NASA TM-87739

NASA-TM-87739 19860021867

NASA Technical Memorandum 87739

USAAVSCOM TECHNICAL MEMORANDUM 86-B-3

STRUCTUREBORNE NOISE IN AIRCRAFT -- MODAL TESTS

S. A. CLEVENSON
V. L. METCALF

JULY 1986

National Aeronautics and Space Administration

Langley Research Center
Hampton, Virginia 23665

STRUCTUREBORNE NOISE IN AIRCRAFT — MODAL TESTS

Sherman A. Clevenson and Vern L. Metcalf

SUMMARY

As part of an investigation to develop measurement techniques for structureborne noise, three modal surveys have been conducted on an OV-10A aircraft and the results have been presented. The purpose of the modal surveys was to identify suitable locations for mounting accelerometer and strain gages in subsequent tests in which transfer functions relating wing vibration to interior noise were to be determined. These surveys consisted of a: 1) wing/fuselage modal survey utilizing one shaker under the right wing; 2) complete wing modal survey utilizing two shakers, one under each wing; and 3) fuselage side panel modal survey utilizing a small instrumented hammer. The predominant frequencies and damping ratios for each analysis were listed in tables. The primary mode shapes at the lower frequencies and at frequencies near the expected engine driving frequencies have been shown for each survey.

INTRODUCTION

A potential problem of the new generation proposed advanced turboprop or propfan aircraft is the high level of interior noise compared to other types of conventional take-off and landing aircraft (References 1 and 2). The interior noise spectra of these aircraft are characterized by low frequency discrete tones at the propeller blade passage frequency and its harmonics. The interior noise levels are functions of the airborne noise and structureborne noise (via a structural path). Because of the high correlation of the airborne and structureborne noise, it is impossible to determine the contribution of each source from only measurements of interior and exterior levels.

The present study was conducted as part of a larger investigation to develop a measurement technique for structureborne noise. The approach, utilizing an OV-10A aircraft, figure 1, consist of four phases. The first phase, which is reported in this paper, consists of performing modal analyses of the aircraft to determine appropriate locations for vibration sensors and shakers. The second phase is to measure structureborne noise transfer functions for shaker inputs. The third phase consists of measuring similar transfer functions for engine operation (ground run-ups). The fourth phase consists of attenuating the airborne noise sufficiently for direct measurement of the structureborne noise. After completion of these phases, the data and results will be combined to fulfill the objective of developing a measurement technique for structureborne noise.

A structure such as an aircraft has many degrees of freedom and therefore has many resonance modes. When external forces such as engine vibration and propeller wake (Reference 3) are coupled to the structure some of these modes are be excited and thus provide an efficient path for this vibrational energy to reach the aircraft interior. Energy of this type may then radiate into the interior space and be recorded as interior noise. In the present study, a modal survey was conducted on the OV-10A aircraft to identify the major structural modes including their resonance frequencies, characteristic mode shapes and damping ratios. The purpose of the study is to aid in the determining the location of sensors and shakers to measure transfer functions. The locations, in turn, will be used to determine the structureborne/airborne noise transmission characteristics of the aircraft.

The results of these analyses are shown in this paper in three sections: (1) Wing/fuselage modal survey, using a 100 pound force shaker with the aircraft fully fueled including its full auxiliary fuel tank; (2) aircraft wing modal survey with minimal fuel on board (no auxiliary fuel tank), using two 50 pound force shakers; and (3) a modal survey of a 39" x 39" fuselage right side panel under the aircraft wing.

TEST METHODS AND ANALYSES

Three modal surveys were conducted using the NASA Lewis Research Center OV-10A aircraft. The aircraft is shown in figure 1. The test set-up including shaker location is shown in figure 2. The aircraft is a cantilever shoulder-wing monoplane, without dihedral or sweep, powered by two 715 shaft horsepower turboprop engines with three-bladed propellers. It can carry 258 US gallons of fuel in its inter-spar fuel tank and 230 US gallons of fuel in its under fuselage jettisonable tank. It has a wingspan of 40 feet, a length of 41 feet 7 inches, a height of 15 feet 2 inches, and an empty weight of 6969 pounds. This aircraft was selected because of its unique construction which ensures all of the structureborne noise reaching the interior is transmitted via the wing.

The experimental modal analysis technique used for these surveys started with the unit forced response input data and extracted the modes of vibration directly from the frequency response functions. It was therefore unnecessary to make any assumptions about the mass and stiffness distribution of the structure (Reference 4). Typical to the extraction of analytical information from test data, modal extraction involves a number of curve fitting operations. With the use of a multichannel FFT analyzer and the Modal Plus software for analysis (reference 5), resonance frequency and modal damping are estimated by a polyreference analysis technique which solves for the modal parameters in the time domain. The technique manipulates multiple response functions for up to three reference locations to obtain least square estimates of the modal parameters. The first model analysis used one reference; the second, two references; and the third used three references.

The first modal analysis consisted of obtaining frequency response function data for a geometry arrangement which included the fuselage and four center sections of the aircraft wing. Data for this configuration were collected with about 2100 lbs. of fuel on the aircraft including its full auxiliary fuel tank. A single 100 lb. force shaker, placed about 132 inches from the fuselage centerline on the lower surface of the right wing rear spar, was used to provide the input forcing function of white noise band-limited to less than 512 Hz and is shown in figure (2). A total of 109 specific accelerometer locations, 73 on the wing (figure 3) and 36 on the fuselage were used to establish the frequency response function data base.

The second modal analysis consisted of obtaining a set of frequency response functions for a geometry arrangement which included only the complete aircraft wing. Data for this configuration were collected with a minimum amount of fuel (about 300 lbs.) on the aircraft. Two 50 lb. force shakers operating simultaneously from a single random source of white noise band-limited to less than 512 Hz were used to provided the input forcing function. One shaker was placed in the identical location on the right wing as described in the previous paragraph. The second shaker was placed about 164 inches from the fuselage centerline on the lower surface of the left wing front spar.

The third modal analysis consisted of a survey to identify the mode shapes and resonance frequencies associated with a large panel area. A fuselage right side panel (aft center body frame) was selected for this investigation. The area, about 39 inches square, was directly adjacent to and under the right wing. Included along the upper boundary were both the aft and forward wing attachment points; the lower boundary was along the fuselage floor. A 0.3 pound instrumented force hammer was used to provide an impulse loading at 49 locations on the panel's framework (figure 4). Frequency response function data were collected for each of the 49 input locations and the three reference accelerometer locations.

RESULTS AND DISCUSSION

The results and discussion of the three modal analyses are in three sections: 1) Wing/fuselage, fueled; 2) wing alone, defueled; and 3) fuselage side panel. In addition, a discussion of suitable transducer locations for obtaining structureborne noise transfer coefficients is included.

Wing/Fuselage Modal Analysis

The results of the modal analysis are shown in Table I. Undeformed structure and typical mode shapes are shown in figures 5-16.

Table I lists 14 modes and their respective frequencies in the range from 0 to 256 Hz. In addition, the corresponding values of damping ratio are shown. Of primary interest were any modal frequencies that occur at or near the predominant engine rotational frequency, propeller blade passage frequency or their harmonics. The blade passage frequency of the aircraft is 99 Hz at rated thrust. Thus mode number 5 at 105.7 Hz is near the blade passage frequency and mode number 11 at 201.5 Hz is near the second harmonic of blade passage frequency.

The mode shapes of modes 5 and 11 are illustrated with the aid of computer-graphics. Figure 5 shows the undeformed wing sections and fuselage, and indicates the right and left sides of the aircraft. All views assume that the observer is to the rear and above the left side of the aircraft. Deflections of each section are first indicated, and then the sections are combined to show the aircraft motion for the particular mode. Deflections all utilize the same scale factors; thus figures may be directly compared.

Figure 6 indicates the deflection of the left side of the fuselage during one cycle of oscillation for mode 5. The lines indicate amplitudes of deflection from the zero condition (figure 5) to maximum deflection in both directions. The nodal points are clearly defined, and the envelope of the deflection is clearly indicated.

Figure 7 indicates the corresponding deflections of the right side of the fuselage for one cycle of oscillation. It may be noted that the deflections of the left and right sides (compare to figure 6) are very dissimilar. A comparison of the deflections of the upper rear portion of the fuselage sides indicate that they are of dissimilar amplitudes (figures 6 and 7). A similar comparison of the forward sections indicate greater deflection of the right side compared to the left side of the fuselage.

Figure 8 shows the left and right sides of the fuselage connected together. It is difficult to distinguish phase between deflections on these figures; however, the relative amplitudes of the sides is apparent.

Figure 9 shows a planform of the aircraft's wing. The numbering system (300-1620) indicated on the figure identify the wing geometry for computer analysis purposes. For the analysis of the wing and fuselage, only the four center sections, identifed in figure 9 by locations between points 620 and 1420 and between points 680 and 1360, were included. The engine and shaker locations are also indicated on the figure.

Figure 10 shows the deflections of the main wing sections (between points 620 and 1420) for one cycle of oscillation at 105 Hz. Note that the predominant deflection occurs on the right hand wing. It should be remembered that only one shaker was used on the right wing and that the aircraft was fueled with more than 2100 pounds of aviation fuel in both wing tanks and a below fuselage tank. It is believed that the fuel acts as a blocking mass and provides viscous damping, the results of which are indicated by the very small relative deflections of the left wing. Considerable torsional motion is apparent.

