Experimental Modal Analysis of an Aero Commander Aircraft

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National Aeronautics and Space Administration

Langley Research Center
Hampton, Virginia 23665
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I. FOREWORD

This report documents the results of the dynamic testing and analysis of the fuselage panels of a 1957 Rockwell Aero Commander light aircraft. The work was performed in accordance with Wyle Labs purchase contract order No. 9-0636S and in support of NASA funded Columbia University research on noise transmission in light twin engined aircraft.

The work was performed during the week of March 23-27, 1981 at Columbia University, Engineering School under the direction of Professor R. Vaicaitus in the Civil Engineering Department. Limited data analysis was performed the first week of April.

Those participating in the project were as follows:

Columbia University
   R. Vaicaitus, Professor Civil Engineering

Wyle Laboratories
   A. L. Abrahamson, Ph.D., Engineering Contract Administrator

SDRC
   D. Geisler, Project Engineer, Vehicle Systems Testing
   D. Bricker, Engineering Assistant, Vehicle Systems Testing
II. INTRODUCTION

The reduction of interior noise in light aircraft has been the aim of several ongoing research efforts funded by NASA. These efforts have been concerned with both the noise source and the noise transmission mechanism. One such effort concerns one of the dominant source-path combinations for interior cabin noise, the thin fuselage sidewall. This effort is being carried out by a team at Columbia University School of Engineering headed by Professor R. Vaicaitus. The approach used by Professor Vaicaitus is theoretical in nature and involves sophisticated modeling of the sidewall panels and stiffeners. The work presented in this report was performed to provide experimental data from which to verify and supplement the dynamic characteristics predicted by the analytical models.

The test article for these experiments was the scraped, gutted fuselage of a twin engined 1957 Rockwell Aero Commander. This aircraft was the subject of many other tests investigating cabin noise of light aircraft while at the Langley Research Center in Hampton, Virginia. Some of the previous tests performed on this aircraft involved studying the effects of mass and stiffness treatments to the sidewalls. Therefore the results of this work will include the dynamic characteristics of some treated panels to contrast with the untreated original sidewall panels.

The results documented in this report were obtained using experimental modal analysis techniques. See references for technical papers describing this technique. These results include the natural frequencies, modal damping, and mode shapes of selected panels. Frequency response functions are provided for most panels whereas the modal parameters were extracted for specific panels. A summary of the modal parameters of those selected panels is presented first. Subsequent to the summary is a bulk data presentation including the test setup documentation, data acquisition techniques and data analysis techniques.

While the primary objectives of this work were focused on local panel dynamic responses, some additional data was acquired relating to the global (involving two or more panels) fuselage response. Also, acoustic response measurements were made. This involved measuring the sound pressure level of several locations in the pilot/passenger cavity due to a single point mechanical excitation of the fuselage sidewall. These global response and acoustic response results are presented in the bulk data sections.
III. SUMMARY

This section summarizes the data presented in the subsequent pages of this report. Included in the summary is a table of the data acquired during the testing performed on-site and also a table summarizing the panel frequencies and damping. Actual frequency response functions and mode shapes are presented in the bulk data sections of the report.

Table III.1 summarizes the data acquired. It indicates the type of data acquired, impact driving point transfer functions or random excitation cross transfer functions, and how much of each. Impact frequency response functions were used to determine the local panel resonances whereas random excitation data was used to generate global resonances. This was done because impacting the panels did not excite global modes, whereas the random excitation excited many fuselage modes. Table III.1 also indicates the panels for which mode shapes and damping were calculated.

Tables III.2 and III.3 summarize the panel frequencies and modal damping for the panels for which this information was generated. Panel frequencies are presented with a tolerance of +/- 2 Hz in most cases. Modal damping was calculated for well defined modes on some panels, and is presented as percent of critical viscous damping. Following are some general observations of the data.

1. Modal damping ranges from .3 - 1.0 percent of critical for the untreated panels.
2. Damping on the honeycomb treated panels range from .6 to 2.6 percent of critical.
3. Damping on the windows and door is much higher at 4 - 10 percent of critical.
4. Individual panel frequencies start above 100 Hz, the first being at 113 Hz for panel 2P.
5. The first fuselage sidewall global mode appears to be at approximately 60 Hz (possibly around 50 Hz). The term global mode is meant to imply a mode involving more than one individual panel.
6. Global modes are apparent up to 200 Hz such that some modes presented in this report in the range 100 - 200 may be more global than local in nature.
7. Honeycomb stiffening increases the first panel mode in frequency only slightly. For example the first panel mode of panel 1S, honeycomb stiffened, is at 167 Hz while the first mode of panel 1P, untreated, is at 150 Hz.
8. However, most noteworthy of the honeycomb treated panels is that the panel mode shapes are changed drastically. Compare the mode shape at 169 Hz of panel 1S, page 78, with the mode shape of panel 1P at 161 Hz, page 128. The honeycomb stiffened panels demonstrate more continuous behavior across the stringer and frame boundaries than the unstiffened panels do.
9. Acoustic response of the cavity due to single point mechanical excitation of the sidewall appeared linear with respect to excitation force in the range tested. That is, when the force is doubled from 1 pound RMS to 2 pounds RMS the sound pressure level increased by approximately 6 dB.
### Table III.1
Data Summary

