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TRANSONIC INTERFERENCE EFFECTS IN TESTING OF OSCILLATING
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The Ohio State University

PH.D. 1982

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TRANSONIC INTERFERENCE EFFECTS IN
TESTING OF OSCILLATING AIRFOILS

DISSERTATION

Presented in Partial Fulfillment of the Requirements
for the Degree of Doctor of Philosophy in the
Graduate School of the Ohio State University

By

James A. Davis, BSME, MSME

The Ohio State University

1982

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2. 0.3-Scale Open Port ALL Turret Wind Tunnel Test Results;
AFWL-TR-73-17, March 1973.
3. Measurement and Analysis of Temperature Data on a Segmented Window for the Airborne Pointing and Tracking (APT) System,
AFWL-TN-73-LRL-1, May 1973.
4. "Acoustic Phenomenon of Open Cavity Airborne Cassegrainian Telescopes," AIAA Paper No. 74-195, 1 February 1974, 12th Aerospace Sciences Meeting, Washington, D.C.
5. Investigation of the 0.3-Scale Open Port APT Turret at NASA Ames 141: Phase 11; AFWL-TR-74-316, December 1974.
6. Airborne Laser Laboratory NKC-135A Flight Test Pressure Data for the Forward Ramp; AFWL-TR-75-82, December 1974.

7. "0.3-Scale APT Tests at NASA Ames 14-Foot Transonic Wind Tunnel," Laser Digest; AFWL-TR-74-241, September 1974.
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10. "Unsteady Transonic Aerodynamics"; Paper presented at the Symposium on Airfoils and Aviation, Ohio State University, 19 May 1978.
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TABLE OF CONTENTS

	Page
ACKNOWLEDGEMENTS.....	ii
VITA.....	iii
LIST OF TABLES.....	vii
LIST OF FIGURES.....	viii
CHAPTER	
I. INTRODUCTION.....	1
1.1 Historical Perspective of General Flutter Problems...	6
1.2 History of Oscillating Airfoil Work.....	7
1.2.1 Unsteady Analytical Efforts.....	7
1.2.2 Unsteady Experimental Work.....	11
1.3 Current Experimental Work at OSU/AARL.....	17
1.3.1 Test Description.....	18
1.3.2 Test Objectives.....	18
II. OSU/AARL EXPERIMENTAL APPROACH.....	20
2.1 Test Facility Requirements.....	20
2.2 Model Descriptions.....	54
2.3 Data Acquisition and Run Procedures.....	68
2.4 Unsteady Pressure Data Accuracy.....	74
III. ANALYTICAL BACKGROUND.....	77
3.1 Formulation of the Small-Disturbance Equation.....	77
3.2 Formulation of Boundary Conditions.....	88
3.3 Steady-State Interference.....	95

	Page
3.4 Unsteady Interference Effects.....	101
IV. STEADY-STATE DATA.....	111
4.1 Steady-State Pressure Profiles.....	113
4.2 Quasi-Steady Pressure Amplitudes.....	126
V. DISCUSSION OF UNSTEADY DATA.....	131
5.1 Typical Unsteady Pressure Plots.....	132
5.2 Power-Spectral-Density Data.....	150
5.3 Correlation Plots.....	175
5.4 Oscillating Airfoil Pressure Coefficients.....	183
5.5 Unsteady Pressure Data From the Wind Tunnel Walls....	213
VI. CONCLUSIONS AND RECOMMENDATIONS.....	222
APPENDICES	
A. UNSTEADY DATA ACQUISTIION ROUTINES AND PROCEDURES.....	231
B. ANALYSIS TECHNIQUES FOR UNSTEADY AERODYNAMIC DATA.....	246
C. TUBE-VOLUME RESPONSE ANALYSIS.....	277
D. UNSTEADY SCALING RELATIONS.....	285
E. DETAILED ERROR ANALYSIS.....	291
BIBLIOGRAPHY.....	309

LIST OF TABLES

TABLE	TITLE	PAGE
1.	GENERAL FLUID FLOW EQUATIONS	79
2.	COMPUTATIONAL COMPARISON OF DIRECT VS. FFT ALGORITHMS	249
3.	DATA WINDOWS	258
4.	SCALING PARAMETERS	287
5.	TYPICAL RUN CONDITIONS	290

LIST OF FIGURES

FIGURE	TITLE	PAGE
F-1	Transonic Flutter Speed Dip	4
F-2	AARL 6 x 22 Airfoil Tunnel Schematic and Operating Envelope	22
F-3	Unsteady Pressure Survey of 6" x 22" Tunnel	34
F-4	Static Probe PSD, Open Walls, $M = 0.60$	35
F-5	Lower Plenum PSD, Open Walls, $M = 0.60$	36
F-6	Static Probe PSD, Open Walls, $M = 0.69$	38
F-7	Static Probe PSD, Taped Walls, $M = 0.61$	41
F-8	Effects of Wall Taping on Static Probe Auto-Correlation, $M = 0.61$	43
F-9	$C_{p_{rms}}$ vs Mach at Constant Reynolds Number	45
F-10	$C_{p_{rms}}$ vs Mach at Constant Stagnation Pressure	45
F-11	Comparison of Unsteady Pressures in Various Transonic Facilities	48
F-12	Original Chamber (4 Screens)	51
F-13	New Chamber (10 Screens)	51
F-14	Total Probe Fluctuations During Settling Chamber Modifications ($C_{p_{rms}}$ vs Mach)	52
F-15	Total and Static Transducer Probes	55
F-16	Static Probe and Tunnel Transducer Installations	57
F-17	10° Cone Model	58
F-18	NACA 64A010 Airfoil Model (3-Inch Chord)	60
F-19	NACA 64A010 Airfoil Model (6-Inch Chord)	61
F-20	NACA 64A010 Model Photographs (6-Inch Chord)	63

F-21	Unsteady Pressure Locations	PAGE 64
F-22	Model Drive System	66
F-23	Unsteady Instrumentation Schematic	70
F-24	6" x 22" Tunnel Test Section	107
F-25	Tunnel Resonance Frequencies	108
F-26	Critical Pressure Coefficient	112
F-27	Steady Pressure Distribution, $M = 0.35$	114
F-28	Steady Pressure Distribution, $M = 0.52$	114
F-29	Steady Pressure Distribution, $M = 0.70$	116
F-30	Steady Pressure Distribution, $M = 0.80$	116
F-31	Steady Pressure Distribution, $M = 0.84$	119
F-32	Steady Pressure Distribution, $M = 0.89$	119
F-33	Comparison of Theoretical and Experimental Steady Pressure Distributions	121
F-34	Lift Coefficient vs Pitch Angle	123
F-35	Lift Coefficient vs Pitching Moment Coefficient	123
F-36	Drag Polars	125
F-37	Quasi-Steady Pressure Distribution, $M = 0.35$	127
F-38	Quasi-Steady Pressure Distribution, $M = 0.52$	127
F-39	Quasi-Steady Pressure Distribution, $M = 0.70$	127
F-40	Quasi-Steady Pressure Distribution, $M = 0.80$	127
F-41	Quasi-Steady Pressure Distribution, $M = 0.84$	127
F-42	Quasi-Steady Pressure Distribution, $M = 0.89$	127
F-43	Unsteady Pressure Trace, $M = 0.35$, $\alpha = 0^\circ$, 15.2 cm Model, a) 45 percent x/C ; b) 90 percent x/C	134
F-44	Unsteady Pressure Trace, $M = 0.35$, $k = 0.46$, 10 percent x/C , a) 7.6 cm Model; b) 15.2 cm Model	136

		Page
F-45	Unsteady Pressure Trace, $M = 0.35$, Static Probe, a) $k=0.46$, 7.6 cm Model; b) $k=0.90$, 15.2 cm Model	137
F-46	Unsteady Pressure Trace, $M = 0.80$, 10 percent x/C , a) $k=0.43$, 15.2 cm Model; b) $k=0.22$, 7.6 cm Model	139
F-47	Unsteady Pressure Trace, $M = 0.80$, 40 percent x/C , a) $k=0.43$, 15.2 cm Model; b) $k=0.22$, 7.6 cm Model	140
F-48	Unsteady Pressure Trace, $M = 0.80$, 60 percent x/C , a) $k=0.43$, 15.2 cm Model; b) $k=0.22$, 7.6 cm Model	141
F-49	Unsteady Pressure Trace, $M = 0.80$, a) $k=0.43$, 15.2 cm Model, Static Probe; b) $k=0.22$, 7.6 cm Model, Static Probe; c) $k=0.43$, 15.2 cm Model, Plenum; d) $k=0.22$, 7.6 cm Model, Plenum	143
F-50	Unsteady Pressure Trace, $M = 0.84$, 10 percent x/C , a) $k=0.40$, 15.2 cm Model; b) $k=0.21$, 7.6 cm Model	144
F-51	Unsteady Pressure Trace, $M = 0.84$, 50 percent x/C , a) $k=0.40$, 15.2 cm Model; b) $k=0.21$, 7.6 cm Model	145
F-52	Unsteady Pressure Trace, $M = 0.84$, a) $k = 0.40$, 15.2 cm Model, 55 percent x/C ; b) $k=0.40$, 15.2 cm Model, 60 percent x/C ; c) $k=0.21$, 7.6 cm Model, 60 percent x/C	146
F-53	Unsteady Pressure Trace, $M = 0.84$, 70 percent x/C , a) $k=0.40$, 15.2 cm Model; b) $k=0.21$, 7.6 cm Model	148
F-54	Unsteady Pressure Trace, $M = 0.89$, $k = 0.38$, 15.2 cm Model, a) 70 percent x/C ; b) 90 percent x/C	149
F-55	Pressure PSD, $\alpha = 0$, 15.2 cm Model, 50 percent x/C , a) $M=0.35$; b) $M=0.52$	152
F-56	Pressure PSD, $M = 0.70$, $\alpha = 0$, 15.2 cm Model, a) 60 percent x/C ; b) static probe	154
F-57	Pressure PSD, $M = 0.80$, $\alpha = 0$, 15.2 cm Model, 60 percent x/C	155
F-58	Pressure PSD, $M = 0.84$, $\alpha = 0$, 15.2 cm Model, a) 50 percent x/C ; b) 60 percent x/C	156
F-59	Pressure PSD, $M = 0.89$, $\alpha = 0$, 15.2 cm Model, a) 55 percent x/C ; b) 60 percent x/C	158

F-60	Pressure PSD, 15.2 cm Model, 0 percent x/C, a) M=0.35, k=0.46; b) M=0.35, k=0.90; c) M=0.70, k=0.47; d) M=0.84, k=0.40	PAGE 160
F-61	Pressure PSD, M = 0.52, 10 percent x/C, a) k=0.31, 15.2 cm Model; b) k=0.31, 7.6 cm Model; c) k=0.62, 15.2 cm Model	161
F-62	Pressure PSD, M = 0.80, 30 percent x/C a) k=0.21, 15.2 cm Model; b) k=0.22, 7.6 cm Model; c) k=0.42, 15.2 cm Model	163
F-63	Pressure PSD, 15.2 cm Model, 50 percent x/C, a) M=0.70, k=0.23; b) M=0.80, k=0.21; c) M=0.84, k=0.20; d) M=0.89, k=0.21	164
F-64	Pressure PSD, M = 0.84, 50 percent x/C, a) k=0.20, 15.2 cm Model; b) k=0.21, 7.6 cm Model; c) k=0.40, 15.2 cm Model	166
F-65	Pressure PSD, M = 0.84, 70 percent x/C a) k=0.21, 7.6 cm Model; b) k=0.40, 15.2 cm Model; c) k=0.38, 15.2 cm Model	167
F-66	Pressure PSD, M = 0.89, 15.2 cm Model, 60 percent x/C, a) k=0.19; b) k=0.38	169
F-67	Pressure PSD, 15.2 cm Model, 90 percent x/C, a) M=0.52, k=0.31; b) M=0.80, k=0.21; c) M=0.89, k=0.19	170
F-68	Pressure PSD, M = 0.80, k = 0.22, 7.6 cm Model, a) 50 percent x/C; b) Low Wall; c) High Wall; d) Static Probe; e) Top Wall; f) Plenum	172
F-69	Auto-Correlation, M = 0.52, $\alpha = 0$, 15.2 cm Model, a) 50 percent x/C; b) Top Wall	176
F-70	Auto-Correlation, M = 0.80, $\alpha = 0$, 15.2 cm Model, 40 percent x/C	178
F-71	Auto-Correlation, 15.2 cm Model, 0 percent x/C; a) M=0.52, k=0.62; b) M=0.84, k=0.40	180
F-72	Auto-Correlation, M = 0.84, a) k = 0.40, 15.2 cm Model, 80 percent x/C; b) k=0.40, 15.2 cm Model, 70 percent x/C; c) k=0.21, 7.6 cm Model, 70 percent x/C	181
F-73	Auto-Correlation, M = 0.89, k = 0.19, 15.2 cm Model, a) k=0.19, 90 percent x/C; b) k=0.38, 70 percent x/C; c) k=0.38, 90 percent x/C	182

F-74	Unsteady Pressure Distributions, Fixed Pitch, a) $M = 0.35$ b) $M=0.52$; c) $M=0.70$; d) $M=0.80$; e) $M=0.84$; f) $M=0.89$	PAGE 185
F-75	Oscillating Airfoil Pressure Data, $k = 0.13$, $M = 0.35$	188
F-76	Oscillating Airfoil Pressure Data, $k = 0.23$, $M = 0.35$	188
F-77	Oscillating Airfoil Pressure Data, $k = 0.46$, $M = 0.35$	189
F-78	Oscillating Airfoil Pressure Data, $k = 0.64$, $M = 0.35$	189
F-79	Oscillating Airfoil Pressure Data, $k = 0.76$, $M = 0.35$	190
F-80	Oscillating Airfoil Pressure Data, $k = 0.90$, $M = 0.35$	190
F-81	Oscillating Airfoil Pressure Data, $k = 0.09$, $M = 0.52$	192
F-82	Oscillating Airfoil Pressure Data, $k = 0.16$, $M = 0.52$	192
F-83	Oscillating Airfoil Pressure Data, $k = 0.31$, $M = 0.52$	193
F-84	Oscillating Airfoil Pressure Data, $k = 0.43$, $M = 0.52$	193
F-85	Oscillating Airfoil Pressure Data, $k = 0.52$, $M = 0.52$	194
F-86	Oscillating Airfoil Pressure Data, $k = 0.62$, $M = 0.52$	194
F-87	Oscillating Airfoil Pressure Data, $k = 0.07$, $M = 0.70$	195
F-88	Oscillating Airfoil Pressure Data, $k = 0.12$, $M = 0.70$	195
F-89	Oscillating Airfoil Pressure Data, $k = 0.23$, $M = 0.70$	196
F-90	Oscillating Airfoil Pressure Data, $k = 0.32$, $M = 0.70$	196
F-91	Oscillating Airfoil Pressure Data, $k = 0.39$, $M = 0.70$	197
F-92	Oscillating Airfoil Pressure Data, $k = 0.47$, $M = 0.70$	197
F-93	Oscillating Airfoil Pressure Data, $k = 0.06$, $M = 0.80$	199
F-94	Oscillating Airfoil Pressure Data, $k = 0.10$, $M = 0.80$	199
F-95	Oscillating Airfoil Pressure Data, $k = 0.21$, $M = 0.80$	199
F-96	Oscillating Airfoil Pressure Data, $k = 0.29$, $M = 0.80$	200
F-97	Oscillating Airfoil Pressure Data, $k = 0.35$, $M = 0.80$	200
F-98	Oscillating Airfoil Pressure Data, $k = 0.42$, $M = 0.80$	201

F-99 Oscillating Airfoil Pressure Data, $k = 0.43$, $M = 0.80$	PAGE 201
F-100 Oscillating Airfoil Pressure Data, $k = 0.06$, $M = 0.84$	203
F-101 Oscillating Airfoil Pressure Data, $k = 0.10$, $M = 0.84$	203
F-102 Oscillating Airfoil Pressure Data, $k = 0.20$, $M = 0.84$	204
F-103 Oscillating Airfoil Pressure Data, $k = 0.28$, $M = 0.84$	204
F-104 Oscillating Airfoil Pressure Data, $k = 0.34$, $M = 0.84$	205
F-105 Oscillating Airfoil Pressure Data, $k = 0.40$, $M = 0.84$	205
F-106 Oscillating Airfoil Pressure Data, $k = 0.05$, $M = 0.89$	208
F-107 Oscillating Airfoil Pressure Data, $k = 0.09$, $M = 0.89$	208
F-108 Oscillating Airfoil Pressure Data, $k = 0.19$, $M = 0.89$	209
F-109 Oscillating Airfoil Pressure Data, $k = 0.27$, $M = 0.89$	209
F-110 Oscillating Airfoil Pressure Data, $k = 0.32$, $M = 0.89$	210
F-111 Oscillating Airfoil Pressure Data, $k = 0.38$, $M = 0.89$	210
F-112 Oscillating Pressure Data on Tunnel Walls, $M = 0.35$	215
F-113 Oscillating Pressure Data on Tunnel Walls, $M = 0.52$	215
F-114 Oscillating Pressure Data on Tunnel Walls, $M = 0.70$	215
F-115 Oscillating Pressure Data on Tunnel Walls, $M = 0.80$	218
F-116 Oscillating Pressure Data on Tunnel Walls, $M = 0.84$	218
F-117 Oscillating Pressure Data on Tunnel Walls, $M = 0.89$	218
F-118 Typical Scaled and Spectral Data	226
F-119 Unsteady Aero Program Data Flow	232
F-120 Unsteady Aero Programs	235
F-121 FFT Algorithm for $N=8$	251
F-122 Rectangle Window and Transform	257
F-123 Raised Hamming Window and Transform	261

	PAGE
F-124 Examples of $G_{xx}(f)$ vs $R_{xx}(f)$	266
F-125 Cascade Form IIR Filter	271
F-126 Typical Transducer Installation	284
F-127 Error Train	292
F-128 Transducer Calibration Plot	298

I. INTRODUCTION

The analysis of fluid flow about two-dimensional oscillating airfoils was pioneered by Theodorsen (1935). The value of such analysis has been to determine the aerodynamic input loads for such problems as aeroelastically induced flutter of aircraft wings and tail sections, control surface buzz at transonic speeds, and aircraft buffeting response to atmospheric gust loads. The oscillating airfoil flow is most important in determining the nature and magnitude of the airloads generated by harmonic pitch and plunge oscillations corresponding to torsional motion of the wing and variations in the effective direction of the freestream air velocity. The efforts reported here were directed at measuring the aerodynamic loading associated with rigid airfoil pitch oscillation and the numerical computation of corresponding case examples. Parameterizations of the run sequence and computational matrix were performed over various geometric and flow variables in order to achieve an understanding of the relationship of the experimental procedures and the computational methods in predicting the free-air unsteady aerodynamic loads. Also, an attempt is made to both measure and numerically analyze the unsteady interference effects imposed on the test data by the ventilated wind tunnel walls.

The majority of the numerical methods available for the analysis of unsteady aerodynamic loads in a two-dimensional transonic flow over an oscillating airfoil are based on potential theory which assumes inviscid, isentropic flow. However, even with these somewhat restrictive stipulations, reasonable estimates of the unsteady loading are obtained at the low to moderate Mach numbers and at low reduced frequencies. Yet, the problems of numerical stability and convergence rate continue to plague even the most sophisticated of the current methods such that at the high subsonic Mach numbers (above critical flow conditions on the airfoil) and at the higher reduced frequencies the predictions are not reliable.

The problems of transonic flutter are due to the highly unstable flow induced by localized shock/boundary layer interaction on the airfoil surface. The formation of shock waves on airfoils of finite thickness in the upper subsonic speed regime is followed by shock strengthening and shift towards the trailing edge as speed increases. At near-sonic conditions, the first shock is positioned at the airfoil trailing edge and another shock is generated ahead of the airfoil as the Mach number passes through 1.0. The speed range between the initial formation of local supersonic flow regions and the locating of the shock at the trailing edge is the so called supercritical transonic range. Between these two limits, a shock of finite strength is located on the airfoil surface. This shock can cause significant changes in the boundary layer such as flow separation or self-induced flow oscillations. The unstable separation phenomenon is coupled with a motion of the shock wave and a trailing-edge feedback process. Since

disturbances cannot propagate upstream through the localized supersonic zones on the airfoil, the major path of communication is through the subsonic zones on the airfoil and separated flow wake around the trailing edge of the airfoil. This path can generate large flow oscillations which result in high intensity lift and moment coefficients.

Further coupling of the unsteady flow conditions with the structural bending and torsion modes of a wing can lead to the transonic flutter behavior. The flutter motion of a wing section can be described as a pitch-and-plunge combination which generates and sustains its own forcing function. While the plunging motions for long wings tend to be low frequency and relatively well damped, the torsional wing oscillation modes tend towards higher frequencies and less damping.

The criticality of the transonic flutter issues is exemplified by F-1 where relative flutter speeds for a supersonic transport wing [Sanford, et al. (1974)] and a space shuttle type wing [Goetz and Dogget (1974)] are shown. The "transonic dip" in the flutter boundary is evident. This behavior can restrict operations in the near-sonic regions of the flight envelopes due to the increased coupling between the unsteady airfoil motion and flow oscillations.

Renewed interests in transonic cruise aircraft and tactical maneuverability in this region have created a revival of transonic flutter research. As with the steady, transonic flow problems, the unsteady, transonic flow cases are difficult (if not impossible for the general case) to analyze. Therefore, experimental studies on

F-1 Transonic Flutter Speed Dip

oscillating airfoils in a transonic wind tunnel at OSU/AARL were conducted. An accompanying survey of existing data and analytical methods was also performed. The final objective of those studies was to provide an assessment of the unsteady tunnel interference effects. The oscillating, rigid airfoil models were used to measure a two-dimensional aerodynamic driving function for flutter response. These measurements were made in a blowdown airfoil tunnel having ventilated walls for transonic operation. The results of that work are reported here.

This dissertation discusses both experimental data obtained at OSU/AARL and computation results generated by existing analytical tools. A general outline for this report is given below.

<u>Section</u>	<u>Topic</u>
I	Introduction and Review of Previous Analytical and Experimental Work
II	Description of the OSU/AARL Experimental Setup
III	Steady and Unsteady Tunnel Interference Work
IV	Discussion of NACA 64A010 Steady Data
V	Discussion of Unsteady Data From NACA 64A010 Airfoil Oscillating in Pitch and From the Tunnel Test Section
VI	Conclusions and Recommendations
Appendix A	Data Acquisition Computer Routines (Descriptions Only)
Appendix B	Unsteady Data Analysis Techniques
Appendix C	Tube-Volume Response Analysis
Appendix D	Scaling Relations and Discussions of Run Conditions
Appendix E	Detailed Error Analysis

1.1 HISTORICAL PROSPECTIVE OF GENERAL FLUTTER PROBLEMS

The historical definition of flutter given by Bisplinghoff (1955) is: "A dynamic instability occurring in aircraft in flight, at a speed called the flutter speed, where elasticity of the structure plays an essential part in the instability." A dictionary definition might be given as: "A periodic oscillation set up in any part of an airplane by mechanical disturbances and maintained by inertia, structural characteristics, etc." Both of these definitions are inadequate in terms of the context studied here and the following working definition is proposed for these studies: "Flutter is the periodic bending or torsion of a wing due to aeroelastic coupling of the structural modes with the aerodynamic forcing function." The problem studied in this work will be restricted to the pitch oscillation mode that corresponds to torsional bending of a wing about its midchord axis. The problems of servoelastic and control feedback induced oscillations are not considered here.

One of the earliest examples of flutter phenomena was reported by Lanchester (1916) in a case where the structural characteristics and stiffness of the control system were coupled in phase with the aerodynamic response of the tail empennage section. The early analytical work in flutter had its origins with Theodorsen (1935) who provided the first, successful linearized treatment of the unsteady aerodynamic coefficients. Kussner (1940) developed the generalized lifting surface theory based on integral methods with a kernel function approach. Garrick provided a long series of reviews on aeroelasticity including articles in Applied Mechanics Reviews (1966), the Princeton

Series on High-Speed Aerodynamics and Jet Propulsion (1957), the AGARD Manual on Aeroelasticity (1962), and numerous NACA and NASA publications. The work of Watkins, Runyan, and Woolston (1955) brought the kernel function method to fruition and and later work delved into the experimental techniques and problems. Jones (1962) reported on the various aspects of unsteady flow in the Sixth Minta Martin Lecture including the status of unsteady airfoil work. The Symposium Transsonicum proceedings from 1962 and 1975 are devoted to transonic flow issues including a wide range of unsteady flow discussions. Unsteady stall and rotor dynamics associated with helicopter aerodynamics were reviewed by Ericsson and Reding (1970) and more recently by Jones, McCroskey, and Costes (1976). Several AGARD status reviews and supplements to the Manual on Aeroelasticity have been prepared by Rodden (1976, 1977). The most recent review of unsteady aerodynamics - to both the numerical and the experimental treatments of oscillating airfoils and flaps - was presented by Tijdeman and Seebass (Annual Review of Fluid Mechanics, 1980).

1.2 HISTORY OF OSCILLATING AIRFOIL WORK

A literature survey is presented for both theoretical and experimental efforts. The literature has been reviewed in a rough chronological fashion with grouping of works dealing with a particular analytical approach or series of experiments.

1.2.1 Unsteady Analytical Efforts

The unsteady airfoil analyses have generally been based on the small disturbance potential equation. Linearized differential forms or integral/kernel function methods are variations of the basic equation.

Most of the techniques developed before 1970 were applicable to the subsonic or supersonic speeds but failed in the critical transonic flutter range due to the nonlinear coupling of the steady flow terms into the unsteady problem. As with steady transonic flow analysis the thickness effects (neglected according to the small disturbance assumption) play a dominant role in controlling the shock position. Furthermore, the shock-wave/boundary-layer interaction has a great deal of influence on the shock location and motion. Hence, in the transonic range, unsteady aerodynamic analyses continue to be expensive for full potential and viscous cases although several local linearization schemes and improved numerical techniques have been applied to the problem.

Theodorsen's linearized method (1935) was the first exact solution (obtained at the cost of several limiting assumptions). He used a conformal mapping approach for the potential flow problem with circulation. The non-circulatory portion of the oscillating airfoil/flap combination was analyzed separately and later combined with circulatory elements modelled by vortices to satisfy an unsteady Kutta condition. The vortex distribution analyses evolved with a complex-valued function, $C(k)$, called Theodorsen's function, which has been tabulated and generalized by several authors since its original inception.

Küssner (1940) provided the basic tools for generalized lifting surface theory which led to the first computer solutions for unsteady wing aerodynamics. Watkins, Runyan, and Woolston (1955) used integral transform methods to develop a kernel function which related the

downwash to lift. The resulting downwash integral provided the basis for most integral method approaches since then. A separation of the sonic singularities was obtained with a simple indefinite integral. Landahl's monogram (1961) dealt with a linearized small perturbation theory for unsteady transonic flow near Mach 1 where the "sonic freeze" phenomenon occurs.

Extensive development of Multhopp's Lifting Surface Theory (for finite wings) and applications of the technique to analyzing oscillatory lift coefficients took place in Great Britain during the 1950's. Nearly all of the earlier efforts failed in the transonic region because of the inherent singularities occurring at the shock waves. Modern integral methods have overcome many of those problems through development of a mixed kernel function approach which provides separate treatments of subsonic and supersonic regions. Most recently, Fromme (1980) has performed oscillating airfoil calculations which include modelling of wind tunnel interference effects. His recent work is based on Possio's Kernel and Bland's transform methods which perform integration operations based on Gaussian quadrature by an orthogonal set of airfoil and pressure polynomials. It handles generalized airfoil mode shapes but does not account for the shock wave or supersonic portions of transonic flutter cases.

The finite-difference formulation for the unsteady, transonic, small-disturbance equation has been independently developed by Weatherill, Sebastian, and Ehlers and by Traci, Albano, and Farr. Both methods are based on the nonlinear, harmonic potential formulation which linearizes the steady and unsteady problems separately. Traci,

et al. use the steady solution (obtained by relaxation methods in their STRANS computer code) to provide the nonlinear coefficients for the unsteady solution (in their UTRANS computer code). These formulations can account for the shock influence from the steady flow and hence, give reasonable estimates for the unsteady pressure oscillations. The numerical instability of the approach restricts its use to low reduced frequency cases and even the steady-state solution has convergence difficulties in the supercritical transonic speed range.

Some of the numerical convergence problems can be overcome by application of the implicit, approximate factorization scheme (the AF2 routine) to solve the time-dependent, small-disturbance equation. The numerous publications by Ballhaus, et al. on the development of this method have shown that the LTRAN2 code (based on the AF2 scheme) has successfully treated many of the transonic unsteady problems. The code was originally based on the low-frequency, transonic approximations to the potential flow equation, and hence, the results were applicable only if the combination of Mach number, airfoil thickness, and reduced frequency fell below limiting criteria. Subsequent modifications by Houwink, et al. of NLR in the Netherlands and by Rizzetta (1979a and 1979b) have extended the usefulness of the LTRAN2 code by including additional perturbation terms previously neglected and by improving the treatment of the wake boundary condition.

Detailed numerical solutions for the full potential equations were outlined by Isogai (1978), and subsequent case studies of NACA 64A010 and 64A410 airfoils were computed with a code development by Magnus and Yoshihara (1975). These cases have been used extensively for

analytical and experimental correlations. However, the large amounts of computing time required for these cases make use of the full Euler equation method impractical for general engineering applications.

More recently, Yates (1979), and Brown and Daniels (1979) have each worked on viscous solutions for oscillating airfoil cases. These approaches have the potential to provide for the shock/boundary layer interaction and viscous effects which couple with the trailing edge feedback mechanisms. Steger, et al. (1979) have applied implicit numerical methods to analyze the unsteady boundary layer separation case known as buzz.

1.2.2 Unsteady Experimental Work

The oscillating airfoil experiments have historically suffered the same problems as have the steady-state airfoil testing. The interest for transonic testing grew out of every-increasing aircraft flight speeds and during the 1940's and 1950's, several large transonic testing facilities were constructed. The problems of steady interference were attacked by using ventilated test section designs. However, little was or has actually been done to date to design facilities specially suited to the unsteady transonic testing regime. In addition, the development of reliable, relatively inexpensive instrumentation was delayed until the late 1960's. The lack of adequate numerical tools to correlate observed behavior and the shift of research to supersonic problems during the 1960's led to a decline in unsteady transonic research. In the interim, much of the flutter research was carried out through testing of dynamically-scaled, structural models. The flutter model testing provided design

validation data but little significant research or engineering data that could be applied in a comprehensive manner. However, during the last decade, several experimental efforts to study oscillating airfoils have been undertaken; most notable among these are the work of Tijdeman, et al. at NLR in the Netherlands (the Nationaal Lucht-en Ruimtevaartlaboratorium...the Dutch National Aerospace Laboratory).

Oscillating airfoil tests were performed by Clevenson (1950) where a NACA 65010 profile was pitched about the quarter-chord axis by natural oscillation of a tuned spring system which provided testing over a reduced frequency range of $k = 0.15$ to $k = 0.35$ (for an incompressible flow case where $k = 2\pi fc/u_\infty$, reduced frequency based on chord). Bratt (1950) examined the oscillating wake structure behind a NACA 0015 at moderate to high reduced frequencies ($k = 0.65$ to 8.5) by use of smoke visualization in a low speed tunnel. Laidlaw (1954) examined an oscillating rectangular wing at a velocity of 40 m/sec. Pitch motions about the midchord combined with plunge oscillations were studied using in-situ pressure transducers to measure reduced frequencies over the range of $k = 0.20$ to 0.60 . Comparisons were made with predictions by lifting surface theories. Abramson et al. (1963) conducted wing bending experiments (plunging motion) in both wind tunnels and water tunnels. They obtained unsteady lift and moment coefficients at reduced frequencies of $k = 0.5$ to $k = 1.25$. Fischler (1968) performed fixed frequency pitch oscillation experiment with a NACA 65A008 airfoil. The tests were conducted at Mach numbers of 0.6 and 0.8, with a mean angle of attack of 4 degrees.

Recent work (since 1970) has benefited from improved instrumentation and a renewed interest in transonic cruise speeds and airfoil optimization for supercritical Mach numbers. Also, time-dependent computations with implicit solution techniques became available and required experimental correlation. Ruiters (1971) used hydrogen bubble flow visualization in a low Reynolds number water tunnel to examine leading edge separation at reduced frequencies less than $k = 0.2$. Pitch oscillations about the midchord axis were performed over an angle of attack range from 0° to 30° for a NACA 0009 airfoil and NACA 0015 flapped airfoil. Nazarenko and Nevezhina (1972) used twelve pressure sensors to examine the behavior of a forced aileron oscillation at Mach 0.974. Commerford and Corta (1974) used the natural vortex shedding from a cylinder to generate relative pitch oscillations at high reduced frequency ($k = 3.9$). Archibald (1975) examined the unsteady Kutta condition at high reduced frequency ($k = 7$ to $k = 50$) using an acoustic excitation of the flow. Parker (1976) studied an airfoil (NACA 0012) oscillating through stall at Reynolds numbers of 1×10^6 and 3×10^6 . Stall vortex shedding was observed. Drane (1970) examined a low aspect-ratio wing (NACA 0006 profile) with an oscillating control surface at reduced frequencies of 0.73 to 8.45. A low-speed, open-jet wind tunnel was used along with a combination of in-situ transducers and the tube-volume measuring system developed at NLR in the Netherlands. Mehta (1977) performed separated boundary layer studies on a model helicopter rotor. Schlieren data was obtained at reduced frequencies of $k = 0.25$ and $k = 0.50$ for angles of attack between 0° and 20° . Lambourne (1977) examined a tapered wing of

finite aspect ratio for $k = 0.5$ to 2.8 . St. Hilaire et al. (1979) performed oscillating experiments on swept NACA 0012 airfoils. In addition to those most recent experiments listed above, studies at NASA Ames and NLR will be discussed separately.

The oscillating airfoil work at NLR in the Netherlands has provided valuable experimental data for correlation with recent computer code developments. Bergh (1958) reported on experiments with a NACA 0012 airfoil with an oscillating 40 percent flap. Tests were conducted at a Reynolds number of four million and reduced frequencies of $k = 0$ to 0.8 . A tube-volume transducer system developed at NLR was used for measurement of the unsteady pressures. Through close matching of connecting tube lengths, a single transducer and a rotary pneumatic valve are used to obtain data using ordinary model static taps by stepping from port-to-port. Extensive analyses and experimental development were conducted by Tijdeman and Bergh to verify accuracy of the system (see Appendix C). Later, Tijdeman and Bergh (1967) conducted another series of oscillating flap experiments with a NACA 64A006 airfoil with a 25 percent rear flap. Testing was conducted in the NLR Pilot Transonic Tunnel over a Mach number range of 0.5 to 1.02 with both unsteady pressure measurements and schlieren.

Measurements were also obtained on the test section walls. These pressure data were again obtained with the NLR tube-volume measuring technique. Main flow effects were later applied to the tube-volume data to correct the previous experimental data as reported by Tijdeman and Bergh (1972). The updated results were reported at Mach numbers of 0.6 and 0.8 and moderate reduced frequencies. A more complete

reporting of the corrected data was given by Tijdeman and Schippers (1973) to supersede previous reporting. The data ranged over Mach numbers of 0.5 to 1.0 and reduced frequencies, $k = 0$ to 0.39. A series of joint NLR/ONERA tests provided a direct comparison of the tube-volume measuring technique and corresponding in-situ transducers (ONERA is the Office National d'Etudes et de Recherches Aérospatiales in France). In those tests, a low aspect-ratio, swept, tapered wing was examined by Destuynder and Tijdeman (1974) for forced oscillations and for undamped flutter cases. Further testing of the NACA 64A006 with quarter-chord rear flap oscillation was reported by Tijdeman (1975a). Measurements of a shock motion were made from flow visualization and pressure data; Tijdeman provides a classification of the shock motion in three type categories. An unsteady shock analytical model was correlated with the observed experimental results. Additional theoretical development was reported in Tijdeman (1975b) along with three-dimensional testing of a swept, tapered NACA 65A006 with oscillating flap. Tijdeman and Destuynder (1976) again corroborated in a joint NLR/ONERA examination of unsteady flow due to transonic wing/store interaction. Tijdeman, Schippers, and Persoon (1977) tested a supercritical airfoil designated as the NLR 7301 profile. Pressure data was obtained for pitch oscillations about the 40 percent chord axis for Mach numbers of 0.5 and 0.7, and reduced frequencies of $k = 0.03$ to 0.20. A comprehensive review of the NLR work is given in Tijdeman's (1977) Ph.D. thesis. Roos (1980) reports results for harmonic pitch of a wing/store model at transonic speeds of $M = 0.6$ to 1.35 with reduced frequencies less than $k = 0.4$. The NLR

experiments have served to establish a basis for other empirical correlations and analytical comparisons.

Parallel efforts at NASA Ames Research Center have been conducted primarily with in-situ transducers rather than with the NLR tube-volume measuring technique. Davis (1976) made observations of trailing edge noise generated by a NACA 64A006 airfoil in a special acoustic wind tunnel. Later, Satyanarayana and Davis (1977) examined the trailing edge pressures from an oscillating NACA 64A010 airfoil at reduced frequencies up to about one in the same acoustic wind tunnel facility. Going to larger scale size, Davis and Malcolm (1978) developed an oscillating airfoil drive system for the Ames 11-foot transonic wind tunnel. It was a hydraulically operated support system with an active feedback control system. In the same paper, they reported on a dedicated data acquisition and digital processing system designed to handle the oscillating airfoil data. In addition, Davis and Satyanarayana (1978) reported on the use of vertical splitters in the 11-foot tunnel to allow 2-D airfoil testing. Results from fixed pitch angle testing of a 12 percent biconvex airfoil were correlated with numerical predictions obtained from the MacCormack and TSFOIL computer codes. Davis and Malcolm (1979) reported the results of testing a NACA 64A010 airfoil oscillating in pitch and plunge motions. The data was obtained in the 11-foot tunnel and was used for comparisons with the full potential solutions obtained by Magnus and Yoshihara. The following year, Davis and Malcolm (1980) reported an additional testing of the NACA 64A010 and the NLR 7301 airfoil sections and direct comparison was made with Tijdeman's results.