The deflections of the combined wing sections and fuselage are shown for one cycle of oscillation at 105 Hz in figure 11. For detailed deflections, one should refer back to the earlier figures for segmented oscillation.

Figure 12 indicates the deflection of the left side of the fuselage during one cycle of oscillation for mode 11 (201.5 Hz). At this higher frequency, there appears to be considerably less deflection than occured for mode 5 (see figure 6). The deflections of the right side of the fuselage are indicated in figure 13. Figure 14 shows the fuselage with the deflections of each side and the connecting links.

Figure 15 shows the deflections of the left and right wing sections, the same wing components as in figure 10, for one cycle of oscillation. The modes shapes of mode 5 and 11 are similar in that they both have nodal points near the engine locations on the right wing. However, mode 11 at 201.5 Hz appear to be primarily a bending mode whereas mode 5 appears to have considerable twist. Again, probably due to the fuel load, the deflections of the left wing are minimal. Figure 16 shows the deflections for the combined wing sections and fuselage, both for one cycle of oscillation. A comparison with mode 5, (figure 11) indicates that mode 5 has greater deflections of both the wing and fuselage and that the mode shapes are quite different.

Wing Alone Modal Analysis

The results of the modal analysis is shown in Table II. Typical mode shapes are shown in figures 17-24.

Table II lists 34 wing modes and their respective frequencies in the frequency range from 0 to 256 Hz. In addition, the corresponding values of damping ratio are also shown. Again, of primary interest are the modes that occur near the blade passage frequency or its harmonics, namely modes 16 or 17 and 23 at frequencies of 95.5 or 108.9 and 200.5 Hz. Also of interest are the first few lowest modal frequencies of 9.9, 18.1, and 31.6 Hz. It should be noted that 31.6 Hz is close to the engine rotational frequency of 33 Hz.

The first beam mode of the wing (driven by two shakers with minimal internal fuel) is shown in figure 17 for one-quarter cycle of oscillation at 9.9 Hz. It may be noted that the deflection of the wing tips are in phase with each other and out of phase with the center section where the fuselage is connected. The irregularities in the deflected curves are probably the result of low signal levels. It should be remembered that the wing is excited with white noise; thus exciting all frequencies.

Mode 2 at 10.1 hz was unrecognizable as to mode shape and nodes even though the damping ratio for mode 2 was much smaller than the damping ratio of the 1st beam mode.

The wing second beam mode at 18.1 Hz is shown for one-quarter cycle of oscillation in figure 18. Note that the wing tips are out of phase and that there is essentially a node line at the center of the wing.

The wing third beam mode at 31.6 Hz, close to the engine rotational frequency, is shown for one-quarter cycle of oscillation in figure 19. From figure 19, it is seen that although the wing tips are in phase (both going downward) there apparently is some twisting of the wing tips. The twisting is more pronounced in the full cycle of oscillation view (figure 20). At all modes occuring at higher frequency, there is some degree of wing torsion combining with wing bending.

Mode 6 at 95.5 Hz is shown in figure 21 and mode 7 at 108.9 Hz is shown in figure 22 for a full cycle of oscillation. Since both modes are close to the blade passage frequency, both are shown for contrast. Mode 6 has relatively small deflections, primarily of the left wing tip and less on the right wing tip. In contrast, mode 7 has considerable twisting of the left wing tip and both twisting and bending of the right wing tip. In neither case is there much deflection of the fuselage.

Figures 23 and 24 show mode 23 at 200.5 Hz for one-quarter cycle and full cycle oscillation respectively. Although this mode appears to be primarily bending with 5 nodal points along the trailing edge of the wing, the left wing tip appears to be rotating around the outer left nodal point. The outboard right trailing edge of the wing also appears to be rotating.

Fuselage Right Sidewall Panel Modal Analysis

The results of the modal analysis is shown in table III. Typical mode shapes are shown in figure 25-33.

Table III lists the 12 fuselage right side panel modes and their respective frequencies in the frequency range from 0-380 Hz. In addition, the corresponding values of damping ratio are also shown. Of primary interest are the modal frequencies that occur at or near the blade passage frequency or its harmonics, namely modes 2 or 3 and 8 or 9. It is interesting to note that the computer determined no modal frequency below 175 Hz for the fuselage side panel. The computer printouts of mode shapes for the first nine modes are shown.

The lowest frequency mode of the panel is shown in figure 25 for one cycle of oscillation. Only one vertical frame appears to be deflecting. A second vertical frame which has very small deflections for this mode, becomes the prominent contribution of the next higher frequency mode as shown in figure 26. The frequency difference between these two modes is small -- only 8.4 Hz. Mode 2 at 183.5 Hz and mode 3 at 221.7 Hz straddle a frequency of interest. The modes shapes also differ considerably. In mode 2 (figure 26), the fourth frame from the front of the panel appears to be the first bending mode. For mode 3 (figure 27), the same frame appears to be in second bending and the upper longeron is starting to deflect. In mode 4 (figure 28), the upper longeron is exhibiting a first bending mode while second bending modes are appearing on some of the vertical frames.

Various bending modes, both vertically and horizontally, appear in mode 5 and 6 of the fuselage side panel (figures 29-30). The most forward frame appears to be in the first bending mode for mode 7 of the panel (figure 31). Mode 8 of the panel at 298.3 Hz (figure 32) is close to a harmonic of the aircraft blade passage frequency. Most of the vertical frames appear to be resonating as does the upper longeron. Panel mode 9 at 312.4 Hz (figure 33) shows considerable deflection near the center of the panel as well as a first bending mode of the upper longeron. Panel modes 10-12 (not shown) also exhibited considerable deformations.

Transducer Locations

The purpose of the modal surveys was to assist in determing transducer location for obtaining structureborne noise transfer functions, or more accurately, in determining locations where transducers should not be located. It was planned to obtain the transfer functions between vibration and strain measurements along the front wing spar and acoustic measurements in the cabin. It was therefore desirable that the vibration and strain measuring transducers not be located on a predominant node point of the wing main spar. The modes of primary interest are those that occur at or near the blade passage frequency or its second harmonic. An examination of these mode shapes show node points occuring at each of the aircraft's engines and at the fuselage. No other predominant node points are observed. Figure 34 indicates five potential locations of transducers on the main spar as determined from physical constraints of the aircraft. Since location four is at node point, it is concluded that it would not be a good location for mounting transducers. However, positions 1, 2, 3, and 5 should be acceptable as locations for determining structureborne noise transfer functions.

CONCLUDING REMARKS

The results of three modal surveys have been presented. The purpose of the modal surveys were to identify suitable locations for mounting accelerometer and strain gages in subsequent tests in which transfer functions relating wing vibration to interior noise were to be determined. The modal surveys consisted of a: 1) wing/fuselage modal survey utilizing one shaker under the right wing; 2) complete wing modal survey utilizing two shakers, one under each wing; and 3) fuselage side panel modal survey utilizing a small instrumented hammer.

The predominant frequencies and damping ratios for each analysis were identified and listed in tables. The primary mode shapes at the lower frequencies and at frequencies near the expected engine driving frequencies have been shown for each survey. The results of these surveys have indicated suitable locations for the sensors and shakers for measuring the transfer functions to be used in determining structureborne noise transmission into this aircraft.

REFERENCES

1. Catherines, John J.; and Mayes, William H.: Interior Noise Levels of Two Propeller-Driven Light Aircraft. NASA TM X-72716, 1975.

2. Catherines, John J.; and Jha, Sunil K.: Sources and Characteristics of Interior Noise in General Aviation Aircraft. NASA TM X-72839, 1976.

3. Metcalf, Vern L.; and Mayes, William H.: Structureborne Contribution to Interior Noise of Propeller Aircraft. SAE Technical Paper Series 830735, 1983.

4. Vold, H.; Kundrat, J.; Rocklin, G. T.; and Russell, R.: A Multi-Input Modal Estimation Algorithm for Mini-Computers. SAE Paper Number 820194, 1982.

5. Structural Dynamics Research Corporation, Modal-Plus User's Manual, General Electric CAE International Inc. 1983.

TABLE I

Wing/Fuselage Modes - Shaker on Right Wing

MODE	FREQUENCY, HZ	DAMPING RATIO
1	34.9	0.091
2	47.5	0.103
3	70.1	0.029
4	88.8	0.081
5	105.7	0.088
6	119.9	0.032
7	135.7	0.072
8	148.5	0.091
9	164.1	0.041
10	173.6	0.037
11	201.5	0.095
12	212.8	0.024
13	223.0	0.031
14	239.0	0.042

TABLE II

Wing Alone Modes — 2 Shakers

MODE	FREQUENCY, HZ	DAMPING RATIO
1	9.9	0.093
2	10.1	0.028
3	18.1	0.015
4	31.6	0.049
5	39.3	0.088
6	39.9	0.028
7	42.4	0.043
8	46.0	0.144
9	47.6	0.098
10	55.9	0.087
11	61.6	0.056
12	63.4	0.285
13	65.8	0.082
14	72.9	0.093
15	83.1	0.071
16	95.5	0.041
17	108.9	0.026
18	110.6	0.058
19	121.6	0.075
20	129.7	0.055
21	140.3	0.075
22	141.6	0.050
23	157.1	0.059
24	161.1	0.042
25	170.2	0.055
26	179.9	0.070
27	184.9	0.041
28	195.2	0.068
29	200.5	0.077
30	205.6	0.024
31	215.8	0.019
32	222.4	0.029
33	237.3	0.028
34	239.8	0.022

TABLE III

Fuselage Right Side Panel Modes - Using an Instrumented Hammer

MODE	FREQUENCY, HZ	DAMPING RATIO
1	175.1	0.045
2	183.5	0.014
3	221.7	0.026
4	234.5	0.060
5	261.0	0.020
6	260.6	0.155
7	267.7	0.056
8	298.3	0.066
9	312.4	0.042
10	322.6	0.085
11	332.1	0.019
12	355.2	0.026

NASA
L-85-6951

Figure 1.- Photograph of aircraft.

