and TE bluntness noise, since the LBL-VS noise was eliminated by the pressure side tripping. The required values of δ_s^* were calculated from equation (14), for an untripped BL. However, the values of δ_p^* were determined from equations (3) and (9), for a tripped BL and then multiplying the result by 0.6 (to reflect the "light" trip condition as discussed for the Schlinker and Amiet study). For the calculations for TE bluntness noise, there was no guidance from the paper for the specification of h and Ψ . A reasonable TE thickness of $h = 0.63$ mm was assumed and the TE flow angle parameter was set at $\Psi = 23^\circ$, because it gave good agreement for the high frequencies in figures 106 and 107. The overall agreement between the total predictions and the data appears good.

6.3. Study of Fink, Schlinker, and Amiet

Fink, Schlinker, and Amiet (ref. 27) conducted tests in the UTRC tunnel to study LBL-VS noise from three airfoil geometries. The untripped BL airfoil models had an NACA 0012 planform and their geometries are shown in figure 108. The first had a constant-chord length of 11.4 cm across the span while the other two were spanwise tapered, having linearly varying chord lengths along the span. Of the tapered airfoils, the first had a taper ratio of 2 to 1 with chord length varying from 15.2 cm down to 7.6 cm. The other airfoil had a taper ratio of 4 to 1 with chord length varying from 18.3 cm to 4.6 cm. The span was $L = 79$ cm for all cases. Because the levels of the LBL-VS noise were sufficiently intense compared with the tunnel background noise, a directional microphone system was not used to measure the noise. Instead, far-field spectra were obtained with individual microphones placed on an arc of 2.25-m radius about the midspan of the models. The noise data from reference 27 which are presented in the present report are all from a microphone for which $\Theta_e \approx 90^\circ$.

Reference 27 presented most noise data in narrow-band form at various bandwidths to allow examination of the tonal character of the LBL-VS noise. To compare these data with the predictions of the present paper, the narrow-band data were digitized and converted to 1/3-octave presentations. As a check on this procedure, as well as a check on the consistency of the data presented in reference 27, overall sound pressure levels (OASPL) were computed from the digitized data and compared with overall levels reported from direct measurement. The values generally agreed to within 1.0 dB.

For the constant-chord airfoil at $\alpha_t = 4^\circ$, 1/3-octave spectra are shown in figure 109 for various tunnel velocities between $U = 37$ m/s and 116 m/s. The number of spectral bands, as well as the frequency range, presented for the spectra varies for the different speeds. This variation is due to the different narrow-band analysis ranges used in reference 27, as all available data were used to generate the 1/3-octave band spectra. For $U = 37$ m/s, figure 109(a), the spectrum is flat at the lower frequencies but is peaked between 1 and 3 kHz. From the narrow-band presentation of reference 27 (fig. 22), one finds that the flat portion is dominated by broadband noise, which is characteristic of tunnel background contamination. It is noted again that these spectra are single microphone results from which the background noise has not been subtracted. The spectral peak region is due to the presence of three quasi-tones, representing the LBL-VS noise portion. At $U = 52, 64,$

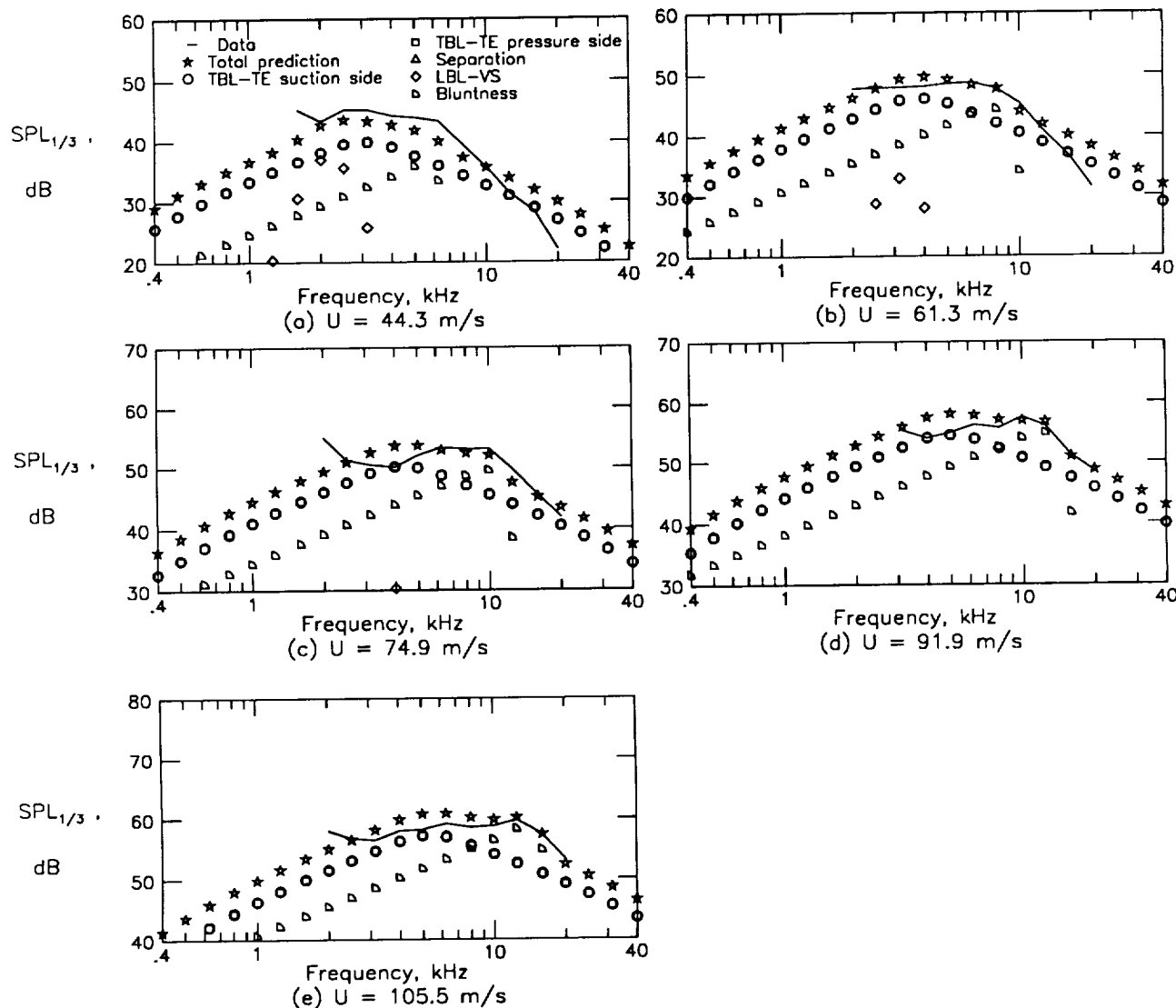


Figure 106. Noise spectra for NACA 0012 airfoil at $\alpha_t = 0^\circ$ ($\alpha_* = 0^\circ$) from reference 26 compared with prediction of present paper.

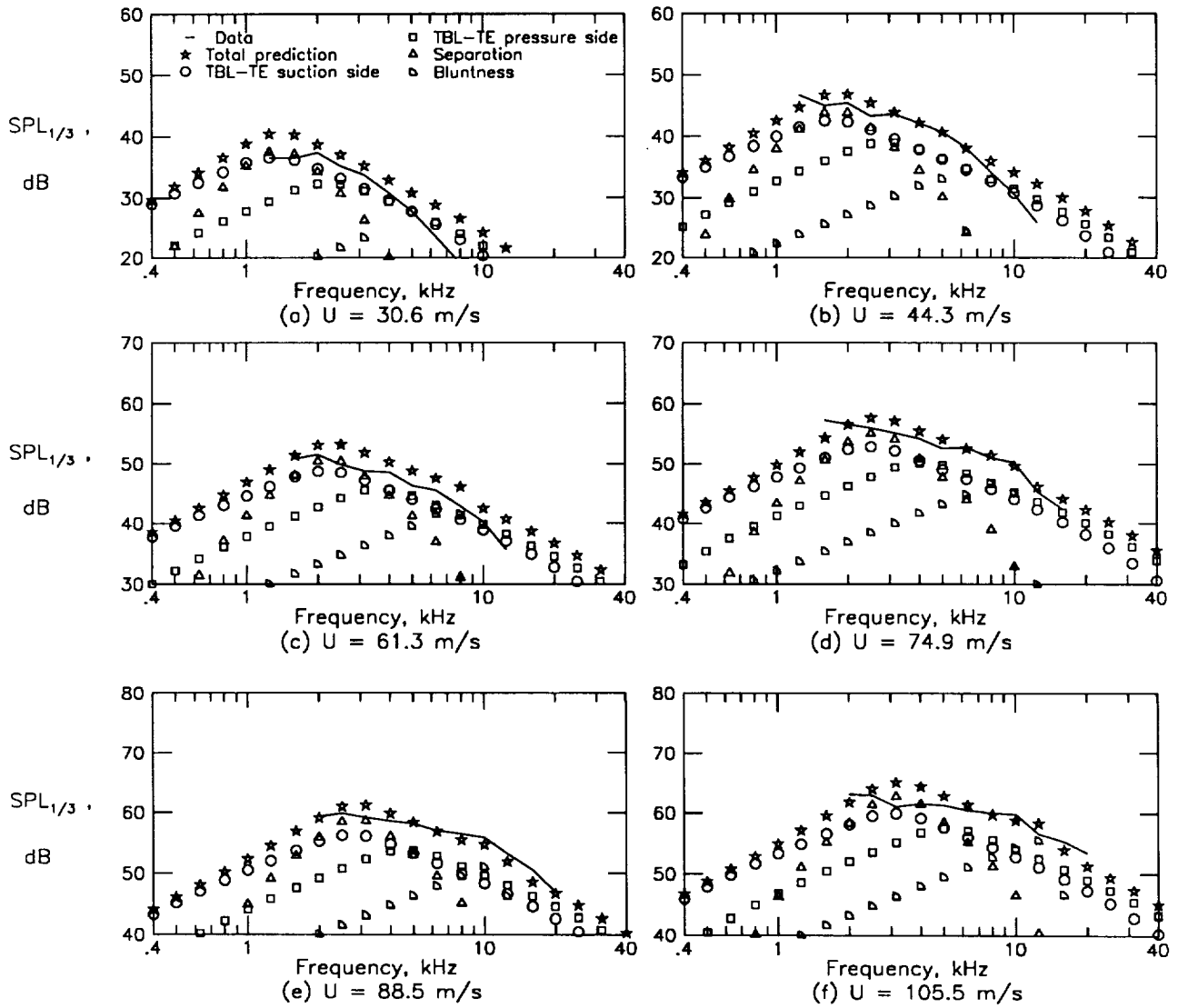


Figure 107. Noise spectra for NACA 0012 airfoil at $\alpha_t = 6^\circ$ ($\alpha_* = 3.9^\circ$) from reference 26 compared with prediction of present paper.

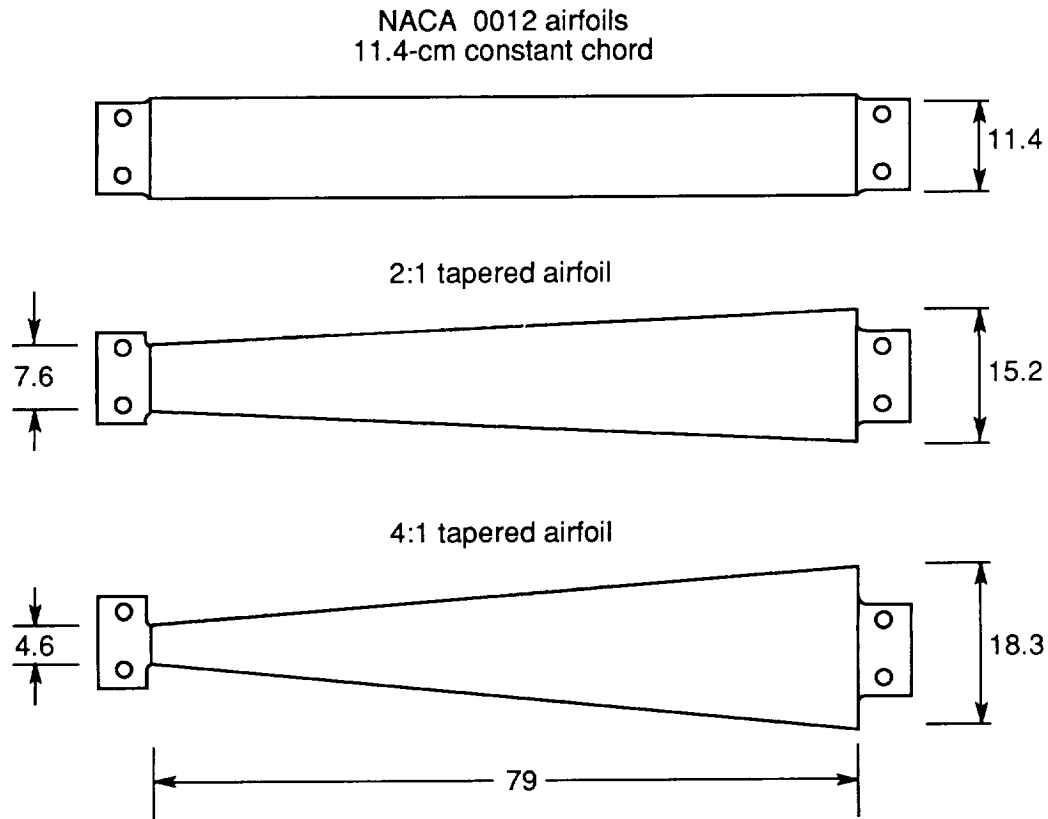


Figure 108. Airfoil models of reference 27. All dimensions are in centimeters.

and 79 m/s, in figures 109(b), 109(c), and 109(d), the spectra are very peaked because of the dominating contributions from large numbers (10 to 15) of LBL-VS quasi-tones. At $U = 98$ and 116 m/s, in figures 109(e) and 109(f), the spectra are less peaked because of a somewhat decreased number of quasi-tones which become submerged within broadband background noise (which itself increases with speed).

The strong velocity dependence of the noise is seen clearly in figure 110 (from fig. 25 of ref. 27) where the OASPL is plotted as a function of velocity. The overall levels were directly measured, for the noise between 200 Hz and 20 kHz, rather than determined by integrating measured spectra. The levels rise and then stabilize with increases in velocity. The resumed increase in levels at the highest speeds (approximately 100 m/s) is where the background noise appears to become dominant.

Compared with the data in figures 109 and 110 are noise predictions of LBL-VS, TBL-TE, and separation noise. No consideration was given to bluntness noise because of the lack of information about the TE geometry as well as the fact that LBL-VS noise dominates the predictions where comparative data are available. For the BL thickness determina-

tions, the equations of section 3 for untripped boundary layers were used. The corrected angles of attack were calculated from equation (1), with $c = 11.4$ cm and $H = 53$ cm, which rendered $\alpha_* = 0^\circ$ and 1.9° for $\alpha_t = 0^\circ$ and 4° , respectively. These were employed with the prediction equations for an observer at $r_e = 2.25$ m, $\Theta_e = 90^\circ$, and $\Phi_e = 90^\circ$. The predictions in figure 109 give good comparisons, except that the peak frequencies are lower than predicted. The previously described background noise contributions explain the differences for the lowest and highest speeds. For the predictions of OASPL in figure 110, the spectra for LBL-VS, TBL-TE, and separation noise were summed. Predictions are presented for not only $\alpha_t = 4^\circ$ but also $\alpha_t = 0^\circ$, 2° , and 6° . This is done to show the great sensitivity of the predictions to airfoil angle of attack. It is seen that the data would most agree with predictions for about $\alpha_t \approx 5^\circ$ rather than $\alpha_t = 4^\circ$. This could be interpreted to mean that the agreement is on the order of possible experimental bias error in angle definition.

The tapered-chord airfoils were used in reference 27 to provide a continuous variation in expected vortex tone frequency to compare with an analogous rotating constant-chord blade. The tone variation

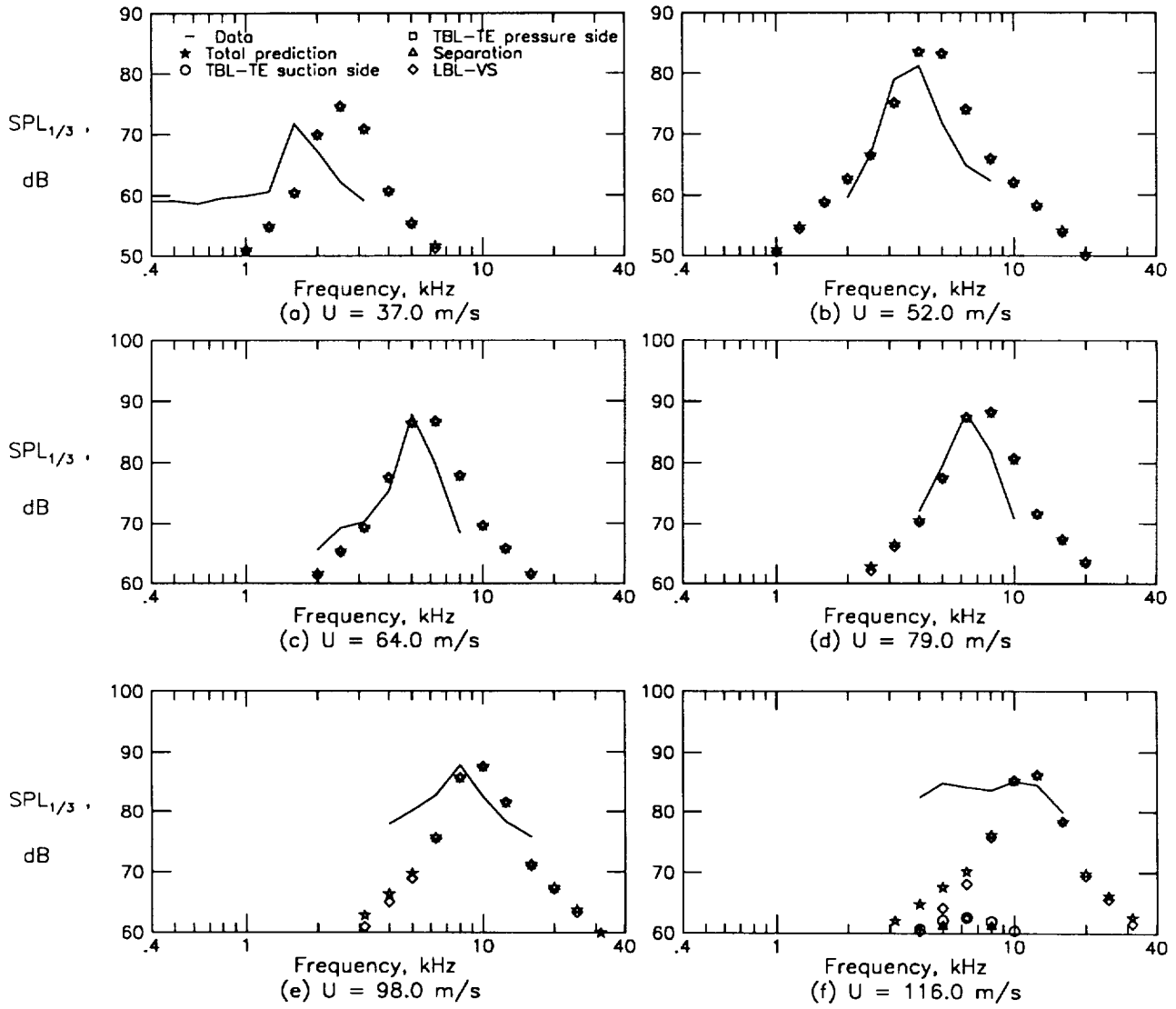


Figure 109. Noise spectra for constant-chord airfoil at $\alpha_t = 4^\circ$ ($\alpha_* = 1.9^\circ$) from reference 27 compared with prediction of present paper.

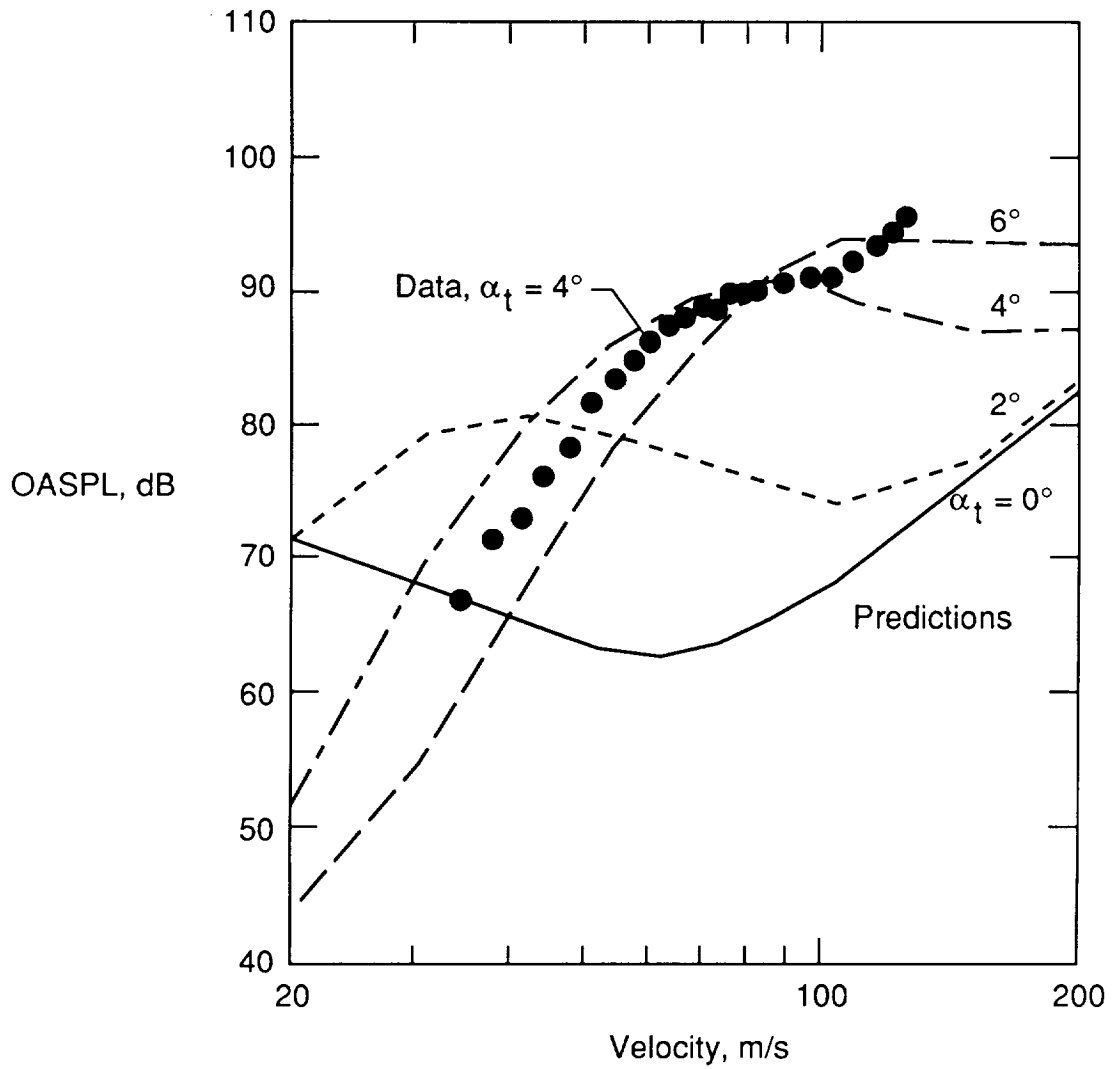


Figure 110. Overall sound pressure level versus velocity for constant-chord airfoil from reference 27 compared with predictions of present paper.

was found not to be continuous; however the tapered models did produce spectra containing a large number of peaks spread over a somewhat wider frequency range than those for the constant-chord airfoil at about the same velocities. In figure 111, 1/3-octave spectra are shown for the 2-to-1 taper airfoil at $\alpha_t = 4^\circ$ for tunnel velocities between $U = 27$ and 107 m/s. The data are similar to those for the constant-chord model, except that the peaks are generally less well defined. In figure 112, corresponding OASPL variations with tunnel velocity are shown for $\alpha_t = 4^\circ$. Also in this figure, OASPL is shown for a range of velocities where $\alpha_t = 0^\circ$.

