FLOW INDUCED NOISE CONSIDERATIONS FOR THE WIND TUNNEL TESTING OF A NACA 0015 AIRFOIL WITH SLOTS

Robert Bruce Alstrom, Pier Marzocca, Goodarz Ahmadi
Department of Mechanical and Aeronautical Engineering

Erik Bollt
Mathematics and Computer Science

Clarkson University, Potsdam, New York 13699-5725

ABSTRACT

A wind tunnel experiment was conducted in order to perform an aerodynamic frequency domain identification on a NACA 0015 airfoil with a simple flap. The experimental results revealed the existence of two flow noise sources; mainly a wind tunnel resonance, also referred to as a Parker Mode, and flow induced noise resulting from uncovered directed synthetic jet actuator cavities. The Parker Mode registers as 423 Hz for the given wind tunnel facility with an unstaggered cascade and variable passage (due an articulated wing in cross flow) in its test section. A Parker Mode changes in frequency are reported due to changes in tunnel speed, and to angle of attack, the latter topic not reported extensively in the literature. The flow induced noised produced by the leading edge synthetic jet actuators result from the boundary layer interacting with the open cavities. As a result, acoustic radiation propagates downstream of the 30° angled slots producing distinct tones in the power spectra. The amplitude of these tones will increase and sharpen or collapse due to the change in angle of attack varying from 0° to 24° as the wind tunnel speed is held constant at 35 m/s (M=0.102)

Keywords: Parker Mode, flow induced noise, resonator, NACA 0015 wind tunnel experiment

I. Introduction

The purpose of this paper is twofold: to infer the presence of a Parker Mode found during a wind tunnel test of a two-dimensional rectangular NACA 0015 airfoil, under laminar flow condition, and to observe and investigate the presence of a leading edge cavity noise on the aerodynamic test article which is currently used for active flow control studies. Specifically, two flow noise sources have been identified in the current experimental results; they are flow induced acoustic radiation from the leading synthetic jet array cavities and an acoustic signature caused by the test section configuration. The wing section has engineered flow cavity slots, diffuser like opening up into a shaped cavity, prepared to host 2 arrays of directed synthetic jet actuators located at 10% and 65% of the chord. This test was performed at a Reynolds number of 7.189×10^5 (M=0.102) for Angles-of-Attack (AOA) ranging from 0 deg to 24 deg with the slots open. An examination of the power spectra revealed that there was significant amplification of the boundary layer aft of the leading edge synthetic jet array. The cause of the amplification is believed to be the result of sound radiating from the slots located at the leading edge of the airfoil. The two dimensional cavities/slots cut into an aerodynamic surface generate self-excited acoustic radiation which is a function of the slot dimensions, the type of boundary layer and Mach number [1]. The phenomena witnessed in the current experimental results is similar to the rectangular cavity flow problem explored extensively by Samimy et al. [2] and Rowely [3] but differs in that the leading edge synthetic jet arrays are in fact resonator cavities. For resonator cavities, the frequency of radiation is governed by the distance between the cavity lips and not the depth/height and width/diameter of the cavity. In addition to the cavity generated aerodynamic noise, it was also noted that in the frequency spectra, a peak less than 500 Hz was present in all the power spectra at each AOA. Parker, [4, 5] presented experimental results on the acoustic resonances excited between cascades of parallel plates; the resonances were excited by vortex shedding from blunted trailing edges. An analytical exposition of this phenomenon is discussed in detail by Runyan et al. [7, 8], Koch [9] and Lucas [6]. It is our assertion that this is the cause of the persistent tone seen at different angles-of-attack and is due to the geometry of the test section configuration. In Section 2 of the note the experimental setup will be described in detail, followed by a discussion on the two phenomena in Section 3 and the conclusions in Section 4.

II. Experimental Setup

A. The High Speed Wind Tunnel

The Clarkson University wind tunnel shown in Figure 1 is an open circuit wind tunnel equipped with a Howden Buffalo 179 HP AXIVANE Fan. The tunnel is capable of producing 70 m/s airflow velocity at the test section (M0.205 at sea level under standard day conditions). The test section is 121.92 cm x 91.44 cm x 152.4 cm.

Figure 1: Wind Tunnel Test and Test Section

The test section floor and ceiling are made of medium density fiber wood; the removable floor and ceiling make it possible to modify the test section for specific aerospace and automotive research testing. The side walls of the test section are made of clear Plexiglas for optical access. For turbulence reduction, there are two conditioning screens and settling chamber up stream of the test section. Currently, the wind tunnel is not treated acoustically.

B. NACA 0015 Airfoil

A NACA 0015 airfoil fitted with a simple trailing edge flap was used in these tests. The airfoil is made of 6061 Aluminum with a chord of 30 cm and a span of 40 cm with a maximum thickness of 4.5 cm at 30 % of the chord. Aluminum 6061l was selected because of its manufacturability and light weight. The wing is designed to accommodate two arrays of synthetic jet actuators and instrumentation; specifically the wing has a cavity which provides access to the model instrumentation. There are eleven holes of 1 mm diameter; eight on the main element and three on the trailing edge flap. These holes are for the pressure transducers.

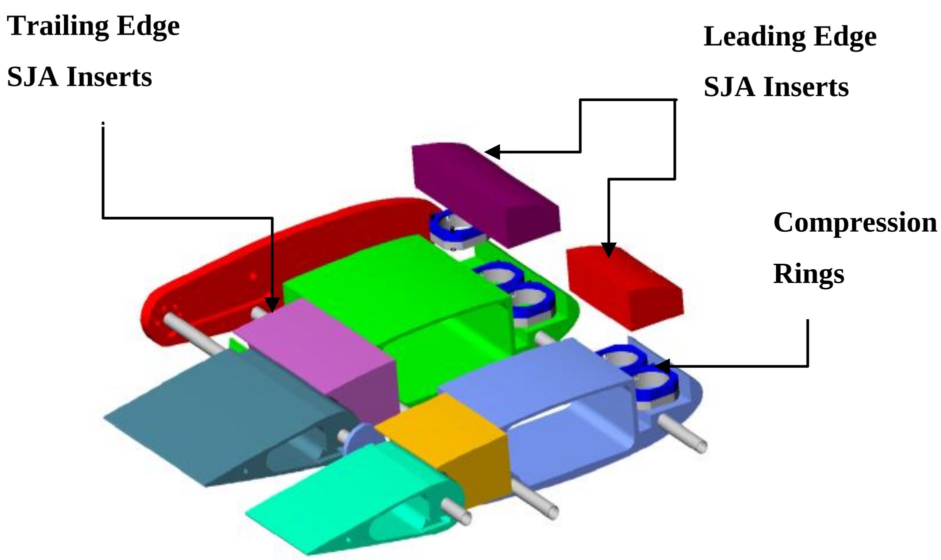

Figure 2: Partial Wing Assembly

The wing assembly is modular, and is comprised of 6 pieces in the spanwise direction that are fastened together by three brass tubes and held together by three corresponding threaded compression rods (Figure 2). The flap is located at 75% of the chord; this flap location was selected because is recommended in McCormick [Ref] that the optimal flap chord ratio is 0.25. The trailing edge flap is connected to the main element via one the brass tubes and threaded compression rod. To ensure that the flap angle is the same throughout the span of the wing, the flap segments are held in place with pins. The brass tube and compression rod is located at the center of the leading edge of the flap that serves as the hinge line for the trailing edge flap. The flap is fabricated with holes that allow for setting specific flap angles. of 0, 5, 10, 20, 40 degrees as is seen in Figure 3. The flap has a wire channel on the right side, which allows for a wire harness access from the main element cavity to the cavity inside the flap.

