NASA Technical Memorandum 101717

Flutter Clearance of the F-14A Variable-Sweep Transition Flight Experiment Airplane — Phase 2

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SUMMARY

An F-14A aircraft was modified for use as the test-bed aircraft for the variable-sweep transition flight experiment (VSTFE) program. The VSTFE program was a laminar flow research program designed to measure the effects of wing sweep on laminar flow. The airplane was modified by adding an upper surface foam and fiberglass glove to the right wing. An existing left wing glove had been added for the previous phase of the program. Ground vibration and flight flutter testing were accomplished to verify the absence of aeroelastic instabilities within a flight envelope of Mach 0.9 or 450 knots, calibrated airspeed, whichever was less. Flight test data indicated satisfactory damping levels and trends for the elastic structural modes of the airplane. Presented in this report are ground vibration test data, in-flight frequency and damping estimates, time histories and power spectral densities of in-flight sensors, and pressure distribution data.

INTRODUCTION

Structural modifications to aircraft are a common occurrence in many flight research projects. However, it is important to ensure that these structural modifications do not adversely affect the aeroelastic characteristics of the aircraft. The procedure to verify the absence of flutter or aeroelastic instabilities for modified aircraft ideally involves a balanced combination of predictive analysis, as well as wind tunnel, ground, and flight testing.

Aircraft modifications may vary from altering a specific part of the structure, such as the wingtip (Kehoe, 1981; Freudinger, 1989), to carriage of different external stores (Smith et al., 1981), to altering a major lifting surface (Kehoe and Ellison, 1985). Each aircraft modification must be individually assessed because the aeroelastic sensitivity for a given modification will vary from one type of aircraft to another.

An F-14A aircraft was modified for use as the test-bed aircraft for the variable-sweep transition flight experiment (VSTFE) program. The VSTFE program was a laminar flow research program designed to measure the effects of wing sweep on boundary layer transition from laminar to turbulent flow (Anderson et al., 1988). The F-14A aircraft was chosen for its variable wing-sweep capabilities, wing planform, and Mach-Reynolds number envelope.

Phase 1 of the program consisted of adding a foam-fiberglass glove to the left wing beginning at wing butt line 130 and extending to wing butt line 350. The glove extended from the 5-percent chord line on the bottom of the wing, forward around the leading edge, and then aft on top of the wing to the 60-percent chord line. This glove was constructed to "cleanup" the airfoil of the basic airplane wing. The airplane, with this modification, underwent ground vibration and flight flutter testing to verify that the aeroelastic characteristics had not been adversely altered (Kehoe, 1987b).

Phase 2 of the research program involved the installation of a similar glove of similar overall dimensions on the right wing. However, though the left glove (or cleanup glove) had the same airfoil shape as the standard F-14A wing, the right glove had a different airfoil cross-sectional shape. The leading-edge chord extension of the right wing glove varied from 4.0 in. at the inboard section to 0.5 in. at the outboard section of the wing. The glove extended from the 5-percent chord line on the lower surface of the wing, forward and around the leading edge, and then aft along the upper surface to the 60-percent chord line. A comparison of a typical cross section between the modified left and right wings and the general layout of the gloves on the airplane are shown in figure 1.

Other aircraft modifications included:

1. Locking the leading-edge slats in the retracted position.
2. Disabling the maneuvering flaps.
3. Keeping the wing fuel tanks empty for the duration of the research program.

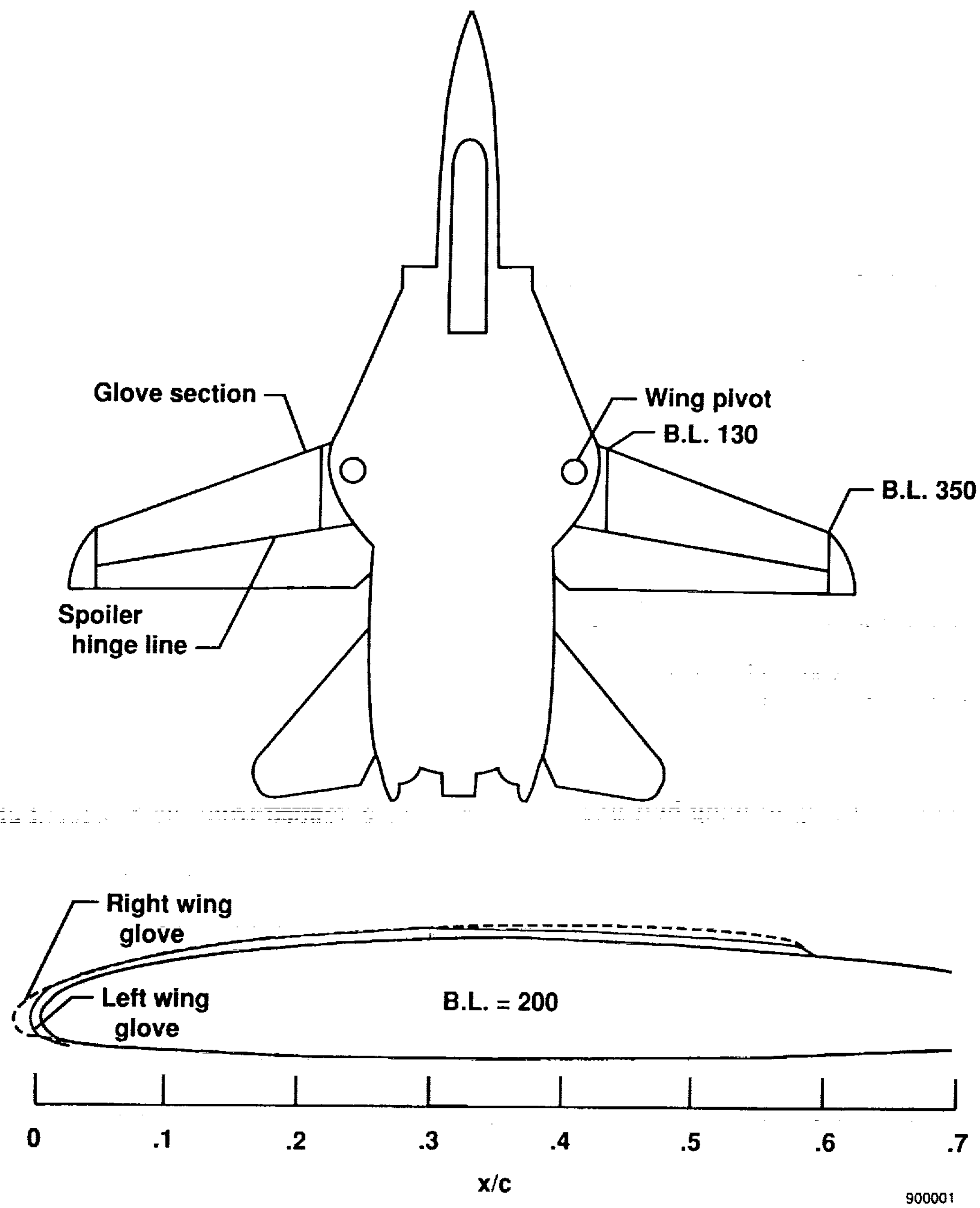


Figure 1. Typical cross sections and positions of the laminar flow gloves.

4. Preventing wing-sweep angles from exceeding 35° to avoid structural interference between the glove and wing fairing. A guard was mounted on the wing-sweep control panel to implement this restriction.

The aeroelastic concerns for this modification were similar to the phase 1 modification. The concerns were changes in wing weight, wing stiffness, and airfoil shape and their effect on the aeroelastic stability of the aircraft. The approach taken to flight qualify this modification was to conduct a ground vibration test (GVT) on the modified airplane and then compare the measured modal data with the modal data for the phase 1 modification and the basic airplane. The measured differences in frequencies and mode shapes were sufficient to warrant a conservative flight flutter clearance program. This report documents the methods used to clear the flight envelope sufficiently for the research program and provides a summary of the ground and flight test results.

NOMENCLATURE

B.L.	butt line
C_p	pressure coefficient
ETA	wing spanwise position
G	structural damping coefficient
GVT	ground vibration test
g	acceleration due to gravity
KCAS	knots, calibrated airspeed
NACA	National Advisory Committee for Aeronautics
V	velocity
VSTFE	variable-sweep transition flight experiment
x/c	chordwise position

VEHICLE DESCRIPTION

The F-14A is a variable-geometry midwing airplane with leading-edge slats, maneuvering flaps, and twin vertical stabilizers. The aircraft is powered by two afterburning turbofan engines and features a full-flying horizontal stabilator.

The net increase in the weight of the right wing was approximately 210 lb with a shift in the center of gravity of approximately 1 in. inboard and 2.6 percent of the chord length forward. The weight increase of the previously modified left wing was approximately 290 lb with a shift in the center of gravity of approximately 3 in. outboard and 3.5 percent chord length forward.

TEST PROCEDURE

Ground Vibration Test

A GVT was conducted to measure structural frequencies and mode shapes. These data were compared with modal data for the basic airplane and for the airplane with the left wing glove installed (Kehoe, 1987b). Previous tests provided baseline data to identify structural changes that resulted from the addition of the right wing glove. For all ground tests, the aircraft was supported on its landing gear with the struts deflated and the tire pressure reduced to half of the normal operating pressure to provide a soft support. Testing was accomplished with the aircraft in the 20° wing-sweep configuration and with the electrical and hydraulic systems operating. The aircraft was fully fueled, except for the wing tanks which were empty.

Instrumentation

Piezoelectric accelerometers were attached to the airplane to measure the structure's modal response. The accelerometers had a 10-g acceleration range with a sensitivity of 0.5 V/g. Force links, with a sensitivity of 50 mV/lb, were used to measure the input forces from the electrodynamic shakers. A minicomputer-based structural analysis system was used to acquire and analyze the structural responses at 170 locations on the airplane.

Excitation

Multishaker sine dwell and multi-input random excitation techniques were used to excite the structure (Kehoe, 1987a). The first technique, sinusoidal excitation, was used to excite the rigid body modes, to perform frequency sweeps, and to tune the wing fore and aft modes. The second technique, multi-input random, was used to excite all of the remaining structural modes.

The sine sweeps were performed with a vertically-oriented shaker under each wingtip. These sweeps were used to identify approximate resonant frequencies and to separate symmetric and antisymmetric modes. All frequency sweeps were conducted from 3 to 45 Hz at a logarithmic sweep rate of 0.3 decades/min. These data were compared to corresponding sine sweeps obtained from previous tests conducted on the airplane.

To excite the wing fore and aft modes, one horizontally oriented shaker was attached at each leading-edge wingtip by suspending the shaker from a steel cable. The cable was of sufficient length to provide a primary pendulum frequency of approximately one order of magnitude below the first elastic fore-and-aft mode frequency. Sine sweeps were performed to locate approximate frequencies, followed by sine-dwell tuning to accurately identify the modal frequencies for the fore-and-aft modes.

Multi-input, uncorrelated random excitation was used for the remainder of the modal survey with a shaker placed at each rear spar wingtip. This choice of shaker location was adequate for exciting the wing modes of interest.

Structural Mode Measurements

The wing fore-and-aft modes were fine-tuned using a coincident-quadrature analyzer to minimize the coincident (real) component and to maximize the quadrature (imaginary) component of a preselected structural response relative to the sinusoidal input force. An additional check for a properly tuned mode was done by terminating electrical power to the shakers and observing the decay of the response traces on a strip chart. Indications of a properly tuned mode included getting a smooth decay free of beats or frequency shifts. Damping estimates for the fore-and-aft modes were obtained by the logarithmic-decrement method using strip chart traces of the free decays.

Structural response data from random excitation were acquired, averaged, and stored in the frequency domain. The data were sampled at 128 samples/sec into blocks of 1024 samples with the antialiasing filters set at 50 Hz. A Hanning window was applied to reduce leakage errors. After fifty averages of data were acquired for the aircraft, modal parameters were estimated with a multi-input time domain method known as Polyreference (Hunt and Peterson, 1983).

Flight Test

The objectives of the flight flutter clearance program were to:

1. Estimate frequency and damping.

2. Establish stability trends for critical structural modes.

3. Verify freedom from flutter within a flight envelope of 450 knots, calibrated airspeed (KCAS) or Mach 0.90, whichever was less. Flutter testing was accomplished at altitudes of 5,000, 17,000, and 27,500 ft. The planned flutter envelope expansion points are shown in figure 2. The aircraft's stability augmentation system was active and the wing sweep was 20° for all expansion points. From a flutter standpoint, the 20° wing-sweep configuration was considered the most critical (Grumman, 1970).

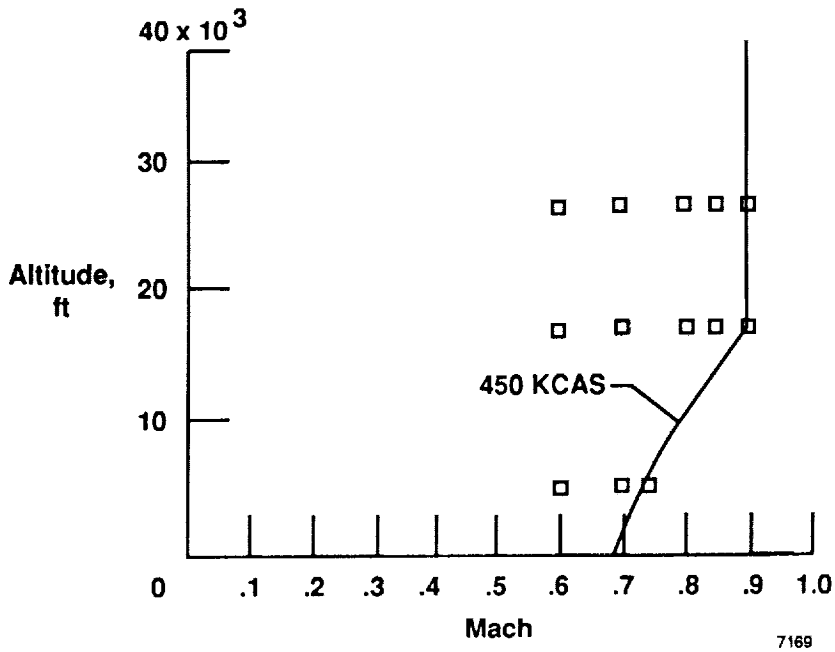
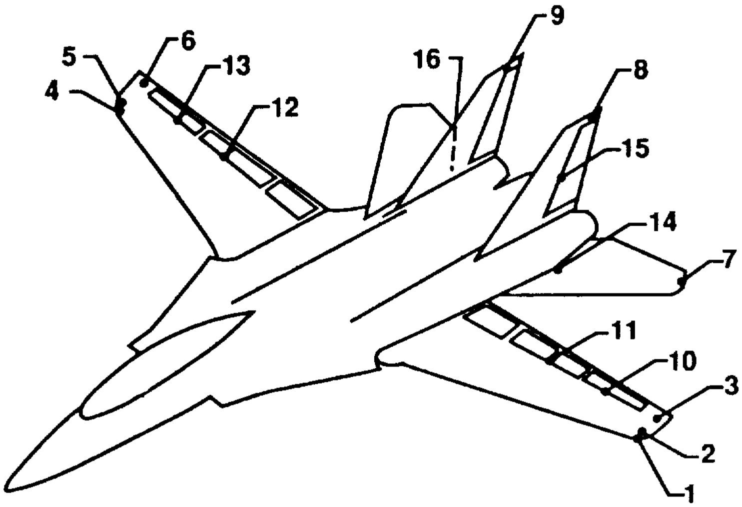


Figure 2. Planned flutter test envelope.

Instrumentation

Piezoelectric accelerometers were used to measure the structure's response. The accelerometer ranges and locations are shown in figure 3. Velocity and altitude information were obtained from a standard NACA noseboom and aircraft instrumentation. Digital telemetry was used to transmit all flight information to the ground where it was displayed in real time, both in frequency and time domain.

Four rows of flush static pressure orifices were used to acquire static pressure information during the stabilized test points on the right wing. Each row contained 24 surface pressure orifices. The approximate locations of the orifices on the wing are also shown in figure 3.