The predictions shown in figures 111 and 112 were obtained by dividing the models into 10 segments of constant chord (where actual chord length for each segment varied according to the blade taper), then making predictions for each segment, and summing on a pressure-squared basis the contributions of each. Angle-of-attack corrections for each segment were made by calculating the correction based on the mean chord (11.4 cm) across the span. This correction was then applied to the angle of attack for each of the blade segments. The corrected angles, therefore,

were the same as for the constant-chord model, that is, $\alpha_* = 0^\circ$ and 1.9° for $\alpha_t = 0^\circ$ and 4° , respectively. The comparisons between predictions and data for the 2-to-1 taper airfoil appear about as good as those for constant-chord comparisons. It appears that the predictions for OASPL at $\alpha_t = 4^\circ$ (fig. 112) would best agree if $\alpha_t \approx 3.5^\circ$ had been used rather than 4° . This again indicates that agreement is on the order of possible experimental angle definition error. The OASPL comparisons for zero angle of attack show the predicted trends to be quite good but the levels to be overpredicted by 5 to 7 dB.

In figures 113 and 114 are the data and prediction comparisons for the 4-to-1 taper model at $\alpha_t = 0^\circ$. The predictions are not as good as for the constant-chord and the less tapered model, although the data still fall within a predictive range of $\alpha_t = 2^\circ$ to 3° . One should bear in mind that the flow behavior in the vicinity of the tapered models would be expected to deviate from the idealized 2D behavior assumed to be occurring over the small spanwise segments employed for the predictions. This makes it difficult to assess the meaning of the comparison deviations for the tapered models.

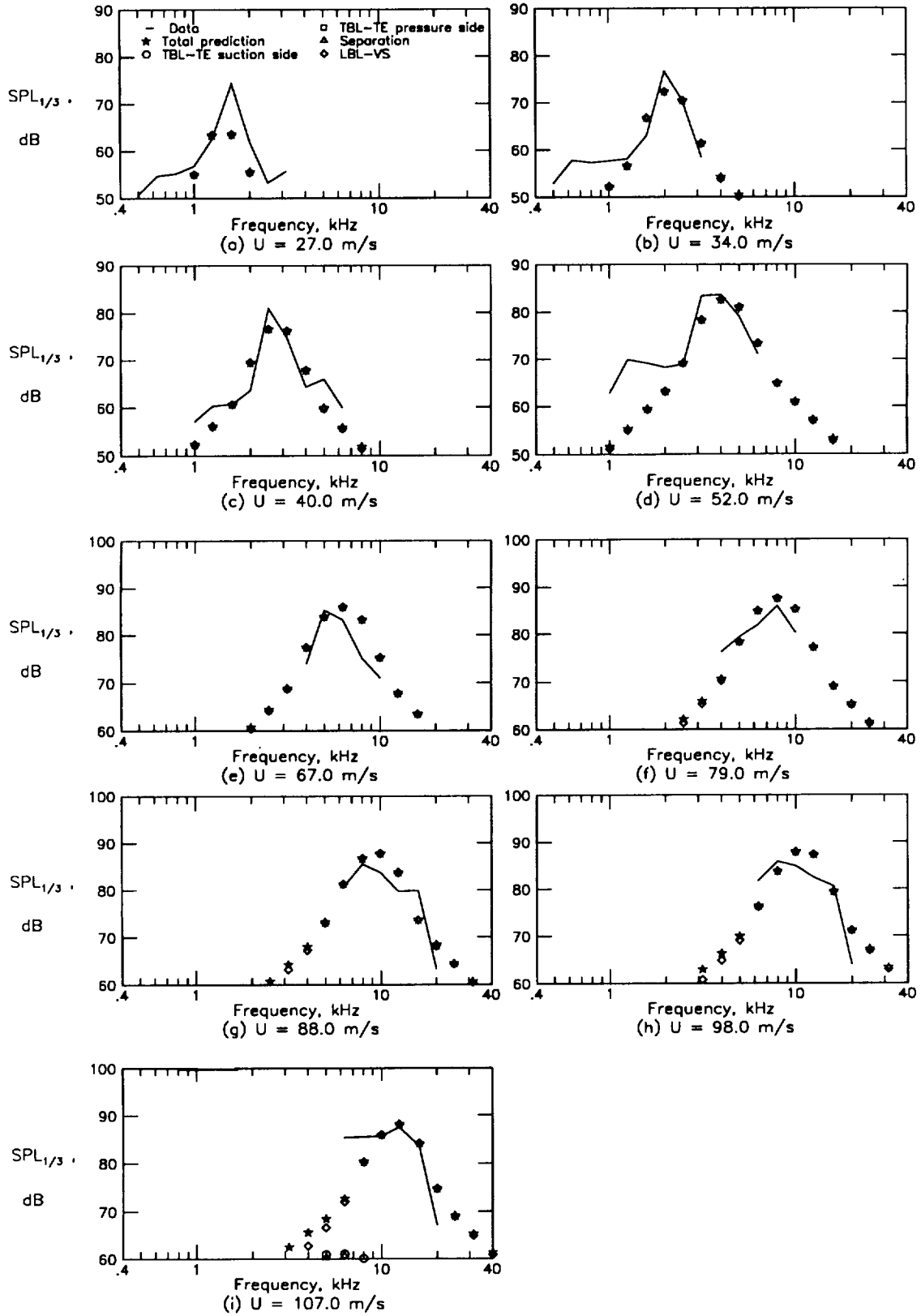


Figure 111. Noise spectra for 2-to-1 tapered-chord airfoil at $\alpha_t = 4^\circ$ ($\alpha_* = 1.9^\circ$) from reference 27 compared with prediction of present paper.

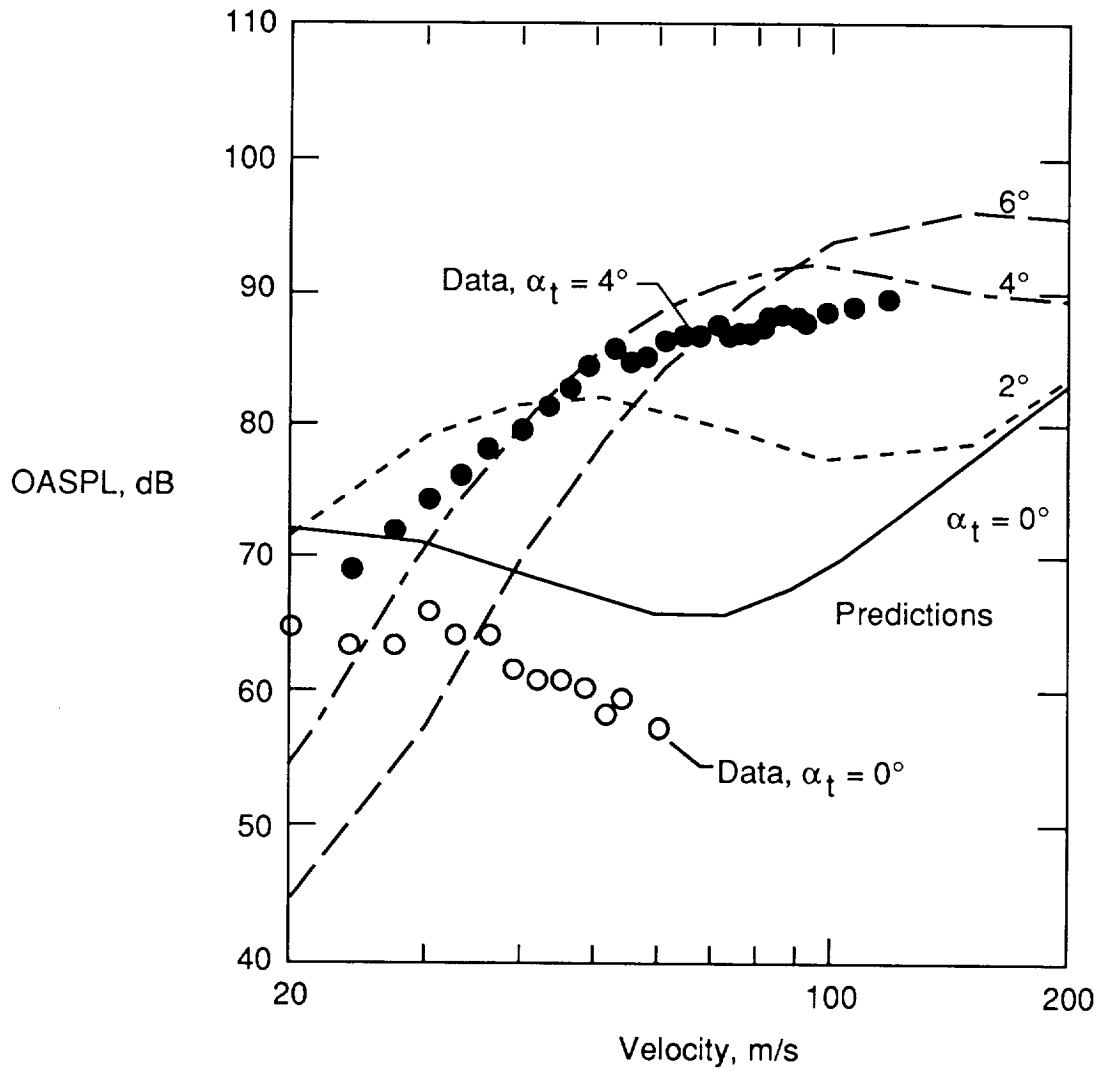


Figure 112. Overall sound pressure level versus velocity for 2-to-1 tapered-chord airfoil from reference 27 compared with predictions of present paper.

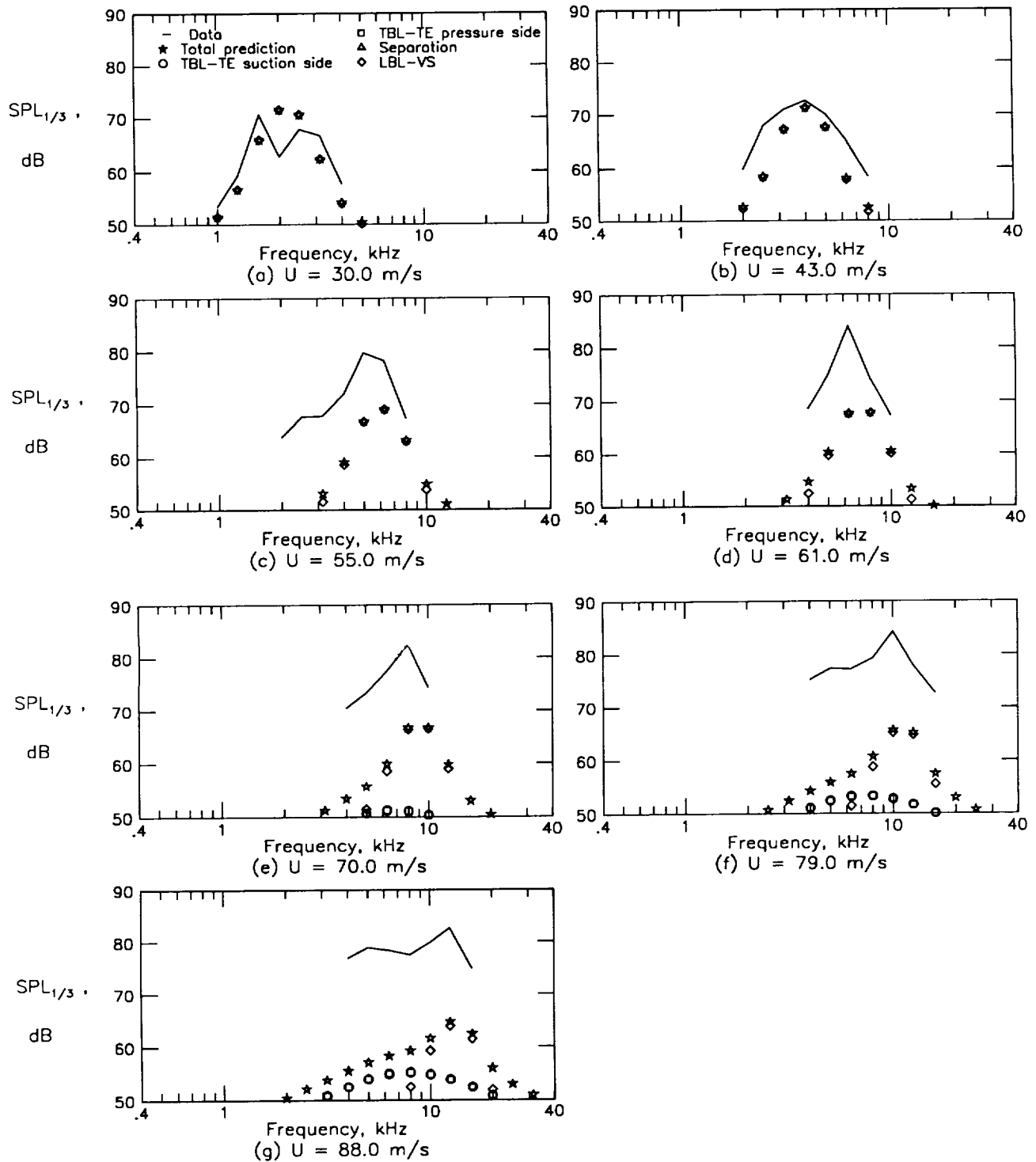


Figure 113. Noise spectra for 4-to-1 tapered-chord airfoil at $\alpha_i = 0^\circ$ ($\alpha_* = 0^\circ$) from reference 27 compared with prediction of present paper.

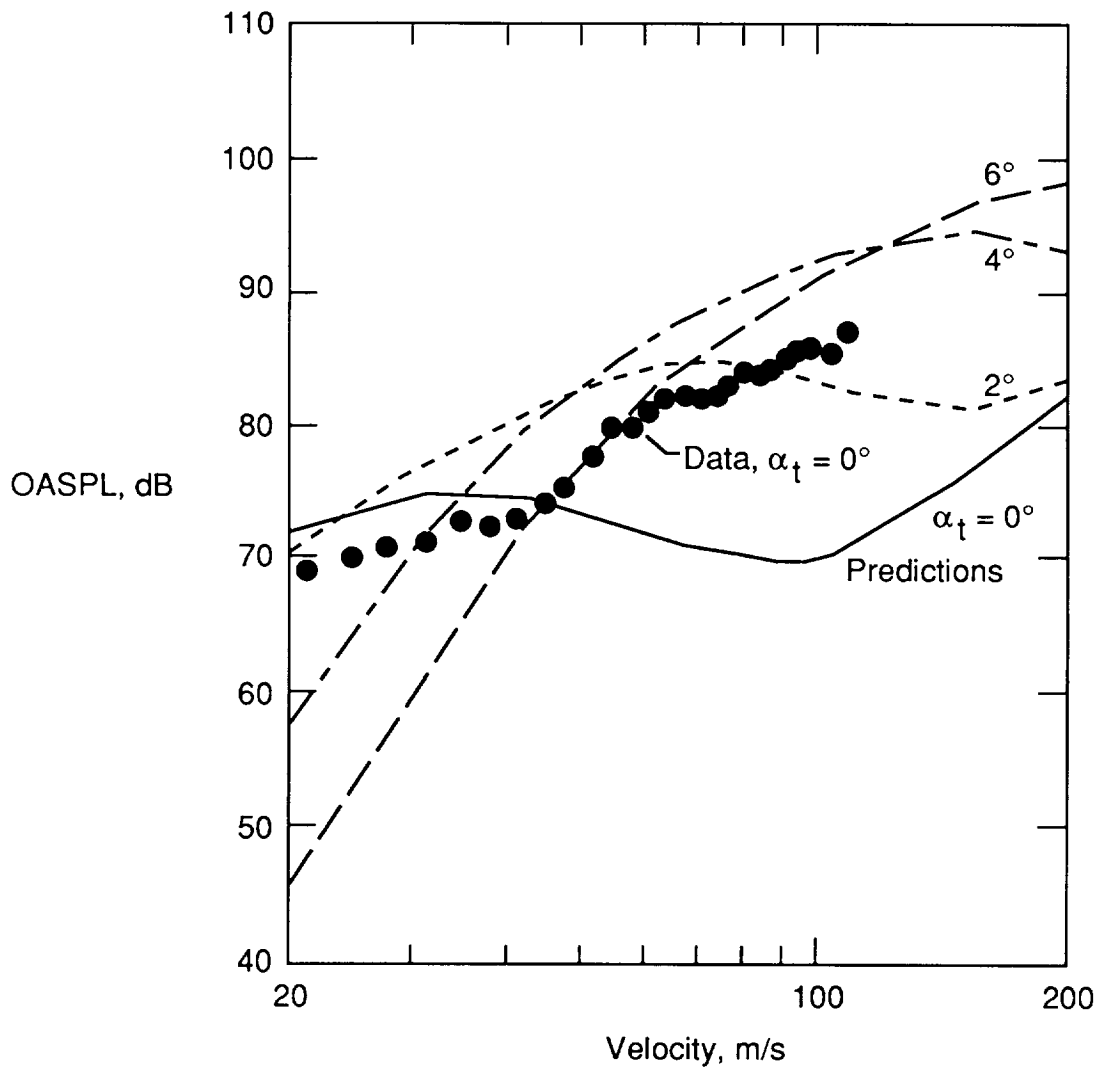


Figure 114. Overall sound pressure level versus velocity for 4-to-1 tapered-chord airfoil from reference 27 compared with predictions of present paper.

7. Conclusions

This paper documents the development of an overall prediction method for airfoil self-noise. The approach is semiempirical and is based on previous theoretical studies and data from a series of aerodynamic and acoustic tests of isolated airfoil sections. The acoustic data processing employed a correlation editing procedure to obtain self-noise spectra uncontaminated by extraneous noise. Five self-noise mechanisms, each related to specific boundary-layer phenomena, are identified in the data and modeled. For each mechanism, the data are first normalized by fundamental techniques using scaled aerodynamic parameters. The spectral shape, level, and frequencies are then examined and modeled for dependences on parameters such as Reynolds number, Mach number, and geometry.

The modeling accuracy of the resulting self-noise prediction methods is established by comparing predictions with the complete data base. The methods are shown to have general applicability by comparing predictions with airfoil self-noise data reported in the literature from three studies. A successful application of the methods is reported for a large-scale-model helicopter rotor broadband noise test.

Conclusions can be drawn regarding the specific self-noise mechanisms. For the turbulent-boundary-layer-trailing-edge noise and separation noise sources, an accurate and generally applicable predictive capability is demonstrated, especially for the important conditions of high Reynolds number and low to moderate angle of attack. The mechanism which can dominate the spectra for low Reynolds number, laminar-boundary-layer-vortex-shedding noise, is also demonstrated to have good predictive capability. For this quasi-tonal noise

mechanism, there are some issues, not fully addressed herein, about how to apply the formulations in the most appropriate way to different airfoil geometries. The tip vortex formation noise source appears to be well predicted, although its relative lack of importance compared with the other self-noise sources prevents a full assessment of accuracy. The trailing-edge-bluntness-vortex-shedding noise source is shown to be very important and predictable by the method developed. For this source, there is an associated "flow angle" parameter which is found to be constant for any given trailing-edge geometry, but is difficult to determine a priori. However, for application of the bluntness noise prediction method, reasonable estimates for this parameter can be made based on the examples in this report.

The unique prediction capability presented should prove useful for the determination of broadband noise for helicopter rotors, wind turbines, airframe noise, and other cases where airfoil shapes encounter low-to moderate-speed flow. For modern propeller designs, the present equations should be applied with some caution because the high-speed, high-loading, and skewed-flow conditions existing about propeller blades do not match the low- to moderate-speed and generally 2D flow conditions of the present data base. The computer codes given herein can be readily incorporated into existing or future noise prediction codes. The documentation provided in this report should provide the means to evaluate where and how any needed future refinements can be made in the prediction codes for particular applications.

NASA Langley Research Center
Hampton, VA 23665-5225
April 19, 1989

Appendix A

Data Processing and Spectral Determination

In section 4, the special processing approach used to determine the self-noise spectra for the 2D airfoil models was summarized. Details are given here.

A.1. Data Acquisition and Initial Processing

Signals from the microphones shown in figure 4 were recorded during the test on a 14-channel FM analog tape recorder, operated to provide a flat frequency response up to 40 kHz. Individual amplifiers were used to optimize signal-to-noise ratio for each microphone channel, and pure-tone and white-noise insertions were used to calibrate amplitude and phase response, respectively. These calibrations and signal-conditioning techniques were the same as in reference 2, where additional details are given. The data were reduced from tape on a spectrum analyzer interfaced with a minicomputer. Pairs of microphones were used to obtain 1024-point cross-correlations at an analysis range of ± 4.167 milliseconds.

A.2. Correlation Editing

The correlation records are modified to eliminate contributions from extraneous noise sources prior to taking the Fourier transforms to obtain the spectra. The first step is to remove, to the extent possible, the noise from the test hardware by subtracting the correlation $R_{45}(\tau)$ without the airfoil in place (the background noise) from $R_{45}(\tau)$ with the airfoil in place. (See fig. 9.) The resulting record should then be comprised of correlation peaks from the desired TE noise, LE noise, and other extraneous noise related to interaction between the model and test rig not accounted for in the subtraction. The TE and LE noise peaks in the cross-correlation are assumed to represent the autocorrelation of the TE and LE noise, respectively.

To eliminate the LE contribution, the correlation record on the right-hand-side (RHS) of τ_{TE} is discarded and replaced by the mirror image of the left-hand-side (LHS). However, for this folding process, it was found that it is important to preserve the basic shape of the TE peak to more accurately represent the spectra at higher frequencies. Because this is a digital correlation, made up of discrete points which are $\Delta\tau$ apart, it is likely that the true TE noise peak falls somewhere between two discrete values of τ . Folding about a discrete point instead of the actual effective peak center would introduce error by distorting the peak shape. In figure A1, the discrete points of the TE correlation peak are illustrated to

show how the folding was accomplished. The discrete center is at τ_{TE} , whereas the effective center is to the left. The correlation values at $\tau_{TE} + \Delta\tau$ and $\tau_{TE} + 2\Delta\tau$ must not be changed to avoid modifying the shape near the very peak. The correlation value at $\tau_{TE} + 2\Delta\tau$ is projected to the LHS to intercept a line connecting $\tau_{TE} - 3\Delta\tau$ and $\tau_{TE} - 2\Delta\tau$. This defines the constants \bar{a} and \bar{b} which are shown. These constants then are used to interpolate between the points on the LHS to determine values at the points on the RHS, that is

$$R(\tau_{TE} + N\Delta\tau) = \frac{\bar{b}}{\Delta\tau} R(\tau_{TE} - N\Delta\tau) + \frac{\bar{a}}{\Delta\tau} R(\tau_{TE} - (N+1)\Delta\tau) \quad (A1)$$

for $N > 2$. The entire LHS of the correlation is folded about the effective peak center using this interpolation scheme.

