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Abstract

The Structural Dynamics Division at NASA Langley Research Center has started a wind tunnel activity referred to as the Benchmark Models Program. The primary objective of the program is to acquire test data that will be useful for developing and evaluating aeroelastic type CFD codes currently in use or under development. The program is a multi-year activity that will involve testing of several different models to investigate various aeroelastic phenomena. This paper describes the progress achieved in testing the first model in the Benchmark Models Program. Experimental flutter boundaries are presented for a rigid semi-span model (NACA 0012 airfoil section) mounted on a flexible mount system. Also, steady and unsteady pressure measurements taken at the flutter condition are presented. The pressure data were acquired over the entire model chord located at the 60 percent span station.

Introduction

Unsteady aeroelastic computational fluid dynamic (CFD) codes require experimental data to validate the accuracy of the analytical results. Therefore, the Benchmark Models Program was initiated by the Structural Dynamics Division at NASA Langley Research Center. The primary objective of this program is to acquire aeroelastic test data useful for code refinements and validation. In some instances, experimental results may also be required to verify that the right types of flow phenomena are being mathematically modeled.

The Benchmark Models Program has identified several aerodynamic configurations to be tested in the NASA Langley Transonic Dynamics Tunnel (TDT). Some configurations will be rigid models for testing on a flexible mount system referred to as the Pitch and Plunge Apparatus (PAPA). This paper describes the progress achieved in testing the first of these models.

Nomenclature

Alpha	angle of attack, deg
C_p	pressure coefficient
C_p'	magnitude of pressure coefficient
c	chord, ft
f	frequency, Hz
M	Mach number
q	dynamic pressure, psf
x	distance from leading edge, ft

Apparatus

Wind Tunnel

The wind-tunnel tests were conducted in the Langley Transonic Dynamics Tunnel (TDT)[1]. The TDT is a continuous flow, single return wind tunnel with a 16-foot square test section (with cropped corners) having slots in all four walls. It is capable of operating at Mach numbers up to 1.2 and at stagnation pressures from near vacuum to atmospheric. The tunnel is equipped with four quick-opening bypass valves which can be used to rapidly reduce test-section dynamic pressure and Mach number when an instability occurs. Although either

air or a heavy gas can be used as a test medium, only air was used for the present tests.

Model

The model has a rectangular planform with a NACA 0012 airfoil. A photograph of the model mounted in the TDT test section is shown in figure 1. The model has a semi-span of 32 in. and a chord of 16 in. The model is mounted outboard of a large fixed splitter plate. The flexible mount system, which allows pitch and plunge motion of the model, is located behind the splitter plate.

Figure 1. NACA 0012 airfoil model.

The model was designed to allow installation of 80 pressure transducers for measurement of unsteady wing surface pressures. Forty of the transducers are located at the 60 percent span station, and forty at the 95 percent span station near the wing tip. The orifice and corresponding pressure transducer locations for each chord are shown in figure 2. During the first test of this model in July of 1990 only the forty pressure transducers at the 60 percent span station were installed. For the second test in January and February of 1991, an additional 40 pressure transducers were installed at the 95 percent span station.

Details of the model construction can be seen in the photographs of figure 3. The lower photograph shows that the model was fabricated in three sections. Each section was machined from solid aluminum stock. The sections were bolted together after the pressure transducers were installed. In the upper left photograph is an expanded portion of the mid section which shows some drilled holes in the edge of the section which were used for insertion of the pressure transducers. Two pressure transducers are shown next to the model. One of the pressure transducers is shown mounted in a brass tube. The brass tube is used to protect the transducer when it is inserted and removed from the model. The associated orifice holes for the pressure transducers are located about 1 inch from the inboard edge of the mid section. There are also four accelerometers in the model, one near each corner. These are mounted in pockets, one of which is shown in the photograph in the upper right of figure 3.
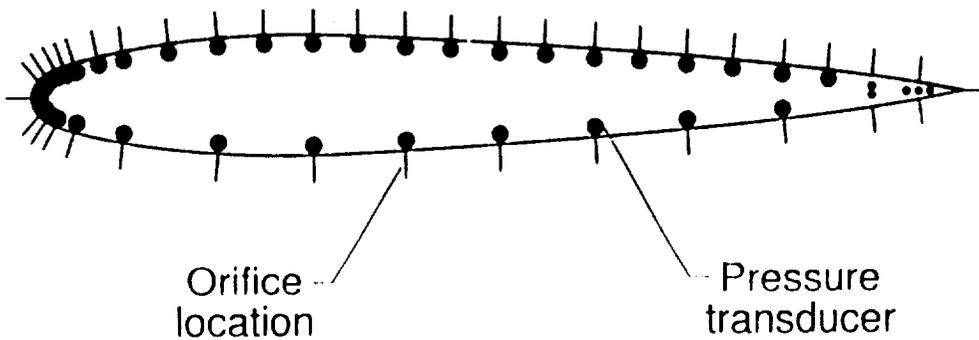