<table>
<thead>
<tr>
<th>Panel ID</th>
<th>Frequency Response Functions Acquired</th>
<th>Mode Shapes Calculated</th>
<th>Damping Calculated</th>
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<tr>
<td></td>
<td>Impact</td>
<td>Random</td>
<td></td>
</tr>
<tr>
<td>1P</td>
<td>6</td>
<td>Full Grid</td>
<td>Yes</td>
</tr>
<tr>
<td>2P</td>
<td>3</td>
<td>Full Grid</td>
<td>Yes</td>
</tr>
<tr>
<td>3P</td>
<td>2</td>
<td>3</td>
<td>Yes</td>
</tr>
<tr>
<td>4P</td>
<td>1</td>
<td>3</td>
<td></td>
</tr>
<tr>
<td>5P</td>
<td>1</td>
<td>3</td>
<td></td>
</tr>
<tr>
<td>6P</td>
<td>2</td>
<td>3</td>
<td></td>
</tr>
<tr>
<td>7P</td>
<td>1</td>
<td>3</td>
<td>Yes</td>
</tr>
<tr>
<td>8P·</td>
<td>1</td>
<td>3</td>
<td></td>
</tr>
<tr>
<td>Port Fuselage</td>
<td>1</td>
<td>3</td>
<td>Yes</td>
</tr>
<tr>
<td>Port Windscreen</td>
<td>1</td>
<td>3</td>
<td>Yes</td>
</tr>
<tr>
<td>1S</td>
<td>4</td>
<td>Full Grid</td>
<td>Yes</td>
</tr>
<tr>
<td>4S</td>
<td>4</td>
<td>6</td>
<td></td>
</tr>
<tr>
<td>6S</td>
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<td>7S</td>
<td>4</td>
<td>6</td>
<td></td>
</tr>
<tr>
<td>8S</td>
<td>2</td>
<td>Full Grid</td>
<td>Yes</td>
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Table III.2
Aero Commander, Starboard Side Panel Frequencies
and Damping (percent of critical)

<table>
<thead>
<tr>
<th>Mode Number</th>
<th>Panel Identification</th>
<th>1S</th>
<th>4S</th>
<th>6S</th>
<th>7S</th>
<th>8S (window)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>f</td>
<td>( \delta )</td>
<td>f</td>
<td>( \delta )</td>
<td>f</td>
<td>( \delta )</td>
</tr>
<tr>
<td>1</td>
<td>167</td>
<td>.7</td>
<td>171</td>
<td>.7</td>
<td>135</td>
<td>107</td>
</tr>
<tr>
<td>2</td>
<td>179</td>
<td>.8</td>
<td>134</td>
<td>1.0</td>
<td>242</td>
<td>172</td>
</tr>
<tr>
<td>3</td>
<td>185</td>
<td>1.0</td>
<td>140</td>
<td>.4</td>
<td>263</td>
<td>209</td>
</tr>
<tr>
<td>4</td>
<td>197</td>
<td>1.0</td>
<td>150</td>
<td>.7</td>
<td>276</td>
<td>230</td>
</tr>
<tr>
<td>5</td>
<td>210</td>
<td>1.4</td>
<td>153</td>
<td>.6</td>
<td>330, 335</td>
<td>248</td>
</tr>
<tr>
<td>6</td>
<td>266</td>
<td>.6</td>
<td>348</td>
<td>.6</td>
<td>328</td>
<td>276</td>
</tr>
<tr>
<td>7</td>
<td>271</td>
<td>.7</td>
<td>380, 390</td>
<td>390</td>
<td>390</td>
<td>363</td>
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<td>8</td>
<td>295</td>
<td>.4</td>
<td>444</td>
<td>403</td>
<td>1.7</td>
<td>400</td>
</tr>
<tr>
<td>9</td>
<td>320</td>
<td>.7</td>
<td>474</td>
<td>.6</td>
<td>457</td>
<td>.</td>
</tr>
<tr>
<td>10</td>
<td>338</td>
<td>.7</td>
<td>265</td>
<td>.4</td>
<td>537</td>
<td>473</td>
</tr>
</tbody>
</table>

1 Pairs of modes show up at approximately these frequencies.
2 Most likely a global mode, involving at least 2 panels.
Table III.3
Aero Commander, Port Side Frequencies
and Damping (percent of critical)

<table>
<thead>
<tr>
<th>Mode Number</th>
<th>Panel Identification</th>
<th>Port Fuselage</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>1P</td>
<td>2P</td>
</tr>
<tr>
<td>1</td>
<td>150</td>
<td>113</td>
</tr>
<tr>
<td>2</td>
<td>159</td>
<td>162</td>
</tr>
<tr>
<td>3</td>
<td>167</td>
<td>.6</td>
</tr>
<tr>
<td>4</td>
<td>185</td>
<td>.5</td>
</tr>
<tr>
<td>5</td>
<td>192</td>
<td>.6</td>
</tr>
<tr>
<td>6</td>
<td>203</td>
<td>276</td>
</tr>
<tr>
<td>7</td>
<td>212</td>
<td>312</td>
</tr>
<tr>
<td>8</td>
<td>227</td>
<td>.4</td>
</tr>
<tr>
<td>9</td>
<td>234</td>
<td>.7</td>
</tr>
<tr>
<td>10</td>
<td>243</td>
<td>.4</td>
</tr>
</tbody>
</table>

<sup>①</sup> Modes of the smaller panel which is one of two panels comprising panel 2P.