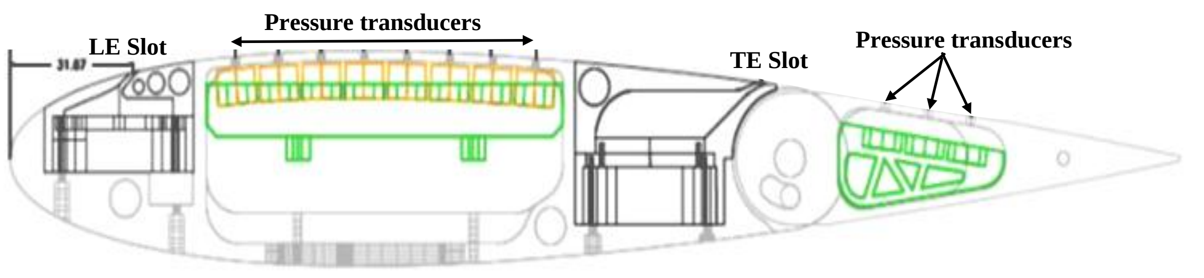

Figure 3: Wing Schematics

The wing is mounted to the six axis force balance by a strut made by aerodynamic tubing. The strut is attached to a support plate that fastened to the pressure side of the airfoil. The strut also accommodates wire harnesses for the synthetic jet actuators and the pressure transducers that are embedded inside the wing. In order to ensure two-dimensional airflow about the wing, false walls are installed in the test section. One of the walls is made of Plexiglas for optical access for image capture and the other is made of plywood. The inside of the plywood wall is painted black as is the wing in order to prevent any reflection during the flow visualization process and PIV measurements.

C. Aerolab Six Axis Force Balance

A six axis internal force balance systems is used to obtain force and moment measurements on the airfoil and control pitch (AOA) and yaw of the airfoil model. The force balance system includes a specially machined load cell fitted with strain gauges.

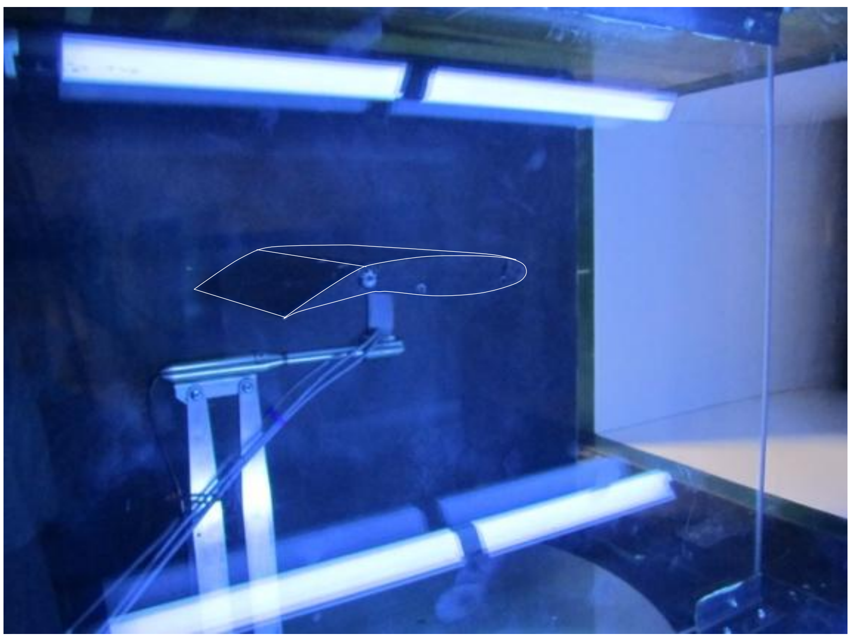

Figure 4: Wing mounted on Force Balance

The tip of the sting is located at the center of the test section. The motion of the force balance is computer controlled via Labview codes. The calibration of the force balance is accomplished by applying weight to a calibration bar and measuring the output voltages. A calibration matrix aids in the conversion of voltages. The force balance is rated at 444 N in the normal direction, 111 N in the axial or drag direction and maximum side force of 333 N. The pitch, yaw, and rolling moments are rated, respectively, at 22.5 Nm, 16.9 Nm, and 16.9 Nm,

D. Dynamic Pressure Sensors

The wing is instrumented with eleven equally spaced PCB 103B01 pressure transducers placed in series along the chord of the wing. There are eight transducers located on the main element of the airfoil and three transducers located in the flap. The pressure transducers are signal conditioned and powered by three 4 channel PCB Model 482 C Series Signal Conditioners. Each transducer comes with a calibration chart and a sensitivity value. The transducer locations are listed in Table 1.

Table 1 Pressure transducer locations

Sensor Serial Number	**X/C**
5413	0.1826
5414	0.2293
5415	0.2660
5416	0.3026
5417	0.3393
5418	0.3760
5419	0.4126
5420	0.4493
5421	0.7483
5422	0.7846
5449	0.8210

The pressure transducers have a resonant frequency of 13 kHz; the data is sampled at 8 kHz. In addition to the pressure sensors, an accelerometer is added to the instrumentation package in order to assist in the investigation of flow induced vibration caused by bluff body vortex shedding at high AOA, as well as, for providing the means to obtain the true angle of attack in the ‘wind on’ condition; this is needed because of the deflection of the sting when due to the aerodynamics forces acting on the wing.

E. Real Time Data Acquisition and Control Hardware & Software

The DS1103 PPC Controller Board is specially designed for the development of high speed multivariable digital controllers and real time simulation. It is a real time control system based on a Power PC processor. The dSPACE 1103 ACE Kit provides a means to rapidly develop controller and implement control system designs and assess their performance. dSPACE has been used extensively in closed loop active flow control. The DS1103 board has a bus frequency of 133 MHz. It also has 16 multiplexed channels equipped with 4 sample and hold analogue to digital conversion (ADC) with 16 bit resolution and 4 channels each equipped with one sample and hold ADC. The schematic in Figure 5 show the integrated system used in the experiment. Included in the instrumentation package is a wire harness from the AEROLAB force balance which allows obtaining lift and drag time series from the force balance strain gauges.

