

## A FLIGHT INVESTIGATION OF OSCILLATING AIR FORCES: EQUIPMENT AND TECHNIQUE

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### Abstract

A description is given of the equipment and techniques to be used in a project aimed at measuring oscillating air forces and dynamic aeroelastic response of a swept wing airplane at high subsonic speeds. Electro-hydraulic inertia type shakers installed in the wing tips will excite various elastic airplane modes while the related oscillating chordwise pressures at two spanwise wing stations and the wing mode shapes are recorded on magnetic tape.

The data reduction technique, following the principle of a "wattmeter" harmonic analyzer employed by Bratt, Wight, and Tilly, utilizes magnetic tape and high speed electronic multipliers to record directly the real and imaginary components of oscillatory data signals relative to a simple harmonic reference signal. Through an extension of this technique an automatic flight-flutter-test data analyzer is suggested in which vector plots of mechanical admittance or impedance would be plotted during the flight test.

### INTRODUCTION

Most theoretical methods for computing oscillating air forces are based on linear potential flow theory, and as such may be expected to deteriorate in accuracy as shock wave and flow separation effects come into play at high subsonic and transonic Mach numbers. The experimental data available for evaluating the accuracy of theory in this Mach number range is extremely limited, and the accuracy of the data is frequently uncertain because of wind tunnel interference effects. In view of the need for accurate predictions of flutter, it is important that we extend

our knowledge of oscillating air forces in this speed range.

To help meet this need, the Flight Research Division at NACA-Langley has undertaken a project aimed at measuring oscillating air forces in flight. It is hoped that these measurements, obtained under full scale flight conditions and free from wind tunnel interference effects, may serve as a check on the accuracy of unsteady aerodynamic theory. In essence, the test method will consist of exciting various elastic modes of the airplane in flight by means of sinusoidal shakers installed in each wing tip. Oscillating air forces will then be investigated two ways: First the aeroelastic response of the airplane to known force inputs will be studied to obtain information on the integrated effects of oscillating air forces; and, second, the oscillating chordwise pressure distribution at two spanwise stations will be measured to gain a detailed insight into the nature of oscillatory flows at high subsonic speeds. Experimental measurements of both the forced response and pressure distributions will then be compared with theoretical predictions.

While obtaining experimental data on oscillating air forces is the primary goal of the project, a secondary, and perhaps equally important, aim is to gain experience which would be applicable to flight flutter testing techniques. This experience would include the development of excitation equipment and instrumentation, data reduction techniques and flight test methods involving the measurement of forced response.

This paper discusses some of the equipment and testing techniques planned for the project and points out, where possible, their application to flight flutter testing.

## FORCED RESPONSE

### Theoretical Forced Response Method

We will first consider the forced response phase of the project — but before discussing the experimental techniques, it is of interest to take a brief look at the theoretical analysis with which the experiment will be compared. The mathematical representation of the airplane wing panel is shown in the first figure.

An influence coefficient type dynamic analysis is used wherein the inertia, the aerodynamic, and the excitation forces acting on the wing are assumed to be concentrated at the eight discrete points shown. Associated with these points are a set of measured flexural influence coefficients and lumped masses representing the wing structure. The aerodynamics of the problem are obtained from the kernel function method of Watkins, Runyan and Woolston (ref. 1). As used here, the method, which is a three dimensional lifting surface theory, provides the aerodynamic load distribution in terms of the wing displacements at the influence points. The air loads concentrated at each of the influence points, are then obtained by integrating the load distribution over the appropriate areas that are shown by the dashed lines in the figure. The response problem approached in this manner has several advantages. It can be conveniently programmed on large scale digital computers. The mode shapes are defined directly by the vector displace-

ments of the influence points. And the accuracy of the mathematical representation of the airplane structure can be readily assessed by comparing ground measurements of forced response and mode shapes with calculated results in which the air forces have been omitted.

Matrix equations for aeroelastic forced response have been formulated by C. E. Watkins and J. L. Sewall of the Dynamic Loads Division, NACA-Langley, for use with an existing program of the kernel function method on the IBM 704 computer. Preliminary results obtained by the method for the forced response of a wind tunnel model show good agreement with experimental data. In the present tests, measurements will be made of the response at the 8 influence points shown in the figure together with the shaker input force. The next figure (Figure 2) shows these and other vibrations pick-up locations on the test airplane.

### Airplane and Instrument

The test airplane is an F-86D. The sweepback angle of the 1/4 chord of the wings is 35°, the aspect ratio is 5, and the thickness ratio is about 10 percent. The normal slotted leading edge for this airplane has been replaced by a fixed leading edge in order to eliminate certain flow irregularities and structural vibrations presented with the slotted configuration.