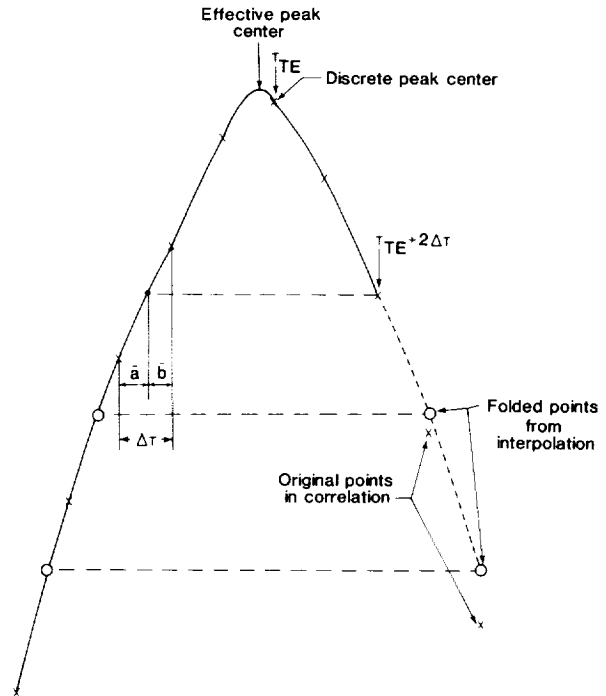


Figure A1. Sample correlation peak.

A.3. Separation of TE and LE Peaks

As indicated in section 4, for some of the correlation data for the three smaller chord lengths, the LE and TE peaks are so close that the LE contribution overlaps and distorts the TE peak shape. For many

such cases a procedure was found to successfully remove the distortion prior to implementing the TE peak folding process described above. This procedure is explained by way of example for the 5.08-cm-chord airfoil shown in the bottom trace of figure A2. The predicted locations of the TE and LE noise peaks in the correlations are indicated and agree well with the peaks in the actual trace. Note the proximity of the two peaks.

The procedure to separate these two peaks involves combining the original $R_{45}(\tau)$ at the bottom of figure A2 with time-shifted versions of itself, so that the peaks are separated by larger time delays. The procedure depends on the implied symmetry of the LE and TE peaks, inherent in the assumption that they represent the autocorrelations for the LE and TE noise, respectively. The first step is to invert $R_{45}(\tau)$ in sign, and reverse it in time, by “flipping” the correlation about τ_{LE} . The result of combining these two curves is seen in the second trace from the bottom of figure A2, denoted $R'_{45}(\tau)$. The two peaks seen here are the original TE noise peak, and an inverted TE noise peak at $2\tau_{LE} - \tau_{TE}$. There is some increase in level and some distortion in the correlation record away from the peaks, as should be expected. The LE noise peak has been removed, but the inverted TE noise peak still affects the original peak at τ_{TE} . To remove the inverted peak, the initial $R_{45}(\tau)$ must be shifted by $2(\tau_{LE} - \tau_{TE})$ and summed with the previous result. This produces the third curve from the bottom in figure A2, denoted $R''_{45}(\tau)$. The TE noise peak has remained at τ_{TE} , while the LE noise peak is now at $3\tau_{LE} - 2\tau_{TE}$. The peaks are now separated in time so that details of each peak can be seen. Note that as the peaks no longer affect one another’s shape, their basic symmetry is evident. This helps to validate the initial assumption that the peaks represent the autocorrelations of TE and LE noise. If the peaks must be further separated, this procedure can be successively repeated, with the results of the next two iterations seen in the top two traces of figure A2, $R'''_{45}(\tau)$ and $R''''_{45}(\tau)$. It should be noted that only the inner portion of the correlation is shown (the correlation was performed for ± 4.067 ms). Because of the data record manipulations, much of the outer portions of the correlations did not overlap and were thus zeroed out.

A.4. Determination of Spectra

Once the correlation records, or their modified forms after the separation processing, are folded about the effective peak center, the resulting TE noise correlations are transformed into spectra of the noise. Because the correlation record lengths are reduced by varying amounts (typically 20 per-

cent) because of the editing described above, the use of fast Fourier transform techniques is not convenient. Instead, regular Fourier transform techniques are used in an approach based on chapter 9 of reference 28. In summary, a data window is applied to the correlation (eq. 9.116, ref. 28) and is used to provide the real and imaginary portions of the spectrum (eqs. 9.167–9.168, ref. 28). The resulting cross-spectra (eqs. 9.172–9.174, ref. 28) are presented in terms of magnitude and phase.

With the cross-spectra produced, amplitude corrections are applied to account for shear layer effects, using the technique of Amiet (ref. 22), as well as self-noise directivity effects, which are described in appendix B. The spectrum for each microphone pair was corrected to an effective position of 90° with respect to the airfoil chord line. The combined effect of both of these corrections tended to be small, with the corrections for many test conditions being less than 1 dB. Since cross-spectra were obtained, the corrections for each of the two microphones involved were averaged to correct the cross-spectral magnitude.

The results obtained from this method are given in figure A3 for the example correlation records of figure A2. Figure A3(a) shows the cross-spectrum obtained from the correlation of the original $R_{45}(\tau)$ record, which is the bottom curve of figure A2, while figure A3(b) shows the cross-spectrum obtained after folding the $R_{45}(\tau)$ record about the TE noise peak. Note that the cross-spectral phase ϕ is a partial indicator of how well the cross-spectrum represents the total TE self-noise. Ideally the phase should vary linearly with frequency, $\phi = 360^\circ f\tau_{TE}$. The breaks seen in this phase line and the corresponding spectral peculiarities indicate regions adversely affected by contamination, which was not removed by the background subtraction and, in the case of figure A3(b), the folding process. The contamination from the LE is seen to primarily affect the cross-spectrum of figure A3(a) below around 4 kHz. Folding the correlation removes most of this, leaving a dip in the spectrum of figure A3(b) at about 1.5 kHz. Figure A3(c) shows the spectrum for the third curve from the bottom in figure A2, $R''_{45}(\tau)$, which is the modified correlation after two manipulations have separated the TE and LE noise peaks. The phase difficulty and spectral dip at about 1.5 kHz in figure A3(b) are eliminated in figure A3(c). Figure A3(d) shows the spectrum for the top curve of figure A2, $R''''_{45}(\tau)$, which is for four manipulations. This spectrum is similar to that of figure A3(c) except for some apparent increase in contamination at the low- and high-frequency ends. For the airfoil presented here, a choice was made to use the spectrum of figure A3(c), based on two

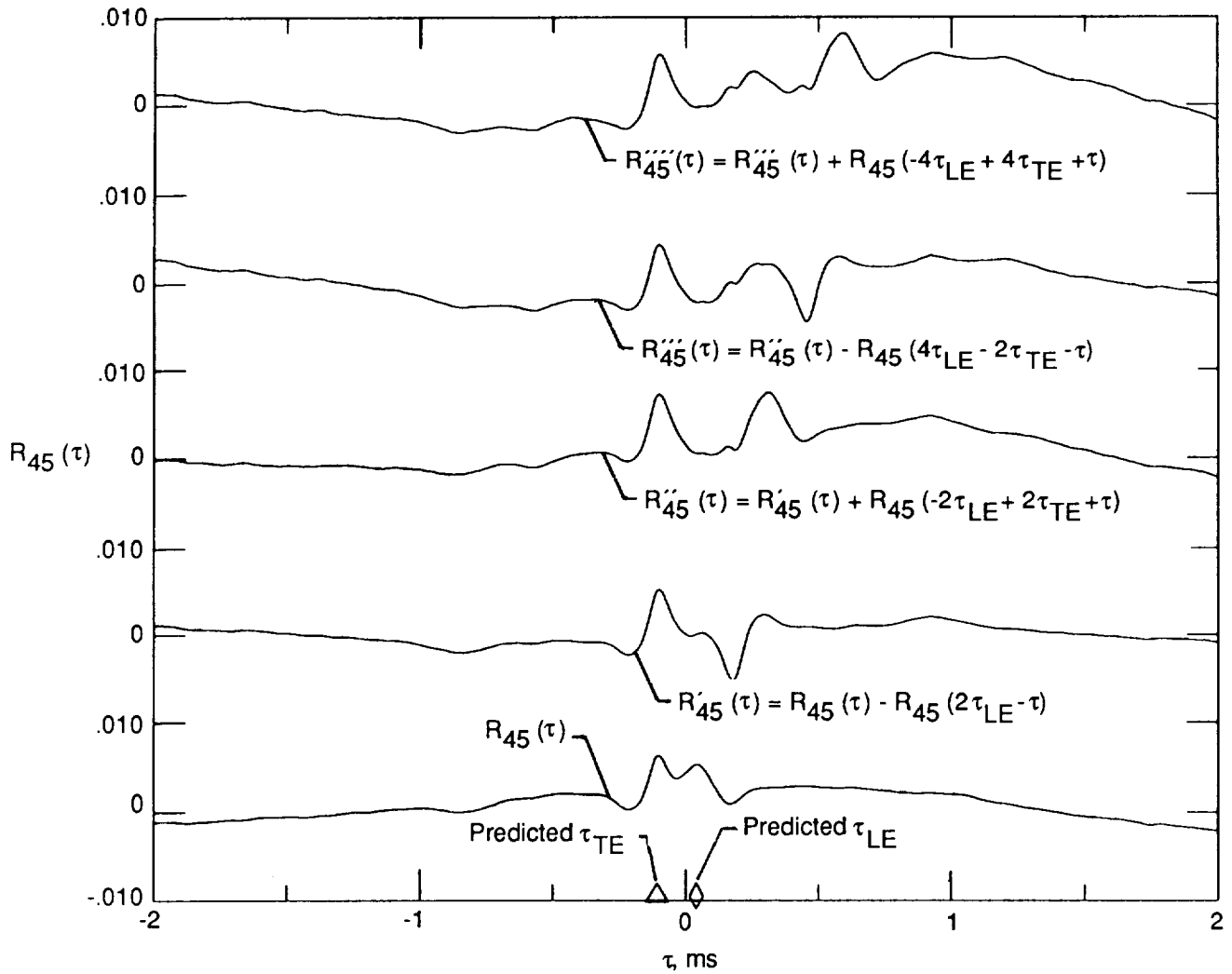
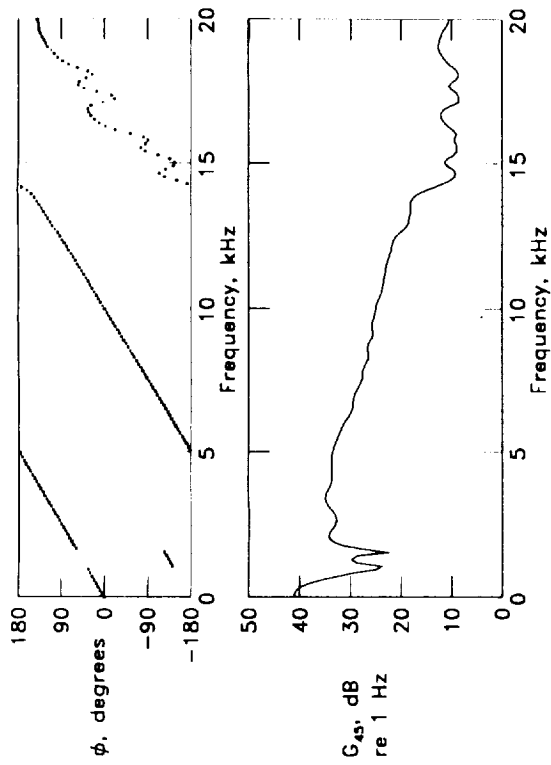
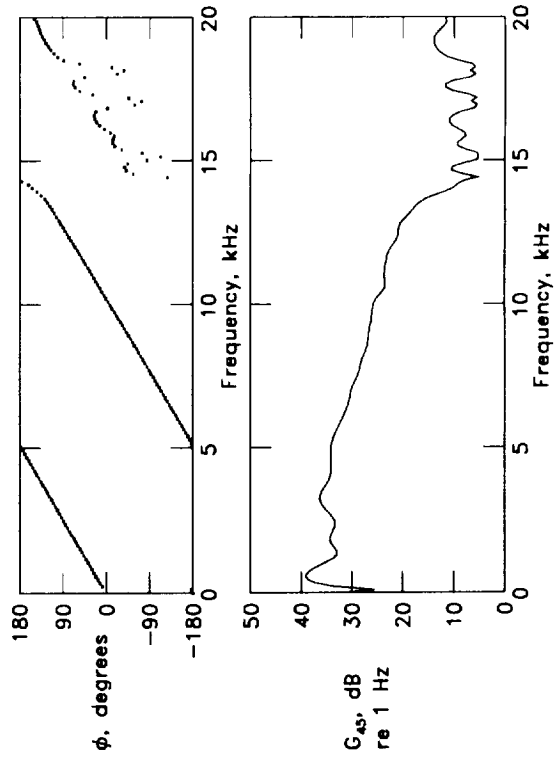


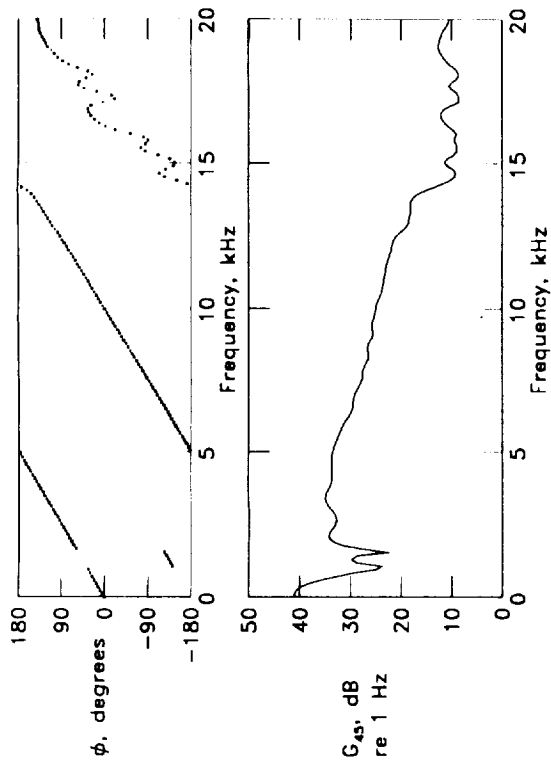
Figure A2. Separation of TE and LE peaks in a cross-correlation. Example is cross-correlation between microphones M4 and M5 for the 5.08-cm-chord airfoil with tripped BL. $\alpha_* = 0^\circ$, $U = 71.3$ m/s.



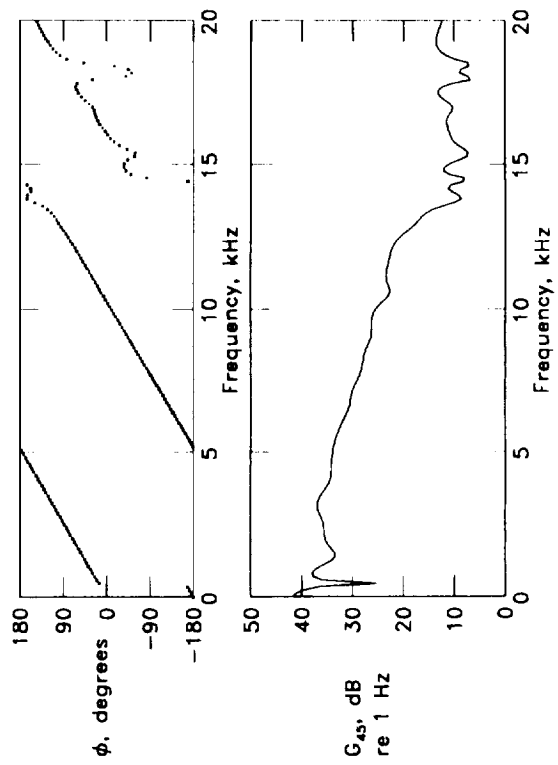
(a) Transform of $R_{45}(\tau)$ with no folding.



(b) Transform of $R_{45}(\tau)$ with folding about TE peak.



(c) Transform of $R_{45}''(\tau)$ with folding about TE peak.



(d) Transform of $R_{45}'''(\tau)$ with folding about TE peak.

Figure A3. Cross-spectra obtained by transforming correlations of figure A2.

separation manipulations, to represent the self-noise. The lower and upper limits to which the spectrum is believed to be accurate are from about 0.8 to 13 kHz. For the other airfoils of this paper, similar evaluations of the limits were made and the spectra are cut off beyond these limits in their presentation, as indicated in section 4.

To increase confidence, all the 2D airfoil spectra presented in figures 11 to 74 were found by averaging independently determined spectra from two microphone pairs. After the shear layer and directivity corrections were applied, the spectra from the two microphone pairs generally agreed to within 1 dB. In tables 1 and 2, the data processing and manipulations, and whether the correlations were folded or not prior to taking the spectra, are specified for each test case. It is seen that for the three larger airfoils, no correlation manipulations were needed to separate the LE and TE correlation peaks. For the three smaller airfoils, performing two separation ma-

nipulations was advantageous for about a quarter of the cases. The table shows that a substantial number of correlations were not folded. For airfoils at sufficiently high angles of attack, low frequencies can dominate the noise. This results in large correlation humps, rather than the relatively sharp peaks which are needed in the folding process. For these cases, the raw cross-correlations are transformed, with only the background subtraction being performed. Also the correlations were not folded in the presence of strong LBL-VS noise. This noise can dominate all other self-noise sources, as well as the LE noise contamination, negating the need for the correlation editing. This correlation editing would have proved difficult, in any case, since vortex shedding produces noise at small bands of frequencies, appearing as damped sinusoidals in the correlation, which tended to mask other peaks. The effect of folding the correlation in such cases was examined, however, and found to have little effect on the spectra.

Appendix B

Noise Directivity

The purpose of this appendix is to define the directivity functions \bar{D}_h and \bar{D}_ℓ , which are employed in the tunnel noise data processing and proposed for use in the prediction equations for the self-noise sources.

B.1. Retarded Coordinates

The retarded coordinate system is explained by first referring to figure B1 where the airfoil is at zero angle of attack to the tunnel flow. If the velocity were zero everywhere, sound from the model which reaches the microphones (M2 is shown) would follow the ray path defined by the measured distance r_m and angle Θ_m . But with the velocity in the free jet equal to U , the ray which reaches the microphones follows first the radiation angle Θ_c until it encounters the shear layer where it is refracted. It emerges at angle Θ_t with an amplitude change and travels to the microphone. The theoretical treatment employed in this study for the angle and amplitude corrections is that due to Amiet (refs. 22 and 23). A convenient reference for the corrected microphone measurements is a retarded coordinate system where the source and the observer are at corrected positions. The angle Θ_e is referenced to a retarded source position and a corrected observer position where the distance between the positions is $r_e = r_m$. As defined, if there were no shear layer present with flow extending to infinity, the center of the wave front emitted from the source would be at the retarded source position when the wave front reaches the corrected observer position. The retarded coordinates are equivalent to the emission time coordinates employed in the literature, for example, see reference 29, for moving sources and stationary observers. Figure B2 shows a source flyover geometry corresponding to the open jet wind tunnel geometry of figure B1. Physical equivalence between the cases is attained by accounting for the Doppler-related frequency shifts due to the relative motion between the source and observer in one instance and no relative motion in the other. There are no Doppler-related amplitude corrections required between the flyover and wind tunnel cases as the effect of the flow on the source definition is already included in the wind tunnel environment.

B.2. Directivity Functions

In figure B3, a 3D retarded coordinate system is defined where the origin is located at the trailing edge of a thin flat plate, representing an airfoil. The flat

plate is in rectilinear motion of velocity U in direction of the negative x_e axis. The observer is stationary. Trailing-edge noise is produced when boundary-layer turbulence and its associated surface pressure pattern convect downstream (with respect to the plate) at a velocity U_c (Mach number M_c) past the trailing edge. If the noise-producing turbulence eddies are sufficiently small and the convection velocities are sufficiently large to produce acoustic wavelengths much shorter than the chord length, the directivity can be shown to be (based on analysis of Schlinker and Amiet, ref. 3)

$$\bar{D}_h(\Theta_e, \Phi_e) \approx \frac{2 \sin^2(\Theta_e/2) \sin^2 \Phi_e}{(1 + M \cos \Theta_e)[1 + (M - M_c) \cos \Theta_e]^2} \quad (\text{B1})$$

where the h subscript indicates the high-frequency (or large-chord) limit for \bar{D} . The overbar on \bar{D}_h indicates that it is normalized by the TE noise radiated in the $\Theta_e = 90^\circ$ and $\Phi_e = 90^\circ$ direction, so $\bar{D}_h(90^\circ, 90^\circ) = 1$. For the flyover plane ($\Phi_e = 90^\circ$), equation (B1) is the same as equation (32a) of reference 3. In reference 3, the equation was compared favorably with measured airfoil TE noise results, for limited M and Θ_e ranges, as well as with previous theoretical results. The directivity expression used in reference 2 was found to give virtually identical results for low Mach numbers.

Although developed for when the velocity U is parallel to the plate along the x_e axis, equation (B1) can be applied when the plate or airfoil is at an angle of attack α to the flow. In application (refer to fig. B3), one should define the angles with respect to a coordinate system that is fixed with respect to the airfoil with the x_e axis fixed along the chord line, rather than one where the x_e axis is fixed along the direction of motion. Note, however, that any analysis of Doppler frequency shifts (not treated in this paper) should reference angles with respect to the direction of motion. Applications of equation (B1) at angles of attack should result in little additional error to that already built into the relation. Because it was derived with the plate assumed to be semi-infinite, \bar{D}_h becomes inaccurate at shallow upstream angles ($\Theta_e \rightarrow 180^\circ$), when applied to finite airfoils even for high frequencies. As frequency is lowered, the wavelengths become larger with respect to the chord and the directivity becomes increasingly in error. However, \bar{D}_h should be of sufficient accuracy to define the directivity of all the self-noise sources discussed because of their high-frequency character. The one exception to this is the stalled-airfoil noise.

When the angle of attack of the airfoil is increased sufficiently, the attached or mildly separated TBL flow on the suction side gives way to large-scale

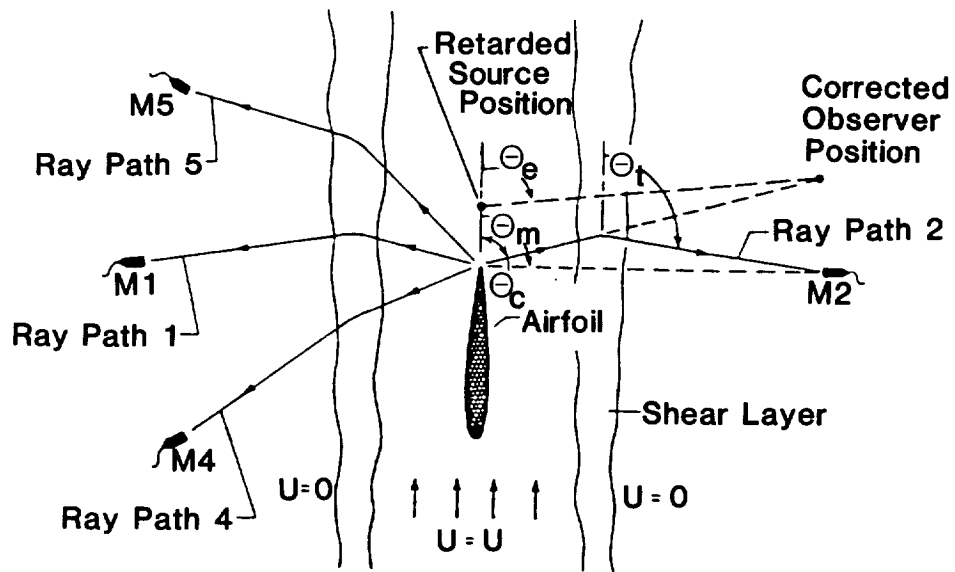


Figure B1. Sketch of shear layer refraction of acoustic transmission paths.