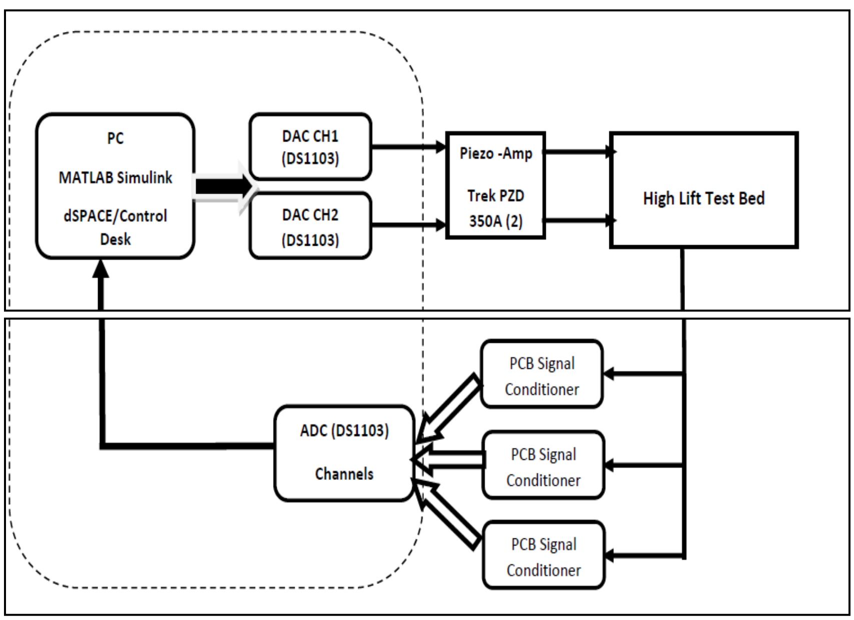

Figure 5 Schematic of experimental setup

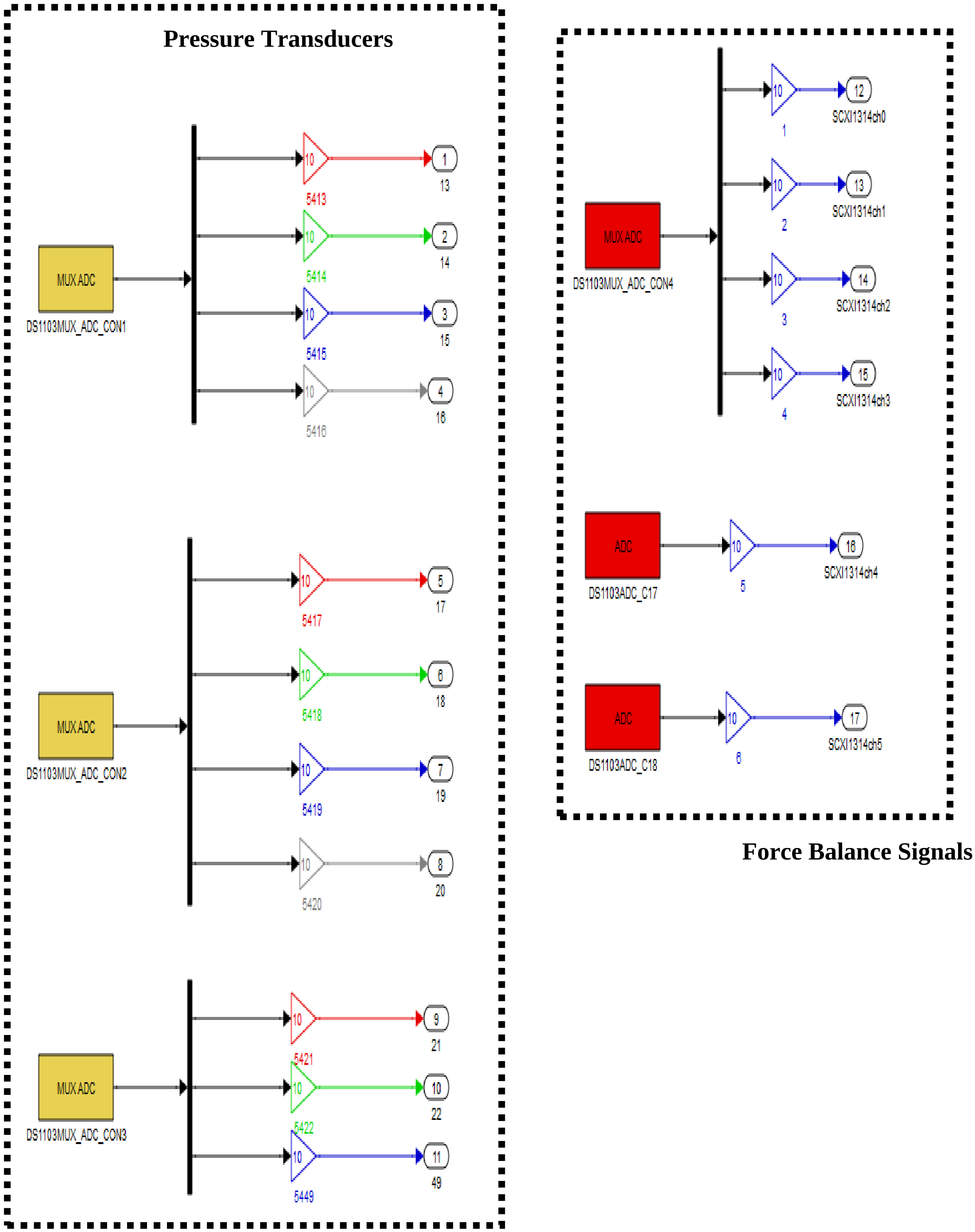

Figure 7: Data acquisition block diagram

A Control Desk is used to collect simultaneous data along with the control algorithm development. The data for the aerodynamic system identification experiments is collected via a Simulink block system that is constructed to read the pressure transducer and force balance signals, as shown schematically in Figure 6. On the left hand side, the three MUX-ADC blocks are a special dSPACE block set in the Simulink library. These blocks contain the channels which the transducers and force balance signals, on the right hand side, are plugged into. These blocks perform the analogue to digital conversion; the practitioner can collect the data. Once in the Control Desk environment, there is are many available options for displaying the real time signals as well as data acquisition configurations.

III. Results and Discussion

A. Flow Noise Generated by Open Leading Edge SJA Slots

Acoustic radiation is generated by the open leading edge SJA cavities, and are quite pronounced at 4, 6 and 8 degrees AOA. This is evident from the foremost pressure transducer PCB SN 5413, located immediately downstream the leading edge SJA slot, as shown in Fig. 7.