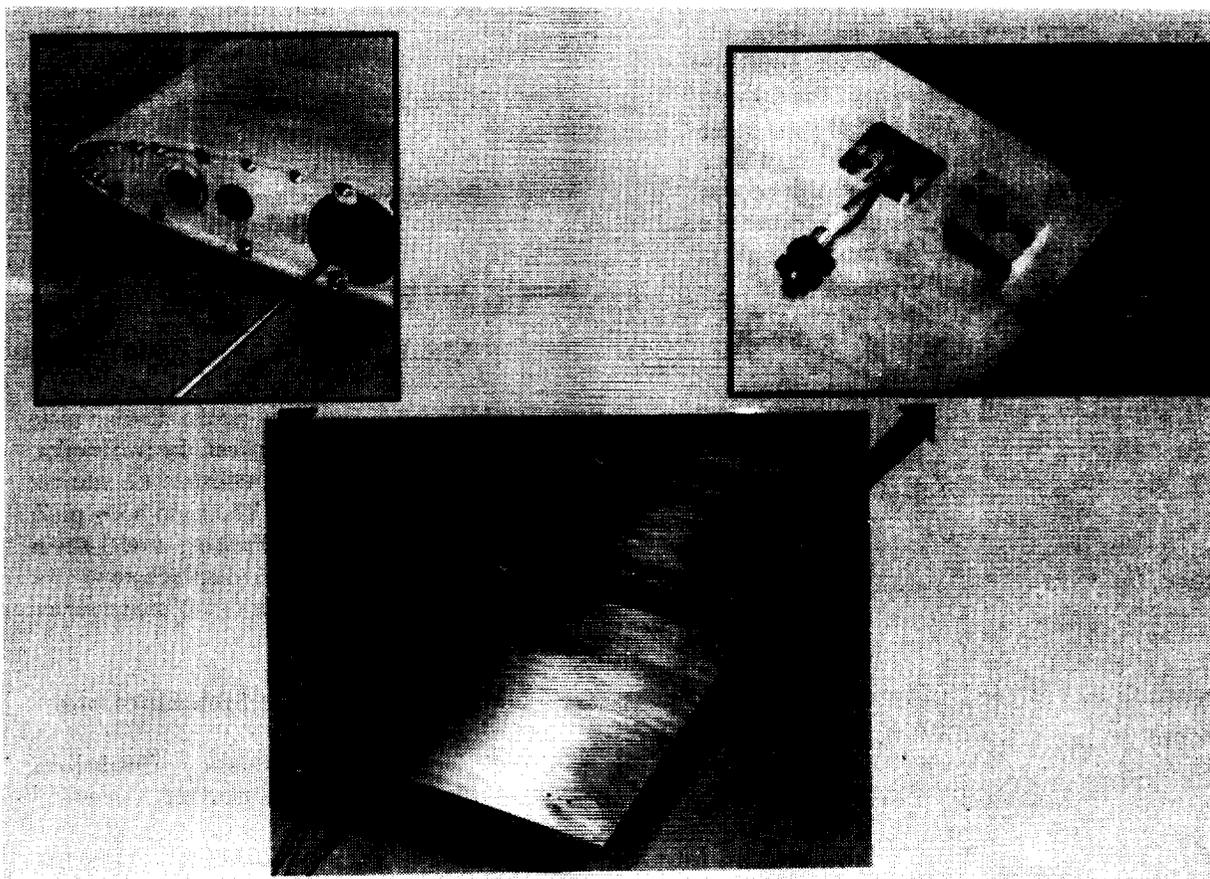Figure 2. Orifice and pressure transducer locations.

Figure 3. Model details.