Item	Parameter identification	Range
1	Left wingtip fore-and-aft acceleration	± 2 g
2	Left forward wingtip normal acceleration	± 5 g
3	Left aft wingtip normal acceleration	± 5 g
4	Right wingtip fore-and-aft acceleration	± 2 g
5	Right forward wingtip normal acceleration	± 5 g
6	Right aft wingtip normal acceleration	± 5 g
7	Left horizontal stabilator normal acceleration	± 20 g
8	Left vertical stabilizer lateral acceleration	± 20 g
9	Right vertical stabilizer lateral acceleration	± 20 g
10	Left outboard spoiler position	0° - 60°
11	Left inboard spoiler position	0° - 60°
12	Right inboard spoiler position	0° - 60°
13	Right outboard spoiler position	0° - 60°
14	Left horizontal stabilator position	-30°, 12°
15	Left rudder position	-30°, 30°
16	Right horizontal stabilator position	-30°, 12°

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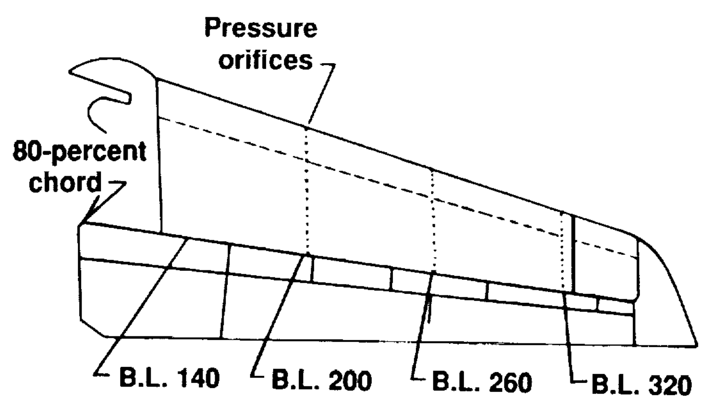

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Figure 3. Flight test accelerometer and pressure orifice locations.

Excitation

Natural atmospheric turbulence and pilot-generated control surface pulses were used as structural excitation. One min of stabilized data were collected at each test point. The signal-to-noise ratio of the resulting response data was typically low.

Pilot-induced control surface pulses were also used to excite the structure. However, these pulses provided only satisfactory excitation for the symmetric wing bending mode. The amplitude of the pulses were eventually reduced to avoid the possibility of cracking the fiberglass gloves. This reduction in amplitude further decreased the effectiveness of the pulses.

Envelope Clearance Procedure

A conservative procedure was used to clear the aircraft for each test point. The aircraft was slowly accelerated 2 to 4 knots/sec to the specified Mach number and test altitude. At this time, accelerometer time history responses were monitored on the strip charts for indications of any adverse aeroelastic behavior. Once stabilized on the specified test point, 60 sec of random data were acquired followed by pilot-induced control surface pulses.

All accelerometer response data channels were band-pass filtered with upper and lower cutoff frequencies of 40 Hz and 2 Hz, respectively. The data were displayed on strip charts and acquired by a Fourier analyzer. Fourier analyzer data were acquired at 100 samples/sec into data blocks of 1024 samples, which provided six averages of information/60 sec of stabilized flight. The Fourier analyzer was then used to immediately extract estimates of frequency and damping for critical modes. Satisfactory damping trends as a function of Mach number were then used as the basis to grant clearance for acceleration to the next test point. The same algorithms (Kehoe, 1988) were used postflight to estimate frequency and damping for a more complete set of modes.

RESULTS AND DISCUSSION

Ground Vibration Test Results

A comparison of measured modal data for the basic airplane, the gloved left wing (phase 1) airplane, and airplane with gloves on both wings (phase 2) is shown in the following table. Only data for the 20° wing sweep is shown because the configuration was considered the most critical for flutter (Grumman, 1970). In general, there were few significant changes in modal frequencies between the unmodified, the left-wing-gloved, and the both-wings-gloved configurations. The exceptions are discussed in the following paragraphs.

Table 1. Ground vibration test results and comparisons.
Wing sweep = 20°

Mode description	Left glove (phase 1)		Both gloves (phase 2)		Unmodified	
	Frequency, Hz	Damping, G	Frequency, Hz	Damping, G	Frequency, Hz	Damping, G
Symmetric						
Wing bending	4.62	0.02	4.57	0.02	4.71	0.03
Wing fore and aft	9.76	0.10	10.35	0.07	---	---
Left	---	---	---	---	9.23	0.09
Right	---	---	---	---	9.96	0.07
Second wing bending	13.98	0.04	13.93	0.03	14.39	0.04
Wing torsion	19.26	0.06	20.23	0.04	19.78	0.05
Left only	21.26	0.07	---	---	---	---
Horizontal stabilator bending	17.08	0.05	17.26	0.09	---	---
Fuselage bending	7.70	0.05	7.74	0.03	7.86	0.06
Engine nacelle roll	10.26	0.06	10.57	0.06	10.77	0.05
Antisymmetric						
Wing bending	6.54	0.03	6.50	0.04	6.49	0.06
Wing fore and aft						
Left	9.39	0.06	9.68	0.04	9.34	0.08
Right	11.23	0.11	11.34	0.04	10.35	0.07
Second wing bending	16.37	0.06	16.11	0.04	16.04	0.05
Wing torsion	23.79	0.05	23.25	0.04	23.54	0.04
Fuselage torsion	8.38	0.04	8.26	0.04	8.48	0.05

The most significant changes in the aircraft's modal frequencies and mode shapes involved the wing torsion modes. In Kehoe (1987b), the aircraft configuration with the left wing glove exhibited a left-wing-only torsion mode in addition to the typical torsion modes that involved both wings. The left-wing-only mode was eliminated with the addition of the right wing glove. Figure 4 shows a comparison of the torsion mode shapes for the baseline, left-glove-only (phase 1), and the left-plus-right-glove (phase 2) aircraft configurations. The symmetric wing torsion mode frequency for the left-plus-right-glove configuration was increased by 0.45 Hz above the basic airplane value. This frequency increase was attributed to the added torsional stiffness caused by the glove on each wing. The effect of the wing gloves on bending stiffness appeared to be small, as indicated by the comparison of measured bending frequencies shown in table 1.

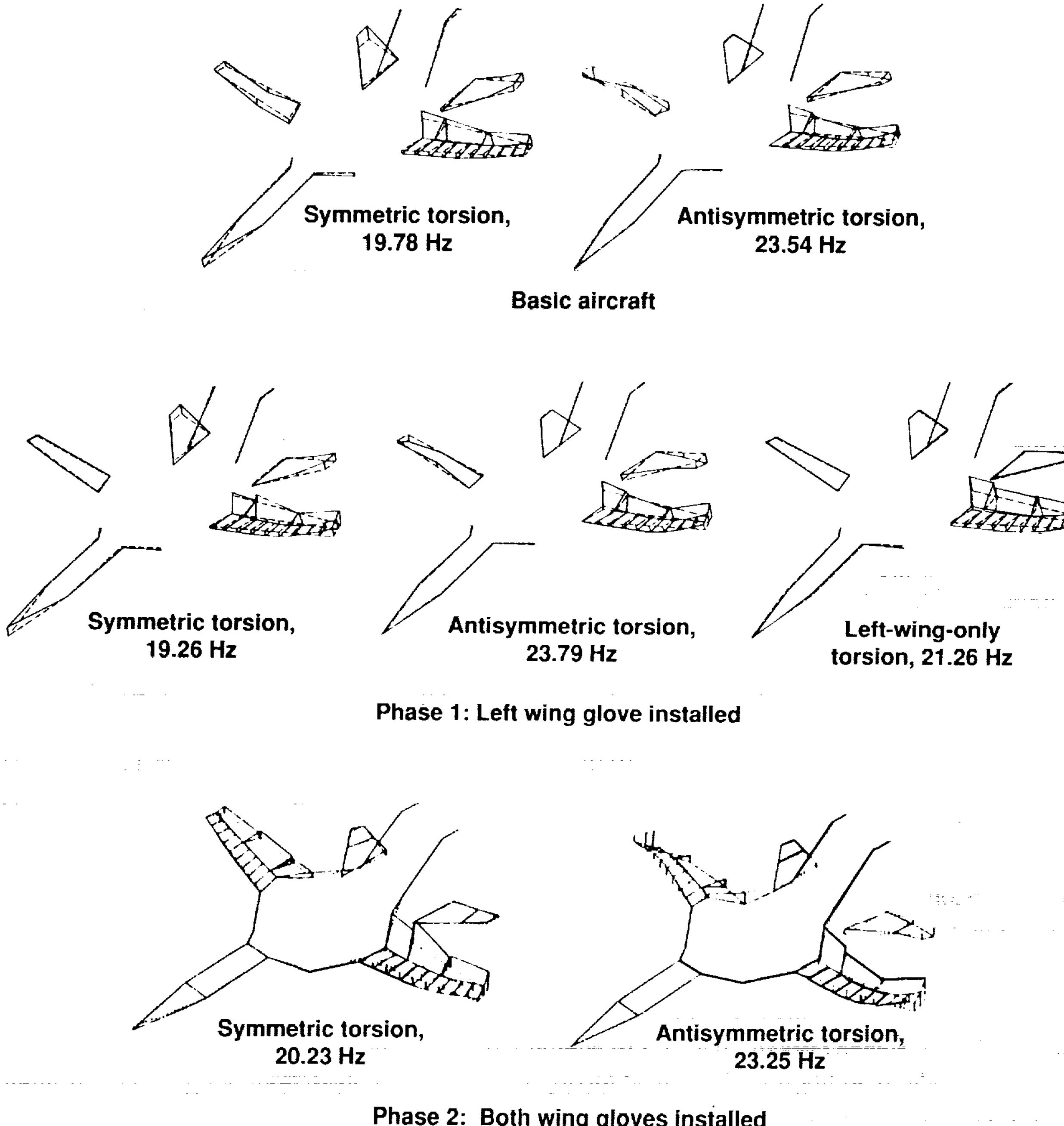

Figure 4. Wing torsion mode shape comparisons at 20° sweep angle.

The frequency of the antisymmetric wing fore-and-aft mode was a different value depending on which wing response was used to tune the mode. This finding was consistent with data acquired for the previous aircraft configurations. For all three aircraft configurations, the antisymmetric fore-and-aft modes were tuned at higher frequencies when tuned with the right wing. The symmetric fore-and-aft wing mode tuning was less difficult and did not exhibit the left-right anomalies except for the unmodified airplane. The differences in the left and right wing gloves, and the wing pivot freeplay, most likely contributed to the tuning difficulty in these modes.

Flight Test Results

Frequency and damping values were estimated for the structural modes excited during the flight flutter envelope expansion. Frequency and damping plots as a function of Mach number for the 20° wing sweep at altitudes of 5,000, 17,000, and 27,500 ft are shown in the appendix.

Damping levels were relatively high and trends were considered satisfactory for each structural mode. At velocities above Mach 0.80, an increase in response level on the right wing was observed and attributed to Mach effects. The response levels were considered high enough to potentially crack the right wing glove and, for this reason, the planned flight envelope maximum speed of Mach 0.90 was reduced to Mach 0.84. Cracking of the right wing glove would have been detrimental to accomplishing the objectives of the laminar flow research program.

Figure 5 shows accelerometer time histories for the left and right wings at Mach 0.84 and 17,000 ft. These traces show significant differences in amplitude with the right wing approximately twice the amplitude of the left wing. The higher amplitude of the right wing is believed to be caused by the aerodynamics of that wing.

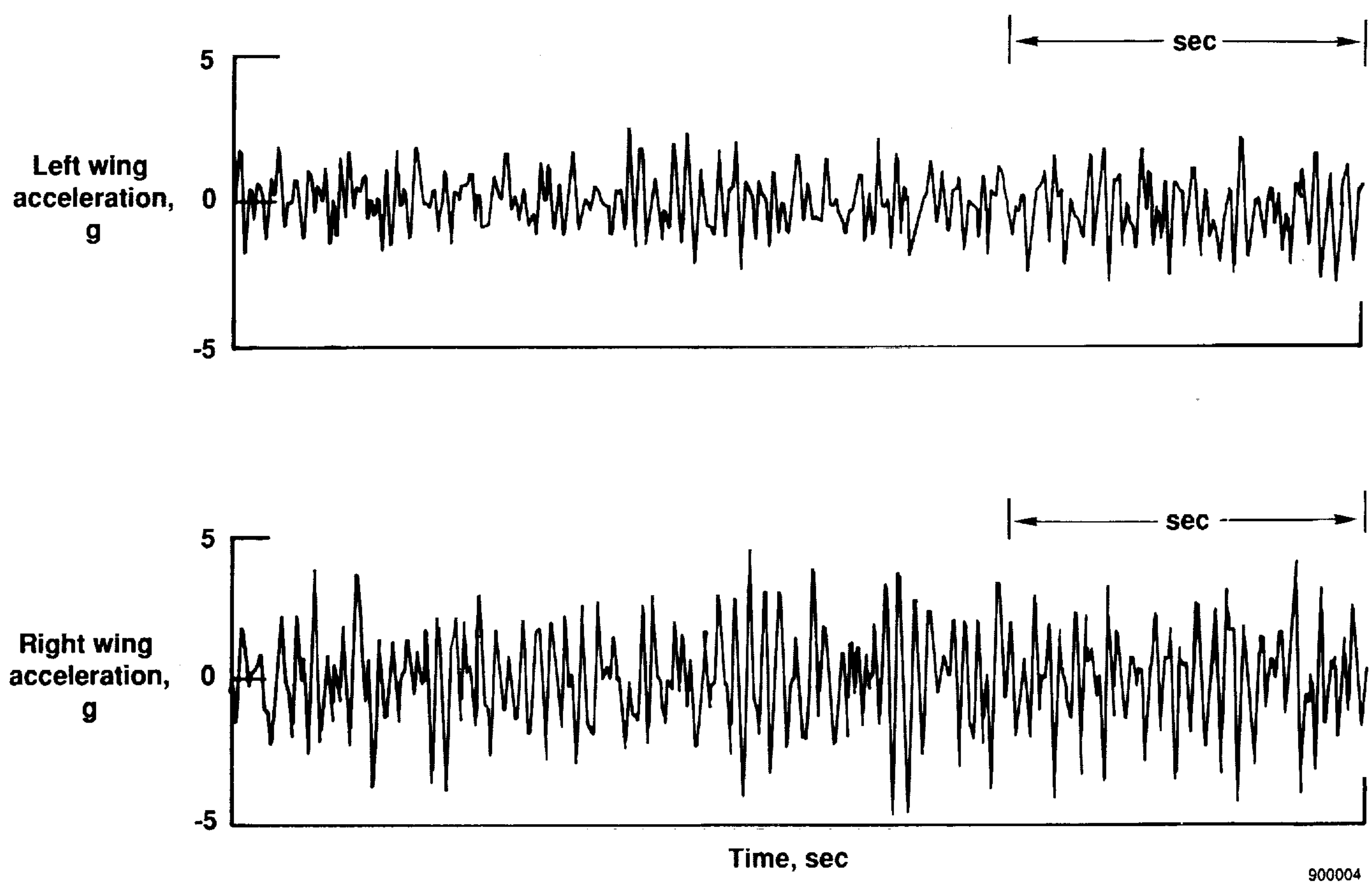

Figure 5. Left and right wing accelerometer time histories for Mach 0.84 at 17,000 ft.