<sup>②</sup> Most likely a global mode, involving surrounding panels.
IV. TEST DESCRIPTION

This section of the report describes the test configuration, the test article definition and the test techniques used in the modal data acquisition task.

IV.1 Test Configuration

The test specimens were defined as particular panel sections of an Aero Commander fuselage sidewall. The panels bordered the passenger/pilot cavity beginning with the door and continuing forward to and including the pilot windscreen. The aircraft fuselage with wings, tail and nose gear removed was supported in a sling type cradle. The sling straps supported the fuselage underbody just aft of the pilot side window and just aft of the door. It was assumed that the support mechanism does not affect the local panel properties. Figure IV.8 shows the aircraft fuselage in its cradle support.

Figures IV.1a and IV.2a show the panel identification scheme. The data presented in this report is labeled according to this scheme. For example, panel 6S is panel no. 6 (Figure IV.1a) on the starboard side.

Data was measured at discrete locations throughout the sidewall. For identification of where the data was taken these points are labeled with integer numbers. Figures IV.1b and IV.2b present an overview of the point labeling scheme. These figures do not show all data point locations because the grid work was too dense on certain panels. Subsequent figures will show all points on particular panels. Grid numbers in the range 1 through 250 are located on the starboard sidewall. All points on the port side are placed symmetrically opposite points on the starboard side and the grid points on the port side are labeled with numbers 1000 greater than the corresponding starboard point.

Detail data acquisition surveys were made on panels 1S, 6S, and 8S (starboard side) and on panels 1P and 2P on the port side. Figures IV.3 through IV.6 show the grid point detail for these panels. Figure IV.7 shows a longitudinal line and vertical line on the port sidewall where data was collected to document the global sidewall modes. The longitudinal line in Figure IV.7 represents the frame member which joins the top row of panels to the bottom row of panels. The vertical line in Figure IV.7 represents the frame that joins panels 6P and 7P and panels 2P and 3P.
The coordinate system used for displaying the panel geometry and labeling the data is the right hand cartesian system. The X direction corresponds to the fore-aft direction, positive being forward. The Y direction is positive up in the vertical direction while the Z direction corresponds to the lateral direction positive to the starboard side.

All data presented in this report is labeled with the point number and direction X, Y, or Z that correspond to the location and sense at which the data was obtained.

IV.2 Data Acquisition

The objective of the data acquisition task of this work was to experimentally measure the appropriate frequency response functions, FRF, from which to extract the desired panel modal parameters. The frequency response functions, sometimes referred to as a transfer function, is the ratio of the output response to the input force. In this case the FRF is acquired, stored and displayed in units of g's/pound (acceleration response/input force) as a function of frequency. The functions presented in this report are labeled with two coordinates. The first coordinate represents force input location and direction while the second coordinate represents the response location and direction.

For example, FRF 1Z-, 35Z represents the response at point 35 in the +Z direction due to a force at point 1 in the -Z direction. The FRF's were obtained using two different techniques. The first method involved single point random excitation. This was accomplished by attaching an electromechanical shaker to the fuselage sidewall via a uniaxial stinger. The stinger was instrumented with force transducer in order to measure the input force. A roving small accelerometer was then attached to the desired location for the response measurement. Figure IV.10 demonstrates this excitation method while Figure IV.11 shows a close up of the stinger attachment. This method proved excellent for exciting the entire fuselage sidewall and was used for extracting panel mode shape information. The input force spectra for this technique was essentially flat out to 1000 Hz. The exact point of excitation was in line with a fuselage frame, which is close to a nodal line for local panel modes. Consequently, the measured response was often composed predominantly of global fuselage modes making it difficult, in some cases, to identify individual frequencies associated with a given panel.

The second data acquisition method involved impulse techniques. This technique requires the input to be supplied by the impact of an instrumented hammer. The impulse response was then measured using the same small accelerometers as were used before.
The corresponding FRF was then calculated by averaging 5 such impacts. This method was used to single out the local panel modes and typically only driving point FRF's were acquired. These driving point FRF's exhibit well separated, well defined response peaks and in most cases these functions were used for damping calculations. The actual FRF's are presented in the bulk data sections.

There was not enough energy input with the impact technique to excite global modes or even excite the modes of surrounding panels significantly. In fact the input energy was dissipated in local deformation of the impacted panel. The force input spectra dropped drastically at the frequencies associated with the panel flexibility. Figure IV.31 exemplifies the force input spectra for impacts on panels of different flexibility. Figure IV.31a shows the impact on panel 1S a honeycomb stiffened panel while Figure IV.31b is the force spectra for an impact on a large very flexible panel, panel 4S.