1.3 CURRENT EXPERIMENTAL WORK AT OSU/AARL

Research in unsteady aerodynamics in the OSU/AARL 6 inch x 22 inch transonic airfoil tunnel began in 1976 with initial surveys of test section internal noise levels. Surveys were made to determine the ambient noise levels and sources that would possibly contaminate oscillating airfoil data. A standard for comparison with other transonic test facilities has been noise measurements on a 10° cone. A cone model was fabricated which replicated the features of the AEDC standard 10° cone, and it was used to measure noise levels in the 6 x 22 inch tunnel during a second series of tests. Following the second tunnel survey, modifications to the settling chamber were made--including rework of the perforated cylinders at the inlet and replacement of the turbulence damping screens with ten new screens. Another survey of the test section noise levels showed little change in the data obtained with the 10° cone model.

Oscillating airfoil tests began in May 1978 and three test series were conducted over the period up to August 1979. The final series involved testing of two models; both airfoils have a NACA 64A010 profile but were different chord lengths, namely, 7.6 cm (3 inches) and 15.2 cm (6 inches). The airfoils were oscillated in pitch about the mid-chord axis over a frequency range of 20 Hertz to about 120 Hertz. Six Mach numbers were explored over the range of 0.35 to 0.89. The results of the final series of testing are described here for comparison with other analytical and experimental efforts.

1.3.1 Test Description

The final series of airfoil tests were conducted as both fixed angle-of-attack (AOA) tests and pitch oscillation tests. The fixed AOA tests were conducted at each Mach number first to determine the zero lift angle to be used as the reference zero pitch angle for the symmetric NACA 64A010 airfoil models. Next, both positive and negative pitch angles were tested to obtain the quasi-steady lift information. These fixed AOA tests were repeated for each Mach number. Finally, the model drive system was set up for a zero mean pitch, $\pm 1^\circ$ oscillation at various frequencies. At each driven frequency, the model was tested at each Mach number condition. The tests were conducted so that the selected frequencies were at near even multiples of each other. When subsequent testing of the other scale model was conducted, there was a direct overlap of reduced frequencies as well as physical frequencies. The testing was conducted over the frequency range of 20 Hertz to 120 Hertz for reduced frequencies between $k = 0.05$ to $k = 0.90$.

1.3.2 Test Objectives

The goals of these test series were: (1) develop experimental techniques to examine oscillating airfoil behavior in transonic flow; (2) measure unsteady loads on a representative airfoil for correlation with other empirical and numerical results; and (3) assess the nature of the unsteady interference effects in transonic testing of oscillating airfoils. Accomplishment of the first objective involved determination of the tunnel background noise levels, development of the data acquisition and data reduction computer routines, and construction of the oscillating airfoil models and drive system. The remaining two

objectives were addressed in the final test series where a complete matrix of Mach numbers and reduced frequencies were examined.

II. OSU/AARL EXPERIMENTAL APPROACH

The experimental facilities, airfoil models, data acquisition systems, and test conditions are discussed in detail in the following sections. The general design features of transonic wind tunnels are outlined by means of a literature review. The developments of the ventilated-wall and of the intermittent-operation tunnel designs are reviewed following a description of the test facility used for these experiments. Data on the test conditions experienced in the unsteady testing at OSU/AARL is also presented. Drawings, photographs, and design details for the models and instrumentation are shown. The test procedures and data processing routines are outlined in Section 2.3. Comments on unsteady data sampling statistics and on the measurement error are given in Section 2.4.

2.1 TEST FACILITY

The OSU/AARL 6 inch x 22 inch transonic airfoil tunnel was constructed as a two-dimensional test facility to study the transonic flow over airfoils over a wide range of Reynolds numbers. It is a blowdown test facility. The high-pressure air supply at OSU/AARL uses storage tanks with a maximum pressure of 2600 psia and 1500 cubic feet capacity. The flow is controlled by two hydraulically actuated valves; a rotary-plug valve provides the start-stop operation and a conical throttling valve controls flow rate. The start/stop operation is

directly linked to the on-line computer system for operator control. The flow control valve is preset by the tunnel operator through a pneumatic feedback system. Control of the mass flow rate determines the test section pressures and Reynolds number. The flow enters a 90 psia settling chamber through an 8-inch diameter supply line (see F-2). The air is throttled through three concentric perforated cylinders and "squirted" into the upstream end of the settling chamber. A perforated bulkhead provides additional flow distribution. Ten 60-mesh screens are installed to provide turbulence damping. A contoured, double-contraction nozzle reduces the flow area by a ratio of 13.7:1 into the test section. The test section measures 6 inches (15.24 cm) wide by 22 inches (55.88 cm) by 44 inches (111.76 cm) long. The perforated upper and lower walls are a uniform 10 percent porosity along the entire test section length. The straight 1/8-th inch diameter holes feed into 3-inch deep plenums. The plenums are independently aspirated by the flow at the downstream end of the test section. Viewing ports in the solid sidewalls are used to mount full-span, two-dimensional airfoil models. Modified window inserts were installed to accommodate the oscillating airfoil testing. Five Kulite unsteady pressure transducers were installed in the test section and settling chamber during the initial noise surveys. A vertically translating pitot probe is used to measure momentum deficit downstream of the airfoil models for determining section drag coefficients. Test section Mach number is controlled by the choke bar arrangement located in the diffuser exhaust section. The air is exhausted directly to the atmosphere outside the test facility.

● KULITE PRESSURE TRANSFUCER LOCATIONS

F-2 AARL 6 x 22 Airfoil Tunnel Schematic and Operating Envelope

The operating envelope for the 6 inch x 22 inch tunnel is also shown in F-2 where it is observed that Mach numbers over the range of 0.2 to 1.1 are attainable. The various choke bar arrangements give a nearly continuous series of Mach number conditions with control to a ± 0.001 Mach number tolerance. Unit Reynolds number can be varied continuously over this speed range from 0.35 million per foot at low Mach number to a maximum of 34.0 million per foot at the highest flowrate allowed. This span of Reynolds number is controlled by operating the blowdown tunnel at stagnation pressures of 20 to 90 psia (138 to 621 KPa). Temperature and pressure variations during the short run times are relatively constant, and as seen below, the operator controls the precise moment when the data is taken with the aid of an on-line computer display. Therefore, the Reynolds number can be accurately controlled from one data point to the next.

The general requirements for transonic wind tunnel designs include simulation of the appropriate nondimensional flow parameters such as specific heat ratio, Mach number, and Reynolds number. In addition to the transonic interference problems, it is often difficult to obtain full Reynolds number simulation with geometrically scaled models. Accurate replication of the boundary layer characteristics is required for transonic flow simulation since the boundary layer/shock interaction is highly dependent upon the boundary layer growth. The mean shock location can be greatly altered by viscous effects and tunnel interference; this leads to erroneous results. More general wind tunnel requirements such as low levels of freestream turbulence and uniform flow conditions through the test section hold for transonic

tunnels as well. Ambient noise levels have been observed to have an influence on boundary layer transition results, and they must be minimized if unsteady pressure measurements are desired.

Historically, wind tunnels were first constructed during the late 19th century by Wenham (1871). The Wright Brothers constructed a wind tunnel to examine airfoil characteristics for both the propellers and wings of their early aircraft. Both open-jet and closed test sections were built and problems encountered were traced to the influences on the model downwash velocities at the airfoil trailing edge. Rigid walls tended to decrease the downwash whereas the free-jet boundary allowed greater downwash than would result in a freestream flow condition. Theodorsen (1931) studied these effects and proposed a balance of open and closed portions of the test section walls to achieve a balanced effect on the downwash velocity. Allen and Vincenti (1944) extended the theory of interference to cases involving compressibility. The ventilated test section designs (both perforated and slotted configurations) have been incorporated in transonic wind tunnels since Theodorsen's proposal and many subsonic and supersonic tunnels as well have been outfitted with ventilated walls.

Noise in wind tunnels can be attributed to several major sources, namely: compressor noise for closed loop tunnels, diffuser boundary layer separation, inadequate turbulence management upstream of the test section, wall-induced resonances in the slots or perforations, valve flow noise, etc. Many investigations into wind tunnel noise and its treatment have been conducted. Dryden and Abbott (1948) reported on the development of low turbulence tunnels for testing low-drag

airfoils. Their paper includes a discussion of the effectiveness of damping screens and contraction sections in reducing turbulent fluctuations in the test section. Hensel (1964) performed research in the area of transonic pressure fluctuations, presenting early unsteady pressure measurements of the boundary layer/shock interaction which results in buffet phenomena. Hosier and Mayes (1972) investigated noise associated with blowdown wind tunnels, namely, valve noise, test section boundary layer disturbances, and exhaust jet pressure fluctuations. Loehrke and Nagib (1972) provide an extensive review of turbulence management techniques including design of screens, honeycombs, foams, etc. Among their findings, they concluded that high solidity perforated plates can generate jets which coalesce rather than dissipating and that multiple screens are better than a single high pressure-drop element. The LEHRT Group (1974) was organized to review transonic tunnel design ideas for a large European test facility. They defined the major technology issues as : (1) facility noise, (2) model design for loads in high-pressure tunnels, (3) contraction section design, and (4) test section designs. The resulting MiniLaws Report (1975) provided data and guidelines for addressing the current transonic test problems. Morel (1975) reported on optimum designs for tunnel contractions and concluded that two matched cubic contours would provide the best flow quality. Concurrently, Ramjee and Hussain (1975) investigated combinations of screens and contraction ratios to provide the best turbulence damping characteristics. Goethert (1976) reviewed the new trends in transonic tunnels including the use of cryogenic test conditions for high Reynolds number simulation and the advent of

adaptive-wall wind tunnels which are intended to remove boundary interference effects. Mabey (1976) reported on tunnel designs with low unsteadiness. He performed measurements on the influence of turbulence on models, deduced limits for acceptable inlet conditions, and isolated the major sources of wind tunnel noise. Nearly all of the efforts reported above have been aimed at improving the flow quality for steady-state transonic testing. Looking at the sources of test section noise, one must first consider the pressure fluctuations in the turbulent boundary layers on the test section walls and any influence due to interaction with shock waves generated by the model. Willmarth (1958) first characterized the turbulent boundary layer pressure disturbances as being approximately 0.35 percent of the freestream dynamic pressure (q_∞) over a wide range of Mach numbers and Reynolds numbers. Willmarth (1959) updated the unsteady pressure levels to 0.6 percent of q_∞ . He deduced that for eddies that were greater than 30 percent of δ , the boundary layer thickness, the average convection velocity was 82 percent of the freestream velocity, and such eddies are preserved over a distance of about 10δ . These data were obtained with space-time correlations and universal spectral shapes were derived. Laufer (1961) studied supersonic turbulent boundary layers and how noise was radiated from the boundary layers into the core of a supersonic wind tunnel. Willmarth and Wooldridge (1962) correlated the pressure fluctuations to wall shear stress. They also found that the pressure spectra scaled with freestream velocity and boundary layer displacement thickness. Data for low frequency (large-size) eddies again showed that convection speeds near 80 percent of freestream

velocity, and data for high frequency (small-scale) eddies indicated convection speeds near 65 percent of freestream speeds. The unsteady pressure level was reported as $P_{RMS} = 0.0047 q_{\infty}$ (previous experimental results were discounted due to transducer size effects). Bull (1967) discussed two wave number families for eddy size distribution. The high wave number disturbances were said to be generated in the constant shear stress region of the boundary layer and appeared to dissipate at rates inversely proportional to wave number. At larger spatial separations, the low wave number eddies were observed to persist longer and to be convected at speeds up to 0.825 times the freestream velocity. Willmarth (1971 and 1975) provided reviews of instrumentation techniques and boundary layer noise results. He hypothesized that pressure fluctuations in an incompressible, turbulent, boundary layer are the result of momentum fluctuations within the field, and hence, pressure-velocity correlations are negligible. The mean lifetime of pressure-producing eddies was found to be such that they propagated a distance of about six times their lateral extent before dissipating or distorting to uncorrelatable characteristics.

An added source of noise during two-dimensional transonic testing is the pressure fluctuation associated with boundary layer/shock interaction on the sidewalls of the test section. Fiszdon and Mollo-Christensen (1959) performed oscillating shock experiments and examined the interaction with a laminar boundary layer. Coe (1969) investigated transonic shock/boundary layer interaction and deduced typical pressure spectra corresponding to various local flow

conditions. In that paper (and a later experiment in 1973), he discussed test techniques to survey tunnel noise, and he determined the effects of transducer size and flushness on data quality. Ericsson (1975) examined the unsteady, shock-induced, boundary layer separation and structural mode coupling commonly called "buffet". Telionis (1977) gave a review of unsteady boundary layer research. In his own research, he concluded that the mean boundary layer profile was relatively unaffected by large forced oscillations. Simpson (1977) used laser-Doppler velocimeter and hot-wire measurements to examine a turbulent boundary layer with a periodic driving function. A series of experiments reported by Seegmiller, Marvin, and Levy (1978a, 1978b, 1980) examined unsteady transonic flow through both experimental and theoretical studies. An 18 percent thick circular arc airfoil was tested and analyzed in the supercritical Mach number range of 0.76 through 0.79 at a fixed Reynolds number of eleven million. The resulting oscillating shock/boundary layer interaction was investigated with an LDV setup synchronized to signals from selected pressure taps.

Tunnel noise was extensively investigated in supersonic tunnels due to the pronounced influence it has on laminar boundary layer transition. Laufer (1961) postulated that noise in supersonic tunnels is primarily due to turbulent boundary layers in the nozzle section, radiating noise into the test rhombus. Pate (1969 and 1971) investigated boundary layer transition on a 10° cone tested at Mach 3 and 6. Microphone measurements in the cone indicated that transition was directly affected by tunnel noise. Credle (1969) used the same 10° cone to examine noise levels in the AEDC 4-foot PWT transonic test

facility. He obtained unsteady pressure levels and spectral data from 30 Hertz to 10 Kilohertz. The 4-T wind tunnel was surveyed over the Mach number range of 0.2 to 1.2 at pressures of 0.5, 1.0, and 1.5 atmospheres. Credle and Shadow (1970) continued the survey of AEDC facilities with examination of the 16-T tunnel noise levels over a range of 0.6 to 1.3 at unit Reynolds numbers of six (6) million to twenty-two (22) million per meter. A maximum unsteady pressure coefficient, C_p RMS, of 0.035 was detected at Mach 0.75. It was hypothesized that in the range of Mach 0.7 to 0.8 a noise generated in the perforations of the ventilated test section walls caused large increases in the ambient noise levels. Dods and Hanly (1972) reported on tunnel noise studies at NASA Ames in the 12-foot pressure tunnel, the unitary 11' x 11' transonic tunnel, and the 14-foot transonic tunnel. Data sampled over the frequency range of 10 Hertz to 20 Kilohertz indicated that each of these tunnels had resonances associated with the corrugated slots in the ventilated test sections. Tests performed by Ross and Rohne (1973) in the NLR High Speed Tunnel indicated a maximum C_p RMS of 0.01 at Mach 0.8 (due to acoustic wall resonance). Measurements were made with the AEDC 10° cone. Noise propagating upstream from the compressor and diffuser was detected, but it was effectively blocked at speeds approaching Mach one. Comparisons are made with the results obtained at AEDC, NASA Ames, etc. Varner (1974) provided a general review of transonic tunnel noise. The wall noise was postulated as being generated by three possible sources: (1) jet flow through holes or slots, (2) acoustic resonance due to organ-pipe modes in the perforations or slot corrugations, and (3)

random pressure fluctuations radiating from the turbulent sidewall boundary layers.

Parker (1975) conducted a study of tunnel wall hole configurations in the AEDC 1T in an effort to damp the ventilated wall resonances using the 10° cone model. Various hole patterns and hole slant angles were tested along with screens and hole splitters aimed at decoupling resonant characteristics from the flow. Schutzenhofer and Howard (1975) used screens over the walls of the NASA Marshall 14-inch Transonic Wind Tunnel to reduce the ambient noise levels to those of a flat plate turbulent boundary layer. Dougherty et al. (1976) continued the AEDC experiments on noise reduction techniques for perforated walls. They tried spoilers and splitters in the wall perforations and it was found that single longitudinal splitters in each hole would reduce test section noise by a factor of three. Bauer and Chapkis (1977) examined perforated inlet liners and determined the wall resonance frequencies were observed to have wavelengths corresponding to Strouhal frequencies (based on hole diameter) between 0.1 and 0.2. Laderman (1977) provided updated noise measurements for supersonic wind tunnel testing. The investigations above have indicated that the major noise source in transonic wind tunnels is wall resonance through the upper subsonic Mach number range. Also, treatments for hole resonances and new tunnel wall designs are reviewed.

Noise surveys in the OSU/AARL transonic airfoil tunnel were conducted as a part of these studies. These measurements used Kulite transducers mounted in and around the test section (as seen in F-2). Measurements were made at five locations. A position in the stilling

chamber, upstream of the last screen gave the nominal stagnation pressure fluctuations ahead of the test section. Two sidewall locations just ahead of the center of the test section measured pressures within the test section and inside the lower plenum, respectively. For the initial noise survey, two probes were positioned near the center of the test section to measure fluctuating pitot and static pressures. The probes were cantilevered from the model view ports and were equally spaced from each other and the sidewalls. During the second noise survey, these probes were replaced with an instrumented cone model with transducers installed at two axial locations. During airfoil model tests, data was not obtained from these locations, however, unsteady pressure data was obtained in and near the upper plenum (those results are discussed in section 5.5 below).

Three test series were performed over a period of 18 months. The initial test series examined the baseline noise levels and investigated the influence due to the straight perforations in the upper and lower walls. A correlation with published tunnel noise data from other wind tunnel facilities was obtained from testing a 10° cone model during the second series. The effects of modifications to the settling chamber (changing from 4 screens to 10 screens and adding another perforated cylinder and screens around the inlet) were observed during the final noise survey.

The data presented here and in the subsequent airfoil testing are reported in the conventional non-dimensional forms of pressure, lift and moment coefficients. The steady-state pressures are denoted by

$C_p = (p - p_\infty) / q_\infty$, where the sensed pressure, p , is relative to the freestream static pressure, p_∞ , and the freestream dynamic pressure, q_∞ , as determined from the sensed tunnel pressure and calculated from facility calibration data. The root-mean-square amplitude of the unsteady pressure signals, P_{rms} are non-dimensionalized similarly as: $C_{p\ rms} = P_{rms} / q_\infty$. The oscillatory pressure coefficient for fluctuations at the model oscillation frequency take a form of $|C'_p| = |p'| / (q_\infty \Delta\alpha)$. The single-frequency, peak pressure amplitude, $|p'|$, is non-dimensionalized by both dynamic pressure and model motion amplitude, $|\Delta\alpha|$, in radians ($\pm 1^\circ$ pitch oscillation was maintained for all oscillating airfoil tests). The $|C'_p|$ data was obtained through a complex demodulation processing described in Appendix B. Other unsteady data results, including power-spectral-densities (PSD's) and correlation functions (also described in Appendix B) are presented. The PSD data and the unsteady pressure traces are displayed in raw engineering units in Chapter 5; however, scaling criteria for the PSD amplitude and frequencies are presented in Appendix D. The PSD magnitudes have been appropriately non-dimensionalized for the tunnel noise survey results presented below so that comparisons between various Mach numbers are facilitated. The correlation data is normalized by the standard deviation of the signal, and hence, is already in a non-dimensional form.

The unsteady pressure levels measured during the initial empty tunnel surveys show a near linear dependence with dynamic pressure at a given Mach number, whereas there appears to be roughly an inverse proportionality versus Mach number for the ambient pressure

fluctuations (F-3). These results show the highest levels were detected in the stagnation chamber while the lowest levels were observed in the lower plenum chamber behind the perforated wall. The static wall location indicated 10 to 20 percent lower levels than the stagnation chamber while the static probe showed still lower values (by 20 to 30 percent over sidewall data). The influence of the turbulent boundary layer at the solid sidewall is observed here as the difference between the static probe and the sidewall. As discussed above, this contribution is at least an additional $C_{p_{rms}}$ value of 0.005. At the lower Mach numbers, the variation of $C_{p_{rms}}$ from the static probe to the sidewall is about 0.01 but above Mach 0.40, the 0.005 predicted variation is observed. The total probe measurements exhibit a 5 to 10 percent larger fluctuating pressure average compared to the static probe data.

The spectral contents observed from data obtained at lower Mach numbers showed generally random, non-resonant structure. Minor spectral peaks and broadband maxima were seen in PSD and CSD plots. The highest energy levels were in the 1000 to 1500 Hz range and this noise appeared stronger inside the settling chamber. The cross-correlation results indicated highly random pressure fluctuations. There were apparently no strong turbulent structure convected from the settling chamber into the test section. At Mach numbers of 0.49 and below, there was no apparent generation of acoustic resonances at the perforated walls.

Strong resonant structure was noted at Mach 0.61. Sharp peaks were detected near 300, 900, 3300 and 8300 Hz at levels at about 118 dB or

F-3 Unsteady Pressure Survey of 6" x 22" Tunnel

F-4 Static Probe PSD, Open Walls, M = 0.60

F-5 Lower Plenum PSD, Open Walls, M = 0.60

less (see F-4 and F-5). The cross-correlations generally showed irregular long-wave correlation with superimposed higher frequency disturbances. The static sidewall transducer PSD displayed a prominent peak at 8300 Hz (15 dB above neighboring bands) with a 126 dB level. The associated auto-correlation showed an amplitude modulated, high frequency signal (suggesting a side lobe structure associated with the 8300 Hz peak). In general, broadband peaks were observed in the 600 to 1000 Hz range in the test section along with the sharp 8300 Hz peak (up to 127 dB levels were measured at the high frequency). However, this high frequency was not observed in the stilling chamber data, and while the test section correlations showed the short wave periodically, these high frequency components did not correlate with the stilling chamber signal. Instead, the PSD from the stilling chamber transducer indicated a maximum over the 1000 to 1300 Hz range with a small secondary peak near 3500 Hz and its auto-correlation showed irregular, long-wave patterns.

The high frequency peak is not excited at Mach 0.69 but there are still distinct peaks. The total probe measured broad energy peaks at 1000 Hz and 3000 Hz while the static probe measured resonant peaks at 1250 Hz, 3630 Hz, and 7250 Hz (see F-6) corresponding to a fundamental mode and the first two harmonics. The apparent frequency shift between the total and static probes may indicate the disturbances to be "flow-related" (i.e., convected disturbances or affected by Doppler-shift due to motion of the transmitting medium). The static sidewall and lower plenum locations indicate the major peak at 3630 Hz measured at levels to 126 dB. The auto-correlations show: (1) long

F-6 Static Probe PSD, Open Walls, $M = 0.69$

wave structure for the total probe; (2) a harmonic waveform structure for the static probe; and (3) moderately periodic structure for sidewall and plenum signals, indicating significant broadband contributions.

At Mach 0.80, the major resonant peak occurred near 4800 Hz. However, despite the dominance of the 4800 Hz peak in the spectra, cross-correlations of the various data channels showed only small quasi-periodic structure indicating only a slight coupling of these signals. At higher flow speeds, the resonant structures were not apparent as they were at the Mach 0.61 through 0.80 conditions. These resonances were apparently edge-tone oscillations generated by the wall perforations.

Investigations into the lower frequency, broadband peaks between 1000 and 1500 Hz showed that there were no significant correlations between the stilling chamber fluctuations and test section noise. Acoustic transit times of 2 to 8 milliseconds showed no increased correlation coefficients associated with radiation of jet noise from the perforated bulkhead or screens into the test section. Also, flow convection required time delays on the order of 80 milliseconds between the stilling chamber transducer site to the test section locations. Additional test runs at extended sample periods showed no increase above the low values associated with random fluctuations for up to 200 millisecond delays. Hence, it was concluded that no coherent acoustic or turbulent velocity waves were transmitted into the cavity from upstream.

To isolate the resonant peaks in the mid-subsonic range, additional test runs were made with the perforations sealed with tape inside the test section. Overall rms pressure levels showed few significant changes. The static pressure probe showed a 12 percent increase in $C_{p_{rms}}$ at Mach 0.61 yet decreases of 4 and 17 percent were noted at Mach 0.69 and 0.81, respectively. Similar results were observed for the sidewall transducer. Further examination revealed that averaged data points at Mach 0.61 were strongly influenced by data at the lower Reynolds number conditions. Eliminating those points showed that covering the wall perforations gave reductions of between 4 and 14 percent at higher Reynolds number conditions for Mach 0.61 for sidewall and probe data, respectively.

The power-spectral-density plots from each of the transducer locations indicate that, in general, the peak values of the resonant oscillations are reduced by sealing the wall perforations. However, few distinct changes in the base levels were observed, and as a result, the overall power was not significantly reduced.

The total pressure probe measurements at Mach 0.61 show reductions of over 50 percent in the two major peaks at 720 Hz and 8400 Hz; however, the sealed-wall data shows a general increase in the overall level and additional peaks near 1000 Hz and 7300 Hz. Results on the static probe are more dramatic with reductions of 96 percent in power near 6210 Hz and 740 Hz peaks and 82 percent near 8420 Hz (see F-7 in comparison with F-4). These are relatively narrowband peaks and contain little power when compared to the overall integrated power under the PSD curves. The sidewall transducer showed similar effects

F-7 Static Probe PSD, Taped Walls, M = 0.61

with 90 to 95 percent peak power reductions in the lower frequencies but only a 55 percent reduction at 8420 Hz. The lower plenum initially showed the large resonant peak at 8420 Hz but was not measured with the taped wall configuration. Measurements in the stagnation chamber showed few significant changes in peak values with no recognizable change in the general spectral shape or overall level.

The corresponding auto-correlation plots show little change in character for the total pressure probe data, which shows only slightly periodicity with a high frequency wave superimposed over a rapidly decaying lower frequency disturbance. The change in correlation coefficients for the static probe data is more apparent (F-8) where there appears to be greater damping of the high frequency periodic disturbance as a result of taping the perforated walls. Similar results are observed in the sidewall data, but the change in amplitude is less (partially due to smaller correlation values in the original open wall data). While the lower plenum auto-correlation data exhibits the same periodicity seen in the static probe and sidewall data, the stagnation chamber data indicates only low frequency correlation, that is relatively unchanged by covering the test section perforations.

Large reductions in resonant peaks were observed at higher Mach numbers (0.69 and 0.81). The high frequency peaks were attenuated by about 75 to 90 percent by sealing the wall perforations. In some cases, the resonant peaks were attenuated to levels obscured by the random noise content of the signals. Similarly, the corresponding auto-correlation plots showed that taping the perforations severely attenuated the periodic disturbances to random signal values.

F-8 Effects of Wall Taping on Static Probe
Auto-Correlation, $M = 0.61$

The wall perforations were found to be a source of high frequency oscillations in the Mach number range of 0.61 to 0.81. These frequencies are well above any harmonics that were observed during the airfoil pitch oscillation testing and hence were not considered in reducing unsteady pressure data from those runs. Small reductions in overall power levels were noted when the perforations were covered; the large attenuations noted from the major peaks resulted in only small benefits since these narrow-band resonances contained relatively small percentages of the total energy.

The second noise survey used a 10° cone model similar to the AEDC dynamic calibration cone described in reports by Pate (1969) and Credle and Shadow (1970). An eight-inch long (20.3 cm) cone was constructed with transducers at 2.0 inches (5.08 cm) and 5.5 inches (12.7 cm). Steady-state pressure taps were used to insure alignment of the cone to the test section flow. The model was mounted on a transverse strut between the two sidewall model ports. The area blockage was 3.25 percent of the 6 x 22 inch cross section. Local sonic effects on the aft portion of the cone were observed in the steady-state pressure data--due primarily to the three-dimensional effects of the cone flow interacting with the sidewalls.

Unsteady pressure coefficients measured during the cone testing are plotted in F-9 and F-10 for four transducer locations. The sidewall and stagnation chamber data closely duplicated the results reported above. The high $C_{p_{rms}}$ values at low Mach numbers were evident at all transducer locations and the influence of the test section wall resonances appear in the sidewall data for the intermediate Mach

F-10 $C_{p_{rms}}$ vs Mach at Constant Stagnation Pressure

numbers. The forward cone location appears to experience the effects of boundary layer transition. The sharp rise in unsteady pressure levels near transition has been observed in AEDC tests and the range of Reynolds' numbers at the forward transducer location are comparable to the cone transition Reynolds numbers measured at AEDC by Pate (1970). Therefore, the aft cone location is assumed to be in a turbulent boundary layer region and will be used for comparisons with the other facilities. The lower noise levels detected at the aft location are not strongly dependent on Reynolds number as may be observed in comparing F-9 and F-10. However, in the intermediate Mach number range the fore cone data show apparent shifts in the transition location for the constant stagnation pressure data at Mach 0.61 and 0.69, whereas, the constant Reynolds' number data (where the transition should remain fixed) reflect only the additions of resonant peak energies at Mach 0.69 through 0.80. The fore cone data and previous static probe data were both likely influenced by transition since the local axial Reynolds' numbers were of nearly equal magnitudes.

The power-spectral-density plots obtained from analysis of the cone data are not significantly different from those discussed above. At the low Mach numbers, the noise levels are dominated by sources within the stagnation chamber. The spectra all display a broadband maximum near 1000 Hz (attributed to subsonic jet noise in the stilling chamber as hypothesized during the initial noise survey). Resonant peaks again appear at the Mach 0.60 flow condition. The resonant peaks were not as dominant in transducer signals from the cone model. Correlation processing showed that the periodic waveforms generated at the

perforated walls were not generally coupled to the cone model pressures. Longer wavelength disturbances are however observed but the amplitudes and periods are too erratic to describe the behavior as a sinusoidal disturbance. The short wavelengths are rapidly damped (i.e., there is a rapid drop in correlation for the high frequencies). The sharper roll-off near zero time delay and more erratic appearance in the auto-correlation of the stagnation chamber data are indicative of the fact that this data had a greater proportion of random, uncorrelated noise than did the cone data. This is due in part to the instrumentation mounting techniques used in the cone where the transducers were recessed from the surface and coupled via a short, small-diameter hole. Calculated transfer function response of such an installation showed slight amplification of signals near 2500 Hz and some attenuation of signals above 5000 Hz. These non-linearities did not significantly affect measurement of overall levels, but they did tend to poorly filter the high frequency disturbances.

Cross-correlations of the fore and aft cone data reflected, more than anything, the similarity of the separate auto-correlations, since no prominent correlation peaks or strong periodic structures were observed.

The 10° cone tests performed in the OSU-AARL 6 x 22 inch transonic airfoil tunnel show that this facility is acceptable for testing oscillating airfoils at the higher transonic Mach numbers since the ambient unsteady pressure coefficients at those speeds are within acceptable limits. Comparison data from the 10° cone studies in other facilities--data from AGARD-AR-83 (1975)--shows favorable performance for the OSU facility above Mach 0.6 (see F-11). The edge-tones

F-11 Comparison of Unsteady Pressures in Various Transonic Facilities

associated with flow across wall perforations are relatively narrowband in nature and at rather high frequencies in comparison with anticipated airfoil oscillation frequencies of less than 200 Hz. The disturbances in the stagnation chamber appear to dominate at lower Mach numbers; improved flow distribution techniques need to be applied to redesign of the upstream end of the stagnation chamber. The data curve obtained for $C_{p_{rms}}$ versus Mach in the OSU facility is typical of a blowdown facility and large improvements obtained in other blowdown facilities after modification to the 6 x 22 inch tunnel can reduce these ambient levels. The lack of a "transonic noise peak" in the intermediate Mach number range contrasts the behavior observed in several large, continuous-flow, transonic wind tunnels. The relatively small proportion of ventilated wall area required in the two-dimensional tunnel design is a contributing factor to the good performance in this speed region.

Subsequent modifications to the stilling chamber were monitored with the unsteady total pressure probe. The goal was to reduce the level of turbulence and permit lower Reynolds number testing. It was found that drag data on laminar flow airfoils at zero lift conditions was not as low as had been observed in other facilities. This third noise survey was performed both before and after the modifications. The original chamber consisted of two concentric perforated cylinders at the inlet pipe, a perforated bulk head, and four (4) fine mesh screens. The final configuration included an additional perforated cylinder and screen on the inlet, a screen against the perforated bulk head, and replacement of the four screens of various mesh sizes by ten

screens of 60-mesh (about 23 percent porosity stainless steel screen) as seen in F-12 and F-13 (schematics, not scaled drawings). Data was obtained at various stages during the modification work.

The pressure fluctuations were found to be near the previous levels measured for the original chamber configuraion (see F-14). Check points at Mach 0.35 and Mach 0.80 were obtained after the first and second settling screens were removed. The first configuration tested in this series included a squirter screen wrapped around the perforated inlet pipe. This modification had been added since the original survey, but was not documented. It showed a 20 percent to 23 percent improvement in $C_{p_{rms}}$ over the original setup. The original settling chamber design was duplicated by removing the squirter screen and the pressure levels were about 9 percent higher at the low Mach number but were nearly identical at Mach 0.80. A 55 to 63 percent increase in the unsteady pressure coefficient was obtained when the first settling screen was removed, while a 95 to 127 percent increase was noted after two screens had been removed. After the ten new screens were installed, the pressure levels remained 8 to 25 percent above those measured in the original survey. The relatively constant level of the P_{rms} levels over the Mach number range indicate the disturbances may be dominated by the effects of test section turbulent boundary layer noise and perforated wall resonances. No significant effects were observed in the PSD plots from this test series.

The effect of the screens on vorticity fluctuations was not measured directly, however, the relations proposed in Pope and Harper

F-12 Original Chamber (4 Screens)

F-13 New Chamber (10 Screens)

F-14 Total Probe Fluctuations During Settling Chamber Modifications ($C_{p_{rms}}$ vs Mach)

(1966) indicate that the turbulence reduction by screens is as follows:

$$\frac{U'_3}{U'_1} = 1/(1 + k)^{n/3} \quad (2.1)$$

where

U'_1 = entrance velocity fluctuations,

U'_3 = exit velocity fluctuations (downstream of the damping region
for screen-induced, fine-grain eddys),

n = number of screens, and

k = dynamic pressure drop across one screen.

This gave reductions in velocity fluctuations ranging from 65 percent from two screens to 99.5 percent for ten screens. Therefore, the improvement in the airfoil drag measurements apparently resulted from reduced velocity fluctuations rather than any change in the acoustic levels measured.

The noise surveys of the OSU/AARL 6 x 22 inch airfoil tunnel indicated that it was suitable from an ambient noise viewpoint for testing oscillating airfoils. Signal-to-noise ratios of at least 10:1 were observed during subsequent testing. The facility exhibited high unsteady pressure coefficients at the lowest Mach number but did not show a large resonance influence in the mid- to upper-subsonic range. The aft 10° cone data showed that above about Mach 0.50, the tunnel was relatively quiet and ranked favorably with other blowdown and continuous-flow facilities. Edge-tone oscillations were observed at frequencies far above those for the oscillating airfoils.

2.2 MODEL DESCRIPTIONS

Several models and probes were fabricated for the oscillating airfoil experiments and tunnel noise surveys. The total and static transducer probes were designed to provide freestream measurements of the unsteady pressure field. A 10° cone model containing two pressure transducers was made to specifications of the AEDC 10° cone dynamic calibration model. Two NACA 64A010 airfoil models of 3-inch (7.6 cm) and 6-inch (15.2 cm) chord lengths were constructed. An oscillating airfoil drive system was assembled and mounted on the 6 x 22 inch wind tunnel. Design details of the experimental test hardware are given below.

Both the total and static pressure probes were used during the initial noise surveys. The total probe was installed to monitor the stilling chamber modifications. The static probe was relocated near the upper perforated wall during the airfoil test series. Additional transducers were mounted on the tunnel sidewalls, perforated wall, and outer plenum wall to monitor pressures during oscillating airfoil tests. The probes were instrumented with Kulite XCQL-14-093-25 transducers (as shown in F-15). The total probe was a 1/8-th inch diameter tube with the Kulite transducer flush-mounted in one end. The transducer in the static probe was mounted twelve (12) diameters back from the elliptical nose. A single port and minimum volume cavity coupled the transducer to the tunnel pressures. Both probes extended twenty-five diameters upstream of their faired mounting struts. The struts held the probes 2.0 inches (5.08 cm) from the tunnel wall so that when mounted in the sidewall model support windows the total and static probes were 2.0 inches (5.08 cm) apart. The probes were

F-15 Total and Static Transducer Probes

mechanically leveled and aligned with the tunnel walls during installation. The probes were fixed in position during the test runs. The upper wall installation was also spaced 2.0 inches away from the wall. A photograph (see F-16) shows the static probe mounted in a section of the perforated upper wall. Also shown are transducer mounts in the sidewall window and the outer plenum wall. These probes and tunnel wall transducers provided data on the ambient tunnel noise levels as discussed in section 2.1 above and on pressures radiated from the oscillating airfoils as discussed in section 5.5 below.

The 10° cone model was mounted on a faired strut which spanned the tunnel width between the sidewall ports. The solid brass model was machined to a fine finish and then hand polished to a smooth surface. Actual surface finish measurements were not made but the nominal finish was determined visually to be equivalent to a fine surface grind with a surface texture in the range of 8 to 16 microinch. The nose tip radius was examined visually to be 0.005 inch radius (see F-17). The Kulite transducers were mounted as shown with a single passage to the model surface. The two halves of the cone were rough machined and assembled with a roll pin prior to final machining and polishing. Eight steady-state static pressure taps were mounted in the model including four taps at the 6.0 inch station to help align the cone to the flow. The indicated range of boundary layer transition was based on correlation with data from the AEDC cone model [see Pate (1969)]. The Reynolds number at the forward transducer location (based on axial length) were between 1.4×10^6 and 3.2×10^6 while the aft location was at values of 3.9×10^6 to 8.7×10^6 . Use of the AEDC cone

F-16 Static Probe and Tunnel Transducer Installations

F-17 10° Cone Model

model was precluded due to its size (8.6 inch base diameter and 49.05-inches long). The OSU/AARL cone model was made as a replica of the AEDC cone, resized to fit into the 6 x 22 inch tunnel.

The airfoil models were two-dimensional NACA 64A010 profiles spanning the six-inch tunnel width. The airfoil coordinates were computed by a NACA 6-digit airfoil generation program used to compute surface contours for the 3-inch (7.6 cm) and 6-inch (15.2 cm) chord models. These surface coordinates were examined to remove any undesirable slope inflections and then input to a computer program to generate the appropriate milling-ball coordinates for machining the airfoil contours. The models were cut from brass stock and hand-polished (as above to about a 10 microinch surface finish). The final coordinates were hand-checked for accuracy.