NASA
L-85-6663

Figure 2.- Front view of aircraft showing shaker location.

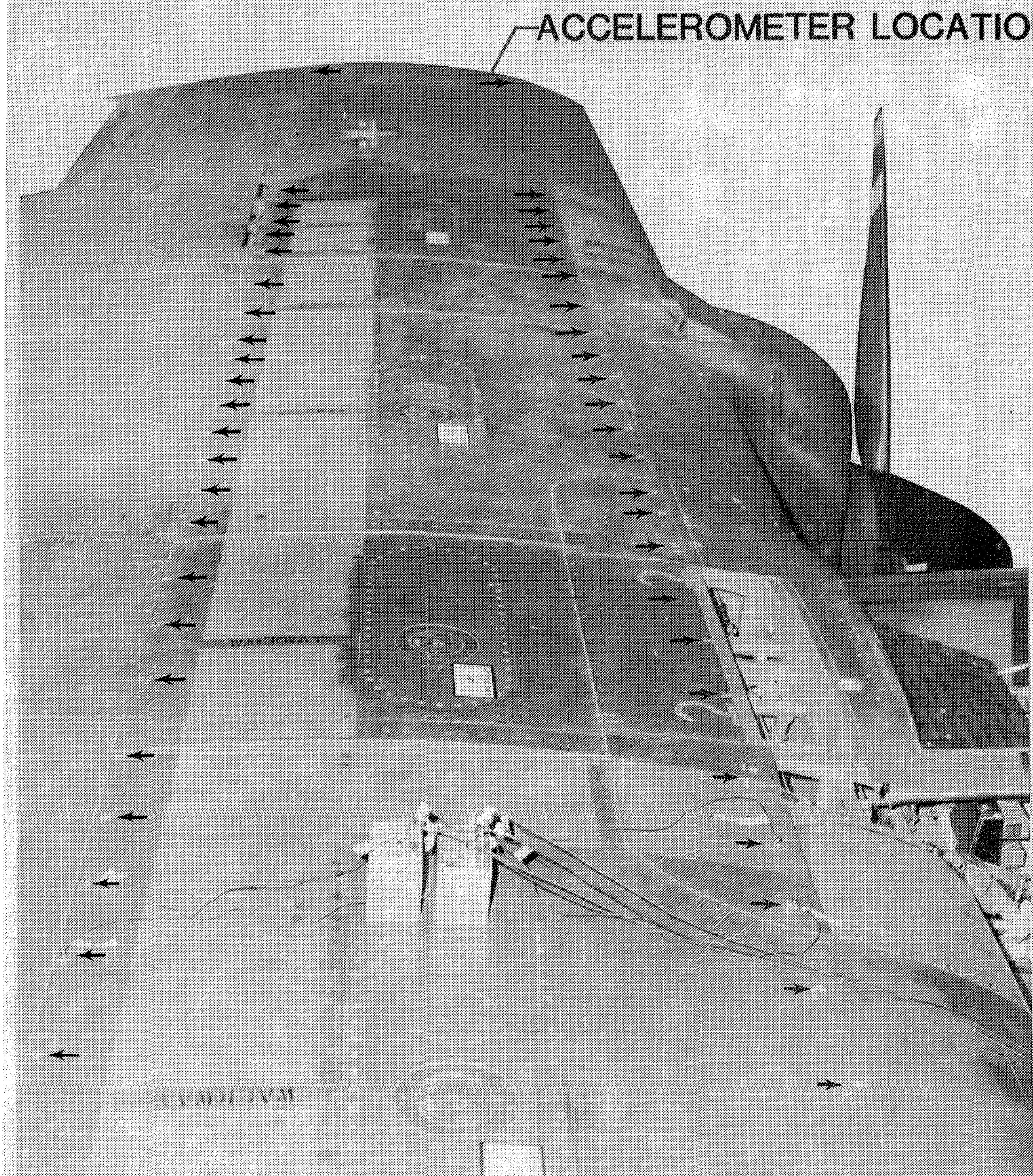

Figure 3.- Top view of aircraft wing showing most of 73 accelerometer locations.

NASA
L-85-6666

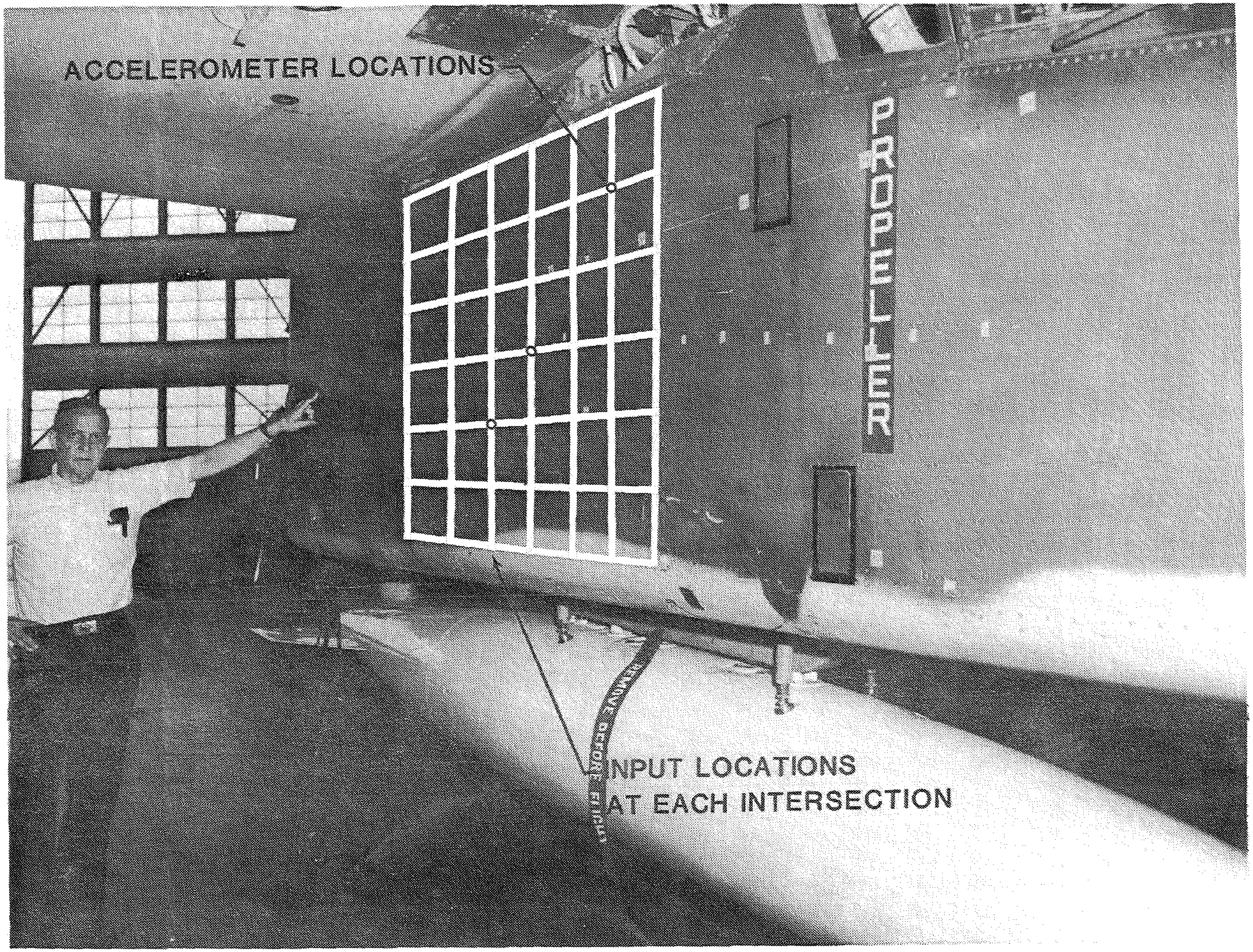

Figure 4.- Side view of aircraft right side panel showing 49 input locations and three refererence accelerometer locations.