It was hypothesized that a relatively strong shock wave had formed over the right wing. Right wing averaged pressure distribution plots for Mach 0.60, 0.70, 0.80, and 0.84 at altitudes of 17,000 and 27,500 ft are shown in figures 6 and 7, respectively. A shock wave is indicated by a sudden drop in the pressure coefficient with increasing chordwise position (x/c). For example, figure 7 shows a shock of moderate strength at Mach 0.80 at the 52-percent wingspan position (ETA = 0.52). The shock tapers off quickly to

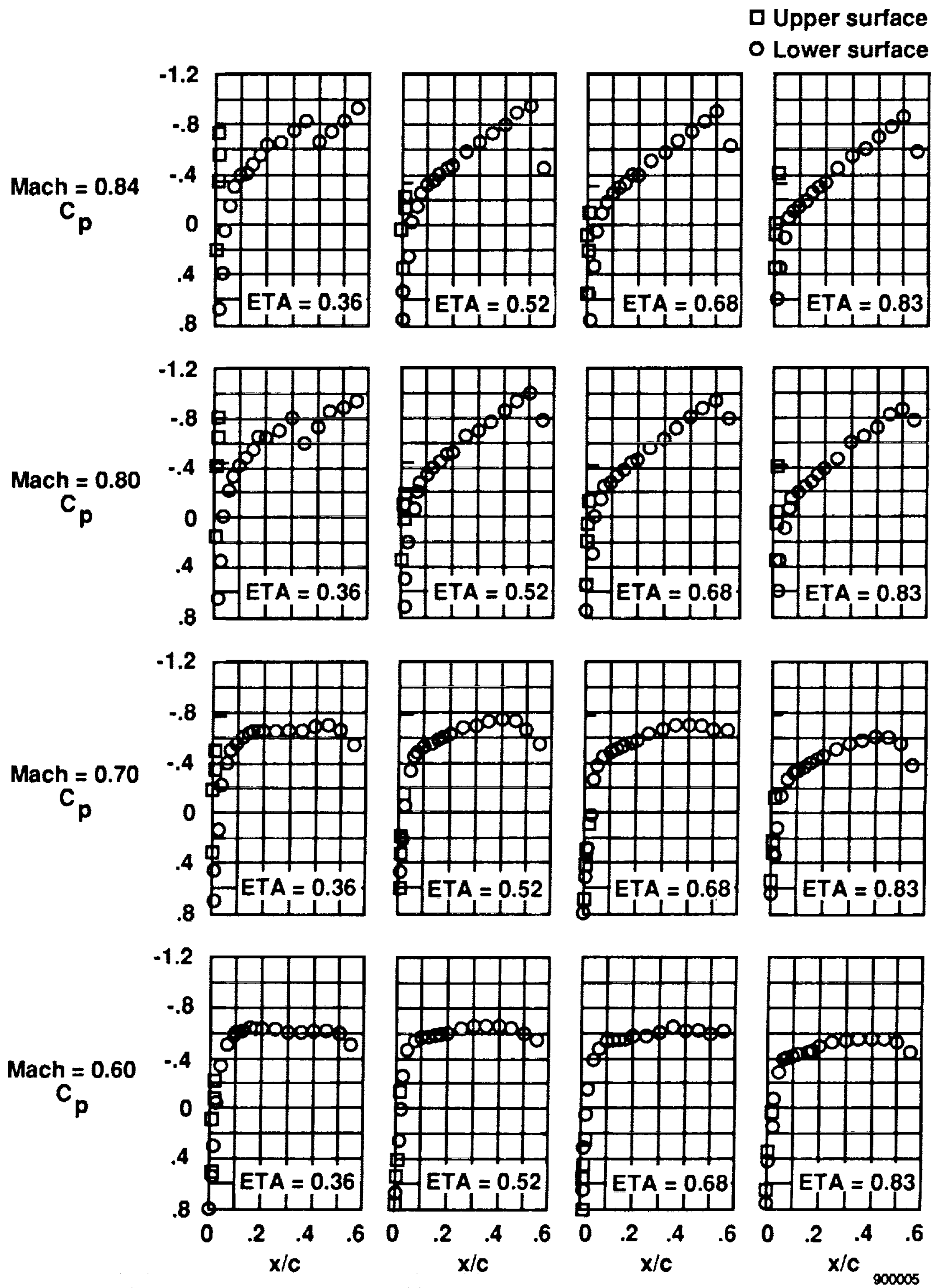

Figure 6. Right wing pressure distributions at 17,000 ft for 20° wing sweep.

a weak shock at the outboard measurement station (ETA = 0.83). A comparison of pressure distribution at Mach 0.80 and 0.84 shows that the strength of the shock has increased significantly at Mach 0.84,

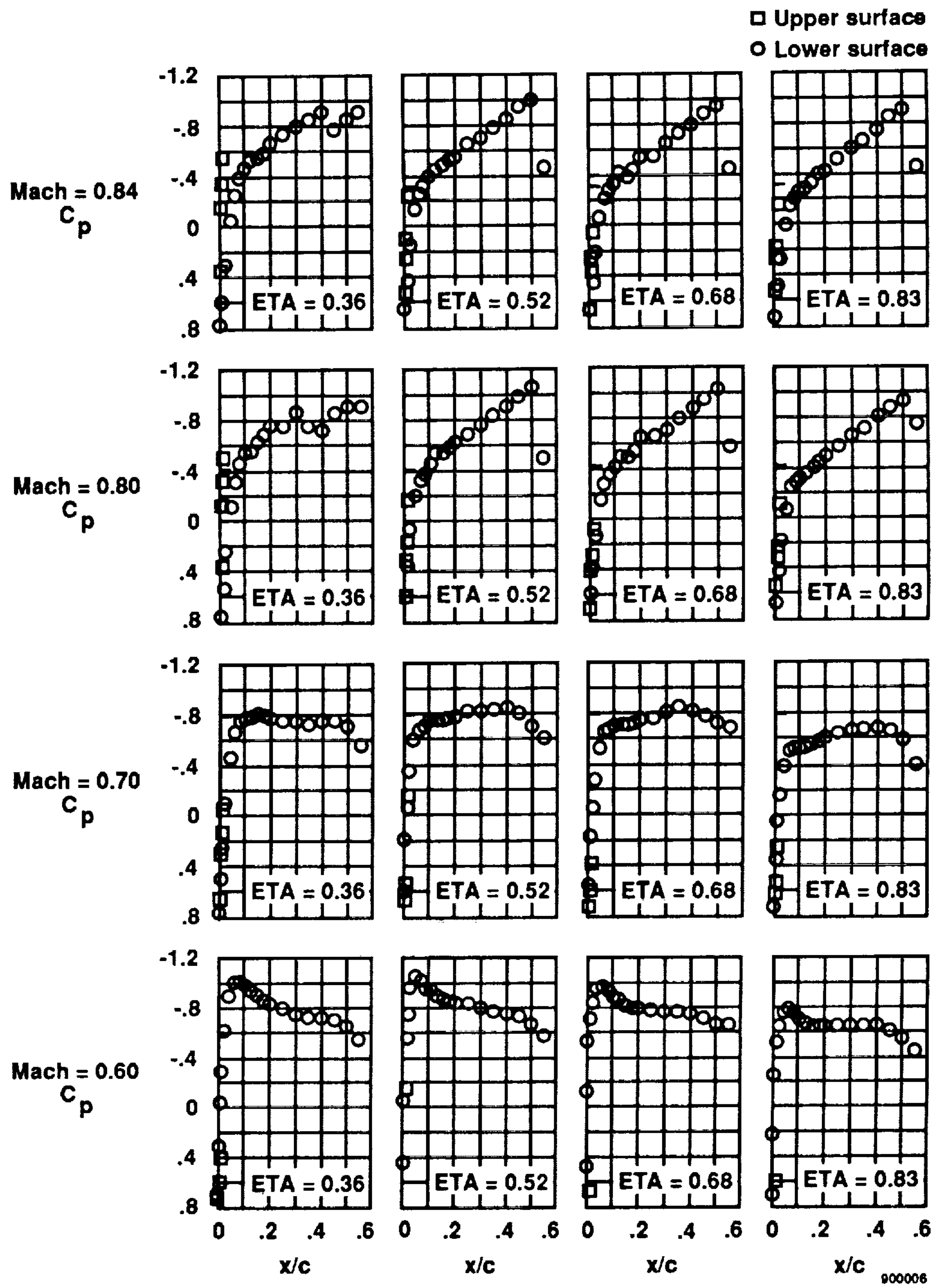

Figure 7. Right wing pressure distributions at 27,500 ft for 20° wing sweep.

particularly at the outboard section of the wing. The same comparison at 17,000 ft (fig. 6) shows similar results except that the strength of the shock is notably less.

A comparison of the left and right wing pressure distribution data is shown in figure 8. The data provide an explanation for the increased response amplitude of the right wing. The right wing experienced a significant increase in response amplitude because of the shock formation over the wing outboard section. The amplitude of the left wing response was less since the shock had not completely formed over the outboard section. These data indicate a shock formation over the entire outboard section of the right wing but not over the tip section of the left wing. The airplane configured with the glove on the left wing only did not encounter Mach buffet until Mach 0.87.

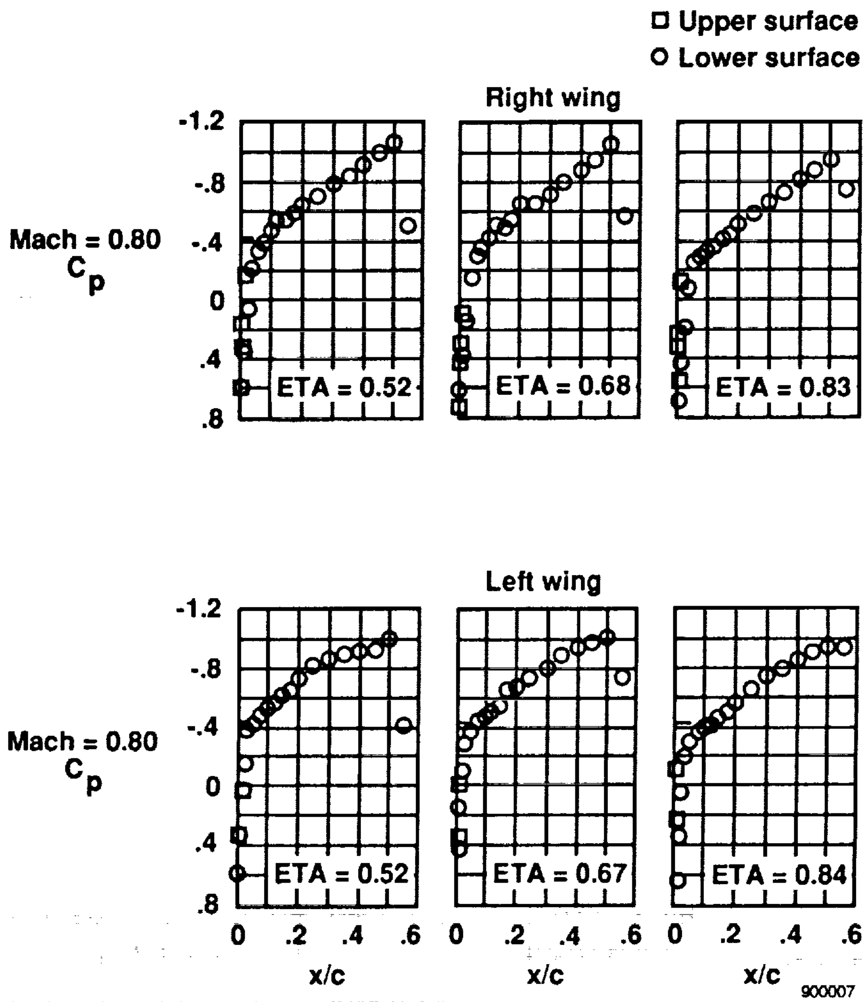

Figure 8. Comparison of left and right wing pressure distributions at 27,500 ft for 20° wing sweep.

Figures 9 and 10 show the shock's effect on structural mode excitation. As the Mach number increased to 0.80, the amplitude of the structural response remained relatively constant. At Mach 0.84, there was a significant increase in the structural response. Flight flutter testing was then terminated because the structural response might be of sufficient amplitude to crack the right wing glove.

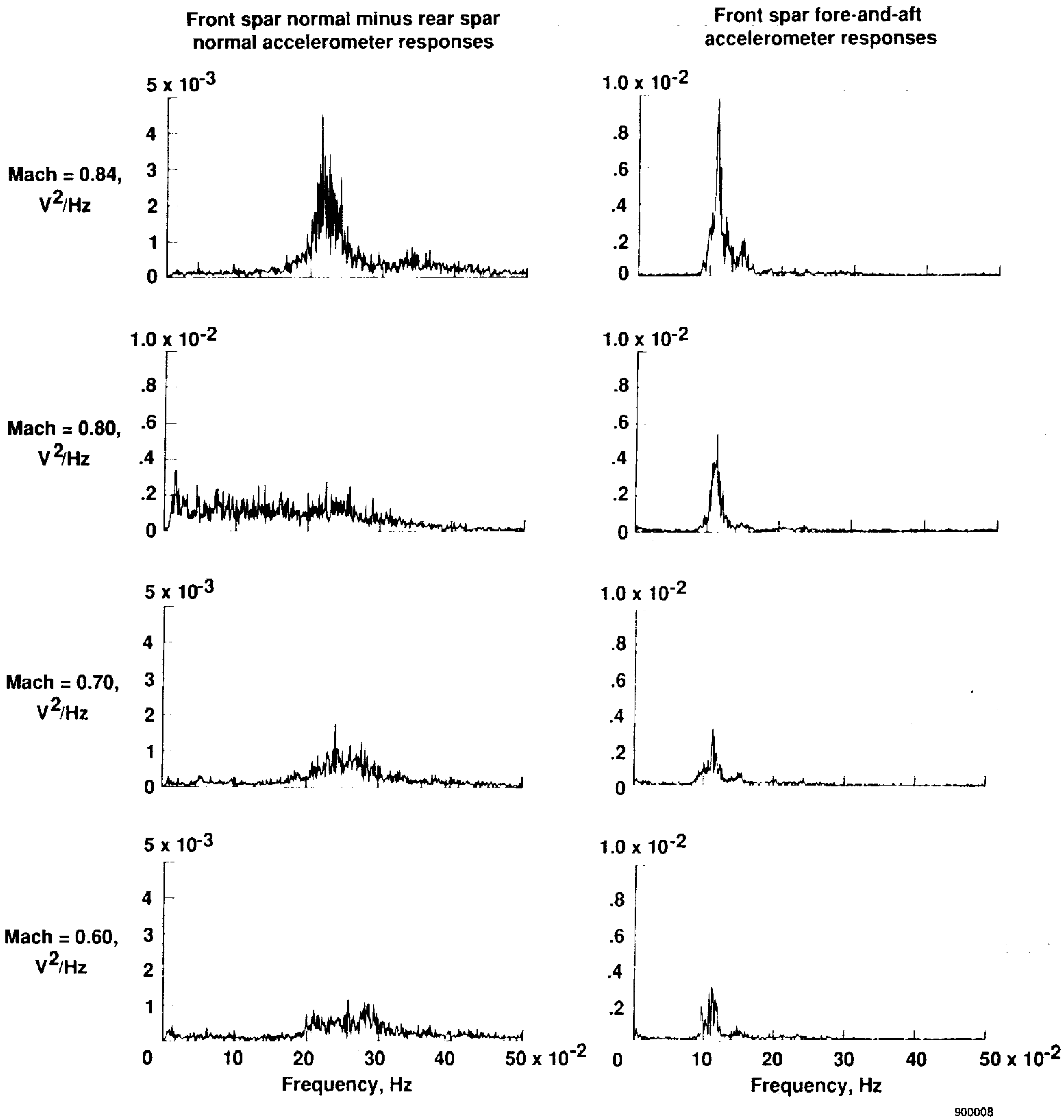

Figure 9. Right wing accelerometer response power spectra at 17,000 ft.

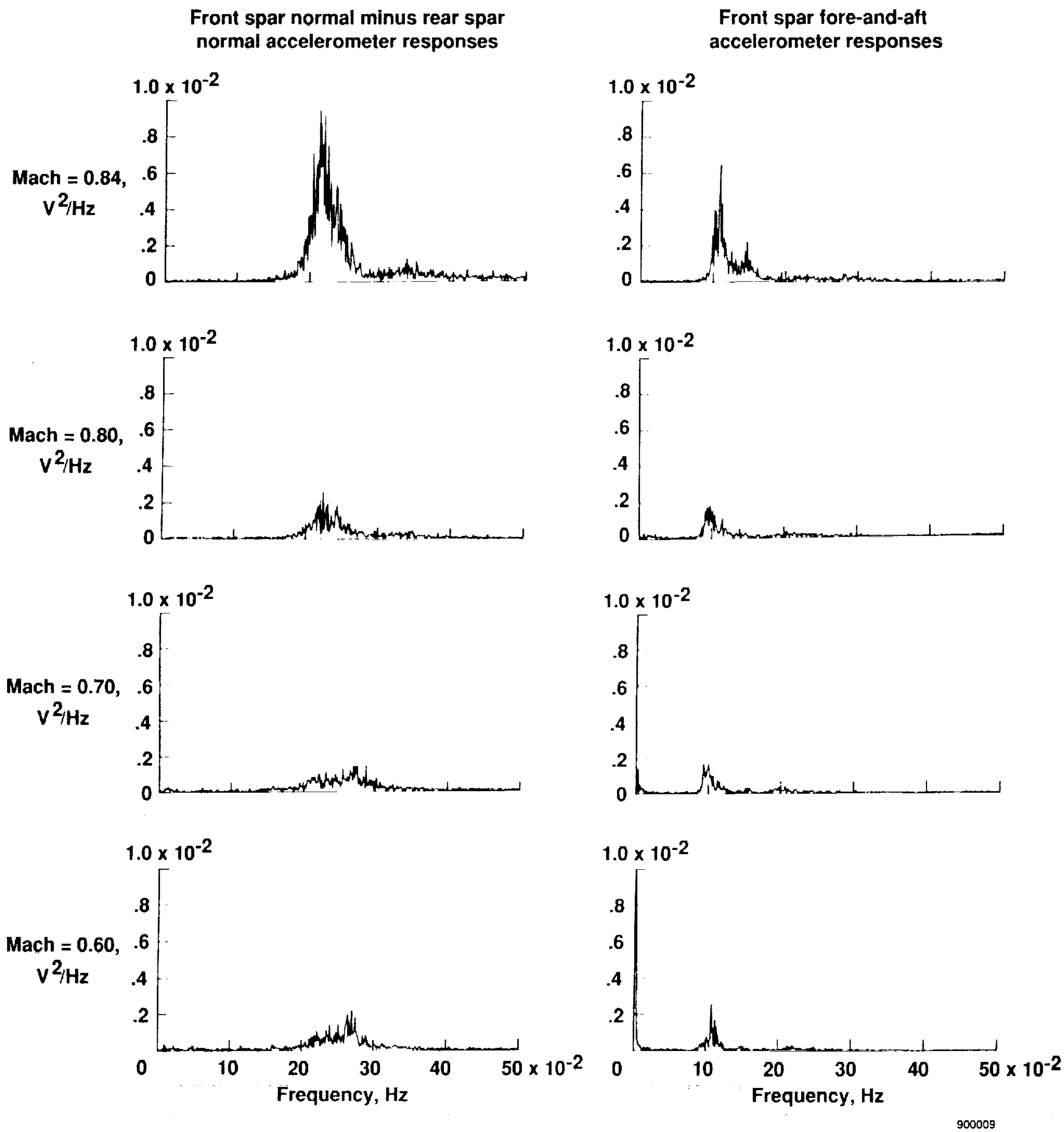

Figure 10. Right wing accelerometer response power spectra at 27,500 ft.