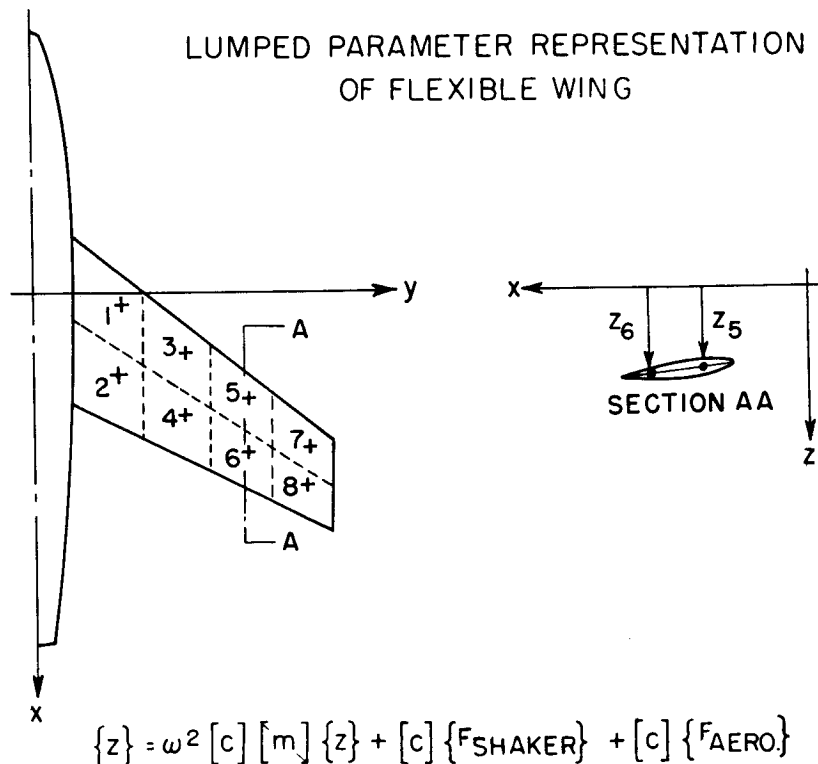


Figure 1. Lumped Parameter Representation of Flexible Wing

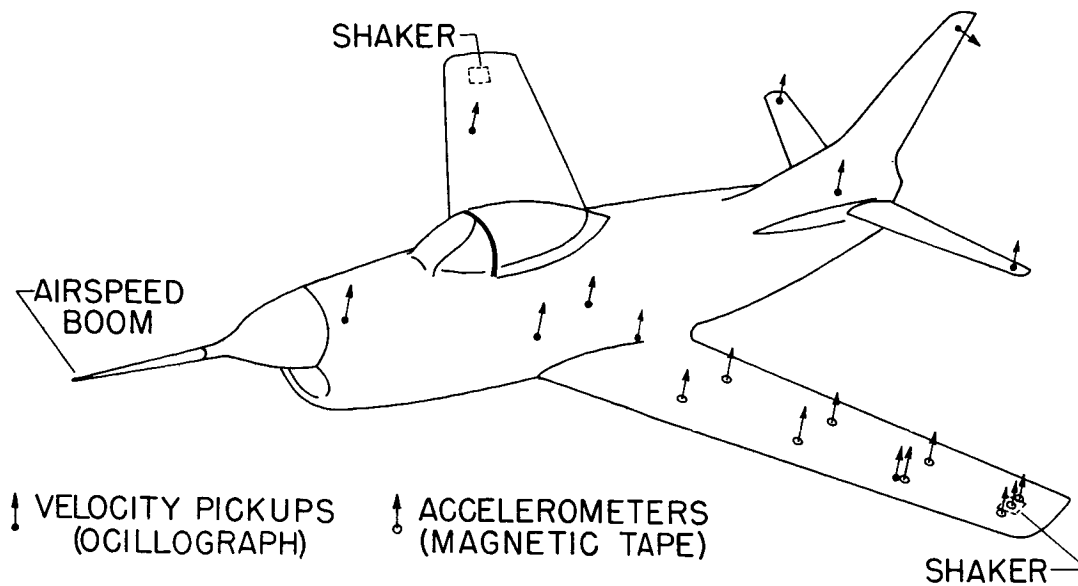


Figure 2. Location of Vibration Pickups on Test Airplane

The vibration pick-up locations are indicated by the arrows in the figure and the direction of the arrows depicts the sensing axis of the transducer. The primary measurements, indicated on the figure by the circles, are from accelerometers located at the 8 influence points on the wing and also in one of the shaker masses. These accelerometers are NACA variable inductance telemetering transducers equipped with temperature regulated ovens to minimize the

effect of outside temperature on the damping of the units. The accelerometer outputs are telemetered to a ground recording station and recorded on magnetic tape. Vibration data from other locations on the airplane, shown in the figure by arrows without circles, are obtained from MB type 124 self-generating velocity pick-ups and are recorded in the airplane on a recording oscillograph. A complete listing of the flight instrumentation is given in Table I.