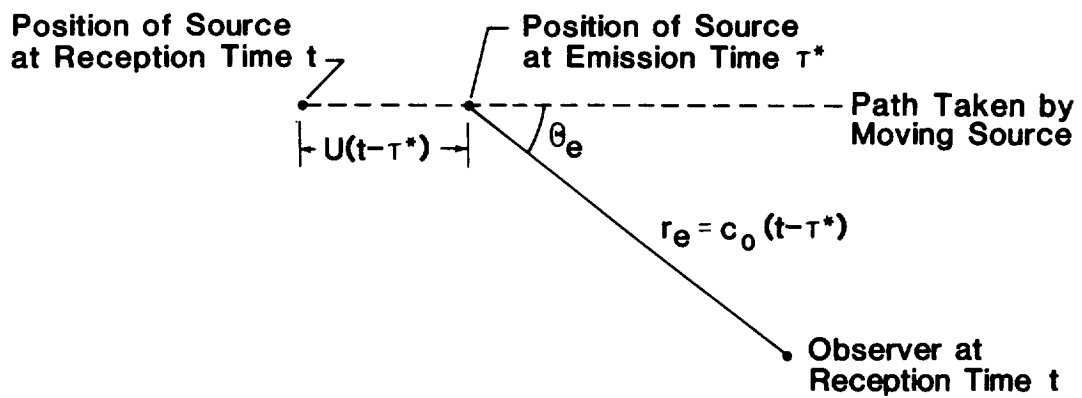


Figure B2. Emission from a source moving with a constant velocity.

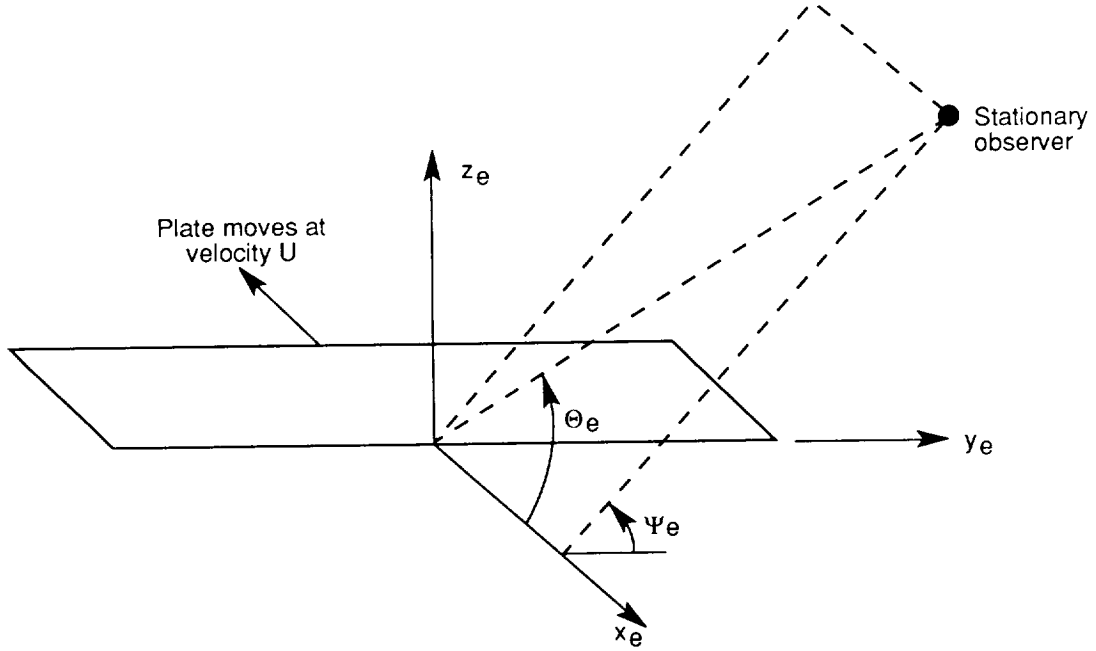


Figure B3. Flat plate in rectilinear motion.

separation. The turbulence eddies are then comparable in size to the airfoil chord length and the eddy convection speeds are low. The directivity for this low-frequency noise is more properly defined as that of a translating dipole, which is

$$\bar{D}_\ell(\Theta_e, \Phi_e) \approx \frac{\sin^2 \Theta_e \sin^2 \Phi_e}{(1 + M \cos \Theta_e)^4} \quad (\text{B2})$$

where the ℓ subscript indicates a low-frequency limit. The coordinate system and comments about angle definitions in equation (B1) apply also in equation (B2). Equation (B2) is employed for the directivity in the expression for stalled flow noise (eq. (30)).

For the noise data reduction in the present study, equation (B1) was used in the determination of the self-noise spectral levels for the reference observer position, at $r_e = 122$ cm and $\Theta_e = 90^\circ$. First, shear layer refraction corrections were calculated to determine the spectral level adjustments, to add to measured values, and a resultant source-observer location at r_e and Θ_e . This was done while keeping track of the actual physical coordinates of the trailing edge which varied with airfoil angle of attack. Finally, equation (B1), with $\Phi = 90^\circ$ and an assumed convection Mach number of $M_c \approx 0.8M$, was used to determine final level adjustments required to match results to the $\Theta_e = 90^\circ$ location.

Appendix C

Application of Predictions to a Rotor Broadband Noise Test

An acoustics test of a 40-percent scale model BO-105 helicopter main rotor was conducted in the German-Dutch Wind Tunnel (DNW). Figure C1 shows an overview of the test setup in the large open anechoic test section. The 4-meter-diameter rotor is shown positioned in the flow between the nozzle on the right and the flow collector on the left. A key aim of the test was to produce a large benchmark aeroacoustic data base to aid and verify rotor broadband noise prediction. In reference 30, the present authors compared data with predictions of rotor broadband self-noise for a number of rotor operating conditions. The predictions employed the self-noise prediction methods, which are documented in section 5 of the present paper, and the NASA ROTONET program (ref. 31) to define rotor performance and to sum contributions of noise from individual blade segments.

In this appendix, the experiment is not reviewed in detail nor are data-prediction comparisons presented, as reference 30 is complete in this regard. Rather, reference 30 is complemented by specifying how the self-noise prediction methods of the present paper were applied. Given below is a summary of the rotor prediction method, a definition of the rotor blade geometry and test modifications and a specification of input parameters for the individual source predictions. The degrees of success of data-prediction comparisons in reference 30 are discussed along with recommended refinements to the prediction methods.

To produce a rotor prediction, the rotor geometry definition and flight conditions, specified as thrust, rotor angle, rotor speed, flight velocity, and trim condition, are provided as inputs to the ROTONET rotor performance module. The particular module used assumes a fully articulated rotor with rigid blades and a simple uniform inflow model. The module determines local blade segment velocities and angles of attack for a number of radial and azimuthal positions. Ten radial segments were considered at 16 azimuthal positions. The BL thicknesses and other parameters needed are calculated. The noise due to each source is predicted for each blade segment, and the ROTONET noise radiation module is used to sum contributions from all blade segments to obtain, after accounting for Doppler shifts and the actual number of blades, the noise spectrum at the observer.

As indicated in reference 30, the accuracy of predictions depends on a number of factors including the accuracy of the performance module used. One may question the quasi-steady assumptions used in

defining the local BL characteristics, which ignore unsteadiness and resultant hysteresis effects. Likely more important is how well the aeroacoustic scaling determined from low-speed data extends to higher speed. The Mach number at the tip of the blades is 0.64 for rotor hover, whereas the 2D airfoil model tests were limited to Mach 0.21. Also there are questions on how to apply scaling obtained from symmetrical NACA 0012 sections with particular TE geometries to the cambered NACA 23012 rotor blade with different TE geometries.

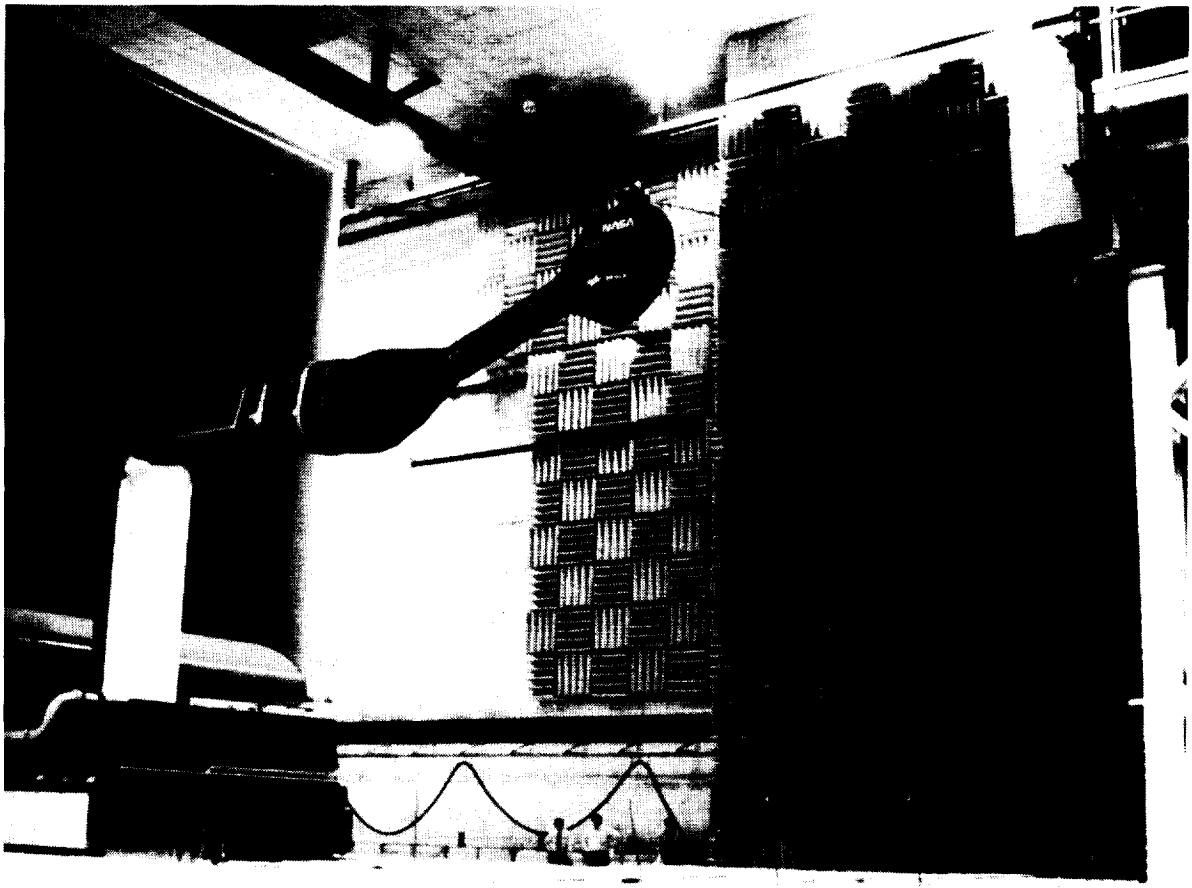
The model rotor is a 40-percent-scale, four-bladed, hingeless BO-105 rotor, with a diameter of 4.0 m and a chord of 0.121 m. A blade and its details are shown in figure C2. The blades have -8° linear twist and a 20-percent cutout from the hub center. The effects of several blade modifications were examined, including (1) application of Carborundum grit from the blade leading edge to 20 percent chord to match the BL trip condition for the 2D blade sections described in section 2 of the present paper, (2) taping of the TE with 0.064-mm-thick plastic tape to modify the "step tab" geometry, and (3) attachment of a rounded tip to each blade (the standard blades have a squared-off tip).

C.1. Boundary-Layer Definition

With the local blade segment mean velocities and angles of attack determined by the rotor performance module, the equations of section 3 were directly applied to determine the BL thicknesses required in the noise predictions. Most noise comparisons in reference 30 are for the blades with untripped BL. For the tripped BL, the fact that the BL trip conditions for the rotor blades matched the 2D test models assured the appropriateness of using the equations for a heavy trip rather than modifying the equations as required for the UTRC comparisons reported in section 6. For all BL thickness calculations, the aerodynamic angles of attack were used in the equations. The aerodynamic angle is referenced to the zero lift angle, which is -1.4° from the geometric angle for the NACA 23012 airfoil.

C.2. TBL-TE and Separation Noise Prediction

Given the definitions of segment chord length, span width, velocity, aerodynamic angle of attack, and BL thicknesses, the calculation of TBL-TE and separation noise is straightforward as specified in section 5. From the data-prediction comparisons of reference 30, it is concluded that the TBL-TE and separation noise calculations demonstrated a good predictive capability for these mechanisms. The rotor was tested from hover to moderately high flight



L-89-44

Figure C1. Test setup in DNW for helicopter main rotor broadband noise study reported in reference 30.

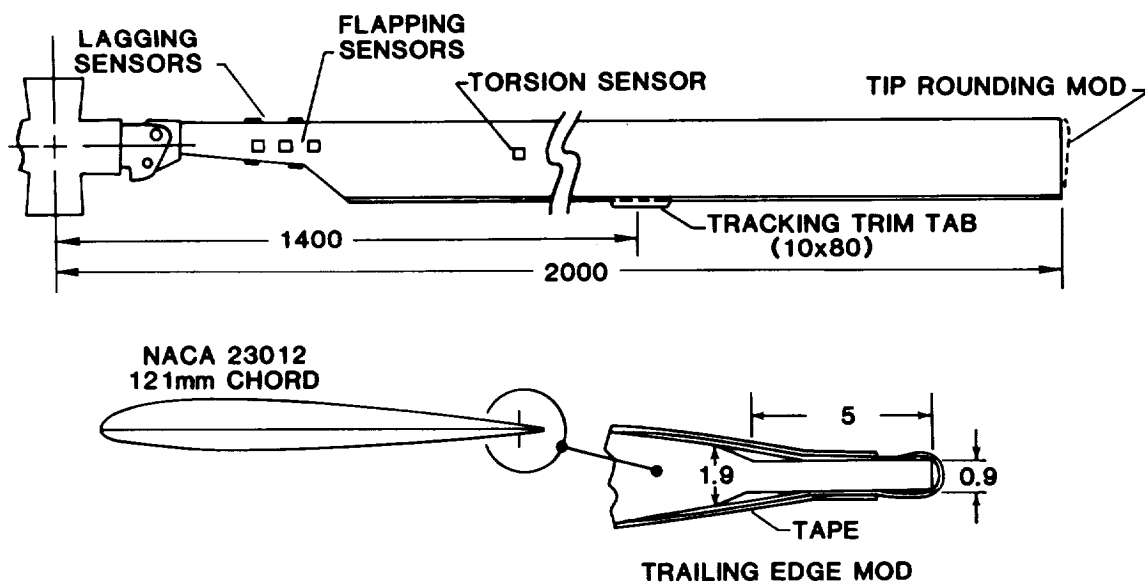


Figure C2. Model BO-105 blade details. All dimensions in mm.

speeds for various climb and descent rates at different thrust settings. Diagnostics included 1/2 rotor speed tests and the BL tripping tests. It is noted that the TBL-TE and separation noise predictions for a number of rotor conditions fell below contributions of LBL-VS, especially at the 1/2 rotor speed, and of TE bluntness noise. This represents a limitation of the comparisons which prevents sweeping statements regarding predictive accuracy of TBL-TE and separation noise sources. Still the agreements were quite good except when the rotor operated at full speed (tip speed of $M = 0.64$) and the boundary layers were tripped. Then the noise was underpredicted by about 6 dB. It is believed that for this high speed the heavy trip disturbed the flow substantially, made it dissimilar to the 2D model cases, where the speed was limited to $M = 0.21$, and perhaps changed the controlling noise mechanisms. Comparisons for the tripped BL rotor at 1/2 speed and the untripped BL blades at full and 1/2 speed produced good results.

C.3. LBL-VS Noise Prediction

The comparisons for LBL-VS noise in reference 30 showed, for a broad range of rotor conditions, very good predictions. As with the TBL-TE and separation noise predictions, the calculation of LBL-VS noise is straightforward given the specification of local flow conditions at the blade segments. A special note should be made for one key parameter involved in the calculations. The angle of attack α_* employed in the LBL-VS noise prediction (eqs. (53) to (60)) was the geometric angle rather than the aerodynamic angle for the NACA 23012 airfoil section. The BL thickness calculations, however, used the aerodynamic angle, as previously stated. The use of the geometric angle for the noise calculation is justified by (1) the better rotor data-prediction comparisons found using the geometric rather than the aerodynamic angle and (2) the lack of guidance one has in applying the acoustic scaling laws which were based on symmetrical airfoil results, to airfoils that are cambered. Remember that the controlling mechanism of LBL-VS noise is the presence of aeroacoustic feedback loops between the trailing edge and an upstream location on the airfoil surface where laminar instabilities occur. This geometric connection indicates that a purely aerodynamic angle definition for the LBL-VS mechanism would not likely be correct. An alternate viewpoint of the angle definition problem would be that the aerodynamic angle should be used, which would increase the noise predicted over that measured, but that allowance should be made for the fact that the inflow to the rotor blade segments is not the assumed smooth quasi-steady flow. The presence of sufficiently unsteady flow con-

ditions over portions of the rotor would prevent the establishment of the LBL-VS mechanism and related noise. Limiting LBL-VS noise production to some measure of inflow turbulence offers promise as a refinement to the self-noise prediction method.

C.4. Tip Vortex Formation Noise

The tip noise predictions were made for both the rounded and the squared-off blade tips tested. The performance module was used to determine the local flow velocities and angles for the tips at different azimuth locations. The α_{TIP} used was the NACA 23012 aerodynamic angle. Because the tip loading characteristics for the rotor blades differed from the reference case of the tip noise model, which was an untwisted large-aspect-ratio blade with uniform flow over the span, the sectional lift term of equation (66) was evaluated. The sectional lift slopes for the rotor blades were analyzed by employing a lifting-surface model adapted from reference 18. The velocity and angle of attack were linearly varied over the span near the tip of the lifting surface blade. It was found that the tip loading is increased over the reference case by a small amount. For equation (66), the redefined α'_{TIP} angle was then given by $\alpha'_{TIP} = 1.1\alpha_{TIP}$.

The predictions for tip noise in reference 30 were in all cases significantly below predictions for TBL-TE noise. This makes it impossible to truly assess the accuracy of the tip noise modeling for the rotor. However, since the data comparisons with the total levels predicted were good for both low and normal rotor speeds, the tip noise is apparently well predicted. It is noted that a review of data for a number of rotor cases, not all given in reference 30, indicated no significant effect due to the blade tip modification. This is in line with prediction for this rotor.

C.5. TE-Bluntness-Vortex-Shedding Noise

Given the flow definition for the blade segments from the performance module, the bluntness predictions require the specification of thickness h and flow angle parameter Ψ . As with the UTRC test comparisons of section 6, it is not clear how to apply scaling laws obtained from an airfoil with a particular TE geometry to a rotor blade with a different TE. For the step tab TE geometry, shown in figure C2, h was specified as the actual 0.9 mm and Ψ was taken as 14° , which is actual solid angle of the surface at the TE of the NACA 23012 airfoil (same as the NACA 0012 airfoil). However, because of the 0.5-mm step 5 mm upstream of the TE, 0.5 mm was added to the calculated value of δ_{avg}^* to approximately account for the anticipated step-caused BL flow deficit. For the TE tape modification case, δ_{avg}^* was taken as

that calculated, because the step was removed, but h was increased by four tape thicknesses. Had the tape remained fully attached to the TE surface (see fig. C2) during the test, two thicknesses would have been added. The flow angle Ψ was taken as 18° . The choice of this specific number was rather arbitrary, but is in line with that used for the UTRC comparisons (section 6) for rounded trailing edges. The tape rounded the TE bluntness which should reduce the persistence of and noise due to the separated flow in the near wake. The larger Ψ angle value (18° compared with 14°) results in less noise predicted.

The comparisons of reference 30 obtained using the above "reasonable" choices for the TE parameters give good results for all 1/2 rotor speed cases.

For the full rotor speed cases the levels were consistently overpredicted. This is believed to be due to a speed dependence for the bluntness mechanism that could not have been anticipated from the low speed airfoil data, from which the scaling laws were developed. Subsequent analysis indicates that much better agreement with data could have been obtained if the bluntness noise contribution was eliminated for blade segments exceeding Mach numbers of 0.45 or 0.5. This is in some conflict with comparisons in section 6 for the blade section noise of Schlinker and Amiet (ref. 3), which shows apparently strong bluntness noise at $M = 0.43$ and 0.5. However, based on the rotor results, an upper limit of 0.45 for the bluntness noise contribution is recommended as a refinement to the prediction method.

Appendix D

Prediction Code

The airfoil self-noise prediction method is available as a computer code written in standard FORTRAN 5 specifically for the Digital Equipment Corp. VAX-11/780 series machine running under the VMS operating system. To the extent possible, the code has been made machine independent. There is one input file to the code and one output file. Input consists of user supplied NAMELIST parameters while output is a table of 1/3-octave centered frequencies with corresponding sound pressure levels for

each noise mechanism followed by their total. The user selects which of the mechanisms to calculate.

The airfoil section for which a prediction is desired is assumed to be composed of a number of segments, each having its own chord, span, angle of attack, free-stream velocity, trailing-edge bluntness, and angle parameter, as well as observer directivity angles and distance. This permits a variety of configurations such as taper, twist, spanwise-varying free-stream velocity (for rotor blades), etc. The user may specify as many or few segments as desired depending on the complexity of the geometry. Characteristics for each segment are specified in the input file, which contains the FORTRAN variables given in table D1.

Table D1. Segment Characteristics Specified in Input File

FORTRAN name	Symbol	Description
NSEG		Number of segments
C	c	Chord length, m
L	L	Span, m
R	r_e	Observer distance, m
THETA	Θ_e	Observer angle from x -axis, deg
PHI	Φ_e	Observer angle from y -axis, deg
ALPSTAR	α_*	Aerodynamic angle of attack, deg
ALPTIP	α'_{TIP}	Tip flow angle, deg
H	h	Trailing-edge bluntness, m
PSI	Ψ	Trailing-edge angle, deg
U	U	Free-stream velocity, m/sec
ITRIP		0—Use untripped BL condition 1—Use tripped BL condition 2—Use lightly tripped BL condition
ILAM		1—Compute LBL-VS noise 0—Do not compute LBL-VS noise
ITURB		1—Compute turbulent TBL-TE noise 0—Do not compute TBL-TE noise
IBLUNT		1—Compute TE bluntness noise 0—Do not compute TE bluntness noise
ITIP		1—Compute tip noise 0—Do not compute tip noise
IROND		.TRUE.—Use rounded tip in tip calculation .FALSE.—Use square tip in tip calculation
VISC	ν	Kinematic viscosity, m^2/sec
CO	c_o	Speed of sound, m/sec

The prediction shown in figure 45(a) was obtained using the following input:

```

$INDATA
  NSEG      = 1,
  C         = .3048,
  L         = .4572,
  R         = 1.22,
  THETA     = 90.,
  PHI       = 90.,
  ALPSTAR   = 1.516,
  U         = 71.3,
  ITRIP     = 0,
  ILAM      = 1,
  ITURB     = 1,
$SEND

```

Note that all parameters need not be included in the input if their default values are desired (see program listing for default values). In this example, only the laminar and turbulent mechanisms are computed and the untripped boundary layer condition is used in both mechanisms. The airfoil consists of one segment of constant chord and the observer is 122 cm directly beneath the trailing edge at the midspan. The free-stream velocity has a constant value of 71.3 m/sec across the span. For this example, the output file is given in table D2.