In summary, the impulse technique applied on the panel itself, was best suited for identifying the individual panel frequencies and extracting the modal damping for those frequencies. The random excitation applied at a centrally located frame member, was best suited for acquiring the modal deformation patterns of the individual panels as well as the overall fuselage. The random excitation also excited acoustic response in the cavity which is described in a subsequent section.
Figure IV.1
Starboard Sidewall, as Viewed from Port Side
(a) Panel Identification Scheme

(b) Data Point Labeling Scheme (Overview)
(All Points Except Detail Grid on Panels 1, and 2)

Figure IV.2
Port Sidewall, as Viewed from Port Side
Figure IV.3
Panel 6S, Data Point Identification

Figure IV.4
Panel 1S, Data Point Identification

a) outside skin

b) inside frame and stringers

Frames and Stringers

Frames and Stringers
Figure IV.5
Panel 8S, Data Point Identification

Figure IV.6
Panel 1P and 2P
Data Point Identification
Figure IV.7
Port Side Global Fuselage Measurement Location Identification
Figure IV.8
Test Article Supported in Sling Type Cradle

Figure IV.9
Instrumented Impact Hammer and Response Transducer

Figure IV.10
Random Excitation Setup
Electromechanical Shaker

Figure IV.11
Force Transducer Closeup for Electromechanical Shaker
Figure IV.12
Panel 1S
Exterior

Figure IV.13
Panel 1S, Interior
(Honeycomb Stiffened)

Figure IV.14
Panel 6S
Exterior

Figure IV.15
Panel 6S, Interior
(Honeycomb Removed)
Figure IV.24
Panel 5P
Pilot Side Glass

Figure IV.25
Panel 6P

Figure IV.26
Panel 7P
Passenger Side Glass

Figure IV.27
Panel 8P
Door Glass Exterior
Figure IV.28
Port Side Windscreen

Figure IV.29
Microphone Location 101
View from Rear

Figure IV.30
Microphone Location 101
View from Starboard Side
Figure IV.31

a) Input Force Spectra, Impact Technique
Honeycomb Stiffened Panel 1S

b) Input Force Spectra, Impact Technique
Panel 4S

Figure IV.31
IV.3 Data Analysis

Data Analysis consisted of the determination of the: (1) mode shapes, and (2) modal damping corresponding to the panel resonant frequencies. This section describes the procedures used for determining the parameters desired.

The modal deformation patterns (or mode shapes) presented in this report were generated by extracting the imaginary component at the desired frequency for each transfer function of the panel. This type of mode shape is referred to as a quadrature mode shape. These coefficients are stored in a file for each mode and can be superimposed on the geometry of the panel for visual display of the deformation pattern. This display can be animated at the computer console or statically displayed for hard copy documentation.

The use of quadrature mode shapes assumes well separated real normal modes at resonance. This implies that the imaginary component of the transfer function is at a maximum (absolute value) at resonance while the real component is at zero. Thus the modal amplitude is equal to the amplitude of the imaginary component of the function at the resonant frequency. In general, this approach is adequate for the lower order panel modes.

Figures IV.32a and b graphically depict the above discussion. Figure IV.32a is the Bode frequency response plot for response location 32Z on panel 6S while exciting at location 1Z. This plot is an amplitude and phase versus frequency display. Figure IV.32b displays the real and imaginary components of the same function shown in Figure IV.32a. The tabulated points at the side of each plot correspond to cursorred points on the plots. In Figure IV.32a the tabulated numbers correspond to the frequency and total response magnitude whereas in Figure IV.32b the tabulated numbers correspond to frequency and imaginary component amplitude. This imaginary component amplitude corresponds to the mode coefficients on point 32Z presented in subsequent sections.
Phase ______________

Log Amplitude
A/F

Real Component
Imaginary Component

Log Frequency

2.00E-03 2.00E 02 4.00E 02

A1: AERO PANEL 6S, RAND -

a) Total Amplitude and Phase versus Frequency

Real Component
Imaginary Component

Linear Frequency

2.00E 00

A3: AERO PANEL 6S, RAND -

b) Real and Imaginary Components versus Frequency

Figure IV.32
Transfer Function 1Z - 32Z, Panel 6S
Modal Damping is determined by using a multi-degree of freedom, MDOF, curve fitting algorithm on selected transfer functions. This technique is implemented in the SDRC modal analysis software package. The results of this technique include natural frequency modal amplitude proportional to modal mass and stiffness and modal damping ratio. The damping is viscous type ratioed to critical damping.

The general procedure involved in modal parameter estimation is outlined as follows. First the MDOF algorithm is applied to a selected function over a desired frequency range. Table IV.1 presents the results of such an estimation. The driving point function at location 1106Z on panel 1P was curvefit over the frequency range of 210 – 250 Hz. The results of this curvefit can be checked by generating a transfer function based on the analytical results. This is accomplished by choosing the significant roots of the estimation (the significant roots in Table IV.1 are underlined) and building a modal parameter table. Table IV.2 is such a table that contains the significant results shown in Table IV.1 as well as the results of other MDOF estimations. An analytical expression representing the sum of all the modes represented in Table IV.2 can then be generated for comparison with the original function from which the parameters were extracted. Figure IV.33 shows such a comparison for the parameters presented in Table IV.2.