TABLE I — AIRPLANE INSTRUMENTATION LIST

(a) Response Data

(Telemetered and recorded on magnetic tape)

<u>No. of Channels</u>	<u>Measurement</u>	<u>Description or Location</u>
8	Acceleration	8 influence points on left wing
1	Acceleration	Shaker mass on left shaker
2	$E_0 \cos \omega t, E_0 \sin \omega t$	Shaker input signal and input signal with 90° phase shift
1	Timer	
1	Voice	

(b) Oscillating Pressure Data

(Recorded in airplane on magnetic tape)

<u>No. of Channels</u>	<u>Measurement</u>	<u>Description or Location</u>
9	Pressure	9 chordwise locations at 0.60 $\lambda$ or 0.85 $\lambda$ spanwise station
1	Acceleration	Front spar at 0.60 $\lambda$ or 0.85 $\lambda$ spanwise station
1	Acceleration	Rear spar at 0.60 $\lambda$ or 0.85 $\lambda$ spanwise station

TABLE I (cont)

(c) Miscellaneous Data

(Recorded in airplane on oscillograph)

No. of Channels	Measurement	Description or Location
2	Velocity (vertical)	Right and left wing on front spar at 0.70Z spanwise station (to check symmetry of airplane response)
2	Velocity (vertical)	Fuselage nose, fuselage tail
3	Velocity (vertical)	Fuselage and wing center section (to determine rigid body pitch, roll, and translation)
2	Velocity (vertical)	Stabilizer tips
1	Velocity (horizontal)	Vertical tail tip
3	Angular displacement	Position transducers located on left wing at 3 aileron hinge points
1	Shaker input signal	From shaker input signal generator
2	Shaker feedback	Right and left shaker displacement potentiometers
2	Airspeed, altitude	Airspeed head on nose boom
3	Maneuver Acceleration	3-component low-frequency accelerometer mounted near airplane cg
2	Log of wing tip acceleration	Vibration amplitude from accelerometer on left wing tip; vibration frequency from input signal generator. Data recorded whenever shaker operates.

**Shakers**

Shakers are installed in each wing panel in the vicinity of the tip. In the next slide (Figure 3) is shown a schematic diagram of one of the shakers.

The principle of operation is that of a simple electro-hydraulic servo system having position feed-

back. A mass, which is free to translate in a direction perpendicular to the plane of the wing, is driven hydraulically by means of an electro-hydraulic servo valve. The valve is actuated by an electrical error signal proportional to the difference between the position of the mass called for by the input signal generator and its actual position which is sensed by a slide wire potentiometer. The force

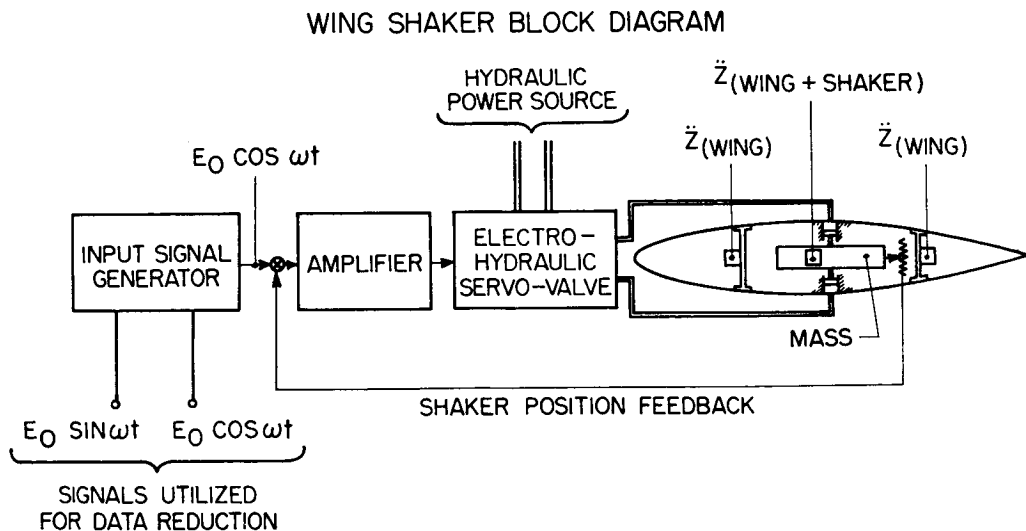


Figure 3. Wing Shaker Block Diagram

output and frequency of the shaker can be controlled independently through adjustment of the voltage level  $E_0$  and frequency  $\omega$  of the electrical input signal which is obtained from a mechanically driven sine-cosine potentiometer. Note that in addition to providing the input signal  $E_0 \cos \omega t$ , the signal generator also provides a signal that is  $90^\circ$  out of phase with the input, i.e.,  $E_0 \sin \omega t$ . Both of these signals are recorded on magnetic tape for use in data reduction which will be discussed later.