Figure 11 shows the cleared flight flutter envelope for the laminar flow research program. The envelope has a do-not-exceed limit of Mach 0.84 and is valid for wing sweeps of 20° and greater for this aircraft.

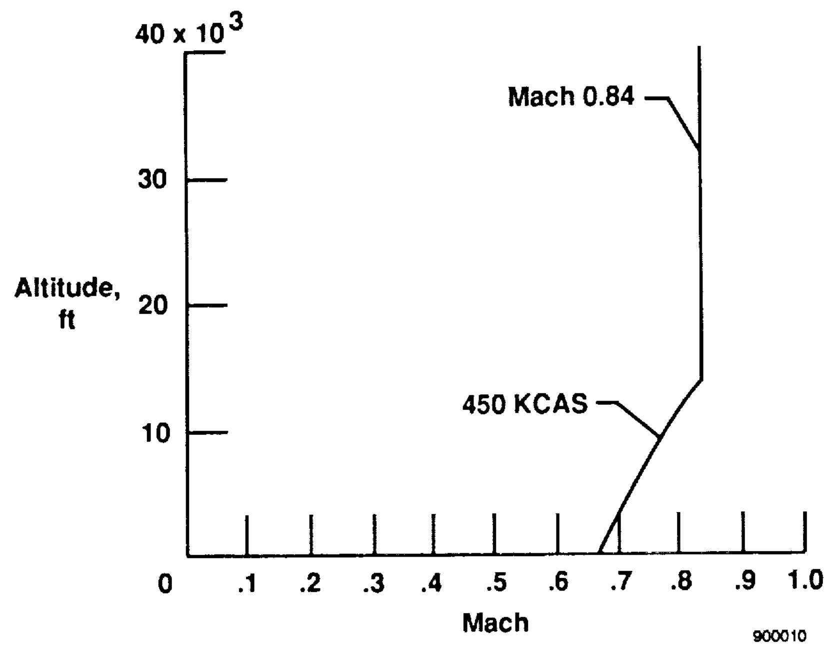

Figure 11. Cleared flight envelope.

CONCLUSIONS

A ground vibration test was conducted on an F-14A research aircraft modified with an aerodynamic glove on each wing to determine the change in modal characteristics. Results indicated that the addition of the foam and fiberglass glove did not affect the wing bending modes but did alter the frequency of the torsion modes. The change in modal response characteristics was sufficient to warrant a flight flutter test.

Flight flutter testing was conducted with the modified aircraft at altitudes of 5,000, 17,000, and 27,500 ft to maximum Mach numbers of 0.74, 0.80, and 0.84, respectively. Damping levels were satisfactory from a flutter standpoint for the modes monitored. To avoid cracking the wing glove, flight testing did not continue beyond Mach 0.84. As a result, the cleared flight flutter envelope for this research aircraft was Mach 0.84 or 450 knots, calibrated airspeed, whichever was less.

Ames Research Center
Dryden Flight Research Facility
National Aeronautics and Space Administration
Edwards, California, January 16, 1990

APPENDIX

FREQUENCY AND DAMPING TRENDS

Frequency and damping trends established for the airplane are presented in this appendix. The presentation of the data is a plot of frequency and damping as a function of Mach number.

Random atmospheric turbulence was used as the source of structural excitation. One disadvantage of using turbulence is that some structural modes may not be excited at every test point. An extreme example is the symmetric torsion mode at 27,500 ft where no reliable estimates could be extracted from the data.

Pilot-induced control surface pulses were also used to excite the structure but only the first wing bending mode was excited. Data acquired from these pulses are not presented in this report.

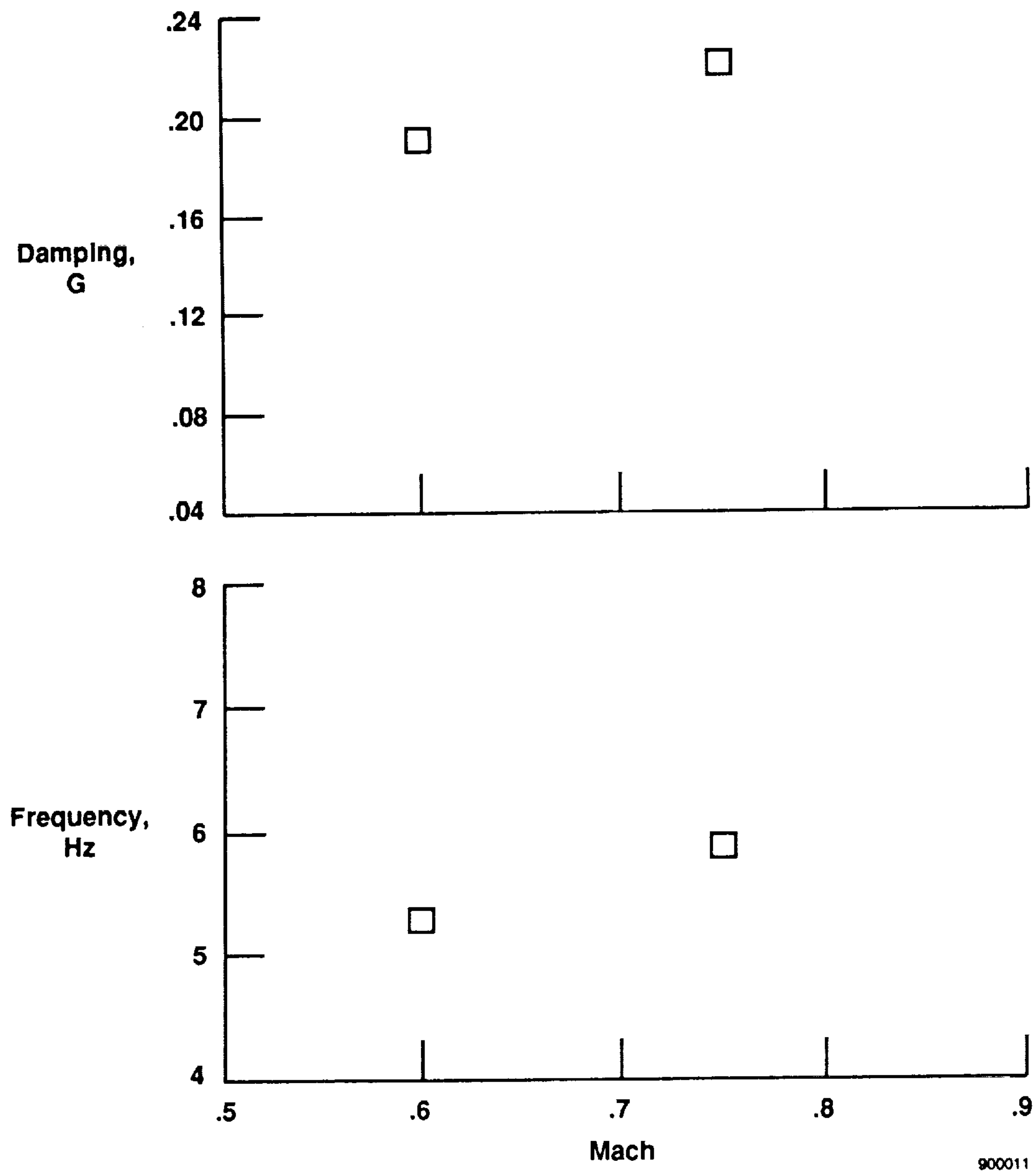

Figure A-1. First symmetric wing bending modal data at 5,000 ft.

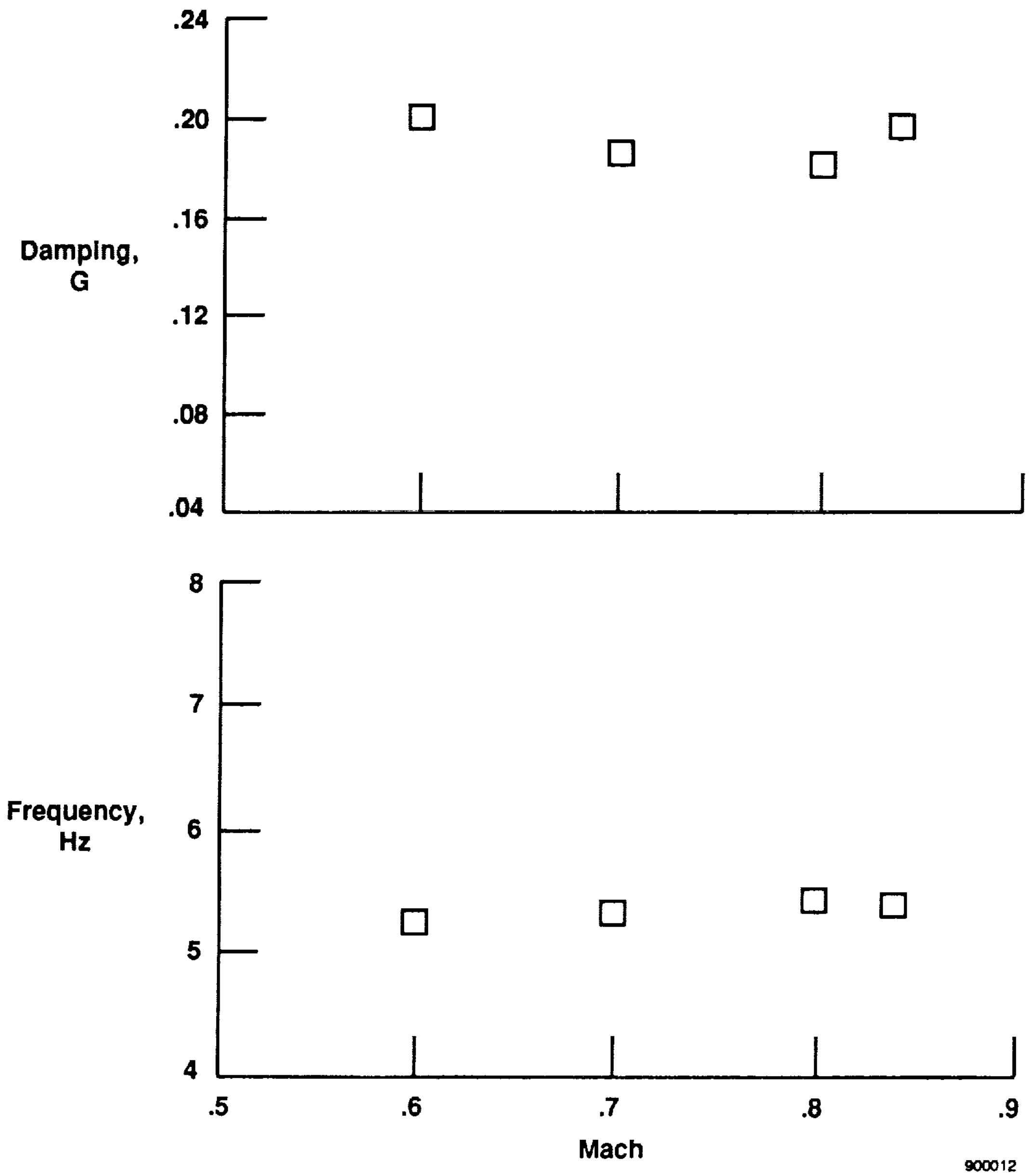

Figure A-2. First symmetric wing bending modal data at 17,000 ft.

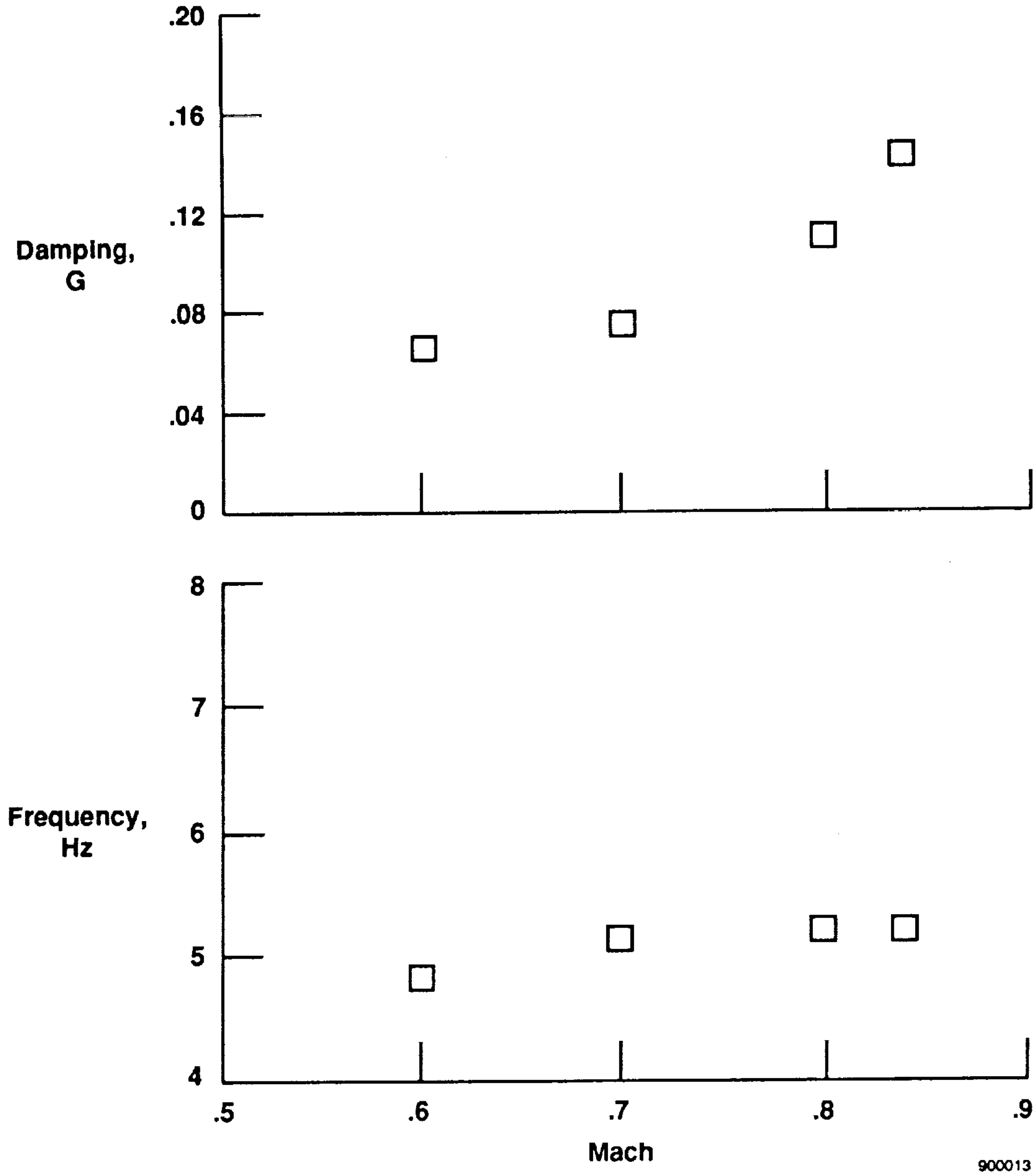

Figure A-3. First symmetric wing bending modal data at 27,500 ft.

