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INFLIGHT AIRCRAFT VIBRATION MODES
AND THEIR EFFECT ON AIRCRAFT RADAR
CROSS-SECTION

Final Technical Report

August 1977-August 1978

Prepared By:

William J. Anderson
Dipak L. Sengupta
Sanjay Correa

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Rome Air Development Center
Griffiss Air Force Base, New York 13441

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

A theoretical study is done for the identification of aircraft by type through elastic inflight modes and their radar signature. Three contemporary U.S. fighter and fighter-bomber aircraft are studied for inflight vibration amplitudes due to sharp-edged gusts and random turbulence. The resulting elastic and rigid body motions are found to be relatively small and at relatively low frequencies for radar detection. The rigid body response to turbulence at frequencies above 2 hz for these aircraft is found to be less than 0.01 inch rms over 95 percent of the flight path.

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Response to sharp-edged gusts can be more pronounced. Elastic wing tip deflections range from 3/4" to 1/8" over 2 ft/sec. gusts; gusts of this severity are routinely observed over only 5 percent of the typical flight path.

In addition to small deflections, another problem lies in the unique characterization of aircraft by their modes. For two of the three aircraft, the modal frequencies and shapes vary greatly with fuel and armament load. The third aircraft (the fighter-bomber), however, has a fundamental elastic frequency which is relatively invariant with change in airspeed, fuel load and wing sweep. This mode is predominantly fuselage-bending and was used as a candidate for radar cross-section studies.

Radar cross-section studies were done for varying view angles lying in the vertical plane of symmetry for the aircraft. An RCS model was based on a collection of independent scatterers identified with various components of the aircraft. Elastic deflections due to a 2 ft/sec. sharp-edged gust were observable with 3 cm radar wavelength. A number of figures are given for static and dynamic radar scattering cross-section. The frequency content of the radar return has quite strong third harmonic components at 3 cm.

EVALUATION

1. For noncooperative tactical aircraft identification, it is essential to get as much supportive evidence as possible to make valid decisions. Studies at RADC/EE raised the question of whether sub-structural motions of a target could generate identification information. The present study by the University of Michigan was initiated as part of a program in target modulated signatures. Its specific goal was to determine whether aeroelastic airframe motions induced by atmospheric forces could be used to obtain a radar signature. The approach involved studying the mechanical phenomena involved in such interactions to obtain data about the frequencies, mode shapes, and displacements for three classes of tactical aircraft. The variations that occur for ranges of velocity, fuel loading, external stores, and wing position were examined. The scope of the study was restricted; the implications of the calculated substructural motions in relation to corresponding changes of electromagnetic scattering centers were addressed only peripherally. The complexity of the vibrational patterns and the limited deflections that result would tend to make it difficult to observe characteristic modulations of a radar signal.

John F. Lennon

JOHN F. LENNON
Contract Monitor

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I. INTRODUCTION

This report covers theoretical studies of structural motion, aeroelastic vibration and radar scattering characteristics of aircraft subject to gusts and turbulence. Three aircraft have been chosen as test cases, a variable configuration fighter/bomber (hereafter called Type A and typified by the F-111), a swept-wing fighter (Type B, typified by the F-4) and a small fighter with relatively straight wings (Type C, typified by the F-5). The weights of these aircraft are given in Table I. The weight range studied varies from 77,302 lb. (Type A, with wet wing and fuselage) to 15,265 (Type C). The modes of vibration for these aircraft have been studied as functions of fuel load, armament load and airspeed. Both frequencies and mode shapes in flight have been determined.

A major tool for this study has been the computer program "FACES," developed by the Flight Dynamics Laboratory at Wright-Patterson Air Force Base [1, 2, 3]. This program considers the elastic forces in the fuselage and wings, the inertial characteristics of the entire plane, and the aerodynamic forces on the wings (strip theory). With substantial effort and computational cost, the aerodynamic forces can be extended to the fuselage through the use of the doublet-lattice method. Strip theory has been used here, however.

This report (1) studies 12 combinations of aircraft type and weight, (2) considers mode <u>shape</u> as well as frequency, (3) discusses the invariance of modes and frequencies, (4) establishes expected amplitudes of motion, and (5) constructs and studies a radar scattering model of the aircraft.

In several regards, the work done here differs from conventional structural studies of aircraft in flight. Most structural studies concentrate on stresses at the wing root, accelerations, or possibly interaction of structure and the fluid flow (as in flutter). The emphasis here, however, is on the motion of the aircraft as perceived by a distant radar site. In this regard, the internal stress, strain and accelerations of the aircraft are of no importance, whereas the displacement field is very important.

II. TURBULENCE

A literature survey has been carried out for references on (1) atmospheric turbulence, (2) methodology for calculating elastic aircraft response to turbulence, (3) structural data and vibration characteristics of specific aircraft, (4) effects of aging on structural response. Approximately 250 references have been cited in this survey, which have been entered into a computer file for convenient modification. Twenty-five of the more relevant papers have been read, and a brief critical summary given following the references. The most often cited journals are the AIAA Journal, Journal of Aircraft and Journal of Atmospheric Sciences as well as Air Force and NASA reports. The bibliography is included as an Appendix to this report.

The literature in the field of atmospheric turbulence appears to be well developed. There are also many papers available in the area of elastic aircraft response to turbulence. Papers on specific aircraft (such as the F4) are less prevalent in the open literature and the few papers on aging seem to concentrate on composites and glue strength. The literature appears to be adequate for the purpose of obtaining general information on aerodynamic motion required by the project.