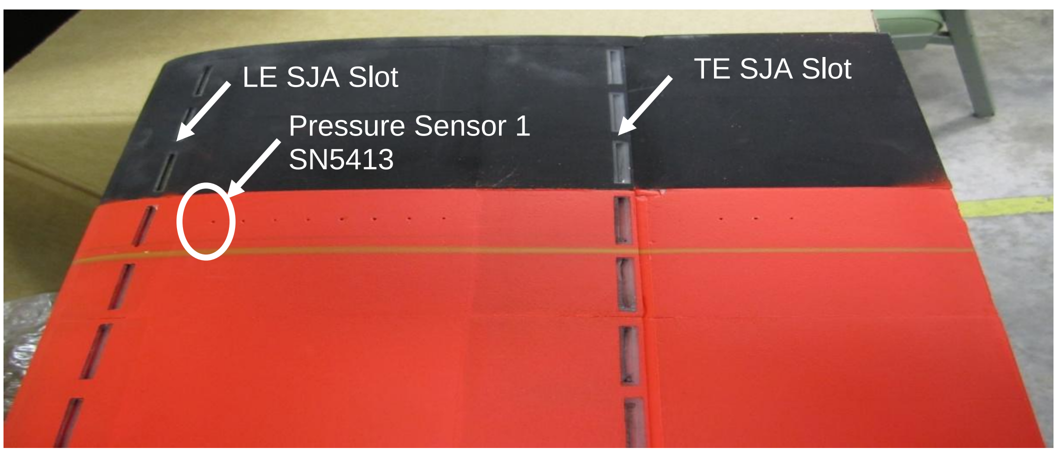

Figure 7: Location of Leading Pressure Transducer

Figures 8-11 show the development of cavity modes with angle of attack, the following provides a more detailed description of the observations. In Figure 8 it is seen there is a frequency peak that fluctuates about 423 Hz in all of the power spectra plots. At 4 degrees three distinct frequency peaks spaced at 825.5 Hz, 1238 Hz, and 2100 Hz appear, as illustrated in Figure 8. At 6 degrees, the 825.5 Hz peak decrease in amplitude and peak at 590.5 Hz appears; the peak at 1238 Hz shifts to 1014 Hz and is considerably more pronounced. It is also observed that 1438 Hz appears and the rounded 2.1 kHz remains. At 8 degrees, it is conjectured that the 590.5 Hz shift to 715.5 Hz and is very sharp; the other modes is suppressed with the increase in AOA, see Figure 9. Typically, a cavity that is exposed to a fast moving boundary layer will develop resonant modes. As the velocity increases these modes establish a standing wave pattern similar to that which is developed in open cavities. Beyond the critical velocity, the frequency peaks will lose their sharpness. In this case, the velocity was held constant and the AOA was changed; increasing the AOA had the same effect as velocity in that the peaks at 4 degrees rounded and at 8 degrees, certain peaks sharpen and increase in amplitude, indicating the presence of coherent structures. As with cavity flows, once the resonant modes reach the equilibrium limit cycle state often indicated by a large amplitude frequency peak, there are reflections that take place inside the cavity and spill out into the shear layer; the spillover results in the propagation of the acoustic radiation downstream of the open SJA slots. Flow phenomena of this nature are often characterized by the Rossiter equation [2]. However, a closer evaluation of the data presented revealed that there is not enough resolution in the AOA to show a trend. Furthermore, the literature reveals that there are no Rossiter like relationships, neither *open cavity* or *closed-box* below a Mach number of 0.4 [10]. There are no published results that address the change in AOA. All the frequency peaks to the right of the 423 Hz are due to the open slots, this is confirmed by the fact that at 10 deg all that remains is the frequency peak about 423 Hz, as illustrated in Figure 9. Note that as the AOA increases, spectrum becomes increasingly more broadband. The broadband response is an indication that the flow over the wing is turbulent and the pressure gradient is becoming more severe with the increase in AOA.