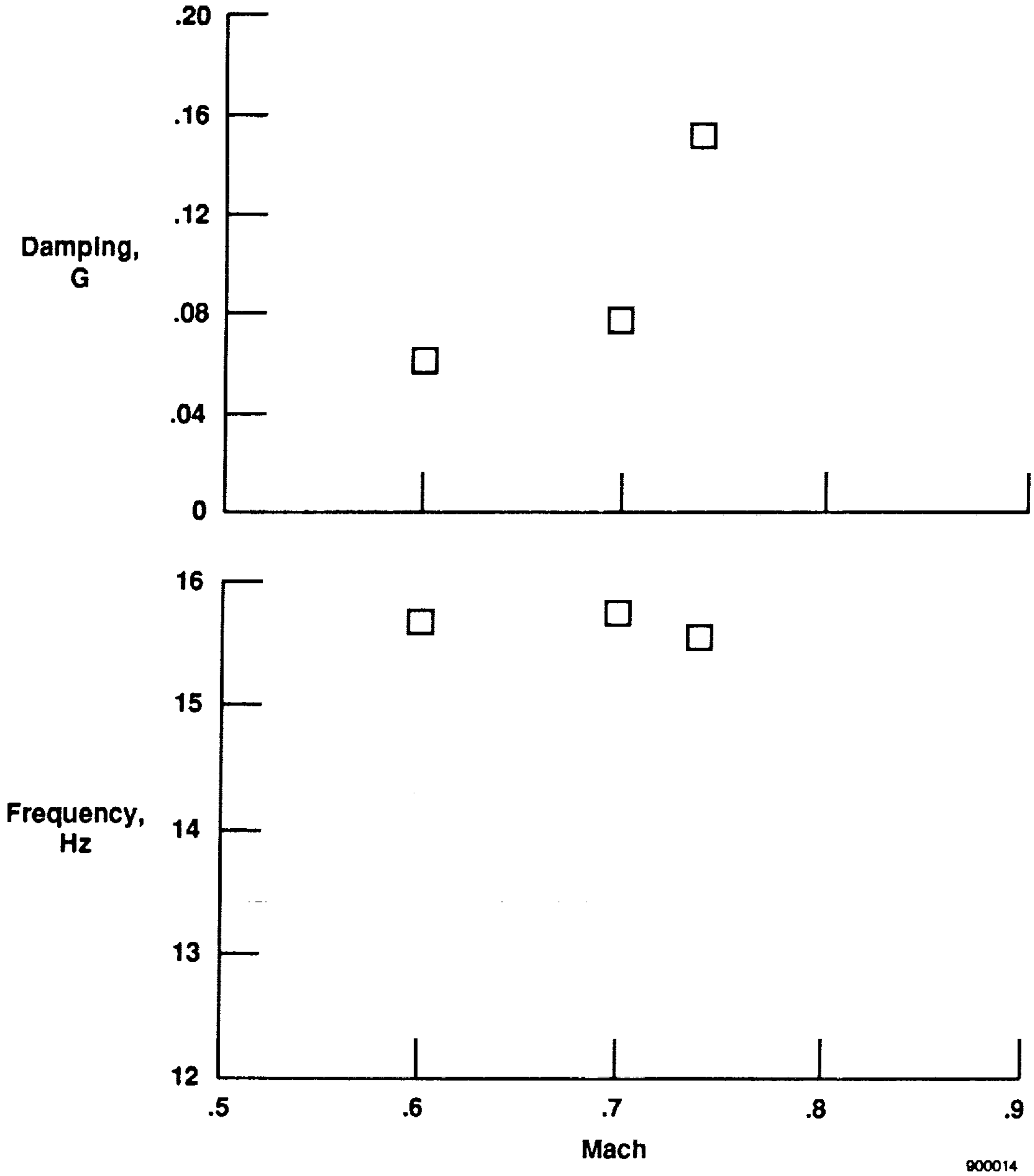

Figure A-4. Second symmetric wing bending modal data at 5,000 ft.

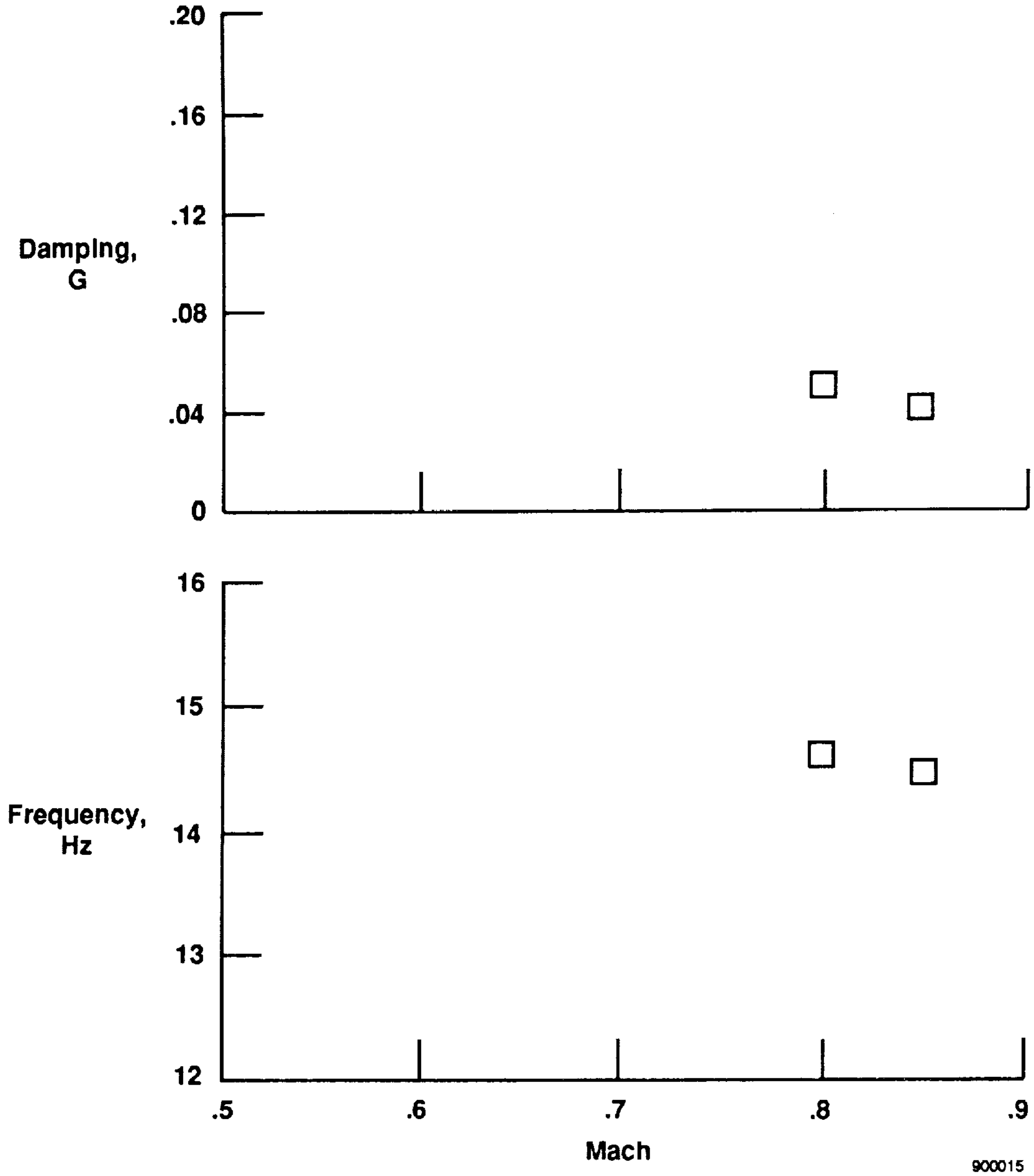

Figure A-5. Second symmetric wing bending modal data at 17,500 ft.

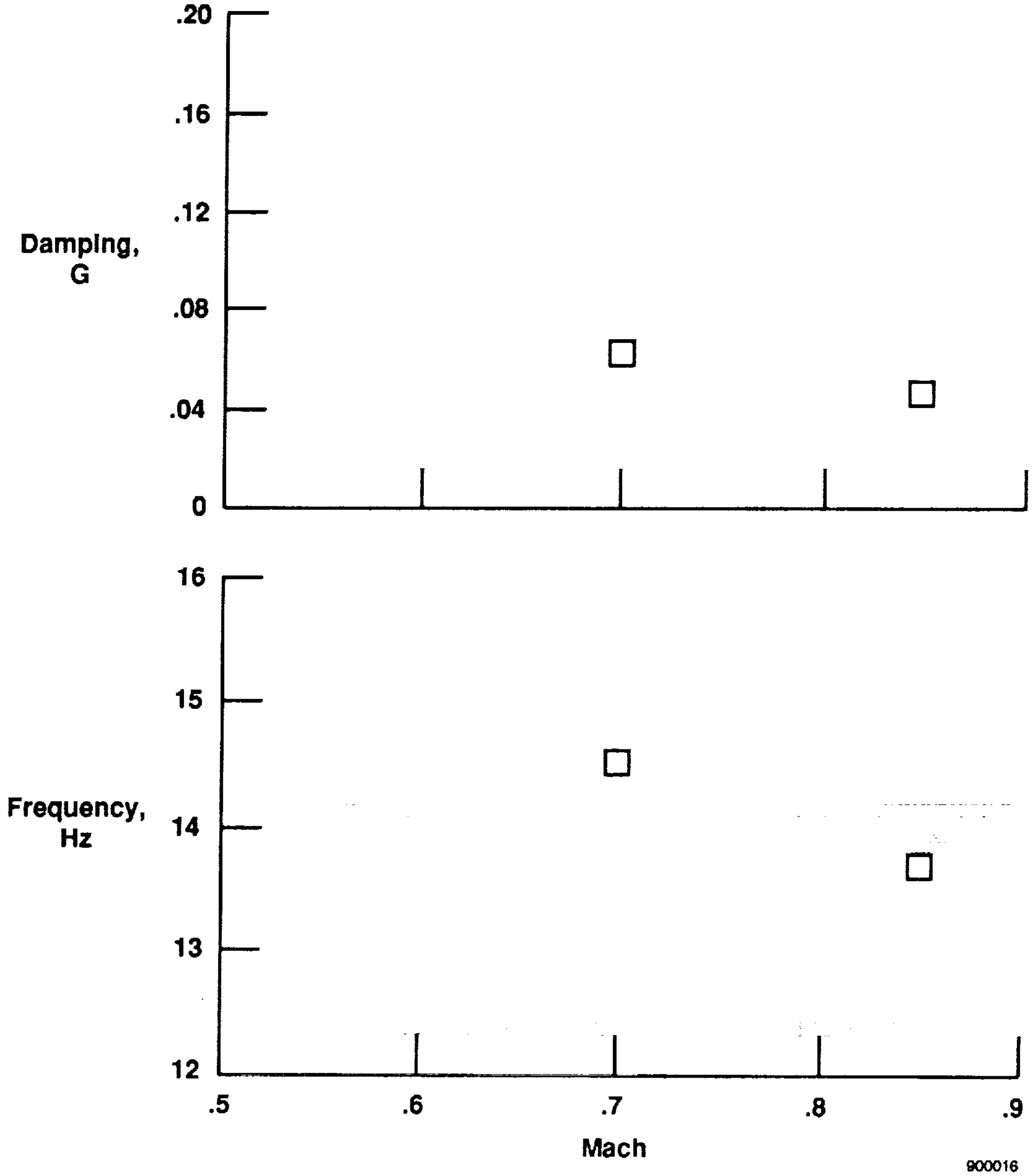

Figure A-6. Second symmetric wing bending modal data at 27,500 ft.

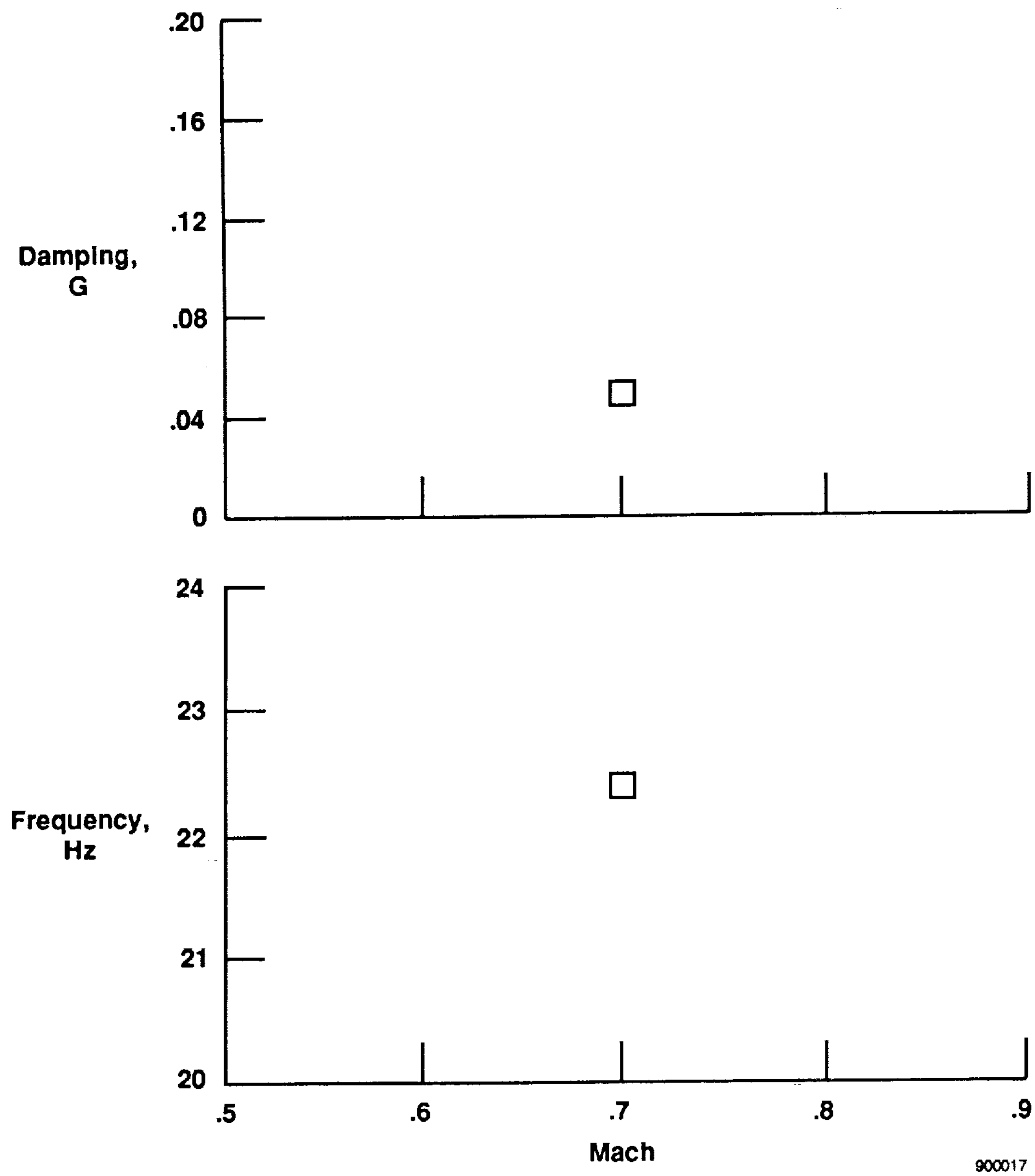

Figure A-7. First symmetric wing torsion modal data at 5,000 ft.