There are certain requirements of the MDOF routine to assure accuracy in the estimation results. One of these requirements is that the frequency resolution be such that damping can be accurately estimated. The less damping present the greater the resolution required. For some of the panels extra data was acquired with greater resolution for this reason.
Table IV.1

MDOF Curvefitting Results
Significant Roots Underlined

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<tr>
<th>ROOT</th>
<th>FREQUENCY</th>
<th>DAMPING</th>
<th>AMPLITUDE</th>
<th>PHASE</th>
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<td>17.81</td>
<td>-3.142</td>
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<td>3</td>
<td>216.2</td>
<td>0.6153E-02</td>
<td>13.89</td>
<td>2.339</td>
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<tr>
<td>4</td>
<td>221.3</td>
<td>0.4965E-02</td>
<td>105.8</td>
<td>-1.827</td>
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Table IV.2

Parameter Table of Significant Roots

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<th>DAMPING</th>
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Figure IV.33
Comparison of Experimental Results versus Analytically Generated Results
IV.4  Acoustic Frequency Response

Random excitation on the fuselage sidewall created motion throughout the fuselage and consequently a significant sound pressure level. During these tests, microphones were positioned in the pilot/passenger cavity to measure the acoustic transfer function. These acoustic frequency response functions are calibrated in psi/lb and were measured during excitation on both the port side, 1001Z, and the starboard side, 1Z. Four acoustic response locations were chosen and their locations are described in Figure IV.34 and in the following table.

Table IV.3

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The measured acoustic frequency response functions are presented in Figures IV.35 through IV.43. Accompanying the FRF's are power spectral densities (PSD) of the input force, the input acceleration and the sound pressure level (SPL). Figure IV.38 shows the background noise present. Table IV.4 summarizes the overall acoustic response in terms of RMS.
Table IV.4
Acoustic Response to Single Point
Random Mechanical Input Summary

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<th>Force Input Location</th>
<th>Acoustic Response Location</th>
<th>Pounds Force Input (RMS)</th>
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<th>Accelerations At Input G's RMS</th>
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Figure IV.34
Microphone Location Description
for Acoustic Frequency Response Measurements
Figure IV.35
Acoustic Frequency Response
Location 101, Starboard Excitation
Figure IV.36
Acoustic Frequency Response
Location 102, Starboard Excitation
Figure IV.37
Acoustic Response Excitation
Starboard Side, Random Excitation
Figure IV.38
Acoustic Background Noise
Location 101
Figure IV.39
Acoustic Frequency Response
Location 101, Port Side Excitation
Figure IV.40
Acoustic Frequency Response
Location 102, Port Side Excitation
Figure IV.41
Acoustic Frequency Response
Location 103, Port Side Excitation
Figure IV.42
Acoustic Frequency Response
Location 104, Port Side Excitation
Figure IV.43
Acoustic Response Excitation
Port Side Random Excitation
V. DATA PRESENTATION

This section presents the dynamic response of the panels in terms of mode shapes and frequency response function. The modal deformation patterns are presented statically with the deformed shape shown with dotted lines overlayed on top of the undeformed shape as a solid line. Modal coefficients are also listed for selected modes. Frequency Response Functions, (FRF), acquired via the impact technique follow the mode shapes. These functions are plotted in a Bode diagram, log amplitude and linear phase versus log frequency. The plots contain cursor points with the frequency and amplitude of the cursored points tabulated on the side.
V.1 Modal Coefficients, Deformation Patterns and Frequency Response Functions
Panel 6S

The mode shapes in this section are shown from an isometric viewing position. Two separate traces are shown, one of the panel skin and one of the frame and stringer behind the panel. The trace links are separated so as not to confuse the pattern. There are more frames attached to panel but data was acquired only on those shown in the mode shape display.
MR PANEL 65
1. 1Z- COMP,F= 72.000 KHZ < 1.0, 1.0, 1.0, 0.0=VIEW

MR PANEL 65
2. 1Z- COMP,F= 93.000 KHZ < 1.0, 1.0, 1.0, 0.0=VIEW

MR PANEL 65
3. 1Z- COMP,F= 166.000 KHZ < 1.0, 1.0, 1.0, 0.0=VIEW
MR PANEL 6S
4: 12- COMP, F= 170.600 HZ < 1.0, 1.0, 1.0, 0.0 VIEW
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MODE SHAPE 4: 1Z- REAL, FREQ = 170.000 Hz
MR PANEL 63

COEFF 1

COEFF 2

COEFF 3
MR PANEL 6S
5: 12- COMP, F = 174.800 HZ ( 1.0, 1.0, 1.0, 0.0 ) VIEW
### Mode Shape 5: 12 - REAL, FREQ = 174.000 HZ

**MR PANEL 63**

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MR PANEL 6S
6: 1Z-COMP, F = 219.000 HZ < 1.0, 1.0, 1.0, 0.0> VIEW
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MODE SHAPE 5: 12- REAL, FREQ = 219.000 HZ
MR PANEL 53