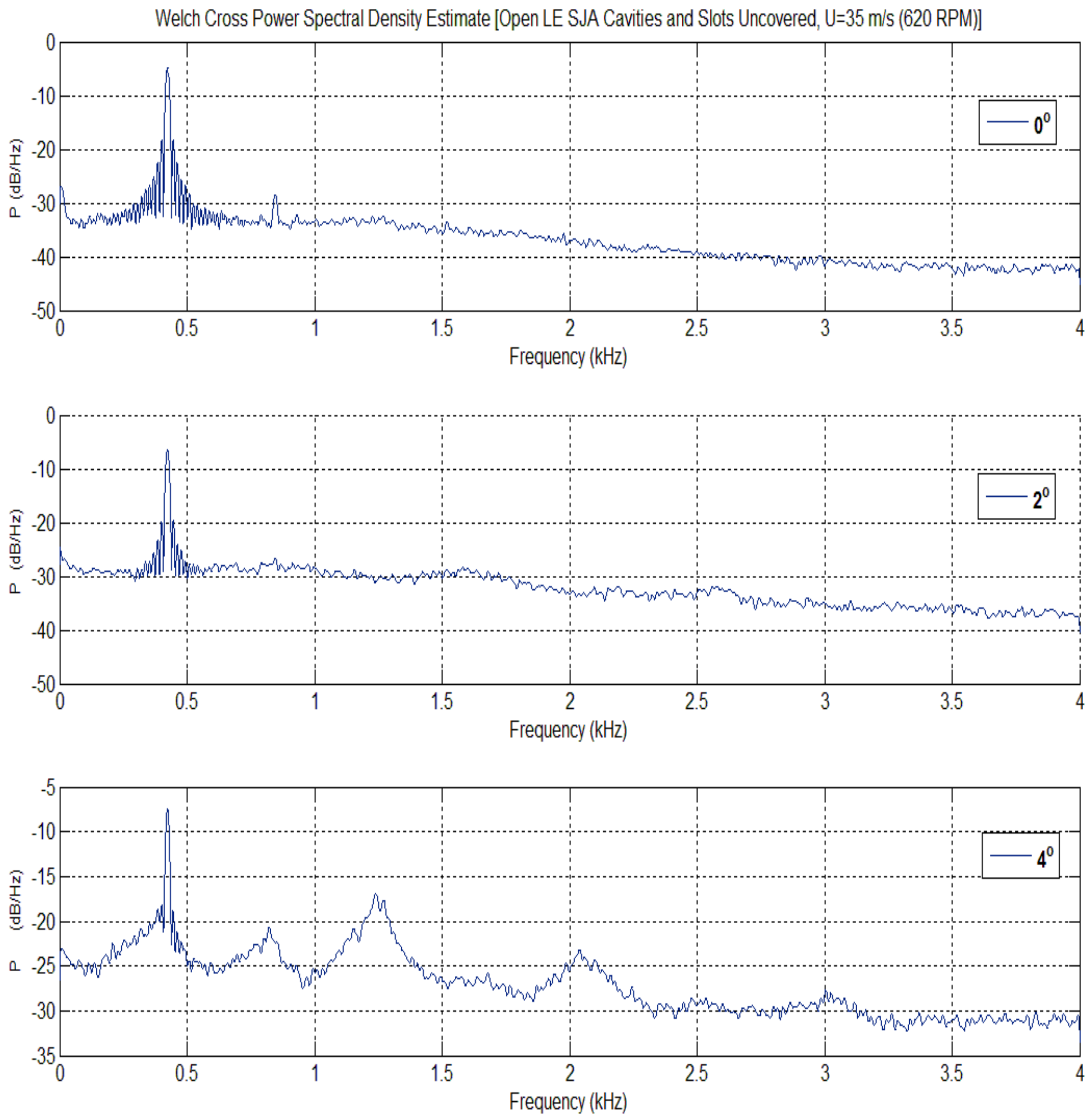

Figure 8: Pressure Spectra for AoA=0-4 degs

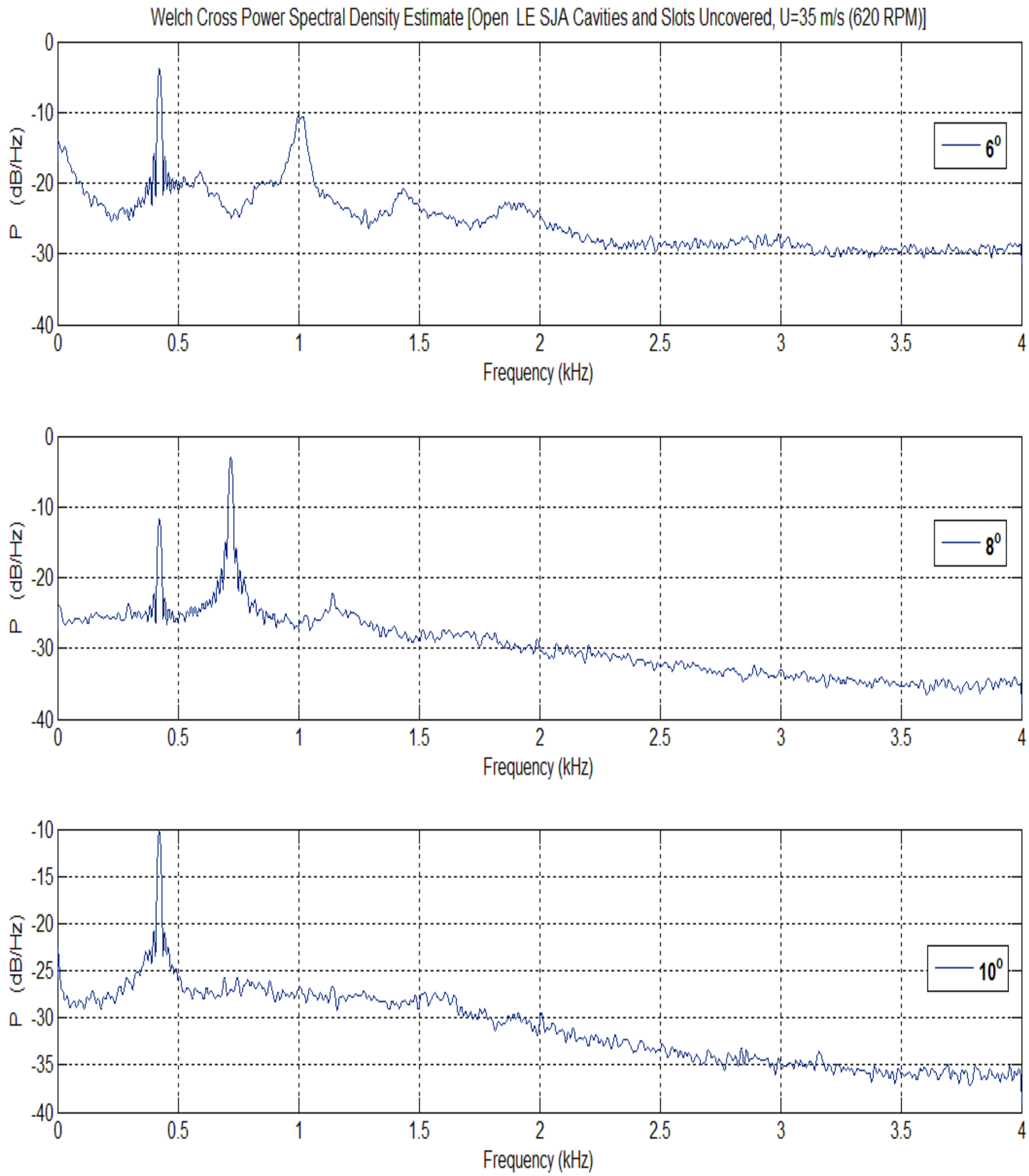

Figure 9: Pressure Spectra for AoA=6-10 degs

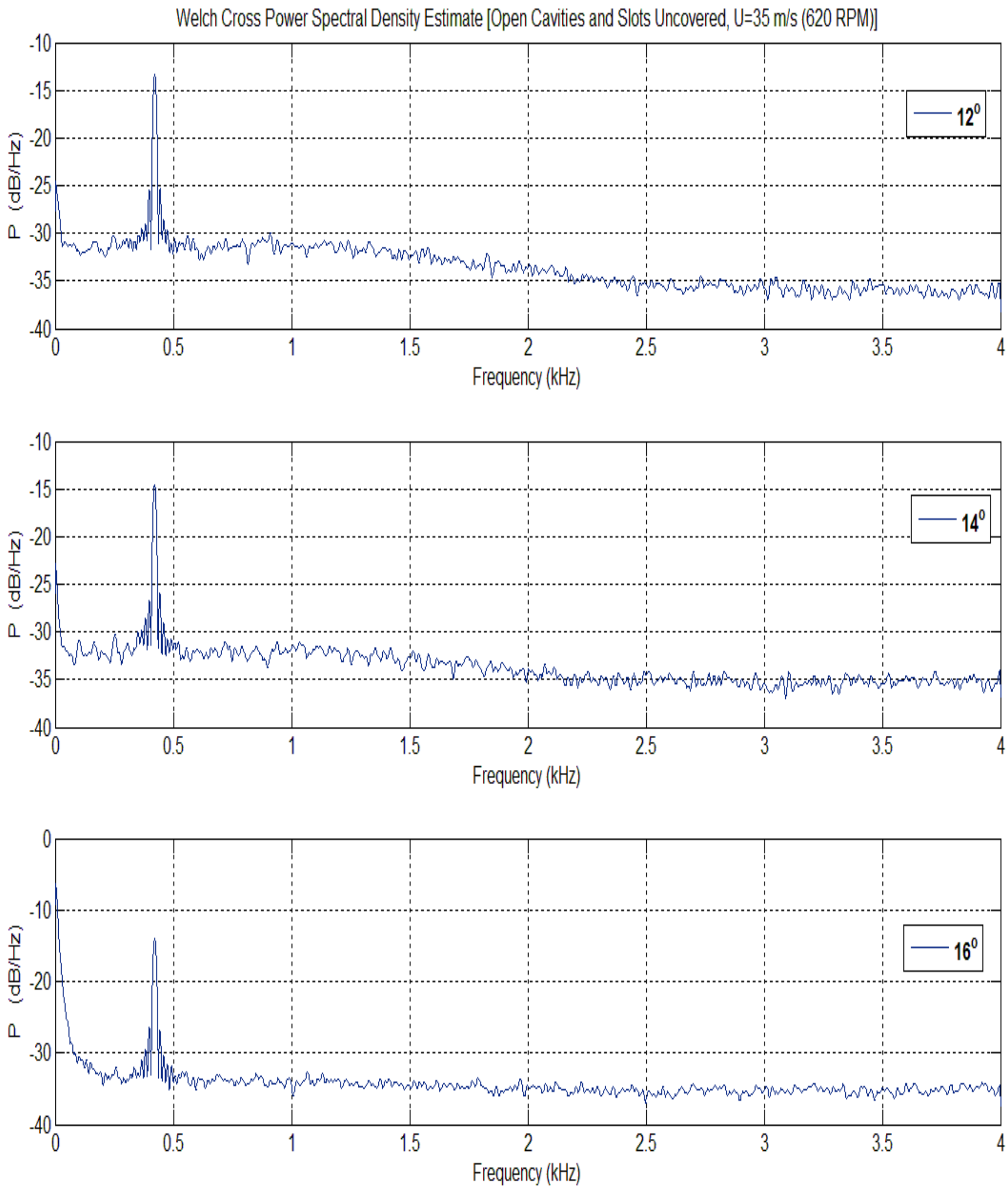

Figure 10: Pressure Spectra for AoA=12-16 degs

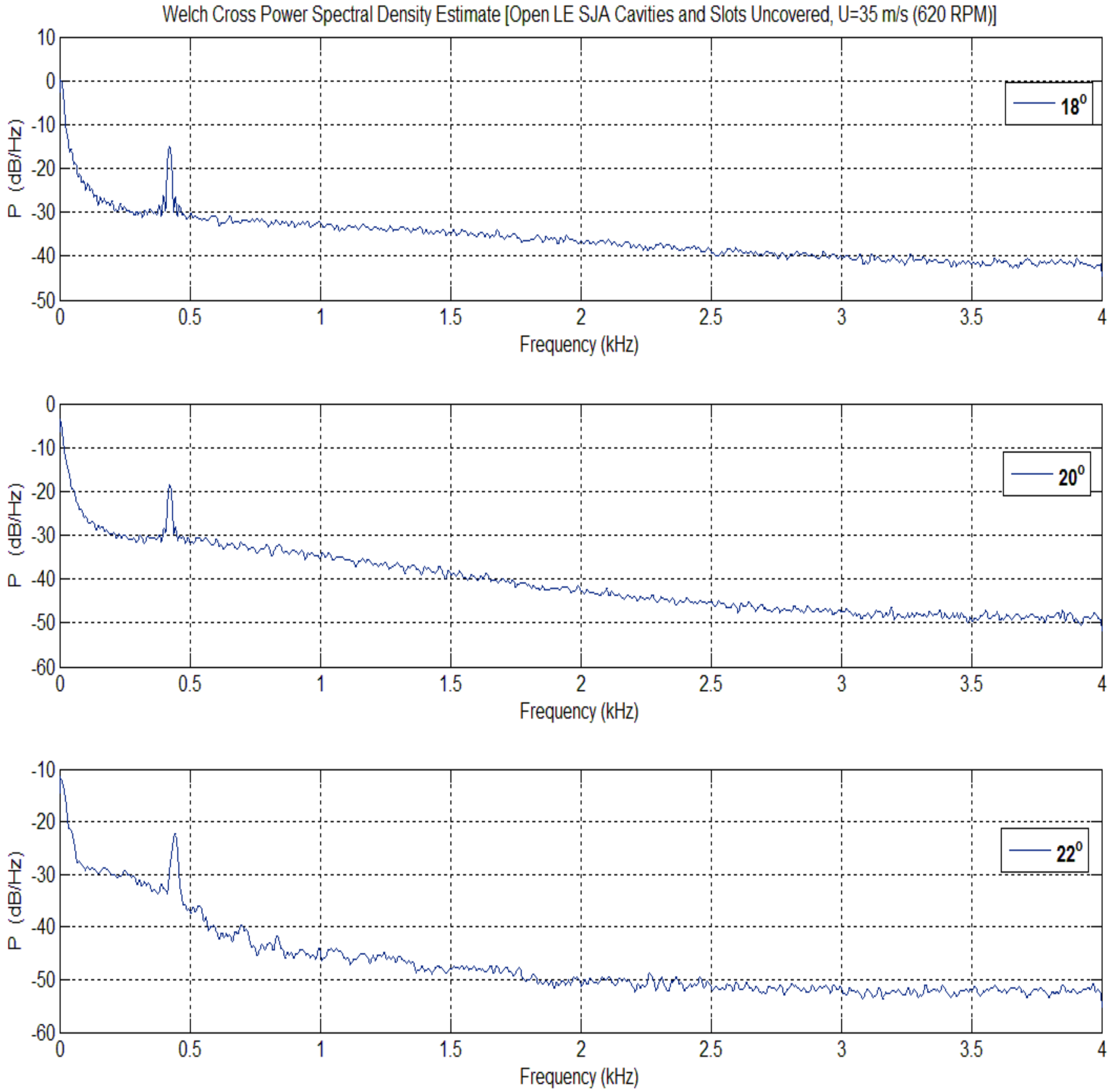

Figure 11: Pressure Spectra for AoA=18-22 degs

Above 10 degrees, the cavity tones are also not present in the spectra, and the flow progresses toward stall indicated by the broadband frequency response shown in Figures 10 and 11 at angles of attack 12 through 22 degs. The experiment was repeated with the slots covered. The corresponding pressure response spectra for AoA of 4 and 12 degrees are shown in Figures 12 and 13. Figure 12 shows that the rolling peaks observed in the first experiment for the airfoil at AoA of 4 degrees, are no longer present; Similarly, Figure 13 shows that for AoA= 8 degrees the peak at 715.5 Hz has been removed.