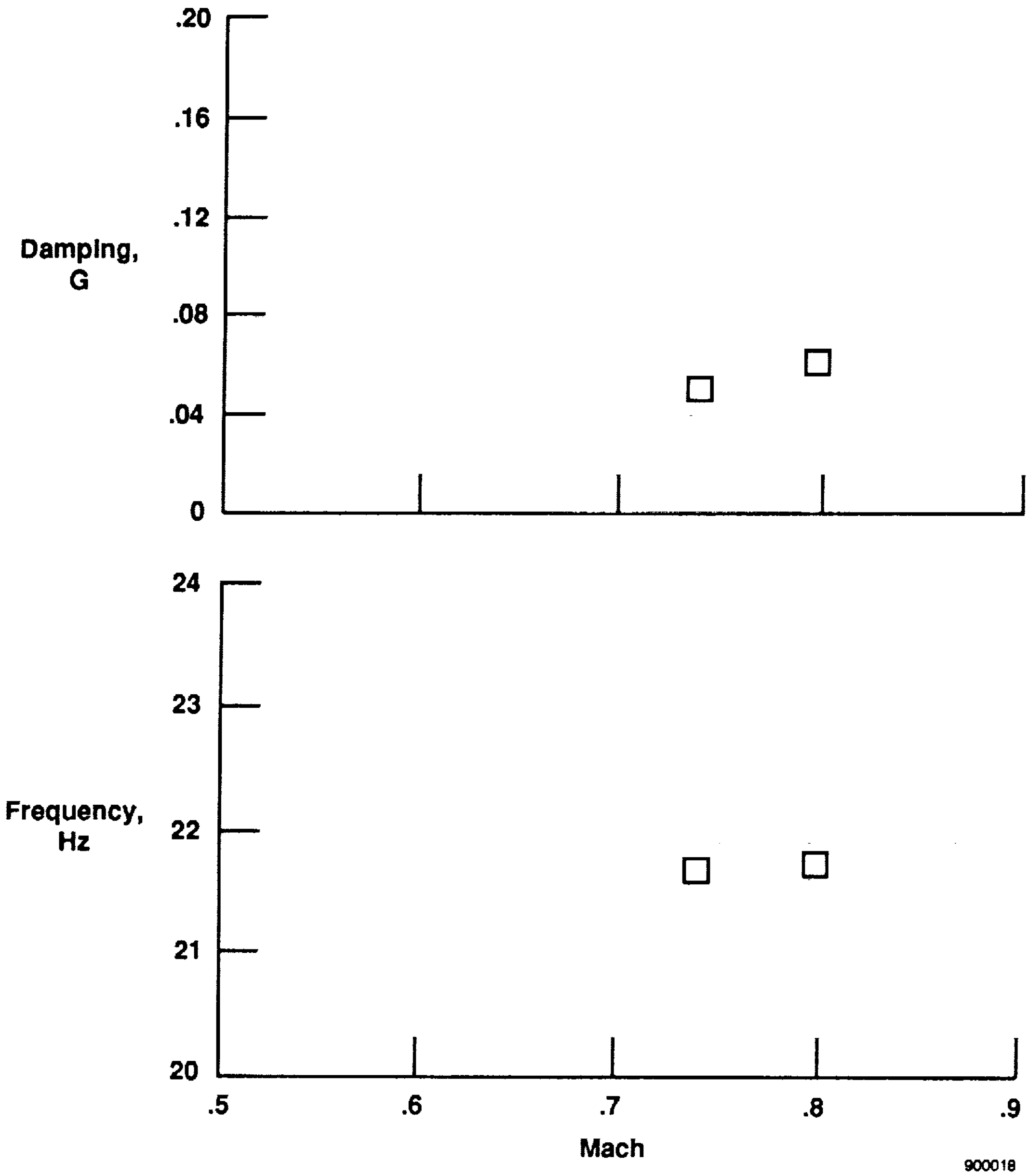

Figure A-8. First symmetric wing torsion modal data at 17,000 ft.

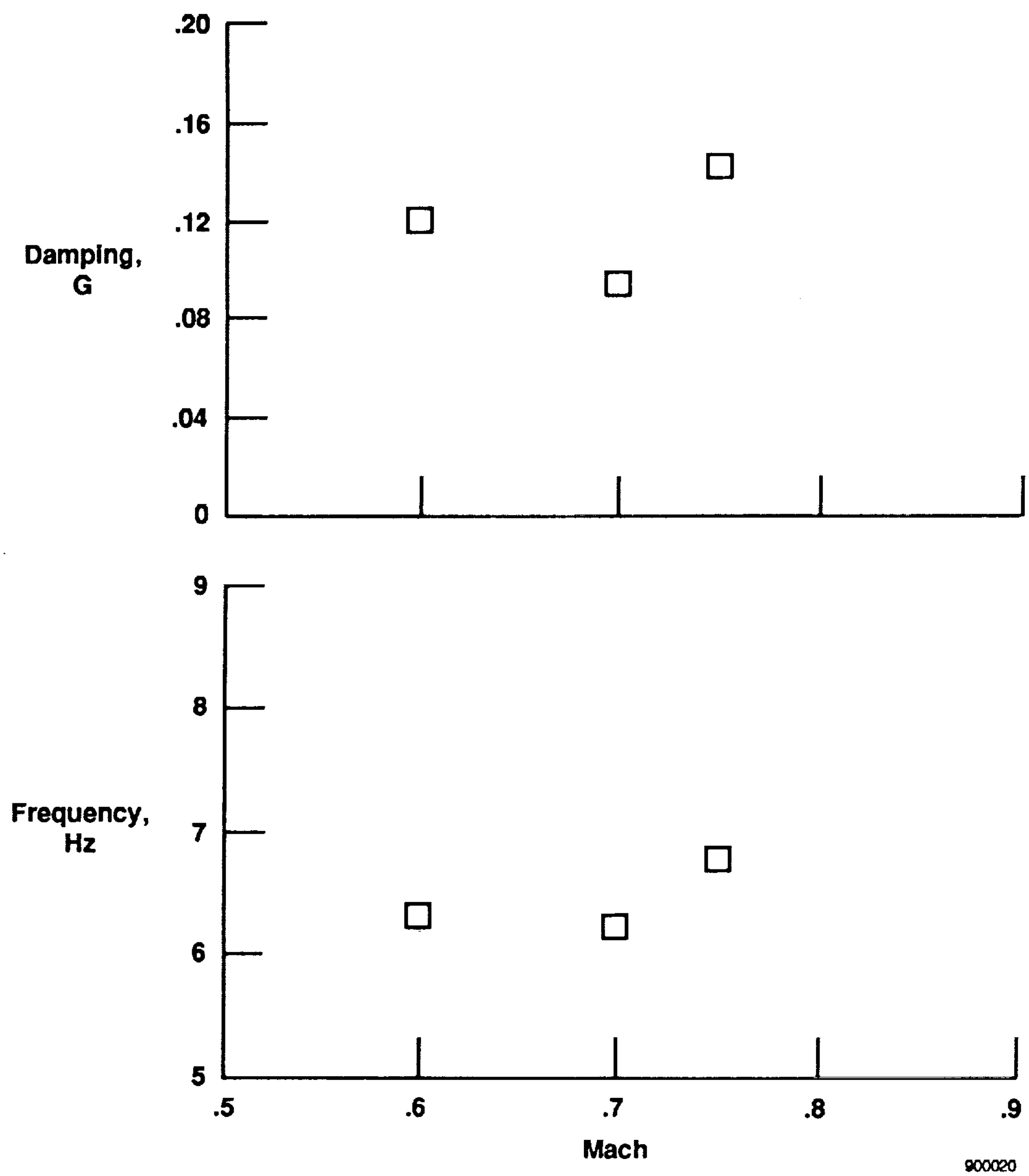

Figure A-9. First antisymmetric wing bending modal data at 5,000 ft.

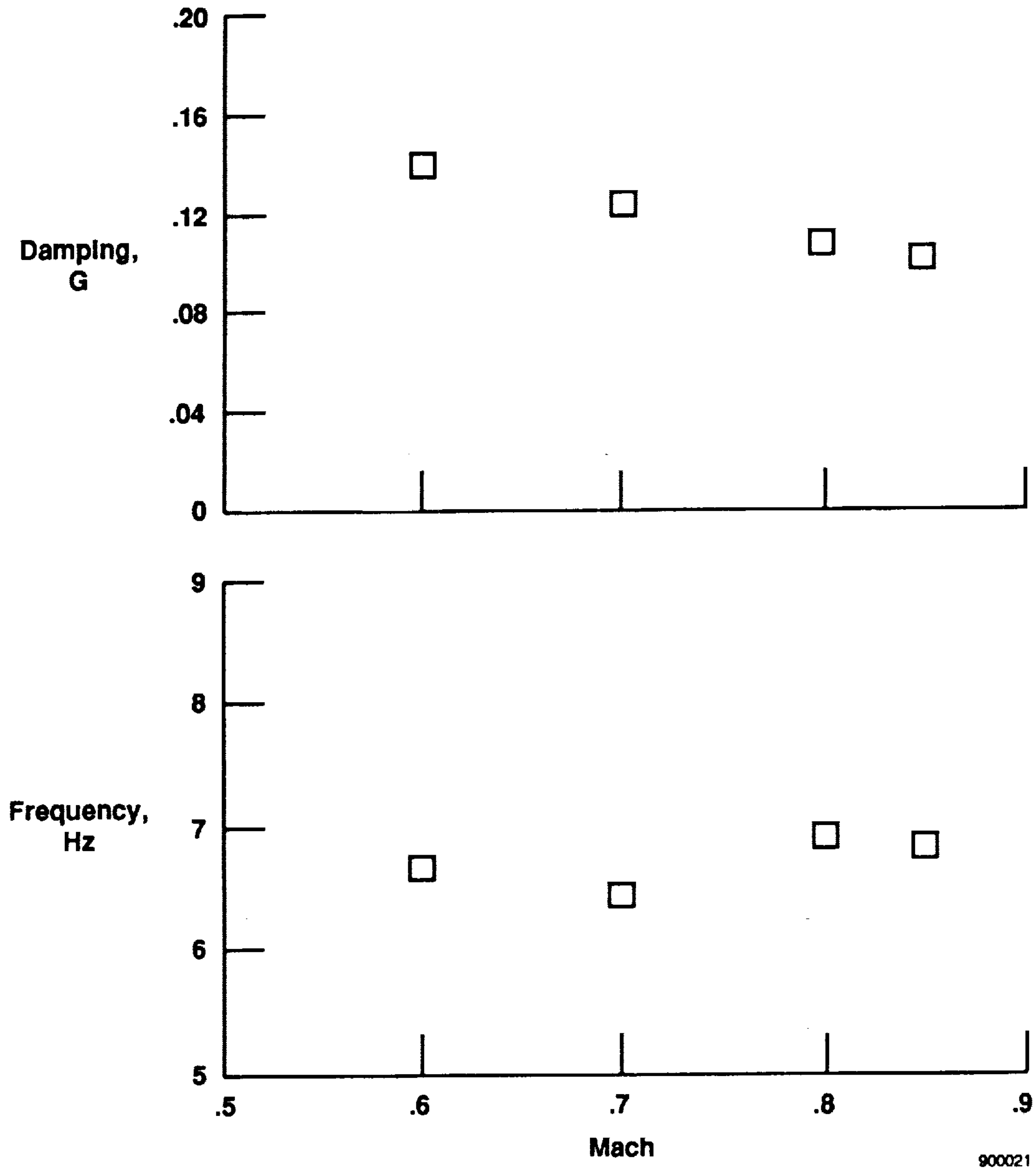

Figure A-10. First antisymmetric wing bending modal data at 17,000 ft.

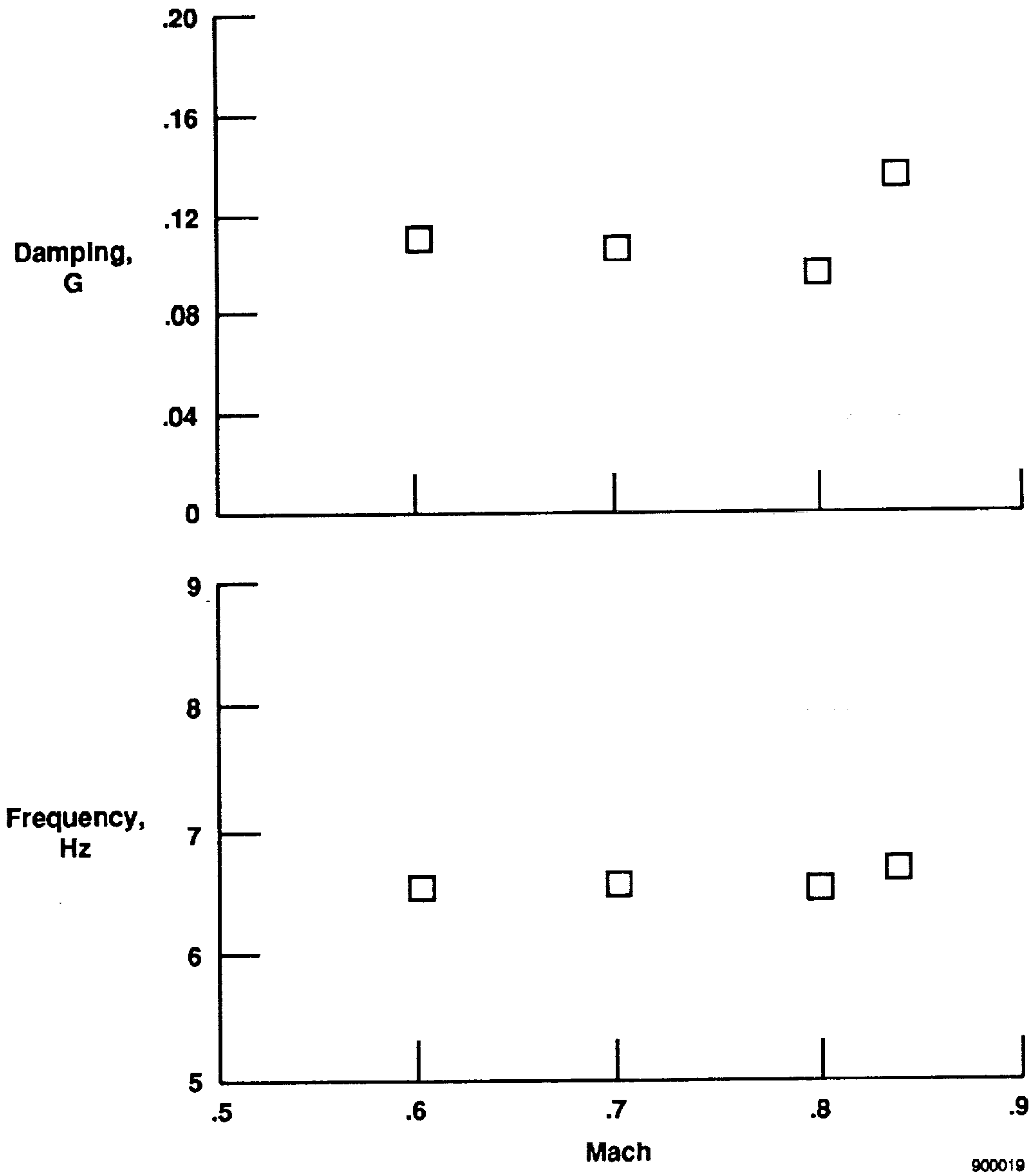

Figure A-11. First antisymmetric wing bending modal data at 27,500 ft.

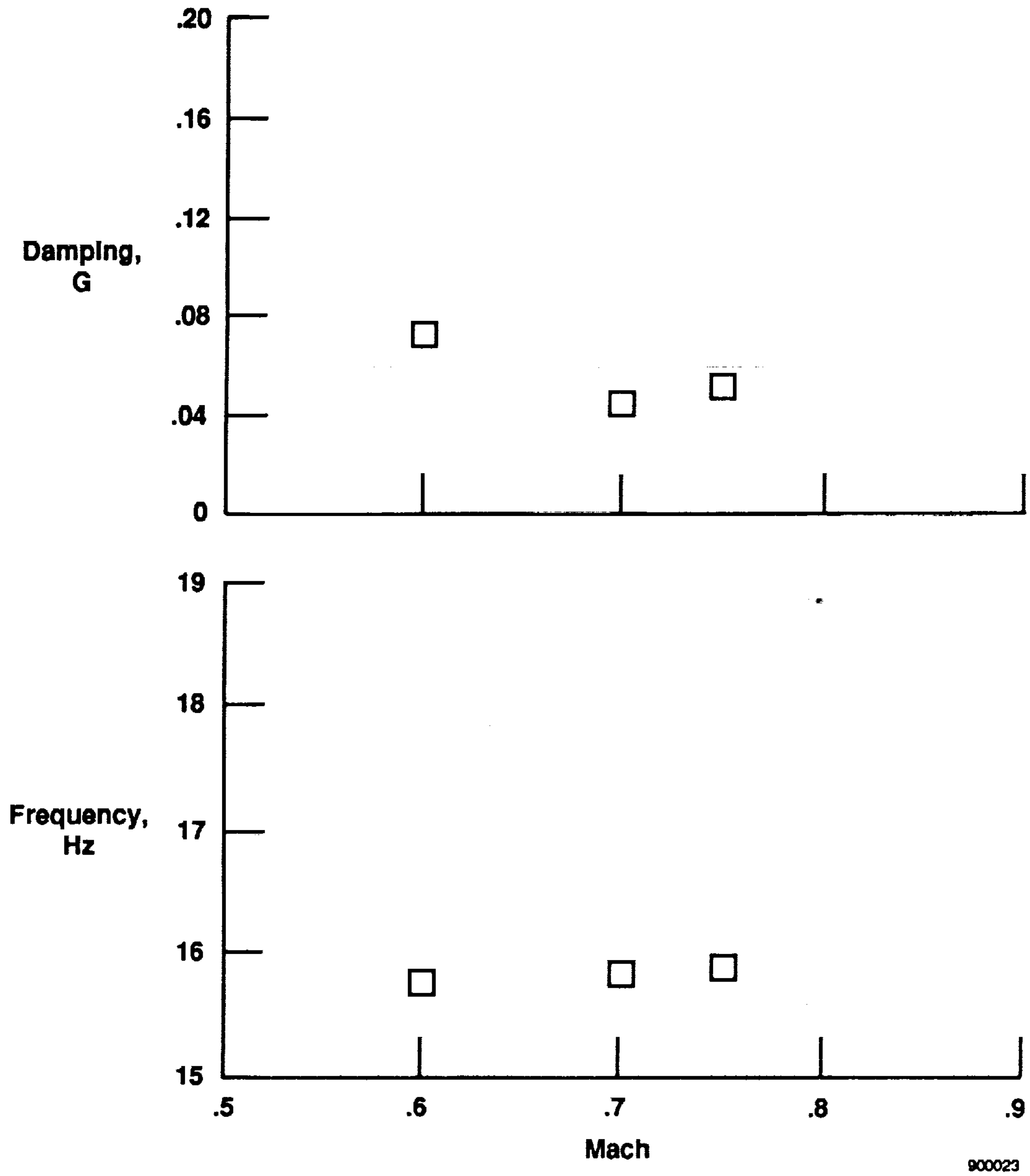

Figure A-12. Second antisymmetric wing bending modal data at 5,000 ft.