There are several causes of atmospheric turbulence with the primary ones being the sun's energy and the whirling motion of the earth [4]. The mechanics of turbulence involve wind shear and convection along with some man-made effects, such as wake turbulence behind aircraft.

For the radar return problem, turbulence caused by convection will be of the most interest since it is the only source of excitation over most flight paths. This convective turbulence occurs in patches hovering over the earth's surface. An aircraft spends perhaps five percent of a typical flight in one or more of these patches. The patches consist of repeated patterns of convection cells of which two varieties have been observed. Hardy and Ottersten [5] state: "One pattern consists of small thermal-like cells which are 1-3 km in diameter and several hundred meters in height. ... The other pattern is made up of clear air Bernard-like convection cells ... which are 5-10 km in diameter and 1-2 km in height ...".

An airplane passing through such a turbulence patch experiences a random force field due to velocity fluctuations u, v and w in the relative velocity between aircraft and air mass. The most significant perturbation is the upward perturbation component (indeed, it is the only one considered in the related problem of sharp-edged gusts). The velocity fluctuations are approximately isotropic and a stationary random theory is usually employed. There is general agreement that von Karman has presented the best expression for power spectral density of velocity fluctuations in isotropic turbulence. The turbulence is often considered "cylindrical," i.e., constant in magnitude along the wing span. Along with the assumption of stationarity, this two-dimensional assumption makes the problem of finding aircraft response tractable.

One could imagine the nonstationary random problem to be important, i.e., the short-time behavior of an airplane suddenly exposed to turbulence might be more severe than the stationary case. This has been studied by several researchers using an envelope-modulated stationary random input. Fujimora [6,7] found that sudden onset of a stationary random forcing function can cause 28% more acceleration at the aircraft center of gravity than stationary random forcing. This fact is much more of a concern in stress analysis than in the radar return problem, however, and should be ignored here.

Finally, the cylindrical nature of the turbulence is examined by Coupry [8]. He claims that spanwise variations in the turbulence cause enough cancellation of lift to smooth out the predicted ride. For the purposes of radar modulation by elastic modes, this is important because it means that simpler theories ("cylindrical" waves) will over-predict the aircraft response. There is a scale factor involved, i.e., the ratio of length of coherence over wingspan. For aircraft as large as the Concorde, the spanwise effect reduces the peak response by a factor of two. For fighter aircraft, the cylindrical assumption will probably overpredict response by about ten percent.

Several solutions using classical methods have been carried out for aircraft response to continuous random turbulence and to sharp-edged gusts. (See Section V.) These provide some feeling for the amplitudes of motion. In one calculation, a type B aircraft flying at 600 mph at sea level has an rms vertical velocity in rigid-body plunging of 3 in/sec while passing through continuous, moderate turbulence with rms vertical velocity component of 6 in/sec. In other calculations, the elastic response to a 2 ft/sec sharp-edged vertical gust is found to be less than 3/4-inch over the entire aircraft and as little as 1/8-inch in many cases. Observation of motion this small may require X-band or shorter wavelength radar.

III. INFLIGHT STRUCTURAL MODES AND FREQUENCIES

3.1 Inflight Frequencies

The program FACES provides the natural frequencies and modes of the aircraft structure on the ground and in flight. These results stem from a solution of the coupled eigenvalue problem including effects of elastic fuselage, elastic wing, elastic stores and aerodynamic flow. The inflight frequencies are given directly in numerical tables whereas the inflight modes are given indirectly in tables of modal participation factors. A coordinate transformation must therefore be done to recover the inflight modes. No information is given in FACES about the response problem (specific motion due to external forces); however, the eigenvalue work from it serves as the background for all such response work done here.

Volume I of the FACES manual [1] illustrates many of the natural frequencies and natural mode shapes for the type B aircraft. Volume II of the FACES manual [2] gives extensive inflight modal frequency data for the type B aircraft. Inflight mode shapes are not given in these manuals, however.

In the current study, the type A, B and C aircraft have been modelled using data provided by the FACES manual (type B) and by Wright-Patterson Air Force Base (types A and C). In each case the properties of the elastic wing, elastic fuselage and elastic stores are separately found and entered into the program. These data can be calculated with some accuracy and have been tabulated in company reports on basic data for each aircraft. Perhaps the weakest aspect of the structural model of the airplane is the choice of elastic root restraint. The stiffness of the wing carry-through structure in the fuselage is not well documented; indeed, the "attachment point" called for in the FACES program is an artifice.* The root restraint, which accounts for the fuselage-wing

*The exception is in the swing-swing aircraft where the pivot point is literally an attachment point. Even here, however, the properties are not tabulated in the form needed by FACES.

interaction, is given values that are "strictly the user's own choice" (Ferman [1], p. 93). In practice, stiffnesses about ten times the wing stiffness at the root give reasonable answers.

3.2 Frequencies for Type A Aircraft (Swing-wing Fighter Bomber)

Inflight modal frequencies are given in Figures 1-5 (full forward sweep), and Figures 6-10 (full rearward sweep). A sequence of cases is studied for dry, partially fueled and fully-fueled cases. Several of the modal frequencies remain constant with airspeed; these same frequencies have mode shapes with little phase lag in time, and tend to remain relatively "pure," i.e., do not couple with adjacent modes. Figures 4 and 9 are crossplots of Figures 1-3 and 6-8 respectively, taking data at 500 knots and considering the effect of fuel load on flight frequencies. The fuel is carried internally in the fuselage and wings and has a moderate effect on the modal frequencies. The numbering system given on the curves has to do with mode shapes and will be discussed in Section 3.6.

Ground vibration frequencies for forward and swept wings are shown in Figures 5 and 10 respectively. These are helpful for mode identification studies done later.