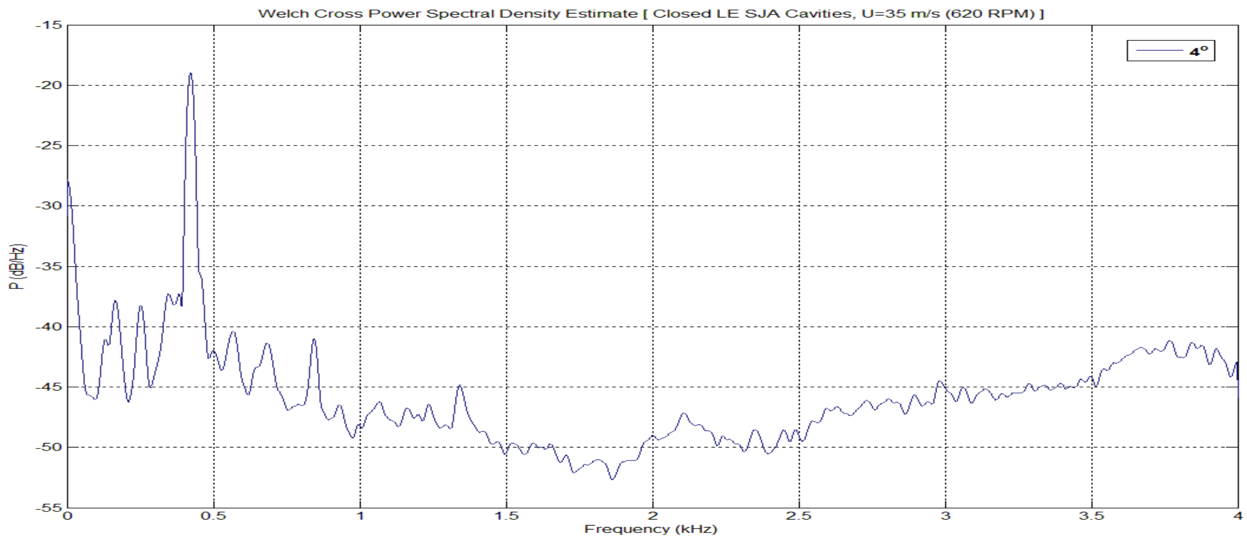

Figure 12: Pressure Spectra at 4 deg AoA, Slots Closed

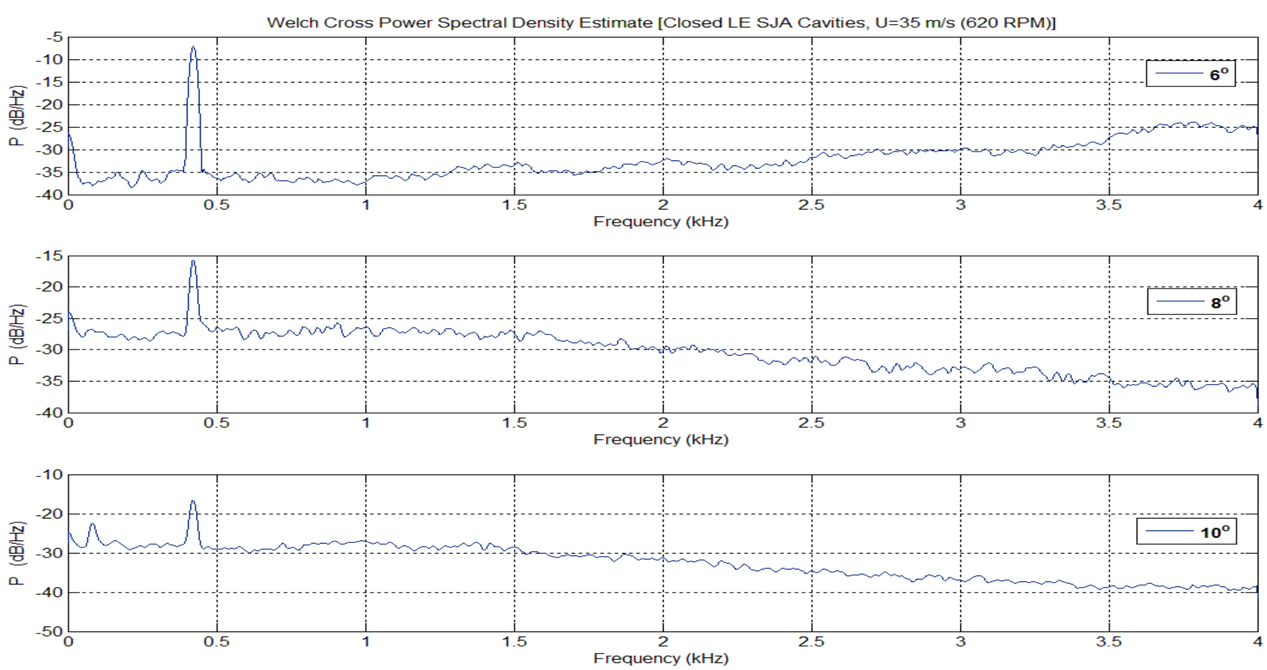

Figure 13: Pressure Spectra 6-10 deg AoA, Slots Covered

B. Wind Tunnel Resonance

In the power spectra presented above, there is a tone that persists from 0 to 22 degrees AOA. The 423 Hz tone at flat pitch is nearly 3 times that of the blade passage frequency of the lift fan at 165 Hz and is too low to be a shear layer mode. It is conjectured that that this frequency is coming from the wind tunnel test section configuration. Specifically, the test section configuration at Clarkson University is such that each false wall is parallel to the sides of the wind tunnel test section hence the walls form a converging duct; the false walls form a flat plate cascade. In general, a typical rectangular or unstaggered cascade system, such as the one represented in Figure 14, is defined as an array of parallel or annular plates in the plane of a gas or any compressible fluid, and is generally assumed to be subsonic. The on-coming flow produces a wake at the trailing edge of the false wall cascade which generates an acoustic resonance. These acoustic resonances can interact with the structural resonances of the cascade itself.

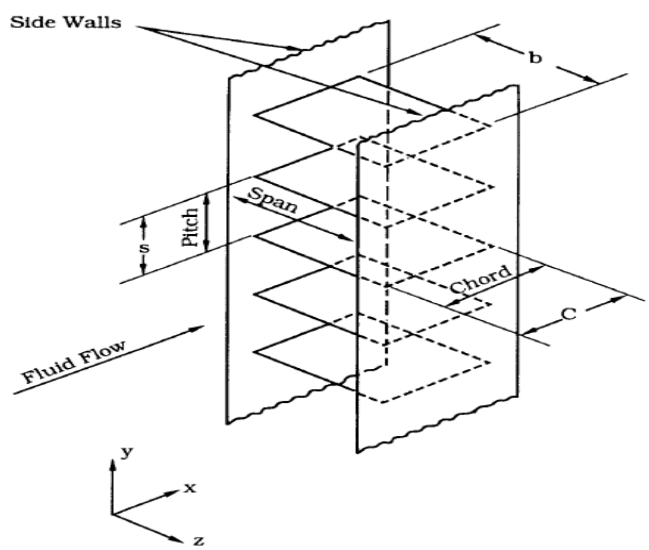

Figure 14: Flat Plate Cascade [4]