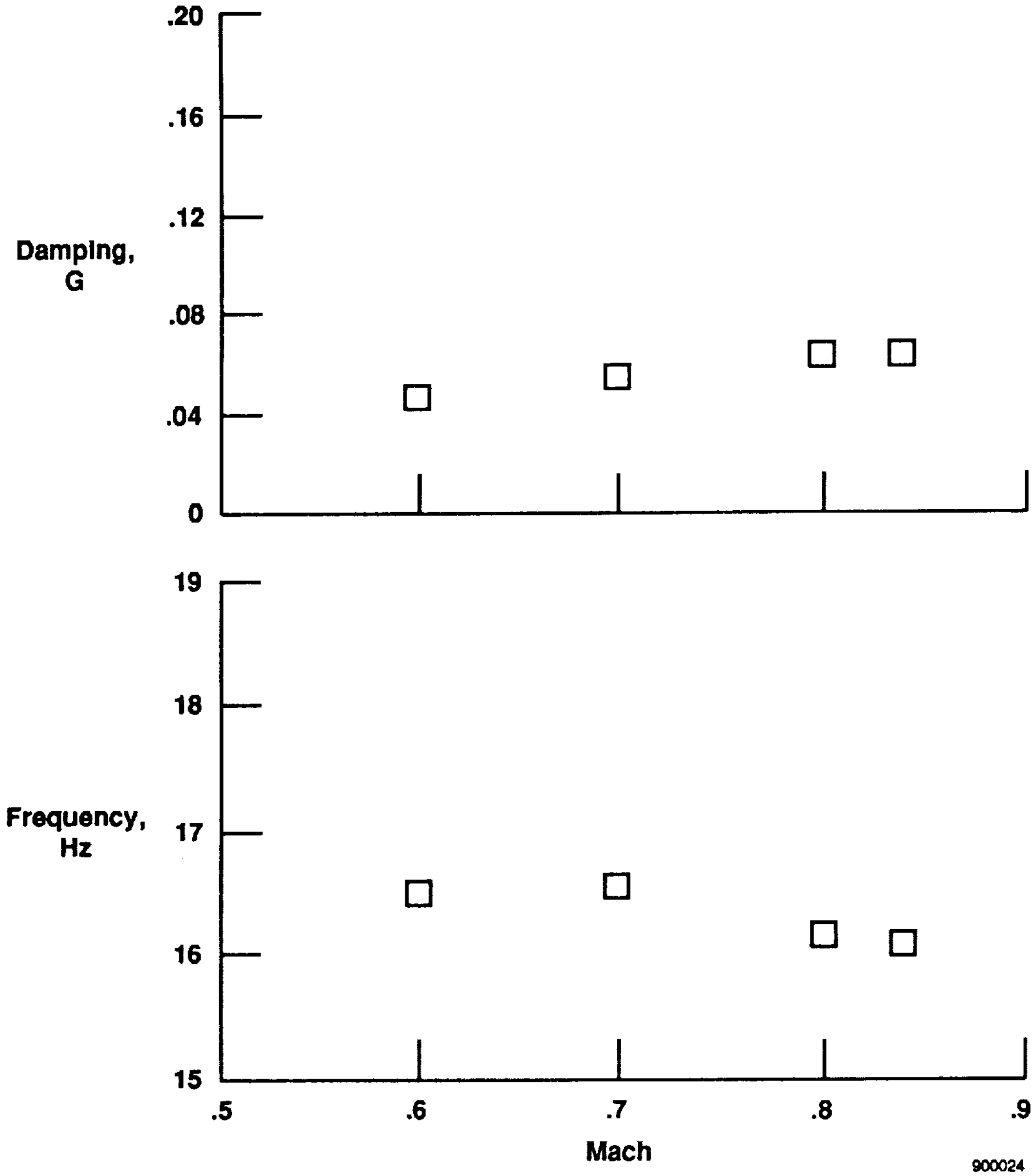

Figure A-13. Second antisymmetric wing bending modal data at 17,000 ft.

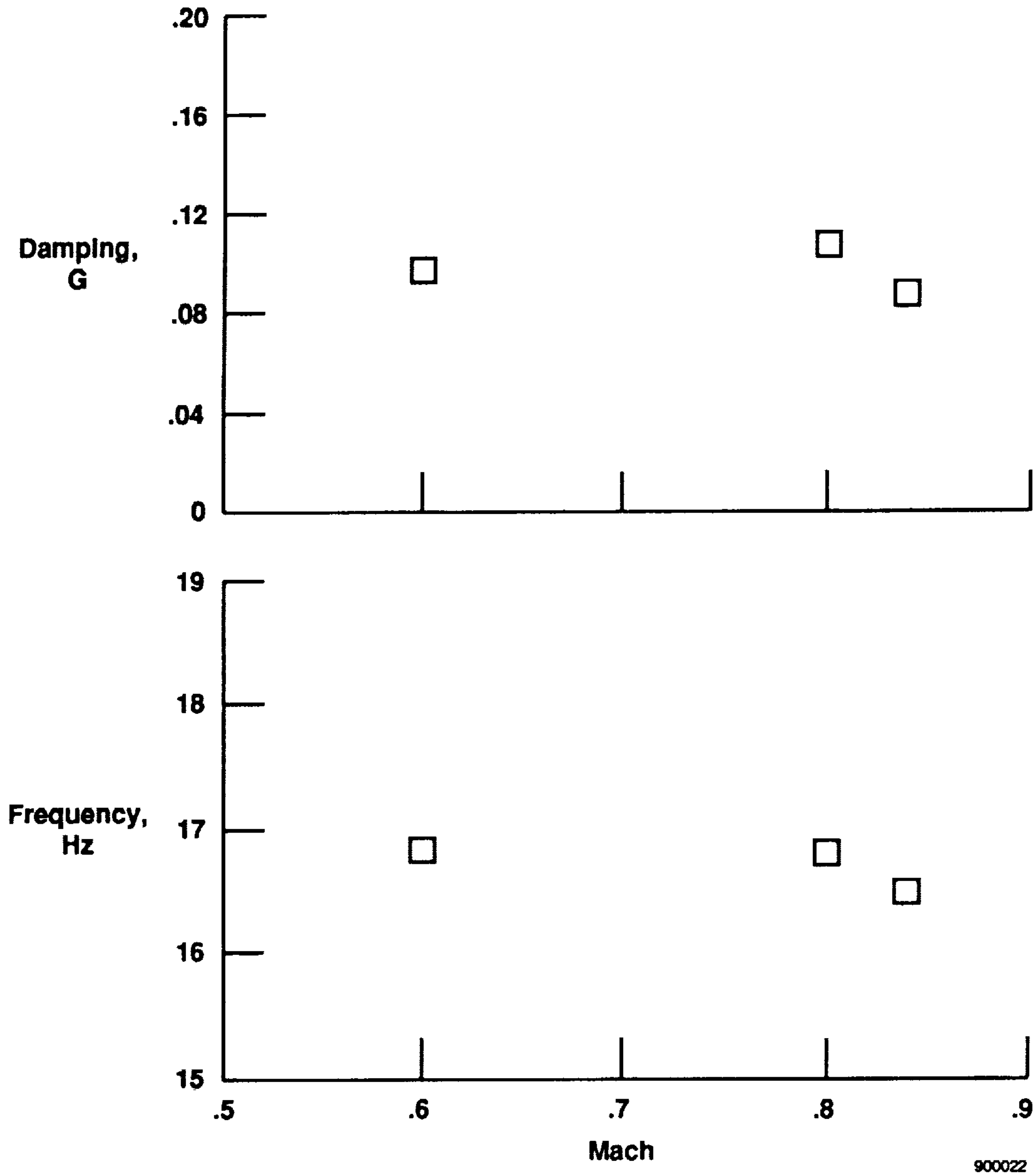

Figure A-14. Second antisymmetric wing bending modal data at 27,500 ft.

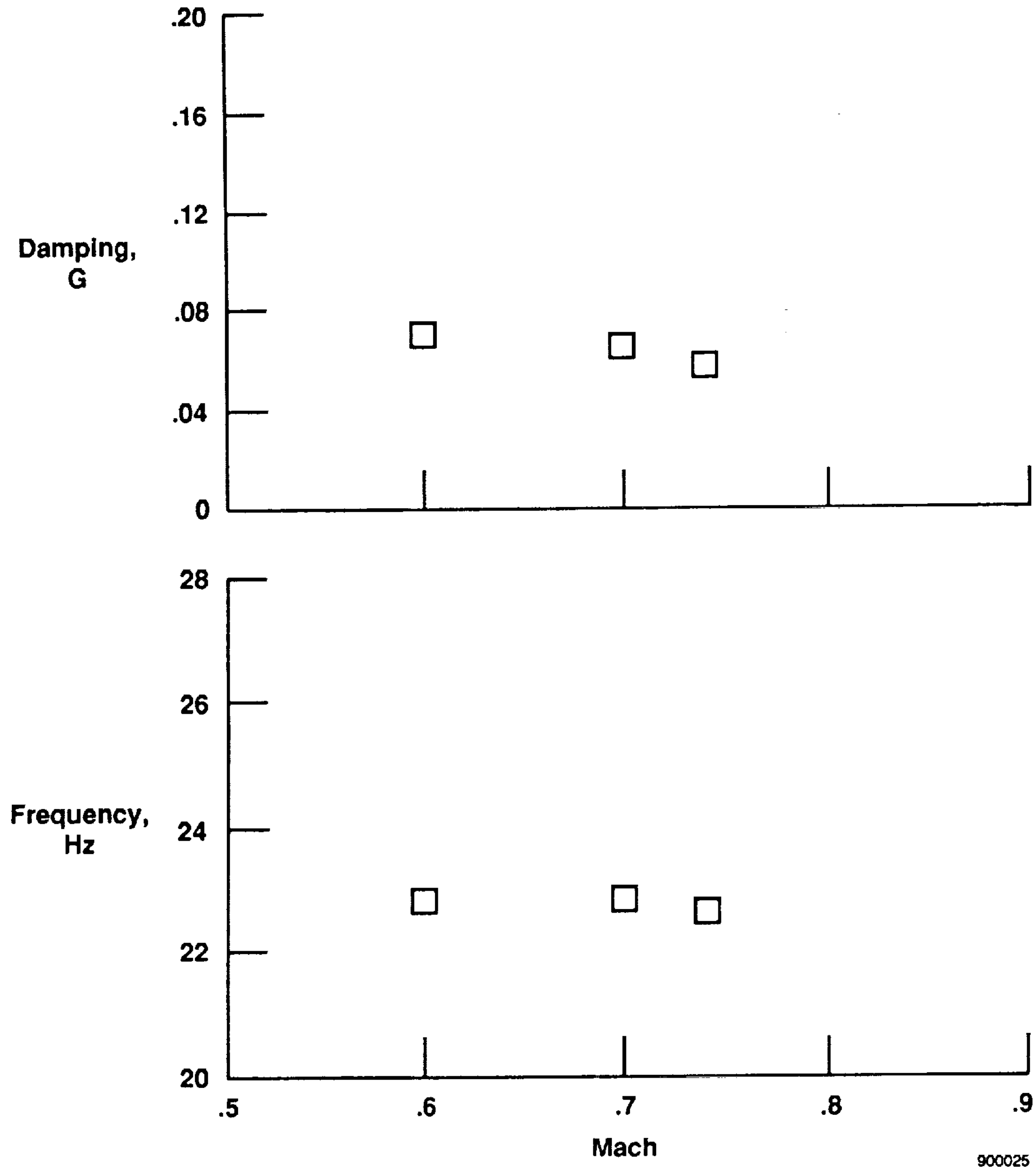

Figure A-15. First antisymmetric wing torsion modal data at 5,000 ft.

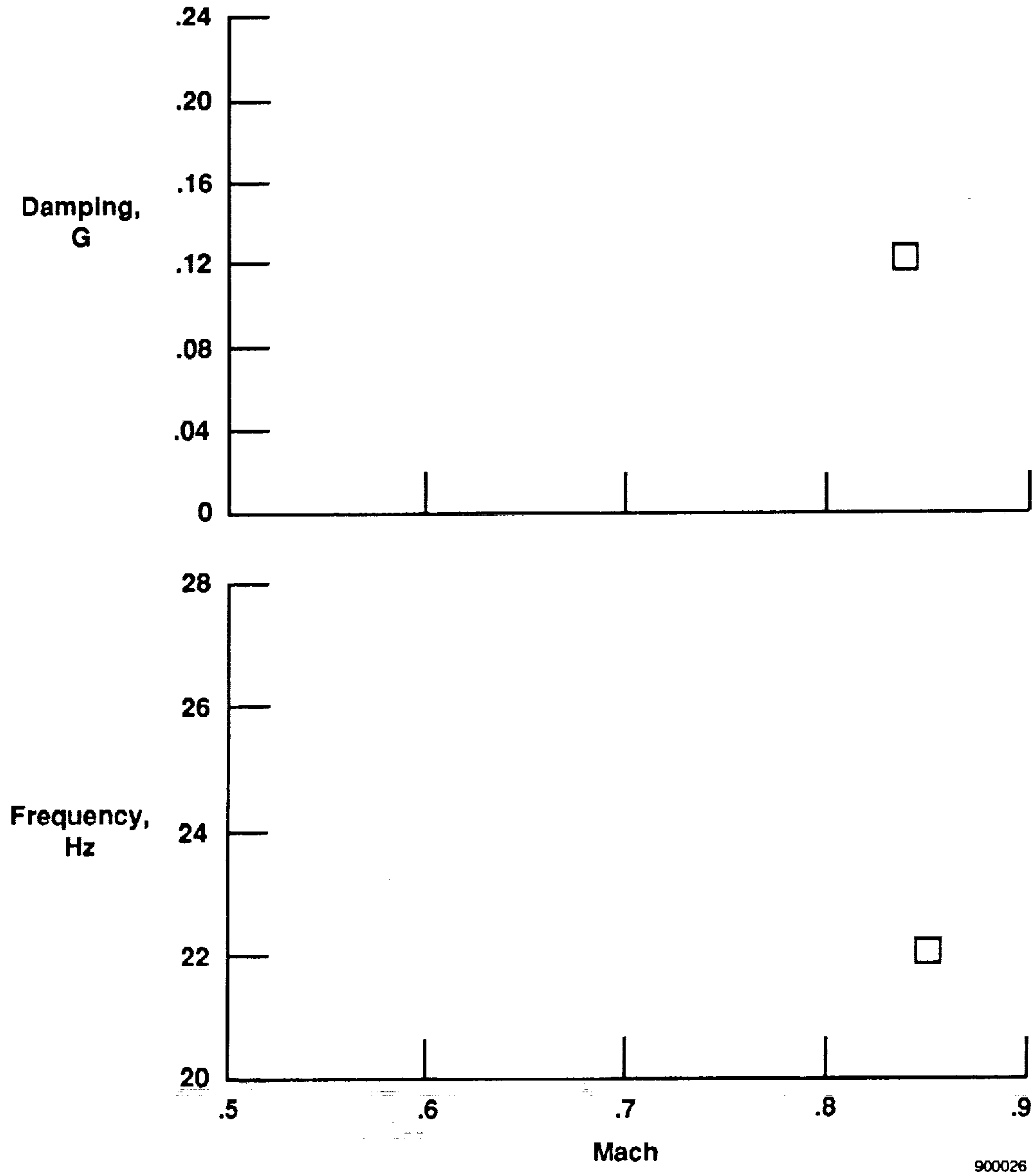

Figure A-16. First antisymmetric wing torsion modal data at 17,500 ft.