Similarly, the prediction shown in figure 91, was obtained using the following input:

```

$INDATA
  NSEG      = 10.,
  C         = 10*.1524
  L         = 10*.0305

```

```

R         = 10* 1.22,
THETA     = 10* 90.,
PHI       = 10* 90.,
ALPSTAR   = 10* 5.4,
ALPHTIP   = 7.7,
U         = 10* 71.3,
ITRIP     = 1,
ILAM      = 0,
ITURB     = 1,
ITIP      = 1,
ROUND     = .TRUE.,
$SEND

```

This is an example of a multisegmented case where each segment has the same geometry and inflow conditions. Turbulent-boundary-layer noise and tip noise are calculated where the tip is rounded and at an effective angle of attack of 7.7°. All segments are summed to yield a total prediction for each mechanism as shown in table D3.

For the VAX-11/780 machine running under VMS, the following commands will compile, link, and execute the code (assumed to reside on PREDICT.FOR), read input from a file EXAMPLE.IN, and write results to a file EXAMPLE.OUT:

```

$ FOR PREDICT
$ LINK PREDICT
$ ASSIGN EXAMPLE.IN FOR004
$ ASSIGN EXAMPLE.OUT FOR005
$ RUN PREDICT

```

The details of execution for other machines or operating systems may vary. A listing of the code follows.

Table D2. Output File From Prediction Code for Test Case of Figure 45(a)

ONE-THIRD OCTAVE SOUND PRESSURE LEVELS							
FREQUENCY(HZ)	PRESSURE SIDE TBL	SUCTION SIDE TBL	SEPARATION SIDE TBL	LAMINAR	BLUNTNES	TIP	TOTAL
100.000	20.654	28.704	-100.000	-17.142	0.000	0.000	29.336
125.000	24.461	31.965	-100.000	-13.285	0.000	0.000	32.676
160.000	28.291	35.244	-75.254	-9.018	0.000	0.000	36.042
200.000	31.437	37.937	-49.243	-5.161	0.000	0.000	38.815
250.000	34.309	40.400	-27.506	-1.304	0.000	0.000	41.356
315.000	37.023	42.736	-9.030	2.690	0.000	0.000	43.768
400.000	39.577	44.949	6.266	6.820	0.000	0.000	46.057
500.000	41.761	46.859	17.532	10.677	0.000	0.000	48.034
630.000	43.845	48.706	26.603	14.671	0.000	0.000	49.954
800.000	45.839	50.503	33.718	18.801	0.000	0.000	51.849
1000.000	47.581	52.106	38.756	22.658	0.000	0.000	53.568
1250.000	49.233	53.664	42.692	26.515	0.000	0.000	55.255
1600.000	50.987	55.368	46.294	30.782	0.000	0.000	57.106
2000.000	52.533	56.907	49.334	37.725	0.000	0.000	58.817
2500.000	54.074	57.750	51.298	47.262	0.000	0.000	60.167
3150.000	55.570	57.500	50.766	48.959	0.000	0.000	60.496
4000.000	56.044	56.082	47.711	41.796	0.000	0.000	59.455
5000.000	55.399	54.541	44.617	32.428	0.000	0.000	58.208
6300.000	53.840	52.942	40.974	28.433	0.000	0.000	56.553
8000.000	52.190	51.253	36.227	24.304	0.000	0.000	54.821
10000.000	50.638	49.614	30.419	20.447	0.000	0.000	53.192
12500.000	49.044	47.890	22.834	16.590	0.000	0.000	51.523
16000.000	47.202	45.851	11.842	12.323	0.000	0.000	49.591
20000.000	45.436	43.863	-0.924	8.466	0.000	0.000	47.731
25000.000	43.549	41.710	-16.833	4.609	0.000	0.000	45.737
31500.000	41.440	39.279	-37.092	0.614	0.000	0.000	43.503
40000.000	39.065	36.522	-62.593	-3.515	0.000	0.000	40.987

Table D3. Output File From Prediction Code for Test Case of Figure 91

ONE-THIRD OCTAVE SOUND PRESSURE LEVELS							
FREQUENCY(HZ)	PRESSURE SIDE TBL	SUCTION SIDE TBL	SEPARATION SIDE TBL	LAMINAR	BLUNTNES	TIP	TOTAL
100.000	19.913	43.883	-19.803	0.000	0.000	-34.005	43.900
125.000	23.788	46.159	-0.396	0.000	0.000	-24.312	46.184
160.000	27.673	48.459	16.851	0.000	0.000	-14.255	48.498
200.000	30.853	50.372	29.124	0.000	0.000	-5.769	50.452
250.000	33.746	52.155	38.723	0.000	0.000	2.145	52.407
315.000	36.470	53.894	46.334	0.000	0.000	9.738	54.662
400.000	39.024	55.609	52.245	0.000	0.000	16.940	57.320
500.000	41.202	57.165	56.460	0.000	0.000	23.074	59.897
630.000	43.274	58.766	59.996	0.000	0.000	28.824	62.489
800.000	45.252	60.360	63.297	0.000	0.000	34.121	65.130
1000.000	46.980	60.940	65.719	0.000	0.000	38.475	67.016
1250.000	48.620	60.473	65.697	0.000	0.000	42.257	66.917
1600.000	50.364	58.874	62.909	0.000	0.000	45.774	64.582
2000.000	51.911	57.328	59.818	0.000	0.000	48.349	62.363
2500.000	53.456	55.775	56.383	0.000	0.000	50.351	60.580
3150.000	54.709	54.122	51.975	0.000	0.000	51.821	59.364
4000.000	54.799	52.336	45.974	0.000	0.000	52.694	58.443
5000.000	53.761	50.565	38.550	0.000	0.000	52.917	57.439
6300.000	52.162	48.597	28.510	0.000	0.000	52.544	56.204
8000.000	50.507	46.387	15.081	0.000	0.000	51.512	54.736
10000.000	48.936	44.132	-0.755	0.000	0.000	49.955	53.078
12500.000	47.311	41.665	-20.241	0.000	0.000	47.826	51.110
16000.000	45.415	38.655	-46.603	0.000	0.000	44.802	48.594
20000.000	43.583	35.650	-75.275	0.000	0.000	41.466	46.075
25000.000	41.611	32.347	-90.000	0.000	0.000	37.557	43.405
31500.000	39.390	28.582	-90.000	0.000	0.000	32.904	40.555
40000.000	36.873	24.291	-90.000	0.000	0.000	27.449	37.552

0001 PROGRAM PREDICT

0002

0003

0004 C

0005 C

0006 C

0007

0008 C

0009 C

0010

0011 C

0012 C

0013 C

0014 C

0015 C

0016 C

0017 C

0018 C

0019 C

0020 C

0021 C

0022 C

0023 C

0024 C

0025 C

0026 C

0027 C

0028 C

0029 C

0030 C

0031 C

0032 C

0033 C

0034 C

0035 C

0036 C

0037 C

0038 C

0039 C

0040 C

0041 C

0042 C

0043 C

0044 C

0045 C

0046 C

0047 C

0048 C

0049 C

0050 C

0051 C

0052 C

0053 C

0054 C

0055 C

0056 C

0057 C

0058 C

0059 C

0060 C

0061 C

0062 C

0063 C

0064 C

0065

0066

0067

0068

0069

0070

0071

0072

0073

0074

0075

***** VARIABLE DEFINITIONS *****

VARIABLE NAME	DEFINITION	UNITS
ALPHTIP	TIP ANGLE OF ATTACK	DEGREES
ALPSTAR	SEGMENT ANGLE OF ATTACK	DEGREES
ALPRAT	TIP LIFT CURVE SLOPE	---
C	SEGMENT CHORDLENGTH	METERS
C0	SPEED OF SOUND	METERS/SEC
FRCEN	1/3 OCTAVE CENTERED FREQUENCIES	HERTZ
H	SEGMENT TRAILING EDGE THICKNESS	METERS
IBLUNT	FLAG TO COMPUTE BLUNTNESS NOISE	---
ILAM	FLAG TO COMPUTE LBL NOISE	---
ITIP	FLAG TO COMPUTE TIP NOISE	---
ITRIP	FLAG TO TRIP BOUNDARY LAYER	---
ITURB	FLAG TO COMPUTE TBLTE NOISE	---
L	SEGMENT SPAN LENGTH	METERS
MAXFREQ	MAXIMUM NUMBER OF FREQUENCIES	---
MAXSEG	MAXIMUM NUMBER OF SEGMENTS	---
NFREQ	NUMBER OF 1/3 OCTAVE FREQUENCIES	---
NSEG	NUMBER OF SEGMENTS	---
P1	PRESSURE ASSOCIATED WITH TBLTE PREDICTION	NT/M2
P2	PRESSURE ASSOCIATED WITH TBLTE PREDICTION	NT/M2
P3	PRESSURE ASSOCIATED WITH TBLTE PREDICTION	NT/M2
P4	PRESSURE ASSOCIATED WITH TOTAL PREDICTION	NT/M2
P5	PRESSURE ASSOCIATED WITH LBLVS PREDICTION	NT/M2
P6	PRESSURE ASSOCIATED WITH BLUNTNESS PREDICTION	NT/M2
P7	PRESSURE ASSOCIATED WITH TIP NOISE PREDICTION	NT/M2
PHI	DIRECTIVITY ANGLE	DEGREES
PSI	BLUNTNESS ANGLE	DEGREES
R	SEGMENT TO OBSERVER DISTANCE	METERS
ROUND	LOGICAL INDICATING ROUNDED TIP	---
SPL	TOTAL SOUND PRESSURE LEVEL	DB
SPLALPH	SOUND PRESSURE LEVEL ASSOCIATED WITH TBLTE PREDICTION	DB
SPLBLNT	SOUND PRESSURE LEVEL ASSOCIATED WITH BLUNTNESS PREDICTION	DB
SPLLBL	SOUND PRESSURE LEVEL ASSOCIATED WITH LBL PREDICTION	DB
SPLP	SOUND PRESSURE LEVEL ASSOCIATED WITH TBLTE PREDICTION	DB
SPLS	SOUND PRESSURE LEVEL ASSOCIATED WITH TBLTE PREDICTION	DB
SPLTBL	TOTAL PRESSURE LEVEL ASSOCIATED WITH TBLTE PREDICTION	DB
SPLTIP	SOUND PRESSURE LEVEL ASSOCIATED WITH TIP NOISE PREDICTION	DB
ST	STROUHAL NUMBER	--
THETA	DIRECTIVITY ANGLE	DEGREES
U	SEGMENT FREESTREAM VELOCITY	METERS/SEC
VISC	KINEMATIC VISCOSITY	M2/SEC

PARAMETER (MAXSEG = 20, MAXFREQ = 27)

DIMENSION	FRCEN(MAXFREQ)	C(MAXSEG)	L(MAXSEG)
1	ST(MAXFREQ)	SPLLBL(MAXFREQ)	SPLTBL(MAXFREQ)
2	U(MAXSEG)	SPLP(MAXFREQ)	SPLS(MAXFREQ)
3	SPLALPH(MAXFREQ)	SPL(7,MAXFREQ)	R(MAXSEG)
5	SPLBLNT(MAXFREQ)	PHI(MAXSEG)	SPLTIP(MAXFREQ)
7	THETA(MAXSEG)	ALPSTAR(MAXSEG)	PSI(MAXSEG)

```

0076      8          H (MAXSEG)          ,P1 (MAXFREQ)          ,P2 (MAXFREQ)          ,
0077      9          P3 (MAXFREQ)        ,P4 (MAXFREQ)        ,P5 (MAXFREQ)        ,
0078      1          P6 (MAXFREQ)        ,P7 (MAXFREQ)
0079
0080
0081      REAL          L
0082      LOGICAL       ROUND
0083
0084      C      DEFINE DEFAULT VALUES FOR NAMELIST DATA
0085      C      -----
0086
0087
0088      DATA C      /  MAXSEG*1.0      /
0089      DATA L      /  MAXSEG*.10     /
0090      DATA R      /  MAXSEG * 1.    /
0091      DATA THETA  /  MAXSEG * 90.   /
0092      DATA PHI    /  MAXSEG * 90.   /
0093      DATA ALPSTAR /  MAXSEG * 0.0   /
0094      DATA H      /  MAXSEG * .0005 /
0095      DATA PSI    /  MAXSEG * 14.0  /
0096      DATA U      /  MAXSEG * 100.  /
0097      DATA ITRIP  /  0             /
0098      DATA ILAM   /  0             /
0099      DATA ITURB  /  0             /
0100      DATA IBLUNT /  0             /
0101      DATA ITIP  /  0             /
0102      DATA ALPHTIP /  0.0         /
0103      DATA NSEG   /  10           /
0104      DATA VISC   /  1.4529E-5    /
0105      DATA C0     /  340.46       /
0106      DATA ALPRAT /  1.0         /
0107      DATA ROUND  /  .FALSE.      /
0108
0109      DATA NFREQ   /  27           /
0110
0111      C      SET UP VALUES OF 1/3 OCTAVE CENTERED FREQUENCIES
0112      C      -----
0113
0114      DATA FRCEN / 100. , 125. , 160. , 200. , 250. ,
0115      1          315. , 400. , 500. , 630. , 800. ,
0116      1          1000. , 1250. , 1600. , 2000. , 2500. ,
0117      3          3150. , 4000. , 5000. , 6300. , 8000. ,
0118      2          10000. , 12500. , 16000. , 20000. , 25000. ,
0119      3          31500. , 40000. /
0120
0121
0122      NAMELIST /INDATA / C      ,L      ,R      ,
0123      1          THETA ,PHI    ,ALPSTAR ,
0124      2          H      ,PSI    ,U      ,
0125      1          ITRIP ,ILAM   ,ITURB  ,
0126      2          IBLUNT ,ITIP   ,ROUND  ,
0127      3          ALPHTIP ,NSEG   ,C0     ,
0128      4          VISC
0129
0130
0131
0132      C      READ IN NAMELIST DATA AND ECHO INPUT TO OUTPUT FILE
0133      C      -----
0134
0135      OPEN(UNIT=4, STATUS = 'OLD')
0136      READ(4,INDATA)
0137
0138      OPEN(UNIT=5, STATUS = 'NEW')
0139      WRITE(5,INDATA)
0140
0141
0142      C      INITIALIZE ALL PREDICTED PRESSURES AND SOUND
0143      C      PRESSURE LEVELS TO ZERO
0144      C      -----
0145
0146      DO 6001 I=1,NFREQ
0147          P1(I) = 0.0
0148          P2(I) = 0.0
0149          P3(I) = 0.0
0150          P4(I) = 0.0

```

```

0151         P5(I) = 0.0
0152         P6(I) = 0.0
0153         P7(I) = 0.0
0154
0155         DO 6002 J=1,7
0156             SPL(J,I) = 0.0
0157 6002     CONTINUE
0158 6001 CONTINUE
0159
0160
0161 C     FOR EACH BLADE SEGMENT, MAKE A NOISE PREDICTION ACCORDING
0162 C     TO THE MECHANISMS SELECTED. TIP NOISE IS PREDICTED FOR
0163 C     THE LAST SEGMENT ONLY.
0164 C     -----
0165
0166         DO 6000 III=1,NSEG
0167
0168             IF (ILAM .EQ. 1)
0169 1         CALL LBLVS(ALPSTAR(III),C(III),U(III),FRCEN,SPLLBL,
0170 2             THETA(III),PHI(III),L(III),R(III),NFREQ,
0171 3             VISC,C0)
0172
0173             IF (ITURB .EQ. 1)
0174 1         CALL TBLTE(ALPSTAR(III),C(III),U(III),FRCEN,ITRIP,SPLP,
0175 1             SPLS,SPLALPH,SPLTBL,THETA(III),PHI(III),L(III),R(III),
0176 2             NFREQ,VISC,C0)
0177
0178             IF (IBLUNT .EQ. 1)
0179 1         CALL BLUNT(ALPSTAR(III),C(III),U(III),FRCEN,ITRIP,SPLBLNT,
0180 1             THETA(III),PHI(III),L(III),R(III),H(III),PSI(III),
0181 2             NFREQ,VISC,C0)
0182
0183             IF ((ITIP .EQ. 1) .AND. (III .EQ. NSEG))
0184 1         CALL TIPNOIS(ALPHATIP,ALPRAT,C(III),U(III),FRCEN,SPLTIP,
0185 2             THETA,PHI,R(III),NFREQ,VISC,C0,ROUND)
0186
0187 C     ADD IN THIS SEGMENT'S CONTRIBUTION ON A MEAN-SQUARE
0188 C     PRESSURE BASIS
0189 C     -----
0190 C
0191
0192         DO 989 I=1,NFREQ
0193
0194             IF (ILAM .EQ. 1) THEN
0195                 P5(I) = P5(I) + 10.**(SPLLBL(I)/10.)
0196             ENDIF
0197
0198             IF (ITURB .EQ. 1) THEN
0199                 P1(I) = P1(I) + 10.**(SPLP(I)/10.)
0200                 P2(I) = P2(I) + 10.**(SPLS(I)/10.)
0201                 P3(I) = P3(I) + 10.**(SPLALPH(I)/10.)
0202             ENDIF
0203
0204             IF (IBLUNT .EQ. 1) THEN
0205                 P6(I) = P6(I) + 10.**(SPLBLNT(I)/10.)
0206             ENDIF
0207
0208             IF ((ITIP .EQ. 1) .AND. (III .EQ. NSEG)) THEN
0209                 P7(I) = P7(I) + 10.**(SPLTIP(I)/10.)
0210             ENDIF
0211
0212
0213 C     COMPUTE TOTAL PRESSURE FOR THE SEGMENT FOR ALL MECHANISMS
0214 C     -----
0215
0216         P4(I) = P1(I) + P2(I) + P3(I) + P5(I) + P6(I) + P7(I)
0217
0218 989     CONTINUE
0219 6000 CONTINUE
0220
0221 C     CONTRIBUTIONS FROM ALL SEGMENTS ARE NOW ACCOUNTED FOR.
0222 C     COMPUTE SOUND PRESSURE LEVELS FOR EACH MECHANISM AND
0223 C     FOR THE TOTAL
0224 C     -----
0225

```

```

0226      DO 6003 I=1,NFREQ
0227          IF (P1(I) .NE. 0.) SPL(1,I) = 10.*ALOG10(P1(I))
0228          IF (P2(I) .NE. 0.) SPL(2,I) = 10.*ALOG10(P2(I))
0229          IF (P3(I) .NE. 0.) SPL(3,I) = 10.*ALOG10(P3(I))
0230          IF (P4(I) .NE. 0.) SPL(4,I) = 10.*ALOG10(P4(I))
0231          IF (P5(I) .NE. 0.) SPL(5,I) = 10.*ALOG10(P5(I))
0232          IF (P6(I) .NE. 0.) SPL(6,I) = 10.*ALOG10(P6(I))
0233          IF (P7(I) .NE. 0.) SPL(7,I) = 10.*ALOG10(P7(I))
0234      6003 CONTINUE
0235
0236
0237      C      WRITE OUTPUT FILE
0238      C      -----
0239
0240          WRITE(5,7000)
0241
0242          DO 6005 I=1,NFREQ
0243
0244              WRITE(5,7100) FRCEN(I), (SPL(J,I),J=1,3), (SPL(J,I),J=5,7),
0245              1          SPL(4,I)
0246              IF (MOD(I,5) .EQ. 0) WRITE(5,7200)
0247      6005 CONTINUE
0248
0249      7000 FORMAT(1H1,52X,'ONE-THIRD OCTAVE',/,50X,'SOUND PRESSURE LEVELS'
0250      1          ////,5X,'          ' ,',',', 'PRESSURE',',',
0251      2          '          SUCTION',',', 'SEPARATION',',',
0252      3          5X,' FREQUENCY(HZ)',',', 'SIDE TBL',',',
0253      4          '          SIDE TBL',',', 'SIDE TBL',',',
0254      5          '          LAMINAR',',', 'BLUNTNESNESS',',',
0255      6          '          TIP',',', 'TOTAL',',',
0256      7          /,5X,8('-----'),/)
0257
0258      7100 FORMAT(8F15.3)
0259      7200 FORMAT(' ')
0260      8000 FORMAT(I3)
0261      8002 FORMAT(4I10)
0262
0263
0264          STOP
0265      END

```

```

0001      SUBROUTINE LBLVS(ALPSTAR,C,U ,FRCEN,SPLLAM,THETA,PHI,L,R,
0002      1          NREQ,VISC,C0)
0003
0004      PARAMETER (MAXFREQ = 27)
0005
0006
0007
0008      C          -----
0009      C          ***** VARIABLE DEFINITIONS *****
0010      C          -----
0011
0012      C          VARIABLE NAME          DEFINITION          UNITS
0013      C          -----          -----          -----
0014
0015
0016      C          ALPSTAR          ANGLE OF ATTACK          DEGREES
0017      C          C          CHORD LENGTH          METERS
0018      C          C0          SPEED OF SOUND          METERS/SEC
0019      C          D          REYNOLDS NUMBER RATIO          ---
0020      C          DBARH          HIGH FREQUENCY DIRECTIVITY          ---
0021      C          DELTAP          PRESSURE SIDE BOUNDARY LAYER
0022      C          THICKNESS          METERS
0023      C          DSTRP          PRESSURE SIDE BOUNDARY LAYER
0024      C          DISPLACEMENT THICKNESS          METERS
0025      C          DSTRS          SUCTION SIDE BOUNDARY LAYER
0026      C          DISPLACEMENT THICKNESS          METERS
0027      C          E          STROUHAL NUMBER RATIO          ---
0028      C          FRCEN          1/3 OCTAVE FREQUENCIES          HERTZ
0029      C          G1          SOUND PRESSURE LEVEL FUNCTION          DB
0030      C          G2          OVERALL SOUND PRESSURE LEVEL
0031      C          FUNCTION          DB
0032      C          G3          OVERALL SOUND PRESSURE LEVEL
0033      C          FUNCTION          DB
0034      C          ITRIP          FLAG TO TRIP BOUNDARY LAYER          ---
0035      C          L          SPAN          METERS
0036      C          M          MACH NUMBER          ---
0037      C          NREQ          NUMBER OF FREQUENCIES          ---
0038      C          OASPL          OVERALL SOUND PRESSURE LEVEL          DB
0039      C          PHI          DIRECTIVITY ANGLE          DEGREES
0040      C          R          OBSERVER DISTANCE FROM SEGMENT          METERS
0041      C          RC          REYNOLDS NUMBER BASED ON CHORD          ---
0042      C          RC0          REFERENCE REYNOLDS NUMBER          ---
0043      C          SCALE          GEOMETRIC SCALING TERM          ---
0044      C          SPLLAM          SOUND PRESSURE LEVEL DUE TO
0045      C          LAMINAR MECHANISM          DB
0046      C          STPRIM          STROUHAL NUMBER BASED ON PRESSURE
0047      C          SIDE BOUNDARY LAYER THICKNESS          ---
0048      C          ST1PRIM          REFERENCE STROUHAL NUMBER          ---
0049      C          STPKPRM          PEAK STROUHAL NUMBER          ---
0050      C          THETA          DIRECTIVITY ANGLE          DEGREES
0051      C          U          FREESTREAM VELOCITY          METERS/SEC
0052      C          VISC          KINEMATIC VISCOSITY          M2/SEC
0053
0054
0055
0056      DIMENSION STPRIM(MAXFREQ) ,SPLLAM(MAXFREQ) ,FRCEN(MAXFREQ)
0057
0058      REAL      L          ,M
0059
0060
0061      C          COMPUTE REYNOLDS NUMBER AND MACH NUMBER
0062      C          -----
0063
0064      M          = U / C0
0065      RC          = U * C/VISC
0066
0067
0068      C          COMPUTE BOUNDARY LAYER THICKNESSES
0069      C          -----
0070
0071      CALL THICK(C,U ,ALPSTAR,ITRIP,DELTAP,DSTRS,DSTRP,C0,VISC)
0072
0073
0074
0075      C          COMPUTE DIRECTIVITY FUNCTION