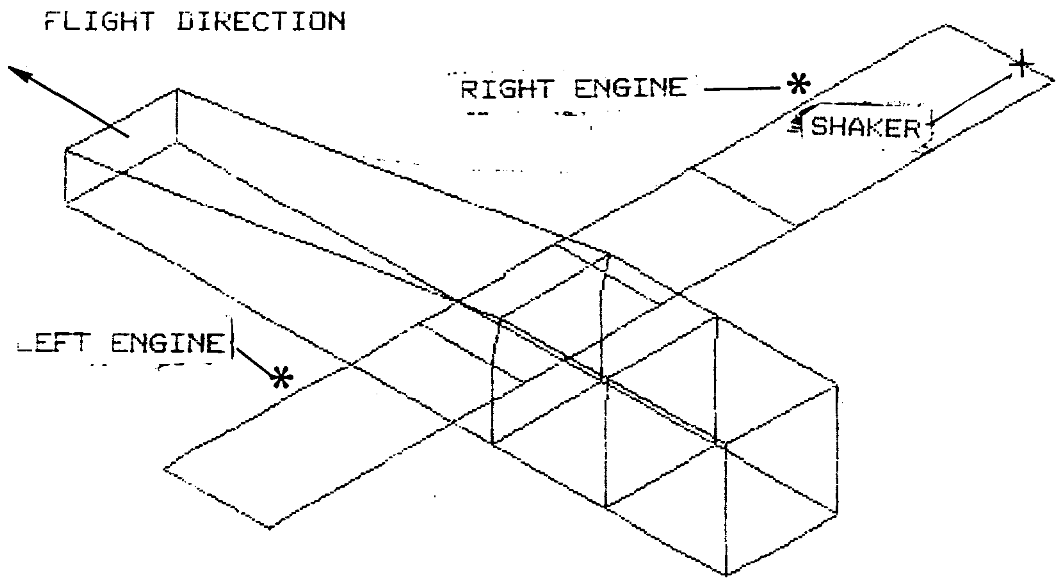

Figure 5.- Undeformed wing sections and fuselage.

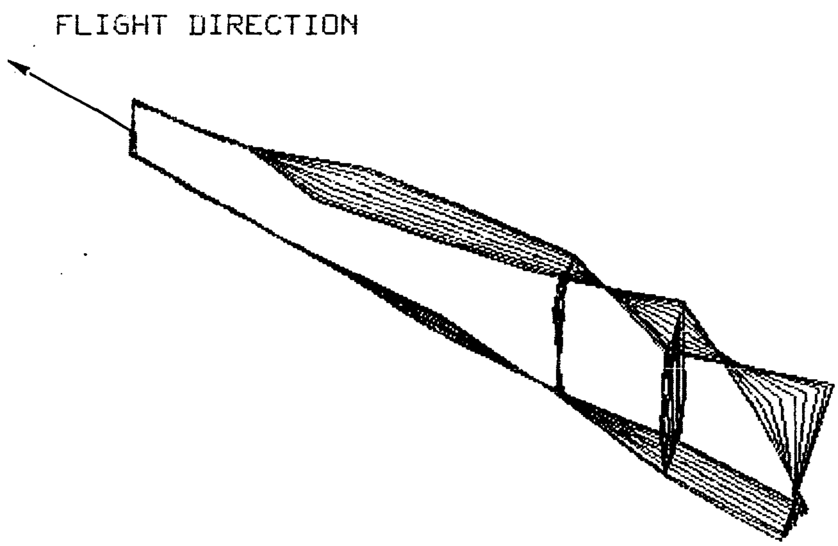

Figure 6.- Left side of fuselage: full cycle of oscillation: f = 105.7 Hz.

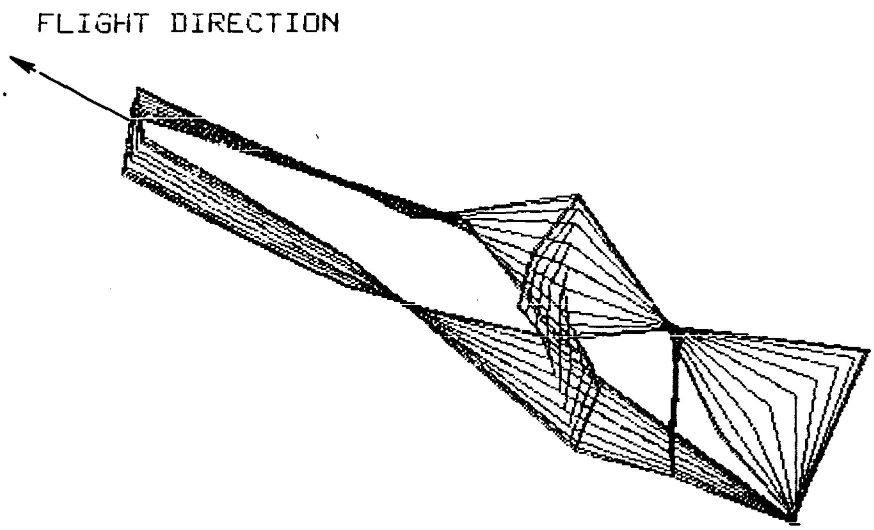

Figure 7.- Right side of fuselage: full cycle of oscillation: f = 105.7 Hz.

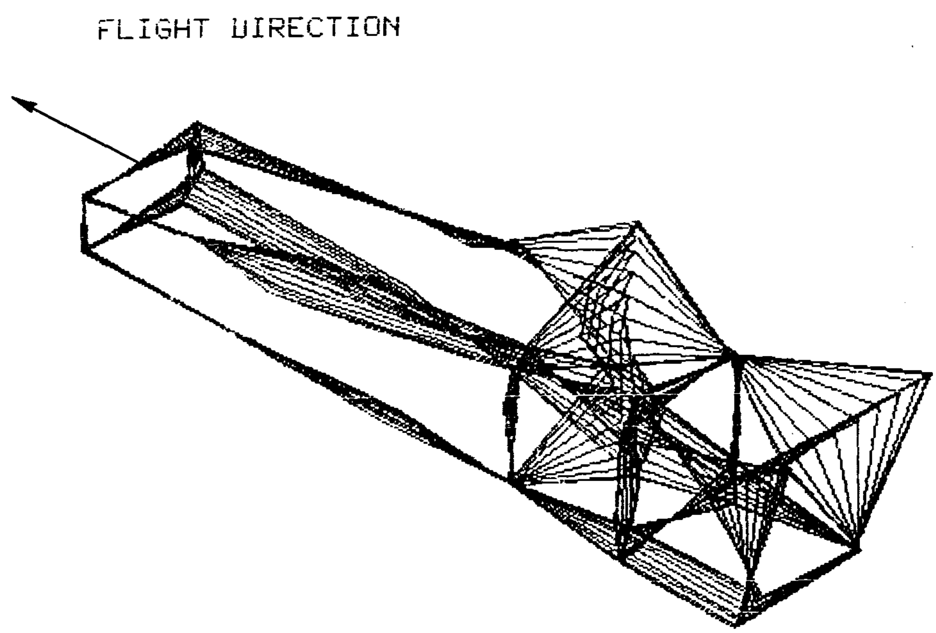

Figure 8.- Fuselage: full cycle of oscillation: f = 105.7 Hz.

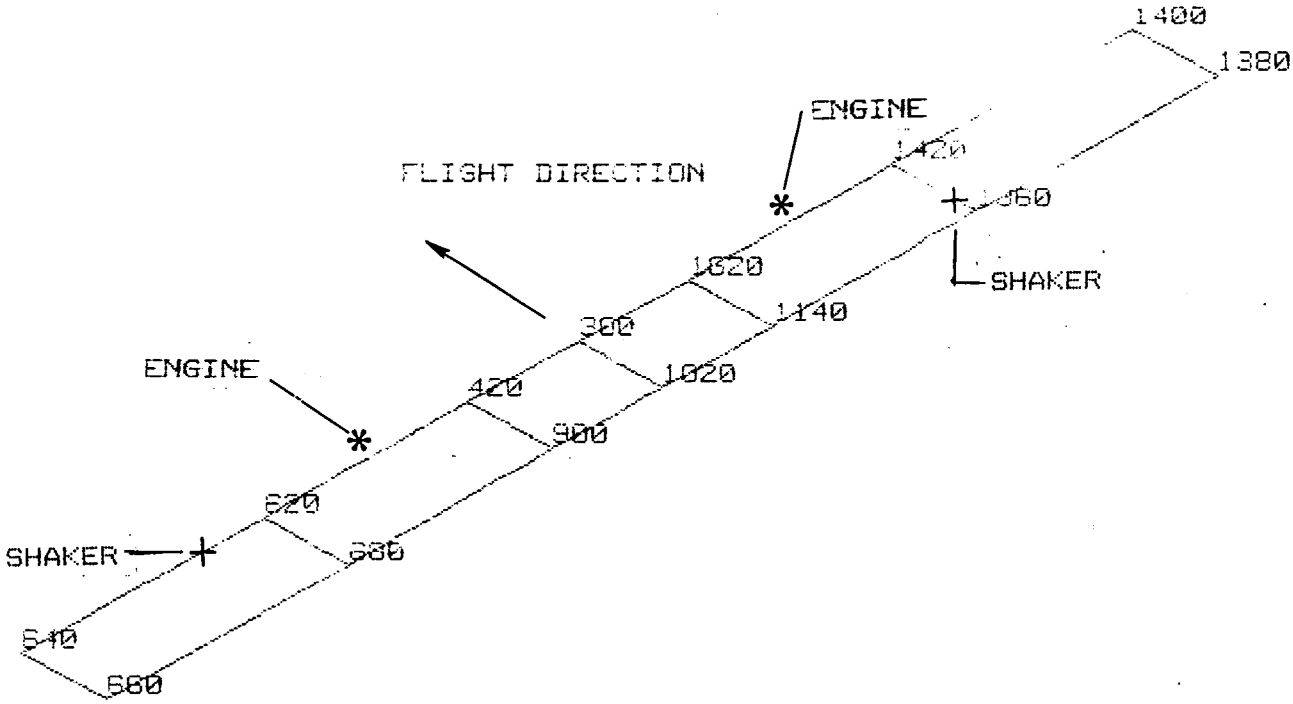

Figure 9.- Plan form of full wing indicating engine and shaker locations.