Both models were driven by a 1" x 1" x 5" block of material contiguous with the airfoil contour section. This block mounted in a rotating sidewall port connected to the model drive system (see F-18 and F-19). The non-driven end of the 3-inch model was formed as another contiguous block of the brass material. It mounted in a steel collar which rotated inside a bearing in the sidewall port opposite the drive system. The six-inch model was supported on the non-driven side by a steel insert which fit into the end of the airfoil model. The insert also had a shaft sized to fit into the sidewall bearing. Both models were instrumented with steady-state static taps and Kulite transducers. The static taps and transducer ports were drilled spanwise through the airfoil model and then cross-drilled normal to the surface contour. The static pressures were transmitted through

F-18 NACA 64A010 Airfoil Model (3-Inch Chord)

NOTE: CUT CONTOUR TO MILLING-BALL COORDINATES.
MAT'L: BRASS

F-19 NACA 64A010 Airfoil Model (6-Inch Chord)

stainless steel tubes bonded into the model ports. The transducers were inserted into the model and bonded in place to minimize the acoustic coupling volume. The tubes and transducer leads were routed through the sidewall bearings for external hookup. The photographs in F-20 show the six-inch chord model mounted in the sidewall ports. The lower view shows the tunnel installation with pressure tubes routed through the non-driven sidewall. The driver sidewall port contains a large diameter, double-row, spherical roller bearing which accommodated the large brass block on the end of the airfoil model. The 8-inch face plate rotated with the model.

The model instrumentation locations were chosen so that there were matching chord positions in both models for steady-state taps and pressure transducers. The locations of unsteady pressure transducers are shown in F-21 for both models. The larger model contained thirteen transducers with a closer spacing in the 40 to 70 percent chord region to more accurately resolve the local shock influence at supercritical Mach numbers. Only seven transducers were installed in the 3-inch model due to space limitations. The thin trailing edge region was difficult to instrument and special transducers were mounted flush to the surface at the 90 percent chord location of the 6-inch model. There were 21 steady-state pressure taps in the small model and 30 in the larger one. Both upper and lower surface pressures were measured by the static taps with increased emphasis on the upper surface. The in-situ transducers read only upper surface pressure fluctuations and were each paired with a static tap to register steady-state pressures (for fixed angle-of-attack tests). All taps and transducer ports were

F-20 NACA 64A010 Model Photographs (6-Inch Chord)

F-21 Unsteady Pressure Locations

located as close to mid-span as possible (the only notable exception was the trailing edge tap bonded to the 6-inch model out at 1.5 inches from the sidewall).

Additional sidewall ports located 3 and 9 inches above the model were instrumented during the testing of the 3-inch chord model. These ports were instrumented with unsteady transducers and static taps. The region near the upper plenum was instrumented during both small and large airfoil model tests. The unsteady static pressure probe used during the tunnel noise survey was installed on centerline, directly above the airfoil midchord position. It was supported 2.0 inches (5.08 cm) below the perforated wall at the same 9.0 inch height above the model as the upper sidewall transducer. Directly above the probe, an unsteady transducer and static tap were flush-mounted in the perforated wall (facing the test section). Another flush-mounted transducer and static tap were mounted in the upper plenum wall.

The model oscillations were produced by an eccentric crankshift system (see F-22) driven by a 5-horsepower, alternating-current, electric motor. The motor speed was constant at 1760 rpm. The oscillation frequencies were controlled by changing the pulley size ratio for the V-belt drive. The crankshift was counterbalanced for minimum vibration and was supported by two high-speed ball bearings. A piston-rod drive link oscillated up and down on the eccentric cam and drove the model positioning arm. The arm directly drove the sidewall port and airfoil model. As stated above, the model was supported by two bearings: a large, double-row, spherical roller bearing set on the driver side (resistant to axial bearing forces normal to the tunnel

F-22 Model Drive System

wall and a smaller double-row, angular-contact ball bearing set on the non-driven side also resistant to axial loads). The model motion amplitude and mean angle-of-attack were set by positioning the crankshaft at various distances and locations with respect to the model rotation axis. The model oscillation frequency and phase were determined from an accelerometer attached to the model positioning arm. A reference arm is attached to the model drive shaft and was used to statically check model position for the fixed angle of attack tests to manually set the model oscillation limits. The drive system rigidity provided a pure sinusoidal model pitching motion about the mid-chord axis at frequencies up to 120 Hz. The airfoil moment of inertia ($I_{xx} = 4.8 \text{ in.}^4$) was small compared to that of the sidewall port and drive system and the rigidity of the solid airfoil model was sufficient to prevent significant torsional bending in the airfoil section.

This drive system was operated over a frequency range of 19.5 Hertz to 120 Hertz. The operating frequency was checked by analyzing the accelerometer output during pre-run checkout of the model drive setup procedures. These tests showed that the oscillation frequency was stable and subsequent spectral analyses showed no evidence of higher harmonic components in the airfoil motion or of broadening of the fundamental peak associated with frequency shifting. The potential of phase lag due to torsional bending of the drive elements or airfoil model were limited to a bending of the main drive shaft and driver arm. That phase lag was computed to be less than one degree (1°) at the highest frequency tested. Instrumentation problems encountered

during testing of the 3" chord model necessitated installation of an additional buffer amplifier to reduce noise levels from the accelerometer. This change apparently introduced a systematic phase error in the model oscillation reference signal which was corrected for during data analysis.

2.3 DATA ACQUISITION AND RUN PROCEDURES

Procedures and computer control software were developed to provide sensor calibrations, sequence tunnel operations, acquire data, and provide data reduction support. The computer routines were written in FORTRAN for the Harris SLASH 6 computer system at OSU/AARL. The Purdue Extensions to FORTRAN were used to handle the analog/digital interface operations. Computer routines originally developed for steady-state airfoil testing in the OSU/AARL facilities were adapted and modified for use in the unsteady aerodynamic testing.

Pretest pressure calibrations were performed on all steady-state and unsteady pressure sensors. Water and mercury manometers were used to calibrate transducers over the appropriate ranges rather than using traceable standard instruments. These manometer calibrations were conducted with the aid of an on-line calibration routine. The transducers were thus calibrated through the entire signal conditioning, digitizing, and computer system. As a part of the calibration sequence, electrical resistances (R-cals) were automatically placed in the strain-gage bridge circuit to provide calibration values for the on-line run calibrations. A special routine was written to calibrate the Kulite transducers together through a common manifold of their reference pressure tubes. An average of ten

to twelve calibration points were set and then digitized, with a least squares linear fit to the data used to provide bias and sensitivity values (zero and slope).

Steady state measurements are made with individual transducers for monitoring and measuring the total pressure, total temperature, and static pressure while a trapped-volume Scanivalve system is used to record model pressures and additional static sidewall taps. Rotary guillotine cutoff valves are used to capture pressures during the run, and post-run cycling of the multiport Scanivalve system allows sequential readout of the pressures via direct digitizing by the on-line computer system. Corrections for volume carry-over are accomplished by pre-calibrations and software data processing. Some individual pressure sensors were monitored continuously during the test run to provide a time history of the flow conditions. Others were recorded during certain data operations such as the wake traverse scans to measure pitot pressure behind the airfoil models.

The in-situ, subminiature Kulite pressure transducers were used throughout the noise surveys and oscillating airfoil tests. Figure 23 shows a schematic of the unsteady instrumentation system. The analog instrumentation system at OSU/AARL includes several pairs of signal conditioning units and amplifiers. The signal conditioners provide a stable d.c. transducer excitation and were used in the constant voltage mode for these experiments. Balancing and zeroing circuits are also included in the signal conditioner along with an interchangeable bridge completion circuit card. This card contains the four (4) calibration resistors for the automatic, R-cal, on-line calibrations. The

F-23 Unsteady Instrumentation Schematic

amplifiers were d.c. - wideband differential amplifiers using integrated circuit designs. The multiple gain levels were manually controlled over a gain range of unity to 1000. An RC circuit which served as a high-pass filter with a cutoff frequency of about one Hertz was used to remove any unbalanced bias during the data acquisition runs.

A multiplexer relay box was used to allow recording of up to twenty channels of data with only five sets of signal conditioners and amplifiers. This hardware was used extensively throughout the oscillating airfoil tests. The relay unit switched the output signal leads of each of four transducers per channel upon commands from the AARL computer system. The d.c. excitation was not switched, thus avoiding transients coupling to the regulated power supplies and transducers. The reed relays were multiplexed to individual buffer amplifiers for each output channel. The sixteen unsteady transducers were multiplexed through the relay unit to four output channels. The accelerometer reference channel was not multiplexed through the unit. Four Kulite transducer signals plus the accelerometer were sampled during each of the four data windows covering a total period of about six seconds. Sufficient time was allowed between relay stepping to buffer the previous data from the computer memory to a temporary disc file and then collect a new set of unsteady data. This system permitted acquisition of four times the amount of data than could normally be accommodated by the main computer memory.

The analog/digital interface allowed direct digitizing through two A/D converter systems and a series of digitally controlled output

relays. The relays were used to sequence the tunnel operations; i.e., they controlled the R-cal switching, energized control circuits to operate the large air valves to run the tunnel, provided stepper signals for cutoff valves and Scanivalves, traverse control of the wake survey probe, and sequencing of the relay multiplexer unit used for the Kulite transducers. A low-level, auto-ranging RTP digitizing system was utilized to record the quasi-steady signals associated with monitoring the tunnel test conditions, post-run readout of the Scanivalve data, and recording of the wake probe traces. The throughput of this system is about 8000 samples per second. The second A/D conversion system is a Datel 256 high-speed multiplexer/digitizer with an overall sampling rate of about 110 Kiloherzt. The Datel device was operated in the Direct Memory Access (DMA) mode to allow full speed throughput of multiple channels. This allowed high-speed transfer of the digital data from the Datel data registers to the main computer memory on a first priority basis.

The digital computer system at OSU-AARL uses a Harris SLASH 6 central processing unit. A Universal Common System is used to provide part of the analog/digital interface. It includes D/A conversion units, external interrupt circuits, external relay switches, and real time clocks. It operates through a dedicated Universal Common Block located in the main computer memory and activated by changes in the binary bit patterns stored there. The processor is a 24-bit machine with 64K words of semiconductor memory using a high-speed, floating-point arithmetic processor. The operating system is a Disc Monitoring System capable of foreground/background processing,

input/output spooling operations, graphics support, and high-level language processors such as FORTRAN and BASIC. The input/output channels are individually microprocessor-controlled to allow independent buffering operations. Peripheral terminals are allowed on-line, interactive editing privileges through the re-entrant ACRONIM processor modules. Link editor services are provided along with an overlay cataloger to allow execution of large background jobs. Communications software can transmit jobs to IBM and CDC computer centers over a 2000 baud telephone link.

Peripheral equipment includes various input/output devices and mass storage units. Two high-speed disc drives provide a combined mass storage capacity in excess of 90 megabytes. Two digital tape units allow storage of archival records for data and disc files. A medium-speed line printer and a hard-copy drum plotter provide the permanent output records. Several keyboard terminals including alpha-numeric CRT's, teletypewriters, and graphic display CRT's are available for operator communications and file editing tasks. A punched card reader and paper tape reader provide input functions in addition to the two digitizer systems described above.

The data acquisition and processing software consists of a series of re-entrant foreground programs which are interrupt initiated and controlled. Each on-line routine communicates to the other routines through master disc files and a Universal Common Block called SYSCOM which was located in the main computer memory. The operator controls the sequence of programs and provides inputs for run options and calibration data through a graphic CRT display terminal and external

interrupt switches. The interrupt processing functions are handled by a resident foreground routine called SORTIL. It processes the interrupt requests while releasing most of the core memory for background processing when the operator is waiting for model changes or other tasks not requiring on-line dedication of the computer resources. The routines used in the on-line data acquisition and setup tasks are described in detail in Appendix A.

Additional data processing functions are provided as off-line routines using data retrieval from the magnetic tape files. These routines are executed in an interactive mode where the operator can actively select data processing options, display the graphical results, and construct output spool files for the hard-copy plots and tabulated printout.

These routines were developed for the specific purpose of acquiring and analyzing data from the oscillating airfoil tests. However, a wide-range of applications exist for these data acquisition and processing procedures.

2.4 UNSTEADY PRESSURE DATA ACCURACY

The analysis of the confidence bands which must be applied to data is often as important as the actual data values obtained. The accuracy of the unsteady pressure data is a combination of an entire train of error biases. A detailed discussion of the experimental errors encountered during these tests and the resultant measurement uncertainties is given in Appendix E.

The transducer calibration data showed that sensitivities were determined to within a ± 2 percent band. Periodic recalibration of the

transducers ensured that the sensor characteristics did not change over long periods of time. Pre-run and post-run electrical calibration checks reduced the errors due to signal conditioning instrumentation variations to near negligible levels. The subsequent digitizing and processing errors were also small since there was a wide dynamic range at the digitizer interface and careful consideration of computer roundoff error provided algorithms with minimal error accumulation. The overall uncertainty in the unsteady pressure measurements was determined to be approximately ± 5 percent, which was deemed acceptable for this research.

The steady-state pressure instrumentation was calibrated in a similar manner using periodic pressure calibrations and pre-run and post-run electrical checks. The Scanivalve, trapped volume system has been thoroughly characterized by multiple-scan checkouts and calibrations. The instantaneous trapping of all steady pressures with the rotary guillotine valves avoids errors associated with the transient pressure history of a blowdown tunnel facility. This data system has been proven to be highly accurate and produces data of consistently high quality. The data accuracy coupled with the low steady-state interference are presented in detail by Lee, et al. (1978).

The statistical accuracy of the unsteady data is based primarily upon the sampling criteria used to obtain the raw digital data. For these tests, the data sampling concentrated on obtaining results associated with the airfoil oscillations. The sample size was based on a minimum acceptable number of data points to get reliable data and a maximum limit imposed by the available computer memory. The selected

sample size was 4100 points per channel gathered at a 4 Kiloherzt rate. For each single run, twenty channels were digitized, producing 82,000 samples to be analyzed. The relative accuracy of the PSD processing is limited by the effective bandwidth based on the number of samples and total sample period. For single frequency data (associated with the model oscillation) the confidence band limits for the spectral estimates do not give sufficiently accurate results. Therefore, the complex demodulation processing technique was used to obtain the oscillatory pressure coefficients C'_p . The principal error for that algorithm was in determining the exact frequency of oscillation. Therefore, an optimizing routine which corrected the demodulation frequency for minimum error was developed. The subsequent error was reduced to less than 1 percent.

These experimental errors are within acceptable limits. The reliability of the unsteady data obtained during the course of these studies has been determined to be within a ± 5 percent band of the data reported. The steady-state data has previously been demonstrated to be of even better accuracy.

III. ANALYTICAL BACKGROUND

Experimental simulation of transonic airfoil characteristics - both steady and unsteady phenomena - can be performed in suitable wind tunnels. That simulation requires matching non-dimensional parameters based on full-scale conditions. The major problem in transonic testing is often centered on attaining adequate Reynolds number conditions on subscale models. Such models are restricted to small sizes so as to avoid the influence of the tunnel flow boundaries on data. Generally, this interference is empirically or analytically estimated and corrections are applied to the data or effective test conditions to reduce the interference effects on the data.

3.1 FORMULATION OF THE SMALL-DISTURBANCE EQUATION

The time-dependent, transonic, small-disturbance potential equations are derived in general by applying simplifying assumptions to the basic fluid flow equations. Investigators have derived several approximating forms of the fluid flow equations applied to specific cases and numerical methods. Since the problem of analyzing an oscillating transonic airfoil involves the time-dependent terms in addition to the gradients in the space coordinates, solving the two-dimensional unsteady equation is tantamount to solving a three-dimensional steady flow problem. Therefore, solutions of the

transonic unsteady airfoil flow have been obtained primarily from simplified small disturbance potential flow equations.

The primal formulation for the problem could be the Kinetic Gas Theory and statistical fluid mechanics. However, it is quite reasonable to make the assumption that the mean free path is insignificant in comparison with the smallest dimension over which a change can occur. This continuum assumption will allow us to use the Navier-Stokes approach to describe the fluid motions. The set of equations required (continuity, momentum, energy, and thermodynamic state equations) is quite formidable. For typical aerodynamic problems in the transonic flow regime, Table 1 contains a complete set of equations for the solution of two-dimensional, unsteady gas flow problems. The closure models listed are based on the assumptions of thermally- and calorically-perfect gases and an approximate viscosity law proposed by Sutherland. While these equations are complete field equations, in practice, applications to turbulent flow cases necessitate additional turbulence closure models for ensemble-averaged forms of the Navier-Stokes and energy equations. There are no inherent inaccuracies in the formulation in Table 1 but resolving the fine scale fluctuations over relatively short time steps would require astronomical computer memories and near infinite computational rates (even by current standards). It is therefore difficulties with the numerical solution issues that dictate the use of simplified formulations.

TABLE 1. GENERAL FLUID FLOW EQUATIONS
(Space Coordinates x and y, Time Coordinate t)

Continuity (Conservation of Mass)

Navier-Stokes (Conservation of Momentum)

Energy

$$\begin{aligned} \frac{\partial}{\partial t} \begin{pmatrix} \rho \\ \rho u \\ \rho v \\ \rho e \end{pmatrix} + \frac{\partial}{\partial x} \begin{pmatrix} \rho u \\ \rho u^2 + p \\ \rho uv \\ u(\rho e + p) \end{pmatrix} + \frac{\partial}{\partial y} \begin{pmatrix} \rho v \\ \rho vu \\ \rho v^2 + p \\ v(\rho e + p) \end{pmatrix} = - \frac{\partial}{\partial x} \begin{pmatrix} 0 \\ \tau_{xx} \\ \tau_{xy} \\ q_x + (u\tau_{xx} + v\tau_{xy}) \end{pmatrix} \\ - \frac{\partial}{\partial y} \begin{pmatrix} 0 \\ \tau_{yx} \\ \tau_{yy} \\ q_y + (u\tau_{yx} + v\tau_{yy}) \end{pmatrix} + \rho \begin{pmatrix} 0 \\ g_x \\ g_y \\ ug_x + vg_y \end{pmatrix} \end{aligned} \quad (3.1)$$

where

- $\rho \equiv$ fluid density
- $u \equiv$ x-velocity
- $v \equiv$ y-velocity
- $e \equiv$ internal energy
- $p \equiv$ thermodynamic pressure
- $\mu \equiv$ dynamic viscosity

Stress tensor:

$$\tau_{xx} \equiv -2\mu \frac{\partial u}{\partial x} + \frac{2}{3} \mu \left(\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} \right) \quad (3.2)$$

$$\tau_{xy} = \tau_{yx} \equiv -\mu \left(\frac{\partial u}{\partial y} + \frac{\partial v}{\partial x} \right) \quad (3.3)$$

$$\tau_{yy} \equiv -2\mu \frac{\partial v}{\partial y} + \frac{2}{3} \mu \left(\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} \right) \quad (3.4)$$

TABLE 1 (Continued)

where

$q_x, q_y \equiv$ heat transfer rates

$k \equiv$ Fourier heat conduction coefficient

$T \equiv$ thermodynamic temperature

$C_v \equiv$ specific heat at constant volume

$R \equiv$ gas constant

$g_x, g_y \equiv$ body force (gravitational) components

Closure formulas:

Equations of State

$$\text{Ideal-gas law: } p = \rho RT \quad (3.5)$$

$$\text{Caloric equation: } e = C_v T \quad (3.6)$$

Fourier heat conduction law:

$$q_x = -k \frac{\partial T}{\partial x} \quad q_y = -k \frac{\partial T}{\partial y} \quad (3.7)$$

Sutherland's law (simplified power law)

$$\mu = \mu_{\text{ref}} \left(\frac{T}{T_{\text{ref}}} \right)^{0.76} \quad (3.8)$$

The first simplification reduces the formulation to the Euler equations under the assumptions that:

- (1) Viscosity effects are negligible--this is accurate for the major portion of the flowfield external to the boundary layers and near wake regions;
2. Heat transfer rates are negligible--only for high supersonic flow conditions or very slow flow convection cases do heat fluxes play an important role; and
3. The body forces are insignificant--again, only in slow flow convection cases or meteorological analyses are these terms regarded.

These assumptions reduce the main transport equation to a homogeneous, partial differential equation as follows (the Euler equation):

$$\frac{\partial}{\partial t} \begin{pmatrix} \rho \\ \rho u \\ \rho v \\ \rho e \end{pmatrix} + \frac{\partial}{\partial x} \begin{pmatrix} \rho u \\ \rho u^2 + p \\ \rho uv \\ u(\rho e + p) \end{pmatrix} + \frac{\partial}{\partial y} \begin{pmatrix} \rho v \\ \rho vu \\ \rho v^2 + p \\ v(\rho e + p) \end{pmatrix} = 0 \quad (3.9)$$

with

$$e = \frac{p}{\gamma - 1} + \frac{1}{2} \rho(u^2 + v^2) \quad \text{Note: } \gamma \equiv \text{specific heat ratio.} \quad (3.10)$$

This formulation was solved by Magnus (1977) in a time-marching, finite-difference computation for an oscillating airfoil. However, the large computing costs make this approach currently uneconomical for

analytical studies of several flow conditions and frequencies. At this stage, we have retained all the salient features of the shock discontinuity in the mathematical formulation, however, numerical techniques are generally unstable when such discontinuities* are present. Numerical diffusion or artificial viscosity are often used to stabilize the finite-difference solution methods at the cost of a sharp resolution of the shock discontinuity. Furthermore, the effects of unsteady shock/boundary layer interaction are not treated. For steady flow, Euler computations, boundary layer displacement corrections could be applied using integral methods and the predicted external flow velocities and pressures, however, such efforts have not yet reached fruition for the unsteady case.

The next stage of simplification uses the assumptions of irrotationality and isentropic flow to reduce the system of transport equations to a single second-order, partial differential equation. Irrotationality of a flow is mathematically defined as the case where the curl of the velocity vector is zero throughout the field ($\partial V/\partial x - \partial U/\partial y = 0$). This is also called the conservation of vorticity in that it implies that vorticity is neither generated or dissipated in the field. This condition is consistent with the assumptions of negligible frictional forces or body forces invoked for the Euler equations but these are necessary, not sufficient conditions. The isentropic

*These discontinuities arise from the mathematical formulations and numerical algorithms. They are not a violation of the original continuum assumption although the rapid change in properties across the shock can take place in a relatively few mean free paths giving the appearance of a physical discontinuity.

criterion reduces the equations of state to the polytropic relation having the specific heat ratio as an exponent ($p/\rho^\gamma = \text{constant}$).

Finally, introduction of the scalar velocity potential ($\vec{v} = \nabla\phi$) where

$u = \phi_x$ and $v = \phi_y$ results in the full potential equation:

$$\left(1 - \frac{\phi_x^2}{a^2}\right) \phi_{xx} + \left(1 - \frac{\phi_y^2}{a^2}\right) \phi_{yy} - 2 \frac{\phi_x}{a^2} \phi_y \phi_{xy} - 2 \frac{\phi_x}{a^2} \phi_{xt} - 2 \frac{\phi_y}{a^2} \phi_{yt} - \frac{1}{a^2} \phi_{tt} = 0 \quad (3.11)$$

where the isentropic speed of sound is defined as:

$$a^2 = a_\infty^2 - (\gamma-1) [u_\infty \phi_x + \phi_t + \frac{1}{2} (\phi_x^2 + \phi_y^2)] \quad (3.12)$$

This formulation was applied to unsteady airfoil cases by Isogai (1978) using a finite-difference procedure and shock capture method. The isentropic assumption is invalidated by finite strength shock waves. Therefore, it becomes necessary to provide shock-capturing tools in order to re-introduce the inherent non-linearity of the shock process. This is most often done by invoking a mixed-differencing technique where upwind differencing is applied in supersonic regions with central differencing in subsonic zones. In addition, the Rankine-Hugoniot conditions may be applied at the shock location. Steady potential flows may be analyzed by deleting the time-dependent terms from the full potential equation as was done by Bauer, et al. (1972) for computing transonic flows about super-critical airfoils.

Small-disturbance theory is next used to further simplify the potential equation. For thin airfoils at low pitch angles and small

oscillation amplitudes, the assumption that the x-component of velocity varies negligibly in the y-direction for conditions near Mach 1.0 eliminates the ϕ_{xy} term. The other assumption used here is that the y-velocity is small compared to the x-component. This criterion is satisfied by essentially collapsing the profile onto the mean chord line and establishing boundary conditions consistent with the surface slope of the original contour. This creates singularities at the leading and trailing edges of the airfoil which must be treated. Clustering grid points in these regions and in the shock zone help overcome the numerical stability problems associated with this approach. The general form of the transonic small-disturbance equation is obtained by neglecting terms containing the ϕ_y term and combining the acoustic velocity relation with the remaining terms:

$$(1 - M_\infty^2 - M_\infty^2 \frac{\gamma+1}{u_\infty} \phi_x) \phi_{xx} + \phi_{yy} - 2 \frac{M_\infty^2}{u_\infty} \phi_{xt} - \frac{M_\infty^2}{u_\infty^2} \phi_{tt} = 0 \quad (3.13)$$

This equation was analyzed in general by Rizzetta and Yoshihara (1980) while additional restrictions to low frequency oscillations eliminates the ϕ_{tt} term as in Ballhaus and Goorjian (1978). The zero frequency or steady case was analyzed by Murman and Cole (1971).

Time-linearization of the transonic, small-disturbance equation is accomplished with the use of a harmonic disturbance potential. The scalar velocity potential is considered to be a linear combination of the steady-state potential ϕ_0 and a scalar harmonic component, $\phi_1 e^{i\omega t}$. This relation is based on the assumption that the airfoil oscillations are infinitesimal and the pressure oscillations are sinusoidal and of amplitude linearly proportional to the amplitude of

the airfoil motion. This restriction can be stated as the condition that the amplitude of oscillation be much smaller than the airfoil thickness and mean pitch angle (note: the thickness and pitch parameters are still limited by the conditions of the small-disturbance formulation given above). Introducing the reduced frequency parameter, $k = 2\pi fc/u_\infty$, for an airfoil of unit chord, the linearized, transonic, harmonic-potential equation can be stated as:

$$\begin{aligned} (1-M_\infty^2 - M_\infty^2 \frac{\gamma+1}{u_\infty^2} \bar{\Phi}_{0xx}) \phi_{xx} + \phi_{yy} - 2ikM_\infty^2 \phi_x + k^2 M_\infty^2 \phi \\ - (\gamma+1) \frac{M_\infty^2}{u_\infty^2} \bar{\Phi}_{0xxx} \phi_x = 0 \end{aligned} \quad (3.14)$$

This approach was implemented by Traci, et al. (1975) to analyze the unsteady pressures on airfoil oscillating in transonic flow. It relies upon having a reliable steady-state solution for the nonuniform transonic flow. It theoretically does not account for shock motion relative to the airfoil and contains only single frequency oscillations corresponding to model motion. The assumptions of linearity may be invalidated by attempts to analyze anything other than small reduced frequencies. The method was found to be stable only at frequencies below a guideline defined as an inverse function of Mach number and thickness. Therefore, the range of calculable frequencies tended to near zero at sonic freestream conditions. The solution relies on a steady-state potential function, and, hence, the accuracy is highly dependent on shock strength and placement for the steady case. As before, the inviscid solutions without boundary layer displacement corrections or proper shock-capturing tools can be highly inaccurate

for supercritical flow conditions. An added complication of the harmonic potential method is that complex mathematics are introduced. The real and imaginary parts of the potential function represent in-phase and out-of-phase components. The fixed shock location and sinusoidal disturbance assumption do not allow detailed studies of the unsteady pressure loads in the region of the oscillating shock.

The final stages of simplification provide us with the lifting-surface methods as described by Giesling, et al. (1976). These methods are generally applied in strictly subsonic or strictly supersonic cases. The appropriate elliptic or hyperbolic operators can be implemented in an integral formulation. Such methods cannot handle discontinuities within the range of integration; therefore, either shock-free regions over the airfoil must be present or a complex integration domain with a cutout around the shock must be constructed. The condition of a shock-free flow over the airfoil also exists for a narrow speed range near Mach 1.0 where Landahl (1961) presented a formulation for high reduced frequencies such that:

$$k \gg |1 - M_\infty| \quad (3.15)$$

The "acoustical" equation is written as:

$$\phi_{yy} - 2ikM_\infty^2 + k^2M_\infty^2\phi = 0 \quad (3.16)$$

With suitable boundary conditions, this method can still give reasonable results for thin airfoils in the near-sonic speed range.

The hierarchy of unsteady, transonic flow models described above are progressively simplified by qualifying assumptions. The general

accuracy and utility decreases as the equations become simpler to solve. A compromise between computational effort and accuracy must be attained if these formulations are to be used in engineering applications. The transonic, small-disturbance potential theory appears to have achieved that status in light of current computer capabilities and numerical analysis techniques. Therefore, a major portion of the efforts in transonic, oscillating airfoil analysis are performed with the finite-difference methods based on the transonic, small-disturbance potential equations. However, as with any of these mathematical approaches, the solution is dependent on the imposed boundary conditions. The appropriate modelling of the boundary conditions is often of greater difficulty than setting up the primary solution technique.

Lighthill (1972) discussed the various processes associated with sound propagation in a moving field. He classified solutions to the potential equation according to the amplitude of the disturbance and its wavelength relative to the source dimensions. For the oscillating airfoil, moderate wave strengths are anticipated and the wavelength of the pressure wave associated with the airfoil oscillations considered here are far greater than the airfoil chord length. Lighthill described such cases as "acoustically compact" flow fields, and he suggested modelling this problem as a point monopole of specified volume outflow (source flow) and a joint dipole of a given strength (associated with the Theodorsen function) moving at constant velocity relative to the medium. The radiating sound field was that determined from geometric acoustics and scattering from the dipole source.

3.2 FORMULATION OF BOUNDARY CONDITIONS

The uniqueness of a solution to a given differential equation such as the unsteady potential equation is determined by the boundary conditions. The tunnel interference problems are essentially generated by a mismatch between the conditions at the tunnel walls and the equivalent free-air conditions that would exist at the same location. Ideally, the free air boundary conditions are specified as asymptotic formulas for large distances away from the airfoil.

The homogeneous boundary conditions for various tunnel boundaries are outlined by Gopinath (1976). The free-jet boundaries are generally modelled as constant pressure lines. A closed-wall tunnel is specified by setting the normal component of the velocity at the wall equal to zero. The ventilated wall designation is generally formulated as an equivalent homogeneous boundary following development by Baldwin et al. (1954). The slotted wall condition is a combination of the closed wall and open jet boundaries of the form:

$$\phi_x + K \phi_{xn} = 0 \quad (3.17)$$

where

$$K = (d/\pi) \ln[\csc(\pi a/2d)] ,$$

a = slot width, and

d = slot spacing .

This relation establishes a compatibility condition between the streamwise velocity and the streamwise gradient of the normal velocity component at the wall taken as a homogeneous boundary rather than alternating open and closed areas. The perforated wall case is expressed as a compatibility condition between the streamwise and

normal velocity components:

$$\phi_x + \phi_n/R = 0 \quad (3.18)$$

where R is an empirically determined porosity factor.

In small-perturbation theory, the airfoil surface slopes are projected onto a mean chord line and these slopes constitute the airfoil boundary condition. For the unsteady airfoil case, the slopes are designated as having oscillating pitch components to simulate airfoil motion. The trailing edge and wake conditions can be modelled as a series of vortices satisfying an unsteady Kutta condition, although it is somewhat uncertain that at higher reduced frequencies an "unsteady Kutta condition" can be rigorously enforced.

For the OSU/AARL test setup a homogeneous boundary condition at the ventilated wall can be based on an acoustic impedance analysis of the wall holes. Ronneberger (1972) hypothesized that orifice flow resistance (the real part of the complex impedance) for low-frequency, quasi-steady cases is directly proportional to the flow velocity across the orifice. This assumes that the boundary layer displacement thickness is significantly less than the hole diameter. Hersh (1976) found that for moderate sound pressure levels, the orifice impedance is a weak nonlinear function of jet effects. At greater sound pressure levels, the flow through the wall perforations would correspond to a predominantly quasi-steady behavior characterized by strong, nonlinear jet effects. For intense noise levels, the effective orifice Reynolds number is large and the viscous effects of the jet can be ignored. Hersh suggests use of a time-averaged discharge coefficient for the

case of a resistance dominated orifice impedance. Hersh (1979) further examined clustered orifices and found that interaction between the holes occurred only for hole centers aligned parallel to the grazing flow. The resistance and reactance of the wall holes are relatively insensitive to orifice spacing or number. However, the orifice and correction is dependent on spacing dimensions for closely clustered holes because of changes in the apparent mass contribution. The reactance was observed to be decreased in grazing flow cases.

The equivalent homogeneous boundary condition for the OSU/AARL 6 x 22 inch tunnel walls can be deduced from previous work in steady and unsteady interference. The experimentally determined steady-state interference is relatively small as noted in the next section. The well known "equivalent homogeneous boundary condition", was introduced by Baldwin, et al. (1954) as:

$$\phi_x + K \phi_{xy} + \frac{1}{R} \phi_y = 0 \quad (3.19)$$

where

x = streamwise coordinate

y = coordinate normal to the tunnel wall,

K = wall geometry factor, and

R = porosity parameter .

For a perforated wall, K was taken to be zero. Hence, the porosity parameter can be solved for as:

$$R = - \left. \frac{\phi_y}{\phi_x} \right|_{\text{wall}} = \frac{2v/u_\infty}{\Delta p/q_\infty} \quad (3.20)$$

where v = equivalent average velocity normal to the wall ,

Δp = pressure drop across perforated wall ,

u_{∞} = freestream velocity, and

q_{∞} = freestream dynamic pressure

This linearized boundary condition is based on an approximation of viscous flow through a porous media. The value of R can thus be determined by experimental measurement of wall crossflow characteristics as done by Chew (1955). More recent work reported by Smith (1976) suggest that the boundary condition should be rewritten as:

$$\phi_x + K \phi_{xy} + \frac{1}{R} \phi_y = C \quad (3.21)$$

retaining K as a correction for nonlinear compressibility effects and C as a measure of the difference between freestream and plenum pressures. The value of C is computed directly from measured pressure

as:

$$C = \frac{p_{\infty} - p_p}{\rho_{\infty} u_{\infty}^2} = - \frac{C_{pp}}{2} \frac{u_{\infty}}{u_{\infty}} \quad (3.22)$$

where C_{pp} = plenum pressure coefficient (second p subscript denotes plenum, ∞ subscript denotes freestream).

This formulation allows distinction between isolated plenums for lifting conditions. The resulting interference analysis could not be correlated with experiment because of inaccuracies in the data. The applicability of an averaged plenum pressure was questioned for moderate-to-high lift conditions.

A more complete treatment of boundary conditions is given by Mercer, et al. (1981) where the full potential equation was analyzed for transonic wind tunnel flow about a wing. The paper suggest two possible sets of boundary conditions. The first (a Neuman condition) states simply that the average velocity normal to the wall is equal to the gradient of the potential ($v_n = \partial\phi/\partial n$). This requires that the

wall crossflow be specified by some other means. For conservation of mass, the downstream boundary must be corrected for the new outflow through the walls by:

$$\left. \frac{\partial \phi}{\partial x} \right|_{x=L} = u_{\infty} + \delta \quad (3.23)$$

where

$$\delta = \frac{1}{\rho_d A_d} \int_{\text{tunnel wall area}} \rho v_n dS + \left(\frac{\rho_{\infty} - \rho_d}{\rho_d} \right) u_{\infty} \quad (3.24)$$

The downstream boundary of the test section is adjusted by integrating the mass flux normal to the walls and at the downstream boundary and correcting the test section exit velocity condition. (Note: d denotes downstream exit conditions). The alternate set of boundary conditions (Dirichlet type) start with a specified pressure coefficient such that $C_p = -2(u-u_{\infty})$ where u is the local velocity and the potential is then calculated as:

$$\phi = \int_0^x u \, dx \text{ (integrated along the wall)} \quad (3.25)$$

This form of the pressure coefficient is a simplified expression whose accuracy is sufficient on the walls. The compatibility of the downstream condition is satisfied by setting the test section exit velocity equal to that at the downstream edge of the walls ($\partial \phi / \partial x = u_d$). Both of these approaches require a priori knowledge of wall conditions and do not provide values based on computed wall cross-flow conditions. These specified values can be obtained experimentally and input to the calculation to determine tunnel interference impact. Whether this procedure could be extended to an unsteady boundary value problem is unclear at this point.

The work of Mabey (1980) in predicting the acoustic resonance of porous and slotted-wall wind tunnels presents a set of boundary conditions applied at the ventilated wall and at the plenum outer wall. The problem is solved as a coupled between two flowfields with the velocity potentials ϕ and ψ designating the conditions in the tunnel test section and plenum, respectively. For the plenum potential at the outer wall, the solid wall boundary condition prevails - namely, a zero normal gradient (velocity) - as $\psi_y = 0$. At the ventilated test section boundary, the conservation of mass requires that velocities be equal at the wall ($\phi_y = \psi_y$). The flow through the wall is specified according to the physical characteristics. Based on the assumption that in the presence of the main flow, the acoustic reactance of the perforation does not change significantly and also that the flow resistance is much smaller than the total acoustic impedance, Mabey (1978) shows a derivation of the homogeneous boundary condition. The pressures of the freestream and plenum are defined respectively as:

$$p - p_\infty = - \rho [u \phi_x + \phi_t] \quad (3.26)$$

and

$$p_p - p_\infty = - \rho [\psi_t] \quad (3.27)$$

This results in a pressure across the wall of: $\Delta p = p - p_p = - \rho [u \phi_x + \phi_t - \psi_t]$. The results of Hersh (1976) showed that the oscillatory velocity lags the pressure wave by 90° at higher frequencies. The acoustic impedance of a single orifice was given as

the resistance (R) and reactance (X) components:

$$R = \sqrt{8\mu\rho\omega} \quad (1 + L/D) \quad (3.28)$$

$$X = \rho\omega (L + 0.85 D) \quad (3.29)$$

where $\omega = 2\pi f$, angular frequency,

ρ = fluid density,

μ = dynamic viscosity,

D = orifice diameter, and

L = orifice length .

Mabey uses just the reactance term for the wall pressure difference and writes:

$$\Delta p = i\rho\omega (L + 0.85 D) v_0 \quad (3.30)$$

where

v_0 = oscillatory velocity amplitude through the orifice.