```

```

0076 C -----
0077
0078 CALL DIRECTH(M,THETA,PHI,DBARH)
0079
0080
0081
0082 C COMPUTE REFERENCE STROUHAL NUMBER
0083 C -----
0084
0085 IF (RC .LE. 1.3E+05) ST1PRIM = .18
0086 IF ((RC .GT. 1.3E+05).AND.(RC.LE.4.0E+05))ST1PRIM=.001756*RC**.3931
0087 IF (RC .GT. 4.0E+05) ST1PRIM = .28
0088
0089 STPKPRM = 10.**(-.04*ALPSTAR) * ST1PRIM
0090
0091
0092
0093 C COMPUTE REFERENCE REYNOLDS NUMBER
0094 C -----
0095
0096 IF (ALPSTAR .LE. 3.0) RCO=10.**(.215*ALPSTAR+4.978)
0097 IF (ALPSTAR .GT. 3.0) RCO=10.**(.120*ALPSTAR+5.263)
0098
0099
0100
0101
0102 C COMPUTE PEAK SCALED SPECTRUM LEVEL
0103 C -----
0104
0105 D = RC / RCO
0106
0107 IF (D .LE. .3237) G2=77.852*ALOG10(D)+15.328
0108 IF ((D .GT. .3237).AND.(D .LE. .5689))
0109 1 G2 = 65.188*ALOG10(D) + 9.125
0110 IF ((D .GT. .5689).AND.(D .LE. 1.7579))
0111 1 G2 = -114.052 * ALOG10(D)**2.
0112 IF ((D .GT. 1.7579).AND.(D .LE. 3.0889))
0113 1 G2 = -65.188*ALOG10(D)+9.125
0114 IF (D .GT. 3.0889) G2 =-77.852*ALOG10(D)+15.328
0115
0116
0117 G3 = 171.04 - 3.03 * ALPSTAR
0118
0119 SCALE = 10. * ALOG10(DELTAP*M**5*DBARH*L/R**2)
0120
0121
0122
0123 C COMPUTE SCALED SOUND PRESSURE LEVELS FOR EACH STROUHAL NUMBER
0124 C -----
0125
0126 DO 100 I=1,NFREQ
0127
0128 STPRIM(I) = FRCEN(I) * DELTAP / U
0129
0130 E = STPRIM(I) / STPKPRM
0131
0132 IF (E .LT. .5974) G1=39.8*ALOG10(E)-11.12
0133 IF ((E .GE. .5974).AND.(E .LE. .8545))
0134 1 G1 = 98.409 * ALOG10(E) + 2.0
0135 IF ((E .GE. .8545).AND.(E .LT. 1.17))
0136 1 G1 = -5.076+SQRT(2.484-506.25*(ALOG10(E))**2.)
0137 IF ((E .GE. 1.17).AND.(E .LT. 1.674))
0138 1 G1 = -98.409 * ALOG10(E) + 2.0
0139 IF (E .GE. 1.674) G1=-39.80*ALOG10(E)-11.12
0140
0141 SPLLAM(I) = G1 + G2 + G3 + SCALE
0142
0143 100 CONTINUE
0144
0145 RETURN
0146 END

```


0001 SUBROUTINE TBLTE(ALPSTAR,C,U ,FRCEN,ITRIP,SPLP,SPLS,
0002 1 SPLALPH,SPLTBL,THETA,PHI,L,R,NFREQ,VISC,C0)
0003

0004
0005
0006 C -----
0007 C ***** VARIABLE DEFINITIONS *****
0008 C -----
0009
0010

LINE	CHAR	VARIABLE NAME	DEFINITION	UNITS
0011				
0012	C	VARIABLE NAME	DEFINITION	UNITS
0013	C	-----	-----	-----
0014				
0015	C	A	STROUHAL NUMBER RATIO	---
0016	C	A0	FUNCTION USED IN 'A' CALCULATION	---
0017	C	A02	FUNCTION USED IN 'A' CALCULATION	---
0018	C	AA	'A' SPECTRUM SHAPE EVALUATED AT	
0019	C		STROUHAL NUMBER RATIO	DB
0020	C	ALPSTAR	ANGLE OF ATTACK	DEGREES
0021	C	AMAXA	MAXIMUM 'A' CURVE EVALUATED AT	
0022	C		STROUHAL NUMBER RATIO	DB
0023	C	AMAXA0	MAXIMUM 'A' CURVE EVALUATED AT A0	DB
0024	C	AMAXA02	MAXIMUM 'A' CURVE EVALUATED AT A02	DB
0025	C	AMAXB	MAXIMUM 'A' CURVE EVALUATED AT B	DB
0026	C	AMINA	MINIMUM 'A' CURVE EVALUATED AT	
0027	C		STROUHAL NUMBER RATIO	DB
0028	C	AMINA0	MINIMUM 'A' CURVE EVALUATED AT A0	DB
0029	C	AMINA02	MINIMUM 'A' CURVE EVALUATED AT A02	DB
0030	C	AMINB	MINIMUM 'A' CURVE EVALUATED AT B	DB
0031	C	ARA0	INTERPOLATION FACTOR	---
0032	C	ARA02	INTERPOLATION FACTOR	---
0033	C	B	STROUHAL NUMBER RATIO	---
0034	C	B0	FUNCTION USED IN 'B' CALCULATION	---
0035	C	BB	'B' SPECTRUM SHAPE EVALUATED AT	
0036	C		STROUHAL NUMBER RATIO	DB
0037	C	BETA	USED IN 'B' COMPUTATION	---
0038	C	BETA0	USED IN 'B' COMPUTATION	---
0039	C	BMAXB	MAXIMUM 'B' EVALUATED AT B	DB
0040	C	BMAXB0	MAXIMUM 'B' EVALUATED AT B0	DB
0041	C	BMINB	MINIMUM 'B' EVALUATED AT B	DB
0042	C	BMINB0	MINIMUM 'B' EVALUATED AT B0	DB
0043	C	BRB0	INTERPOLATION FACTOR	DB
0044	C	C	CHORD LENGTH	METERS
0045	C	C0	SPEED OF SOUND	METERS/SEC
0046	C	DBARH	HIGH FREQUENCY DIRECTIVITY	---
0047	C	DBARL	LOW FREQUENCY DIRECTIVITY	---
0048	C	DELK1	CORRECTION TO AMPLITUDE FUNCTION	DB
0049	C	DELTAP	PRESSURE SIDE BOUNDARY LAYER THICKNESS	METERS
0050	C	DSTRP	PRESSURE SIDE DISPLACEMENT THICKNESS	METERS
0051	C	DSTRS	SUCTION SIDE DISPLACEMENT THICKNESS	METERS
0052	C	FRCEN	ARRAY OF CENTERED FREQUENCIES	HERTZ
0053	C	GAMMA	USED IN 'B' COMPUTATION	---
0054	C	GAMMA0	USED IN 'B' COMPUTATION	---
0055	C	ITRIP	TRIGGER TO TRIP BOUNDARY LAYER	---
0056	C	K1	AMPLITUDE FUNCTION	DB
0057	C	K2	AMPLITUDE FUNCTION	DB
0058	C	L	SPAN	METERS
0059	C	M	MACH NUMBER	---
0060	C	NFREQ	NUMBER OF CENTERED FREQUENCIES	---
0061	C	PHI	DIRECTIVITY ANGLE	DEGREES
0062	C	P1	PRESSURE SIDE PRESSURE	NT/M2
0063	C	P2	SUCTION SIDE PRESSURE	NT/M2
0064	C	P4	PRESSURE FROM ANGLE OF ATTACK	
0065	C		CONTRIBUTION	NT/M2
0066	C	R	SOURCE TO OBSERVER DISTANCE	METERS
0067	C	RC	REYNOLDS NUMBER BASED ON CHORD	---
0068	C	RDSTRP	REYNOLDS NUMBER BASED ON PRESSURE	
0069	C		SIDE DISPLACEMENT THICKNESS	---
0070	C	RDSTRS	REYNOLDS NUMBER BASED ON SUCTION	
0071	C		SIDE DISPLACEMENT THICKNESS	---
0072	C	SPLALPH	SOUND PRESSURE LEVEL DUE TO ANGLE OF	
0073	C		ATTACK CONTRIBUTION	DB
0074	C	SPLP	SOUND PRESSURE LEVEL DUE TO PRESSURE	
0075	C		SIDE OF AIRFOIL	DB

```

0076 C      SPLS          SOUND PRESSURE LEVEL DUE TO SUCTION
0077 C              SIDE OF AIRFOIL                      DB
0078 C      SPLTBL        TOTAL SOUND PRESSURE LEVEL DUE TO
0079 C              TBLTE MECHANISM                      DB
0080 C      STP          PRESSURE SIDE STROUHAL NUMBER      ---
0081 C      STS          SUCTION SIDE STROUHAL NUMBER      ---
0082 C      ST1          PEAK STROUHAL NUMBER             ---
0083 C      ST1PRIM      PEAK STROUHAL NUMBER             ---
0084 C      ST2          PEAK STROUHAL NUMBER             ---
0085 C      STPEAK      PEAK STROUHAL NUMBER             ---
0086 C      SWITCH      LOGICAL FOR COMPUTATION OF ANGLE
0087 C              OF ATTACK CONTRIBUTION                ---
0088 C      THETA        DIRECTIVITY ANGLE                 DEGREES
0089 C      U           VELOCITY                          METERS/SEC
0090 C      VISC         KINEMATIC VISCOSITY              M2/SEC
0091 C      XCHECK      USED TO CHECK FOR ANGLE OF ATTACK
0092 C              CONTRIBUTION                          ---
0093 C
0094
0095      PARAMETER (MAXFREQ = 27)
0096
0097
0098      DIMENSION SPLTBL(MAXFREQ) ,SPLP(MAXFREQ) ,SPLS(MAXFREQ) ,
0099      1          SPLALPH(MAXFREQ) ,STP(MAXFREQ) ,
0100      1          STS(MAXFREQ) ,FRCEN(MAXFREQ)
0101
0102      LOGICAL SWITCH
0103      REAL      L,M,K1,K2
0104
0105      RC        = U * C / VISC
0106      M         = U / C0
0107
0108
0109 C      COMPUTE BOUNDARY LAYER THICKNESSES
0110 C      -----
0111
0112      CALL THICK(C,U ,ALPSTAR,ITRIP,DELTAP,DSTRS,DSTRP,C0,VISC)
0113
0114 C      COMPUTE DIRECTIVITY FUNCTION
0115 C      -----
0116
0117      CALL DIRECTL(M,THETA,PHI,DBARL)
0118      CALL DIRECTH(M,THETA,PHI,DBARH)
0119
0120
0121 C      CALCULATE THE REYNOLDS NUMBERS BASED ON PRESSURE AND
0122 C      SUCTION DISPLACEMENT THICKNESS
0123 C      -----
0124
0125      RDSTRS = DSTRS * U / VISC
0126      RDSTRP = DSTRP * U / VISC
0127
0128 C      DETERMINE PEAK STROUHAL NUMBERS TO BE USED FOR
0129 C      'A' AND 'B' CURVE CALCULATIONS
0130 C      -----
0131
0132      ST1     = .02 * M ** (-.6)
0133
0134      IF (ALPSTAR .LE. 1.333) ST2 = ST1
0135      IF ((ALPSTAR .GT. 1.333).AND.(ALPSTAR .LE. 12.5))
0136      1  ST2 = ST1*10.**(.0054*(ALPSTAR-1.333)**2.)
0137      IF (ALPSTAR .GT. 12.5) ST2 = 4.72 * ST1
0138
0139
0140      ST1PRIM = (ST1+ST2)/2.
0141
0142
0143      CALL A0COMP(RC,A0)
0144      CALL A0COMP(3.*RC,A02)
0145
0146 C      EVALUATE MINIMUM AND MAXIMUM 'A' CURVES AT A0
0147 C      -----
0148
0149      CALL AMIN(A0,AMINA0)
0150      CALL AMAX(A0,AMAXA0)

```

```

0151
0152      CALL AMIN(A02,AMINA02)
0153      CALL AMAX(A02,AMAXA02)
0154
0155      C      COMPUTE 'A' MAX/MIN RATIO
0156      C      -----
0157
0158      ARA0 = (20. + AMINA0) / (AMINA0 - AMAXA0)
0159      ARA02 = (20. + AMINA02) / (AMINA02 - AMAXA02)
0160
0161      C      COMPUTE B0 TO BE USED IN 'B' CURVE CALCULATIONS
0162      C      -----
0163
0164      IF (RC .LT. 9.52E+04) B0 = .30
0165      IF ((RC .GE. 9.52E+04).AND.(RC .LT. 8.57E+05))
0166 1      B0 = (-4.48E-13)*(RC-8.57E+05)**2. + .56
0167      IF (RC .GE. 8.57E+05) B0 = .56
0168
0169      C      EVALUATE MINIMUM AND MAXIMUM 'B' CURVES AT B0
0170      C      -----
0171
0172      CALL BMIN(B0,BMINB0)
0173      CALL BMAX(B0,BMAXB0)
0174
0175      C      COMPUTE 'B' MAX/MIN RATIO
0176      C      -----
0177
0178      BRB0 = (20. + BMINB0) / (BMINB0 - BMAXB0)
0179
0180      C      FOR EACH CENTER FREQUENCY, COMPUTE AN
0181      C      'A' PREDICTION FOR THE PRESSURE SIDE
0182      C      -----
0183
0184      STPEAK = ST1
0185
0186      DO 100 I=1,NFREQ
0187      STP(I) = FRCEN(I) * DSTRP / U
0188      A      = ALOG10( STP(I) / STPEAK )
0189      CALL AMIN(A,AMINA)
0190      CALL AMAX(A,AMAXA)
0191      AA      = AMINA + ARA0 * (AMAXA - AMINA)
0192
0193      IF (RC .LT. 2.47E+05) K1 = -4.31 * ALOG10(RC) + 156.3
0194      IF((RC .GE. 2.47E+05).AND.(RC .LT. 8.0E+05))
0195 1      K1 = -9.0 * ALOG10(RC) + 181.6
0196      IF (RC .GT. 8.0E+05) K1 = 128.5
0197
0198      IF (RDSTRP .LE. 5000.) DELK1 = -ALPSTAR*(5.29-1.43*
0199 1      ALOG10(RDSTRP))
0200      IF (RDSTRP .GT. 5000.) DELK1 = 0.0
0201
0202      SPLP(I)=AA+K1-3.+10.*ALOG10(DSTRP*M**5.*DBARH*L/R**2.)+DELK1
0203
0204
0205
0206
0207      GAMMA = 27.094 * M + 3.31
0208      BETA  = 72.650 * M + 10.74
0209      GAMMA0 = 23.430 * M + 4.651
0210      BETA0  = -34.190 * M - 13.820
0211
0212      IF (ALPSTAR .LE. (GAMMA0-GAMMA)) K2 = -1000.0
0213      IF ((ALPSTAR.GT.(GAMMA0-GAMMA)).AND.(ALPSTAR.LE.(GAMMA0+GAMMA)))
0214 1      K2=SQRT(BETA**2.-(BETA/GAMMA)**2.*(ALPSTAR-GAMMA0)**2.)+BETA0
0215      IF (ALPSTAR .GT. (GAMMA0+GAMMA)) K2 = -12.0
0216
0217      K2 = K2 + K1
0218
0219
0220
0221      STS(I) = FRCEN(I) * DSTRS / U
0222
0223      C      CHECK FOR 'A' COMPUTATION FOR SUCTION SIDE
0224      C      -----
0225

```

```

0226      XCHECK = GAMMA0
0227      SWITCH = .FALSE.
0228      IF ((ALPSTAR .GE. XCHECK).OR.(ALPSTAR .GT. 12.5))SWITCH=.TRUE.
0229      IF (.NOT. SWITCH) THEN
0230          A      = ALOG10( STS(I) / ST1PRIM )
0231          CALL AMIN(A,AMINA)
0232          CALL AMAX(A,AMAXA)
0233          AA = AMINA + ARA0 * (AMAXA - AMINA)
0234
0235          SPLS(I) = AA+K1-3.+10.*ALOG10(DSTRS*M**5.*DBARH*
0236      1      L/R**2.)
0237
0238      C      'B' CURVE COMPUTATION
0239      C      -----
0240
0241          B = ABS(ALOG10(STS(I) / ST2))
0242          CALL BMIN(B,BMINB)
0243          CALL BMAX(B,BMAXB)
0244          BB = BMINB + BRB0 * (BMAXB-BMINB)
0245          SPLALPH(I)=BB+K2+10.*ALOG10(DSTRS*M**5.*DBARH*L/R**2.)
0246
0247      ELSE
0248
0249      C      THE 'A' COMPUTATION IS DROPPED IF 'SWITCH' IS TRUE
0250      C      -----
0251
0252          SPLS(I) = 0.0 + 10.*ALOG10(DSTRS*M**5.*DBARL*
0253      1      L/R**2.)
0254          SPLP(I) = 0.0 + 10.*ALOG10(DSTRS*M**5.*DBARL*
0255      1      L/R**2.)
0256          B = ABS(ALOG10(STS(I) / ST2))
0257          CALL AMIN(B,AMINB)
0258          CALL AMAX(B,AMAXB)
0259          BB = AMINB + ARA02 * (AMAXB-AMINB)
0260          SPLALPH(I)=BB+K2+10.*ALOG10(DSTRS*M**5.*DBARL*
0261      1      L/R**2.)
0262      1      ENDIF
0263
0264
0265      C      SUM ALL CONTRIBUTIONS FROM 'A' AND 'B' ON BOTH
0266      C      PRESSURE AND SUCTION SIDE ON A MEAN-SQUARE PRESSURE
0267      C      BASIS
0268      C      -----
0269
0270
0271      IF (SPLP(I) .LT. -100.) SPLP(I) = -100.
0272      IF (SPLS(I) .LT. -100.) SPLS(I) = -100.
0273      IF (SPLALPH(I) .LT. -100.) SPLALPH(I) = -100.
0274
0275      P1 = 10.**(SPLP(I) / 10.)
0276      P2 = 10.**(SPLS(I) / 10.)
0277      P4 = 10.**(SPLALPH(I) / 10.)
0278
0279      SPLTBL(I) = 10. * ALOG10(P1 + P2 + P4)
0280
0281      100 CONTINUE
0282
0283      RETURN
0284      END

```

```

0001      SUBROUTINE AMIN(A,AMINA)
0002
0003      C      THIS SUBROUTINE DEFINES THE CURVE FIT CORRESPONDING
0004      C      TO THE A-CURVE FOR THE MINIMUM ALLOWED REYNOLDS NUMBER.
0005      C
0006
0007      X1 = ABS(A)
0008
0009      IF (X1 .LE. .204) AMINA=SQRT(67.552-886.788*X1**2.)-8.219
0010      IF((X1 .GT. .204).AND.(X1 .LE. .244))AMINA=-32.665*X1+3.981
0011      IF (X1 .GT. .244)AMINA=-142.795*X1**3.+103.656*X1**2.-57.757*X1+6.006
0012
0013      RETURN
0014      END

```

```

0001      SUBROUTINE AMAX(A,AMAXA)
0002
0003      C      THIS SUBROUTINE DEFINES THE CURVE FIT CORRESPONDING
0004      C      TO THE A-CURVE FOR THE MAXIMUM ALLOWED REYNOLDS NUMBER.
0005      C
0006
0007      X1 = ABS(A)
0008
0009      IF (X1 .LE. .13)AMAXA=SQRT(67.552-886.788*X1**2.)-8.219
0010      IF((X1 .GT. .13).AND.(X1 .LE. .321))AMAXA=-15.901*X1+1.098
0011      IF (X1 .GT. .321)AMAXA=-4.669*X1**3.+3.491*X1**2.-16.699*X1+1.149
0012
0013      RETURN
0014      END

```

```

0001      SUBROUTINE BMIN(B,BMINB)
0002
0003      C      THIS SUBROUTINE DEFINES THE CURVE FIT CORRESPONDING
0004      C      TO THE B-CURVE FOR THE MINIMUM ALLOWED REYNOLDS NUMBER.
0005      C
0006
0007      X1 = ABS(B)
0008
0009      IF (X1 .LE. .13)BMINB=SQRT(16.888-886.788*X1**2.)-4.109
0010      IF((X1 .GT. .13).AND.(X1 .LE. .145))BMINB=-83.607*X1+8.138
0011      IF (X1 .GT. .145)BMINB=-817.81*X1**3.+355.21*X1**2.-135.024*X1+10.619
0012
0013      RETURN
0014      END

```

```

0001      SUBROUTINE BMAX(B,BMAXB)
0002
0003      C      THIS SUBROUTINE DEFINES THE CURVE FIT CORRESPONDING
0004      C      TO THE B-CURVE FOR THE MAXIMUM ALLOWED REYNOLDS NUMBER.
0005      C
0006
0007      X1 = ABS(B)
0008
0009      IF (X1 .LE. .1) BMAXB=SQRT(16.888-886.788*X1**2.)-4.109
0010      IF((X1 .GT. .1).AND.(X1 .LE. .187))BMAXB=-31.313*X1+1.854
0011      IF (X1 .GT. .187)BMAXB=-80.541*X1**3.+44.174*X1**2.-39.381*X1+2.344
0012
0013      RETURN
0014      END

```

```

0001      SUBROUTINE A0COMP(RC,A0)
0002
0003      C      THIS SUBROUTINE DETERMINES WHERE THE A-CURVE
0004      C      TAKES ON A VALUE OF -20 dB.
0005
0006      IF (RC .LT. 9.52E+04) A0 = .57
0007      IF ((RC .GE. 9.52E+04).AND.(RC .LT. 8.57E+05))
0008      1      A0 = (-9.57E-13)*(RC-8.57E+05)**2. + 1.13
0009      IF (RC .GE. 8.57E+05) A0 = 1.13
0010      RETURN
0011      END

```

```

0001      SUBROUTINE DIRECTH(M,THETA,PHI,DBAR)
0002
0003      C      THIS SUBROUTINE COMPUTES THE HIGH FREQUENCY
0004      C      DIRECTIVITY FUNCTION FOR THE INPUT OBSERVER LOCATION
0005
0006      REAL M,MC
0007
0008      DEGRAD = .017453
0009
0010      MC = .8 * M
0011      THETAR = THETA * DEGRAD
0012      PHIR = PHI * DEGRAD
0013
0014      DBAR=2.*SIN(THETAR/2. )**2.*SIN(PHIR)**2./((1.+M*COS(THETAR)) *
0015      1      (1.+(M-MC)*COS(THETAR))**2.)
0016      RETURN
0017      END

```

```

0001      SUBROUTINE DIRECTL(M,THETA,PHI,DBAR)
0002
0003      C      THIS SUBROUTINE COMPUTES THE LOW FREQUENCY
0004      C      DIRECTIVITY FUNCTION FOR THE INPUT OBSERVER LOCATION
0005
0006      REAL M,MC
0007
0008      DEGRAD = .017453
0009
0010      MC = .8 * M
0011      THETAR = THETA * DEGRAD
0012      PHIR = PHI * DEGRAD
0013
0014      DBAR = (SIN(THETAR)*SIN(PHIR))**2/(1.+M*COS(THETAR))**4
0015
0016      RETURN
0017      END