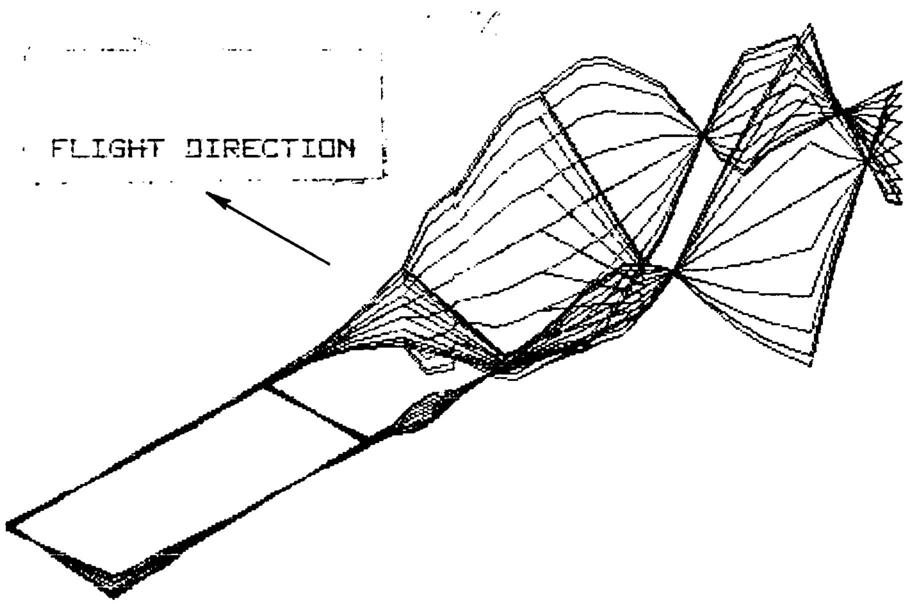

Figure 10.- Left and right wing sections: full cycle of oscillation:
f = 105.7 Hz.

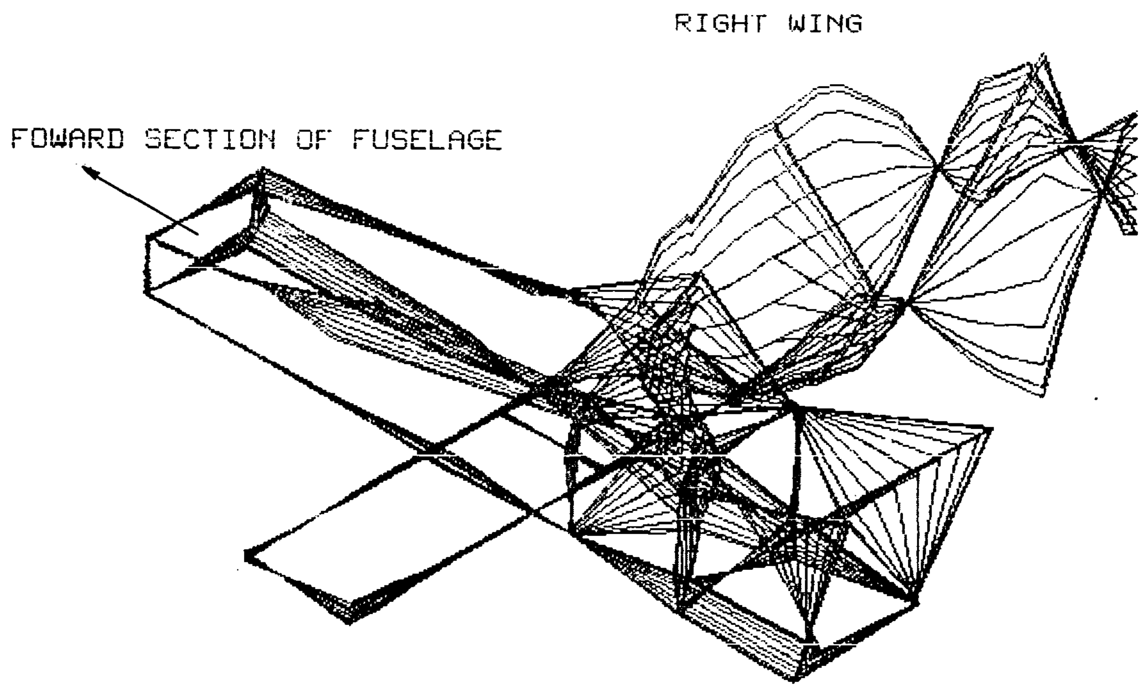

Figure 11.- Fuselage and wing sections: full cycle of oscillation: f = 105.7 Hz.

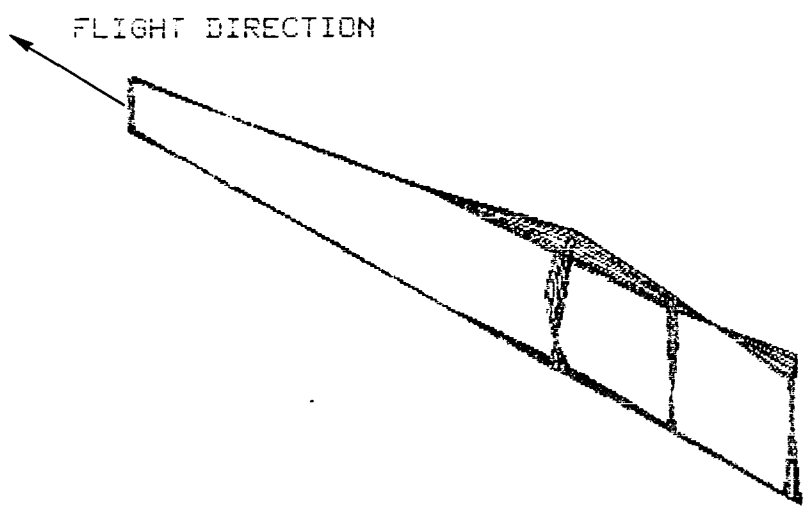

Figure 12.- Left side of fuselage: full cycle of oscillation: f = 201.5 Hz.

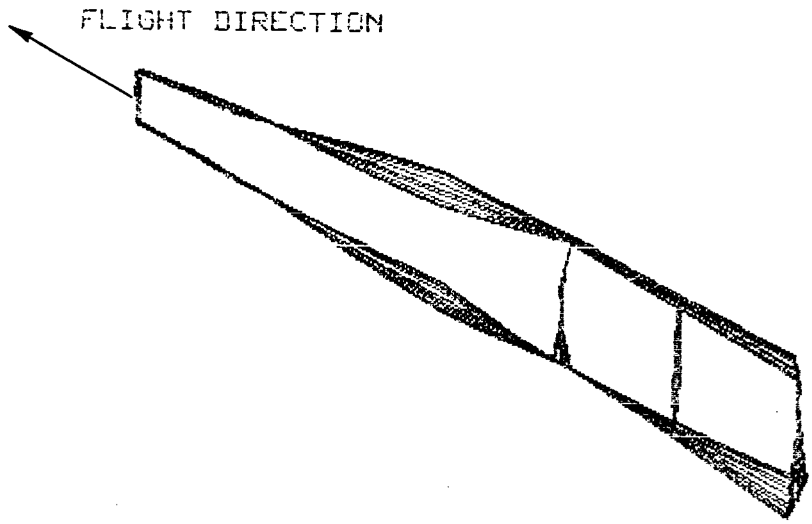

Figure 13.- Right side of fuselage: full cycle of oscillation: f = 201.5 Hz.

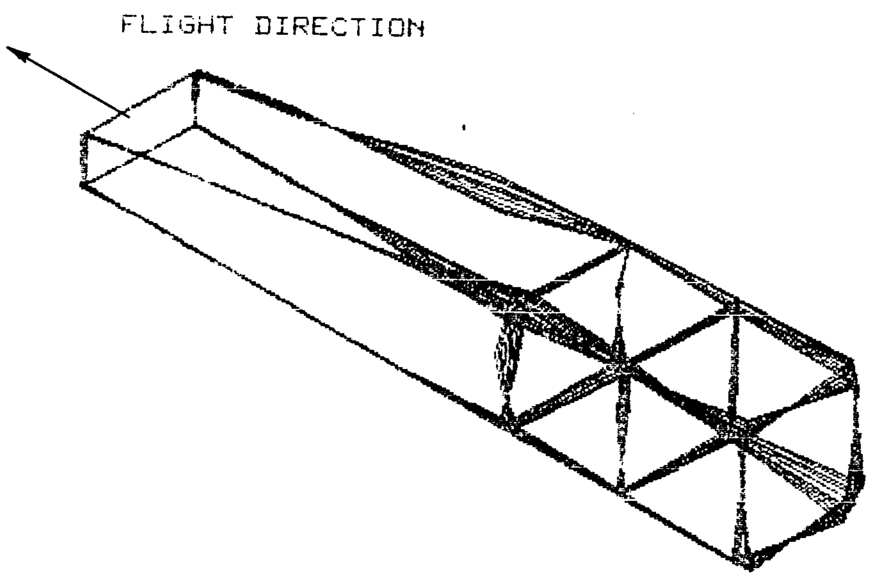

Figure 14.- Fuselage: full cycle of oscillation: f = 201.5 Hz.