The weight of the moving part of the shaker can be varied on the ground from a minimum of 60 lbs. to a maximum of 100 lbs. The maximum displacement amplitude of the moving mass is  $\pm 0.8$  inches. With shaker mass known, the input force to the wing can then be derived from acceleration measurements on the moving mass together with similar measurements on the wing structure ahead of and behind the shaker location. The maximum force output of each shaker is limited by the hydraulic system to a value of about 1,000 lbs. which, for the heavy shaker condition, occurs at frequencies of 11 cps and higher.

By flight flutter testing standards, a forcing function of this magnitude is probably several times greater than would be necessary for adequate response of an airplane of the size used here. In the present application, however, force inputs of this magnitude are believed necessary in order to provide measurable oscillating pressures in the pressure measuring phase of the project.

In the next figure (Figure 4) is shown a listing of the primary shaker controls and indicators to the pilot.

The shaker frequency may be varied either by manual tuning to any desired frequency in the range from 4.5 to 40 cps or by scanning the frequency range by means of a programmed automatic frequency sweep device. With the shaker in automatic sweep operation, the variation of frequency with time is such that the percent change of frequency per cycle is constant ( $\dot{\omega}/\omega^2 = \text{constant}$ ). Thus the sweep rate  $\dot{\omega}$  increases as the square of the frequency. It can be shown, on the basis of the response of a lightly damped single degree of freedom dynamic system, that use of the above frequency sweep relation makes the errors due to sweep independent of where in the surveyed range of frequencies resonance occurs (ref. 2). The time required to cover the frequency range in one direction is adjustable from 15 to 100 seconds.

The amplitude of both shakers is controlled simultaneously by means of one knob which controls the voltage level of the input signal.

Selector switches are provided for choosing between symmetrical and antisymmetrical excitation. To make the excitation as nearly symmetrical or antisymmetrical as possible an effort has been made to match the dynamic characteristics of the two shaker servo systems.

The input signal can be either a sine wave for forced response measurements or a square wave, having a period of 10 seconds for transient response measurements. Note that the square wave signal calls for a succession of abrupt position changes of the mass. Therefore, the force input to the wing is dependent on the dynamic response characteristics of the shaker to a step input signal.

## PRIMARY SHAKER CONTROLS

- |                     |                     |
|---------------------|---------------------|
| 1. FREQUENCY        | 4. INPUT SELECTOR   |
| (a) MANUAL TUNING   | (a) SINE WAVE       |
| (b) AUTOMATIC SWEEP | (b) SQUARE WAVE     |
| 2. AMPLITUDE        | 5. QUICK CUT OFF    |
| 3. PHASE SELECTOR   | (FOR DECAY RECORDS) |
| (a) SYMMETRICAL     |                     |
| (b) ANTISYMMETRICAL |                     |

## INSTRUMENT PANEL DISPLAY

1. FREQUENCY
2. WING TIP ACCELERATION
3. SHAKER AMPLITUDE

Figure 4. Primary Shaker Controls

The last control is the quick cut-off switch, with which the shaker can be stopped within half a cycle. This will be used when measuring the decay of various modes excited by the forced response technique.

To aid the pilot in tuning to resonance and keeping the amplitude of wing response at the desired level, meters are provided which give an indication of the shaker frequency, the amplitude of acceleration at the wing tip, and the amplitude of shaker displacement relative to the wing.

This concludes the discussion of the forced response phase of the project. We will next consider the second phase which is aimed at measuring oscillating chordwise pressure distributions by shaking the wing at various resonant frequencies.

### PRESSURE MEASUREMENTS

The primary measurements in this phase are the pressure differences between the upper and lower surface of the wing at the 60 and 85 percent semispan stations together with acceleration measurement on the front and rear spar at these stations. Again the data will be recorded on magnetic tape. The pressure pick-ups to be used are NACA miniature inductance type gages designed to accurately measure high frequency fluctuating pressures (see ref. 3). These gages, schematically illustrated in Figure 5, utilize

a flat stretched diaphragm which is installed vertically in the wing in order to minimize acceleration effects. By referencing an oscillating pressure to its steady state value through a suitable acoustical filter, only the oscillating part is detected by the gage. The pressure difference between the upper and lower surfaces at a given chordwise location is then obtained by electrically combining the outputs of the upper and lower gages.