```

```

0001      SUBROUTINE BLUNT(ALPSTAR,C,U ,FRcen,ITRIP,SPLBLNT,THETA,PHI,
0002 1          L,R,H,PSI,NFREQ,VISC,C0)
0003
0004
0005 C          -----
0006 C          ***** VARIABLE DEFINITIONS *****
0007 C          -----
0008
0009 C          VARIABLE NAME          DEFINITION          UNITS
0010 C          -----          -----          -----
0011
0012 C          ALPSTAR          ANGLE OF ATTACK          DEGREES
0013 C          ATERM          USED TO COMPUTE PEAK STROUHAL NO.          ---
0014 C          C          CHORD LENGTH          METERS
0015 C          C0          SPEED OF SOUND          METERS/SEC
0016 C          DBARH          HIGH FREQUENCY DIRECTIVITY          ---
0017 C          DELTAP          PRESSURE SIDE BOUNDARY LAYER          ---
0018 C          THICKNESS          METERS
0019 C          DSTARH          AVERAGE DISPLACEMENT THICKNESS          ---
0020 C          OVER TRAILING EDGE BLUNTNES          ---
0021 C          DSTRAVG          AVERAGE DISPLACEMENT THICKNESS          METERS
0022 C          DSTRP          PRESSURE SIDE DISPLACEMENT THICKNESS          METERS
0023 C          DSTRS          SUCTION SIDE DISPLACEMENT THICKNESS          METERS
0024 C          ETA          RATIO OF STROUHAL NUMBERS          ---
0025 C          FRcen          ARRAY OF 1/3 OCTAVE CENTERED FREQ.          HERTZ
0026 C          F4TEMP          G5 EVALUATED AT MINIMUM HDSTARP          DB
0027 C          G4          SCALED SPECTRUM LEVEL          DB
0028 C          G5          SPECTRUM SHAPE FUNCTION          DB
0029 C          G50          G5 EVALUATED AT PSI=0.0          DB
0030 C          G514          G5 EVALUATED AT PSI=14.0          DB
0031 C          H          TRAILING EDGE BLUNTNES          METERS
0032 C          HDSTAR          BLUNTNES OVER AVERAGE DISPLACEMENT          ---
0033 C          THICKNESS          ---
0034 C          HDSTARL          MINIMUM ALLOWED VALUE OF HDSTAR          ---
0035 C          HDSTARP          MODIFIED VALUE OF HDSTAR          ---
0036 C          ITRIP          TRIGGER FOR BOUNDARY LAYER TRIPPING          ---
0037 C          L          SPAN          METERS
0038 C          M          MACH NUMBER          ---
0039 C          NFREQ          NUMBER OF CENTERED FREQUENCIES          ---
0040 C          PHI          DIRECTIVITY ANGLE          DEGREES
0041 C          PSI          TRAILING EDGE ANGLE          DEGREES
0042 C          R          SOURCE TO OBSERVER DISTANCE          METERS
0043 C          RC          REYNOLDS NUMBER BASED ON CHORD          ---
0044 C          SCALE          SCALING FACTOR          ---
0045 C          SPLBLNT          SOUND PRESSURE LEVELS DUE TO          ---
0046 C          BLUNTNES          DB
0047 C          STPEAK          PEAK STROUHAL NUMBER          ---
0048 C          STPPP          STROUHAL NUMBER          ---
0049 C          THETA          DIRECTIVITY ANGLE          ---
0050 C          U          FREESTREAM VELOCITY          METERS/SEC
0051 C          VISC          KINEMATIC VISCOSITY          M2/SEC
0052
0053
0054      PARAMETER (MAXFREQ = 27)
0055
0056      DIMENSION SPLBLNT(MAXFREQ) ,FRcen(MAXFREQ) ,STPPP(MAXFREQ)
0057
0058      REAL M,L
0059
0060 C          COMPUTE NECESSARY QUANTITIES
0061 C          -----
0062
0063      M = U /C0
0064      RC = U * C / VISC
0065
0066
0067 C          COMPUTE BOUNDARY LAYER THICKNESSES
0068 C          -----
0069
0070      CALL THICK(C,U ,ALPSTAR,ITRIP,DELTAP,DSTRS,DSTRP,C0,VISC)
0071
0072 C          COMPUTE AVERAGE DISPLACEMENT THICKNESS
0073 C          -----
0074
0075      DSTRAVG = (DSTRS + DSTRP) / 2.

```

```

0076      HDSTAR = H / DSTR AVG
0077
0078      DSTARH = 1. / HDSTAR
0079
0080      C      COMPUTE DIRECTIVITY FUNCTION
0081      C      -----
0082
0083      CALL DIRECTH(M, THETA, PHI, DBARH)
0084
0085
0086      C      COMPUTE PEAK STROUHAL NUMBER
0087      C      -----
0088
0089      ATERM = .212 - .0045 * PSI
0090
0091      IF (HDSTAR .GE. .2)
0092 1      STPEAK = ATERM / (1.+ .235*DSTARH-.0132*DSTARH**2.)
0093      IF (HDSTAR .LT. .2)
0094 1      STPEAK = .1 * HDSTAR + .095 - .00243 * PSI
0095
0096      C      COMPUTE SCALED SPECTRUM LEVEL
0097      C      -----
0098
0099      IF (HDSTAR .LE. 5.) G4=17.5*ALOG10(HDSTAR)+157.5-1.114*PSI
0100      IF (HDSTAR .GT. 5.) G4=169.7 - 1.114 * PSI
0101
0102
0103      C      FOR EACH FREQUENCY, COMPUTE SPECTRUM SHAPE REFERENCED TO 0 DB
0104      C      -----
0105
0106      DO 1000 I=1,NFREQ
0107
0108          STPPP(I) = FRCEN(I) * H / U
0109          ETA      = ALOG10(STPPP(I)/STPEAK)
0110
0111          HDSTARL = HDSTAR
0112
0113          CALL G5COMP(HDSTARL,ETA,G514)
0114
0115          HDSTARP = 6.724 * HDSTAR **2.-4.019*HDSTAR+1.107
0116
0117          CALL G5COMP(HDSTARP,ETA,G50)
0118
0119
0120          G5 = G50 + .0714 * PSI * (G514-G50)
0121          IF (G5 .GT. 0.) G5 = 0.
0122          CALL G5COMP(.25,ETA,F4TEMP)
0123          IF (G5 .GT. F4TEMP) G5 = F4TEMP
0124
0125
0126          SCALE = 10. * ALOG10(M**5.5*H*DBARH*L/R**2.)
0127
0128          SPLBLNT(I) = G4 + G5 + SCALE
0129
0130
0131      1000 CONTINUE
0132
0133      RETURN
0134      END

```



```

0001      SUBROUTINE G5COMP(HDSTAR,ETA,G5)
0002
0003
0004      REAL M,K,MU
0005
0006
0007      IF (HDSTAR .LT. .25) MU = .1211
0008      IF ((HDSTAR .GT. .25).AND.(HDSTAR .LE. .62))
0009 1      MU=-.2175*HDSTAR + .1755
0010      IF ((HDSTAR .GT. .62).AND.(HDSTAR .LT. 1.15))
0011 1 MU = -.0308 * HDSTAR + .0596
0012      IF (HDSTAR .GE. 1.15)MU = .0242
0013
0014      IF (HDSTAR .LE. .02) M = 0.0
0015      IF ((HDSTAR .GE. .02).AND.(HDSTAR .LT. .5))
0016 1      M=68.724*HDSTAR - 1.35
0017      IF ((HDSTAR .GT. .5).AND.(HDSTAR .LE. .62))
0018 1      M = 308.475 * HDSTAR - 121.23
0019      IF ((HDSTAR .GT. .62).AND.(HDSTAR .LE. 1.15))
0020 1      M = 224.811 * HDSTAR - 69.354
0021      IF ((HDSTAR .GT. 1.15) .AND. (HDSTAR .LT. 1.2))
0022 1      M = 1583.28 * HDSTAR - 1631.592
0023      IF (HDSTAR .GT. 1.2) M = 268.344
0024      IF (M .LT. 0.0) M = 0.0
0025
0026      ETA0 = -SQRT((M*M*MU**4)/(6.25+M*M*MU*MU))
0027
0028      K      = 2.5*SQRT(1.-(ETA0/MU)**2.)-2.5-M*ETA0
0029
0030      IF (ETA .LE. ETA0) G5 = M * ETA + K
0031      IF ((ETA .GT. ETA0).AND.(ETA .LE. 0.))G5=2.5*SQRT(1.-(ETA/MU)**2.)-2.5
0032      IF((ETA.GT.0.).AND.(ETA.LE..03616))G5=SQRT(1.5625-1194.99*ETA**2.)-1.25
0033      IF (ETA .GT. .03616) G5=-155.543 * ETA + 4.375
0034
0035      RETURN
0036      END

```

```

0001      SUBROUTINE TIPNOIS(ALPHTIP,ALPRAT,C,U ,FRCEN,SPLTIP,THETA,PHI,
0002 1      R,NFREQ,VISC,C0,ROUND)
0003
0004      C      -----
0005      C      ***** VARIABLE DEFINITIONS *****
0006      C      -----
0007
0008      C      VARIABLE NAME      DEFINITION      UNITS
0009      C      -----      -----      -----
0010
0011      C      ALPHTIP      TIP ANGLE OF ATTACK      DEGREES
0012      C      ALPRAT      TIP LIFT CURVE SLOPE      ---
0013      C      ALPTIPP      CORRECTED TIP ANGLE OF ATTACK      DEGREES
0014      C      C      CHORD LENGTH      METERS
0015      C      C0      SPEED OF SOUND      METERS/SEC
0016      C      DBARH      DIRECTIVITY      ---
0017      C      FRCEN      CENTERED FREQUENCIES      HERTZ
0018      C      L      CHARACTERISTIC LENGTH FOR TIP      METERS
0019      C      M      MACH NUMBER      ---
0020      C      MM      MAXIMUM MACH NUMBER      ---
0021      C      NFREQ      NUMBER OF CENTERED FREQUENCIES      ---
0022      C      PHI      DIRECTIVITY ANGLE      DEGREES
0023      C      R      SOURCE TO OBSERVER DISTANCE      METERS
0024      C      ROUND      LOGICAL SET TRUE IF TIP IS ROUNDED      ---
0025      C      SCALE      SCALING TERM      ---
0026      C      SPLTIP      SOUND PRESSURE LEVEL DUE TO TIP
0027      C      MECHANISM      DB
0028      C      STPP      STROUHAL NUMBER      ---
0029      C      TERM      SCALING TERM      ---
0030      C      THETA      DIRECTIVITY ANGLE      DEGREES
0031      C      U      FREESTREAM VELOCITY      METERS/SEC
0032      C      UM      MAXIMUM VELOCITY      METERS/SEC
0033      C      VISC      KINEMATIC VISCOSITY      M2/SEC
0034
0035      PARAMETER (MAXFREQ =27)
0036
0037      DIMENSION SPLTIP(MAXFREQ),FRCEN(MAXFREQ)
0038      REAL L,M,MM
0039      LOGICAL ROUND
0040
0041
0042      ALPTIPP = ALPHTIP * ALPRAT
0043      M      = U / C0
0044
0045      CALL DIRECTH(M,THETA,PHI,DBARH)
0046
0047      IF (ROUND) THEN
0048          L = .008 * ALPTIPP * C
0049      ELSE
0050          IF (ABS(ALPTIPP) .LE. 2.) THEN
0051              L = (.023 + .0169*ALPTIPP) * C
0052          ELSE
0053              L = (.0378 + .0095*ALPTIPP) * C
0054          ENDIF
0055      ENDIF
0056
0057
0058      MM      = (1. + .036*ALPTIPP) * M
0059
0060      UM      = MM * C0
0061
0062      TERM = M*M*MM**3.*L**2.*DBARH/R**2.
0063      IF (TERM .NE. 0.0) THEN
0064          SCALE = 10.*ALOG10(TERM)
0065      ELSE
0066          SCALE = 0.0
0067      ENDIF
0068
0069
0070      DO 100 I=1,NFREQ
0071          STPP      = FRCEN(I) * L / UM
0072          SPLTIP(I) = 126.-30.5*(ALOG10(STPP)+.3)**2. + SCALE
0073      100 CONTINUE
0074      RETURN
0075      END

```

```

0001      SUBROUTINE THICK(C,U ,ALPSTAR,ITRIP,DELTAP,DSTRS,DSTRP,C0,VISC)
0002
0003      C
0004      C          -----
0005      C          ***** VARIABLE DEFINITIONS *****
0006      C          -----
0007      C          VARIABLE NAME          DEFINITION          UNITS
0008      C          -----          -----          -----
0009
0010      C          ALPSTAR          ANGLE OF ATTACK          DEGREES
0011      C          C          CHORD LENGTH          METERS
0012      C          C0          SPEED OF SOUND          METERS/SEC
0013      C          DELTA0          BOUNDARY LAYER THICKNESS AT
0014      C          C          ZERO ANGLE OF ATTACK          METERS
0015      C          DELTAP          PRESSURE SIDE BOUNDARY LAYER
0016      C          C          THICKNESS          METERS
0017      C          DSTR0          DISPLACEMENT THICKNESS AT ZERO
0018      C          C          ANGLE OF ATTACK          METERS
0019      C          DSTRP          PRESSURE SIDE DISPLACEMENT
0020      C          C          THICKNESS          METERS
0021      C          DSTRS          SUCTION SIDE DISPLACEMENT
0022      C          C          THICKNESS          METERS
0023      C          ITRIP          TRIGGER FOR BOUNDARY LAYER TRIPPING  ---
0024      C          M          MACH NUMBER          ---
0025      C          RC          REYNOLDS NUMBER BASED ON CHORD          ---
0026      C          U          FREESTREAM VELOCITY          METERS/SEC
0027      C          VISC          KINEMATIC VISCOSITY          M2/SEC
0028
0029
0030      C          COMPUTE ZERO ANGLE OF ATTACK BOUNDARY LAYER
0031      C          THICKNESS (METERS) AND REYNOLDS NUMBER
0032      C          -----
0033
0034      M          = U / C0
0035
0036      RC          = U * C/VISC
0037
0038      DELTA0     = 10.**(1.6569-.9045*ALOG10(RC)+
0039      1          .0596*ALOG10(RC)**2.)*C
0040      IF (ITRIP .EQ. 2) DELTA0 = .6 * DELTA0
0041
0042
0043      C          COMPUTE PRESSURE SIDE BOUNDARY LAYER THICKNESS
0044      C          -----
0045
0046      DELTAP     = 10.**(-.04175*ALPSTAR+.00106*ALPSTAR**2.)*DELTA0
0047
0048
0049      C          COMPUTE ZERO ANGLE OF ATTACK DISPLACEMENT THICKNESS
0050      C          -----
0051
0052      IF ((ITRIP .EQ. 1) .OR. (ITRIP .EQ. 2)) THEN
0053          IF (RC .LE. .3E+06) DSTR0 = .0601 * RC **(-.114)*C
0054          IF (RC .GT. .3E+06)
0055      1          DSTR0=10.**(3.411-1.5397*ALOG10(RC)+.1059*ALOG10(RC)**2.)*C
0056          IF (ITRIP .EQ. 2) DSTR0 = DSTR0 * .6
0057      ELSE
0058          DSTR0=10.**(3.0187-1.5397*ALOG10(RC)+.1059*ALOG10(RC)**2.)*C
0059      ENDIF
0060
0061      C          PRESSURE SIDE DISPLACEMENT THICKNESS
0062      C          -----
0063
0064      DSTRP     = 10.**(-.0432*ALPSTAR+.00113*ALPSTAR**2.)*DSTR0
0065      IF (ITRIP .EQ. 3) DSTRP = DSTRP * 1.48
0066
0067      C          SUCTION SIDE DISPLACEMENT THICKNESS
0068      C          -----
0069
0070      IF (ITRIP .EQ. 1) THEN
0071          IF (ALPSTAR .LE. 5.) DSTRS=10.**(.0679*ALPSTAR)*DSTR0
0072          IF((ALPSTAR .GT. 5.) .AND. (ALPSTAR .LE. 12.5))
0073      1          DSTRS = .381*10.**(.1516*ALPSTAR)*DSTR0
0074          IF (ALPSTAR .GT. 12.5)DSTRS=14.296*10.**(.0258*ALPSTAR)*DSTR0
0075      ELSE

```

```
0076         IF (ALPSTAR .LE. 7.5)DSTRS =10.**(.0679*ALPSTAR)*DSTRO
0077         IF((ALPSTAR .GT. 7.5).AND.(ALPSTAR .LE. 12.5))
0078 1         DSTRS = .0162*10.**(.3066*ALPSTAR)*DSTRO
0079         IF (ALPSTAR .GT. 12.5) DSTRS = 52.42*10.**(.0258*ALPSTAR)*DSTRO
0080         ENDIF
0081
0082         RETURN
0083         END
```

References

1. Brooks, Thomas F.; and Schlinker, Robert H.: Progress in Rotor Broadband Noise Research. *Vertica*, vol. 7, no. 4, 1983, pp. 287-307.
2. Brooks, T. F.; and Hodgson, T. H.: Trailing Edge Noise Prediction From Measured Surface Pressures. *J. Sound & Vibration*, vol. 78, no. 1, Sept. 8, 1981, pp. 69-117.
3. Schlinker, Robert H.; and Amiet, Roy K.: *Helicopter Rotor Trailing Edge Noise*. NASA CR-3470, 1981. (See also AIAA Paper 81-2001.)
4. Liu, N. S.; and Shamroth, S. J.: *On the Application of a Hairpin Vortex Model of Wall Turbulence to Trailing Edge Noise Prediction*. NASA CR-177938, 1985.
5. Ffowcs Williams, J. E.; and Hall, L. H.: Aerodynamic Sound Generation by Turbulent Flow in the Vicinity of a Scattering Half Plane. *J. Fluid Mech.*, vol. 40, pt. 4, Mar. 1970, pp. 657-670.
6. Brooks, Thomas F.; and Marcolini, Michael A.: Scaling of Airfoil Self-Noise Using Measured Flow Parameters. *AIAA J.*, vol. 23, no. 2, Feb. 1985, pp. 207-213.
7. Chou, S.-T.; and George, A. R.: Effect of Angle of Attack on Rotor Trailing-Edge Noise. *AIAA J.*, vol. 22, no. 12, Dec. 1984, pp. 1821-1823.
8. George, Albert R.; and Chou, Shau-Tak: *Broadband Rotor Noise Analyses*. NASA CR-3797, 1984.
9. Grosveld, Ferdinand W.: Prediction of Broadband Noise From Horizontal Axis Wind Turbines. *AIAA J. Propuls. & Power*, vol. 1, no. 4, July-Aug. 1985, pp. 292-299.
10. Glegg, S. A. L.; Baxter, S. M.; and Glendinning, A. G.: The Prediction of Broadband Noise From Wind Turbines. *J. Sound & Vibration*, vol. 118, no. 2, Oct. 22, 1987, pp. 217-239.
11. Fink, M. R.; and Bailey, D. A.: *Airframe Noise Reduction Studies and Clean-Airframe Noise Investigation*. NASA CR-159311, 1980.
12. Paterson, Robert W.; Amiet, Roy K.; and Munch, C. Lee: Isolated Airfoil-Tip Vortex Interaction Noise. AIAA Paper No. 74-194, Jan.-Feb. 1974.
13. Tam, Christopher K. W.: Discrete Tones of Isolated Airfoils. *J. Acoust. Soc. America*, vol. 55, no. 6, June 1974, pp. 1173-1177.
14. Wright, S. E.: The Acoustic Spectrum of Axial Flow Machines. *J. Sound & Vibration*, vol. 45, no. 2, Mar. 22, 1976, pp. 165-223.
15. Fink, M. R.: *Fine Structure of Airfoil Tone Frequency*. UTRC78-10, United Technologies Research Center, Jan. 1978.
16. Paterson, Robert W.; Vogt, Paul G.; Fink, Martin R.; and Munch, C. Lee: Vortex Noise of Isolated Airfoils. *J. Aircr.*, vol. 10, no. 5, May 1973, pp. 296-302.
17. George, A. R.; Najjar, F. E.; and Kim, Y. N.: Noise Due to Tip Vortex Formation on Lifting Rotors. AIAA-80-1010, June 1980.
18. Brooks, Thomas F.; and Marcolini, Michael A.: Airfoil Tip Vortex Formation Noise. *AIAA J.*, vol. 24, no. 2, Feb. 1986, pp. 246-252.
19. Blake, William K.: *Aero-Hydroacoustics for Ships, Volume II*. Rep. No. DTNSRDC-84/010, U.S. Navy, June 1984.
20. Chou, S.-T.; and George, A. R.: Effect of Blunt Trailing Edge on Rotor Broadband Noise. *AIAA J.*, vol. 24, no. 8, Aug. 1986, pp. 1380-1382.
21. Brooks, Thomas F.; Marcolini, Michael A.; and Pope, Dennis S.: Airfoil Trailing-Edge Flow Measurements. *AIAA J.*, vol. 24, no. 8, Aug. 1986, pp. 1245-1251.
22. Amiet, R. K.: Refraction of Sound by a Shear Layer. *J. Sound & Vibration*, vol. 58, no. 4, June 1978, pp. 467-482.
23. Schlinker, R. H.; and Amiet, R. K.: Refraction of Sound by a Shear Layer—Experimental Assessment. AIAA Paper 79-0628, Mar. 1979.
24. Paterson, Robert W.; and Amiet, Roy K.: *Acoustic Radiation and Surface Pressure Characteristics of an Airfoil Due to Incident Turbulence*. NASA CR-2733, 1976. (See also AIAA Paper 76-571.)
25. Fink, Martin R.: Noise Component Method for Airframe Noise. AIAA Paper 77-1271, Oct. 1977.
26. Schlinker, R. H.: Airfoil Trailing Edge Noise Measurements With a Directional Microphone. AIAA Paper 77-1269, Oct. 1977.
27. Fink, M. R.; Schlinker, R. H.; and Amiet, R. K.: *Prediction of Rotating-Blade Vortex Noise From Noise of Non-rotating Blades*. NASA CR-2611, 1976.
28. Bendat, Julius S.; and Piersol, Allan G.: *Random Data: Analysis and Measurement Procedures*. John Wiley & Sons, Inc., c.1971.
29. Dowling, A. P.; and Ffowcs Williams, J. E.: *Sound and Sources of Sound*. John Wiley & Sons, c.1983.
30. Brooks, Thomas F.; Marcolini, Michael A.; and Pope, D. Stuart: Main Rotor Broadband Noise Study in the DNW. *J. American Helicopter Soc.*, vol. 34, no. 2, Apr. 1989, pp. 3-12.
31. Golub, R. A.; Weir, D. S.; and Tracy, M. B.: Application of the Baseline Rotonet System to the Prediction of Helicopter Tone Noise. AIAA-86-1904, July 1986.