Converting to a homogeneous boundary with an average local crossflow velocity for a wall of porosity σ , he gets:

$$v_n = \sigma v_0; \text{ and } \Delta p = \frac{i\rho\omega}{\sigma} (L + 0.85 D) v_n \quad (3.31)$$

This neglects interaction of clustered orifices, strong grazing flow, and high amplitude pressure fluctuations. More recently, Hersh (1979) has described detailed approaches to correct for those effects. However, he also describes a simplified correction for the interaction of clustered orifices for the specific reactance, X_0 (distributed uniformly over the perforated plate) as:

$$x_0 = \frac{\rho\omega}{\sigma} \left[L + \frac{0.85 D}{\Psi'(\xi)} \right] \quad (3.32)$$

where $\Psi'(\xi) = 1.66$, the Fok interaction parameter based on the

clustering variable, $\xi = \frac{D}{\sqrt{\frac{4A_f}{\pi}}}$, (A_f = zone area of each orifice).

This gives:

$$\Delta p = \frac{i\rho\omega}{\sigma} (L + 0.512 D) v_n \quad (3.33)$$

The reduction in the reactive mass end correction from 0.85 D to 0.512 D is the result of interacting flows from neighboring orifices reducing the virtual mass addition to the fluid element oscillating within the orifice. Modifying Mabey's effective hole diameter to $T = (L + 0.512 D)$, the wall pressure differential can finally be written as:

$$\Delta p = i\rho\omega kT v_n \quad (3.34)$$

where $k = (1-\sigma)/\sigma$, an empirical function of the open area ratio.

Substituting and rearranging this result into the boundary condition gives:

$$\phi_x + \frac{\phi_t - \psi_t}{u} + \frac{i\omega kT}{u} \phi_y = 0 \quad (3.35)$$

This boundary condition provides the basis of Mabey's tunnel resonance calculation. It also forms a reasonable unsteady wall boundary condition for the OSU/AARL facility.

3.3 STEADY-STATE INTERFERENCE

Steady-state wind tunnel interference has historically been treated as the combined influence on aerodynamic data due to the presence of the wind tunnel boundaries and model support mechanisms. For two-dimensional airfoil testing the model supports are generally just the tunnel sidewalls and are therefore treated as part of the boundary interference. Various interference analysis methods and testing guidelines have been developed to handle wind tunnel interference.

Allen and Vincenti (1944) discussed the different steady-state interference phenomena and methods to handle them. Lock's method of

images was discussed as a tool to examine pressure deviations due to the basic model profile in wind tunnel experiments. It was concluded that such blockage was characterized by an effective increase in the simulated free stream velocity. The wake effect was said to be a consequence of the low-momentum and low mass-flux in the model wake...necessitating increased flow velocity between the wake and the walls. Analytical modelling of the effects was deduced to be equivalent source at the airfoil. Suitable compressibility corrections were derived from the Prandtl-Glauert similarity arguments. It was suggested that model camber be modelled as a sheet of bound vortices located at the mean camber line. Interference was designated to be the combined effect of all of the above issues in the presence of the tunnel flow boundaries. The most severe effect at high speeds was seen to be tunnel choking for solid-wall test sections, and it was predicted according to the chord-to-tunnel height ("drag density factor") and Mach number. Later Allen (1954) traced the development of transonic tunnels for NACA and the solid blockage limits were eased by slotted wall test section designs. In the survey of transonic tunnel research, Allen provides a review of slotted and variable porosity perforated walls. Goethert (1954) discussed the USAF efforts for transonic tunnel development. He examined perforated wall cross-flow characteristics and recommended the use of a porous wall construction with auxiliary pumping. Brescia (1957) analytically examined ventilated wall interference in terms of "reflected" and "transmitted" singularities meant to represent perforated walls as a cascade of lifting surfaces. Wright (1959) developed a first-order treatment of the combined

interference effects. For subsonic testing lift, solid blockage, wake blockage, and compressibility effects were discussed. For transonic testing the additional influences of choking and wave reflection were noted. Arguments were presented for slotted versus perforated test section designs. Wright had observed that two-dimensional tunnel testing suffered greater lift interference effects than did three-dimensional experiments. Also, he observed that perforated tunnels apparently reduce shock wave reflections more effectively than slotted wall tunnels. Reynolds, et al. (1959) presented experimental results for two models of different sizes tested in a slotted test section for Mach 0.4 to 1.3. The models were wing/body combinations having 3 percent thick biconvex wings and stabilizers with solid blockage values of 0.06 percent and 0.96 percent. The results were compared with small-disturbance potential theory results using the Baldwin model for wall boundary conditions. Pearcey, et al. (1959) reported on two-dimensional transonic interference effects including solid blockage effects, lift influence, local supersonic flow, and wake blockage effects due to boundary layer separation. His test facility was operated with isolated plenums, and he examined end wall boundary layer effects. Much of the early work in tunnel interference resulted more in providing testing guidelines than for developing data correction models for general use. Adjustments to the effective test conditions were more often used than actual interference analysis based on detailed data values combined with analytical effects.

Collins and Krupp (1974) examined tests of a pair of two-dimensional circular arc airfoils (6 percent and 12 percent

biconvex shapes) in a solid-wall wind tunnel. Theoretical results were obtained with the mixed operator method of Murman and Cole for direct comparison with the empirical observations. Berndt (1975) made measurements inside the slots of a transonic tunnel wall to aid in a derivation of the wall boundary condition. Through an accounting of the wall boundary layer/slot interaction, Berndt provided an improved numerical derivation of steady-state interference effects. Starr (1975) performed some Ludweig tube experiments with a cone-cylinder model to measure the effects of boundary layer displacement thickness on ventilated wall crossflow characteristics. Lo (1975) examined the use of a non-uniform porosity distribution to reduce lift and pitch interference incurred during testing of V/STOL models with finite wings. He also discussed some of the early ideas for self-correcting or adaptive wind tunnel designs. Kraft (1976) examined transonic, two-dimensional airfoil interference through the application of integral methods to the nonlinear small disturbance equations. The transonic form of the small disturbance equation was modified using a transonic similarity parameter proposed by Krupp and wall boundary conditions based on Baldwin's theory to form the appropriate kernel functions. Smith (1976) examined the use of separated plenum chambers. He proposed a modification to Baldwin's theory which treated cases where the plenum pressure differed from the freestream static conditions.

Kacprzynski (1976) stated that: "The component of velocity normal to the wind tunnel wall is a nonlinear function of the difference of the static pressures on the wall and in the plenum chamber." He

suggested that the inflow and outflow characteristics at the ventilated boundary were inherently different and were not adequately modelled by inviscid techniques. Kraft and Lo (1977) linearized the transonic, small-disturbance equation and Fourier transformed the boundary conditions to apply integral methods to the nonlinear problem. Comparison with earlier 6 percent biconvex airfoil data was made. Blackwell and Pounds (1977) examined transonic interference for a supercritical airfoil design tested at various Reynolds number and wall porosities. The experimental data was treated with computational correction techniques. Starr (1978) studied the lift and downwash interference effects in the AEDC 4T and 16T tunnels using an ONERA calibration model which is a high aspect-ratio wing/body combination. The lift interference was shown to be related to chord, tunnel height, and normal force coefficient. Lower aspect-ratio models were not as sensitive to interference. Barnwell (1979) performed sidewall boundary layer measurements in two-dimensional testing and found that the viscous influence created the effect of a Mach number deficit at the model location. Although the sidewall boundary layer may be relatively thin (as estimated by flat plate growth estimates) the local thickness becomes dominated by local pressure and edge-velocity gradients. Through a momentum integral analysis he showed the effect of the local pressure terms. Barnwell also showed how the empirical corrections could be based on measured tunnel pressures to determine outer boundary conditions. Experimental verification was conducted with 10 and 20 percent thick supercritical airfoils. Stahara and Spreiter (1979) used experimentally measured pressures to predict boundary conditions at a

reference location near the walls. Data from the outer wall and an internal rail were used to define a Dirichlet boundary condition at an outer control surface. This result was applied to the transonic small disturbance equation to assess the potential flow interference effects. Chan (1980a) examined the use of sidewall suction near the model to control the boundary layer influence and avoid shock-induced separation. He concluded that sidewall suction had little potential benefit. Mokry and Ohman (1980) developed a numerical method using the Fast Fourier Transform (FFT) to solve the Dirichlet problem where the outer boundary values are determined from measured pressure distributions. The method requires that all outer boundaries are subcritical but supersonic pockets inside the flow are allowed. Another use of experimental wall conditions in conjunction with the transonic, small-disturbance equation by Chen (1980b) where certain nonlinear terms were empirically determined to provide tunnel corrections.

Experimental verification of the low steady-state interference in the 6" x 22" transonic airfoil tunnel has been reported by Lee (1976) where tests with a NACA 0012 model showed negligible interference even in supercritical flows. Three chord lengths of 3-, 6-, and 12-inch (7.6, 15.2, and 30.4 centimeters) were tested over a wide range of Mach numbers and angles of attack. An induced downwash correction of -0.17° times the section lift coefficient was suggested to correct the effective pitch angle. Comparisons with data obtained on a 6-inch (15.2 centimeter) model in the CALSPAN 8-foot tunnel indicated the pressure distribution was negligibly influenced by the tunnel

boundaries in the OSU 6" x 22" facility. The low interference effects observed were attributed to the independent plenum designs. A continuation of the OSU interference work was reported by Lee, et al. (1978) through testing of several other airfoil sections including the NACA 64A410 and GAW-1 profiles.

3.4 UNSTEADY INTERFERENCE EFFECTS

The influence of the tunnel boundaries on the unsteady pressure measurements measured during oscillating airfoil experiments has been the subject of research for several decades. The causes of unsteady wind tunnel interference include both periodic and random effects. For high reduced frequencies, acoustic reflection from the walls can cause inaccuracies in the data. The forward feed of pressure disturbances in the boundary layers can cause boundary layer/shock interaction on the sidewalls; for the shock motion observed in oscillating airfoil tests, such interaction improperly influences the unsteady pressure loading. Turbulent boundary layers can radiate a random noise which unduly influence the signal-to-noise ratio in unsteady pressure measurements. A strong, high frequency resonance can occur due to flow across the perforated or slotted walls. The resulting acoustic modes can be attributed to geometric properties of the perforations or slots. The streamline confinement associated with steady-state interference (perturbations in local Mach number distribution and mean shock position) has a nonlinear effect on the unsteady pressure distributions since the highest loadings are generally seen in the region near the shock wave. The effects of the oscillating wake result from the vortex confinement which can create an oscillating upwash influence and

possible acoustic feedback if the vortices impinge on downstream struts or choke bar controls. Tunnel resonance observed during early oscillating airfoil tests occurs when the natural acoustic modes with wavelengths corresponding to significant test section dimensions are excited by model oscillation at nearly the same frequencies. All of the above effects contribute to reduced data quality in the oscillating airfoil experiments and few of these interference sources are directly amenable to detailed interference analysis.

Early unsteady tunnel flow research was limited due to a lack of adequate instrumentation and data processing techniques. However, significant contributions were made in understanding the unsteady interference effects. Jones (1950) developed a theory for predicting the influence on aerodynamic coefficients measured on an oscillating airfoil in the incompressible flow cases. The predicted effects were minimized if the airfoil pitched about its aerodynamic center. Examples were computed for the mid-chord (pitch) oscillation, and comparisons with experiment were discussed. The work of Runyan and Watkins (1953) examined the effects of walls on subsonic, two-dimensional, compressible flow testing of oscillating airfoils. Their technique uses the method of images for analysis, and from that it was theoretically demonstrated that tunnel resonance would occur at certain reduced frequencies. The transverse oscillation mode was coupled to the model motion if the reduced frequency corresponded to:

$$k = \frac{\pi c \sqrt{1-M^2}}{H M} (2n-1) \quad (3.36)$$

for $n = 1, 2, \dots$; where c = airfoil chord length, M = freestream Mach

number, and H = tunnel height. Jones (1953) observed tunnel wall effects at Mach 0.7 corresponding to frequencies below those predicted by Runyan and Watkins. Jones gives guidelines which provide allowance for airfoil thickness and boundary layer displacement effects which he titles the equivalent profile method, wherein an equivalent rigid thin surface is used to approximate the actual airfoil. It was observed that the unsteady interference effects on measured stability derivatives were greatest at tunnel resonance conditions and at low reduced frequencies where quasi-steady interference effects were observed. Runyan, Woolston, and Rainey (1956) gave an integral formulation of the subsonic interference condition and provided experimental confirmation of the typical resonance condition. The testing was performed in the NACA Langley 2x4-foot tunnel with a NACA 65-010 airfoil (1-foot chord) using differential pressure transducers which measure the difference between upper and lower surface pressures at various chord locations. The report includes data on unsteady lift and moment coefficients over a range of Mach 0.3 to 0.8 obtained in the solid-walled rectangular test section.

Molyneux (1964) investigated several sources of dynamic interference including: (1) boundary constraints (closed wall or open wall), (2) compressibility (tunnel resonance), (3) choking, (4) supersonic flow regions (shock interaction with wall boundary layers, (5) noise from the tunnel drive system, diffuser flow separation, turbulent boundary layers, and oscillating wake impingement, and (6) model support interference (for three-dimensional models) produced by diffraction effects in the geometric acoustics field around stings or

support pylons. Many of these sources of unsteady interference are merely counterparts of the effects observed for steady interference behavior. Acum (1966) derived a theoretical treatment of sinusoidally oscillating wing interference based on a time-linearized form of the small-perturbation equation. He provided corrections for the two-dimensional, compressible flow case, predicted tunnel resonance for solid-wall test sections, and examined the coupling of the steady and unsteady interference behavior as a quasi-steady vortex field between solid walls at low values of the frequency-Mach number product.

Recent unsteady interference investigations have delved more into compressible flow cases, and as yet, no unified, transonic interference theory has been developed to correct results from oscillating airfoil tests. However, the renewed interest in transonic aerodynamics has spurred interest in providing high-quality experimental data both for confirmation of numerical modelling techniques and design data requirements. Garner (1972) investigated the use of variable porosity to minimize dynamic interference. He hypothesized that an optimum porosity distribution for any given model could be calculated by an iterative procedure which appears to be relatively insensitive to Mach number. Moore and Wright (1973) performed experiments involving oscillating pitch motion of half-wing models. Variation of tunnel porosity was examined to see what effect on damping and stiffness derivatives would result. These empirical results indicate that the boundary conditions for ventilated walls could be based on flow resistance measurements and the ratio of hole size versus boundary layer displacement thickness. Streather (1974) applied the

lifting-surface method to cases involving subsonic, three-dimensional models, oscillating at low frequencies. An interference kernel (to analyze open, closed, or slotted wall cases) was used to evaluate spanwise variations on finite, swept wings. Ruhlin, et al. (1975) conducted tunnel tests at the ONERA Modane S2 facility to provide wing flutter data on an SST model for comparison with results obtained in the NASA Ames six-foot tunnel and the NASA Langley Transonic Dynamics Tunnel (TDT). An examination of the tunnel resonance effects, of shock reflections, and of wall dynamic interference provided guidelines to set limits on model versus tunnel size for reasonable flutter testing. An asymptotic solution for high frequency oscillation of an airfoil in a transonic tunnel was proposed by Savkar (1976), who found such behavior was relatively insensitive to Mach number.

The experimental observation and analytical characterization of tunnel resonance associated with oscillating airfoil experiments have been the topic of several publications. Runyan and Watkins (1953) supported their theoretical work with experimental observation of resonant acoustic modes associated with the transverse dimensions of a closed-wall test section. Bauer (1976) prepared a note on acoustically transparent walls for a tunnel test section which use an acoustic absorber material to form the flow boundaries. The sintered screen material suggested had a 1 dB transmission loss and was shown to reduce the shear layer noise seen in open-jet, anechoic wind tunnels. Mabey (1978) deduced that ventilated tunnel resonance was a function of plenum depth and wall orifice characteristics. He postulated a more realistic wall condition for testing below Mach 0.6. Mabey (1979)

suggested the use of acoustic liners to reduce the dynamic interference effects in wind tunnel testing. Mabey (1980) provided a ventilated tunnel resonance calculation method which considers the wall perforation details and plenum geometry for both subsonic and transonic test conditions.

Following Mabey's guidelines for computing tunnel resonance for ventilated-wall test sections, it was found that the frequency range explored in the oscillating airfoil tests reported here remained well below any predicted tunnel acoustic modes. The test section geometry (see F-24) shows that for acoustic modes operating between the upper and lower walls, the relative propagation velocity across the test section is defined as: $a_B = a \sqrt{1-M^2}$. The freestream velocity tends to slow the transverse wave propagation in the test section in accordance with the vector diagram shown. The resonance in a solid wall tunnel would be based on a standing half wave as postulated by Acum and would display a rapid decline in resonant frequency as sonic conditions are approached. The more complete analysis of Mabey shows (in F-25) that the tunnel resonance frequencies remain relatively constant for Mach numbers below the critical value of 0.618; above this speed the resonant frequencies increase for freestream Mach number between 0.618 and 0.77. The rapid decline in resonant frequencies at conditions above Mach 0.77 are indicative of the reduced transverse propagation velocity. At supersonic velocities, the acoustic disturbances cannot propagate directly across the test section (but rather travel along Mach lines), and unless an external feedback path (through boundary layers or plenums) is accounted for, resonant modes

F-24 6" x 22" Tunnel Test Section

F-25 Tunnel Resonance Frequencies

cannot be predicted. Lateral modes along the airfoil span would have frequencies over three- and one-half times the solid-wall values. Therefore, it is apparent that the frequencies predicted for tunnel resonance behavior are well above the frequency range examined during the oscillating airfoil tests reported here. Any higher frequency modes associated with the edge-tones generated at the perforations and/or plenum depth modes would not affect the airfoil oscillation tests.

For subsonic, compressible flow cases, the oscillating airfoil interference effects have been characterized by Bland (1970) who derived a kernel function and integral transform technique to account for the ventilated-tunnel wall condition for subcritical conditions. He developed a transform pair based on what he called downwash and pressure polynomials (an orthogonal set) and provided quadrature techniques to evaluate the integrals. Fromme and Golberg (1978 and 1979) implemented Bland's transform technique in a computer routine to analyze thin airfoils oscillating in ventilated wind tunnels. They made comparisons with the results obtained at NLR on a NACA 64A006 flap oscillation model and the earlier works of Kussner. A convergence acceleration technique was suggested for problems dealing with oscillating airfoil control surfaces (the Richardson Extrapolation Method). The computer code was used to examine tunnel resonance effects. Fromme and Golberg (1980a and 1980b) have reformulated Possio's kernel so that it may be used in conjunction with collocation methods developed to treat Bland's formulation above. This work was then used to examine interference effects observed for oscillating flap

tests in ventilated wind tunnels. The computer code developed to solve Bland's transform was used to analyze the equivalent flat plate cases corresponding to the OSU/AARL experiments and the results of that effort will be reported in later sections of this paper.

IV. STEADY-STATE DATA

The steady-state pressure coefficients obtained during fixed angle-of-attack test runs were integrated to give section lift coefficient, C_L and pitching moment coefficient, C_m . Section drag coefficient data were obtained with a traversing pitot probe which surveyed the airfoil wake to measure the momentum deficit in the flow. The static pressure was assumed to be uniform across the wake and it was determined through continuously monitored plenum static pressures.

The critical pressure coefficient, C_{p^*} , is defined as the value of C_p which corresponds to locally sonic flow conditions on the airfoil. It represents the limiting value dividing subcritical flow cases from supercritical transonic flows. If local pressure coefficients exceed that critical C_p (in negative value), then the possibility of shock formation on the airfoil exists, however, observable shock influence is often delayed until the local Mach number exceeds a value of about 1.2. The functional definition of the critical pressure coefficient is given as:

$$C_{p^*} = \frac{\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma}{1-\gamma}} - \left(1 + \frac{\gamma-1}{2} M_\infty^2\right)^{\frac{\gamma}{1-\gamma}}}{\frac{\gamma}{2} M_\infty^2 \left(1 + \frac{\gamma-1}{2} M_\infty^2\right)^{\frac{\gamma}{1-\gamma}}} \quad (4.1)$$

and values of that function are shown in F-26. For Mach numbers less

F-26 Critical Pressure Coefficient

than 0.8 local C_p values did not indicate supersonic conditions for the pitch angles tested. At Mach 0.8 and above, local supersonic flow effects were observed for all test cases.

4.1 STEADY-STATE PRESSURE PROFILES

The steady-state pressure coefficients were obtained on both the 7.6 cm and 15.2 cm airfoil models. These results show differences from the numerical predictions computed with the STRANS and LTRAN2 small-disturbance potential equation routines. These inviscid solution techniques tended to overpredict shock strength and place the shock too far aft as compared to experimental data obtained in the OSU/AARL testing and other wind tunnel facilities.

The subcritical cases were plotted in F-27, F-28, and F-29. The predictions from the steady-state, small perturbation equations were obtained with the over-relaxation technique in the STRANS CODE (this steady-state result was then used in the UTRANS code to obtain the solution to the harmonic form of the unsteady small disturbance equations). Similarly, a steady-state pressure profile was obtained using the LTRAN2 code which develops a result for the mean angle-of-attack through iterative refinement of the small-disturbance solution using the implicit approximate-factorization scheme. As above, the result is used to develop the unsteady solution; however, LTRAN2 uses the steady-state case only as an initial condition for the time-marching method which tracks the pressure response to the time-varying boundary condition on the airfoil surface.

The low Mach number case (shown in F-27) compared the experimental results for the two airfoil models with the numerical results. The two

STEADY PRESSURE DISTRIBUTION

NACA 64A010 AIRFOIL

M = .350, CPX = -4.9564
ALPHA = .000 DEGREE

F-27 Steady Pressure Distribution, M = 0.35

STEADY PRESSURE DISTRIBUTION

NACA 64A010 AIRFOIL

M = .520, CPX = -1.9272
ALPHA = .000 DEGREE

F-28 Steady Pressure Distribution, M = 0.52

sets of empirical data agreed quite well, indicating that the two profiles were geometrically similar and tunnel interference effects were negligible for this case. The STRANS and LTRAN2 tended to respectively underpredict and overpredict the local magnitude of the pressure coefficients.

At Mach 0.52, the correlation of the experimental and numerical results was improved. Both STRANS and LTRAN2 results closely predicted the pressures adequately in the mid-chord region (see F-28). However, the LTRAN2 code showed a greater flow acceleration near the leading edge than was measured or was predicted by STRANS. This may be attributed to the inadequacy of treating the rounded-nose of the NACA 64A010 profile with small-disturbance assumptions. The trailing-edge condition was also incorrectly predicted. Since this was a non-lifting symmetric airfoil case, the upper and lower surfaces had equal values of C_p and there was no Kutta condition to satisfy. The finite angle of the trailing-edge can present a mathematical singularity in the small-disturbance potential formulation as does the blunt leading edge. This factor, combined with the large gradients in velocity potential near a blunt leading edge necessitated clustering of grid points near the leading- and trailing-edge. Such clustering was used in both codes, however, a more accurate treatment of these regions would have to consider the finite thickness effects which would not allow use of the Cartesian, finite-difference grids used here.

The Mach 0.70 data (F-29) showed that the flow in this case was still subcritical and agreement with the numerical predictions was good. The differences near the leading-edge were again attributed to

STEADY PRESSURE DISTRIBUTION

NACA 64A010 AIRFOIL

$M = .700$, $CPX = -.7791$
 $ALPHA = .000$ DEGREE

F-29 Steady Pressure Distribution, $M = 0.70$

STEADY PRESSURE DISTRIBUTION

NACA 64A010 AIRFOIL

$M = .800$, $CPX = -.4346$
 $ALPHA = .000$ DEGREE

F-30 Steady Pressure Distribution, $M = 0.80$

the limited validity of the small-disturbance assumption. Beyond the 10 percent chord location the two numerical predictions were in excellent agreement and the data were well matched beyond the 25 percent chord.

For the above subcritical cases, it was apparent that the Mach 0.35 and 0.52 data were nearly identical. The Mach 0.70 data showed increased compressibility effects with larger magnitudes of the pressure coefficient. The interference effects for these non-lifting cases was assumed to be negligible since there were few observable differences between the two sets of data from different-sized models.

Supercritical conditions were reached during testing at a freestream Mach number of 0.8 for the zero pitch position. The comparison of experimental and numerical results (see F-30) showed that both STRANS and LTRAN2 predicted higher local velocities over the first half of the airfoil than were measured in the OSU/AARL testing. Experimental data obtained in NASA Ames 11-foot Transonic Wind Tunnel were also shown. There was no strong indication of a shock wave in the OSU/AARL data and the STRANS prediction. However, the Ames data and LTRAN2 code both showed a shock located near the 55 percent chord location. The inviscid analysis techniques, in general, tended to predict stronger shocks at locations further aft than were observed in experiments. This discrepancy was attributable to the boundary layer/shock interaction on the airfoil surface. The local viscous influence tends to curve the normal shock forward on the airfoil and decrease its strength to that of an oblique shock. The weaker oblique shock did not decrease the local Mach number as much as the normal

shock would, hence, the pressure ratio across the shock was reduced by viscous effects. The oblique shock resulted from the wedge effect generated by sudden thickening of the boundary layer in the presence of the strong adverse pressure gradient associated with the shock. The relative displacement thickness slope can be seen as a profile change which when applied to the airfoil surface coordinates will more accurately predict supercritical pressure profiles. The disagreement between the two sets of experimental data were partially due to a greater maximum thickness in the Ames model which occurred during their model fabrication process and for a small offset in pitch angle (0.21°).

Observable differences in the LTRAN2 results and the OSU/AARL experimental data were magnified for Mach 0.84 freestream conditions (see F-31). The empirical results placed the shock at about 57 percent chord while the numerical prediction indicated a shock at 70 percent chord. As a result, the comparison was poor over the aft portion of the airfoil and subsequent correlations of the unsteady results reflected this. While this is not typical of most steady-state, transonic airfoil predictions, it is significant in that the steady-state solutions from LTRAN2 serve as the starting point for the time-dependent (unsteady) solution. Improved numerical techniques, especially those accounting for boundary layer displacement effects, tend to give more accurate steady-state solutions. The STRANS code would not converge due to numerical instabilities in the relaxation method used. Hence, no results were obtained for this Mach number or above with the STRANS and UTRANS codes. The experimental data indicated that the shock decelerated the flow to conditions nearer the sonic values than predicted.

STEADY PRESSURE DISTRIBUTION

NACA 64A010 AIRFOIL

$M = 0.84$, $C_{P\infty} = -0.3267$

$\alpha = 0.000$ DEGREE

F-31 Steady Pressure Distribution, $M = 0.84$

STEADY PRESSURE DISTRIBUTION

NACA 64A010 AIRFOIL

$M = 0.89$, $C_{P\infty} = -0.2094$

$\alpha = 0.000$ DEGREE

F-32 Steady Pressure Distribution, $M = 0.89$

This disagreement persisted at the Mach 0.89 test condition as shown in F-32, where the LTRAN2 code placed the shock at the trailing-edge. The experimentally observed shock location was somewhat obscured, however, a mean shock position between the 70 and 80 percent chord locations was apparent. This lack of a strong indication of shock prediction was due to the shock motion attendant to separating flow conditions which can generate an alternating shift in the shocks on the upper and lower airfoil surfaces. It can be driven by an acoustic feedback mechanism around the trailing edge. The time-averaged response of the pressure measuring system tends to smear the actual shock strength (which in itself has a time-varying nature).

These zero pitch pressure profiles served to point out the relatively good agreement between the numerical results of the STRANS and LTRAN2 codes and experimental data at subcritical conditions; yet, the discrepancies of the inviscid assumption were significant for the supercritical flow conditions. Such disagreements were expected to persist in unsteady flow predictions since these two codes both relied on fully developed steady-state solutions.

For lifting conditions, comparisons were made with other experimental data and theoretical codes which account for boundary layer effects as well as the inviscid, small-perturbation solution from the STRANS code (see F-33). The Navier-Stokes solution and STRANS solution predicted nearly identical shock locations but the inviscid code indicated higher local velocities over the front half of the airfoil. The Garabedian-Korn code was specially suited to analyzing transonic airfoils through an iterative procedure of computing the

F-33 Comparison of Theoretical and Experimental Steady Pressure Distributions

inviscid potential flow about airfoil coordinates corrected by adding in the boundary layer displacement thickness. The subsequent computations indicated a shock location ahead of results for the other two codes. The OSU/AARL data was obtained for both positive and negative pitch angles and the results plotted here show that the data were symmetric for upper and lower surface pressures. The other experimental data obtained in NASA Ames Research Center facilities projected a shock location ahead of the OSU/AARL data and the three numerical predictions. The OSU data showed a shock at about 57 percent chord and lower C_p magnitudes than the Ames data and computer code results. These comparisons were representative of the difficulties in obtaining reliable data and numerical results for lifting airfoils at transonic flow conditions.

The integrated lift coefficients obtained from the steady-state pressure data indicated a linear relationship to pitch angles. The nominal lift-curve slope given in Abbott and von Doenhoff (1949) has a value of 0.11 per degree for the NACA 64A010 profile at incompressible flow speeds. The OSU/AARL data (shown in F-34) were obtained over a range of -2° to $+4^\circ$ pitch angle. The effects of compressibility on this lift curve slope were discussed by Abbott and von Doenhoff, but no specific results for the NACA 64A010 profile were found. The OSU/AARL data showed the lift curve slope increasing with freestream velocity up to Mach 0.84. At Mach 0.89, the slope decreased to a value similar to that observed at Mach 0.35; this general trend correlated well with Abbott and von Doenhoff's examples for similar airfoils. The interference correction suggested by Lee, et al. (1978) for the

F-34 Lift Coefficient vs Pitch Angle

F-35 Lift Coefficient vs Pitching Moment Coefficient

OSU/AARL 6" x 22" facility was not applied since its influence was only 0.1° for the highest lift condition obtained.

The pitching moments characteristics observed during the fixed angle of attack test runs were not as clearly defined as the lift data. In F-35, we see that the C_L versus C_m (about the quarter-chord point) plots displayed the two major flow conditions. For the subcritical and low supercritical test conditions (Mach 0.35 through 0.80) the lift-moment diagram indicated that the pitching moment changed by -0.04 per unit lift coefficient. This resulted in a slight rearward shift in the center of pressure as the pitch angle and lift increase. For the flow cases with strong shocks (Mach 0.84 and 0.89), the rearward shift in the aerodynamic center was greater due to increased aft loading on the airfoil. These lift-moment curves indicated a change in pitching moment equivalent to -0.15 times the lift coefficient. This was consistent with the classical shift of the aerodynamic center from the quarter-chord for thin airfoils in subsonic flow to the mid-chord location for supersonic flow conditions.

The section drag data was obtained with a pitot traversing probe to measure momentum deficit approximately two chord lengths behind the larger (15.2 centimeter) model. Those results were typified for two Mach numbers (each examined at two Reynolds numbers) and plotted in F-36. Abbott and von Doenhoff reported that the minimum drag coefficient for the NACA 64A010 airfoil is about 0.0042 for a "smooth" surface and 0.0090 for a "rough" surface. The results were measured at a Reynolds number of six (6) million at incompressible flow speeds. Data obtained during earlier tests at OSU/AARL indicated the minimum

F-36 Drag Polars

drag at Mach 0.30 to be about 0.0070. The "laminar bucket" was not apparent at the Mach 0.30 test condition, however, there was evidence such a drop in the drag coefficient was present at Mach 0.80. While no variation in the drag was apparently dependent on Reynolds number at the low speed, a shift in the minimum drag coefficient was observed at the higher Mach number. The higher values of drag at Mach 0.8 are attributed to compressibility effects and boundary layer separation from the aft portion of the airfoil.

These steady-state results helped insure that the models were of the proper profile, and they were used to determine if any flow angularity or significant wall interference effects were present.

4.2 QUASI-STEADY PRESSURE AMPLITUDES

Apart from the classical use of the steady-state pressures to obtain integrated lift and moment coefficients, an analysis was performed using the differences in C_p for test runs at $+1^\circ$ and -1° pitch to determine the quasi-steady pressure coefficients for reduced frequencies near zero. These results were indicative of the low frequency pitch oscillations and as discussed in Chapter 5, the data correlated well with the oscillating pressure amplitudes for small reduced frequency values. The data from both upper and lower surfaces were plotted for the 15.2 centimeter model. However, the relative "phase" for these quasi-steady results were -180° and 0° for the upper and lower surfaces, respectively.

The quasi-steady pressure amplitudes depicted in F-37 through F-39 showed that the response of the upper and lower surfaces were nearly identical and for these subcritical flow cases, the quasi-steady

F-37 Quasi-Steady Pressure Distribution, $M = 0.35$

F-39 Quasi-Steady Pressure Distribution, $M = 0.70$

F-41 Quasi-Steady Pressure Distribution, $M = 0.84$

F-38 Quasi-Steady Pressure Distribution, $M = 0.52$

F-40 Quasi-Steady Pressure Distribution, $M = 0.80$

F-42 Quasi-Steady Pressure Distribution, $M = 0.89$

response was dominated by the pressure changes due to pitch near the leading-edge. No significant differences existed over most of the airfoil, however, a difference of about 10 percent was noted for the 5 percent chord location. This difference was consistent at all Mach numbers tested and therefore it was attributed to a probable misplacement of the pressure tap on the lower surface. Small changes in chord position corresponded to large changes in the local flow velocity due to the large curvature in the neighborhood of the leading edge. The quasi-steady magnitudes increased slightly between Mach 0.35 and 0.70 but did not change in their general shape.

The first evidence of shock influence on the pressure data was seen at Mach 0.80 for small non-zero pitch angles. At $+1^\circ$ and -1° , the steady pressure profiles indicated that weak shock formed on the upper and lower surfaces, respectively, near the 45 percent chord location. The quasi-steady results vividly showed the influence of the alternate shock formation on the upper and lower surfaces (see F-40). This relatively large pressure difference was located at the 45 percent chord location and directly correlated to the presence of the shock. At low frequencies, the unsteady response was dominated by an alternating shock formation on the upper and lower surfaces.

At Mach 0.84, the steady-state pressure profiles indicates that shock waves existed on both surfaces simultaneously and at low frequencies of pitch oscillation, such shocks would persist and oscillate between the 50 and 60 percent chord locations for an oscillation amplitude of $\pm 1^\circ$. The impact of this movement of the shocks and resulting changes in their strengths is seen in F-41 where

the quasi-steady pressure amplitude was dominated by the shock influence. This alternating shock motion was seen to give the greatest variations in quasi-steady pressure amplitudes. The small differences in quasi-steady pressure amplitudes near the shock regions provided additional evidence that the model was symmetric and aligned to the flow.

The quasi-steady pressure data for Mach 0.89 (see F-42) indicated that the shift in relative shock positions on the upper and lower airfoil surfaces observed at Mach 0.84 were not observable at the higher speed. The shock "locations" were nearly identical for the upper and lower surfaces for both $+1^\circ$ and -1° pitch angle. At the Mach 0.89 condition, boundary-layer induced shock motion apparently would dominate any periodic shock motion due to low frequency pitch oscillations. The disparity between the upper surface and lower surface responses was unexplained unless an undetected setup or data sequencing error occurred. It was observed in F-32 that greater data spread occurred than previously noted and this behavior suggested a slight change in flow angularity at this test condition which was not corrected for in positioning the model.

The above quasi-steady pressure results indicated that for the testing over the Mach number range of 0.35 through 0.84, the upper and lower surfaces were nearly identical. Hence the upper surface pressure ports used to measure unsteady pressures during the pitch oscillation tests gave an adequate picture of the airfoil response. For subcritical flow cases, the quasi-steady response was dominated by the rapid changes in pressure near the leading-edge. However, at Mach

0.80 the formation of shocks on the low pressure (suction) side of the airfoil occurred at non-zero pitch angles and this aspect of the flow generated large quasi-steady pressure values. At Mach 0.84 the data were dominated by the large shift in shock location between positive and negative pitch angles and this behavior led to the largest unsteady pressure amplitudes at low frequencies. The Mach 0.89 data indicated that such strong, periodic shock motion apparently decreased at higher Mach numbers. Further correlations of these quasi-steady results with the low frequency results are made in Chapter 5 below.

V. DISCUSSION OF UNSTEADY DATA

The unsteady data obtained on the two NACA 64A010 airfoil models and on the test section walls were recorded and analyzed for both fixed angle-of-attack testing and pitch oscillation experiments. All pitch oscillation testing reported here was of $\pm 1^\circ$ amplitude, and the center of rotation was the airfoil midchord. The data was obtained at a sampling rate of 2000 samples per second with low pass analog filtering above 1 Kiloherztz. Fourier analysis techniques (described in Appendix B) were applied to obtain power-spectral density plots, auto- and cross-correlations, and single frequency magnitude and phase values (by complex-demodulation processing). The influence of non-sinusoidal inputs on the oscillating pressures were examined to determine if analyses regarding only the input to the fundamental frequency corresponding to airfoil motion were sufficient. The nonlinear influence of shock motion on pressures was investigated.

Due to limitations in the instrumentation, unsteady data with and without model oscillation were obtained at fixed amplifier gain settings. The quantization noise due to the finite resolution of the Datel digitizer in combination with system electrical noise limited the accuracy of data obtained during fixed angle-of-attack test runs. However, due to the facts that the noise was generally broadband in nature and that the oscillating model pressures were much greater in

amplitude, the signal-to-noise ratio was greatly improved for the oscillating airfoil pressure data. Comparing overall root-mean-square (rms) pressure values to the recorded noise levels, the fixed model data had relatively low signal-to-noise ratios (as low as 6 dB in cases of high noise and low pressure fluctuations). Since, in general, the baseline noise was uncorrelated to the random pressure fluctuations in the model and tunnel wall boundary layers, these rms data values were corrected by subtracting the corresponding electrical noise values from pressure data by a root-sum-squares method. The comparison of broadband values of P_{rms} during the pitch oscillation to the electrical noise gave a minimum signal-to-noise ratio of 12 dB and a maximum of 57 dB. Under the conservative assumption that 1/100th of the electrical noise value would be seen in a one Hertz bandwidth of the 1000 Hertz range, the comparable signal-to-noise ratios for the unsteady pressure component obtained by complex demodulation techniques were a minimum of 45 dB and a maximum of 97 dB. The low signal-to-noise ratios were seen at transducer locations near the trailing-edge of the airfoil and in the upper plenum chamber behind the ventilated wall. These locations exhibited relatively low signals yet for the single frequency data displayed acceptable data quality. The large dynamic range of the pressure signals made it difficult to optimize the amplifier gain for improved signal-to-noise ratios.