Pressure measurements at the two spanwise stations will be made at the 9 chordwise locations shown in the figure. In order to improve accuracy when integrating the pressure distributions, the gages have been placed at points given by Gauss's formula for numerical integration (ref. 4). The locations of the four cells within the 0 to 25 percent chord band satisfy the four ordinate Gauss formula and the remaining five cells between 25 and 75 percent band are positioned to satisfy the 5 ordinate formula.

The theoretical pressure distribution given in the figure indicates approximately the magnitude and phase angle of oscillating pressure that might be expected in flight at the 85 percent semispan station. These results were obtained from the kernel function procedure using the ground measured first bending mode shape to define the downwash boundary conditions. The pressures shown are for a Mach number of 0.9, an altitude of 5,000 feet and a wing tip vibration displacement amplitude of  $\pm 2$  inches. Note that the average pressure amplitude is about  $\pm 0.3$  psi, but to provide for the measurement of much larger pressure fluctuations occasioned by the oscillation of a shock

### WING SECTION SHOWING PRESSURE PICKUP LOCATION AND THEORETICAL OSCILLATING PRESSURE DISTRIBUTION

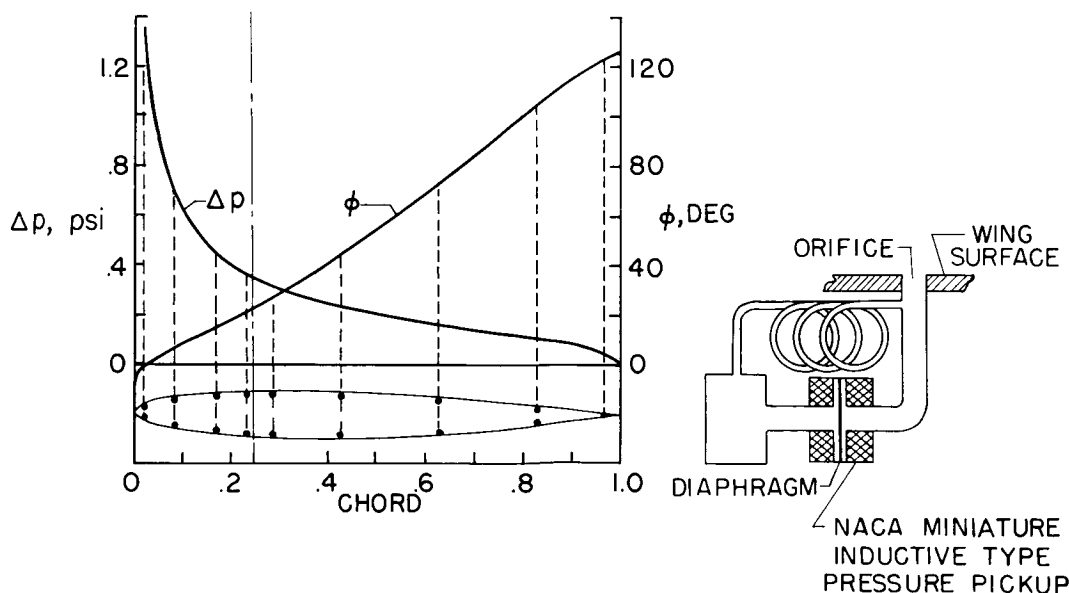


Figure 5. Wing Section Showing Pressure Pickup Location and Theoretical Oscillating Pressure Distribution

wave over an orifice, the gages selected have a range of  $\pm 2.0$  psi.

### MEASURED GROUND MODES

As an indication of the vibration amplitude at the pressure measuring stations, the measured ground mode shapes and node line patterns for the test airplane are shown in the next slide (Figure 6). These modes were excited with electro dynamic shakers attached to the rear spar near the tip of each wing panel. The flight shaker was simulated for these measurements by attaching 130 lb. weights at the location of each of the flight shakers. Note from the plot of node lines that the first antisymmetrical bending ( $f = 9.75$  cps) mode crosses the inboard pressure station at about the  $1/4$  chord point and the second symmetric bending ( $f = 21.7$  cps) node crosses the outboard pressure station at approximately the same chordwise position. The angle between the node lines and pressure orifice bands is about  $45^\circ$  in both cases, indicating that the wing motion at these stations involves considerable torsion. The torsion mode at  $f = 32.5$ , however, may not be adequately excited in the flight tests because the center of the shaker force is very close to the torsional node line.