3.3 Frequencies for Type B Aircraft (Swept-wing Fighter)

Inflight modal frequencies are given in Figures 11-13. Each figure has a different number of pylons and armament. Figure 11 is a partially fueled aircraft with no armament and Figures 12 and 13 consider 4 and 8 pylons respectively. The corresponding armament is listed in Table 1. Some of the modal frequencies remain constant with airspeed, but as seen in the crossplot in Figure 14, there is a substantial drop in modal frequencies with increasing armament load on pylons. The figures include all effects of the coupled elastic fuselage and elastic wing. In this particular study, the pylons were assummed rigid so as to eliminate the additional elastic degrees of freedom, which are interspersed with the dominant wing and fuselage motion and which make problems in identifying modes. The neglect of elastic pylon effects means that candidate modes for identification will have to be studied further for this complication. The fully elastic pylon cases have been run for the Type B airplane and are very difficult to interpret.

Ground vibration frequencies are given in Figure 15.

3.4 Frequencies for Type C Aircraft (Straight-wing, lightweight fighter)

The inflight modal frequencies are given in Figures 16-19. These frequencies are higher than for the larger aircraft. The crossplot of frequency variation with weight in Figure 19 shows a dramatic decrease of frequencies as armament load increases. This aircraft was the most sensitive of the three in this regard.

The ground frequencies are shown in Figure 20.

This aircraft, more than the others, poses a real threat to mode identification as armament weight changes. Both the inflight and ground frequencies become very scrambled as weight changes, which is why no connecting lines are given between data points in Figures 19 and 20. A numerical mode tracking method to be discussed in Section 3.6 attempts to provide the continuity as shown in Figures 21 and 22, but is not very helpful.

3.5 Composite Frequency Study (Types A, B and C)

Because there seems to be a relation between the weight of the aircraft and the modal frequencies, a composite plot of modal frequencies at 500 knots is given in Figure 23. There is no general trend in the data; however, one might speculate whether the modes of the empty aircraft of each fighter type might be somewhat more predictable than heavily-loaded aircraft.

3.6 Inflight Mode Shapes

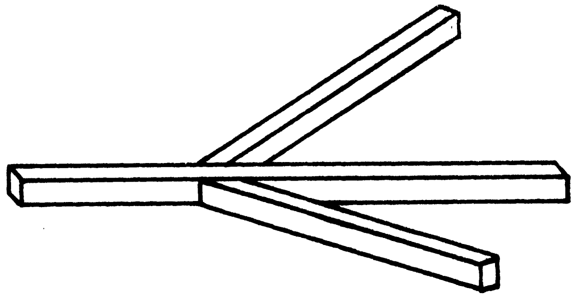
The previous work has dealt primarily with modal frequencies. Let us now turn our attention to the modal shapes.

The inflight mode shapes are the eigenfunctions for the airplane in the presence of an airstream. These modes are aerodynamically damped at speeds below the flutter speed. Above the flutter speed, one or more of the modes are unstable and grow in amplitude with time.

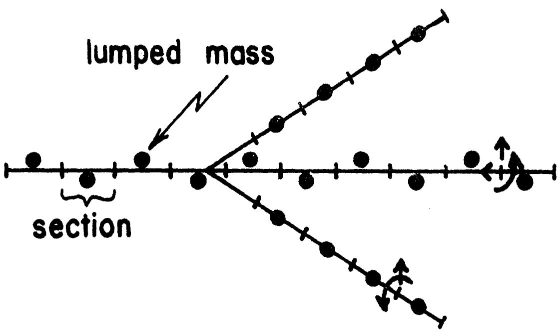
The previous sections considered inflight and ground frequencies (eigenvalues), and the variation of inflight frequencies with airspeed and loading (Figures 1-23). It is necessary, however, to consider the mode shapes for two reasons. The first is that the radar return depends on the relative amplitudes of different reflection points, lines and surfaces. The second is that mode tracking (following each mode as airspeed and weight change) is difficult without knowledge of mode shapes to distinguish them when frequencies are closely packed.

The modes and frequencies are calculated through the idealized models shown in Sketches 1-3. The structural stiffness is developed using finite sections of beams which can be serially kinked. The inertia of fuselage and wing sections is located appropriately within each section at the center of gravity of the section. Each section has rotary as well as translational moment of inertia. The degrees of freedom are identified at section boundaries and consist of wing z deflection and torsional angle about the elastic axis, as well as fuselage vertical deflection, forward displacement and rotation about the pitch axis. The aerodynamic forces used in the model are based on strip theory, and act on the wing only (Sketch 3).

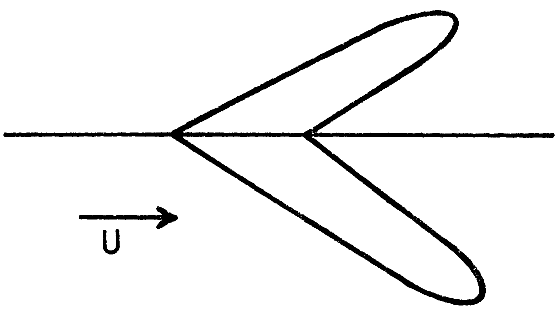
Of the five available degrees of freedom at each section boundary, three are considered relevant to the radar problem for symmetric motion. These are



Sketch 1. Elastic model used in FACES. Beam theory. Fuselage and wing can be kinked.



Sketch 2. Finite sections used in FACES. Can be serially kinked. Three degrees of freedom at fuselage stations and two degrees of freedom at wing stations for symmetric motion.