5.1 TYPICAL UNSTEADY PRESSURE PLOTS

The waveforms of the pressure data were examined to deduce the relative influence of random and sinusoidal fluctuations on the individual pressure signals. Unsteady shock influence on the pressure

traces for both fixed and oscillating model tests were observed at the supercritical Mach numbers. Several of the current numerical approaches use the simplification that the airfoil response to oscillatory motion is manifest as sinusoidal pressure variations on the airfoil surface. However, shock motion and resultant boundary layer interactions tend to distort the sinusoidal waveforms. These additional periodic components are often neglected because they do not contribute significantly to the integrated lift and moment values, yet they are fundamental to understanding the nature of the shock motion.

The boundary layer pressure fluctuations measured during fixed angle-of-attack test runs were limited by the finite bit resolution of the analog/digital converter as seen in F-43. The upper curve showed that the data values covered a range of only six finite digitizer levels out of a possible 1024 binary bit patterns available. The lower curve displayed the random nature of most of the pressure data gathered at various locations on the fixed models and tunnel walls. The random fluctuations were the combined influences of turbulent boundary layer noise, tunnel noise, and the transducer electrical noise.

Additionally, the 60 Hz noise due to pickup of induced currents from the electromagnetic fields around electrical power lines was not discernible in the upper curve, but as shown in the next section, this signal was the dominant frequency in several power-spectral-density plots of the signals from the fixed model tests.

The major observation in these unsteady pressure traces obtained from constant pitch angle tests was that the contributions due to the tunnel, electrical, and normal boundary layer noise sources were

F-43 Unsteady Pressure Trace, $M = 0.35$, $\alpha = 0^\circ$, 15.2 cm Model, a) 45 percent x/C ; b) 90 percent x/C

negligible when compared to the oscillating airfoil responses seen in the rest of this section.

The pressure signals from the two airfoil models were examined for test series having identical model oscillation frequencies (therefore, the smaller model had a reduced frequency about half that of the other). Studying the scaled data plots provided a physical insight into the shock wave influence and the relation of periodic and random components.

At low speeds, the strongest oscillating pressure signals were observed at the 10 percent chord location. Comparison of data from both models at Mach 0.35 (see F-44) showed that the pressure signals were generally sinusoidal in shape with a smaller random noise component. The amplitudes of these signals were comparable for the two model sizes. These observations held for the rest of the airfoil pressures at the Mach 0.35 test condition. Off the model, comparison of the static probe signals showed a greater sinusoidal content in data from the larger model, while the other signal was dominated by random fluctuations (F-45). This probe data indicated more radiated acoustic power was felt near the ventilated walls for the larger model yet there was no apparent tunnel mode in resonance.

For the subcritical Mach numbers, the data traces generally had a single frequency sinusoidal shape with the random boundary layer noise added on. For these cases, time-linearized forms of the small-perturbation equation (e.g., the harmonic disturbance potential used in the UTRANS routine) should be adequate. Yet, for transonic flow conditions at Mach 0.80 and above, the sinusoidal character of the

F-44 Unsteady Pressure Trace, $M = 0.35$, $k = 0.46$,
10 percent x/C , a) 7.6 cm Model; b) 15.2 cm Model

F-45 Unsteady Pressure Trace, $M = 0.35$,
Static Probe, a) $k=0.46$, 7.6 cm Model; b) $k=0.90$, 15.2 cm Model

pressure data became distorted in the vicinity of local shock waves and subsequent boundary layer flow separation.

In the flow ahead of the shock influence region, the signals were strongly sinusoidal as exemplified in F-46, where as before, the small model produced greater signal amplitudes. Some indications of shock influence was seen at the 30 percent chord location, however, these results were not definitive. At 40 percent of chord, the data from the 15.2 centimeter chord model showed a periodic pattern which appeared to be altered from a pure sinusoidal form by a passing shock wave (F-47). This behavior is more readily apparent for the large model. The amplitudes were equivalent for these two models at the corresponding 40 percent chord locations. The rapid change in pressure following the peaks shown in the data from the larger model was indicative of the forward passage of a shock followed by the shock-induced boundary layer fluctuations. Further aft on the models, the shock induced boundary layer noise was present during the entire cycle of pitch oscillation. In F-48 pressure responses at the 60 percent chord showed a strong random component superimposed on the fundamental oscillation frequency. The 15.2 cm model produced a greater random noise component than did the 7.6 cm model. This behavior persisted over the rest of the airfoil surface downstream of the shock location. In general, at Mach 0.8 there appeared to be a relatively weak shock (that apparently dissipated during portions of the pitch cycle) which induced a strong, random pressure fluctuation at downstream locations. The induced boundary layer fluctuations were greater for the larger model

F-46 Unsteady Pressure Trace, $M = 0.80$,
10 percent x/C , a) $k=0.43$, 15.2 cm Model; b) $k=0.22$, 7.6 cm Model

F-47 Unsteady Pressure Trace, $M = 0.80$,
40 percent x/C , a) $k=0.43$, 15.2 cm Model; b) $k=0.22$, 7.6 cm Model

F-48 Unsteady Pressure Trace, $M = 0.80$,
60 percent x/C , a) $k=0.43$, 15.2 cm Model; b) $k=0.22$, 7.6 cm Model

which--based on its size--was operated at twice the Reynolds number of the small model.

The pressure signals near the upper ventilated wall exhibited a strong sinusoidal content as seen in F-49. The larger model still produced about 70 percent greater signal amplitudes at the static probe and upper plenum locations than observed during the 7.6 cm chord model tests. Spectral analyses of these signals confirmed these observations. For the smaller model, the random fluctuations had a more pronounced influence since the fundamental frequency amplitudes are smaller.

The next supercritical test condition (Mach 0.84) displayed several trends similar to behavior observed at Mach 0.80. At 10 percent chord, the small model again responded with greater amplitude than seen on the large model (see F-50). The large amplitude sinusoidal content of signals from both models over the first half of the chord length significantly dominated any random fluctuations. However, near the 50 percent chord location, there were strong signs of shock influence. The larger model displayed a nearly triangular sawtooth waveform while the smaller model exhibited quite different behavior (see F-51). Both models were likely seeing the influence due to shock motion near the forward limit of the shock travel. Slightly further aft at 55 percent chord the amplitude of the signal increased significantly and rapid variations in pressure associated with shock passage were clearly evident (see F-52). In the same figure, the aft extent of the apparent shock motion (near 60 percent chord) showed the influence of the shock passage on the turbulent boundary layer. Following the rapid change in

F-49 Unsteady Pressure Trace, $M = 0.80$,
 a) $k=0.43$, 15.2 cm Model, Static Probe; b) $k=0.22$, 7.6 cm Model,
 Static Probe; c) $k=0.43$, 15.2 cm Model, Plenum; d) $k=0.22$, 7.6 cm
 Model, Plenum

F-50 Unsteady Pressure Trace, $M = 0.84$,
 10 percent x/C , a) $k=0.40$, 15.2 cm Model; b) $k=0.21$, 7.6 cm Model

F-51 Unsteady Pressure Trace, $M = 0.84$,
 50 percent x/c , a) $k=0.40$, 15.2 cm Model; b) $k=0.21$, 7.6 cm Model

F-52 Unsteady Pressure Trace, $M = 0.84$, a) $k = 0.40$, 15.2 cm Model, 55 percent x/C ; b) $k=0.40$, 15.2 cm Model, 60 percent x/C ; c) $k=0.21$, 7.6 cm Model, 60 percent x/C

pressure associated with shock passage, a more gradual pressure rise containing perceptible random fluctuations were observed. This pattern of random boundary layer fluctuations was terminated by another rapid pressure change indicative of shock passage in the alternate direction. These signals contained the largest sinusoidal amplitudes measuring during the current oscillating airfoil testing. While the large model displayed the characteristics of a highly perturbed boundary layer superimposed on the fundamental pressure oscillation, the small model indicated that the shock may travel as far aft as the 70 percent chord location (see F-53). Also, there was a strong indication of a higher harmonic response at that location and this particular aspect of the data was well correlated with the power-spectral-density and autocorrelation results. It is hypothesized that this behavior is either a manifestation of coupling between the tunnel resonance modes and/or a trailing-edge feedback phenomenon which couples the shock motions on the upper and lower surfaces.

The highest Mach number condition tested showed nearly pure sinusoidal variations at pressure locations covering the first 65 percent of the airfoil chord. The shock influence was seen vividly near the 70 and 90 percent locations (see F-54). At these two measuring stations the higher harmonic components were clearly visible although random boundary layer fluctuations were more noticeable near the trailing-edge. While a strong periodicity remained, it was evident that the shock motion influenced the local airfoil pressures at several harmonic frequencies.

F-53 Unsteady Pressure Trace, $M = 0.84$,
70 percent x/C , a) $k=0.40$, 15.2 cm Model; b) $k=0.21$, 7.6 cm Model

F-54 Unsteady Pressure Trace, $M = 0.89$, $k = 0.38$, 15.2 cm Model,
a) 70 percent x/C ; b) 90 percent x/C

The general patterns observed in the scaled data plots provided supporting evidence to the deductions which were obtained through other data forms. Although quantification of these data plots was difficult, their qualitative values in determining shock locations, shock induced boundary layer fluctuations and the relation of signals to various noise levels were significant. The unsteady pressure signals on the airfoil models are seen to have nearly equal amplitude and shape for both models tested at the same frequency at Mach numbers of 0.80 and below. The transducers near the upper wall exhibit the same sinusoidal shape (with superimposed random boundary layer noise) but the amplitudes for the 7.6 cm chord model tests are about one-half as great as for the 15.2 cm model. At Mach numbers of 0.84 and above pressure signals ahead of the shock interaction zones are similar for both models, but those near 70 percent X/C tend to disagree as to higher harmonic influence. These differences are attributed to differences in the shock/boundary layer interaction for the two models.

5.2 POWER-SPECTRAL-DENSITY DATA

The power-spectral-density (PSD) data were obtained by direct Fast Fourier Transform (FFT) methods from the digitized data samples. These pressure spectra were examined for both subcritical and supercritical cases. The growth in the number and relative magnitudes of harmonic components was investigated for regions displaying influence of shock waves and large random boundary layer fluctuations. These observations were correlated with no appearance of non-sinusoidal, periodic waveforms in the pressure traces. Also, the frequency content

attributable to electrical noise and baseline flow sources was analyzed to establish the lower limits of resolution for the spectral data.

The data presented in this section was obtained over the limited bandwidth of 1 to 1000 Hertz and therefore does not comprehensively treat the higher frequency oscillations observed in previous tunnel surveys and in possible oscillating flow phenomena due to shock/boundary layer interaction with higher frequency content. The data were digitized at a 4000 sample per second rate for a total sample period of 1.025 seconds. Anti-aliasing analog filters on the instrumentation amplifiers were set at 1000 Hertz to prevent frequency fold-over that can occur due to the digital sampling process. The equivalent filter resolution was about 0.5 Hertz, however, smoothing over 1 Hertz bandwidths was performed during the PSD computations. Frequencies above 130 Hertz were smoothed over greater bandwidths since they were beyond the model oscillation frequencies.

The data from the fixed angle-of-attack test runs generally had a broadband random noise content superimposed on a 60 Hertz electrical pickup from line power. Typical spectral from subcritical Mach numbers of 0.35 and 0.52 are shown in F-55. The line power component contained nominally an order-of-magnitude greater power than the rest of the spectral components but there was no indication that harmonics of the 60 Hertz were present. Other narrowband peaks were observed, however, their magnitudes were not significantly greater than the neighboring noise levels. There was no indications of acoustic modes corresponding to Mabey's predictions of resonance in ventilated tunnels. Spectra computed for locations exhibiting the largest pressure fluctuations

F-55 Pressure PSD, $\alpha = 0$, 15.2 cm Model,
50 percent x/C, a) M=0.35; b) M=0.52

were not observed to have any significant narrowband peaks but were seen to have slightly greater broadband noise levels. The 60 Hz signal remained below $4 \times 10^{-5}(\text{psi})^2/\text{Hz}$ for all of the data and the magnitude of this peak was nearly constant and corresponded to an rms pressure variation of 0.006 psi. The Mach 0.70 data showed another peak power level at 185 Hertz, however, this peak is not observed at other locations such as at the static probe station and was not a tunnel resonance (see F-56). At these subcritical Mach numbers, the random boundary layer fluctuations and 60 Hz line power noise dominated the PSD data.

For the constant pitch angle of zero degrees, the Mach 0.80 steady-state pressure data showed little indication of a shock wave even though supersonic flow regions occurred near the region of maximum thickness on the airfoil. However, the PSD data showed a rise in the broadband power levels above 200 Hz as seen in F-57. The lower frequencies were relatively unaffected, and the random contributions apparently peaked in the 600 to 700 Hz range. This behavior was attributed to local turbulent boundary layer disturbances which were amplified in the region of unfavorable pressure gradient aft of the maximum thickness at the 40 percent chord.

The supercritical cases at Mach 0.84 and 0.89 had clear evidence of shocks at the zero pitch setting and the boundary layer fluctuations were observed to increase dramatically in the region slightly downstream of the mean shock location. Just ahead of the 55 percent nominal shock location, the pressure spectra showed little evidence of additional boundary layer perturbations (see F-58), however, at 60

F-56 Pressure PSD, $M = 0.70$, $\alpha = 0$, 15.2 cm Model, a) 60 percent x/C; b) static probe

F-57 Pressure PSD, $M = 0.80$, $\alpha = 0$, 15.2 cm Model, 60 percent x/C

F-58 Pressure PSD, $M = 0.84$, $\alpha = 0$, 15.2 cm Model, a) 50 percent x/C ; b) 60 percent x/C

percent chord there was a notable increase in the broadband noise levels over the entire frequency range analyzed. The minor peaks indicated in the PSD at 60 percent chord had little relative power. The same behavior was observed at Mach 0.89 test conditions (F-59) where spectral levels ahead of the mean shock location were not influenced, but those aft were dominated by broadband random noise fluctuations.

The results obtained through spectral analysis of the fixed pitch angle testing can be generalized as follows:

- (1) The baseline spectral content with a random, broadband noise with the 60 Hertz peak corresponding to pickup of line power;
- (2) There was no clear indications of tunnel resonance or excessive freestream turbulence in the frequency range analyzed (1 to 1000 Hertz); and
- (3) The shock boundary layer interaction at supercritical Mach number conditions generated higher random pressure fluctuations in the regions immediately downstream of the mean shock locations. Most of all, the baseline spectral content was observed to be low in comparison with the oscillating model data as is discussed below.

Spectral analyses of the pressures measured during the oscillating airfoil testing showed the relative magnitudes of the fundamental frequency and higher order harmonic components. In most cases, the fundamental mode corresponding to the pitching frequency dominated by two orders-of-magnitude or more over any of the harmonics. The leading edge region experienced a strong first harmonic variation. The

F-59 Pressure PSD, $M = 0.89$, $\alpha = 0$, 15.2 cm Model, a) 55 percent x/C ; b) 60 percent x/C

spectral power in the fundamental was relatively dominant over the first half of the airfoil, however, power in the harmonic peaks increased rapidly near regions influenced by shock motion. Aft of such shock regions and generally over the aft chord location, the broadband random noise levels increased while the fundamental and harmonic peaks went to much lower levels. Very little harmonic content was detected by the transducers located in the upper part of the test section and the relative powers were down about ten-fold from model pressures (even more for the smaller model).

Measurement of the unsteady pressure near the leading-edge showed a strong first harmonic peak which in some cases exceeded power in the fundamental mode. This resulted from the decrease in pressure at the nose of the airfoil as it oscillated either side of the zero mean pitch position. The stagnation point moved either side of the measuring location such that the local pressure decreased for both positive and negative pitch angles. As seen in F-60, the first harmonic contained more power than the fundamental for both models when tested at subcritical Mach numbers and low to moderate reduced frequencies. However, at higher reduced frequencies and at supercritical Mach numbers the fundamental mode contained more acoustic energy while the harmonic peak decreased in amplitude.

At 10 percent chord, the fundamental mode energy increased dramatically and at subcritical Mach numbers this nearly pure sinusoidal pressure variation produced the maximum local unsteady pressure loading. The power in the first harmonic was two to four orders-of-magnitude less than the fundamental (see F-61). The higher

F-60 Pressure PSD, 15.2 cm Model, 0 percent x/C,
a) $M=0.35$, $k=0.46$; b) $M=0.35$, $k=0.90$; c) $M=0.70$, $k=0.47$;
d) $M=0.84$, $k=0.40$

F-61 Pressure PSD, $M = 0.52$, 10 percent x/C ,
 a) $k=0.31$, 15.2 cm Model; b) $k=0.31$, 7.6 cm Model; c) $k=0.62$,
 15.2 cm Model

harmonic peaks were at least another ten-fold lower in magnitude. This behavior was observed for nearly all Mach numbers and frequencies over the first third of the airfoil chord. The broadband random noise levels along with the 70 Hz noise were not significantly greater than those observed during fixed angle tests.

At the 30 percent chord location, the number and energy content of harmonic peaks had increased in comparison with the data at 10 percent chord. This growth was enhanced at the higher driving frequencies and higher Mach number conditions (see F-62). Up to eight harmonic peaks were observed, yet the strongest power--usually found at the first harmonic frequency--was a factor of eighty below that of the basic mode. The relative signal amplitudes between the fundamental and the harmonic modes were still greater than 9 to 1.

Data from the midchord location exemplified the influence of an oscillating shock on local pressure spectra. The PSDs shown in F-63 are for a range of subcritical and supercritical test conditions. At Mach 0.70, the pressure response to pitching motion was nearly a pure sinusoidal function. The increase in freestream velocity to Mach 0.80 conditions combined with the pitching motion, generated locally supersonic flow at the midchord position which was influenced by the periodic formation and motion of a weak shock wave. Power in the fundamental mode increased, and the harmonic content was magnified even more. The first harmonic peak indicated a signal amplitude about 25 percent that of the basic mode. The nominal shock location was apparently near the 55 percent chord, thus it was very likely that the shock passed directly over the midchord during each pitch cycle. At

F-62 Pressure PSD, $M = 0.80$, 30 percent x/C
 a) $k=0.21$, 15.2 cm Model; b) $k=0.22$, 7.6 cm Model; c) $k=0.42$, 15.2 cm Model

F-63 Pressure PSD, 15.2 cm Model, 50 percent x/C,
 a) $M=0.70$, $k=0.23$; b) $M=0.80$, $k=0.21$; c) $M=0.84$, $k=0.20$;
 d) $M=0.89$, $k=0.21$

the next higher Mach number, the steady pressure distributions indicated a mean shock position further aft. The peak power at the driving frequency increased slightly while the harmonic content decreased. The higher order harmonic peaks became visible, however, that event was due more to a relative decrease in the broadband random noise content than an increase in the harmonic power levels. The ratio of fundamental to harmonic power increased to nearly three orders-of-magnitude in value. At the highest Mach number condition ($M = 0.89$), all frequencies saw a relative decrease in acoustic power and up to ten multiple tones were clearly visible. This reduced magnitude midchord signal was still dominated by the power in the fundamental mode.

Local shock influence was strongest during tests at Mach 0.84 near the 60 percent chord position. The subsequent distortion of the scaled pressure seen in F-52 produced large harmonic peaks (see F-64). The amplitudes of the first harmonic components were nearly one-third those of the fundamental waves for the range of reduced frequencies examined. For the large model, the power decreased significantly beyond the first harmonic, but the smaller model had nearly equal power in the second harmonic as in the first one. This would indicate a more rapid change in pressure as the shock passed over the measuring location generating more nearly a square wave signal.

The relative strength of the first harmonic was even greater at the 70 percent chord location during the high Mach number testing (see F-65). First harmonic peaks were observed at Mach 0.84 and 0.89 which had power level corresponding to between 65 and 75 percent as much

F-64 Pressure PSD, $M = 0.84$, 50 percent x/C ,
 a) $k=0.20$, 15.2 cm Model; b) $k=0.21$, 7.6 cm Model; c) $k=0.40$,
 15.2 cm Model

F-65 Pressure PSD, $M = 0.84$, 70 percent x/C
 a) $k=0.21$, 7.6 cm Model; b) $k=0.40$, 15.2 cm Model; c) $k=0.38$, 15.2 cm Model

signal amplitude as did the basic mode. This strong harmonic response could possibly be explained in terms of the acoustic feedback mechanism previously hypothesized as coupling the pressure field above and below the aft portion of the airfoil around the trailing edge. Alternating shocks on the upper and lower surfaces which were closer to 90° out-of-phase than an expected 180° phase lag would produce an influence near the trailing edge which would closely resemble a doubled frequency response superimposed on the basic oscillation. This hypothesis is also supported through comparison with the scaled data signal in F-53 for the 7.6 cm model and F-54 for the 15.2 cm model.

Large unsteady pressures recorded during pitch oscillation tests at Mach 0.89 were observed further aft on the model near the 80 percent chord location. The spectral content shown in F-66 suggested that the hypothesis given above was invalid unless this location was near an apparent acoustic node since the first harmonic peak was nearly three orders-of-magnitude lower in power than the fundamental mode. This was due to an increase in the acoustic power at the model oscillation frequency and a decline in power of the first harmonic. The apparent effect of the local shock oscillation at this location resembled a nearly pure sinusoidal waveform.

However, nearer the trailing edge the doubled frequency component again appeared to contain a non-negligible fraction of the total acoustic power for tests run at Mach 0.89. As shown in the first PSD plot of F-67, the 90 percent chord location indicated a relatively weak sinusoidal response with a broadband random noise content. This response was typical for the subcritical test conditions. At Mach

F-66 Pressure PSD, $M = 0.89$, 15.2 cm Model, 60 percent x/C,
 a) $k=0.19$; b) $k=0.38$

F-67 Pressure PSD, 15.2 cm Model, 90 percent x/C,
a) $M=0.52, k=0.31$; b) $M=0.80, k=0.21$; c) $M=0.89, k=0.19$

0.80, the fundamental and first harmonic modes showed larger power levels, yet in comparison with location near the midchord, the total acoustic power here was still relatively small. The Mach 0.89 spectra showed increased power in both modes and nearly equal amplitudes in the fundamental and first harmonic modes. This observation reinforced the hypothesis about the source of the doubled frequency component. Due to the low magnitudes of all peak power levels detected at the 90 percent chord, higher harmonics were not clearly observable. This was in part due to the expected increase in random pressure fluctuations in the turbulent boundary layers near the trailing edge.

Propagation of the pressure waves away from the oscillating airfoil surface was investigated through analysis of the pressure spectra on the model and various locations around the upper test section. A series of PSD plots obtained from Mach 0.80 test data for the 7.6 cm model given in F-68 illustrated the attenuation of the fundamental mode and the harmonics at various distances from the airfoil. A typical pressure spectra at the 50 percent chord location showed the fundamental mode and several harmonic peaks due to local shock motion on the airfoil. At one chord distance above the airfoil, pressure data from the tunnel sidewall indicated a small loss in power in the fundamental mode, however, power in the first harmonic mode decreased tenfold and over two orders-of-magnitude for the higher harmonics. The broadband random fluctuations were also reduced by an order-of-magnitude. At three chord lengths above the model the static probe mounted five centimeters off the upper wall and a sidewall transducer at an equivalent height both recorded peak levels five times

F-68 Pressure PSD, $M = 0.80$, $k = 0.22$, 7.6 cm Model, a) 50 percent x/C ; b) Low Wall; c) High Wall; d) Static Probe; e) Top Wall; f) Plenum

lower than the probe 7.6 centimeters above the model. The broadband spectral levels on the upper sidewall were higher than those on the probe due to the turbulent boundary layer characteristics. At a sensor location in the upper ventilated wall, the fundamental peak power is nearly the same as measured at the static probe, and the energy content in the boundary layer pressure disturbances were ten times those on the probe. The perforated wall at the top of the test section attenuated the acoustic power in the fundamental mode by nearly a factor of 8, and the harmonic content was obscured by the random content in the plenum and transducer electrical noise. The acoustic field generated by the oscillating airfoil was observed to rapidly dissipate at relatively short distances above the airfoil. No evidence of transverse resonance in the test section could be found. The perforated plate apparently transmitted only about 10 percent of the acoustic energy in the fundamental mode and negligible amounts at the higher harmonic frequencies.

In comparing the spectral content of signals from the two airfoil models, it is apparent that spectra at the same physical frequency are more closely correlated than are spectra of the same reduced frequency (see F-61 and F-62). It is seen that at $M = 0.80$, $k = 0.21$, the smaller model exhibits higher harmonic content. However, for the reduced frequencies of $k = 0.22$ and $k = 0.42$ (for the 7.6 cm and 15.2 cm chord models respectively) the higher harmonic content is comparable. This indicates that there may be some influence on these higher frequencies due to the wind tunnel acoustic properties. These harmonic frequencies are near the range of tunnel resonances predicted

by Mabey's criteria (see F-25). However, this influence of tunnel response to the higher harmonics is not observed to have significant influence on the fundamental mode corresponding to model motion. This behavior is apparently exacerbated at the higher Mach numbers where airfoil locations nearest the region of the oscillating shock have greater harmonic power levels. For cases where the first harmonic is of nearly equal magnitude as the fundamental mode (see F-65), the corresponding pressure traces (F-53 and F-54) had shown the anomalous behavior between the two model sizes. From these results, it is apparent that although tunnel resonance does not appear to directly influence pressure oscillations at the driven model frequencies, there is some coupling with the higher harmonics which may influence shock motion and have a resultant effect on the unsteady pressure coefficients at the higher Mach numbers and frequencies over the aft portions of the airfoil. However, upon examination of the transducer spectra near the upper wall of the tunnel test section, it is not clearly evident that the tunnel resonance modes have been significantly excited (see F-68).

The power-spectral-density data served to explain the relative power levels in fundamental and harmonic modes. For the most part, the power levels at the first harmonic peak were several orders-of-magnitude below those found in the fundamental mode. Only at the leading edge where the pressure decreased from the stagnation value for both positive and negative pitch angles and near the trailing edge at higher Mach number conditions where the proposed acoustic feedback mechanism coupled with local shock motion were the power levels at

harmonic frequencies comparable to those in the fundamental mode. However, since even the fundamental modes have relatively low values at these locations, the harmonic frequencies did not significantly influence the integrated lift and moment coefficients.

5.3 CORRELATION PLOTS

The time-lag correlation results were obtained through FFT methods. Their value here lies in their representation of the components of a signal which are correlated to previous portions of the signal; i.e., the correlations can display the relative periodicity of a signal while suppressing the truly random (and hence uncorrelatable) portions. In computing the auto-correlation values, the raw signal was first Fourier transformed to the frequency domain. Then the moduli of the spectral components are inversely transformed back into the temporal domain to form the auto-covariance coefficients. The coefficients were then normalized with the original signal root-mean-square amplitude for plotting as the autocorrelation function. The existence of a strong sinusoidal mode was reflected directly in the autocorrelation results, however, due to the method of calculation, the relative phase lags for the different frequencies were not preserved. Additional details on the computational method are given in Appendix B.

As seen from the above discussion of the PSD results, most of the unsteady pressure signals were dominated by the single-frequency sinusoidal components corresponding to the pitch oscillation frequency.

During the fixed angle-of-attack testing, the only major periodic component observed was the 60 cycle line power pickup. That component

F-69 Auto-Correlation, $M = 0.52$, $\alpha = 0$, 15.2 cm Model, a) 50 percent x/C ; b) Top Wall

appeared in varying degrees as exemplified by F-69. The 60 Hz component was clearly visible at the 30 percent chord location for Mach 0.52 conditions, yet a large portion of the signal was still random pressure fluctuations. At the upper ventilated wall, the boundary layer generated sufficient broadband fluctuations to obscure the 60 Hz line noise.

The supercritical test conditions at Mach 0.80 produced some higher frequency yet broadband components peaked near 400 Hz. The correlation data at 40 percent chord showed that component (see F-70). The relative amplitude of this component was small compared to the random fluctuations.

These observations of the correlation data for fixed model tests confirmed previous conclusions that the unsteady pressures present in the tunnel did not contain any resonant peaks in the frequency range of interest. The line power pickup present in some of the signals was not a significant influence on the oscillating airfoil pressure data.

The correlation plots generated from the oscillating airfoil data displayed the strong sinusoidal character prevalent in the large majority of the pressure signals recorded. For strong fundamental peaks in the spectra, corresponding correlation plots were essentially pure sine waves whose normalized amplitude approached unity. The only notable exceptions to these general remarks occurred near the leading- and trailing-edges coincidental with observations of strong harmonics in the PSD plots, or near the upper tunnel walls where random noise levels were relatively well attenuated.

F-70 Auto-Correlation, $M = 0.80$, $\alpha = 0$, 15.2 cm Model, 40 percent x/C

The leading-edge condition at subcritical Mach numbers vividly showed the doubled frequency waveform which was characterized in the PSD data as having more power in the first harmonic than in the fundamental mode (see F-71). This behavior was not clearly apparent at the supercritical Mach number conditions. The harmonic response was obscured by the partially correlated fluctuations at higher frequencies.

Wave distortion due to localized shock motion during the pitch oscillation experiments was manifest in several ways. The correlation plots shown in F-72 give three typical waveforms. The first plot showed a sine wave which was "clipped" on one side due to non-sinusoidal shock influence. The second correlation function was typical of most of the data analyzed with only the fundamental model oscillation frequency containing significant correlation. The final plot was indicative of the strong first harmonic content observed at some locations near the oscillating shock regions. These various types of behavior were in good agreement with the observations made with the scaled data signals. The principal value of the correlation plots were exemplified by comparing these plots with the scaled data plots in F-53 where it becomes evident that the small perturbations superimposed over the main periodic waveform were random in nature since no correlation of the higher frequencies was observed.

The doubled frequency hypothesis used to explain the trailing-edge behavior at the Mach 0.89 condition was reinforced by the strongly correlated signal shown in F-73. The first harmonic frequency showed significant amplitude at the 90 percent chord location for reduced

F-71 Auto-Correlation, 15.2 cm Model, 0 percent x/C ;
 a) $M=0.52$, $k=0.62$; b) $M=0.84$, $k=0.40$

F-72 Auto-Correlation, $M = 0.84$, a) $k = 0.40$, 15.2 cm Model, 80 percent x/C ; b) $k=0.40$, 15.2 cm Model, 70 percent x/C ; c) $k=0.21$, 7.6 cm Model, 70 percent x/C

F-73 Auto-Correlation, $M = 0.89$, $k = 0.19$, 15.2 cm Model,
 a) $k=0.19$, 90 percent x/C ; b) $k=0.38$, 70 percent x/C ; c) $k=0.38$,
 90 percent x/C

frequencies of $k = 0.19$ and at 70 percent for $k = 0.38$. The decrease in first harmonic amplitude at the higher model pitching frequency for the 90 percent location suggested a forward movement of the mean shock location for higher reduced frequencies.

In this section we discussed the nature of the autocorrelation data obtained during both fixed and oscillating airfoil testing. The results reinforced the previous conclusions and hypothesis. Pure sinusoidal correlations were observed for the majority of the oscillating airfoil data except for leading-edge pressures and some responses obtained in regions displaying local shock motion behavior. These anomalies were characterized as having a noticeable first harmonic component, and such results were in good agreement with the scaled data plots and PSD curves. As noted in the previous discussions, the harmonic content of signals from the aft portion of the 7.6 cm model experiences higher power in the first harmonic than does the 15.2 cm model at Mach 0.84. The larger model exhibits the same behavior for Mach 0.89 conditions. Therefore, it is apparent that the hypothesis concerning coupling between the tunnel acoustic modes and harmonic behavior of the shock at high transonic Mach numbers is not conclusive. But rather, there is an apparent influence associated with some secondary factors such as Reynold's number, etc.

5.4 OSCILLATING AIRFOIL PRESSURE COEFFICIENTS

The results obtained from the complex demodulation data reduction analysis provided accurate estimates of signal amplitude and phase at a single frequency. The unsteady pressure fluctuations and accelerometer data were processed by the complex demodulation routine UA/SYN which

computed the oscillating pressure coefficients, C_p' , magnitudes and phase lags relative to the model motion. The influence of shock motion on the peak fluctuations was ascertained for both fixed angle and pitch oscillation testing. The quasi-steady values of ΔC_p projected from the steady-state C_p data and theoretical predictions were compared to data obtained at the low frequencies. Experimental values of C_p' were compared with the numerical predictions obtained with the UTRANS, LTRAN2, and TWODI computer codes and with experimental data obtained at NASA Ames.

The broadband unsteady pressure levels measured during fixed pitch angle tests were indicative of the turbulent boundary layer hydrodynamic pressure disturbances and any local shock influences generated by increased boundary layer disturbances or natural shock oscillation. The $C_{p_{rms}}$ values plotted in F-74 showed the unsteady pressure changes due to Mach number and pitch angle changes. At Mach 0.35 the noise levels were dominated by the ambient tunnel noise. As a result the signal levels saw little influence due to change in angle-of-attack, and the levels observed were twice as high as observed at the other Mach numbers. For the other subcritical Mach numbers, 0.52 and 0.70, the maximum unsteady pressure levels were detected at 45 percent chord where the pressure transitions from a favorable gradient to an adverse gradient (just downstream of the maximum airfoil thickness). Data from supercritical test runs distinctly showed the influence of local shock waves. At Mach 0.80, peak unsteady pressure levels were detected at the 45 percent chord location with a pitch setting of zero degrees. For positive angles, this maximum shifted aft

F-74 Unsteady Pressure Distributions, Fixed Pitch, a) $M = 0.35$
 b) $M = 0.52$; c) $M = 0.70$; d) $M = 0.80$; e) $M = 0.84$; f) $M = 0.89$

on the airfoil, and for negative angles it shifted forward. At negative angles, the upper surface saw decreased fluctuation levels. The shock influence was centered at 60 percent chord for 0° angle-of-attack. The forward shift of the peak value for negative pitch angle was again observed, however, the rms levels remained nearly constant. At the positive pitch setting the shock influence apparently was maximized in the region between 65 and 80 percent chord where unsteady pressure data were not obtained due to transducer malfunction. As Mach number increased, the shock influence was progressively shifted aft until it was observed to peak near the 80 percent chord for Mach 0.89. Again, inadequate spatial resolution prevented accurate determination of the locations and magnitudes of the peak unsteady pressure levels.

The unsteady pressure levels measured during the fixed model testing did not represent significant noise levels which contaminated the oscillating airfoil pressure coefficients. As discussed above, the spectral content of these fixed angle data was mostly broadband random fluctuations whose spectral contents at the various model oscillation frequencies were negligible.

The magnitude of the oscillating airfoil pressure (fundamental mode only) will be denoted as $|C_p'|$ and the phase lag as θ . As discussed in Appendices B and E, the complex demodulation processing technique predicts the real and imaginary components of the unsteady signal at the selected frequency. The inherent errors were characterized as a confidence region encircling the estimated values plotted in the complex plane. This circular confidence region was nearly constant in

size, therefore, the relative errors for magnitude and phase at small magnitudes were inherently larger. This limitation was not considered serious since the lower magnitude results did not contribute significantly to the integrated unsteady loads.

The tests conducted at Mach 0.35 produced the oscillating pressure results shown in F-75 through F-80. At low frequency both models experienced similar oscillating pressure magnitudes in comparison with each other and with the quasi-steady data shown in F-54. The maximum value measured was at the 10 percent chord location with nearly equal values recorded for both the 7.6 cm and 15.2 cm models. The phase lag for the larger model varied from near -180° to -140° at $k = 0.16$. The experimental results for moderate frequencies (see F-77) were compared with predictions obtained from the time-marching method in the LTRAN2 code and from the integral method employed in the TWODI code. The pressure amplitudes measured on the 7.6 cm model were slightly greater than for the large model and the two sets of analytical results. The phase lag results were in reasonable agreement for both sets of model data and TWODI results. However, the LTRAN2 code did not give reasonable phase lag results. There were no general trends in the pressure amplitudes due to increased frequency, however, the phase lag data showed a greater variation at $k = 0.90$, spanning from -180° near the leading edge to -80° at the 90 percent location. Static probe data (plotted at the right-hand margins) show relatively low unsteady pressure levels--near values observed close to the trailing edge of the models.

Similar results were observed at the next higher subcritical Mach number of 0.52, where the maximum unsteady pressure amplitudes were

F-75 Oscillating Airfoil Pressure Data,
 $k = 0.13$, $M = 0.35$

F-76 Oscillating Airfoil Pressure Data,
 $k = 0.23$, $M = 0.35$

F-77 Oscillating Airfoil Pressure Data,
 $k = 0.46$, $M = 0.35$

F-78 Oscillating Airfoil Pressure Data,
 $k = 0.64$, $M = 0.35$

F-79 Oscillating Airfoil Pressure Data,
 $k = 0.76$, $M = 0.35$

F-80 Oscillating Airfoil Pressure Data,
 $k = 0.90$, $M = 0.35$

observed at 10 percent chord. The monotonic decrease in both magnitude and phase lag are displayed in F-81 through F-86. At the lowest reduced frequencies examined the two models generated pressure amplitudes equal to the quasi-steady results in F-38. The phase lag data showed nearly identical values. The phase lag over the airfoil demonstrated the same variations observed for Mach 0.35. The analytical and experimental comparisons showed excellent agreement for the pressure magnitudes except for the small model which exhibited higher than expected levels. The phase lag predictions produced by the TWODI code were in excellent agreement with the model data. The amplitudes near the leading edge decreased slightly with the increased frequencies and the phase lag variation along the model increased with frequency. At the highest reduced frequency, the difference in phase lag between the 10 and 90 percent locations was about 100° .

The next subcritical Mach number (0.70) demonstrated good agreement with the quasi-steady pressure amplitudes given in F-39 except for the slightly greater amplitude observed at the 10 percent position for the oscillating airfoil data. The Mach 0.70 data (given in F-87 through F-92) continued to display the same general trends observed in the other subcritical data. The experimental results continued to be in good agreement for pressure amplitudes at the low to moderate reduced frequencies with a notable departure seen over the first 30 percent of the small model from the LTRAN2 predictions and 15.2 cm model. The phase lag results of the two models maintained good agreement with the TWODI results. The oscillating pressure amplitude apparently declined slightly at the moderate reduced frequencies and then maintained nearly

F-81 Oscillating Airfoil Pressure Data,
 $k = 0.09$, $M = 0.52$

F-82 Oscillating Airfoil Pressure Data,
 $k = 0.16$, $M = 0.52$

F-83 Oscillating Airfoil Pressure Data,
 $k = 0.31$, $M = 0.52$

F-84 Oscillating Airfoil Pressure Data,
 $k = 0.43$, $M = 0.52$

F-85 Oscillating Airfoil Pressure Data,
 $k = 0.52$, $M = 0.52$

F-86 Oscillating Airfoil Pressure Data,
 $k = 0.62$, $M = 0.52$

F-87 Oscillating Airfoil Pressure Data,
 $k = 0.07$, $M = 0.70$

F-88 Oscillating Airfoil Pressure Data,
 $k = 0.12$, $M = 0.70$

F-89 Oscillating Airfoil Pressure Data,
 $k = 0.23$, $M = 0.70$

F-90 Oscillating Airfoil Pressure Data,
 $k = 0.32$, $M = 0.70$

F-91 Oscillating Airfoil Pressure Data,
 $k = 0.39$, $M = 0.70$

F-92 Oscillating Airfoil Pressure Data,
 $k = 0.47$, $M = 0.70$

constant values. At a reduced frequency of 0.47, the 15.2 cm model experienced about a 105° variation in phase between the 10 to 90 percent chord regions. The monotonic trends in unsteady pressure amplitudes and phase lags were again confirmed for the subcritical test condition.