### Wing Fatigue Considerations

Mention should be made here of the steps that have been taken to assure that the relatively large amplitude shaking, planned in the pressure measurement phase of the project, will not induce structural fatigue failures in the wing. A check against the occurrence of such failures was made by shaking a duplicate wing which had the same structural modifications and shaker installation as incorporated in the flight wing. In these tests each of the modes shown in the figure was excited at the amplitude desired in flight and for a duration ten times as great as the estimated testing time in flight. No evidence of fatigue was discovered. During the flight tests the amplitude and frequency of vibration at the wing tip will be continuously logged and also monitored by the pilot to assure that the safe limits established by the fatigue tests are not exceeded in flight.

### DATA REDUCTION TECHNIQUES

#### Wattmeter Principle of Data Reduction

In reducing the flight forced response and oscillating pressure data it is essential that accurate

### MEASURED WING MODE SHAPES AND NODE LINES

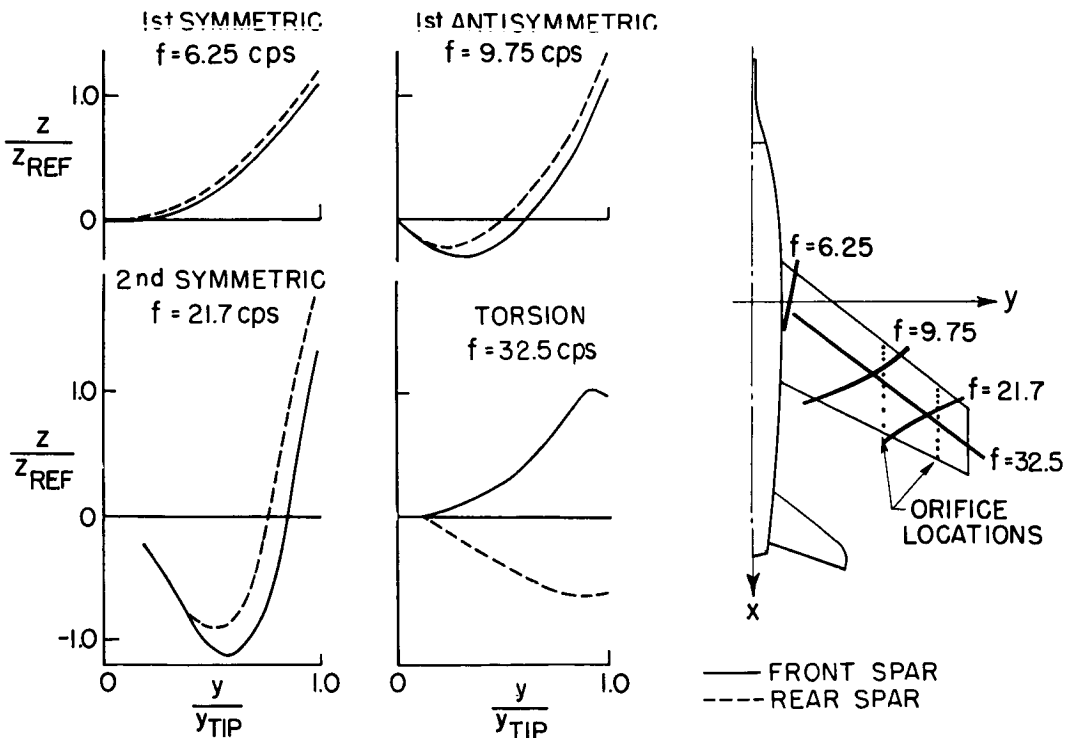


Figure 6. Measured Wing Mode Shapes and Node Lines

measurements are made not only of the amplitude of oscillation but also of the phase angle. With most data reduction techniques the primary difficulty lies in measuring the phase angle accurately. This is especially true when unwanted harmonics are present in the data. This difficulty is avoided, however, by the use of a technique employed by Bratt, Wright, and Tilly (ref. 5) in which separate measurements are made of the vector components of vibration data. The method is known as the "wattmeter" principle of harmonic analysis because just as a wattmeter measures power by indicating the average value of the product of the potential difference and current, the analyzer measures the component of a data signal in phase with a simple harmonic reference signal by indicating the average value of the produce of the two signals.