Sketch 3. Aerodynamic model used in FACES. No aerodynamic forces on fuselage or tail.

the wing bending deflection, wing torsional rotation and fuselage bending deflection, as plotted in Figures 24a to 27c. These twelve figures show the fundamental mode for each of the four major geometries studied, with all at medium weight. The z deflections are in feet and the torsion is in radians. The motion in each case consists of an in-phase and leading component. The motion is not synchronous, i.e., different points along wing and fuselage are not in phase with each other. The deflection for the wing w(y,t) could be written

$$w(y,t) = f_1(y)e^{i\omega_i t} + f_2(y)e^{i(\omega_i t + \pi/2)}$$

for instance, where $f_1(y)$ and $f_2(y)$ are plotted as solid and dotted lines.

Discussing mode shapes for a complex structure is more difficult than discussing frequencies, which are scalars. An attempt is made here to quantify the mode shapes so that modes can be "tracked" and their invariance (or lack of it) determined. The method used is to consider the maximum amplitudes in wing bending, wing torsion and fuselage bending. To compare on the basis of length scales, the wing torsional angle is multiplied by the mean half chord of the wing. This corresponds approximately to the distance the leading edge and trailing edge move vertically and is a reasonable way to judge the effect of torsion on the radar return. (Leading edges may be good reflectors.) A code for each mode shape has been worked out for a normalized vector of length 100, where the mode illustrated in Figures 24-26 would be represented by

(17 02 98)

component of wing bending

wing torsion

fuselage bending

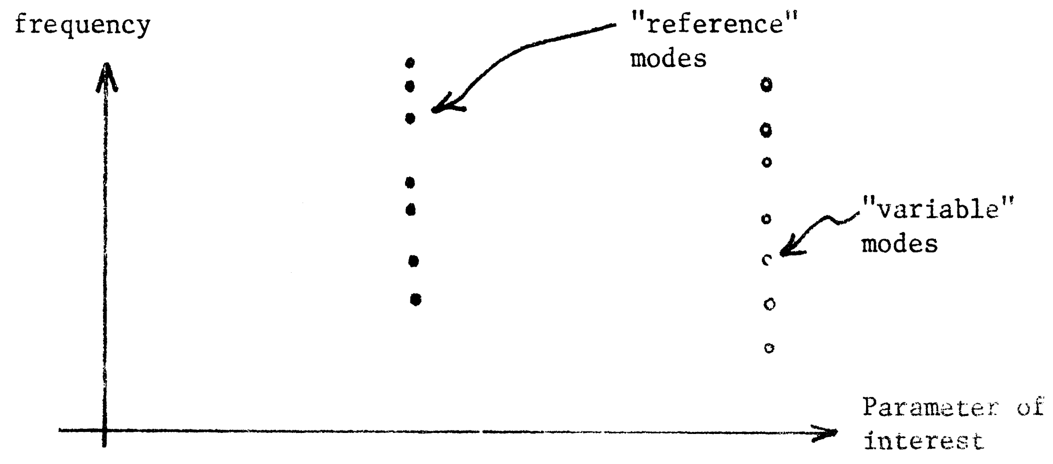
Sketch 4. Mode-shape code.

This mode is easily seen from the code to be dominantly fuselage bending, whereas deducing this from the figures takes some effort. Furthermore, comparisons between mode shapes can be made, and differences quantified. Using the code discussed above, the frequencies displayed in Figures 1-23 have been "tagged" with their corresponding mode shapes.

3.7 "Optimal" Mode Tracking

Both modal shapes and modal frequencies change with the three parameters considered: airplane load, airspeed and wing sweep. Mode-tracking thus involves following a particular mode as any parameter is varied. The modal frequency is not a good indicator of the modal number, because a modal crossover upsets a notation that is frequency-ordered. This leaves the mode shape as the possible "tag" on a mode; the shape has been described with a numerical string quantifying the contributions from wing bending, wing torsion and fuselage bending. This six-digit code is used as the "tracer" in the mode-tracking process.

The mode-tracking is optimized on the basis of shape. The eight "variable" modes in Sketch 5 can be permuted in 8! or 40,320 ways; each permutation represents a unique mapping of the "reference" modes onto the "variable" modes.



Sketch 5. Permutation of modal frequencies.

For each of these mappings, the sum, over all eight modes, of the squares of the differences between the two-digit code numbers, for a reference mode and the variable mode, is computed. The minimum of the 40,320 sums thus obtained corresponds to the "optimal" mode-tracking; a root-mean-square error can be obtained from this minimum sum. This procedure:

(1) searches for a global minimum and so often discards intuitively "comfortable" mappings between lower modes.

(2) disregards the possibility of picking up variable modes from or losing reference modes to higher frequencies.

Typical results of this mode-tracking procedure are illustrated for aircraft Types B and C in Figures 21 and 22. One can appreciate from these figures the difficult in tracking modes for aircraft Types B and C.

If one can track the modes with enough confidence in a given case, the next question is whether the frequencies and mode shapes along the properly tracked mode are invariant or not. This is a more detailed question than tracking and is discussed in the next section.

IV. INVARIANCE OF INFLIGHT MODES

The information contained in Figures 1-23 is the factual basis for determining invariance for the three aircraft. The question of how to define invariance of modes is somewhat subjective, but is basically whether modal frequencies and shapes vary excessively with change in aircraft loading, speed and configuration. What is excessive from the radar return pattern is critical and cannot be completely determined at this point.

Although the first 8 modes have been studied, identification doesn't require invariance of all 8 modes. The aircraft motion can be shown to be dominated by the fundamental mode with some contribution from the second and third modes.