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4  0.0000E-01  0.0000E-01  -3.6478E-01
5  0.0000E-01  0.0000E-01  -2.8260E-01
6  0.0000E-01  0.0000E-01  -2.0168E-01
7  0.0000E-01  0.0000E-01  -1.2595E-01
8  0.0000E-01  0.0000E-01  -9.8138E-02
9  0.0000E-01  0.0000E-01  -8.2492E-02
10 0.0000E-01  0.0000E-01  -6.2566E-02
11 0.0000E-01  0.0000E-01  -2.3429E-02
12 0.0000E-01  0.0000E-01  -2.4982E-02
13 0.0000E-01  0.0000E-01  -2.0164E-02
14 0.0000E-01  0.0000E-01  -1.2595E-02
15 0.0000E-01  0.0000E-01  -9.8138E-02
16 0.0000E-01  0.0000E-01  -8.2492E-02
17 0.0000E-01  0.0000E-01  -6.2566E-02
KPR PANEL 16S
7: 1Z-COMP, F= 232.000 HZ < 1.0, 1.0, 1.0, 0.0> VIEW
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**MR Panel 6S**

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I1R PANEL 68

8: 12-CCMP,F=240.000 HZ < 1.0, 1.0, 1.0, 0.0> VIEW
**MODE SHAPE 3: 1Z REAL, FREQ = 240.000 HZ**

**MR PANEL 6S**

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MODE SHAPE 9: 12- REAL, FREQ = 252.000 HZ

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MR PANEL 66
12: 1Z-COMP,F= 305.000 HZ < 1.0, 1.0, 1.0, 0.0>VIEW
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MODE SHAPE 13: 12- REAL, FREQ = 329.000 Hz
MR PANEL 6S

MODE SHAPE
MR PANEL 6S
14: 1Z- COMP,F= 338.000 HZ < 1.0, 1.0, 1.0, 0.0> VIEW
### MODE SHAPE 14: 12- REAL, FREQ = 338.000 Hz

**MR PANEL 63**

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Frequency Response, Panel 6S

Figure V.1.2
Frequency Response, Panel 6S
Figure V.1.3
Frequency Response, Panel 6S

Figure V.1.4
Frequency Response, Panel 6S
Figure V.1.5
Frequency Response, Panel 6S

Figure V.1.6
Frequency Response, Panel 6S
V.2 Modal Coefficients, Deformation Patterns and Frequency Response Functions
Panel 1S
FR PANEL 19
1: 1Z-CMP,F= 152.000 HZ < 1.0, 1.0, 1.0, 0.0=VIEW
**MODE SHAPE 1:** 1Z- REAL, FREQ = 152.000 HZ

**MR PANEL IS**

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MODE SHAPE 8: 12- REAL, FREQ = 297.000 HZ
MR PANEL IS
MR PANEL 1S
9: 1Z- COMP,F= 328.000 HZ ( 1.0, 1.0, 1.0, 0.0)=VIEW

MR PANEL 1S
10: 1Z- COMP,F= 366.000 HZ ( 1.0, 1.0, 1.0, 0.0)=VIEW
Figure V.2.1
Frequency Response, Panel 1S

Figure V.2.2
Frequency Response, Panel 1S
V.3 Modal Coefficients, Deformation Patterns and Frequency Response Functions
Panel 8S

The mode shapes in this section are presented from three viewing positions, two plan views and an isometric view. The deformed position denoted by the dotted line, is overlayed on the undeformed solid line. Two deformation positions, the plus and minus extremes, are shown for the isometric whereas only the plus extreme deformed position is shown in the plan views.
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MR PANEL 8S (WINDOW)  
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7: 1Z- COMP,F= 158.000 HZ ( 0.0, 0.0, 0.0, 0.0)=VIEW

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MODE SHAPE 9: 12- REAL, FREQ = 174.000 HZ
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MR PANEL 8S (WINDOW)
10: 12- COMP,F= 185.000 HZ ( 1.0, 1.0, 1.0, 0.0)=VIEW
#
MR PANEL

MR PANEL 9S (WINDOW)

11: IZ- COMP,F = 194.000 HZ < 0.0, 0.0, 0.0, 0.0, 0.0> = VIEW

MR PANEL 9S (WINDOW)

11: IZ- COMP,F = 194.000 HZ < 1.0, 1.0, 1.0, 0.0> = VIEW

MODE SHAPE 11: IZ- REAL, FREQ = 194.000 HZ

MR PANEL 9S (WINDOW)

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MR PANEL 05 (WINDOW)
12: 1Z- COMP.F= 209.000 HZ (< 1.0, 1.0, 1.0, 0.0) = VIEW
MODE SHAPE 13: 1Z- REAL, FREQ = 223.000 Hz
MR PANEL 85 (WINDOW)

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MR PANEL 85 (WINDOW)
13: 1Z- COMP, F= 223.000 Hz (1.0, 1.0, 1.0, 0.0)=VIEW
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MR PANEL 8's (WINDOW)
16: 12- COMP,F= 273.000 HZ ( 0.0, 0.0, 0.0, 0.0)=VIEW
#

MR PANEL 8's (WINDOW)
16: 12- COMP,F= 273.000 HZ ( 1.0, 1.0, 1.0, 0.0)=VIEW
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MR PANEL 8S (WINDOW)
17: 12- COMP,F = 283.000 HZ ( 0.0, 0.0, 0.0, 0.0)=VIEW
MR PANEL 8S (WINDOW)
18: 1Z- COMP,F= 321.000 Hz < 0.0, 0.0, 0.0, 0.0>=VIEW
#