In the present application of the principle, use is made of an electronic analog computer coupled with magnetic tape play-back equipment. In Figure 7 we see that the principle involved is precisely that of a Fourier analysis. Thus, a periodic data signal

$$F(t) = \frac{A_0}{2} + \sum_{n=1}^{\infty} (A_n \cos n\omega t + B_n \sin n\omega t)$$

is multiplied by a simple harmonic reference signal having the fundamental frequency of the data signal

$$E(t) = E_0 \cos \omega t$$

The resulting product, when averaged, is proportional to the Fourier coefficient  $A_1$ , the factor of proportionality being  $E_0/2$  which is known or can be measured. This is readily seen from the equation for  $A_1$

$$A_1 = \frac{2}{T} \int_0^T F(t) \cos \omega t dt$$

$$= \frac{2}{E_0} (\text{Average value of product } F(t) E(t))$$

In a similar manner,  $B_1$  may be computed by multiplying the data signal by the reference signal shifted  $90^\circ$  in phase, i.e.,  $E_0 \sin \omega t$ . The computer components used for multiplying the signals are high speed quarter square multipliers. These are commercially available electronic devices which have negligible phase shift at frequencies below 100 cps (ref. 6).

As mentioned earlier, the reference signal and its quadrature component are obtained from the input signal generator which drives the shaker. This assures that the frequency of the reference signal is the same as the fundamental frequency of the forced response. Since the reference signal is used as the common frame of reference to which all data vectors are referred, its phase angle relative to the shaker force is entirely arbitrary. The real and imaginary components of a data vector are, then, respectively, the vector's components in phase and  $90^\circ$  out of phase with the reference signal  $E_0 \cos \omega t$ .

#### An Automatic Flight Flutter Test Data Analyzer

Since this Symposium is concerned primarily with flight flutter testing it is of interest to consider the possibility of utilizing the wattmeter principle as a basis for an automatic flight vibration data reduction and analysis system. Three advantages which make this technique of data reduction particularly attractive for handling flight flutter test data are: first, the data can be reduced as the test is being run; second, undesirable harmonics are automatically filtered from the data; and, third, the reduced data, being in the form of vector components, can be conveniently compared with theoretical results.

The system shown in Figure 8 would display a vector plot of the frequency response or admittance (the ratio of the displacement amplitude of a point on the structure to the amplitude of the sinusoidal force