The first supercritical test condition (Mach 0.80) was investigated for reduced frequencies of 0.06 to 0.43 (see F-93 through F-99). The monotonic variations in pressure amplitudes and phase lags were distorted by the local shock influence near the midchord position. At low frequencies the magnitude of the unsteady pressure decreased from the 10 percent chord to the 45 percent chord, then dipped to relatively low levels over the aft 50 percent of the chord. The phase lag in the supersonic bubble was nearly constant at -180° . This was indicative of a piston response in the supersonic zone. The relative maximum observed at the 45 percent chord location of the quasi-steady pressure data (F-39), was not present in the low frequency oscillating pressure data. The amplitudes measured near the mean shock location were more representative of the "piston response" generated in the local supersonic flow region than that of alternating pressure steps generated by an oscillating shock wave. Downstream of the shock locations, the amplitudes showed a step decrease accompanied by a 10 to 45 degree shift in phase lag. The size of the shift was roughly proportional to the reduced frequency. The monotonic decline in unsteady pressure amplitude was again observed in the subsonic flow over the rear half of the airfoil model. Correlations with experimental data from the NASA Ames 11-foot wind tunnel [from Davis

F-93 Oscillating Airfoil Pressure
Data, $k = 0.06$, $M = 0.80$

F-94 Oscillating Airfoil Pressure
Data, $k = 0.10$, $M = 0.80$

F-95 Oscillating Airfoil Pressure
Data, $k = 0.21$, $M = 0.80$

F-96 Oscillating Airfoil Pressure Data,
 $k = 0.29$, $M = 0.80$

F-97 Oscillating Airfoil Pressure Data,
 $k = 0.35$, $M = 0.80$

F-98 Oscillating Airfoil Pressure Data,
 $k = 0.42$, $M = 0.80$

F-99 Oscillating Airfoil Pressure Data,
 $k = 0.43$, $M = 0.80$

and Malcolm (1979)] and the computational routines were examined. The integral solution fails due to its inability to predict the local supersonic flow conditions and shock waves. Hence, the amplitudes computed with TWODI were in poor agreement. The time marching and harmonic, time-linearized techniques in the LTRAN2 and UTRANS codes respectively had predicted large oscillating pressure amplitudes and phase shifts associated with local shock motion. The local peak in the Ames data also disagreed with the OSU/AARL data. This may be due to slight differences in model thickness mentioned in section 4.1 above and coincidental misplacement of pressure sensors, out of the apparently narrow shock travel range. The 15.2 centimeter model showed a weak response to the shock motion. The phase lag for $k = 0.21$ was observed to increase slightly at positions downstream of the shock wave, in contrast to observations made at higher frequencies and for subcritical conditions. (Note: the phase lags for low amplitude signals were subject to larger error bounds). The phase lags for the small model are more closely correlated to the analytical predictions. This first supercritical test condition, in general, exhibited little shock wave influence.

The shock influence at Mach 0.84 was significantly greater than noted at the supercritical Mach 0.80 condition. Unsteady pressure amplitudes increased over threefold in the neighborhood of the shock for the range of reduced frequencies examined (see F-100 through F-105). The peak amplitudes occurred at the 55 and 60 percent chord locations which span the range of shock positions observed in the steady-state pressure data. The low frequency data agreed well with the

F-100 Oscillating Airfoil Pressure Data,
 $k = 0.06$, $M = 0.84$

F-101 Oscillating Airfoil Pressure Data,
 $k = 0.10$, $M = 0.84$

F-102 Oscillating Airfoil Pressure Data,
 $k = 0.20$, $M = 0.84$

F-103 Oscillating Airfoil Pressure Data,
 $k = 0.28$, $M = 0.84$

F-104 Oscillating Airfoil Pressure Data,
 $k = 0.34$, $M = 0.84$

F-105 Oscillating Airfoil Pressure Data,
 $k = 0.40$, $M = 0.84$

quasi-steady results in F-41 which showed peak amplitudes nearly equal to the data obtained on the 15.2 centimeter model. The phase lag data again exhibited nearly constant values in the regions upstream of the shock for low frequencies. The models experienced nearly a 180° phase lag up to the 60 percent chord location at low reduced frequency ($k = 0.07$) with a strong jump and rapid change across the shock location and in the subsonic flow region downstream. The correlation with the numerical results of LTRAN2 suffered from improper shock placement and overestimation of the peak amplitudes. The predicted unsteady pressure amplitudes were about twice the measured magnitudes for positions 15 percent of the chord length further aft. These results were consistent with the comparisons made for the LTRAN2 steady-state values and steady pressure coefficients measured in the wind tunnel experiments (i.e. disagreements centered on the improper predictions for the strength and location of the oscillating shock wave). At the higher frequencies, the phase lags shifted to more negative values and the peak amplitudes decreased slightly at the increased frequencies. The piston effect evident at low frequencies in the locally supersonic regions were distorted at higher frequencies. The phase lag variations spanned a range from -180° to -75° at low frequencies while the higher pitch frequencies saw variations of -235° to -80° (or a range of about 150°). The results at Mach 0.84 showed the largest unsteady pressure amplitudes and shifts in phase lag due to model oscillation.

At higher Mach numbers the mean shock position shifted toward the trailing edge of the airfoil. At Mach 0.89, the shock position indicated by the steady-state pressure coefficients was between the 70

and 80 percent chord locations. The natural unsteadiness of the shock/boundary layer interaction generated a moderately high level of random pressure fluctuations at the 80 and 90 percent chord locations. The quasi-steady results from F-42 gave no significant indication of a strong shock influence. The oscillating airfoil results in F-106 through F-111 also showed little influence of periodic shock motion. The unsteady magnitudes showed nearly the same levels for each of the frequencies tested. The two models generated equivalent unsteady pressure amplitudes which had the highest levels at the 10 percent chord and the 70 percent chord. The phase lags were almost constant near the -180° value for most of the chord locations. Large jumps in the phase angle were seen at the 70 percent chord for both models (on the order of 150° change at low frequencies down to 25° at $k = 0.38$). The phase lags returned to near the -180° values at the 80 percent location only to be followed by a second set of jumps in phase lag of smaller magnitude than the first. The shock influences in these cases were either the result of a double shock system or attributed to the acoustic feedback hypothesis established during the above discussions of the scaled data, PSD's, and correlation plots. The LTRAN2 results gave reasonable comparisons for unsteady pressure amplitudes and for phase lag data. However, the numerical results placed the shock at the trailing edge of the airfoil, hence, the local shock influence observed at the 70 percent and 90 percent chord positions were not substantiated by the numerical predictions. The majority of the airfoil pressure sensors measured phenomena consistent with the "piston theory" behavior suggested by Landahl for the near sonic flow conditions.

F-106 Oscillating Airfoil Pressure Data,
 $k = 0.05$, $M = 0.89$

F-107 Oscillating Airfoil Pressure Data,
 $k = 0.09$, $M = 0.89$

F-108 Oscillating Airfoil Pressure Data,
 $k = 0.19$, $M = 0.89$

F-109 Oscillating Airfoil Pressure Data,
 $k = 0.27$, $M = 0.89$

F-110 Oscillating Airfoil Pressure Data,
 $k = 0.32$, $M = 0.89$

F-111 Oscillating Airfoil Pressure Data,
 $k = 0.38$, $M = 0.89$

The unsteady pressure magnitudes and phase lags measured during the oscillating airfoil testing at OSU/AARL were compared with the computational results obtained from the LTRAN2, UTRANS, and TWODI oscillating airfoil computer codes, and with experimental data from the NASA Ames 11-foot Transonic Wind Tunnel. The low frequency results were very well correlated with the quasi-steady amplitudes. At subcritical Mach numbers, the pressure data showed monotonic decreases in pressure amplitude along the airfoil chord as well as monotonic variations in the phase lags. At the low supercritical Mach number, the pressure amplitudes did not decline as rapidly over the forward portions of the airfoil model, and local maxima appeared near the mean shock location. At Mach 0.84, the local shock influence became the dominant factor in the pressure amplitudes with unsteady pressure coefficients more than three times greater than seen elsewhere. The data upstream of the shock waves indicated that the phase lags were of almost constant value near -180° . Large phase shifts were observed near the shock regions. At the higher Mach number (0.89) the shock influence was not as great and a trailing-edge acoustic feedback was apparently causing a doubled frequency influence near the 70 and 90 percent chord locations.

The unsteady pressure data obtained on the 7.6 cm and 15.2 cm oscillating airfoil models were analyzed with a complex-demodulation data reduction routine. The comparisons with theoretical results at the subcritical conditions were acceptable, however, the supercritical predictions suffered from incorrect shock placement and shock strength values obtained in the steady-state results. The subsequent unsteady

calculations gave larger unsteady amplitudes at locations further aft than observed experimentally. At low Mach numbers, the phase lags measured agreed well with the predictions from the TWODI integral method formulation. At higher Mach numbers, the character of the phase lag variations predicted by the LTRAN2 code was in qualitative agreement with the experimental data, yet the disagreements in shock position and strength prevented good quantitative agreement. There were no distinct differences in the data from the two models attributable to tunnel interference effects.

Comparisons of the data from the two airfoil models for tests at the same reduced frequencies and at the same physical frequencies show that the correlations are best for models oscillated at equivalent reduced frequency conditions. This leads to the observation that even though there is evidence of some harmonic coupling with tunnel acoustic modes, the fundamental unsteady response to airfoil motion is generally unaffected for the conditions tested during these OSU/AARL experiments. The unsteady pressure amplitudes and phase lags were in excellent agreement between the two models for subcritical conditions and regions of the airfoil ahead of the oscillating shock for supercritical flows. At Mach 0.80, agreement was good in comparison with data from the NASA Ames 11-foot tunnel except for the relatively narrow zone of influence of the oscillating shock. This discrepancy is the result of slight differences in the airfoil profiles and of Reynolds number differences (12 million for the NASA tests versus 5.2 and 2.6 million respectively for the OSU 15.2 cm and 7.6 cm models). Also, inadvertent misplacement of the unsteady sensors in the OSU

models appear to have missed the narrow zone influenced directly by the shock motion. The hypothesis of frequency doubling by a trailing edge acoustic feedback cannot be substantiated by these single frequency demodulation results. The important results are that even for pressure aft of shock zones during supercritical testing, the pressure amplitudes are small (indicating greater possible phase errors) and these signals contribute little to the unsteady airfoil lift and moment values. It is apparent that the unsteady pressure loads can have highly concentrated influence over relatively small portions of the airfoil surface, and hence, a very detailed array of instrumentation should be concentrated near these shock regions.

5.5 UNSTEADY PRESSURE DATA FROM THE WIND TUNNEL WALLS

The unsteady pressures measured on the test section walls and in the upper plenum were also analyzed by the complex demodulation technique to determine the magnitude and phase of the signal relative to the airfoil motion. There was no strong indication of a tunnel resonance behavior or a large interference effect. The effect of acoustic impedance of the ventilated wall was determined from this data by examining the attenuation and phase differences between the upper wall and plenum signals.

The subcritical conditions at Mach 0.35 produced data which showed that the unsteady pressures generated by the 7.6 centimeter model had levels at the upper wall which were approximately half as large in magnitude as for the larger model. The small model exhibited non-systematic differences in the values of attenuation obtained for the unsteady pressures between tunnel and plenum locations. The low

overall pressure values, combined with the relatively high ambient noise levels observed at the low Mach number apparently gave erroneous indications of greater magnitudes inside the plenum and large phase lags (see F-112). For the larger (15.2 cm) model, the amplitudes were about twice those of the 7.6 cm model and of nearly constant magnitude. The attenuation between the test section and plenum varied from 20 to 35 percent for the large model. Only 3 to 10 percent reduction was observed between the static probe and upper tunnel wall transducers. The phase lag variation over the range of the six frequencies investigated can be mostly accounted for by the change in wavelength. The phase lags for the large model were -14° to $+3^\circ$ for the probe versus plenum signal difference, indicating that a small reactive component caused the lag to be more negative than predicted by the ratio of separation length to acoustic wavelength. However, there was not a strong frequency dependence, indicating that the wall impedance was primarily resistive in nature at these conditions.

A slight increase in the signal amplitudes, combined with reduced noise levels, led to more consistent results for both models at Mach 0.52. As above, the larger model generated approximately twice the unsteady pressure coefficient values observed for the small model (see F-113). The 7.6 cm chord model exhibited between 18 to 43 percent attenuation in the oscillating pressure signals at the upper wall while the 15.2 cm chord model showed a nearly identical 17 to 41 percent acoustic reduction. The general trend was for slight increases in the wall pressure amplitudes above reduced frequencies of about $k = 0.15$. However, the differences between plenum and test section probe and wall

F-112 Oscillating Pressure Data
on Tunnel Walls, $M = 0.35$

F-113 Oscillating Pressure Data
on Tunnel Walls, $M = 0.52$

F-114 Oscillating Pressure Data
on Tunnel Walls, $M = 0.70$

pressures decreased for the increasing reduced frequency. The variations for the two models indicated phase lag values increased as frequency increased, again related to the change in effective wavelength of the pressure signal. The phase shifts at the upper perforated wall were again observed to have trends related to the frequency variations noted above and indications of some reactive impedance behavior. The plenum demonstrated a reactive impedance behavior since it tended to slightly lead the test section pressures at low frequencies and slightly lagged them at the higher model oscillation frequencies. This result suggested that the flow resistance at the perforated wall tended to dominate the other impedance sources at the higher frequencies.

The upper subcritical condition (Mach 0.70) demonstrated a more pronounced frequency dependence for both amplitude and phase data obtained near the perforated wall. The 15.2 cm model generated unsteady pressure amplitudes which were about twice those produced by the 7.6 model. The unsteady amplitudes as seen in F-114 were greater than observed at the lower Mach numbers and they increased in a manner proportional to frequency. The wall attenuation effect reduced the amplitudes by 35 to 65 percent between the test section and plenum chamber for both models. The signal amplitude for the small model had increased due to greater dynamic pressures at Mach 0.70, and this combined with low ambient noise levels gave improved results for the 7.6 cm model data. The phase lag information indicated that the walls had a more reactive behavior at greater Mach number along with an increased variation in phase angle versus frequency. The increase in

phase lag proportional to frequency was clearly seen. At low frequencies, the probe and upper wall locations measured phase lags of -165° to -175° with the plenum experiencing a -145° lag. The relative phase angle in the plenum led the test section response, indicating a capacitance-like response for the plenum. The final subcritical tunnel condition had no apparent resonance but showed increased reactive impedance to the pressures generated by the oscillating airfoil. Phase lag comparisons of the 7.6 cm model and the 5.2 cm model data were improved if correlations were done on the basis of physical frequency rather than reduced frequency. This result supports the observation that the upper wall phase lags are those naturally occurring due to the relation of acoustic wavelength to the model-to-wall separation distance.

At the first supercritical Mach number ($M = 0.80$), the unsteady amplitudes increased by about 30 percent for the upper test section data obtained with both models (see F-115). The 15.2 cm model generated static probe and upper wall pressures which were between 1.3 and 2.4 times greater than for the 7.6 cm model. The amplitude ratios were greatest at low frequency. The unsteady amplitudes increased with frequency but did not tend towards any apparent tunnel resonance peak. Attenuation of the signals at the upper wall varied between 50 and 70 percent with slightly greater reduction in unsteady pressure amplitude observed for the large model. Phase lag data demonstrated large shifts for the 15.2 cm chord model over the frequency range examined. The increasing negative phase shift which was exhibited suggested that the reactive influence of the plenum grew proportional to frequency. The

F-115 Oscillating Pressure Data
on Tunnel Walls, $M = 0.80$

F-116 Oscillating Pressure Data
on Tunnel Walls, $M = 0.84$

F-117 Oscillating Pressure Data
on Tunnel Walls, $M = 0.89$

phase difference between static probe and plenum data varied from -23.5° at low frequency to $+5.8^\circ$ at the higher frequencies.

The unsteady pressure amplitudes at the upper wall peaked near Mach 0.84 (as observed for the model surface pressures). The data in F-116 indicated that the larger model generated as much as 2.5 times the unsteady signal amplitudes observed in data from the small airfoil model. Test section-to-plenum attenuation also peaked at 72 percent for the large model at low frequency. The attenuation values for the upper wall transducer were nominally 5 percent less than those seen for the static probe results. The unsteady coefficient amplitudes were two to three times greater at Mach 0.84 than for Mach 0.35. The improved signal-to-noise ratio was enhanced by the increased dynamic pressure and reduced tunnel ambient noise levels. Phase lag variation with frequency became slightly more pronounced with the ranges of -25.6° to -4.5° observed between the static probe and plenum and -18.6° to $+7.3^\circ$ between the upper wall and plenum. These amplitude and phase lag data were still not indicative of any natural resonance behavior at the tunnel walls.

The final set of unsteady wall pressure data obtained at Mach 0.89 showed a decrease in the pressure oscillation amplitudes similar to than seen on the model surfaces (see F-117). Since the strength of the airfoil pressure source was reduced due to the decreased shock influence, the response near the upper wall was attenuated 20 to 30 percent for the smaller model and 35 to 50 percent for the larger model. The wall attenuation at Mach 0.89 ranged between 40 and 60 percent with about a 5 percent difference observed between the static

probe and upper wall at low frequencies. Phase lags similar to those at Mach 0.84 were observed with primarily negative phase lags between the tunnel locations and plenum. The increasingly negative phase lags were observed at all three locations near the upper wall.

In general, the unsteady pressure measurements near the perforated wall gave no indications of tunnel resonance. The amplitudes increased with Mach number up to Mach 0.84 and then declined at Mach 0.89 as was observed in the airfoil pressure data. The 7.6 cm model generated pressure levels roughly half as great as seen with the larger 15.2 cm model. The phase lag data for both models showed an increasingly negative slope versus frequency as the flow velocity approached sonic speeds. Data from both models were much lower than most of the model surface pressure levels.

These results indicate that there are no strong tunnel resonance modes excited directly by the fundamental oscillatory model motion. The possibility of coupling between higher harmonics of the pressure response function and tunnel acoustic modes is not supported by this or previously discussed data. The lack of significant harmonic structure in the spectral plots of the data near the upper wall as well as the absence of otherwise unaccountable phase shifts for higher model oscillation frequencies discount such tunnel resonance hypothesis. Also, the only regions of the airfoil showing strong harmonic content relative to the power in the fundamental mode are those near the trailing edge at the higher supercritical Mach numbers. Such behavior can be explained by noting that for those cases, the power in the fundamental mode has been significantly reduced in comparison with

regions ahead of the shock wave or in subcritical cases with the forward portions of the airfoil surface. Hence, the extent of any tunnel resonance influence on unsteady lift and moment data is negligible.

VI. CONCLUSIONS AND RECOMMENDATIONS

Transonic testing of NACA 64A010 airfoil models has been conducted at the Ohio State University Aeronautical and Astronautical Research Laboratory (OSU/AARL). This dissertation reports on the experimental results of both fixed angle-of-attack and mid-chord pitch oscillation testing conducted in the 6" x 22" transonic airfoil tunnel. Comparisons with both steady-state and unsteady airfoil predictions were made along with other experimental data. The validity of the experimental setup was examined in light of unsteady interference effects at transonic speeds.

The transonic flutter dip, as noted in Chapter 1, remains as a serious performance limitation for high-speed aircraft operating in the supercritical transonic regime. The large shock wave/boundary layer interaction, coupled with a trailing-edge feedback mechanism, can develop large unsteady lift and pitching moment oscillations. Coupled with the natural bending and torsional modes of a wing structure, the unsteady loads drive the flutter motion during flight. In this work the forcing function due to oscillatory pitch motion was addressed.

The historical background for oscillating airfoil analyses ranges from the linearized methods of Theodorsen (1935) to the present Navier-Stokes solutions of Steger, et al. (1979). Most notable among the recent analysis techniques are the small-disturbance, harmonic

potential codes (STRANS/UTRANS) developed by Traci, et al. (1975 and 1976) and the time-dependent, transonic, small-disturbance solution by implicit techniques (LTRAN2 code) originally developed by Ballhaus et al. (1975 through 1978) and extended by Rizzetta (1979 and 1980).

These tools were used to predict unsteady pressure on the NACA 64A010 airfoil for correlation with current data. Previous experimental work includes the early work by NACA personnel and by the British at RAE in England. More recently at NLR and in particular at NASA-Ames, special test equipment and data systems were constructed to measure oscillating pressure coefficients.

The development of ventilated wind tunnels for transonic testing helped decrease the steady-state interference problems near Mach one. Also, the porous walls were found to have a beneficial effect on dynamic model testing. However, the ambient noise levels in some wind tunnels have been observed to peak near the critical transonic flow condition ($M = 0.8$). During noise surveys made in the OSU/AARL 6" x 22" transonic airfoil tunnel, it was found that at Mach numbers above $M = 0.50$, the ambient noise levels were near the 0.5 percent level recommended for unsteady testing. Although edge tones at the perforated walls were observed, their narrow bandwidth peaks contained relatively low energy, and hence, the mid-subsonic speed range did not exhibit a large unsteady pressure rise. The overall signal-to-noise ratio for the oscillating airfoil experiments was concentrated in narrowband peaks rather than more-or-less evenly distributed as random noise, the signal-to-noise ratios for the single frequency data was much greater (on the order of 50 to 80 dB). The tunnel wall resonances

were observed to be at significantly higher frequencies than the model oscillation frequencies tested.

The in-situ unsteady pressure measurements were examined for potential data uncertainty. Calibration errors were seen to be within a ± 2 percent band. The transducer installation geometry was analyzed as a Helmholtz resonator. The predicted natural resonance frequencies were well above the airfoil motion frequencies and a flat response over the 0 to 200 Hz range was predicted. The digital data acquisition and data reduction errors (for the complex demodulation processing) were also predicted to be within a ± 2 percent band. The overall uncertainty was observed to be a ± 5 percent for the oscillating airfoil pressure magnitudes.

The full-potential and small-disturbance potential formulations are appropriate for modeling the unsteady transonic problem along with an appropriate set of boundary conditions for the ventilated walls. Following Mabey's (1978) recommendations, two potential flowfields were established in the test section and in the plenum chambers. At the ventilated wall interface the compatibility conditions were based on continuity and a wall pressure differential derived from the orifice impedance work of Hersh (1976 and 1979). Previous experimental work showed that the steady-state interference is quite low in the 6" x 22" tunnel. Both the solid-wall (Acum-1966) and perforated-wall (Mabey-1980) predictions of tunnel resonance show that for the model frequencies tested, transverse tunnel acoustic modes do not lead to resonant conditions for test conditions encountered in the experiments reported here.

The steady-state NACA 64A010 airfoil data was well-correlated with the small-perturbation predictions of the STRANS and LTRAN2 codes at subcritical Mach numbers. For super-critical conditions, shock strength and position are inadequately predicted by the small perturbation methods, however, excellent correlation was achieved with the inverse holograph predictions of Garabedian and Korn and with experimental data obtained in other facilities. No significant differences were observed between the two model sizes tested. Unsteady pressures measured for fixed, pitch angle tests showed the influence of localized shock wave disturbances. Strong periodic shock motion was not observed. Quasi-steady pressure amplitudes deduced from $\pm 1^\circ$ pitch positions served as the zero reduced frequency cases.

The unsteady experimental data from the oscillating airfoil tests were prepared in several forms including scaled data plots, Fourier-analyzed spectral data, and demodulated (single harmonic) frequency data. The latter type was decomposed into real and imaginary components representing pressure signals in-phase and 90° out-of-phase with the model motion.

An example of the first two data types is presented in F-118 which illustrates various chord-wise pressure signals and spectral content associated with some of the signals. The dominance of the single fundamental frequency is observed to decrease as the flow approaches the nonlinear region near the oscillating shock wave. However, the relative strength of the fundamental frequency decreases for the flow downstream of the shock, and the overall amplitudes are greatly reduced.

F-118 Typical Scaled and Spectral Data

The amplitude and phase components of the experimental data are compared to predictions of three codes: (1) STRANS/UTRANS--a small-perturbation harmonic potential solution obtained through relaxation techniques, (2) LTRAN2--a time-dependent, small-perturbation scheme using implicit approximate-factorization methods, and (3) TWODI--an integral method using Bland's transform and a set of orthogonal airfoil and pressure polynomials. The scaled pressure plots exemplified the sinusoidal nature of data obtained at subcritical Mach numbers and in regions significantly ahead of or behind the shock interaction zones. The strong harmonic content of the data obtained within the shock movement region appears as periodic yet non-sinusoidal pressure traces. The power-spectral-density plots show the dominance of the fundamental mode over both the harmonic and broadband random noise levels. Autocorrelation processing shows the strong periodic content of the majority of the pressure signals.

The complex demodulation data showed good correlations with subcritical airfoil predictions from the three computational methods. The unsteady pressure amplitudes for the two model sizes are in excellent agreement. At supercritical conditions the TWODI integral method fails to predict the shock condition and hence is not a valid comparison. Good agreement was obtained with the UTRANS and LTRAN2 data as well as in comparison with data obtained at NASA-Ames except in the relatively narrow region of shock influence. At the $M = 0.89$ condition, the major disagreement between the experimental and computational results are attributed to the incorrect placement of the shock wave as in steady-state calculations.

Measurements of the unsteady pressures on the sidewalls and in the neighborhood of the upper ventilated wall showed that acoustic reflections were non-resonant, and the plenum chambers exhibited an attenuation of the pressure waves. The phase lag noted between the model motion and wall pressure responses were consistent with the propagation times required for transmitting pressure waves through the flow. The smaller model generated unsteady pressure levels near the upper wall roughly half the amplitude of the large model. This experimental evidence supports the conclusion that the unsteady interference effects are small for these tests.

The study of unsteady interference effects in the OSU/AARL 6" x 22" transonic airfoil tunnel was conducted with two NACA 64A010 airfoil models. The data was examined in terms of harmonic content and the in-phase and out-of-phase components. In comparison with three predictive techniques, it was seen that the experimental data had good correlation with all three at subcritical flow conditions. However, the integral method failed to predict supercritical shock influence while disagreements with the finite-difference codes were the result of incorrect shock placement and strength. The measured unsteady interference effects were small and did not indicate any tunnel resonance phenomena. The relation of the predicted tunnel resonance frequencies to the higher harmonic components of tests conducted at the highest model oscillation frequency along with the increased powers observed for the higher harmonics in certain cases suggested a hypothesized coupling with tunnel acoustics at frequencies greater than the fundamental model motion. However, that premise was discounted

since there was no evidence of resonance near the upper tunnel wall. Also, the pressure amplitudes in regions downstream of the shock are seen to be relatively low in magnitude, and therefore, do not contribute significantly to the overall lift and moment loads. The close correlation of a majority of the unsteady pressure data from the OSU/AARL experiments at subcritical Mach numbers with predicted results indicate that the unsteady testing was interference free for the conditions examined. At the higher Mach numbers, there are disparities in the strength and position of the shock influence. The predictive methods tend to misplace the main shock wave and over-estimate the shock strength. The lack of good correlation between aft portions of the two models at supercritical conditions is relatively inconsequential since the unsteady pressure magnitudes are small. The major cause of discrepancy is likely due to Reynolds number or slight profile differences between the two models. The hypothesized frequency doubling associated with a trailing-edge feedback mechanism can adequately explain the strong harmonic content for those conditions and locations. Finally, it is concluded that the unsteady pressure coefficient data obtained in these studies exhibits no evidence of tunnel interference behavior for the frequencies of the model oscillatory motions. The lack of tunnel resonance has been independently confirmed by the data from instrumentation located near the upper plenum. The upper wall pressures show nearly twice as high values for the larger (15.2 cm) model as compared to the small (7.6 cm) model. However, the levels remain lower than corresponding model data and do not peak as a function of frequency for either model.

Development of future unsteady transonic test facilities should consider specific engineering design features which would reduce acoustic reflections within the test section yet avoid the problems associated with free-jet testing in an open-jet anechoic chamber. The use of a sintered mesh or packed wire screen ventilated wall as suggested by Bauer (1976) could provide a flow boundary that is essentially transparent to the acoustic excitations. A reactance muffler packing installed in the plenum chambers can be designed to attenuate the low frequency range effectively. Also, elimination of the slots and perforations can prevent edge tone resonance in the upper subsonic speed range. Further efforts to reduce stilling chamber noise and upstream valve noise can significantly lower the ambient noise levels at the lower Mach numbers. Such modifications should be considered for existing test facilities as well as in construction of new wind tunnels.

APPENDIX A

UNSTEADY DATA ACQUISITION ROUTINES AND PROCEDURES

The data acquisition and post-run analysis routines used for the unsteady aerodynamics testing at OSU/AARL were a combination of FORTRAN programs and Macroassembler routines coded specifically for the wind tunnel setups and in-house Harris computer system. The programs relied extensively on the Foreground processing capabilities of the Harris Disc Monitoring System (DMS) as modified and implemented at AARL. The Purdue extensions to FORTRAN were used extensively to control the digital/analog interfaces for the digitizers and relay controls. The interrupt-processing features of the DMS system provided direct operator control over the data acquisition and data reduction events at the graphic CRT display console.

The data flow schematic (shown in F-119) indicates the various peripheral functions for data storage and handling afforded the on-line operator. The data manipulation is centered on the high-speed, random-access disc files. The operating system is resident in the main core memory with the various support routines and master copies kept on the disc. The user software, temporary and permanent data storage files, work areas for data sorting and buffers, and input/output spool files are maintained on disc. The real-time data acquisition routines control the run sequence and digitize both low-speed and high-speed

F-119 Unsteady Aero Program Data Flow

data through analog-to-digital (A/D) converters, digital-to-analog (D/A) outputs, and digital relay switches whose operation is controlled through a universal COMMON interface. The calibration and setup routines similarly access the control functions for tunnel operation and data sampling. The high-speed digitizing was accomplished with a direct-memory-access (DMA) input/output channels which would allow direct buffering of the data from the A/D converter into core memory and then buffering out to a temporary storage disc file. This allowed several sets of unsteady data to be acquired during each run without being limited by core memory size. The Fast-Fourier-Transform (FFT) algorithm and data reduction routines operated from work areas created on the disc files containing the raw data run file (including calibration constants and test parameters), the scaled data files, and processed data files for output in printed or plotted form. These data analysis routines operated either on-line by accessing current run data files or off-line by generating the appropriate disc data files by retrieving the archived tape data. The digital magnetic tape files were used to provide a history tape of each data run. Copies of the software routines, run parameter files, and operating system were saved on magnetic tape as backup to the disc files. The operator's console allowed keyboard inputs for data manipulation, software programming, and sequencing of the user programs and a CRT display for both alphanumeric and graphic (vector) outputs. This console was used both on-line to display operator messages, to receive inputs, and to display run conditions and preliminary data, and off-line to access the FFT and data processing options and to display processed data plots prior to

hard-copy output. The printed data tabulations and program code listings were output through spool file arrangements which allowed the formatted data to be recorded on temporary disc files for later output. Similarly, coded data plot instructions were stored in a spool file for output upon operator command. Card inputs were read into an input spool file and held in an input queue for processing according to CPU and memory availability and priorities. The highly-interactive computer system at AARL has been programmed to provide quick, efficient data handling and ease of access to multiple users and real-time data acquisition operators.

The on-line and off-line computer programs used to acquire and process the unsteady test data are nearly all foreground routines initiated and controlled from an interactive display terminal. The main sequence of programs used during tunnel operation and data acquisition are shown in F-120 along with off-line data reduction and calibration procedures. Detailed descriptions of each routine are given below.

The initializer routine, UA/INN, allows proper positioning of the SORTIL interrupt processor at the top of the core memory map. It also preinitializes the 100 word universal COMMON block, SYSCOM, which remains resident for the duration of the test run sequence. SYSCOM contains the digital relay bit pattern, the spool file position counters, and other run parameters. UA/INN initiates loading of the file manager, UA/MAN, and then it exits.

UA/MAN carries out most of the overhead tasks of setting up files and initializing parameters. The SORTIL system routine is loaded and

F-120 Unsteady Aero Programs

initialized with the proper interrupt link to allow operator control over loading and execution of the primary run routines. Links to the UA/CAL, UA/ZRO, UA/RUN, UA/ANA, and UA/RED routines are established and execution of any one module is selected from the interrupt panel located on the operator's console. The UA/MAN routine then sequentially displays the pre-run setup options for operator selection and the instrumentation parameters and current calibration constants for operator review and update. The UA\$DAT file is a resident disc file containing the run information, previous constants, choke bar configuration data, alphanumeric descriptors and comments, work file sizes, Scanivalve parameters, unsteady data sampling parameters, and run timing information. After the parameter update is completed, the spool files and work areas for printed output, hard-copy plots, data buffers, and sorted unsteady data files are created and initialized. The UA/SWT routine is initialized as UA/MAN exits the foreground region.

The UA/SWT routine clears and resets all interrupt levels and displays a list of the interrupt switch assignments on the operator's CRT. Before exiting, the routine connects and enables interrupt control for SORTIL. The foreground processing is then halted to allow background activity and spool operations until the operator resumes the data acquisition processing through an interrupt command.

The now resident SORTIL routine is put in a wait mode following exit of the UA/SWT program. The SORTIL code directs processing of the multi-level interrupt logic, which for this application, includes the five first-level access routines and a second-level access to interrupt other foreground activity and initiate the UA/DIG routine.

The on-line calibration sequences are controlled by UA/CAL. The current instrumentation signals are sampled, scaled, and displayed for review. The steady-state transducer assignments, previous calibration zeros and slopes, parameter names, excitation voltages, and amplifier gain and filter information are displayed for instrumentation setup. Upon command, a resistance calibration sequence is performed and new calibration values are computed and displayed for operator approval. If the new values are acceptable, the UA\$DAT file is updated, otherwise the calibration sequence may be repeated. The thermocouples, unsteady pressure sensors, and position slidewire instrumentation are likewise calibrated upon demand. Both pre-run and post-run calibrations can be performed according to operator desires. When the on-line electrical calibrations are completed, the UA/SWT routine is initialized as UA/CAL exits.

The pre-run setup and zeroing routine is titled UA/ZRO. It samples and displays the current run information and instrumentation outputs. The Scanivalve and wake survey probes are properly homed and positioned. The cutoff valves are reset as necessary. Pre-run values of ambient conditions, transducer values, run parameters, and choke bar configurations are displayed, then the operator is given three options for continued processing. A direct link to the UA/RUN data acquisition routine can be initiated by the operator, or the foreground processing put in a wait mode by re-entering the UA/SWT routine. The third option is to terminate the data acquisition sequence through closing and deallocating the spool files and work areas, disabling the SORTIL

interrupt logic, clearing the SYSCOM memory, and exiting all foreground programs associated with the unsteady aero data acquisition.

The UA/RUN routine contains the necessary instructions to control and monitor tunnel operation and sequence the various data taking functions. After releasing the first-level interrupt, run parameters and transducer calibration values are read from the UA\$DAT run file. The UA/DIG digitizer control routine is loaded into the foreground memory and the SYSCOM area is opened for access by both the UA/DIG and UA/RUN routines. The anticipated stagnation pressure and Mach number ranges are displayed and updating is permitted. The tunnel operation mode (manual- or computer-initiated) is selected. For the computer-assisted mode, the Morgan-Smith (MS) value interlock is armed by the operator and confirmed by the computer. Next, axes are drawn for the real-time traces of stagnation pressure and Mach number on the operator's keyboard; the main shut-off valve is opened and real-time traces of P_0 and M are tracked on the CRT until the second-level interrupt assigned to the UA/DIG execution command is activated by the operator. The UA/DIG routine causes the Scanivalve cutoffs to be stepped to a closed position and sets a data flag in SYSCOM before returning control to UA/RUN. With the flag set, the automatic data acquisition sequence is started.

It consists of: (1) reading current tunnel conditions and plotting P_0 and M on the real-time CRT graph; (2) calling the DIGITD subroutine to activate the DATEL system for digitizing accelerometer and pressure transducer signals; (3) stepping the pressure transducer relays for the next set of data; (4) buffering out the unsteady data to

a temporary disc file; (5) recording the time of day; and (6) querying the buffer out status to ensure completion of the data buffer cycle. The six steps above are repeated over the appropriate number of relay cycles. Following unsteady data acquisition, the wake probe is scanned according to preselected parameters (if desired). Following acquisition of the final data point, the Morgan-Smith valve is instructed to close and the shutdown transient is digitized and displayed on the CRT. Once the P_0 value falls below a specified threshold value, the real-time plot is discontinued and the operator is instructed to disarm this MS valve interlock prior to continue processing. When the computer detects disabling of the MS interlock, the operator selects the option to continue the data processing or link to UA/SWT for a repeat run. Continued processing would first lead to buffering out the readings from the real-time run history and any wake survey data. The Scanivalve is then stepped through the required number of ports and that data, along with other flags, status codes, and run comments, are written on the blocked run file. Control and processing are then transferred to the UA/SRT routine as UA/RUN and UA/DIG are terminated and cleared from memory.

UA/SRT retrieves the buffered data from the different relay positions used during the tests and merges the data into a single sorted work file of raw data. Preliminary results of maxima, minima, mean value, and standard deviation are computed and displayed on the CRT. This preliminary data is written to a print spool file.

Control then transfers directly to UA/SAV, where the raw data is transferred to magnetic tape. The calibration constants, run

parameters, and instrumentation setup information is retrieved from the UASDAT run file, along with the Scanivalve results. These data are written to tape in the first block of run information and steady-state data. The second set of data are buffered in from the raw, sorted unsteady data file, and written to a formatted tape. Any wake survey data is similarly buffered in and copied to magnetic tape. The history file on tape serves as a data archive for each successful test run. If unsteady analysis is desired, then the UA/ANA routine is pre-initialized; otherwise, the UA/SWT routine is accessed to put the foreground processing in a wait mode.