Mount System

A photograph of the flexible mount system referred to as the PAPA (Pitch and Plunge Apparatus)[2,3] is presented as figure 4. It consists of a large moving plate supported out from the tunnel wall by a system of four rods and a centerline flat plate drag strut all with fixed-fixed end conditions. At the tunnel wall the rods and drag strut are attached to a plate mounted on a remotely controlled turntable so that the model angle of attack can be varied.
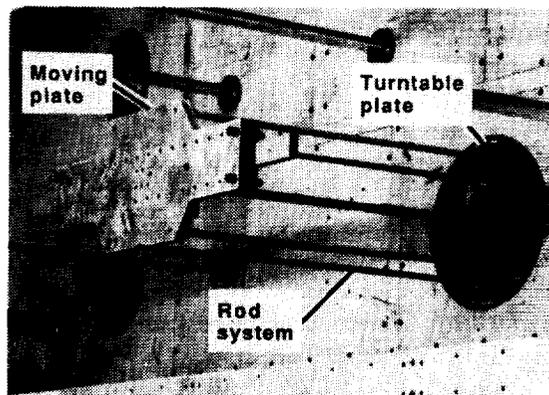

Figure 4. PAPA flexible mount.

The rods and flat plate drag strut provide linear elastic constraints so that the model will oscillate sinusoidally in pitch and plunge if excited. The oscillations are functions of the elastic restraints, the mass properties of the moving apparatus, and the aerodynamic forces on the model. Because the structural properties of the PAPA can be well defined, they can be easily modeled for flutter calculations. This makes the PAPA an invaluable tool in determining experimental data for correlation with analysis because disagreement between theory and experiment can be attributed directly to aerodynamic differences. The PAPA is instrumented with two strain gage bridges oriented to measure bending and torsional moments from which wing model plunge position and pitch angle can be obtained. In addition, the PAPA is instrumented with accelerometers to determine pitch and plunge motion.

The PAPA splitter plate served to separate the model from the test-section-wall boundary layer and is shown in figure 5. As shown, the splitter plate is suspended out from the test-section wall by struts that are about 40 inches long. The splitter plate is 12 feet long and 10 feet high. The center of the PAPA is 7

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feet from the leading edge of the splitter plate. The model is mounted on a small short pedestal attached to the support system moving plate. The pedestal is the portion of the moving system which protrudes through the opening in the splitter plate. The opening in the splitter plate is covered by an end plate at the base of the model. The circular end plate, which has a diameter equal to the model chord length, can be seen in the photograph of figure 5. The gap between the end plate and the splitter plate was made to be less than 0.1 inches. The end plate did not rub against the splitter plate.
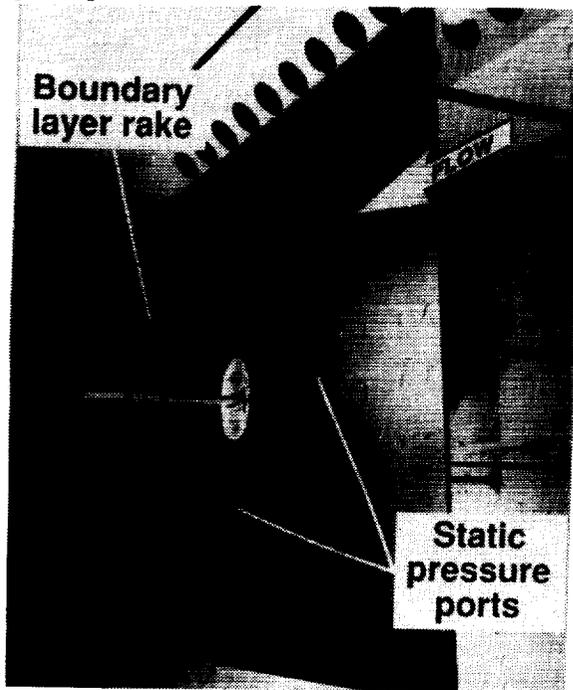

Figure 5. PAPA splitter plate.

The PAPA mount system rods and drag strut were enclosed in a fairing behind the splitter plate. Therefore, the wing and end plate were the only parts of the apparatus that were exposed to the flow in the test section.

The splitter plate is instrumented with 20 pressure transducers to measure splitter-plate surface pressures. These transducers are located above, below, in front of and behind the wing model. A boundary layer rake extending 5 inches into the flow is located above and behind the wing model as shown in figure 5. This rake is instrumented with 10 pressure transducers.

Structural Dynamic Characteristics

The first two natural modes of vibration for the NACA 0012 model/PAPA assembly are the wing-model rigid-body plunge and rigid-body pitch modes

respectively. The frequencies, damping and measured stiffnesses for these two modes are presented in table 1.