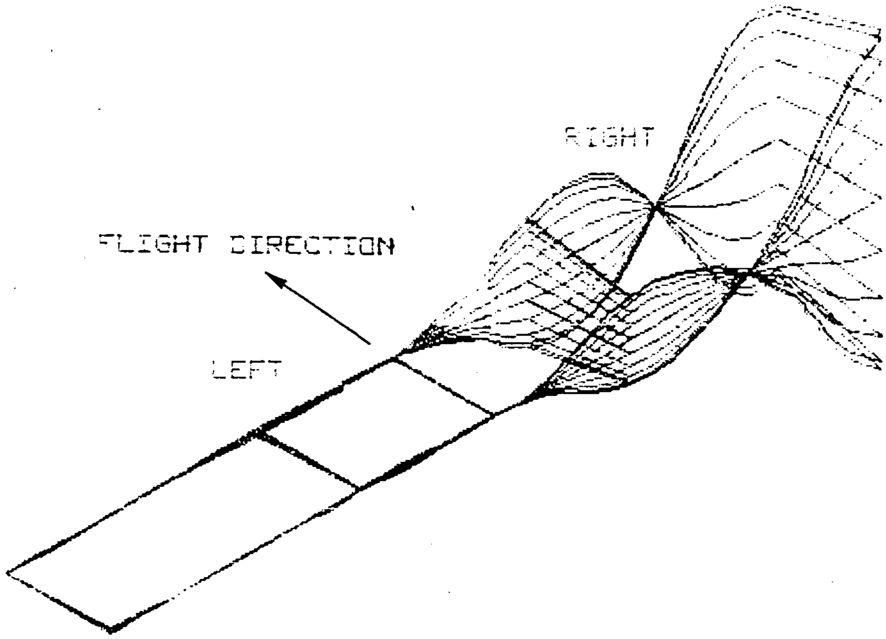

Figure 15.- Left and right wing sections: full cycle of oscillation: f = 201.5 Hz.

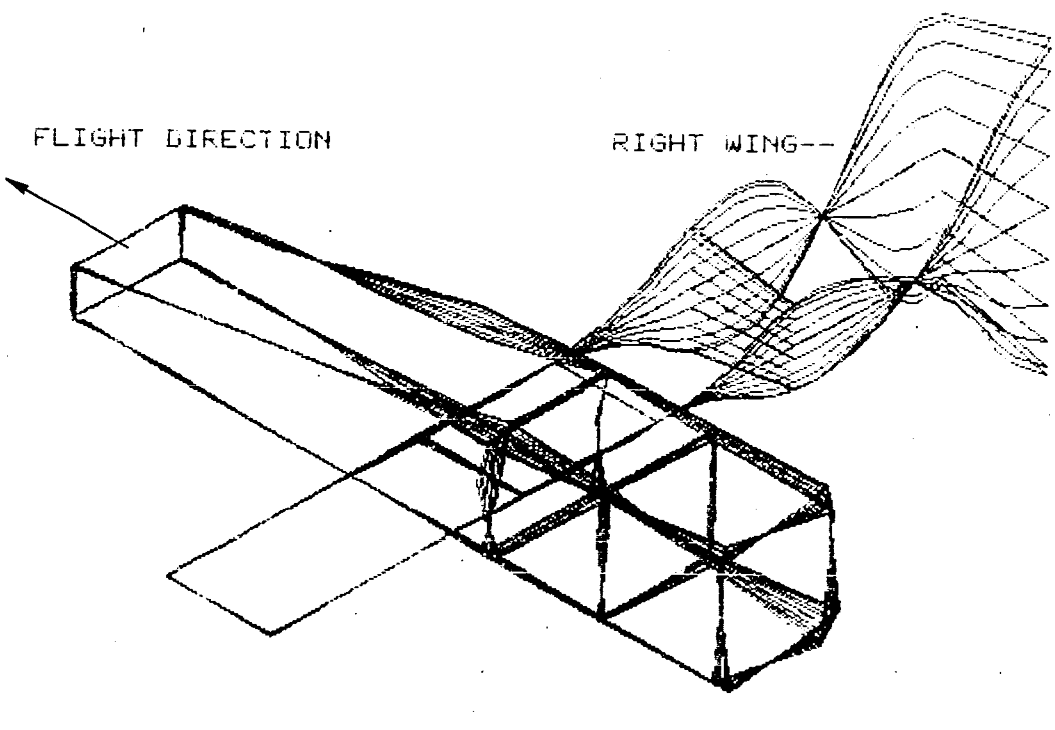

Figure 16.- Fuselage and wing sections: full cycle of oscillation: f = 201.5 Hz.

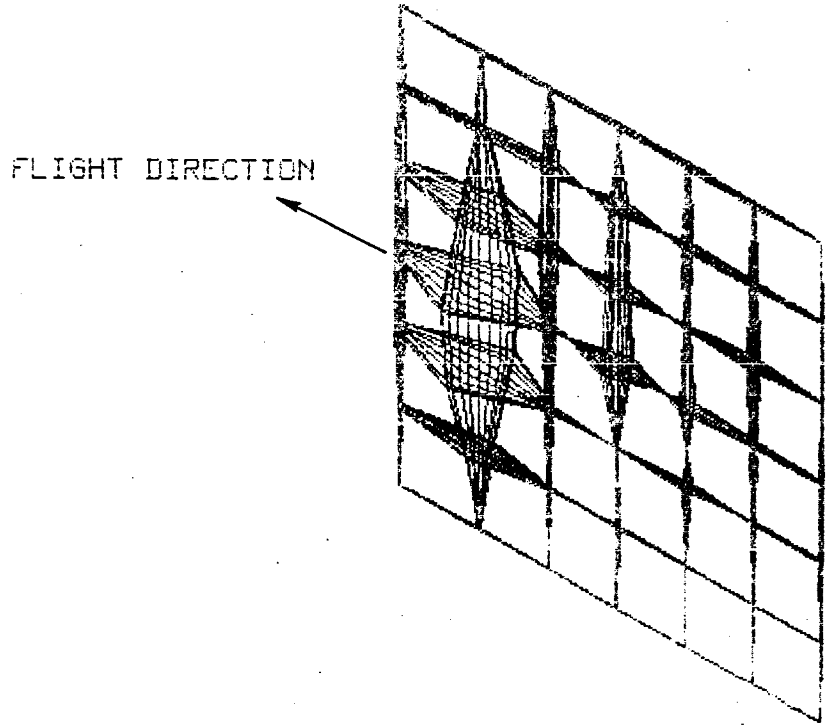

Figure 25.- Mode 1, fuselage right side panel: full cycle of oscillation:
f = 175.1 Hz.

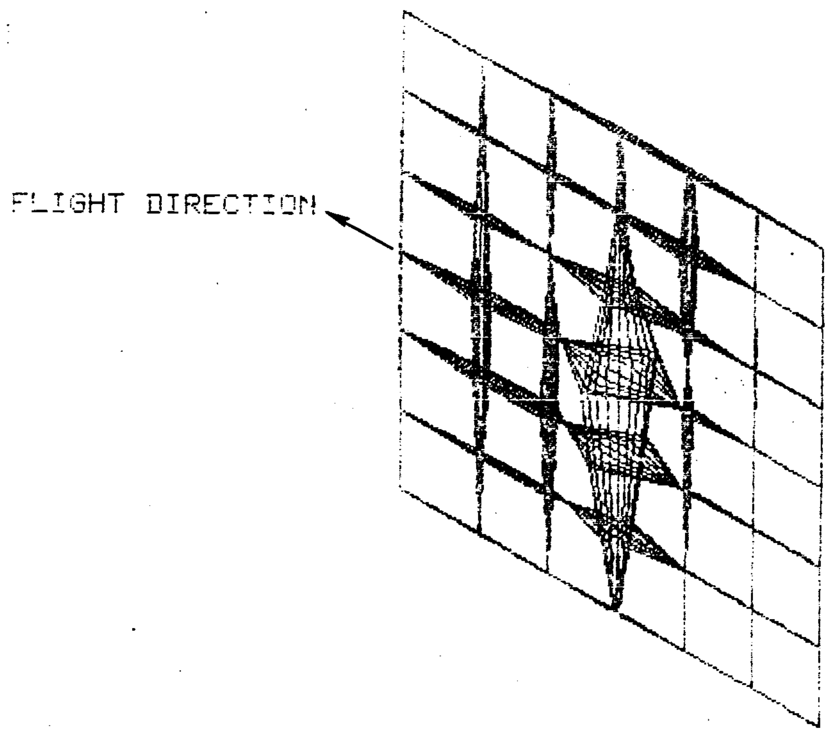

Figure 26.- Mode 2, fuselage right side panel: full cycle of oscillation:
f = 183.5 Hz.