Table 1. Test and Data Processing Conditions for Tripped Boundary-Layer Airfoil Cases of Figures 11 to 43

Fig. no.	Run no.	Chord, C, cm	Tunnel angle, α_T , deg	Effective angle, α^* , deg	Tunnel velocity, U, m/s	Reynolds number, $R_C \times 10^6$	Calculated ratio, δ^*/c	Peak Strouhal no., St _{peak} (fig. 76)	TBL-TTE peak scaled SPL 1/3, dB (fig. 77)	Data processing Microphones No. sep. Fold
11(a)	101	30.48	0.0	0.0	71.3	1.497	0.0087	0.046	128.5	4x5
11(b)	102	30.48	0.0	0.0	55.5	1.164	0.0093	0.056	128.0	4x8
11(c)	103	30.48	0.0	0.0	39.6	0.831	0.0102	0.065	128.5	4x8
11(d)	104	30.48	0.0	0.0	31.7	0.665	0.0109	0.082	128.5	4x8
12(a)	105	30.48	5.4	1.5	71.3	1.497	0.0111	0.050	128.5	4x8
12(b)	106	30.48	5.4	1.5	39.6	0.831	0.0129	0.076	128.0	4x8
13(a)	107	30.48	10.8	3.0	71.3	1.497	0.0140	0.050	129.0	4x8
13(b)	108	30.48	10.8	3.0	55.5	1.164	0.0149	0.069	129.0	4x8
13(c)	109	30.48	10.8	3.0	39.6	0.831	0.0163	0.101	129.0	4x8
13(d)	110	30.48	10.8	3.0	31.7	0.665	0.0174	0.125	128.0	4x8
14(a)	111	30.48	14.4	4.0	71.3	1.497	0.0164	0.057	131.5	4x2
14(b)	112	30.48	14.4	4.0	39.6	0.831	0.0191	0.103	130.5	4x2
15(a)	251	22.86	0.0	0.0	55.5	1.122	0.0094	0.061	128.5	4x8
15(b)	252	22.86	0.0	0.0	39.6	0.873	0.0100	0.069	127.5	4x8
15(c)	253	22.86	0.0	0.0	31.7	0.624	0.0111	0.073	130.0	4x8
15(d)	254	22.86	0.0	0.0	24.8	0.499	0.0119	0.086	130.0	4x8
16(a)	255	22.86	5.4	2.0	71.3	1.122	0.0128	0.067	128.0	4x8
16(b)	256	22.86	5.4	2.0	55.5	0.873	0.0137	0.061	128.5	4x8
16(c)	257	22.86	5.4	2.0	39.6	0.624	0.0152	0.083	128.0	4x8
16(d)	258	22.86	5.4	2.0	31.7	0.499	0.0163	0.110	128.5	4x8
17(a)	259	22.86	10.8	4.0	71.3	1.122	0.0175	0.068	131.0	4x8
17(b)	260	22.86	10.8	4.0	55.5	0.873	0.0187	0.075	131.5	4x8
17(c)	261	22.86	10.8	4.0	39.6	0.624	0.0207	0.099	129.5	4x8
17(d)	262	22.86	10.8	4.0	31.7	0.499	0.0223	0.138	125.5	4x8
18(a)	263	22.86	14.4	5.3	71.3	1.122	0.0229	0.081	133.0	4x2
18(b)	264	22.86	14.4	5.3	39.6	0.624	0.0271	0.085	131.5	4x2
19(a)	265	22.86	19.8	7.3	71.3	1.122	0.0461	0.050	140.0	4x2
19(b)	266	22.86	19.8	7.3	55.5	0.873	0.0493	0.071	135.5	4x2
19(c)	267	22.86	19.8	7.3	39.6	0.624	0.0545	0.120	131.5	4x2
19(d)	268	22.86	19.8	7.3	31.7	0.499	0.0586	0.157	129.5	4x2
20(a)	133	15.24	0.0	0.0	71.3	0.748	0.0105	0.058	128.0	4x8
20(b)	134	15.24	0.0	0.0	55.5	0.582	0.0113	0.065	128.5	4x8
20(c)	135	15.24	0.0	0.0	39.6	0.416	0.0127	0.070	130.0	4x8
20(d)	136	15.24	0.0	0.0	31.7	0.333	0.0137	0.087	130.5	4x8
21(a)	137	15.24	5.4	2.7	71.3	0.748	0.0160	0.057	129.5	4x8
21(b)	138	15.24	5.4	2.7	39.6	0.416	0.0193	0.094	132.5	4x8
22(a)	139	15.24	10.8	5.4	71.3	0.748	0.0263	0.075	132.5	4x8
22(b)	140	15.24	10.8	5.4	55.5	0.582	0.0284	0.081	132.0	4x8
22(c)	141	15.24	10.8	5.4	39.6	0.416	0.0318	0.091	131.0	4x8
22(d)	142	15.24	10.8	5.4	31.7	0.333	0.0345	0.109	130.0	4x8
23(a)	143	15.24	14.4	7.2	71.3	0.748	0.0493	0.086	134.5	4x2
23(b)	144	15.24	14.4	7.2	39.6	0.416	0.0596	0.130	132.5	4x2
24(a)	145	15.24	19.8	9.9	71.3	0.748	0.1268	0.084	134.0	4x2
24(b)	146	15.24	19.8	9.9	55.5	0.582	0.1368	0.105	131.5	4x2
24(c)	147	15.24	19.8	9.9	39.6	0.416	0.1531	0.205	128.0	4x2
25(a)	148	15.24	25.2	12.6	71.3	0.748	0.1659	0.301	129.5	4x2
25(b)	149	15.24	25.2	12.6	39.6	0.416	0.3170	0.170	128.5	4x2
26(a)	293	10.16	0.0	0.0	71.3	0.499	0.3832	1.131	116.5	4x8
26(b)	294	10.16	0.0	0.0	55.5	0.388	0.0130	0.045	132.5	4x8
26(c)	295	10.16	0.0	0.0	39.6	0.277	0.0144	0.071	132.5	4x8
26(d)	296	10.16	0.0	0.0	31.7	0.222	0.0148	0.077	133.0	4x8
27(a)	297	10.16	5.4	3.3	71.3	0.499	0.0201	0.060	133.0	4x8
27(b)	298	10.16	5.4	3.3	55.5	0.388	0.0219	0.074	132.5	4x8
27(c)	299	10.16	5.4	3.3	39.6	0.277	0.0243	0.092	132.5	4x8
27(d)	300	10.16	5.4	3.3	31.7	0.222	0.0249	0.120	127.5	4x8
28(a)	301	10.16	10.8	6.7	71.3	0.499	0.0470	0.086	136.5	4x8

Table 1. Concluded

Fig. no.	Rm no.	Chord, C, Cm	Tunnel angle, α_t' , deg	Effective angle, α_e' , deg	Tunnel velocity, U, m/s	Reynolds number, $R_C \times 10^{-6}$	Calculated ratio, δ^*/C	Strouhal no., St peak (fig. 76)	TEL-TE peak scaled SPL 1/3, dB (fig. 77)	Data processing Microphones No. sep. Fold	
28(b)	302	10.16	10.8	6.7	55.5	0.388	0.0513	0.102	136.0	4x5	F
28(c)	303	10.16	10.8	6.7	39.6	0.277	0.0568	0.125	133.5	4x8	F
28(d)	304	10.16	10.8	6.7	31.7	0.222	0.0583	0.156	131.5	4x8	F
29(a)	305	10.16	14.4	8.9	0.499	0.1024	0.085	0.085	138.0	4x2	NF
29(b)	306	10.16	19.8	12.3	39.6	0.499	0.1238	0.130	138.5	4x2	NF
30(a)	307	10.16	19.8	12.3	55.5	0.388	0.3296	0.209	138.5	4x2	NF
30(b)	308	10.16	19.8	12.3	0.277	0.3594	0.220	0.265	129.0	4x2	NF
30(c)	310	10.16	19.8	12.3	0.222	0.4087	0.4309	0.341	125.0	4x2	NF
31(a)	311	10.16	25.2	15.6	0.499	0.277	0.5209	0.192	132.5	4x2	NF
31(b)	312	10.16	25.2	15.6	0.277	0.5209	0.330	0.330	123.5	4x2	NF
32(a)	179	5.08	0.0	0.0	0.249	0.046	0.044	0.044	133.5	4x8	F
32(b)	180	5.08	0.0	0.0	0.194	0.046	0.046	0.046	134.0	4x8	F
32(c)	181	5.08	0.0	0.0	0.139	0.046	0.046	0.057	134.0	4x8	F
33(a)	182	5.08	0.0	0.0	0.139	0.111	0.111	0.068	134.5	4x8	F
33(b)	183	5.08	5.4	4.2	0.249	0.082	0.082	0.077	134.5	4x8	F
33(c)	184	5.08	5.4	4.2	0.139	0.082	0.082	0.076	133.0	4x8	F
34(a)	185	5.08	10.8	8.4	0.249	0.301	0.301	0.109	133.0	4x8	F
34(b)	186	5.08	10.8	8.4	0.139	0.1051	0.1051	0.156	134.0	4x8	F
34(c)	187	5.08	10.8	8.4	0.139	0.1682	0.1682	0.164	130.0	4x8	F
35(a)	188	5.08	10.8	8.4	0.139	0.124	0.124	0.205	130.0	4x8	F
35(b)	189	5.08	14.4	11.2	0.111	0.111	0.111	0.236	129.5	4x2	F
35(c)	190	5.08	14.4	11.2	0.249	0.2801	0.2801	0.220	135.5	4x2	F
36(a)	191	5.08	19.8	15.4	0.249	0.295	0.295	0.412	131.5	4x2	F
36(b)	192	5.08	19.8	15.4	0.139	0.5215	0.5215	0.42	128.5	4x2	F
36(c)	193	5.08	19.8	15.4	0.139	0.5367	0.5367	0.483	128.5	4x2	F
37(a)	194	5.08	25.2	19.7	0.111	0.577	0.577	0.621	129.0	4x2	F
37(b)	195	5.08	25.2	19.7	0.249	0.572	0.572	0.741	127.0	4x2	F
38(a)	213	2.54	0.0	0.0	0.139	0.668	0.668	0.294	127.0	4x2	NF
38(b)	214	2.54	0.0	0.0	0.125	0.7162	0.7162	0.318	129.5	4x8	NF
38(c)	215	2.54	0.0	0.0	0.097	0.0158	0.0158	0.051	133.0	4x8	F
39(a)	216	2.54	0.0	0.0	0.069	0.0162	0.0162	0.072	135.5	4x8	F
39(b)	217	2.54	5.4	4.8	0.125	0.0173	0.0173	0.063	137.5	4x8	F
39(c)	218	2.54	5.4	4.8	0.097	0.0332	0.0332	0.060	134.0	4x8	F
40(a)	219	2.54	5.4	4.8	0.069	0.0341	0.0341	0.087	134.5	4x8	F
40(b)	220	2.54	5.4	4.8	0.069	0.0355	0.0355	0.116	134.5	4x8	F
40(c)	221	2.54	10.8	9.5	0.055	0.0364	0.0364	0.131	134.5	4x8	F
41(a)	222	2.54	10.8	9.5	0.125	0.1658	0.1658	0.164	135.5	4x8	F
41(b)	223	2.54	10.8	9.5	0.097	0.1707	0.1707	0.190	133.5	4x8	F
41(c)	224	2.54	10.8	9.5	0.069	0.1773	0.1773	0.248	130.5	4x8	F
42(a)	225	2.54	14.4	12.7	0.125	0.1819	0.1819	0.209	130.5	4x8	F
42(b)	226	2.54	14.4	12.7	0.055	0.4787	0.4787	0.356	133.5	4x2	F
42(c)	227	2.54	14.4	12.7	0.069	0.5119	0.5119	0.501	129.0	4x2	F
43(a)	228	2.54	19.8	17.4	0.225	0.6349	0.6349	0.493	129.0	4x2	F
43(b)	229	2.54	19.8	17.4	0.097	0.6533	0.6533	0.518	126.5	4x2	NF
43(c)	230	2.54	19.8	17.4	0.069	0.6789	0.6789	0.413	126.5	4x2	F
43(d)	231	2.54	25.2	22.2	0.055	0.694	0.694	0.402	136.0	4x2	F
43(e)	232	2.54	25.2	22.2	0.125	0.8420	0.8420	0.356	136.0	4x2	F
43(f)	233	2.54	25.2	22.2	0.069	0.9003	0.9003	0.413	135.0	4x2	F

Table 2. Test and Data Processing Conditions for Untripped Boundary-Layer Airfoil Cases of Figures 44 to 74

Fig. no.	Run no.	Chord, C, cm	Tunnel angle, α_t , deg	Effective angle, α_e , deg	Tunnel velocity, U, m/s	Reynolds number, $R_C \times 10^{-6}$	Calculated ratio, δ_p/c	Peak Strouhal no., St peak (fig. 86)	IBL-VS peak scaled level, dB (fig. 88)	Data processing Microphones No. sep. Fold
44(a)	113	30.48	0.0	0.0	71.3	1.497	0.0221	0.235	136.5	4x5
44(b)	114	30.48	0.0	0.0	55.5	1.164	0.0231	0.225	138.0	4x8
44(c)	115	30.48	0.0	0.0	31.7	0.831	0.0246	0.210	147.5	4x8
44(d)	116	30.48	0.0	0.0	31.7	0.665	0.0258	0.205	154.5	4x8
45(a)	117	30.48	5.4	1.5	71.3	1.497	0.0192	0.200	154.0	4x5
45(b)	118	30.48	10.8	3.0	39.6	0.831	0.0169	0.210	147.5	4x8
46(a)	119	30.48	10.8	3.0	55.5	1.164	0.0176	0.210	147.5	4x8
46(b)	120	30.48	10.8	3.0	39.6	0.831	0.0188	0.205	154.5	4x8
46(c)	121	30.48	10.8	3.0	31.7	0.665	0.0197	0.210	148.0	4x8
47(a)	122	30.48	14.4	4.0	71.3	1.497	0.0156	0.210	148.0	4x2
47(b)	123	30.48	14.4	4.0	39.6	0.831	0.0174	0.200	154.0	4x2
48(a)	124	30.48	14.4	4.0	55.5	1.122	0.0232	0.310	126.5	4x5
48(b)	125	22.86	0.0	0.0	71.3	0.873	0.0244	0.220	135.0	4x8
48(c)	126	22.86	0.0	0.0	55.5	0.624	0.0261	0.205	145.0	4x8
49(a)	127	22.86	5.4	2.0	31.7	0.499	0.0274	0.215	150.5	4x8
49(b)	128	22.86	5.4	2.0	71.3	1.122	0.0194	0.200	148.0	4x8
49(c)	129	22.86	5.4	2.0	55.5	0.873	0.0203	0.185	152.5	4x8
50(a)	130	22.86	10.8	4.0	31.7	0.499	0.0229	0.200	148.0	4x8
50(b)	131	22.86	10.8	4.0	71.3	1.122	0.0165	0.165	154.5	4x2
50(c)	132	22.86	10.8	4.0	55.5	0.873	0.0173	0.185	150.5	4x2
51(a)	133	22.86	14.4	5.3	31.7	0.499	0.0194	0.200	148.0	4x2
51(b)	134	22.86	14.4	5.3	71.3	1.122	0.0149	0.165	154.5	4x2
51(c)	135	22.86	14.4	5.3	39.6	0.624	0.0168	0.185	150.5	4x2
52(a)	136	22.86	19.8	7.3	71.3	1.122	0.0131	0.265	136.0	4x2
52(b)	137	22.86	19.8	7.3	39.6	0.624	0.0147	0.205	144.0	4x2
52(c)	138	22.86	19.8	7.3	55.5	0.748	0.0251	0.190	147.0	4x2
53(a)	139	15.24	0.0	0.0	71.3	0.582	0.0265	0.170	153.0	4x8
53(b)	140	15.24	0.0	0.0	55.5	0.416	0.0286	0.180	142.5	4x8
53(c)	141	15.24	0.0	0.0	31.7	0.333	0.0302	0.265	136.0	4x8
54(a)	142	15.24	5.4	2.7	71.3	0.748	0.0197	0.205	147.0	4x8
54(b)	143	15.24	5.4	2.7	39.6	0.416	0.0225	0.190	159.0	4x8
54(c)	144	15.24	5.4	2.7	71.3	0.748	0.0161	0.170	153.0	4x8
55(a)	145	15.24	10.8	5.4	55.5	0.582	0.0183	0.180	142.5	4x8
55(b)	146	15.24	10.8	5.4	39.6	0.416	0.0193	0.265	136.0	4x8
55(c)	147	15.24	10.8	5.4	71.3	0.333	0.0143	0.205	147.0	4x8
56(a)	148	15.24	14.4	7.2	39.6	0.416	0.0162	0.190	159.0	4x2
56(b)	149	15.24	14.4	7.2	71.3	0.748	0.0123	0.170	153.0	4x2
57(a)	150	15.24	19.8	9.9	71.3	0.748	0.0123	0.290	132.0	4x2
57(b)	151	15.24	19.8	9.9	39.6	0.416	0.0140	0.300	138.5	4x2
57(c)	152	15.24	19.8	9.9	71.3	0.748	0.0110	0.240	151.0	4x2
58(a)	153	15.24	25.2	12.6	39.6	0.416	0.0125	0.210	159.5	4x2
58(b)	154	15.24	25.2	12.6	71.3	0.499	0.0274	0.180	142.5	4x2
58(c)	155	15.24	25.2	12.6	55.5	0.388	0.0291	0.290	132.0	4x2
59(a)	156	10.16	0.0	0.0	71.3	0.277	0.0316	0.300	138.5	4x8
59(b)	157	10.16	0.0	0.0	55.5	0.222	0.0335	0.240	151.0	4x8
59(c)	158	10.16	0.0	0.0	31.7	0.199	0.0204	0.180	161.5	4x8
60(a)	159	10.16	5.4	3.3	71.3	0.499	0.0204	0.180	161.5	4x8
60(b)	160	10.16	5.4	3.3	55.5	0.388	0.0217	0.185	159.5	4x8
60(c)	161	10.16	5.4	3.3	39.6	0.277	0.0235	0.190	157.0	4x8
61(a)	162	10.16	10.8	6.7	71.3	0.499	0.0249	0.220	147.0	4x8
61(b)	163	10.16	10.8	6.7	55.5	0.388	0.0161	0.290	132.0	4x8
61(c)	164	10.16	10.8	6.7	39.6	0.277	0.0170	0.300	138.5	4x8
62(a)	165	10.16	14.4	9.9	71.3	0.499	0.0185	0.240	151.0	4x8
62(b)	166	10.16	14.4	9.9	55.5	0.388	0.0195	0.180	161.5	4x8
62(c)	167	10.16	14.4	9.9	31.7	0.222	0.0141	0.190	157.0	4x8

Table 2. Concluded

Fig. no.	Run no.	Chord C, cm	Tunnel angle, α_t' , deg	Effective angle, α_e' , deg	Tunnel velocity, U, m/s	Reynolds number, $R_C \times 10^{-6}$	Calculated Strouhal no., St peak (fig. 86)	IBL-VS peak scaled level, dB (fig. 88)	Data processing Microphones No. sep. Fold
62(b)	326	10.16	14.4	8.9	39.6	0.277	0.0163	4x5	F
63(a)	327	10.16	19.8	12.3	71.3	0.499	0.0122	4x2	NF
63(b)	328	10.16	19.8	12.3	39.6	0.277	0.0140	4x2	NF
64(a)	329	10.16	25.2	15.6	71.3	0.499	0.0111	4x2	NF
64(b)	330	10.16	25.2	15.6	39.6	0.277	0.0128	4x2	NF
65(a)	197	5.08	0.0	0.0	71.3	0.249	0.0325	4x5	F
65(b)	198	5.08	0.0	0.0	55.5	0.194	0.0347	4x8	NF
65(c)	199	5.08	0.0	0.0	39.6	0.139	0.0382	4x8	NF
66(a)	201	5.08	5.4	4.2	71.3	0.111	0.0408	4x8	NF
66(b)	202	5.08	5.4	4.2	39.6	0.249	0.0226	4x8	NF
66(c)	203	5.08	10.8	8.4	71.3	0.249	0.0266	4x8	NF
67(a)	204	5.08	10.8	8.4	55.5	0.194	0.0172	4x8	F
67(b)	205	5.08	10.8	8.4	39.6	0.139	0.0202	4x8	F
67(c)	206	5.08	10.8	8.4	71.3	0.111	0.0216	4x8	F
68(a)	207	5.08	14.4	11.2	71.3	0.249	0.0150	4x2	F
68(b)	208	5.08	14.4	11.2	39.6	0.139	0.0176	4x2	F
69(a)	209	5.08	19.8	15.4	71.3	0.249	0.0132	4x2	F
69(b)	210	5.08	19.8	15.4	39.6	0.139	0.0155	4x2	F
70(a)	211	5.08	25.2	19.7	71.3	0.249	0.0126	4x2	F
70(b)	212	5.08	25.2	19.7	39.6	0.139	0.0148	4x2	NF
71(a)	233	2.54	0.0	0.0	71.3	0.125	0.0394	4x5	NF
71(b)	234	2.54	0.0	0.0	55.5	0.097	0.0425	4x8	NF
71(c)	235	2.54	0.0	0.0	39.6	0.069	0.0473	4x8	NF
72(a)	236	2.54	5.4	4.8	71.3	0.055	0.0510	4x8	NF
72(b)	237	2.54	5.4	4.8	55.5	0.125	0.0264	4x8	NF
72(c)	238	2.54	5.4	4.8	39.6	0.097	0.0284	4x8	F
73(a)	239	2.54	10.8	9.5	71.3	0.069	0.0317	4x8	F
73(b)	241	2.54	10.8	9.5	55.5	0.125	0.0341	4x8	F
73(c)	242	2.54	10.8	9.5	39.6	0.097	0.0197	4x8	F
74(a)	243	2.54	14.4	12.7	71.3	0.069	0.0213	4x8	F
74(b)	244	2.54	14.4	12.7	55.5	0.125	0.0237	4x8	F
74(c)	245	2.54	14.4	12.7	39.6	0.069	0.0255	4x8	F
74(d)	246	2.54	14.4	12.7	71.3	0.125	0.0172	4x2	F
74(e)	246	2.54	14.4	12.7	39.6	0.069	0.0207	4x2	F



Report Documentation Page

1. Report No. NASA RP-1218	2. Government Accession No.	3. Recipient's Catalog No.	
4. Title and Subtitle Airfoil Self-Noise and Prediction		5. Report Date July 1989	6. Performing Organization Code
		8. Performing Organization Report No. L-16528	
7. Author(s) Thomas F. Brooks, D. Stuart Pope, and Michael A. Marcolini		10. Work Unit No. 505-63-51-06	
9. Performing Organization Name and Address NASA Langley Research Center Hampton, VA 23665-5225		11. Contract or Grant No.	
		13. Type of Report and Period Covered Reference Publication	
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546-0001		14. Sponsoring Agency Code	
		15. Supplementary Notes Thomas F. Brooks and Michael A. Marcolini: Langley Research Center, Hampton, Virginia. D. Stuart Pope: PRC Kentron, Inc., Aerospace Technologies Division, Hampton, Virginia.	
16. Abstract A prediction method is developed for the self-generated noise of an airfoil blade encountering smooth flow. The prediction methods for the individual self-noise mechanisms are semiempirical and are based on previous theoretical studies and data obtained from tests of two- and three-dimensional airfoil blade sections. The self-noise mechanisms are due to specific boundary-layer phenomena, that is, the boundary-layer turbulence passing the trailing edge, separated-boundary-layer and stalled flow over an airfoil, vortex shedding due to laminar-boundary-layer instabilities, vortex shedding from blunt trailing edges, and the turbulent vortex flow existing near the tip of lifting blades. The predictions are compared successfully with published data from three self-noise studies of different airfoil shapes. An application of the prediction method is reported for a large-scale-model helicopter rotor, and the predictions compared well with experimental broadband noise measurements. A computer code of the method is given.			
17. Key Words (Suggested by Authors(s)) Airframe noise Helicopter rotor acoustics Rotor broadband noise Propeller noise Wind turbine noise Trailing-edge noise Vortex-shedding noise		18. Distribution Statement Unclassified—Unlimited Subject Category 71	
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 142	22. Price A07