MR PANEL 8S (WINDOW)
18: 1Z- COMP,F= 321.000 Hz < 1.0, 1.0, 1.0, 0.0>=VIEW
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MR PANEL 8S (WINDOW)
19: 12- COMP,F= 351.000 HZ ( 0.0, 0.0, 0.0, 0.0)=VIEW
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MR PANEL 8S (WINDOW)
19: 12- COMP,F= 351.000 HZ ( 1.0, 1.0, 1.0, 0.0)=VIEW
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Figure V.3.1
Frequency Response, Panel 8S

Figure V.3.2
Frequency Response, Panel 8S
V.4 Modal Coefficients, Deformation Patterns and Frequency Response Plots
Panels 1P and 2P

The mode shapes shown in this section are shown from three viewing positions, two plan views and an isometric. The deformed position in dashed lines is overlayed on the undeformed which is the solid line. The isometric view actually shows two deformed positions, the plus and minus extreme positions, whereas the plan views show only the plus extreme position.
MR PORT SIDE PANELS 1P & 2P
1.1001Z+ COMP,F= 61.000 HZ ( 0.0, 0.0, 0.0, 180.0)=VIEW

MR PORT SIDE PANELS 1P & 2P
1.1001Z+ COMP,F= 61.000 HZ ( 1.0, 1.0, -1.0, 0.0)=VIEW
MODE SHAPE 1:1001Z+ REAL, FREQ = 61.000 HZ
MR PORT SIDE PANELS 1P & 2P

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MR PORT SIDE PANELS 1P & 2P
2 1001Z+ COMP,F= 71.000 HZ (< 1.0, 1.0, -1.0, 0.0)=VIEW
MODE SHAPE 2:10012+ REAL, FREQ = 71.000 HZ

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MODE SHAPE 3: 10012+ REAL, FREQ = 86.000 HZ
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4·100Hz COMP.F = 94.000 HZ < 1.0, 1.0, -1.0, 0.0> VIEW

MR PORT SIDE PANELS 1P & 2P
4·100Hz COMP.F = 94.000 HZ < 1.0, 1.0, -1.0, 0.0> VIEW
**MODE SHAPE 4: 1801Z+ REAL, FREQ = 94.000 HZ**

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MR PORT SIDE PANELS 1P & 2P
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MODE SHAPE 6:1001Z+ REAL, FREQ = 137.000 HZ
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MR PORT SIDE PANELS 1P & 2P
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MODE SHAPE 7: 1001Z+ REAL, FREQ = 146.000 Hz
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MR PORT SIDE PANELS 1P & 2P
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MR PORT SIDE PANELS 1P & 2P
0:1001Z+ COMP.F= 157.000 HZ < 1.0, 1.0, -1.0, 0.0 VIEW
MR PORT SIDE PANELS 1P & 2P
9:1001Z+ COMP,F= 161.000 HZ < 0.0, 0.0, 0.0, 100.0> VIEW

MR PORT SIDE PANELS 1P & 2P
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MR PORT SIDE PANELS 1P & 2P
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MR PORT SIDE PANELS 1P & 2P
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MR PORT SIDE PANELS 1P & 2P
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MODE SHAPE 12: 10012+ REAL, FREQ = 177,000 Hz
MR PORT SIDE PANELS 1P & 2P

MODE SHAPE
MR PORT SIDE PANELS 1P & 2P
13:10012+ COMP.F= 183.000 HZ < 0.0, 0.0, 0.0, 180.0) VIEW

MR PORT SIDE PANELS 1P & 2P
13:10012+ COMP.F= 183.000 HZ < 1.0, 1.0, -1.0, 0.0) VIEW
MR PORT SIDE PANELS 1P & 2P

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MODE SHAPE 13: 1001Z+ REAL, FREQ = 183.000 HZ
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MR PORT SIDE PANELS 1P & 2P
14:1001Z+ COMP.F= 189.000 HZ ( 1.0, 1.0, -1.0, 0.0)=VIEW
MR PORT SIDE PANELS 1P & 2P
15:1001Z+ COMP,F= 217.000 HZ ( 0.0, 0.0, 0.0, 100.0)=VIEW

MR PORT SIDE PANELS 1P & 2P
15:1001Z+ COMP,F= 217.000 HZ ( 1.0, 1.0, -1.0, 0.0)=VIEW
MR PORT SIDE PANELS 1P & 2P
16*1001Z+ COMP,F= 224.000 HZ < 0.0, 0.0, 0.0, 100.0>=VIEW
MR PORT SIDE PANELS 1P & 2P
17'1001Z+ COMP.F= 234.000 HZ (< 0.0, 0.0, 0.0, 100.0)=VIEW

MR PORT SIDE PANELS 1P & 2P
17'1001Z+ COMP.F= 234.000 HZ (< 1.0, 1.0, -1.0, 0.0)=VIEW
MR PORT SIDE PANELS 1P & 2P
10 Hz COMP. F= 241.000 Hz (< 0.0, 0.0, 0.0, 180.0)=VIEW