Some generalizations to be drawn from the results include:

(1) Variation of mode shape and frequency with airspeed is generally modest and could be accounted for if it were the sole effect.

(2) Variation of mode shape and frequency with fuel and armament load is great. Modes are scrambled so much that it is difficult to track them, even analytically for a given aircraft where no noise is present (at discrete, calculated points).

(3) Variation of mode shape and frequency with wing sweep for the type A aircraft is not severe and could be accounted for if it were the only effect.

Only the Type A aircraft appears to be a candidate for identification by invariance of an elastic mode. Its fundamental mode (fuselage bending) varies only from 5.8 to 6.8 hz with wide changes in airspeed, loading and wing sweep. Unfortunately, other fighter aircraft in the air with their profusion of frequencies can, for certain stores combinations as seen in Figure 23, mimic the Type A aircraft elastic frequency. Therefore, one would have to search for a unique radar return due to the mode shape of the Type A aircraft.

Another necessary condition for identification is that the modes in question must be excited enough by gusts and turbulence to be observed. The amplitude of this response is studied in the next section.

V. AMPLITUDE RESPONSE TO GUSTS AND TURBULENCE

Three separate gust and turbulence problems will now be considered. The purpose will be to develop insight into the amplitude of response of the aircraft. The cases studied differ in whether the airplane is considered rigid or elastic, and whether the turbulence is modelled by a sharp-edged gust or as a stationary-random process. Of the four possible combinations of these effects, the most complicated one (elastic response to stationary random turbulence) is not considered because of its difficulty and limited scope of this study. One can determine the major effects from the first three cases, however.

5.1 Rigid Body Response to Sharp-Edged Gusts

A simple calculation of the rigid body response to a sharp-edged gust will be made. Only the plunging motion, and not pitching, will be considered.

The type B aircraft will be assumed to have the following flight characteristics:

mass = m = 1696 slugs

weight $\equiv W = 54{,}600$ lb.

wing area $\equiv S = 530$ ft^2

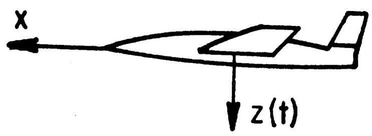
speed $\equiv U = 880$ ft/sec (600 mph)

air density at 10,000 ft =

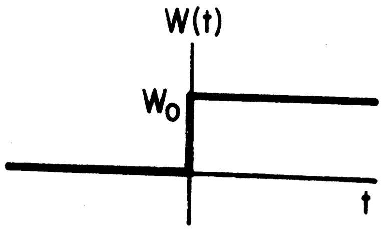
= .000582 slugs/ft^3

air density at sea level =

= 0.002378 slugs/ft^3



Sketch 6. Coordinate System.



Sketch 7. Sharp-edged Gust.

The lift-curve slope, C_{L_α}, could be calculated by procedures outlined by Roskam [9]. The procedure is somewhat involved, however; therefore, C_{L_α} will be estimated at an intermediate value of $C_{L_\alpha} = 3.0$.

The sharp-edged gust will be assumed to have intensity $w_o = 2.0$ ft/sec, which corresponds to the extreme value measured at least once per ten seconds in "turbulent patches." These turbulent patches cover only five percent of the flight path, hence, this magnitude of sharp-edged gust is an upper limit to gusts that could be observed routinely.

The response in plunging of a rigid aircraft to a sharp-edged gust is given by Fung [10]. For a gust velocity

$$w(t) = w_o H(t) \quad \text{(positive upward)}$$

one obtains a response

$$z(t) = \frac{1}{\lambda} w_o (1-e^{-\lambda t}) - w_o t$$

where the characteristic time required to approach a steady motion is $\frac{1}{\lambda}$, where

$$\lambda = \frac{\rho US}{2m} \frac{dC_L}{d\alpha}$$

For our problems, at sea level, $\lambda = 0.981 \text{ sec}^{-1}$ and at 40,000 ft, $\lambda = 0.240 \text{ sec}^{-1}$. Hence

$$z_{SL}(t) = 2.039 (1-e^{-.981t}) - 2t \quad \text{(ft, where t is in seconds)}$$

$$z_{40,000}(t) = 8.333 (1-e^{-.240t}) - 2t \quad \text{(ft, where t is in seconds)}$$

The solution for this plunging response is shown in Figure 28. The figure confirms the characteristic time of 1 second at sea level and 4 seconds at altitude during which the aircraft accelerates upward to a terminal velocity equal to the gust speed.

The maximum acceleration of the aircraft is at t = 0 and is $- w_o\lambda$. To compare this acceleration with that due to gravity, one divides to get the load factor Δn:

$$\Delta n \equiv \frac{|\ddot{z} \max|}{g}$$

$$= \frac{W_o \lambda}{g} .$$

At sea level

$$\Delta n = 0.061 \text{ g.}$$

At 40,000 feet

$$\Delta n = 0.015 \text{ g.}$$

Both values are relatively small and indicate that the gust is mild.

5.2 Elastic Response to Sharp-Edged Gusts

Background

The vibration signatures of structures are characterized by mode shapes, frequencies and amplitudes of response to various inputs. The response is divided between the various modes, with modal content typically decreasing for the higher modes. For aircraft structures, the ambient turbulent field provides aerodynamic inputs which excite these modes. The sharp-edged gust, a simplification of the actual (random) field, provides useful information on the aircraft vibration-signature.