Runyan et al. [8] provide a derivation for an airfoil between solid, reflecting walls using the subsonic integral equation for lifting surface theory. It was noted that the kernel of the integral equation relating lift pressure to the downwash boundary condition became infinite at frequencies equal to:

$$f_{res} = \frac{a_\infty}{2h}\sqrt{1 - M^2}\,(2n - 1), n = 1,2,3 \ldots \qquad [1]$$

where a_∞ is the speed of sound in air and h is the height of the tunnel. When *n=1,* this is the lowest frequency in the series and it is termed the tunnel resonance frequency. The diagram given by Parker (Figure 14) schematically represents h as the separation distance between plates; but Runyan indicates that h is the height of the tunnel. Parker [4] and Lucas et al. [6] indicated that the separation between the plates is the correct parameter to be substituted into Eq. 1, and the acoustic resonant frequency numerically obtained is 423.107 Hz. Note that the experiments reported a 423 Hz at 0 deg AOA. For flat plates with a blunt trailing edge (as is the case for our test section configuration) the vortex shedding frequency will lock on to a higher acoustic resonance. If the velocity were to increase further, the cascade vortex shedding frequency may jump to the next highest acoustic resonance frequency. However, the velocity is held constant at 35 m/s but the AOA is changing. As the AOA is increases the frequency peak tends to exhibit frequency creep to the left and right of 423 Hz in small magnitudes. When the AOA becomes greater than 15 degrees, the Parker mode increases from 423 Hz to 425 Hz. At 22 degrees, the frequency jumps to 440Hz (Figure 24). It is possible that the bluff body wake is interacting with the cascade wake which causes an amplification of the acoustic mode, hence the variation of the Parker mode with AOA, particularly in the post stall regime.

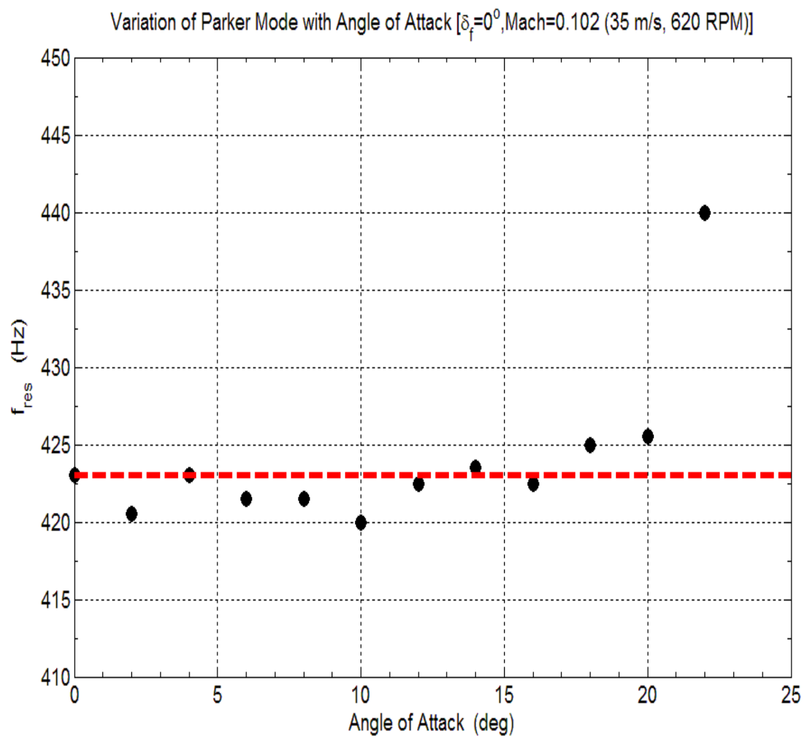

Figure 15 Parker mode variation with angle of attack

IV. Conclusions

This technical note discusses some flow induced noise considerations and lessons learned when performing wind tunnel testing for the purposes of aerodynamic frequency domain identification. Although not often discussed in the literature, the topics presented here should not be overlooked when preparing a wind tunnel test setup. When one is considering measuring pressure fluctuations for frequency domain identification, it is good practice to perform an acoustic assessment of the test section before setting an experiment and after. In this manner, the research has an understanding of the acoustic environment before the study of an external aerodynamic flow begins; this includes identifying Parker Modes (both frequency and velocity), especially if false walls are part of the test section set up. It also is recommended that the trailing edges of the false walls be made sharp. Alternatively, the wing section can span the height of the tunnel eliminating the need for the application of false walls. The existence of a Parker Mode acoustically contaminates the experimental data as does uncovered actuator slots. When not in use, all slots should be covered especially during preliminary aerodynamic frequency domain identification procedures. In addition to the above recommendation, further experiments and analysis are required in order to establish a mathematical relationship between the changes in the cavity acoustic modes with respect to AOA. An initial examination of other frequency domain data collected at different velocities reveal that the acoustic radiation from the leading edge slot/cavities have an effect on the boundary layer, in particular the shear layer, as such it requires further experimental investigation to determine the impact of the passive acoustic excitation on the closed loop active flow control studies.

Acknowledgments

Support for this research has been provided through a grant from the Air Force Office Scientific Research. The authors gratefully acknowledge the Air Forces support for this project.

References

[1] Krsihnamurty, K. ‘Sound Radiation from Surface Cutouts in High Speed Flow’, PhD Thesis, GALCIT, 1965.

[2] Samimy, M. et al, ‘Feedback Control of Subsonic Cavity Flows using reduced-order models’ Journal of Fluid Mechanics, Vol. 579, pp 315-346, May 2007.

[3] Rowley, Clarence, W. ‘Modeling, Simulation and Control of Cavity Flow Oscillations’ PhD Thesis, California Institute of Technology, 2002.

[4] Parker, R. ‘A note on frequency ratios for acoustic resonances of flat plate cascades with

Spacing less than half the longitudinal wavelength', Journal of Sound and Vibration, Vol. 86, Issue 4, pp 594-596, 1983.

[5] Parker, R. 'Resonance Effects in Wake Shedding from Parallel Plates: Calculation Resonant Frequencies', Journal of Sound and Vibration, Vol. 5, Issue 2, pp 330-343, 1967.

[6] Lucas, M.J., Noreen, R, Southerland, L.D., Cole, J and Junger, M., 'The acoustic characteristics of turbomachinery cavities', NASA CR-4671, May 1995.

[7] Runyan, Harry L., Watkins, Charles, E., 'Condsiderations on the Effect of Wind Tunnel Walls on Oscillating Air Forces for Two-Dimensional Subsonic Compressible Flow', NACA Report 1150, 1953.

[8] Runyan et al, 'Theoretical and Experimental Investigation of the Effect of Tunnel Walls on The forces on an Oscillating Airfoil in Two-Dimensional Subsonic Compressible Flow', NACA Report 1262, 1956.

[9] Koch, W., 'Resonant acoustic frequencies of flat plate cascades', Journal of Sound and Vibration, Vol. 88, Issue 2, pp 233-242, 1983.

[10] Clark, R.L., Kuafman II, L.G. and Maciulaitis, A., "Aeroacosutic Measurements for Mach Mach 0.6 to 3.0 Flow Past Rectangular Cavities," AIAA -80-0036, Jan. 1980.