The on-line analysis routine for quick-look power-spectral densities from the unsteady data is titled UA/ANA. It allows the operator to display the raw digital data on the CRT as well as to display Fourier-transformed results. Hard copy plots can be generated using the first 1024 datapoint from each channel selected. The unsteady data can be copied to tape at this point, if not previously done so. Options are available to branch to UA/ZRO for run termination, to UA/SWT for the standby wait mode, or to UA/RED for continued data reduction.

The steady-state data is reduced during the on-line run processor UA/RED. It can be initiated directly from the UA/ANA routine, or by the appropriate interrupt selections through SORTIL. Upon entry, the UA/RED routine accesses the UASDAT run file, the rake data file, the real-time run history buffer, and the appropriate sorted tap file. Using the run file and tap file, the Scanivalve data is scaled and processed to correct for the volume carryover between sequential ports,

and the non-dimensional pressure coefficients are computed. These C_p data are then integrated to obtain section lift, moment, and pressure drag coefficients for the airfoil. The results are displayed on the graphic CRT so the operator can evaluate the data prior to the next test run. Hard-copy plots and tabulated listings are formulated onto the output spool files according to operator desires. The wake survey data is retrieved, scaled, and displayed for the operator via cursor controls on the graphics terminal. The section drag coefficient is then computed using the Betz method, with corrections for static pressure in the wake. These results are displayed a second time with the option to prepare hard-copy plot instructions for the spool file. The run history trace is also prepared for hard-copy plotting upon operator instructions. Following completion of the steady-state data reduction tasks, control can be transferred to the UA/SWT for the wait mode or directly to UA/ZRO for run termination.

The on-line routines are interconnected through the interrupt processor SORTIL and calls to the foreground processor subroutines, FROGS, for chaining the execution of the programs according to operator desires. The off-line routines for the unsteady aerodynamic studies can also be operated in the foreground mode; however, there are no direct links between any of these off-line routines.

The UA/ADJ and UA/KCL programs are utility routines used to construct the tap files and to calibrate the unsteady pressure transducers, respectively. The Scanivalve tap assignments are specified for each model, including the non-dimensional tap location and usage flags. The flags determine whether the data is to be used in

force and moment integration calculations. Both unsorted and sorted output files are prepared. The sorted file contains the sequence appropriate to the steady-state data reduction algorithm. The unsteady pressure transducers are manifolded together on the reference pressure side of the diaphragm, and therefore, the complete set of sensors can be calibrated simultaneously by applying pressure increments to the reference manifold. The UA/KCL program is set up to perform the relay switching functions and to sample the data levels as the pressure levels are set and recorded under operator control. The calibration data are then fit by a least-squares linear fit routine and the sensitivities are computed. The resistance calibrations are then performed to obtain the necessary constants for input to the on-line calibration routine. The individual calibration data sets and curve fits are plotted on the CRT and hard-copy, along with a tabulated listing of the results. A permanent disc file is maintained with the appropriate instrumentation parameters and constants for direct update of the current on-line run files, if desired.

The off-line time series analysis routines are programmed to give a comprehensive capability for Fourier and statistical analyses of the unsteady data obtained during testing. The UA/TSA program was operated in the foreground mode to permit operator interaction during the data reduction process. The routine could process data directly from its resident work file, UA/DAT, or by reloading a selected run file from the magnetic tape archives. The user selects the run number, processing and plotting flags, sampling and bandwidth averaging parameters, and data channel numbers (two at a time) prior to start of

the main processing. The list of options is then reviewed for correctness before continuing. Selected raw data channels are retrieved from the UA/DAT file, along with the calibration constants. Scaled engineering data is then arranged in a large complex data array (one channel in the real coefficient positions and the other in the imaginary storage positions). This array is buffered to a temporary work file for recall at the beginning of each processing sequence after it has been edited for wild points and detrended (removal of any offset or linear slope) by pre-processing subroutines. If desired, the data can be processed by digital filtering techniques prior to continued processing. The processed data is then analyzed to determine the first four statistical moments (mean, standard deviation, skewness, and kurtosis), the non-dimensional unsteady pressure coefficients, and a probability density function in the form of an amplitude histogram. The Fourier analysis routines are then called to generate Power-Spectral-Density, Cross-Spectral-Density, Auto-Correlation, Cross-Correlation, Transfer Gain and Phase, and Coherence functions. The Cooley-Tukey Fast Fourier Transform (FFT) algorithm is used, along with the procedures described in Appendix B, for unsteady data analyses. A variety of data window functions are available to smooth the spectral data results along with bandwidth averaging for reducing the number of output points. These results are then displayed on the graphic CRT for operator review and subsequent preparation of formatted plot instructions and listings for the output spool files. The UA/TSA routine can then be looped back to various points for continued

processing of other run files, data channels, and/or data reduction options.

For single-frequency information corresponding to the model oscillation motion, the UA/SYN routine performs complex demodulation processing. The data is either retrieved from the UA/DAT data file or retrieved from magnetic tape as in UA/TSA. The accelerometer and pressure signals for each relay position are analyzed to determine complex amplitude and phase of each signal at the specified frequency and their phase differences between the respective pressure and accelerometer signals are computed. These amplitude and phase data are scaled according to pitch motion amplitude and dynamic pressure to produce non-dimensional unsteady pressure coefficients. The complex demodulation algorithm convolutes a sine and cosine function with the scaled data, followed by a low-pass digital filtering and averaging to determine the real and imaginary parts of the Complex Fourier coefficients (see Appendix B for details). The tabulated results are sorted and output as a hard-copy listing.

The UA/RRD routine is an off-line version of the steady-state data reduction routine UA/RED. It is identical coding except the data is retrieved from either the UA/DAT data file or the magnetic tape archives, as described above. The real-time run data is not available and therefore, plotting of the P_0 history is not permitted.

Descriptions of the computer routines used to control tunnel operations, to sequence the data acquisition procedures, and to provide a comprehensive data reduction capability have been given. The highly interactive, interrupt activated re-entrant foreground programs used

for on-line data acquisition and analyses facilitated efficient, rapid control of the test progress directly from the computer operator's console. The features available in the off-line data reduction routines for steady-state results, time series analyses, and frequency demodulation were outlined. These programs made extensive use of the foreground processing capabilities, the overlay link cataloger, and random-access disc files available on the OSU/AARL Harris computer system.

APPENDIX B

ANALYSIS TECHNIQUES FOR UNSTEADY AERODYNAMIC DATA

B.1 INTRODUCTION

This appendix discusses analysis of unsteady aerodynamic phenomenon using time series analysis techniques. The increased activity in measuring dynamic data in experimental fluid mechanics has been prompted by several recent advances in measuring technology, in digital computer capability and availability, and in the efficiency of algorithms such as the Fast Fourier Transform (FFT). The interest in unsteady analysis has been prompted by needs to formulate better turbulence closure models, to examine the unsteady aerodynamic inputs to flutter, to analyze aero-acoustic phenomenon, etc.

Historically, Fourier transform techniques gained prominence in designing second-order filter systems and as a mathematical tool for physical wave optics problems. Their use as experimental diagnostics tools was spurred by effort in communication theory for signal detection, both in examining radar and sonar information and in radio telemetry. Today, the Fourier-based time series analysis techniques find wide application in fields from acoustics to economics, and can be applied to any data series of recorded events.

Although techniques are available for interpreting data taken at unequal time steps and with non-stationary statistical properties, this

discussion will restrict its scope to data series with equal sampling intervals and stationary statistical properties. By way of definition, a random variable that is bounded and continuous (integrable in the sense of measure theory) is ergodic and stationary if its statistical moments (i.e., mean, variance, etc.) are independent of time. This assumption allows one to analyze signals over discrete time periods using distinct sampled values at equally-spaced intervals, and it assures that ensemble averages are equivalent to corresponding time averages.

The digital processing of continuous analog signals is founded on the Shannon Sampling Theorem. In the time domain, it may be stated as:

"A band-limited signal which has no spectral components above the frequency, f_{NY} Hertz (called the Nyquist frequency) is uniquely determined by its samples taken at uniform time intervals less than $(1/2 f_{NY})$ seconds apart".

And, the time sampled function is defined as:

$$x_s(t) = x(t)\delta_T(t) = \sum_{n=-\infty}^{\infty} x(nT)\delta(t-nT) \quad (B.1)$$

where $\delta(t)$ is the Dirac-delta function. A frequency-domain statement of the Shannon Sampling Theorem is:

"A time-limited signal which is zero for $t > T$ is uniquely determined by the samples of its frequency spectrum at uniform intervals less than $(1/2T)$ Hertz apart".

These theorem statements establish a foundation for digital sampling and analysis of unsteady data.

The Fourier transform converts time domain data to the frequency domain and vice-versa using the following relations:

$$X(f) = \int_{-\infty}^{\infty} x(t) \exp(-j2\pi ft) dt, \quad (j^2 = -1) \quad (B.2)$$

$$x(t) = \int_{-\infty}^{\infty} X(f) \exp(j2\pi ft) df \quad (B.3)$$

where $x(t)$ is a time-domain function and $X(f)$ represents the corresponding frequency domain function (t is time, in seconds; and f is frequency, in Hertz).

Since one deals with a limited number of data samples in actuality, the finite equivalent of the Fourier transforms are defined as:

$$\tilde{X}(f) = \int_{-T/2}^{T/2} \tilde{x}(t) \exp(-j2\pi ft) dt \quad (B.4)$$

which is equivalent to $X(f)$ for a time limited function:

$$x(t) = \begin{cases} x(t) & \text{for } (-T/2) \leq t \leq (T/2) \\ 0 & \text{for } |t| > (T/2) \end{cases} \quad (B.5)$$

In the discrete sense, the sampled function $x_s(t)$, is then inserted in the finite Fourier transform, resulting in:

$$X_s(f) = \Delta t \sum_{i=-\infty}^{\infty} x_s(i\Delta t) \exp(-j2\pi fi\Delta t) \quad (B.6)$$

where Δt represents the time between uniformly spaced samples. For digital computation, one uses a combined discrete/finite form of the Fourier transform:

$$X(k) = \Delta t \sum_{i=0}^{N-1} x_B(i\Delta t) \exp \frac{-j2\pi i k}{N} \quad (B.7)$$

where k is the array index and N is the total number of samples. It is this final form which one implements with the Fast Fourier Transform (FFT) algorithms.

The rediscovery of the FFT by Cooley and Tukey (1965) produced a significant improvement in the efficiency and accuracy of Fourier Transform computations. Direct computation of Eqn. (B.7) would involve N^2 multiplication operations whereas, the "decimation in time" approach used in the Cooley-Tukey algorithm reduces the number of operations to $N \log_2 N$ for $N = 2^P$ (see Table 2 for comparisons of direct and FFT methods).

TABLE 2
COMPUTATION COMPARISON OF DIRECT vs FFT ALGORITHMS

Number of Samples, N	Speed Ratio $\frac{N^2}{N \log_2 N}$
4	2
16	4
64	10.7
256	32
1024	102.4
4096	341.3
16384	1170.3
65536	4096

By observing that Eqn. (B.7) can be rewritten as:

$$X(k) = \sum_{i=0}^{N-1} x(i) W_N^{ik}; \quad (k = 0, 1, \dots, N-1) \quad (B.8)$$

where $W_N = e^{2\pi j/N}$ is recognized as the N th principle root of unity in the complex plane. The symmetries observed in computing the W_N^{ik} terms aids in pairing terms for computation and as observed in F-121 (a diagram of the FFT procedure), all of the operations can be done within the memory space for the original array. Also, it is evident that each input element is reflected in all of the output elements as would be expected in the direct Fourier transform implementation. The use of bit-reversed address sorting either before or after the FFT operation is required to properly arrange the output data array.

The mixed radix transform described by Singleton (1969) is used where the N data samples may not be a convenient power of two but can be factored into prime integers. In essence, the mixed radix transform uses algorithms written for subseries of a prime number of elements in a pattern similar to the powers-of-two algorithm. For example, if $N = r \cdot s$ (r and s prime), the data sequence is rearranged as an $r \times s$ array, and a two-dimensional transform requiring $N \cdot (r + s)$ operations is performed.

B.2 DATA PRE-PROCESSING

Prior to using Fourier analysis techniques, the raw digitized data must be scaled to engineering units using calibration constants based

F-121 FFT Algorithm for N=8

on transducer sensitivity, amplifier gains, and digitizer range and resolution. Next, this scaled data must be edited for processing by the FFT algorithms or other statistical routines. To maximize the sensitivity in dynamic measurement, the steady-state components of the signal are often removed by either some referencing technique or by some RC coupling technique in the instrumentation (a high-pass filter with a low cutoff frequency). However, these techniques are not successful in removing non-zero mean values and linear trends in the data. Hence, numerical techniques to calculate offset and linear trend are employed to remove these effects. Failure to remove these components will result in spectral data with erroneously high power content at low frequencies and correlation data obscured by the highly-coherent mean value. Accurate numerical methods necessary to decrease round-off errors are outlined in Otnes and Enochson (1978) and several user guides for time series analysis software packages.

Once the data has been detrended, additional editing of spurious points (e.g., random noise spikes) can be performed either manually or by statistical tests. The so-called "wild" points may be detected with a 3σ test (σ = standard deviation) and replaced by linear or exponential interpolation schemes, or they may be deleted altogether if possible. Other schemes include numerical filtering, comparison with data extrapolations, use of the sliding median as a "robust" estimator, etc. If wild points are allowed to remain in the data, the sharp discontinuities they represent will be reflected as increased noise levels in the frequency spectra. If a significant number of wild points are detected, the data sequence should be rejected or at least

detrended again, since the offset and linear slope may be affected by removing or replacing the wild points.

The refined data can then be analyzed for its primary statistical moments and amplitude distribution. This data includes the variance (square of the standard deviation), skewness, and kurtosis, which are measures of how well the data corresponds to a Gaussian random variable distribution. The variance is the most often used moment (i.e., the square root of it -- standard deviation) because it represents an average power level, e.g., acoustic sound pressure levels, structural vibration displacement, turbulence intensity, etc.

The amplitude distribution is often displayed as a histogram of the frequency of samples occurring within a class of amplitude intervals. Such histograms can aid in data interpretation and qualification. The bell-shaped Gaussian data and parabolic sinusoidal data histograms are readily apparent and digitizer failures can manifest themselves as empty class intervals or large occurrence probabilities at the ends of the histogram (indicative of saturation of the amplifiers or over-ranging of the digitizer).

B.3 SPECTRAL ANALYSES

Use of the Fourier transform on a real-valued time series of data points results in a sequence of complex Fourier coefficients in the frequency domain. For the formal definition of the Fourier transform, the continuous time function is transformed into a continuous frequency distribution with infinite resolution. However, in the finite sense, as set of values are produced which statistically represent the true

frequency function as average values over finite intervals (resolution bandwidths). Also difficulties are encountered with the abrupt discontinuities which may occur at the ends of the finite sampling interval.

In the case of the auto-spectra -- or power-spectral-density (PSD) function -- the complex Fourier coefficients represent real- and imaginary- spectra which are composed of in-phase and out-of-phase components of the signal. The PSD is commonly defined as:

$$G_{xx}(f) = \frac{2\Delta t}{N} [X^*(f)X(f)] \quad (B.9)$$

where $X(f)$ is the Fourier transformed data set and $*$ denotes complex conjugation. The transformed array contains both positive and negative frequency information, thus necessitating the factor of 2. The function $G_{xx}(f)$ represents the mean-square variance of the measured property in a finite bandwidth. This is consistent with the earlier analog methods for spectrum analysis, where either a series of narrow band-pass filters or a single sliding frequency filter was used to measure the power about the desired center frequencies. Most digital Fourier analysis routines produce data over a range of constant bandwidths, whereas the proportional bandwidths involved in octave analyses generally use other techniques.

The Fourier equivalent of a convolution of two signals, $x(t)$ and $y(t)$, is represented in the cross-spectral-density (CSD) function defined as:

$$G_{xy}(f) = \frac{2\Delta t}{N} [X^*(f)Y(f)] = C_{xy}(f) - jQ_{xy}(f) \quad (B.10)$$

where $C_{xy}(f)$ is called the covariance-spectrum (or co-spectrum) and $Q_{xy}(f)$ is called the quadrature-spectrum (or quad-spectrum). The co- and quad-spectra are the real and imaginary parts of the CSD function.

The most common use of cross-spectral analysis is to compute mutual coherence γ_{xy} (a measure of statistical correlation at a given frequency) and phase, θ_{xy} , relationships:

$$\gamma_{xy}^2(f) = \frac{G_{xy}(f)^2}{G_{xx}(f) \cdot G_{yy}(f)} \quad (B.11)$$

$$\theta_{xy}(f) = \tan^{-1} \frac{Q_{xy}(f)}{C_{xy}(f)} \quad (B.12)$$

For linear systems (described by linear mathematical operators), a gain or transfer function, $H(f)$, can be defined which describes the frequency response of the system. For input and output signals $x(t)$ and $y(t)$, respectively, the system has a unit impulse response function $h(t)$ which is related by the following convolution integral:

$$y(t) = \int_{-\infty}^{\infty} x(\tau)h(t - \tau)d\tau \quad (B.13)$$

Note the upper bound is t for physically-realizable systems.

Applying the convolution integral theorem (Parseval's theorem for $t = 0$) one sees:

$$y(t) = \int_{-\infty}^{\infty} X(f)H(f)\exp(j2\pi ft)df \quad (B.14)$$

This leads directly to:

$$H(f) = \frac{Y(f)}{X(f)} \quad \text{OR} \quad |H_{xy}(f)| = \frac{|G_{xy}(f)|}{|G_{xx}(f)|} \quad (\text{B.15})$$

Thus, the frequency response of a causal relationship can be represented by the ratio of cross-spectral and input-spectral magnitudes.

The current methods of computing PSD's and CSD's are of two types: the direct FFT and the Blackman-Tukey spectrum techniques. The direct method is more efficient, however, the Blackman-Tukey(B-T) method is more often accepted as the standard for spectral computations. The B-T algorithm first constructs a correlogram (covariance function) and then transforms it to the Fourier coefficients for the PSD values. The direct and B-T methods are asymptotically equivalent for large sample sizes. These methods and others are discussed in detail in several references, including Blackman and Tukey (1958) and Ottes and Enochson (1972 and 1978).

The two common problems of any of the finite Fourier transform methods of computing spectra are leakage and statistical confidence. The leakage is the result of the functional discontinuities at the ends of the sampling intervals and manifests itself as a $\sin x/x$ response in the frequency plane (see F-122). The large sidelobes in the spectral data are attenuated by convoluting the time series with a weighting or tapering function (also called a data lag window) so as to give near zero values at the ends of the data sample. Typical windows and their properties are listed in Table 3. The penalty for decreasing the sidelobes is an increase in the effective resolution bandwidth (see

F-122 Rectangle Window and Transform

TABLE-3: DATA WINDOWS

TYPE	FUNCTION (TIME DOMAIN)	EFFECTIVE BANDWIDTH
Boxcar (Rectangular)	$\left(n = -\frac{N}{2}, \dots, -1, 0, 1, \dots, \frac{N}{2} \right)$ $W(n) = 1.0$	$B_e = \frac{1}{2N\Delta t}$ $1.0 B_e$
Fejer (Triangular)	$W(n) = 1 - \left(\frac{ n }{N} \right)$	
Hanning (Cosine ²)	$W(n) = 0.5 + 0.5 \cos \left(\frac{2\pi n}{N} \right)$	$1.5 B_e$
Hamming (Raised Cosine)	$W(n) = 0.54 + 0.46 \cos \left(\frac{2\pi n}{N} \right)$	$1.36 B_e$
Parzen (Polynomial)	$W(n) = \begin{cases} 1 - 6 \left(\frac{ n }{N} \right)^2 + \left(1 - \frac{ n }{N} \right)^3; & n \leq \frac{N}{4} \\ 2 \left(1 - \frac{ n }{N} \right)^3; & \frac{N}{4} < n \leq \frac{N}{2} \end{cases}$	$1.3 B_e$
Bingham (10% Cosine Taper)	$W(n) = \begin{cases} 1; & n < \frac{2}{5} N \\ \cos^2 \left(\frac{10\pi n}{N} \right); & \frac{2}{5} N \leq n \leq \frac{N}{2} \end{cases}$	$2 B_e$

TABLE-3: DATA WINDOWS

TYPE	FUNCTION (TIME DOMAIN)	EFFECTIVE BANDWIDTH
Ormsby (Trapezoidal)	$W(n) = \begin{cases} 1 & ; n < \frac{2}{5} N \\ 5 \left(1 - \frac{ n }{T}\right) & ; \frac{2}{5} N \leq n \leq \frac{N}{2} \end{cases}$	
Gaussian (Exponential)	$W(n) = e^{-\left[\left(\frac{2n}{N}\right)^2 \frac{\beta^2}{2}\right]}$ <p>β = reciprocal of standard deviation</p>	1.9 B_e
Dolph-Tchebyshev (FFT of $T_n(f)$)	$W(n) = \text{FFT}^{-1} \left\{ \frac{\cos \{ N \cos^{-1} [\alpha \cos(2\pi \frac{n}{N})] \}}{\cosh [N \cosh^{-1}(\alpha)]} \right\}$ <p>where $\alpha = \cosh \left[\frac{1}{N} \ln(10^\beta + \sqrt{10^{2\beta} - 1}) \right]$</p> <p>and β = number of decades down for first sidelobe</p>	
Kaiser-Bessel (Series)	$W(n) = \left\{ \frac{I_0[\beta \sqrt{1 - (2n/N)^2}]}{I_0(\beta)} \right\}$ <p>where β = time-bandwidth product</p>	

TABLE-3: DATA WINDOWS

TYPE	FUNCTION (TIME DOMAIN)	EFFECTIVE BANDWIDTH
------	---------------------------	------------------------

and

$$I_0(x) = 1 + \sum_{k=1}^{\infty} \left[\frac{\left(\frac{x}{2}\right)^k}{k!} \right]^2$$

Frequency Domain Windows

Goodman

$$\tilde{X}(n) = X(n) + \sum_{i=1}^3 a(i)[X(n-i) + X(n+i)] \quad 1.8 B_e$$

$$a(1) = 0.35, a(2) = -0.875, a(3) = 0.0625$$

$$\hat{X}(n) = 0.789 \cdot \tilde{X}(n)$$

G - E - O

(Goodman-Enochson-Otnes)

$$\tilde{X}(n) = X(n) + \sum_{i=1}^3 a(i)[X(n+i) + X(n-i)]$$

$$a(1) = + .1817 ; \quad a(2) = - .1707 ; \quad a(3) = +.1476$$

$$\hat{X}(n) = 0.856 \cdot \tilde{X}(n)$$

F-123 Raised Hamming Window and Transform

F-123). The choice of data window type is dependent upon the purpose and character of the output data and more often simply user preference. Some of the frequency domain windows attempt to correct the $\sin x/x$ behavior by inverting the side-lobe magnitudes and smoothing the spectral estimates. Care must be exercised in selecting data windows for cross-spectral analyses since the averaging can greatly affect the coherence, phase, and transfer function outputs.

The confidence levels for the raw spectral estimates are based on the sampling requirements and smoothing or averaging techniques employed. For most sampling, the experimenter determines the frequency range of interest and the resolution bandwidth desired. These criteria are direct functions of the time between consecutive samples, Δt , and the total number of samples processed, N . The aliasing or Nyquist frequency, f_{NY} , defines the upper limit of frequencies that may be properly identified by a given sampling rate. If frequencies higher than f_{NY} contain significant power and are not blocked out by low-pass filters, they will be ambiguously interpreted as components at a frequency lower than f_{NY} . For example, a sampling rate of 20,000 samples per second cannot adequately distinguish between a 7.5 KHz signal and a 12.5 KHz signal since $f_{NY} = 1/(2\Delta t) = 10$ KHz. The total sampling period, $T = N\Delta t$; gives a measure of the minimum resolution bandwidth as well as the statistical variance of spectral estimates. For Chi-squared confidence level estimates [see Ottes and Enochson (1972 and 1978)], the most important factor is the number of statistical degrees-of-freedom used in determining any one spectral estimate. Since this is roughly proportional to the number of Fourier

coefficients averaged for each estimate, improved confidence can be attained either by ensemble averaging of several consecutive spectra and statistical averaging of neighboring Fourier coefficients. For a given total number of samples, either of these methods increase the effective bandwidth. The use of data windows also decreases the statistical variability at the cost of less resolution. These confidence level bands are not uniform above and below the spectral estimates and are generally based on a Gaussian noise signal (discrete frequencies are more difficult to analyze).

B.4 TIME-LAG CORRELATIONS

Time-lag correlations are a measure of the temporal covariance of a signal, either with itself or another signal. The auto-correlation function is the time average of the product of the signal with itself displaced in time by a time lag, τ . The auto-correlation function is a normalized auto-covariance as the cross-correlation function is a normalized cross-covariance. The auto- and cross-covariances are respectively defined as:

$$r_{xx}(\tau) = \lim_{T \rightarrow \infty} \frac{1}{T} \int_{-T}^T x(t)x(t + \tau)dt \quad (B.16)$$

$$r_{xy}(\tau) = \lim_{T \rightarrow \infty} \frac{1}{T} \int_{-T}^T x(t)y(t + \tau)dt \quad (B.17)$$

and the corresponding auto- and cross-correlation functions (normalizing by the standard deviations) are:

$$R_{xx}(\tau) = \frac{r_{xx}(\tau)}{\sigma_x^2} \quad ; \quad (B.18)$$

$$R_{xy}(\tau) = \frac{r_{xy}(\tau)}{\sigma_x \cdot \sigma_y} \quad . \quad (B.19)$$

These correlation functions have the property that for zero time lag the auto-correlation has unity value and for stochastically random signals, this would be the only non-zero value. Positive cross-correlation implies that the variation of the two signals is in harmony while negative values indicate a dependence, but the variations are opposite in phase. Any periodicity in a signal along with a corresponding periodicity in another signal will be readily apparent in the time-domain correlation function. This property makes correlation a useful tool for extracting signals from noisy background environments. Calculation of the time delay corresponding to a maximum cross-correlation value of two spatially-separated signals can be used to determine the coupling characteristics of the measured phenomenon. For example, in measuring acoustic pressure at separate locations in a flow, one may deduce whether the disturbances are propagated acoustically or by flow convection. Also the magnitude of the correlation indicates how well the character of the disturbance is preserved. In fact, correlation of a large array of sensors can be used to predict the location of a noise source, as in the case of flyover noise from jet aircraft (eg., separation of compressor and exhaust jet noise).

The use of the FFT algorithm to calculate correlation functions is based on the Wiener-Khintchine Theorem, which establishes the equivalence of spectral-densities and correlations by:

$$G_{xx}(f) = \int_{-\infty}^{\infty} R_{xx}(\tau) \exp(-j2\pi f\tau) d\tau \quad (\text{B.20})$$

$$R_{xx}(\tau) = \int_{-\infty}^{\infty} G_{xx}(f) \exp(j2\pi f\tau) df \quad (\text{B.21})$$

These relations are proven using Parseval's theorem which shows the equivalence of the convolution integral to cross products of the corresponding Fourier Transforms. On the bases of Eqn. (B.20) and Eqn. (B.21), it is more efficient to calculate a raw spectrum and apply the inverse Fourier transform requiring $2N\log_2 N$ operations versus N^2 operations for a direct convolution. F-124 illustrates by way of example the spectral/correlation relationship.

The use of time-lag correlations to obtain space-time correlation commonly referred to in many fluid flow problems is straight forward. Correlation of two components of a given measure and (e.g., x and y velocity components simultaneously measured at a fixed point by an x-array hot wire probe or a dual-component laser velocimeter) can be interpreted as the $\langle \overline{uv} \rangle$ correlation tensor used in time-averaged momentum and energy equations. In these cases the correlation value at $\tau = 0$ is commonly used. However, when two measurements are taken simultaneously at various physical separation distances, several cross-correlations are overlaid and the envelope drawn tangent to the maximal correlation values can be used to determine the characteristic

Constant		Constant
Sine wave		Sine wave
White noise		White noise
Low-pass white noise		Low-pass, white noise
Band-pass white noise		Band-pass white noise
Exponential		Exponential
Exponential cosine		Exponential cosine
Exponential cosine, exponential sine		Exponential cosine, exponential sine

PONER-SPECTRAL

DENSITY, $G_{xx}(f)$

CORRELATION

FUNCTION, $R_{xx}(\tau)$

[FROM BENDAT AND PIERSOL (1966)]

F-124 Examples of $G_{xx}(f)$ vs $R_{xx}(\tau)$

scales of various disturbances in the flow. Also, overlaying the cross-correlations, displacing the origins so that the peak correlation values coincide can reveal the time lag due to propagation of the disturbances and this, coupled with the separation distance, can infer a spatial correlation function.

As in spectral estimates, sampling for correlation computations must consider the effects of sampling rate and total sampling period and the computational accuracy. Generally, correlation delay times greater than 20 percent of the total period should be avoided. This is due to the effects of the finite sampling interval and the resulting loss in statistical confidence as the datapoints are lost when the sliding (of $t + \tau$ units) decreases the number of non-zero data points used in the computation. Also, accurate resolution of a disturbance of relatively high frequency may require an increase in sampling rate to 10 to 20 samples per cycle. Examination of the decay of a periodic disturbance due to damping effects may require sampling over 30 to 40 periods. Therefore, the experimenter may judiciously select sampling criteria if he knows some details about the disturbances he wishes to measure.

B.5 DIGITAL FILTERS

In addition to the use of time-lag correlation to separate a signal from the uncorrelated random noise, digital filters may be designed to remove or enhance given portions of the frequency spectrum.

The basis for much of the digital filter design methodology is the so-called z-transform. Since most of the analog filter networks can be

described as a series of poles and zeros in the s-plane defined by the Laplace Transform, the substitution of $a = e^{-st}$ incorporates a time-shifting function which produces the equivalent z-transform:

$$X(z) = \sum_{n=-\infty}^{\infty} x(n)z^{-n} \quad . \quad (B.22)$$

This transform has a discrete characteristic in that for a shifted unit impulse sequence such as

$$x(n) = \begin{cases} 1; & n = 1 \\ 0; & n \neq 1 \end{cases} \quad (B.23)$$

the corresponding z-transform is

$$X(z) = z^{-1} \quad (B.24)$$

Like the Laplace transform techniques, the z-transform reduces the second-order differential equation describing filter behavior to an algebraic difference equation:

$$y(n) = \sum_{i=0}^q L_i \cdot x(n-i) - \sum_{i=0}^m K_i \cdot y(n-i) \quad (B.25)$$

where K_i and L_i are the numerical filter coefficients. Applying the z-transform to Eqn (B.25), one gets the system function:

$$Y(z) = X(z) \left\{ \frac{\sum_{i=0}^q L_i z^{-i}}{1 + \sum_{i=0}^m k_i z^{-i}} \right\} \quad (B.26)$$

where the rational fraction on the right side is the transfer function,

$H(z)$. The unit impulse response function in the time domain, $h(t)$, is the inverse z -transform of $H(z)$. Generally, $h(t)$ is convolved with the sequence $x(t)$ (and $y(t)$ in recursive filter designs) to produce the filter response $y(t)$. The frequency response of a given filter design is obtained when one substitutes $z = e^{j2\pi f\Delta t}$ into the transfer function $H(z)$. The modulus and phase of the response are given as:

$$\text{GAIN}(f) = |H(e^{j2\pi f\Delta t})| \quad (\text{B.27})$$

$$\text{DELAY}(f) = \tan^{-1} \frac{\text{Im}\{H(e^{j2\pi f\Delta t})\}}{\text{Re}\{H(e^{j2\pi f\Delta t})\}} \quad (\text{B.28})$$

These functions are similar to the transfer function gain and phase analyses discussed in the earlier section on spectral analysis. See Peled and Liu (1976) and Helms and Rabiner (1973) for more complete derivations and explanations of z -transforms..

The two basic filter types are the FIR and IIR designs. The FIR-type is a non-recursive linear filter such that all the K_i coefficients are zero. It has a finite-duration impulse response, hence the FIR label. After q time steps, the filter will have zero response to an initial input impulse. The IIR filter design is recursive with a feedback path to use previous output values along with new input values to generate the next successive output. Hence, it is possible to have an infinite-duration impulse response (IIR) when some or all of the K_i coefficients are non-zero. All of the physically-realizable digital filters (causal type) are defined only for previous samples and not for samples yet to be processed, therefore, negative

indices in the z-transform definitions Eqn. (B.22) are not considered. Only predictive filters such as the Kalman or Wiener filters would consider the "future" samples.

Both IIR and FIR types may be implemented in direct, parallel, or cascade configurations. For the parallel configuration, the transfer function must be rewritten as a sum by partial fraction expansion, as:

$$H(z) = \sum_{i=1}^K H_i(z) \quad (B.29)$$

where $H_i(z)$ is the response of the i^{th} parallel stage (K stages). This facilitates using lower-order stages here the numerical errors may be minimized and in certain hardware using parallel processors or several hardwired digital filter modules, the efficiency and speed can be improved. Cascade realizations are defined by:

$$H(z) = c * \prod_{i=0}^x H_i^*(z) \quad (B.30)$$

This method is used to improve numerical stability and reduce the complexity of the digital algorithm. A case example of a cascade IIR filter formed from successive second-order stages is depicted in F-125. For each of these implementations, the K_i and L_i coefficients must be modified.

Computer programs for generating coefficients for both FIR and IIR type filters are given in McClellan, et al. (1973), Peled and Liu (1976), and Steiglitz (1970). Numerous references on the methods and error sources for digital filters are available in current books and

TRANSFER FUNCTION

$$Y(z) = c * \prod_{K=1}^{\chi} \frac{1 + a_{1,k} z^{-1} + a_{2,k} z^{-2}}{1 + b_{1,k} z^{-1} + b_{2,k} z^{-2}}$$

χ = NO. OF SECOND-ORDER CASCADES

$\triangleleft a \equiv$ MULTIPLIER

$\square z^{-1} \equiv$ UNIT SAMPLE TIME DELAY

F-125 Cascade Form IIR Filter

technical literature. In addition to the natural instabilities of filters, the round-off errors and quantization effects are sometimes quite serious in digital filter designs. Whereas, analog filters are stable for distinct pole locations in the negative real half of the s-plane under the Laplace transform, the discrete z-transform maps this region to the inside of the unit circle in the z-plane and digital filters require distinct poles such that $|z| < 1$.

The generation of digital filter coefficients uses several basic methods for computing the proper values. For non-recursive FIR filters, the method of McClellan, et al. is to optimize estimates of the desired ideal frequency response using a minimum weighted Tchebyshev error norm and a calculus-of-variations method called the Remez exchange algorithm. This technique is also called the equiripple method, since the stopband contains several lobes of equal amplitude [see McClellan, et al. (1973)]. The use of Fourier transform methods suffers the same effects of finite-interval computations as PSD processing, and as a result, analogous data windowing techniques are used to minimize the amplitude variation near cutoff frequencies known as the Gibbs phenomenon. The recursive IIR filter design technique of Steiglitz (1970) uses the non-linear optimization Fletcher-Powell algorithm and a bilinear z-transform mapping of poles within the unit circle to duplicate the desired amplitude response in frequency. Rabiner, et al. (1972) discuss three IIR design methods: (1) impulse invariance, (2) bilinear z-transformation, and (3) matched z-transformation. All three of these rely on the continuous-time analog filter designs. The first corresponds to the discrete

equivalent obtained by standard z-transform techniques. The bilinear form relies on mapping the imaginary axis of the Laplace s-plane to the unit circle in the z-plane and optimization of the pole positions within the stable interior of the circle. The matched-z technique relies on a direct mapping of continuous-time filters of desired frequency response to sampled systems by the mapping:

$(s-s_1) \rightarrow [1-\exp(s_1\Delta t) z^{-1}]$. Many of the tables constructed for analog filters (e.g., elliptic, Butterworth, Tchebyshev, sine, tangent, etc.) can be used to design digital frequency domain to the digital frequency domain.

As pointed out by Rader (1968), the practicality of digital filtering is based on its flexibility and compactness (as a result of the large-scale integration - LSI - trend in digital hardware designs). A digital filter can currently operate in the real-time environment as a pseudo-analog component and can be quickly and easily reprogrammed to exact and repeatable characteristics. They can be used to enhance data by eliminating fixed frequency noise inputs (e.g., power line frequency pickup) and data analysis techniques such as the ZOOM FFT where a given frequency range is filtered for a detailed spectral analysis on a sub-set of the spectra. Interpolation, decimation, smoothing, and spectral decomposition are also possible. Adaptive filtering techniques are currently being developed to allow data dependent digital filter responses (non-linear).

B.6 COMPLEX DEMODULATION

The computation of narrow-band spectral analyses prior to the advent of the FFT computer algorithms was performed by complex demodulation techniques. However, even with FFT efficiency, the frequency resolution is insufficient in many cases to examine a highly-periodic signal or to detect a fixed frequency component. The development of complex demodulation has continued in the FFT-era due to the incorporation of digital filters in the process. A general procedure for calculating complex demodulates of a real sequence, $x(t)$, is as follows:

- 1) Generate sine and cosine sequences x_s and x_c by:

$$x_s(t) = x(t) \sin(2\pi f_c t) \quad (B.31)$$

$$x_c(t) = x(t) \cos(2\pi f_c t) \quad (B.32)$$

where f_c is the frequency of interest. This results in a frequency shifting of the f_c -components of the signals to the origin in the frequency plane.

- 2) The frequency-shifting in the first step allows application of low pass filters to the sine and cosine sequences; hence, the sequence is smoothed and higher harmonics of f_c are attenuated. These outputs sequences are denoted $y_s(t)$ and $y_c(t)$.
- 3) The sequences can then be decimated by omitting every k^{th} value and analyzing the resultants complex-valued series for magnitude and phase using:

$$A(f_c) = \frac{1}{T} \int_0^T \sqrt{[y_s(t)]^2 + [y_c(t)]^2} dt \quad (B.33)$$

$$\theta(f_c) = \frac{1}{T} \int_0^T \tan^{-1} \left\{ \frac{-y_s(t)}{y_c(t)} \right\} dt \quad (B.34)$$

where the integrals can be computed numerically. The complex sequences $[y_c(t) + jy_s(t)]$ can also be integrated in separated real and imaginary parts to obtain the complex Fourier coefficients analogous to the FFT outputs.