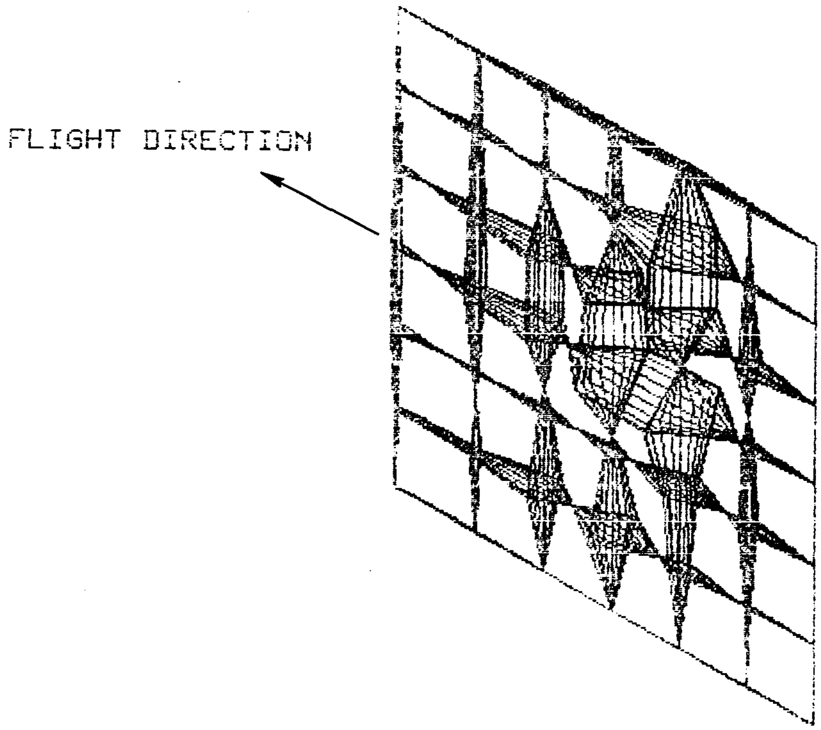

Figure 27.- Mode 3, fuselage right side panel: full cycle of oscillation:
f = 221.7 Hz.

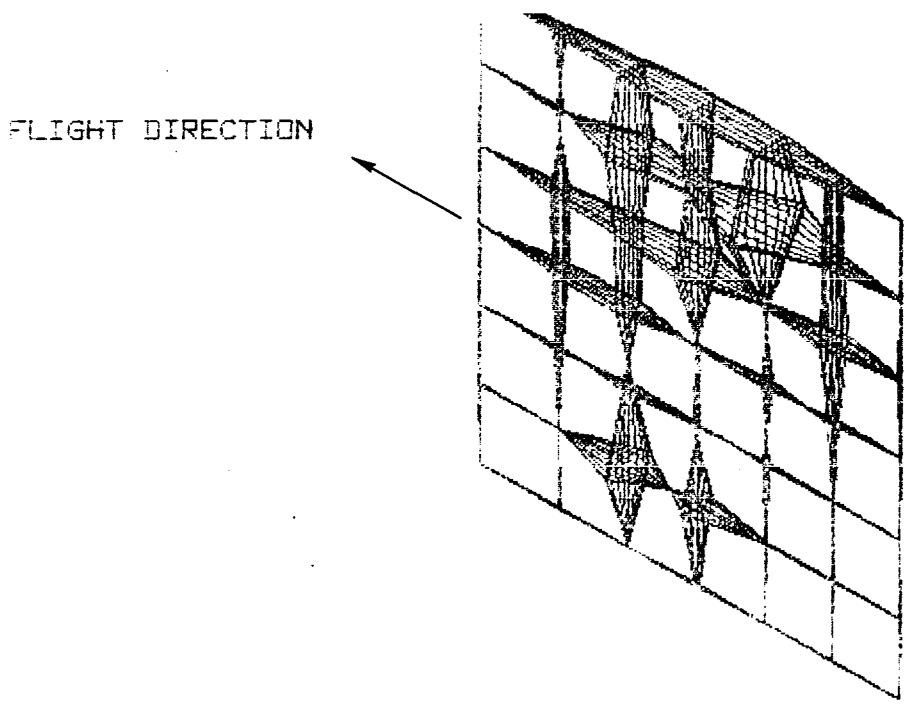

Figure 28.- Mode 4, fuselage right side panel: full cycle of oscillation:
f = 234.5 Hz.

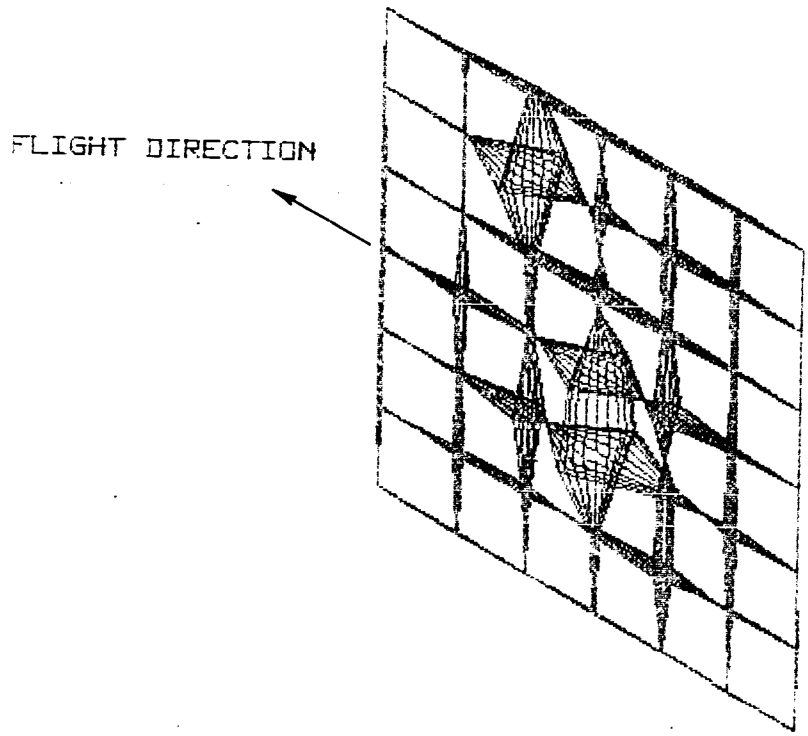

Figure 29.- Mode 5, fuselage right side panel: full cycle of oscillation: f = 261.0 Hz.

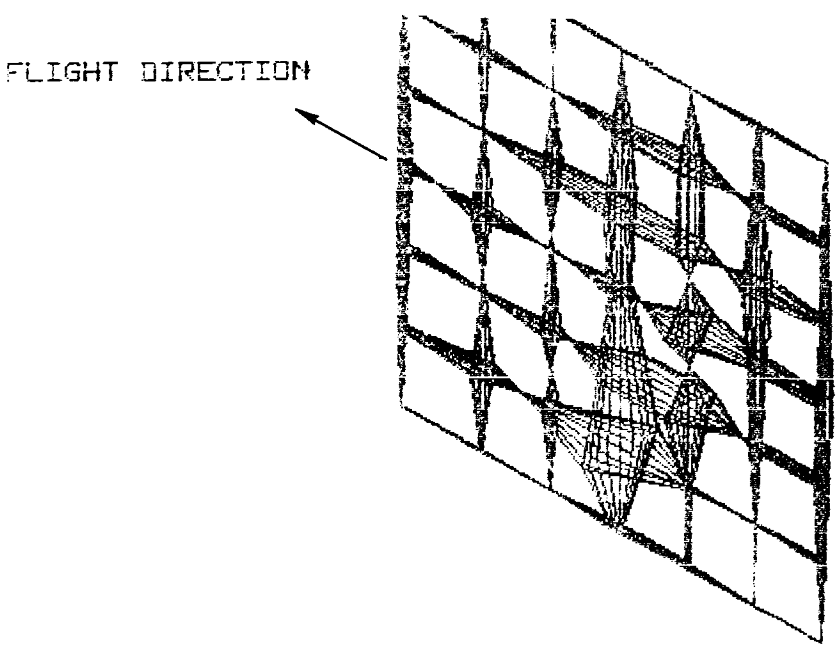

Figure 30.- Mode 6, fuselage right side panel: full cycle of oscillation: f = 260.6 Hz.