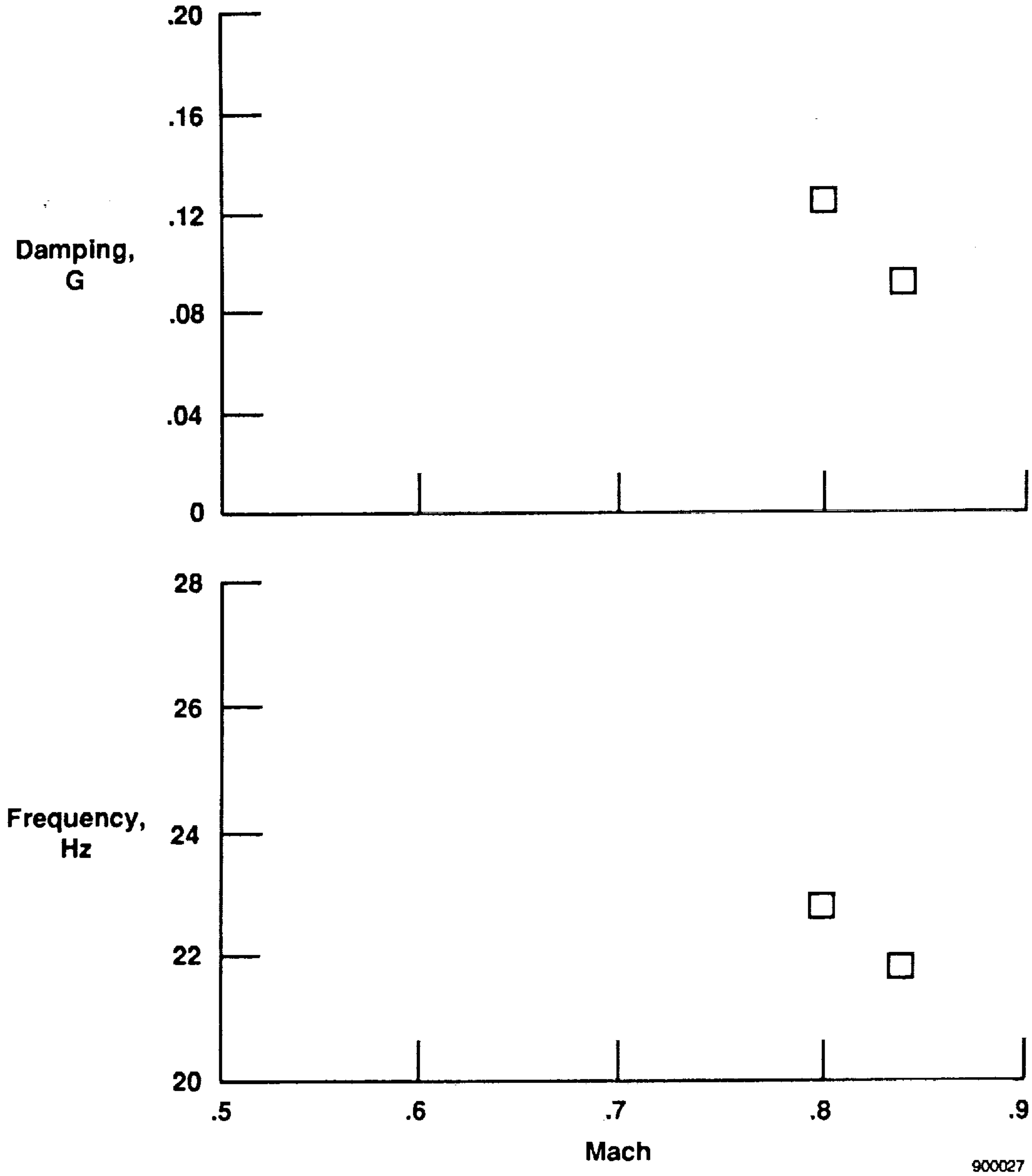

Figure A-17. First antisymmetric wing torsion modal data at 27,500 ft.

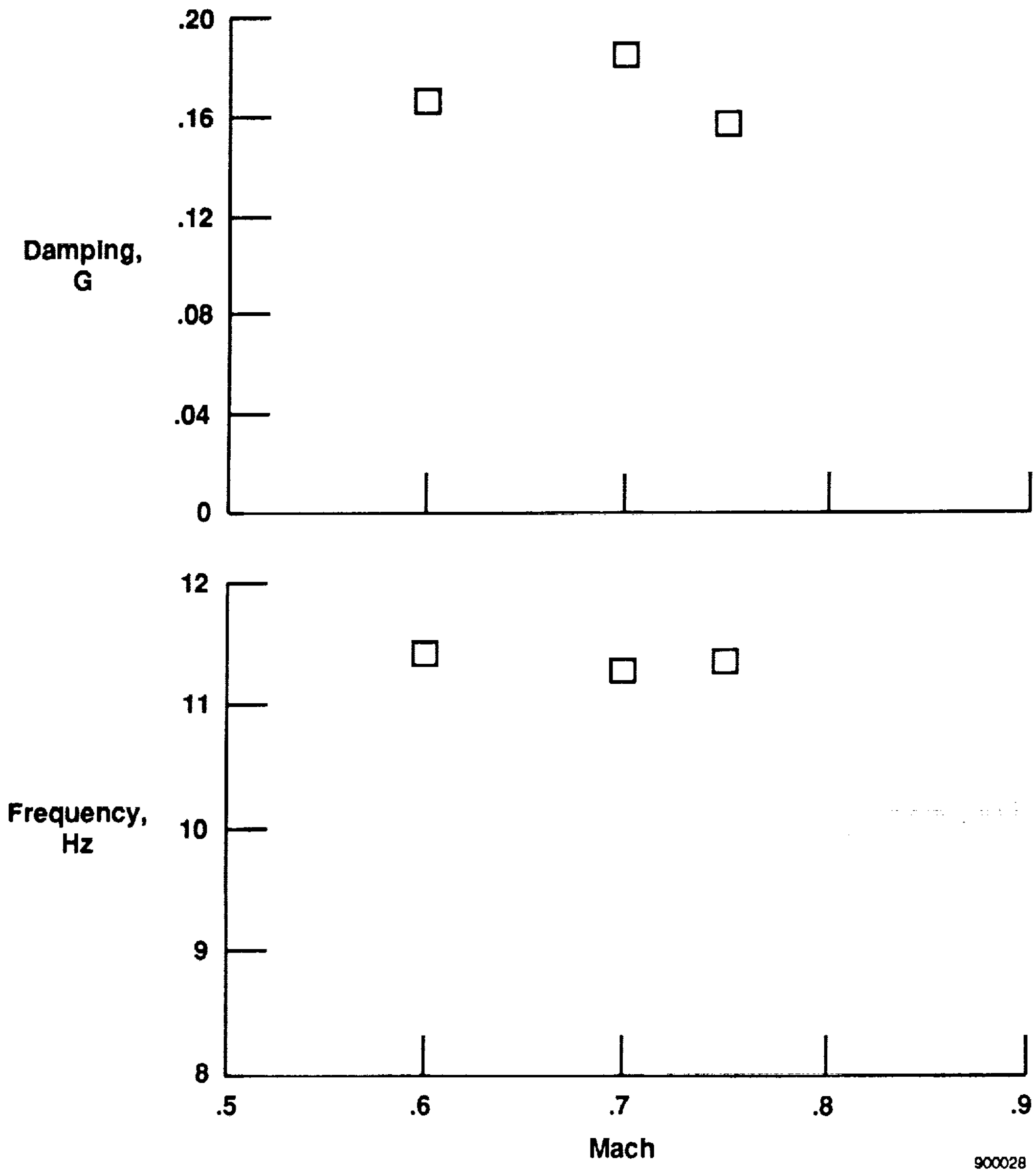

Figure A-18. Wing fore-and-aft modal data at 5,000 ft.

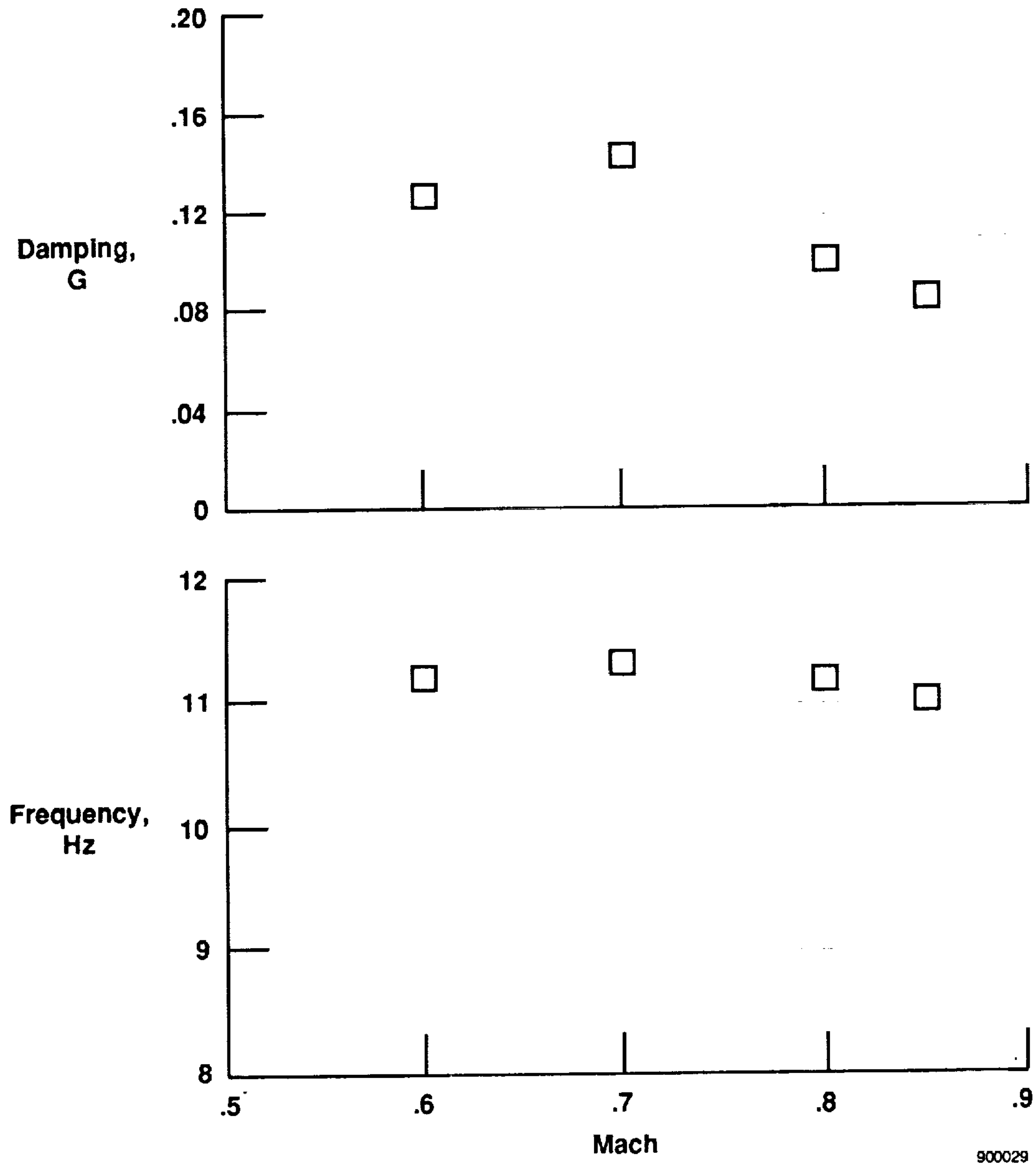

Figure A-19. Wing fore-and-aft modal data at 17,000 ft.

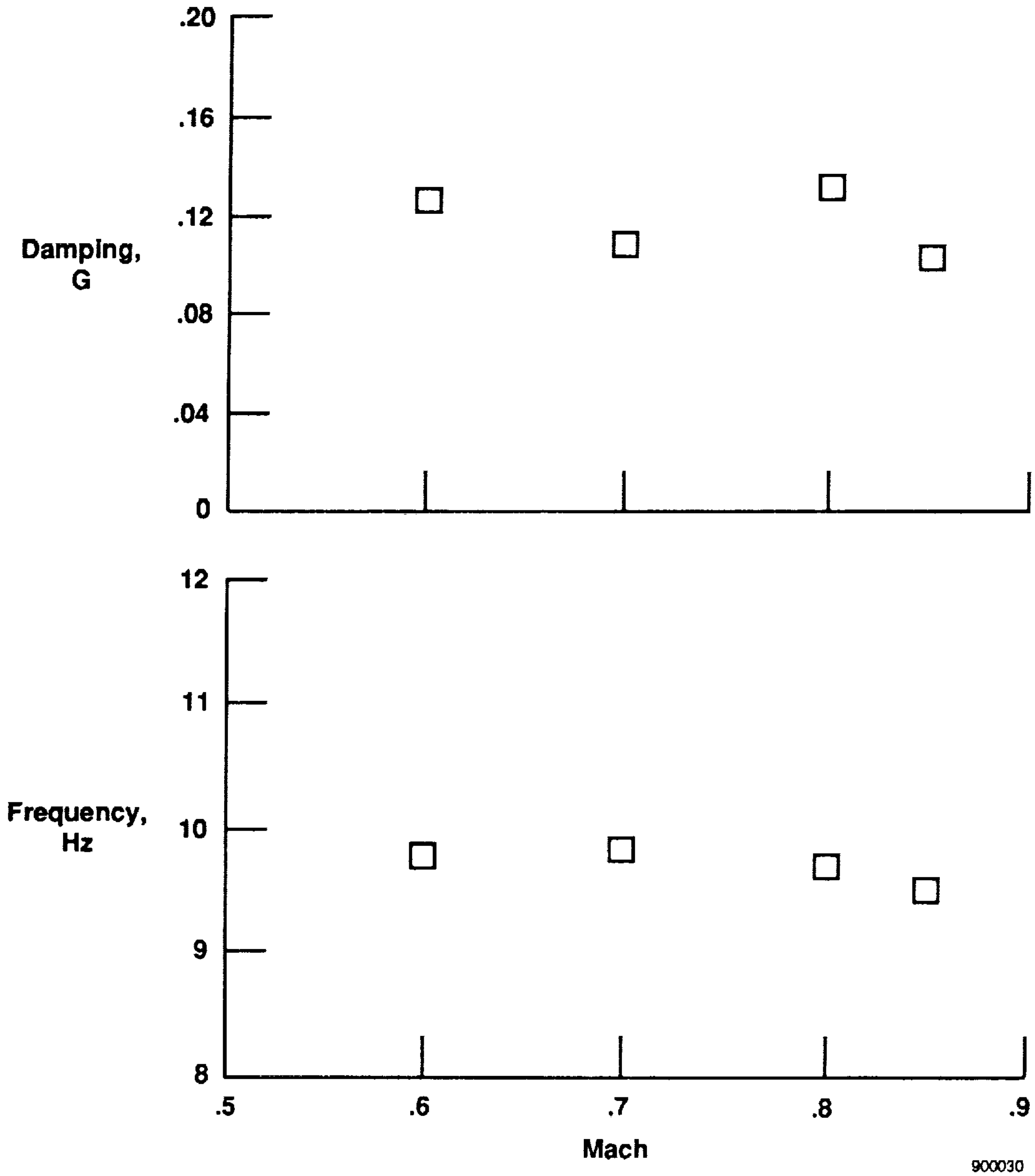

Figure A-20. Wing fore-and-aft modal data at 27,500 ft.

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15. Supplementary Notes

16. Abstract

An F-14A aircraft was modified for use as the test-bed aircraft for the variable-sweep transition flight experiment (VSTFE) program. The VSTFE program was a laminar flow research program designed to measure the effects of wing sweep on laminar flow. The airplane was modified by adding an upper surface foam and fiberglass glove to the right wing. An existing left wing glove had been added for the previous phase of the program. Ground vibration and flight flutter testing were accomplished to verify the absence of aeroelastic instabilities within a flight envelope of Mach 0.9 or 450 knots, calibrated airspeed, whichever was less. Flight test data indicated satisfactory damping levels and trends for the elastic structural modes of the airplane. Presented in this report are ground vibration test data, in-flight frequency and damping estimates, time histories and power spectral densities of in-flight sensors, and pressure distribution data.

17. Key Words (Suggested by Author(s)) Flutter; Ground vibration test; Laminar flow wing; Modal analysis	18. Distribution Statement Unclassified — Unlimited Subject category 05

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