#### WATTMETER PRINCIPLE OF HARMONIC ANALYSIS

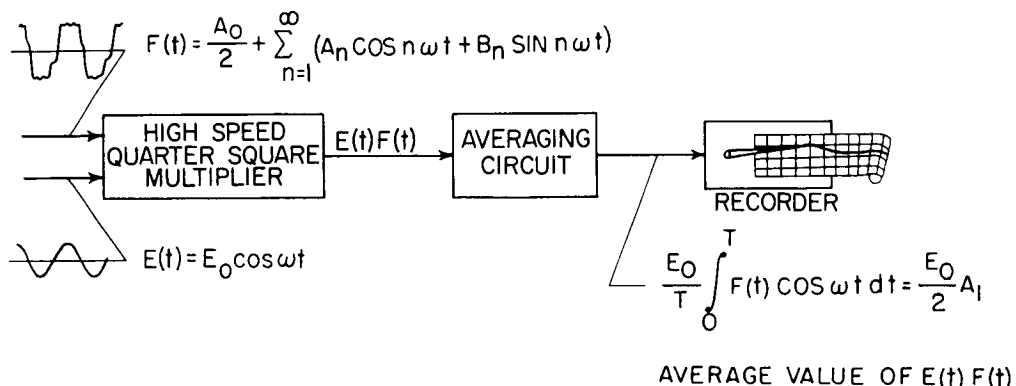


Figure 7. Wattmeter Principle of Harmonic Analysis



# AUTOMATIC DATA REDUCTION AND STABILITY INDEX PLOTTER

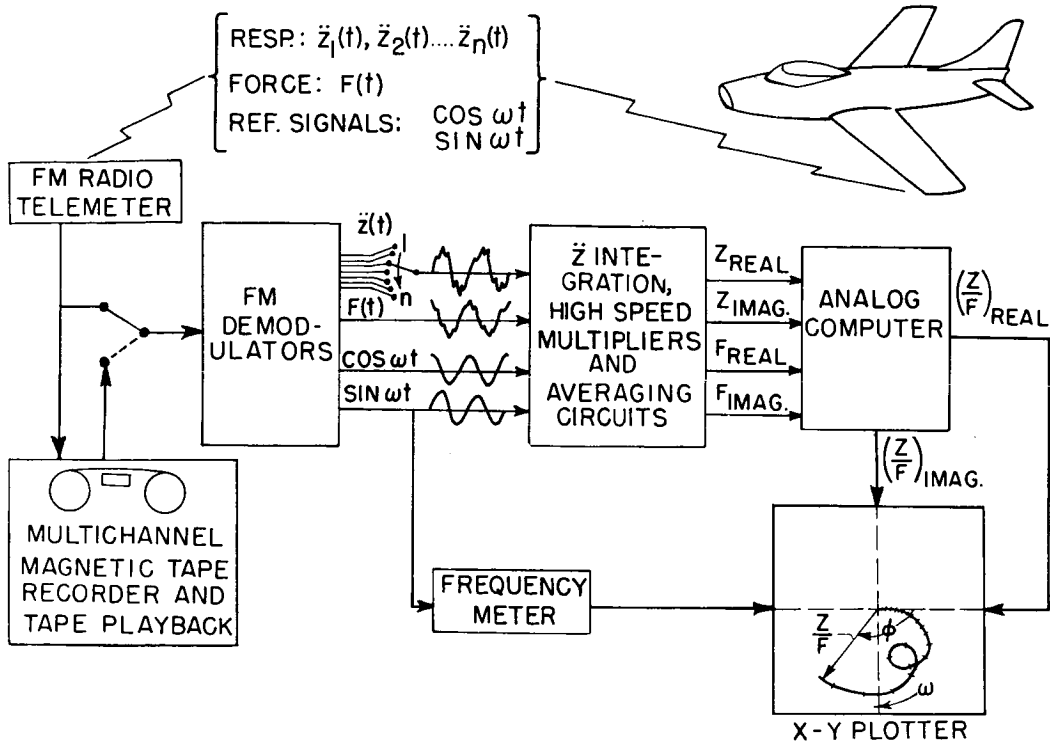


Figure 8. Automatic Data Reduction and Stability Index Plotter

that causes the displacement) for a selected pick-up location on the airplane as the frequency of excitation is varied over the range of interest. The use of vector response plots in the analysis of airplane ground vibration response data has been discussed by Kennedy and Panu in reference 6 and much similar work of this type has been developed for stability analyses relating to feedback amplifiers (reference 7) and servo mechanisms (reference 8).

No attempt will be made here to discuss the merits of vector plotting other than to say that results of theoretical forced response analyses, such as the influence coefficient method discussed earlier in the paper, can also be conveniently presented in the form of vector response plots for ready comparison with experiment.

To illustrate the system, assume that the test vehicle is instrumented to telemeter the following data: acceleration response at various points of interest on the structure  $\ddot{z}_1(t), \ddot{z}_2(t) \dots \ddot{z}_n(t)$  the excitation force  $F(t)$ , a simple harmonic reference signal  $E_0 \cos \omega t$  that has the fundamental frequency of the exciter, and the component  $90^\circ$  out of phase with the reference signal  $E_0 \sin \omega t$ . These data are recorded on magnetic tape at the ground telemeter receiving station while at the same time the acceleration response signal selected to be analyzed during the frequency sweep is fed to the analyzer, together

with the shaker input force signal and the two reference signals. The acceleration response is double integrated to give  $Z(t)$  which, in turn, is multiplied by the reference signals and averaged to give outputs proportional to the real and imaginary components of  $Z$ . In a like manner,  $F(t)$  is multiplied by the reference signals and averaged to give an output proportional to the real and imaginary components of  $F$ . Having the vector components of  $Z$  and  $F$ , an analog computer performs the arithmetic operations required to obtain the vector components of the frequency response  $(Z/F)_{\text{real}}$  and  $(Z/F)_{\text{imag}}$ . Note that since the vector components of  $Z$  and  $F$  are slowly varying quantities whose rates of change with time for a given system depend upon the frequency sweep rate, high speed multipliers are not required in this stage of the analog computer.

Next, the real and imaginary components of  $Z/F$  are connected to an X-Y plotter in a manner such that  $(Z/F)_{\text{real}}$  drives the recorded pen along the X-axis of the plotter and  $(Z/F)_{\text{imag}}$  drives the pen along the Y-axis. Thus, as the shaker frequency is varied, the plotter maps the locus of the vector  $Z/F$ . The amplitude of the vector at a given frequency is determined by the length of a line drawn between the curve and the origin of the real and imaginary axes, and the phase angle is the angle between this line and the positive real axis. The frequency of forced response is indicated by feeding the reference signal

to a frequency measuring device which pulses the recorded pen at equal frequency increments.

Thus the frequency response for one of the pick-up locations is plotted during the test. At a later time, perhaps while the pilot maneuvers for the next test run, the data on magnetic tape can be played back into the analyzer and other channels selected for plotting.

#### CONCLUDING REMARKS

To sum up, we have discussed some of the equipment, instrumentation and data reduction techniques to be used in a project aimed at measuring oscillating air forces in flight. Also, we have considered some possible applications of these techniques to the problem of flight flutter testing. The equipment and instrumentation is now being installed in the airplane and flight test data on the forced response phase of the project should be available in the near future.

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