MR PORT SIDE PANELS 1P & 2P
10 Hz COMP. F= 241.000 Hz (< 1.0, 1.0, -1.0, 0.0)=VIEW
MR PORT SIDE PANELS 1P & 2P
19'1001Z+ COMP,F= 261.000 HZ ( 0.0, 0.0, 0.0, 180.0)=VIEW

MR PORT SIDE PANELS 1P & 2P
19'1001Z+ COMP,F= 261.000 HZ ( 1.0, 1.0, -1.0, 0.0)=VIEW
MR PORT SIDE PANELS 1P & 2P
20:1001Z+ COMP,F= 269.000 HZ < 0.0, 0.0, 0.0, 180.0>VIEW

MR PORT SIDE PANELS 1P & 2P
20:1001Z+ COMP,F= 269.000 HZ < 1.0, 1.0, -1.0, 0.0>VIEW
MR PORT SIDE PANELS 1P & 2P
21.1001Z+ COMP,F = 276.000 HZ < 0.0, 0.0, 0.0, 100.0> VIEW

MR PORT SIDE PANELS 1P & 2P
21.1001Z+ COMP,F = 276.000 HZ < 1.0, 1.0, -1.0, 0.0> VIEW
MR PORT SIDE PANELS 1P & 2P
22.1001Z+ COMP, F = 282.000 HZ < 0.0, 0.0, 0.0, 180.0> = VIEW

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22.1001Z+ COMP, F = 282.000 HZ < 1.0, 1.0, -1.0, 0.0> = VIEW
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MR PORT SIDE PANELS 1P & 2P
23:10812+ COMP,F= 288.000 HZ < 1.0, 1.0, -1.0, 0.0=VIEW
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**MODE SHAPE 20 1001Z+ REAL, FREQ = 196.000 HZ**

**NR PORT SIDE PANELS 1P & 2P**

**MODE SHAPE**

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Figure V.A.1  
Frequency Response, Panel 1P

Figure V.A.2  
Frequency Response, Panel 1P
Figure V.4.3
Frequency Response, Panel 1P

Figure V.4.4
Frequency Response, Panel 1P
Figure V.4.5
Frequency Response, Panel 1P

Figure V.4.6
Frequency Response, Panel 1P
Figure V.4.7
Frequency Response, Panel 2P

Figure V.4.8
Frequency Response, Panel 2P
V.5 Fuselage Port Side Global Mode Shapes

Each mode shape presented in this section is displayed from four viewing positions, three plan views and an isometric. The plus extreme deformation position is shown as a dashed line overlayed on top of the solid line undeformed position. In all cases the view normal to the fuselage sidewall shows no deformation because data was acquired normal to the panel only.
Figure V.5.1
Fuselage Port Side 61 Hz

Figure V.5.2
Fuselage Port Side 71 Hz
Figure V.5.3
Fuselage Port Side 86 Hz

Figure V.5.4
Fuselage Port Side 94 Hz
Figure V.5.5
Fuselage Port Side 114 Hz

Figure V.5.6
Fuselage Port Side 137 Hz
Figure V.5.7
Fuselage Port Side 146 Hz

Figure V.5.8
Fuselage Port Side 157 Hz
Figure V.5.9
Fuselage Port Side 161 Hz

Figure V.5.10
Fuselage Port Side 166 Hz
Figure V.5.11
Fuselage Port Side 172 Hz

Figure V.5.12
Fuselage Port Side 177 Hz
Figure V.5.13
Fuselage Port Side 183 Hz

Figure V.5.14
Fuselage Port Side 189 Hz
V.6 Port Side Panels

This section contains driving point frequency response functions of port side panels, acquired via the impact technique. No mode shape data was acquired on these panels, just the following functions.
Figure V.6.1
Frequency Response, Windscreen

Figure V.6.2
Frequency Response, Panel 3P
Figure V.6.3
Frequency Response, Panel 4P

Figure V.6.4
Frequency Response, Panel 5P
Figure V.6.5
Frequency Response, Panel 6P

Figure V.6.6
Frequency Response, Panel 7P
Figure V.6.7
Frequency Response, Panel 8P
V.7 Starboard Side Panels

This section contains driving point frequency response functions of starboard panels where no mode shape data was acquired. The functions were acquired via the impact technique.
Figure V.7.1
Frequency Response, Panel 4S

Figure V.7.2
Frequency Response, Panel 4S
Figure V.7.4
Frequency Response, Panel 4S
Figure V.7.5
Frequency Response, Panel 7S

Figure V.7.6
Frequency Response, Panel 7S
Figure V.7.7
Frequency Response, Panel 7S

Figure V.7.8
Frequency Response, Panel 7S
VI. REFERENCES


Experimental modal analysis was performed on the fuselage sidewalls of an Aero Commander aircraft for the purpose of assisting and validating the analytical modeling of these fuselage sidewalls. Modal parameters, i.e. modal damping and mode shapes were determined for both the original thin walled fuselage panels and honeycomb treated panels. The honeycomb treatment increased panel natural frequencies slightly but more significant was the change in the modal deformation patterns. Modal damping increased approximately a factor of two on the treated panels.

Ongoing research efforts aimed at the reduction of interior cabin noise of light aircraft was the original impetus for this work. One such effort involves the use of computer modeling to analyze noise transmission through the thin sidewalls of light aircraft, a major contributor to the interior cabin noise.