Beyond this simplification of the input, the aircraft structure itself will be idealized as a rigid fuselage with an unswept, straight, slender wing. Arbitrary spanwise distributions of mass, stiffness and chord are allowed. The entire airplane is free in vertical translation, or 'plunging,' and the wing is elastic in bending; all torsional modes are assumed to be restrained. The modal analysis detailed here follows Bisplinghoff, et al. [11]

It is to be expected that these theoretical results will overestimate the elastic response. The actual wings on the aircraft studied are swept, causing less lift per unit span and initiating lift at different times along the span. In contrast, the theory applies to a larger lift instantaneously along the entire wing, which is a more severe loading condition.

The theory is intended to provide approximate values for elastic wing motion. This will serve as an indication of the wavelength of radar required to observe the motion. Each of the modes will respond with its own amplitude; hence the results should distinguish between observable and nonobservable modes.

Symbols

b_R	reference semi-chord
$a(y)$	spanwise chord distribution
b	semi-chord at station y
S	wing area
m	running wing-mass
ℓ	wing semi-span
M	total airplane mass
U	airplane forward velocity
$w_G(\sigma)$	gust velocity profile
ρ	density of ambient air
n	total number of modes considered
ω_i	bending frequencies of wing, $\omega_1 = 0$
s	nondimensional time
$w(y,t)$	vertical displacement at station y
$\phi_1(y)$	rigid-body mode shape, $\phi_1(y) = 1.0$
$\phi_2(y)\ldots\phi_n(y)$	shapes of vibratory modes of wing
$\xi_1(t)$	response of 'plunging' mode
$\xi_2(t)\ldots\xi_n(t)$	normal coordinates representing responses of vibratory modes
Φ	Wagner function
Ψ	Kussner function
Ξ^{D_i}	generalized forces due to gust
Ξ^{M_i}	generalized forces due to motion
$(\ )'$	prime denotes derivative with respect to the argument

Equations of Motion

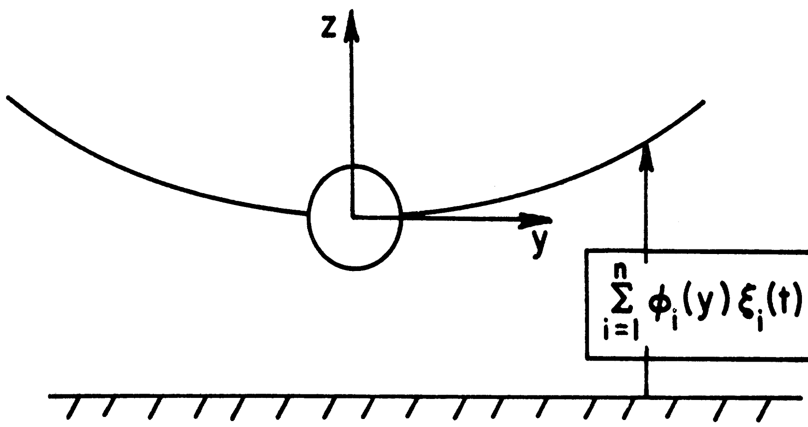
The response of the aircraft is separated into a time-dependent component (t) and a spatial component $\phi(y)$, with the total response then given by

$$w(y,t) = \sum_{j=1}^{n} \xi_j(t)\phi_j(y).$$

The mode shapes are normalized so that

$$M = 2 \int_0^{\ell} m\, \phi_i^2 dy \qquad i=1,\ldots,n.$$

Positive coordinate directions are indicated in Sketch 8.



Sketch 8. Coordinate System.

The response is given by the solution of the differential equations

$$\lambda\xi_i''(s) + \lambda\Omega_i^2\xi_i(s) + \sum_{j=1}^{n} A_{ij}\xi_j''(s) + 2 \sum_{j=1}^{n} B_{ij} \int_0^s \xi_j''(\sigma)\Phi\,(s-\sigma)d\sigma$$

$$= 2\, b_R\, B_{1i} \int_0^s \frac{w_G(\sigma)}{U}\, \Psi'\,(s-\sigma)d\sigma$$

$$i=1,\ldots,n; \qquad \omega_1 = 0$$

where:

$$b = a(y)b_R$$

$$s = Ut/b_R$$

$$\lambda = M/(\pi \rho S b_R)$$

$$\Omega_i = \omega_i b_R/U$$

$$A_{ij} = (b_R/S) \int_{-\ell}^{+\ell} a(y)^2 \phi_i \phi_j dy$$

$$B_{ij} = (b_R/S) \int_{-\ell}^{+\ell} a(y) \phi_i \phi_j dy \tag{5.1}$$

[Eq. 10-149, Ref. [11].]

With a step-function gust velocity input, the gust velocity profile $w_G(\sigma)$ is

$$w_G(\sigma) = w_G(0^+) = w_G,$$

which allows modification of the nonhomogeneous terms in the system of equations. From Eq. 5-382 [11], the unsteady lift due to the gust is

$$L = 2\pi \rho U b \{w_G(0)\Psi(s) + \int_0^s \frac{d w_G(\sigma)}{d\sigma} (s-\sigma)d\sigma\}$$

which, for $w_G(\sigma) = w_G$, reduces to

$$L = 2\pi \rho U b w_G(0)\Psi(s),$$

since

$$\int_0^s \frac{d w_G(\sigma)}{d\sigma} \Psi(s-\sigma)d\sigma = 0.$$

Hence, the spanwise lift due to the gust is

$$L_G(y,s) = 2\pi \rho U a(y)b_R w_G \Psi(s).$$

The definition of generalized force due to the gust is (Eq. 10-143, Ref. [11]):

$$\Xi^{D_i} = \int_{-\ell}^{+\ell} \phi_i \, L_G \, dy$$

or

$$\Xi^{D_i} = 2\pi \rho U b_R w_G \Psi(s) \int_{-\ell}^{+\ell} \phi_i \, a(y) dy \qquad (5.2)$$