Table 1. Frequencies, damping and measured stiffnesses.

Mode	Frequency	Structural damping, g	Measured stiffness
Plunge	3.40 Hz	.0034	2697.2 lbs/ft
Pitch	5.18 Hz	.0016	2854.6 ft-lbs/rad

Ballast weights were positioned on the moving plate to eliminate inertia coupling between the two modes. The rigid-body plunge mode consists of vertical translation of the wing model. The rigid-body pitch mode consists of rotation of the wing model about the 50 percent chord. Modal displacements for corresponding generalized masses of 1.0 are presented in table 2.

Table 2. Modal displacements and generalized mass

Mode	Modal displacement leading edge	trailing edge	Generalized mass
Plunge	+0.4113 ft	+0.4113 ft	1.0 slug
Pitch	+0.4061 ft	-0.4061 ft	1.0 slug-ft^2

Results and Discussion

Flutter Boundary

The initial testing defined the conventional flutter boundary with the model at zero degrees angle of attack as shown in figure 6. The model is stable below the boundary and is unstable above the boundary.
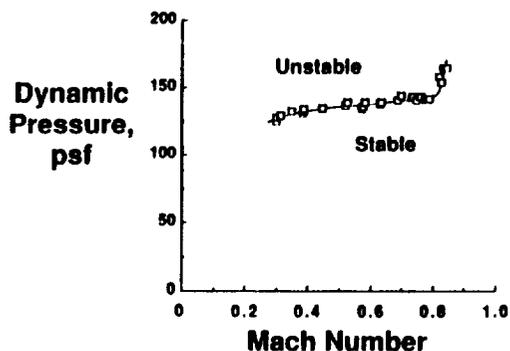

Figure 6. Conventional flutter boundary (alpha=0.0 deg).

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Stall Flutter Boundary

Model flutter dynamic pressure as a function of angle of attack for a fixed test Mach number of 0.78 is presented in figure 7. The results show that the dynamic pressure at which flutter is encountered increases as angle of attack is increased from zero up to about 4 degrees. At angles of attack above 4 degrees there is a rapid drop off in the dynamic pressure at which flutter is encountered. The flutter boundary for angles of attack greater than 4 degrees is associated with model stall conditions during a portion of the oscillation cycle.
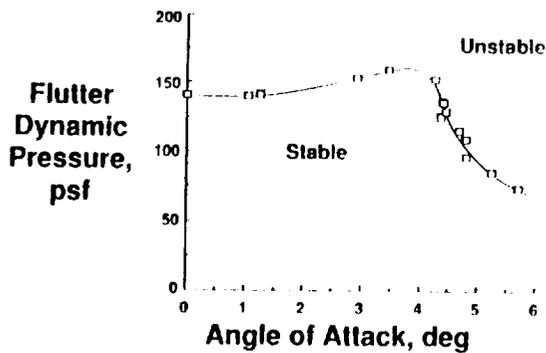

Figure 7. Angle-of-attack effects (M=0.78).

Transonic Plunge Instability

For this model a plunge instability region was observed in a narrow transonic Mach number range from about M=0.88 to 0.92 as shown by the cross hatched area of figure 8. As implied, the model motion was primarily in the plunge mode. At dynamic pressures greater than 140 psf, the model motions were so large that only the low Mach number side of the instability boundary could be identified. This plunge instability is believed to be related to a shock induced phenomenon.[4]
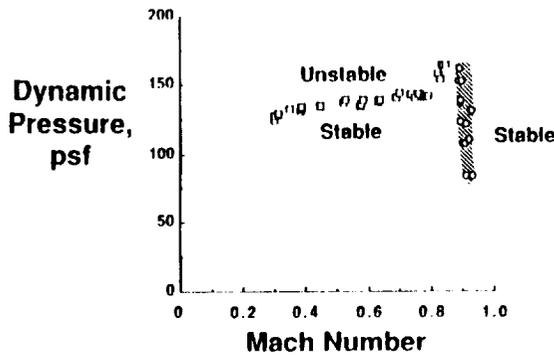

Figure 8. Transonic plunge instability region
(alpha=0.0 deg).