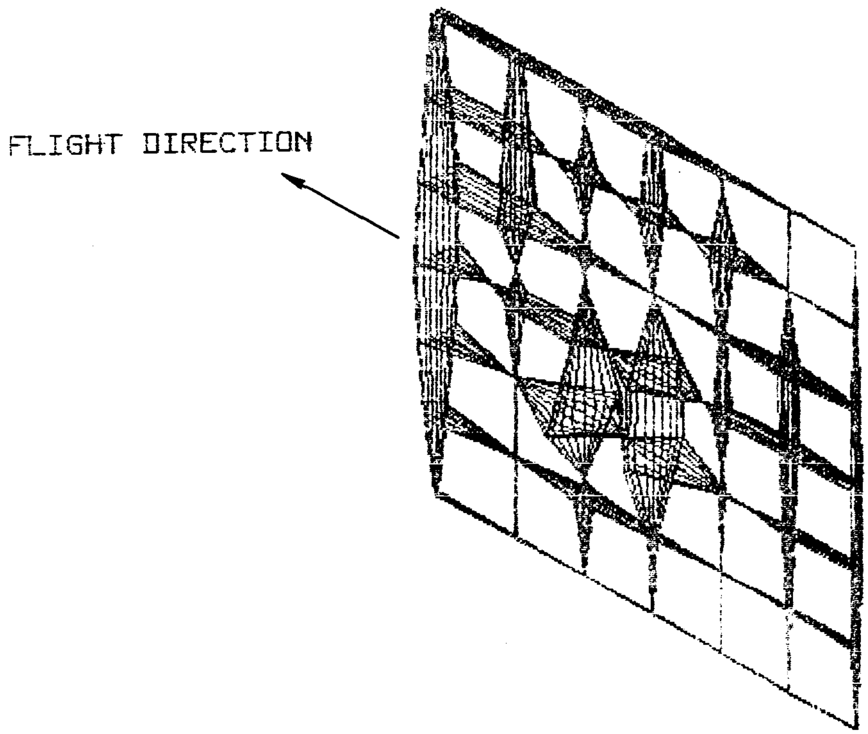

Figure 31.- Mode 7, fuselage right side panel: full cycle of oscillation:
f = 267.7 Hz.

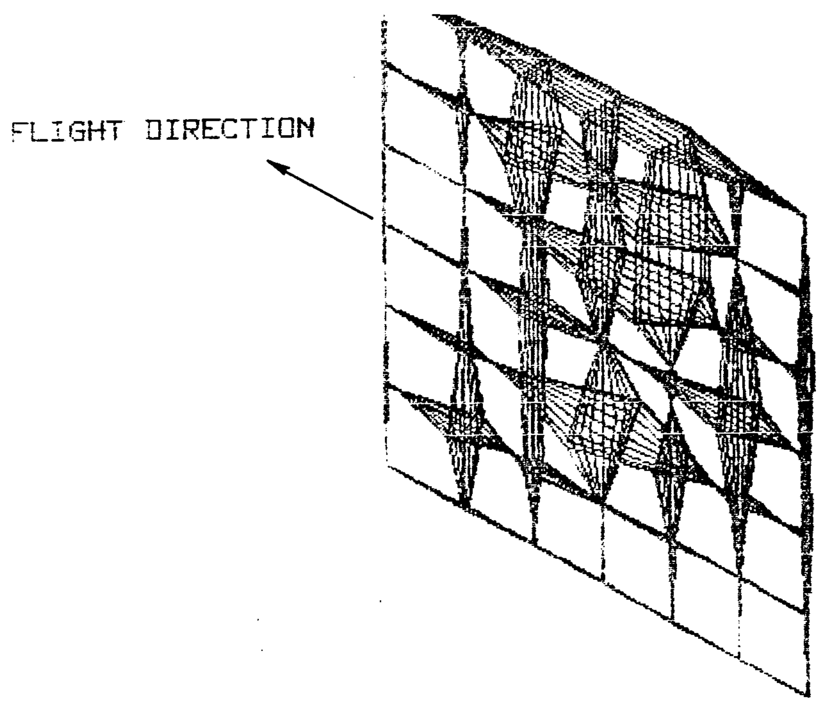

Figure 32.- Mode 8, fuselage right side panel: full cycle of oscillation:
f = 298.3 Hz.

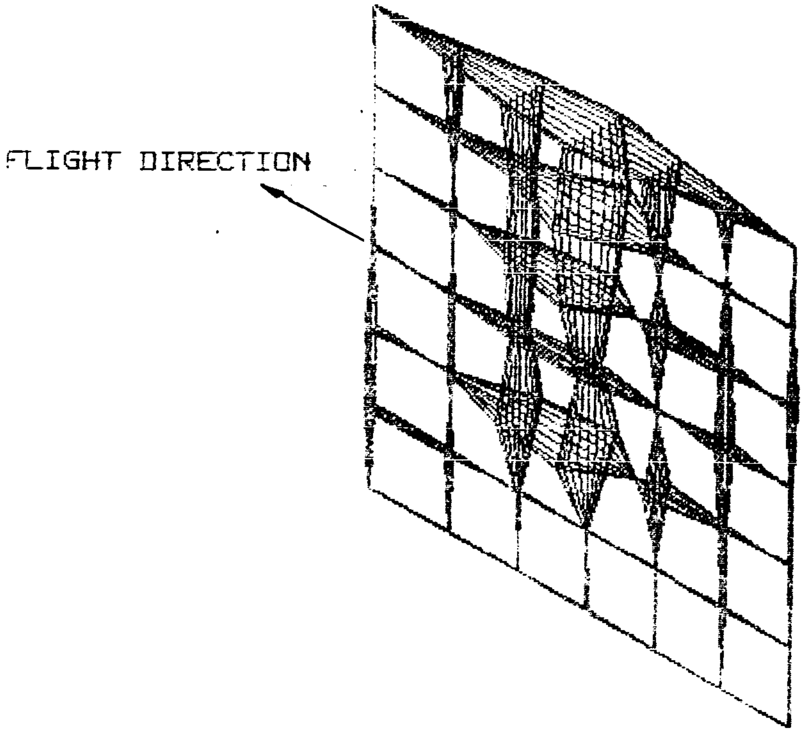

Figure 33.- Mode 9, fuselage right side panel: full cycle of oscillation: f = 312.4 Hz.

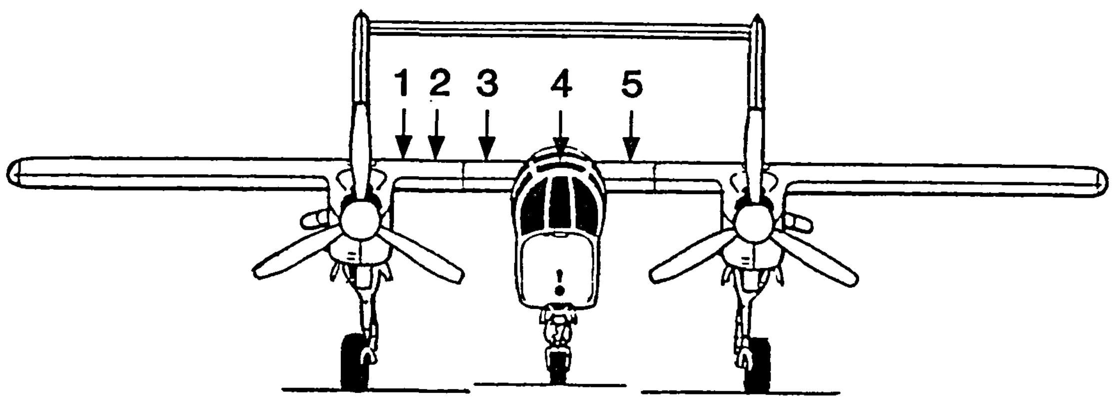

Figure 34.- Potential transducer locations for obtaining structureborne noise transfer functions.

1. Report No. NASA TM-87739 USAAVSCOM TM 86-B-3	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle STRUCTUREBORNE NOISE IN AIRCRAFT--MODAL TESTS		5. Report Date July 1986
		6. Performing Organization Code 535-03-11-03
7. Author(s) Sherman A. Clevenson and Vern L. Metcalf		8. Performing Organization Report No.
		10. Work Unit No.
9. Performing Organization Name and Address NASA Langley Research Center Hampton, VA 23665-5225 and Aerostructures Directorate USAARTA-AVSCOM Langley Research Center Hampton, VA 23665-5225		11. Contract or Grant No.
		13. Type of Report and Period Covered Technical Memorandum
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546-0001 and U.S. Army Aviation Systems Command St. Louis, MO 63120-1798		14. Army Project No. 1L161102AH45

15. Supplementary Notes

Sherman A. Clevenson, NASA Langley Research Center, Hampton, Virginia.
Vern L. Metcalf, Aerostructures Directorate, USAARTA-AVSCOM.

16. Abstract

As part of an investigation to develop measurement techniques for structureborne noise, three modal surveys have been conducted on an OV-10A aircraft and the results have been presented. The purpose of the modal surveys was to identify suitable locations for mounting accelerometer and strain gages in subsequent tests in which transfer functions relating wing vibration to interior noise were to be determined. These surveys consisted of a: 1) wing/fuselage modal survey utilizing one shaker under the right wing; 2) complete wing modal survey utilizing two shakers, one under each wing; and 3) fuselage side panel modal survey utilizing a small instrumented hammer. The predominant frequencies and damping ratios for each analysis were listed in tables. The primary mode shapes at the lower frequencies and at frequencies near the expected engine driving frequencies have been shown for each survey.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Structureborne Noise Aircraft Structureborne Noise Aircraft Interior Noise Vibration Aircraft Modal Analysis	Unclassified - Unlimited Subject Category 71

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	35	A03

*For sale by the National Technical Information Service, Springfield, Virginia 22161

End of Document