From Eq. 10-142 [11], the equations of motion are

$$M_i \frac{d^2}{dt^2} \xi_i + M_i \omega_i^2 \xi_i = \Xi^{M_i} + \Xi^{D_i}.$$

Transforming to nondimensional time,

$$M_i \frac{U^2}{b_R^2} \xi_i'' + M_i \frac{U^2}{b_R^2} \Omega_i^2 \xi_i = \Xi^{M_i} + \Xi^{D_i}$$

and so

$$\lambda \xi_i'' + \lambda \Omega_i^2 \xi_i = \frac{b_R \Xi^{M_i}}{\pi \rho S U^2} + \frac{b_R \Xi^{D_i}}{\pi \rho S U^2}$$

The last term becomes, on substitution from (5.2),

$$\frac{b_R \Xi^{D_i}}{\pi \rho S U^2} = \frac{2 b_R}{U} w_G \Psi(s) B_{1i}$$

Hence, for a step-function gust velocity input and for symmetrical motion, equations (5.1) become

$$\lambda \xi_i''(s) + \lambda \Omega_i^2 \xi_i(s) + \sum_{j=1}^{n} A_{ij} \xi_j''(s) + 2 \sum_{j=1}^{n} B_{ij} \int_0^s \xi_j''(\sigma)\Phi(s-\sigma)d\sigma$$

$$= \frac{2 b_R w_G}{U} B_{1i} \Psi(s) \qquad i=1,\ldots,n; \; \omega_1=0 \qquad (5.3)$$

with b, s, λ, Ω defined as before and

$$A_{ij} = (2\ b_R/S) \int_0^{\ell} a(y)^2\ \phi_i \phi_j\ dy$$

$$B_{ij} = (2\ b_R/S) \int_0^{\ell} a(y)\ \phi_i \phi_j\ dy$$

Rewriting (5.3) in matrix notation

$$\{\xi''(s)\} + \lceil \Omega^2 \rfloor \{\xi(s)\} + \frac{[A]}{\lambda} \{\xi''(s)\} + \frac{2[B]}{\lambda} \int_0^s \{\xi''(\sigma)\}\Phi(s-\sigma)d\sigma$$

$$= \frac{2\ b_R\ w_G}{\lambda\ U} \{\bar{B}\}\ \Psi(s)$$

where

$$\lceil \Omega^2 \rfloor = \frac{b_R^{\ 2}}{U^2} \lceil \omega^2 \rfloor$$

$$\{\bar{B}\} = \lfloor B_{1i} \rfloor^T$$

$$[A] = [A_{ij}]$$

$$[B] = [B_{ij}]$$

or

$$(\lceil I \rfloor + \frac{[A]}{\lambda})\ \{\xi''(s)\} + \lceil \Omega^2 \rfloor\ \{\xi(s)\}$$

$$= \frac{2\ b_R\ w_G}{\lambda\ U} \{\bar{B}\}\ \ \Psi(s) - \frac{2[B]}{\lambda} \int_0^s \{\xi''(\sigma)\}\ \Phi(s-\sigma)d\sigma \qquad (5.4)$$

Numerical Solutions

Numerical solution of equations (5.4) requires some modification to the system, which is implicit in ξ. Define I such that

$$\{I\} = \int_0^s \{\xi''(\sigma)\ \ \Phi(s-\sigma)d\sigma \qquad (5.5)$$

Hamming [12] has shown that this convolution integral has poor convergence properties unless handled carefully. The following procedure is related to Hamming's suggestion for separating out a portion of the integrand.

With a sufficiently small time interval Δs, the displacements can be assumed constant within each interval, and a trapezoidal integration rule can be used; thus

$$\{I\} = \{\xi''(s)\}\int_{(k+\frac{1}{2})\Delta s}^{(k+1)\Delta s} \Phi(s-\sigma)d\sigma + \sum_{i=1}^{k} \{\xi''(i\Delta s)\}\int_{(i-\frac{1}{2})\Delta s}^{(i+\frac{1}{2})\Delta s} \Phi(k\Delta s-\sigma)d\sigma \tag{5.6}$$

Equation (5.6) assumes that

$$s = (k+1)\Delta s \qquad \text{(current time)}$$

and

$$\{\xi''(0)\} = 0$$

The latter is an approximation to an initial steady-flight condition. Adopting a polynomial approximation to the Wagner function

$$\Phi(s) \cong \frac{s+2}{s+4} ,$$

it can be shown that

$$\int_{(i-\frac{1}{2})\Delta s}^{(i+\frac{1}{2})\Delta s} \Phi(k\ \Delta s-\sigma)d\sigma = 2\ln\left(\frac{(k-i-\frac{1}{2})\Delta s + 4}{(k-i+\frac{1}{2})\Delta s + 4}\right) + \Delta s \tag{5.7}$$

$$\int_{(k+\frac{1}{2})\Delta s}^{(k+1)\Delta s} \Phi(s-\sigma)d\sigma = 2\ \ln\left(\frac{4}{\frac{\Delta s}{2} + 4}\right) + \frac{\Delta s}{2} \tag{5.8}$$

Combining equations (5.6), (5.7) and (5.8),

$$\{I\} = \{\xi''((k+1)\Delta s)\}\left(2\ \ln\frac{4}{\frac{\Delta s}{2}+4}+\frac{\Delta s}{2}\right)+\sum_{i=1}^{k}\{\xi''(i\Delta s)\}\left(2\ \ln\left(\frac{(k-i-\frac{1}{2})\Delta s+4}{(k-i+\frac{1}{2})\Delta s+4}\right)+\Delta s\right) \tag{5.9}$$