Steady Pressure Measurements

During a portion of the testing the model mount system was rigidized so that wing surface pressures could be measured with the model in a fixed position. These measurements are referred to as the steady pressure measurements. Steady pressure data were obtained across the Mach number range at the same dynamic pressure at which flutter had been encountered with the model mount system flexible. Tunnel test conditions at which these data were acquired are shown in figure 9. A sample pressure distribution for M=0.78 with the support system rigidized is presented in figure 10. The data in figure 10 are the time average or mean value of measurements for the model upper surface at the 60 percent span station from the leading edge (x/c = 0.0) towards the trailing edge (x/c = 1.0).
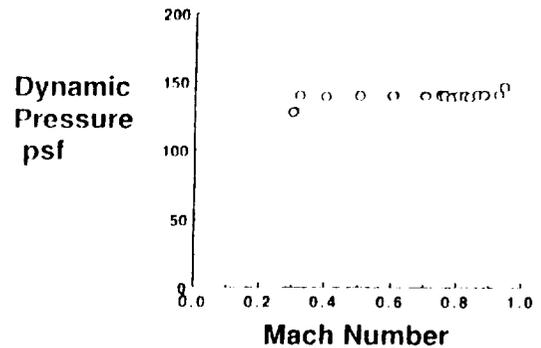

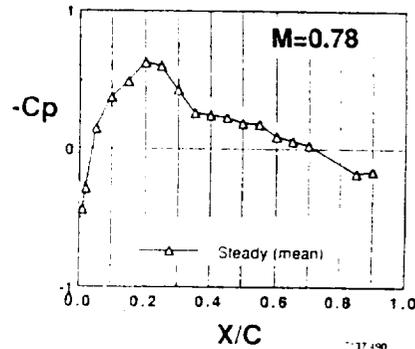Figure 9. Steady pressure measurement tunnel conditions.

Figure 10. Upper surface steady pressure distribution at 60 percent span.

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Unsteady Pressure Measurements

Unsteady pressure measurements refers to data obtained with the model mount system in the flexible mode. Unsteady pressure data were obtained at most of the flutter boundary and plunge instability test points.

The model pitch and plunge motion and a corresponding typical wing unsteady pressure recording for the conventional flutter boundary at M=0.78 are presented in figure 11. The flutter motion involved sinusoidal pitch and plunge motion at 4.15 Hz. The unsteady pressure trace is for the upper surface orifice at the 25 percent chord location. The unsteady pressure trace is essentially sinusoidal and in phase with the plunge motion and out of phase with the pitch motion.
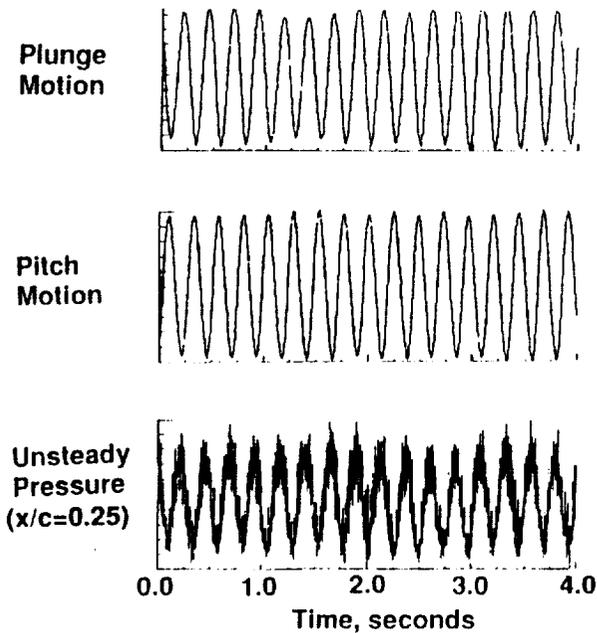

Figure 12. Unsteady pressure statistics (M=0.78).

Figure 11. Sample time histories (M=0.78, f=4.15 Hz).