Since the error in estimating the magnitude of a fundamental component (of possibly unknown frequency) is proportional to $\sin(2\pi n \Delta f T) / (2\pi n \Delta f T)$, where $\Delta f = f - f_c$ (the error in estimating the actual frequency), it is necessary to accurately estimate or to iteratively optimize to the frequency desired. In the case of an oscillating airfoil pressure measurement, the initial estimate can be obtained by determining the maximum power and the corresponding frequency of an FFT-calculated raw spectra. The search is then optimized on the interval $(f_{FFT} - 1/2NT, f_{FFT} + 1/2NT)$, where f_{FFT} is the original estimate. Convergence of the complex demodulation values allows one to have confidence in the final results.

B.7 CONCLUSIONS

The vast improvements in FFT algorithm efficiency and computer speed have brought Fourier analyses to a point of greatly expanded usefulness and to a wide range of applications. While much of the time series analysis work has been aimed at communication theory and more recently voice characterization and encoding, there is a great potential for its use in aerodynamics and fluid mechanics. Recent examples of the use of digital analysis techniques in fluid mechanics

are given by Strahle (1979). Examples of the use of time series analysis in fluid mechanics include turbulence closure modelling based on data cross-correlations reported by Van Atta and Chen (1969), jet engine inlet diagnostics for characterizing the frequency content of unsteady distortion patterns reported by Crites and Heckart (1970), and analysis of aircraft dynamic stability data described in an account by Clark and Croll (1977). In addition, digital filtering techniques are replacing many of the analog counterparts of instrumentation because of flexibility and ease of maintenance factors. Also, custom tailoring of the analysis and acquisition processes is facilitated since the raw digital data can be recorded directly by only moderately sophisticated computer equipment. Re-programming the analysis software is often quicker and easier than acquiring new analog hardware and the resulting interfacing problems.

Interpretation of the time series analysis data is becoming more common and better understood as use of digital techniques expands. Whether the experimenter chooses pre-programmed FFT analyzer equipment or software for a general purpose computer system depends on his resources and goals. The Fourier analysis software can be obtained from sources such as the UCLA Bio-Medical Division (BMD) statistical analysis package or from computer software firms on leasing or purchase agreements. However, it is also possible to generate "home-brew" routines that may have customized benefits for local users. The open literature has available a wide variety of theory, algorithms, computer routines, and application techniques; and bibliographical sources such as Helms (1973) contain a comprehensive listing of the current publications.

APPENDIX C

TUBE-VOLUME RESPONSE ANALYSIS

The problems associated with measurement of unsteady pressures include prediction of the transfer function gain and phase for transducers which are not flush-mounted. The results obtained at NLR depended heavily upon proper prediction and calibration of the tube-volume response characteristics of long pressure tubes and scanning valve pressure adaptors. Their development of methods to compute the acoustic transfer function of such systems was based on Iberall's (1950) derivation of a theoretical model for pressure response in long instrumentation lines. Additional corrections for the effects of the local flow across the opening of the pressure port were incorporated by NLR. Computer codes to analyze the frequency response, based on Iberall's derivations and NLR work, were developed by Nyland, et al. and improved by addition of the flow correction by the present author at OSU/AARL. A detailed discussion of this work is given below.

In Iberall's (1950) paper, the derivation was based on Poiseuille's law of viscous flow in a tube. The oscillatory response calculation accounted for compressibility effects through the use of the polytropic relation ($p = c\rho^n$, where $1 \leq n \leq \gamma$) where the exponent n was adjusted for processes corresponding to isothermal or isentropic compression. The correction introduced for finite amplitude

oscillatory pressures was based on the assumption that a Poiseuille velocity profile (well-developed in a long tube) was preserved at higher pressure amplitudes. The density ratio ($\eta = \rho/\rho_0$) was predicted by the nonlinear, partial differential equation developed from Poiseuille's Law and the finite tube volumes, as given here:

$$\frac{\delta^2 \eta^2}{\delta x^2} = \frac{2\lambda_{T_0} \delta \eta}{L^2 \delta t} \quad (C.1)$$

where λ_{T_0} is a time constant based on instrument volume and the polytropic flow relation. Fourier expansion of Eqn. (C.1) results in a system of differential equations for the Fourier coefficients. Iberall obtained a series of solutions for the coefficients. The solution of this series is based on assumptions that the waves are finite and higher harmonics are rapidly attenuated by viscosity. This gives a rapid series convergence. The accuracy from three terms of the series was found to be adequate for practical processes. Modelling the input functions as a harmonic series expansions and the boundary condition at the sensing end of the tube as:

$$\eta \frac{\delta \eta}{\delta x} = - \frac{\psi_{I0}}{\omega L} \frac{\delta \eta}{\delta t} \quad (C.2)$$

where ψ_{I0} is an attenuation parameter based on instrument volume, ω is the angular frequency, and L is the tube length.

A further correction is applied to account for the local fluid acceleration. This was performed using the more complete Navier-Stokes equations, along with continuity and energy balances to derive a new

attenuation parameter and velocity terms for infinite tubes. The resulting pressure distributions and attenuation factors give improved results for higher frequency pressure oscillations where the assumptions of Poiseuille flow are not as accurate and there is not significant distortion due to finite pressure amplitudes.

The final correction applied by Iberall was for the finite length -- and effects in the entrance flow region where a Poiseuille velocity distribution is not well-developed in the tube. Using one-dimensional relations to correct for the virtual mass effect at the entrance and viscous losses as indicated by Goldstein (Modern Developments in Fluid Mechanics, Vol. 1, p. 302), a pressure excess term is derived that gives a fundamental frequency and second harmonic response for sinusoidal inputs.

Extensive derivations are given for the oscillatory response of long tube and the results are stated as a vast system of equations defining coefficients and attenuation parameters. Iberall presents the complete set of equations for those with great "computational fortitude", and selected graphical representations of the amplitude and phase behavior for a moderately damped behavior so readers could obtain quick estimates for approximate cases.

Bergh and Tijdeman (1965) followed Iberall's procedure, using several simplifying assumptions, and then extended the result to analyze multiple tube-volume sections in series. Their results are restricted to rigid tubes with ideal gases, small disturbances, long tubes of small diameter, and laminar flow throughout the system. A recursive formula for the complex pressure ratio was derived involving

the tube geometry, interconnecting volume, mean gas properties, and frequency. These predictions were experimentally verified for a multiple tube/scanning valve system used at NLR for oscillating airfoil experiments. The calibrations were performed for static crossflow conditions at the tube entrance and the response functions were later corrected when an empirically derived main flow influence was added to the analysis. That result was reported in Tijdeman and Bergh (1972) and was based on a boundary layer displacement model using an empirical constant obtained by correlation of a flush-mounted transducer signal with the tube-volume system measurements. The mathematical details of those static and cross-flow pressure response relations were significantly simplified from Iberall's forms, but are still rather large expressions involving complex-valued Bessel functions and hyperbolic trigonometric terms. The use of these expressions to compute simple cases remains quite tedious.

In 1965, Watts published a computer program, written to compute the pressure response of long tubes used to isolate transducers from the radiation in a nuclear rocket engine. Experimental efforts to verify Iberall's equation were conducted. It was observed that various previous attempts to correlate with Iberall's results suffered from either improper modelling of the input boundary condition, or violation of the assumption that the tube Reynolds number was small (less than 2000). The use of a pressure driver of finite coupling volume at the tube inlet did not produce results in agreement with Iberall's assumption using an infinite inlet volume boundary condition. The analysis was also found to be inadequate for cases where tube

length-to-diameter ratio was small. The errors in determining the frequency and for computing the amplitude of the first resonant peak in the tube response function were found to be inversely proportional to the length-to-diameter ratio and its square, respectively. Watts compared his experimental data with the results computed by the FORTRAN program written to evaluate Iberall's equations and concluded that the results were accurate to within one percent for cases properly obeying the underlying assumptions.

A later effort by Nyland, et al. (1971) undertook to program the pressure response equations as modified by Bergh and Tijdeman for short pitot pressure probes. Their use was to determine the influence of probe response on measurement of dynamic distortion patterns and stall propagation in jet engines. The report includes experimental data obtained with an adjustable organ-pipe resonator and probes with $L/D \sim 8$. It examined straight probes and two other designs for particle impact protection, including a perpendicular transducer installation and a baffle design used to shield the sensing element. Also, the effect of a branch tube to measure steady-state pressure was examined and found to have negligible influence for properly controlled design parameters. The response program was coded to analyze a series of tube-volume segments at prescribed gas conditions. This program was in excellent agreement with the measured results.

An extension to the static response predictions was added to Nyland's code at OSU/AARL when the present author incorporated the main flow correction suggested by Bergh and Tijdeman. This modified code was tested against results published by NLR and other empirical data.

The code failed to predict results for the installation geometry used in the OSU/AARL airfoil testing. This is primarily due to the transducer volume being relatively large compared to the tube length and diameter.

Forsching (1975) presented a review of the wind tunnel test techniques used to examine unsteady aerodynamics and dynamic stability problems. In discussing unsteady pressure data measurements, he outlined the NLR methods for computing a tube-volume response function and indicated a procedure for still air calibration of these systems. The effects of various geometric and mean gas properties were pointed out to illustrate that careful calibrations and analyses were necessary to obtain accurate unsteady pressure results.

The instrumentation setup at OSU/AARL incorporated a direct measuring technique where in-situ transducers were mounted with their diaphragms parallel to the plane of oscillatory motion to reduce sensitivity to the model accelerations. The small cavity used in the mounting technique is not suitable for analysis by Iberall's or NLR's tube-volume response formulas, and is more appropriately modelled as a Helmholtz resonator. The results of such analyses indicated resonances in excess of two kilohertz in frequency -- far above the frequency range of interest.

The natural frequency of a Helmholtz resonator is usually described in terms of the internal volume, V ; the entrance neck area, A_N ; and neck length, ℓ_N . The standard formula for calculating the Helmholtz resonant frequency is:

$$f = \frac{a}{2\pi} \sqrt{\frac{A_N}{V(\ell_N + \alpha)}} \quad (C.3)$$

where a equals the acoustic velocity and α equals the neck length end correction ($\sim .8 A_N$). For a typical unsteady transducer installation (see F-126), the Helmholtz frequency is 6580 hertz, well beyond the frequencies of the oscillating airfoil experiments. Alternatively, looking at the installation as an open-ended organ pipe, resonance is calculated as:

$$f_o = \frac{c}{2\ell_N} = 42.4 \text{ Khz} \quad (C.4)$$

This again is far above the frequencies of concern. An improved method of calculating Helmholtz resonances was developed by Alster (1972), where the internal volume is treated as a form factor which accounts for the geometry and depth of the resonator main volume. The installation in the oscillating airfoil models can be viewed as a cylinder with a lateral hole. For the relatively long neck in the current case, the form factor is approximated as $\mathcal{L}_v = 1.88 r^2/L$, where r equals the radius of the neck and L equals the length of the internal cylinder. The detailed computation of the resonant frequency as proposed by Alster (equation 33), gave a value of 5850 hertz. This lower frequency is still beyond the spectral range of interest. Therefore, it is highly unlikely that the in-situ transducers installed in the NACA 64A010 models suffered response losses over the frequency ranges pertinent to the oscillating airfoil testing.

F-126 Typical Transducer Installation

APPENDIX D
UNSTEADY SCALING RELATIONS

Aerodynamic scaling of the mean flow parameters and local pressure, lift, and moment data observed during subscale studies can be determined through application of similarity analysis to the equations of motion and ideal gas relations. The basic parameters are scalable if certain non-dimensional parameters are properly simulated. The most important parameters to simulate during transonic oscillating airfoil experiments are: specific heat ratio ($\gamma = C_p/C_v$); Mach number ($M = V/a$); Reynolds number [$Re = (\rho V c / \mu)$], and reduced frequency, [$k = (2\pi f c / V)$]. The data obtained for the current experiments can be used most appropriately for conditions where the values of γ , M , Re , and k match the experimental situation.

Use of the subscale geometric airfoil model is necessitated by the large power requirements in operating full-scale test facilities. Although geometric similarity can be preserved by accurate control of model dimensions, the flow dimensions, including boundary layer thicknesses, turbulence scale size, and streamline patterns, are properly scaled by maintaining the ratio of viscous to inertial forces through simulation of Reynold's number. Accounting for the large reduction in the length scale is usually done by increasing the freestream density by pressurization. The viscosity is primarily

affected by temperature and can be significantly altered by massive cooling of the test media, as in the cryogenic wind tunnel designs. For the OSU/AARL facilities, the problem in Reynold's number scaling is most often treated by the use of increased stagnation pressure in the blowdown wind tunnel facilities.

The compressible flow characteristics are directly simulated by using compressed air at an accurately controlled test section Mach number. This ensures duplication of specific heat ratio and local Mach number and shock wave formation (assuming viscous shock/boundary layer interaction affects are adequately modelled also). Six freestream Mach numbers were examined during the current test program. The Mach numbers chosen are judged to be representative of the different regimes of transonic flight: Mach 0.35, subsonic flight with small compressibility effects (cruise conditions for propeller-driven aircraft); Mach 0.52, moderate compressibility condition (climb and high-altitude loiter for jet aircraft); Mach 0.70, subcritical, transonic flight with moderate influence on streamline curvature due to compressibility (nominal cruise condition for earlier commercial jetliners and private jets); Mach 0.80, near critical, transonic flight where weak shockwaves are seen for non-zero lift conditions (cruise conditions for modern jetliners); Mach 0.84, supercritical, transonic flight exhibiting strong shock formation on the aft portion of the airfoil chord (high-speed cruise for subsonic jet aircraft); and Mach 0.89, near sonic conditions corresponding to oblique shock formation at the trailing edge accompanied by a large supersonic flow zone over most of the airfoil chord (high-speed maneuvering of jet fighter aircraft

and emergency procedures for large transport aircraft). These Mach numbers were picked for testing the NACA 64A010 airfoil shape and corresponding flow regimes for other contours (e.g., supercritical airfoil designs) will likely differ.

The basic parameters and their scaling factors are shown in Table 4. The appropriate scaling parameters are derived by substituting the scaled parameters into the flow equations and separating the scale factors from the data variables.

The thermodynamic pressure scaling is based on the inviscid, steady Euler equation:

$$dP = -\rho V dV \quad (D.1)$$

Substituting the parameters for scaling:

$$d(f_P P') = -(f_\rho \rho') (f_V V') d(f_V V') \quad (D.2)$$

TABLE 4. SCALING PARAMETERS

Model Parameters	Full Scale Parameters
l_x , longitudinal flow lengths	$f_x l_x'$
l_y , spanwise length	$f_y l_y'$
l_z , transverse (vertical) flow length	$f_z l_z'$
P local pressure	$f_P P'$
ρ local density	$f_\rho \rho'$
V local velocity	$f_V V'$
a local acoustic velocity	$f_a a'$

*Primed variables denote the scaled results

After final rearrangement and comparison with the original Euler equation, we see:

$$f_p = f_p f_v^2 \quad (D.3)$$

For pressure scaling, the most commonly used parameter is the dynamic pressure (whose form is nearly identical to the scale factor derivation above):

$$q = \frac{1}{2} \rho v^2 \quad (D.4)$$

The non-dimensionalized pressure coefficient, $c_p = \Delta P/q$ is commonly used for both steady and unsteady pressures. For the oscillating airfoil experiments, the unsteady pressure amplitude is seen to be a linear function of the pitch amplitude (from the linearized harmonic potential equation) and the oscillating pressure coefficient is of the form:

$$c'_p = \frac{P_{rms}}{|\Delta\alpha|q} \quad (D.5)$$

For determining the steady and unsteady section lift and moment coefficients, the pressure coefficients are integrated over the chord length and hence, the only additional scale factor needed for computing data for scaled configurations is the chord length, c .

The time-length scaling parameter commonly used for oscillating airfoil testing is the reduced frequency, $k = (2\pi fc/V_\infty)$. Use of this

parameter is based on the time-linearized, harmonic perturbation potential where a dimensional system based on the airfoil chord length (leading edge at the origin and trailing edge at the unity abscissa), the frequency parameter, ik^2 , appear as a coefficient of the disturbance potential term, ϕ . Some sources use a reduced frequency parameter based on model halfchord (leading edge at -1 and trailing edge at +1). The convention used here is based on the full model chord length.

In order to scale the power-spectral-density (PSD) values, the pressure and frequency scaling given above must be used to convert the abscissa values to reduced frequency and the ordinate values to non-dimensionalized spectral densities. The ordinate are converted from units of $(\text{psi})^2/\text{Hz}$ to non-dimensionalized values using the dynamic pressure and reduced frequency scaling. For the engineering unit data, $G_{XX}(f)$, the corresponding non-dimensional quantity is defined as:

$$\Phi_{XX}(k) = \frac{G_{XX}(f)}{q_\infty^2} \left(\frac{2\pi c}{V_\infty} \right) \quad (\text{D.6})$$

This quantity can be used to compare results from other experiments and for predicting values for other comparable conditions.

In order to scale the current wind tunnel data, there are three important scale factors -- chord length ratio, static pressure ratio, and static temperature ratio, which are required to scale data for cases of equivalent Mach number, Reynolds number, and reduced frequency. The nominal values of the pertinent flow parameters are

given in Table 5 for the OSU/AARL test conditions observed during these oscillating airfoil experiments.

TABLE 5. TYPICAL RUN CONDITIONS

MACH	P _o (psia)	T _o (°R)	q (psia)	Re (x 10 ⁶)*	V (ft/sec)
.346	32.2	512	2.49	2.60	380
.516	31.2	505	4.86	3.59	554
.703	32.0	502	7.97	4.62	737
.797	33.2	499	9.70	5.16	822
.840	33.8	497	10.51	5.42	859
.888	33.1	497	10.93	5.43	902

*For six-inch chord model only

APPENDIX E

DETAILED ERROR ANALYSIS

Experimental accuracy must be determined through a detailed examination of the potential sources for measurement uncertainty. For the OSU/AARL oscillating airfoil experiments in the 6" x 22" Transonic Wind Tunnel were found to have negligible calibration, instrumentation, and data processing errors. Each element in the detailed error train was investigated and then the cumulative error determined through root-sum-square (rss) averaging techniques. The instrumentation and calibration procedures used in acquiring the unsteady data were based on the existing steady-state data acquisition techniques which have produced consistently high quality data over many years of experience.

The error train (as seen in F-127) involves several steps including: (1) coupling of the input measurand to the sensor, (2) conversion of the mechanical pressure energy to electrical energy in a transducer, (3) signal conditioning and amplification, (4) signal detection or digitizing, and (5) data recording. For unsteady pressure measurements the acoustic transfer function for the transducer installation geometry and the influence of transducer size on spatial frequency resolution control errors in the first category. If the pressure measurement is not on a surface, then influence of the sensor probe on the local flow must be assessed. For the oscillating airfoil

F-127 Error Train

tests, the background tunnel noise represents an error influencing the pressure measurand associated solely with the airfoil motion and induced flow oscillations. The electro-mechanical conversion of the pressure signal to electrical impulses is influenced by transducer response characteristics such as sensitivity, linearity, hysteresis, resolution, and response time. These sensor properties are determined by careful calibrations. Proper selection of good quality sensors, accurate calibration data and scaling procedures, and proper matching of the sensor capabilities to the intended use will ensure that measurand-sensor coupling errors are minimized. Radio-frequency and 60 Hz line power noise pickup in the sensor and instrumentation leads must be avoided for these low voltage output pressure transducers (signals can be on the order of microvolts to millivolts). Proper cable shielding and grounding practices can eliminate most of this error. Also, impedance-matching becomes important for high frequency response when high-impedance transducers and long cables are needed. Use of the line-driver amplifiers to boost the signal and better match impedance can reduce transmission line losses generally manifest as attenuation of higher frequency signals. The relatively low impedance of the Kulite transducers (nominally 750 ohms), double-shielded cables, and single-point grounding at the amplifier reduced electrical noise pickup below 110 dB SPL in a relatively broadband spectral distribution. The third stage of the error train depends on the electronic instrumentation quality. Properly stabilized and isolated excitation power supplies will have very low voltage ripple or long-term drift as well as providing high common mode rejection (prevention of cross-talk

from one sensor to another by influencing the main power supply). Those issues are resolved by using highly-regulated, individual power supplies for each sensor. The amplifier response and noise characteristics can modify the signal content by nonlinear response at high or low frequencies, noise pickup in the various amplifier stages, and improper input and output impedance matching. High-gain operational amplifiers provide high input impedance so that good voltage response is maintained while not allowing significant ohmic losses in the instrumentation lines and low output impedance to properly drive the output lines with the amplified signal. The resolution of the detector stage (in this case, the analog-to-digital converters must be sufficient to determine changes in the amplified signal commensurate with the baseline electrical noise levels. The digital resolution limit is known as the quantization error and is set by the number of digital bits output and the digitizer range. The 10-bit Datal digitizer used at OSU/AARL can resolve one part in 1024, and hence, it has a quantization error less than 0.1 percent over its ± 10.0 volt range (i.e., ± 10 millivolts). Direct digitizing of these signals rather than recording them on an FM-analog, magnetic tape recorder preserves the 60 dB signal-to-noise ratio between the amplifier output and the digital data rather than providing additional noise generation in the recording and playback stages. For the digital approach, recording noise is virtually eliminated since the digital data is exactly transmitted to the digital tapes and disc file storage. The final sources of error are in the analysis techniques used in data reduction. The digital processing accuracy is limited by

precision errors associated with the computer word size and by mathematical accuracy of the data processing algorithms such as the Fast Fourier Transform (FFT) technique and statistical analysis routines. The 48-bit double-precision, floating-point variable processing in the Harris Slash 6 computer affords more than twelve decimal digits of precision and test data cases show the FFT and averaging routines to be accurate to at least eight decimal digits. This precludes any significant errors due to the data analysis procedures.

The assessment of the overall error must be viewed as two parts to the definition of measurement uncertainty (see Abernethy and Thompson (1973) for a more complete discussion on uncertainty). Uncertainty is the combination of fixed (bias) errors and random (precision) errors. The bias (B) is the summation of fixed (in a time-frame relative to the measurement period) offsets such as thermal drift, nonlinear response, changes in transducer sensitivity, amplifier gain, excitation voltage, etc. Large known biases are determined during calibrations and could be removed by data correction techniques. Small known biases are those that are discovered in calibration but can be considered to be negligible and are generally not worth correction. Large unknown biases occur due to human errors, installation imperfections, and environmental effects. They are controlled by using good measurement techniques and quality control methods in sensor selection, installation, and maintenance of the instrumentation systems. Small unknown biases encompass errors judged acceptable through experience (e.g., thermocouple radiation losses, etc.). The precision, S, is the

data variance computed by root-sum-square combination of the various random, non-repeatable errors. Factors such as digital quantization, RF noise pickup, broadband amplifier noise, etc. all contribute to the precision error. The algebraically summed biases are added to a multiple of the precision to compute the uncertainty. The multiplicative constant is based on the statistical degrees-of-freedom and the desired confidence level for the measurement. Generally, the factor is chosen to equal 2.0 based on a 95 percent confidence level for 30 or more degrees-of-freedom. Therefore, the uncertainty, U , is computed by:

$$U = \pm (B + 2S) \quad (E.1)$$

This sets the range of uncertainty for absolute measurements. However, the unsteady data processing removes the statistical mean prior to determination of the r.m.s. pressure levels and further Fourier analyses. Simultaneously, most of the bias errors are removed from the data. The remaining uncertainty is primarily precision errors associated with ergodic noise sources as well as narrow-band influences such as 60 Hz line power noise. These sources were minimized during the oscillating airfoil tests.

Many of the biases and precision errors can be reduced by such techniques as: (1) cross-parameter correction methods (e.g., adjustment of sensitivities of pressure transducers according to pre-calibrated thermal influence and on-line temperature measurements), (2) careful design of impedance matching circuits, (3) on-line calibrations of in-situ sensors with built-in checks, (4) improved

regulation of excitation sources or direct measurement of excitation values for post-test data corrections, (5) automatic instrumentation checks performed periodically during a test series, (6) increased sample sizes for improved statistics, (7) good quality control and preventive maintenance methods, and (8) careful, well-documented calibration and inspection procedures for all sensors. Such efforts should limit the measurement uncertainty to the inherent sensor accuracy.

The Kulite transducers were calibrated using a water manometer system to pressurize the transducer reference system in a ten-step D.C. calibration and the on-line computer system. The fully-assembled instrumentation system is calibrated with the transducers installed in the tunnel. Slope and linearity of the transducer sensitivity were computed and compared with the manufacturer's test reports. The linearity results agreed to within ± 0.3 percent. In the same sequence, bridge substitution resistors were automatically switched into the circuit to establish pressure values. The values were used in pre-run calibrations performed just before each tunnel run sequence to correct for day-to-day variations in the instrumentation system characteristics.

Typical calibration data are shown in F-128 where the output listing and calibration plot for a Kulite sensor are given. The pertinent hookup data, instrumentation settings and calibration samples are listed. The transducer slope and zero listed are based on a least squares fit to the data. The pressure constants are computed for the R-cal voltages and listed for use in the on-line calibration sequencing. The apparently large difference in sensitivity between OSU

F-128 Transducer Calibration Plot

data and the manufacturer appeared to be somewhat systematic with the OSU transducer slopes about 8 percent higher than those reported on original calibration sheets. For the sixteen transducers calibrated during this single calibration run, the sensitivities were all within ± 2 percent of this 8 percent offset. The direct calibration of the sensors to the OSU/AARL data acquisition system showed that use of the manufacturer calibration data directly could generate up to a 10 percent error. The OSU transducer calibrations were used in order to minimize any system calibration errors. The data plot shows how closely the individual data points match the least squares line fit. On this basis the calibration errors are well within the ± 5 percent error band generally accepted for engineering measurements.

General considerations for unsteady pressure measurements are outlined by Richards (1977) including selection of transducers with: (1) adequate sensitivity and range (2) adequate frequency response characteristics, (3) suitable physical size for the particular installation and for adequate spatial resolution, (4) relative freedom from extraneous noise pickup and change in response characteristics, (5) linear response over the required range, (6) minimal zero drift and maximum repeatability, (7) minimum hysteresis and backlash, (8) relatively easy calibration, and (9) suitable operational characteristics and compatibility with the User's instrumentation system and data acquisition procedures.

The literature dealing with unsteady pressure measurements has been extensive over the past three decades. Both sensor development and techniques for improving the data accuracy have been topics of

several papers and publications. Arnold (1955) indicated that special transducers had been designed to measure pressures during oscillating airfoil test at NACA's Ames Research Center. The sensor was a strain-gage bonded to a 0.003 inch (0.076 millimeter) thick steel diaphragm. Davis (1958) reviewed the NACA-Langley experiences with a tube-volume measuring system and analog methods to integrate pressures to obtain overall aerodynamic coefficients using Gaussian quadrature. Inskeep (1961) outlined transducer evaluation techniques including shock tube response tests, low-frequency tank methods, sirens, etc. Danberg (1962) addressed the effects turbulence might have on static probe response. He deduced that fluctuations create higher mean pressure readings. Bergh (1964) reported on the NLR experiments and oscillating airfoil pressure measurements obtained with a tube-volume system consisting of a scanning valve, equal-length pressure tubes, one in-situ pressure transducer, and one transducer coupled to the scanning valve assembly. The acoustic transfer function was computed (see Appendix c) and experimentally confirmed with the in-situ measurements. Siddon (1969) described a static probe which compensated for cross-flow fluctuations and gave an evaluation of the measurement accuracy. Arndt and Nilsen (1971) used 1/8-inch microphones mounted in pitot and static probes to measure fluctuations in an axisymmetric jet flow. To date, the majority of unsteady pressure measurements made during oscillating airfoil tests are made with the compact, semiconductor strain-gauge sensors or tube-volume systems like those developed in the Netherlands at NLR.

Parallel to the sensor research and development efforts are the reports of work in measuring techniques and methods. Uberoi and Kovasznay (1953) discussed the relation of sensor size (spatial resolution) and the statistical properties to be determined. Demmerle (1964) worked on a resolution and noise relationship for determining the frequency content of a signal using a zero-crossing technique. Alster (1972) developed an improved Helmholtz resonator calculation which can be used to compute the response of transducer installations where the sensor is not flush with the measuring surface. The new prediction method introduces geometric factors accounting for volume, height, shape, neck junction, etc. This prediction method was used to evaluate the response associated with the transducer installation method used in the airfoil models at OSU/AARL. Hanly (1975) examined the effects of transducer flushness and found a large sensitivity to any protrusion. Since surface curvature makes flush installation nearly impossible, it was suggested that slightly submerged transducer installations be used with minimal impact on spectral data and RMS pressure levels. Van Kuren, et al. (1976) examined the transducer sensitivity criteria (based on anticipated local flow conditions) and deduced that data values of less than 1 percent to 20 percent of the dynamic pressure could be expected. Muhlstein and Coe (1979) suggested optimum data sampling times and discussed confidence bands which would be applied to static and dynamic measurements. They developed additional guidelines for estimating required integration times in cases experiencing various flow phenomena, such as buffet, separation, etc.

The sampling statistics and data processing confidence bands associated with the OSU/AARL unsteady airfoil experiments are discussed below. The relative errors in the computation of statistical moments, Fourier spectral analyses, and complex demodulation values are stated in terms of confidence bands or neighborhoods. The raw statistics depend primarily on the number of samples whereas, the Fourier analysis frequency range is more directly tied to sampling rate. The size and sample rates for the unsteady data were adjusted during the various test series. During the final oscillating airfoil tests, the sampling parameters remained fixed at a total of 4100 samples, acquired at a four (4) Kilohertz digitizing rate for a total sample period of 1.025 seconds. (Only 4096 samples were used for subsequent processing.) These parameters proved satisfactory for resolving the oscillating airfoil data.

One parameter useful in discussing data certainty is the correlation time, T , as related to disturbance wavelength, L , and flow velocity, V . For a typical example, disturbances of length equal to the 6-inch airfoil chord at a flow velocity of 500 ft/sec (Mach $\sim$ 0.45) give a correlation time of:

$$T = \frac{L}{V} = \frac{0.5 \text{ ft}}{500 \text{ ft/sec}} = 1 \text{ millisecond} \quad (\text{E.2})$$

Thus, in order to have statistical confidence in a measurement of that disturbance, sampling must cover several correlation times. The length of time needed depends upon the number of occurrences needed for better statistical certainty and the generation frequency of the occurrence. Pure sinusoidal disturbances could theoretically be resolved within a

single correlation time, however, the addition of random noise and harmonics can increase the required sample time to 20 to 50 times the basic time unit, T . For the oscillating airfoil tests, the pressure signals are assumed to be stationary, random processes containing pure sinusoidal waves of constant amplitude and phase. Even though the 6 x 22 inch tunnel operates in a transient blowdown mode, the flow conditions over the one second sampling period are relatively constant (averaged values of pressure and temperature taken both before and after the unsteady data sampling are used in the data analysis). More detailed discussions on sampling criteria are found in Appendix B.

The guidelines for Discrete Fourier processing are given by Otnes and Enochson (1972). The Nyquist or aliasing frequency is:

$$f_{NY} = \frac{1}{2\Delta t} = 2000 \text{ Hz} \quad (E.3)$$

for the sampling time, $\Delta t = 250 \text{ } \mu\text{sec.}$ between digitized data points. This represents the upper limit of frequencies that can be analyzed for these data sets. If higher frequencies containing significant power are present in the signal, they would be reflected or "aliased" into the 0 to 2000 Hz band and appear as false data. This possibility is avoided by using a 1 KHz low pass filter on the amplifier input prior to digitizing. This effectively eliminates aliasing.

Filter resolution or effective bandwidth for digital sampling is based on total sample period, $T = N\Delta t$ (N = total number of samples). Based on Rayleigh's criterion for resolving two sinusoids f_1 and f_2 , where $\Delta f = |f_1 - f_2|$, the resolution bandwidth is set at:

$$\Delta f_{MIN} = \frac{1}{4T} \quad (E.4)$$

for the minimum or just-resolved frequency interval, and $\Delta f = 1/2T$ for a well-resolved case. This analysis is based on the distinction between the $(\sin x/x)$ -shaped spectral components corresponding to the finite-interval, Fourier transform of the two frequencies. The first zero crossings are at a bandwidth of $1/T$, therefore, the just-resolved bandwidth of $1/4T$ corresponds to coincidence of the first zero-crossing with the main lobe peak of the other frequency. The $1/2T$ condition corresponds to frequency separations such that the $(\sin x/x)$ first zero-crossings are overlaid. For the oscillating airfoil a well-resolved bandwidth of 0.488 Hz is predicted. The use of smoothing windows tends to broaden the main lobe slightly, but their main impact is to decrease the amplitude of the side lobes. The well-resolved bandwidth for the oscillating airfoil data is approximately $1/2$ Hz. Spectral smoothing or ensemble averaging for a fixed number of samples will result in greater effective bandwidths for the spectral estimates. This decreases statistical variability and improves the accuracy of the result. The effective smoothing bandwidth, $B_e = M\Delta f$ (where M = number of ensemble averages or spectral points averaged) is used to compute a normalized standard error ($\epsilon_0 = 1/B_e T$). Increasing the averaging bandwidth decreases the error.

The errors in spectral estimation can be discussed in terms of Chi-squared random variables. The confidence level, P , is defined as the expectation that the true value of the spectral function, $G(f)$, lies within the confidence band limits, A and B , about the estimated value, $\langle G(f) \rangle$. Typically, p is chosen as 0.80 to 0.99 corresponding to 80 percent to 99 percent confidence levels; for the case at hand

$p = 0.90$ was selected. The statistical degrees of freedom are defined by $n = 2B_e T$. Hence, the unsteady airfoil data smoothed over 5 Hz bandwidths will have $n = 20$ degrees-of-freedom. The lower and upper confidence limits are defined respectively as:

$$A = \frac{n \langle G(f) \rangle}{\chi^2_{n: \alpha/2}} ; \quad B = \frac{n \langle G(f) \rangle}{\chi^2_{n: 1-\alpha/2}} \quad (E.5)$$

where $\alpha = 1-p$. The current example gives a confidence band about the spectral estimate $\langle G(f) \rangle$ limited by $A = 0.637 \cdot \langle G(f) \rangle$ and $B = 1.843 \cdot \langle G(f) \rangle$. These limits are within a -2.0 dB to +2.7 dB band about the estimates. For a finer frequency resolution ($\Delta f = 2.44$ Hz) it is seen that these 90 percent-confidence bands are increased to -2.6 dB to +4.0 dB limits. It is also noted that these confidence limits apply to general broadband noise spectra and narrow-band peak amplitudes are generally not estimated to a high degree of confidence by Discrete Fourier Transform algorithms.

For single frequency analysis of a nearly sinusoidal function, the statistical variability of finite Fourier analyses is generally not acceptable. Therefore, complex demodulation (frequency synthesis) methods were used to process the oscillating airfoil data. The resulting errors can be classified as statistical variance and frequency mismatch errors. The variance estimates are based on the statistical degrees of freedom and are the same as applied for the usual computation of statistical moments. The synthesizing (for frequency mismatch) error is due to the inaccuracy of selecting the demodulation frequency. As seen above, selection of the frequency from

finite Fourier analysis results leaves an uncertainty at least as great as the resolution bandwidth.

The magnitude of the synthesizing error is now derived for a function, $y(t)$, defined by the Fourier series:

$$y(t) = a_0 + \sum_{m=1}^{\infty} a_m \cos(m\omega t) + \sum_{m=1}^{\infty} b_m \sin(m\omega t) \quad (\text{E.6})$$

where $\omega = 2\pi f$ (angular frequency) and $t = \text{time}$. For a data sample that is over a time period n ($n = \text{number of cyclic oscillations}$, $T = \text{sinusoidal time period}$), the "Real" and Imaginary" Fourier coefficients are given by:

$$\text{Re} \{y, \omega\} = \frac{2}{nT} \int_0^{nT} y(t) \cos \omega t \, dt \quad (\text{Note: } \omega = 2\pi/T) \quad (\text{E.7})$$

$$\text{Im} \{y, \omega\} = \frac{2}{nT} \int_0^{nT} y(t) \sin \omega t \, dt \quad (\text{E.8})$$

Substituting the Fourier series for $y(t)$, we get only one non-zero integral term:

$$\begin{aligned} \text{Re} \{y, \omega\} &= \frac{2}{nT} \int_0^{nT} a_1 \cos^2 \omega t \, dt \\ &= a_1 \left(1 + \frac{\sin 2 \omega nT}{2\omega nT} \right) = a_1 \end{aligned} \quad (\text{E.9})$$

since $2\omega nT = 4\pi n$, and $\sin(4\pi n) = 0$ for $n = \text{an integer}$. Similarly:

$$\text{Im} \{y, \omega\} = b_1 \quad (\text{E.10})$$

Therefore, it is evident that data processing error can exist if an integral number of periods is not used but such potential errors are seen to damp as a $\sin x/x$ function as the sample length is extended. For a frequency mismatch, $\Delta\omega = |\langle\omega\rangle - \omega|$, where $\langle\omega\rangle$ is the estimation of the true frequency ω , the error is derived from the integral forms above. The real and imaginary parts are thus estimated as:

$$\langle \text{Re } \{y, \omega\} \rangle = a_1 \frac{\sin n\tau \Delta\omega}{n\tau \Delta\omega} \quad (\text{E.11})$$

and

$$\langle \text{Im } \{y, \omega\} \rangle = b_1 \frac{\sin n\tau \Delta\omega}{n\tau \Delta\omega} \quad (\text{E.12})$$

The relative errors are computed to be the same magnitude for both parts and is stated as:

$$\epsilon = 1 - \frac{\sin n\tau \Delta\omega}{n\tau \Delta\omega} \quad (\text{E.13})$$

The relative error is bounded by number of cycles analyzed at the frequency estimation error. This provides direction to analyze a minimum number of cycles and to obtain as accurate an estimate of frequency as possible. In the routine written for the OSU/AARL unsteady aerodynamics testing (UA/SYN) this error is minimized by optimizing the frequency estimate in order to maximize the coefficients. If the original estimate was within a bandwidth corresponding to the width of the main lobe of the $\sin x/x$ function, then this technique will stably converge to the true frequency. For a sample case, the error associated with analyzing 10 cycles of a 100 Hz

signal with a frequency mismatch equivalent to the spectral resolution bandwidth of 0.5 Hz, is approximately 1.7 percent of the estimate. Optimizing the frequency will improve on the error.

The unsteady data accuracy has been treated in the paragraphs above. The uncertainties in the sensors, instrumentation, digitizing, and data reduction have all been found to be negligible for the oscillating airfoil data.

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