Unsteady pressure mean values and the range of variation are presented in figure 12 for the wing upper and lower surfaces as a function of chord position x/c for the 60 percent semispan location. The data are from the same flutter test point as the data of figure 11.
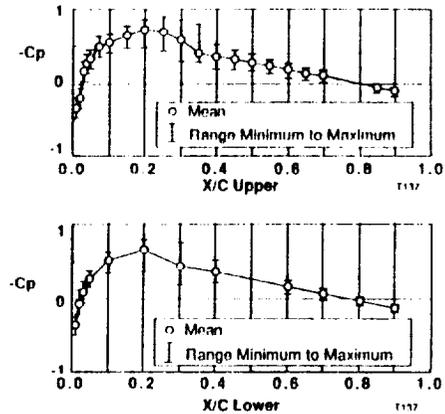
The magnitude of the unsteady pressure oscillations and the phase relationship to the plunge motion during flutter are presented in figure 13. Data are presented on the left for a subsonic Mach number of 0.39 and on the right for a transonic Mach number of 0.78. For the lower Mach number, the unsteady pressure coefficient magnitudes for both the upper and lower surface measurements are largest at the wing leading edge followed by a rapid decrease at locations further aft on the chordline. At the wing leading edge, the upper surface unsteady pressures are in phase with the plunge motion of the model and the lower surface pressures are 180 degrees out of phase. Further aft on the chordline, there is a slight increase in phase between the model motion and the measured unsteady pressure oscillations. For the higher Mach number test condition, the unsteady pressure oscillation magnitudes are about constant from the leading edge back to about the 30 percent chordline position. From the 40 percent chordline position back towards the trailing edge, the magnitudes are small. The large change in magnitude between the 30 percent and 40 percent chord is believed to be due to an oscillating shock at this location for this test condition. There is also a large shift in phase for the upper surface at the 35 percent chordline. This phase shift is believed to occur on the lower surface also. Unfortunately there is not a pressure measurement at this location on the lower surface to substantiate this phase shift.
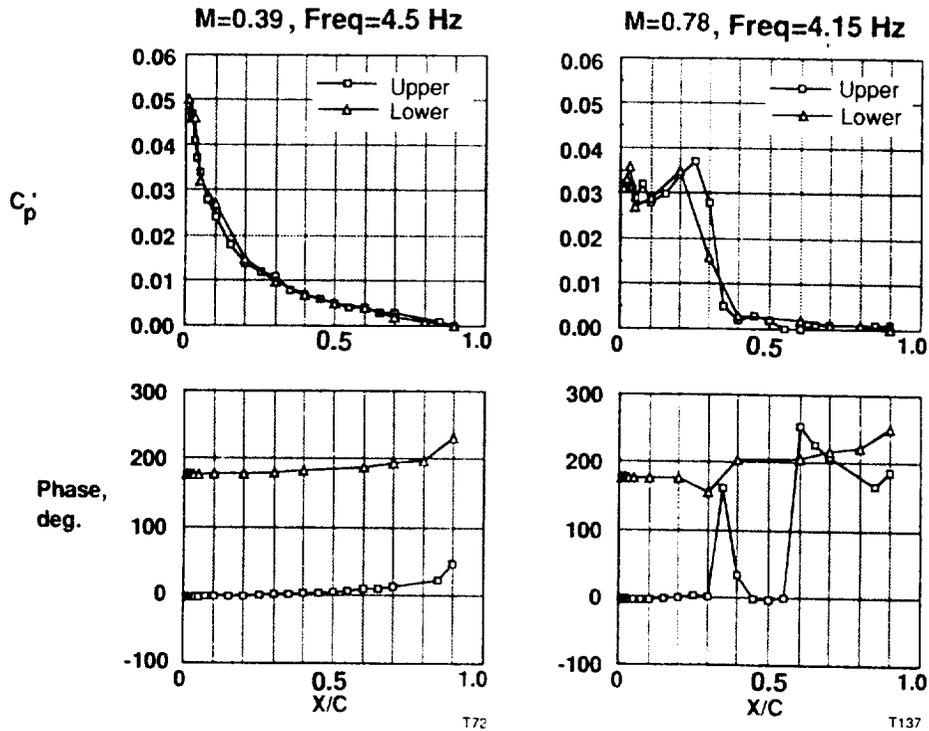
Figure 13. Unsteady pressure response.

Concluding Remarks

The Benchmark Models Program (BMP) has been initiated with the primary objective of obtaining data for aeroelastic CFD code development, evaluation, and validation. Two of three planned tests on the first BMP model have been conducted in the NASA Langley Transonic Dynamics Tunnel. This first BMP model has a NACA 0012 airfoil and is a rigid model mounted on a flexible mount system with both pitch and plunge degrees of freedom. Wind-off structural dynamic characteristics of the wing mounted on a flexible mount system have been presented. The conventional flutter boundary, stall flutter boundaries, and a plunge instability region have been identified. Both steady and unsteady wing-surface pressure measurements have been obtained at the flutter and instability boundaries. An additional test to acquire similar data at a higher Reynolds number is planned. Current activities include further evaluation of the surface pressure measurements with the goal of making the data available for CFD code development and evaluation.

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