Finally, combining equations (5.4), (5.5), (5.6) and (5.9),

$$\left([\ I\ ]+\frac{[A]}{\lambda}+\frac{2[B]}{\lambda}\left(2\ \ln\left(\frac{4}{\frac{\Delta s}{2}+4}\right)+\frac{\Delta s}{2}\right)\right)\{\xi''(k+1)\Delta s)\}+[\ \Omega^2\ ]\{\xi((k+1)\Delta s)\}$$

$$=\frac{2b_R w_G}{\lambda\ U}\{\bar{B}\}\Psi((k+1)\Delta s)-\frac{2[B]}{\lambda}\sum_{i=1}^{k}\{\xi''(i\Delta s)\}\left(2\ \ln\left(\frac{(k-i-\frac{1}{2})\Delta s+4}{(k-i+\frac{1}{2})\Delta s+4}\right)+\Delta s\right) \tag{5.10}$$

which is of the form

$$[M]\{\xi''\}+[K]\{\xi\}=\{R\} \tag{5.11}$$

Equations (5.11) are in a form suitable for direct integration by the Newmark method. The following algorithm may be used (Ref. [13]):

A. Initial Calculations

1. Initialize

$\{\xi\} = \{0\}$ at $s = 0$

$\{\xi'\} = \{0\}$ at $s = 0$

$\{\xi''\} = \{0\}$ at $s = 0$

2. Select 'time' step size Δs and parameters α and δ; calculate integration constants.

$\delta \geq 0.050;\ \alpha \geq 0.25\ (0.5+\delta)^2$

$$a_0 = \frac{1}{\alpha\Delta s^2};\quad a_1 = \frac{\delta}{\alpha\Delta s};\quad a_2 = \frac{1}{\alpha\Delta s}$$

$$a_3 = \frac{1}{2\alpha}-1;\quad a_4 = \frac{\delta}{\alpha}-1;\quad a_5 = \frac{\Delta s}{2}\left(\frac{\delta}{\alpha}-2\right)$$

$$a_6 = \Delta s(1-\delta);\quad a_7 = \delta\Delta s.$$

3. Form mass [M] and stiffness [K] as defined by equations (5.10) and (5.11). Form effective stiffness $[\hat{K}]$:

$$[\hat{K}] = [K] + a_0[M]$$

4. Triangularize $[\hat{K}]$:

$$[\hat{K}] = [L][D][L]^T.$$

B. Iterative Loop

1. Calculate effective loads at s + Δs:

$$\{R\}_{s+\Delta s} = \{R\}_{s+\Delta s} + [M](a_0\{\xi\} + a_2\{\xi'\} = a_3\{\xi''\})$$

2. Solve for displacements at s + Δs:

$$[L][D][L]^T\{\xi\}_{s+\Delta s} = \{\hat{R}\}_{s+\Delta s}$$

3. Calculate accelerations and velocities at s + Δs:

$$\{\xi''\}_{s+\Delta s} = a_0(\{\xi\}_{s+\Delta s} - \{\xi\}_s) - a_2\{\xi'\}_s - a_3\{\xi''\}_s$$

$$\{\xi'\}_{s+\Delta s} = \{\xi'\}_s + a_6\{\xi''\}_s + a_7\{\xi''\}_{s+\Delta s}$$

Results

Each of three aircraft types was studied at the medium weight to provide typical response values. In each case, the aircraft penetrated a two fps sharp-edged gust applied instantaneously along the entire wing. The results yield elastic displacements which are rather small, ranging from 3/4-inch wing tip displacement on the type B aircraft, to response as low as 1/8-inch wing tip displacement for the type A aircraft (swing-wing fighter-bomber). Figures 29-31 show the elastic response at the wing tip. The rigid body plunging mode is not included. The bulk of the motion in all cases was due to the first aircraft mode. The second mode participates in a minor way and the third and higher modes are scarcely excited. In addition to the graphical results in these figures, a wealth of tabular output is available for each aircraft.

5.3 Dynamic Response to Continuous Atmospheric Turbulence: Rigid Body Plunging

An airplane penetrating an atmospheric turbulence field experiences continuous rather than discrete gusts; hence a statistical approach is needed to model the continuous properties of atmospheric turbulence. The general method consists of applying a random input (the power spectrum of atmospheric turbulence) to a linear system (the mechanics of the rigid body plunging mode) and studying the response of this system. Only statistical properties of the response, e.g., root-mean-square displacements, may be determined by this method; explicit time-histories will not be known.

The aircraft is modeled by a rigid wing of constant chord 2b flying at a forward velocity U. The wing thickness and the magnitude of the vertical translations are assumed small compared to the chord. The fluctuating turbulence velocities u,v,w are assumed small compared to U. The components u,v may be neglected as the wing is free in vertical translation only. Thus, the wing is subjected to a fluctuating angle of attack

$$\alpha = \frac{w}{U} .$$

It is further assumed that the gust field is two-dimensional (no spanwise variations) and the turbulence pattern does not change during the time required for one particle of air to traverse the wing, i.e., during time 2b/U.

Input

The turbulence w is assumed to be a stationary random function given by the von Karman power spectrum, $\phi(\Omega)$:

$$\phi(\Omega) = \frac{L}{\pi} \sigma_w^2 \frac{1 + \frac{8}{3}(1.339 \text{ L } \Omega)^2}{[1 + (1.339 \text{ L } \Omega)^2]^{11/6}}$$

where:

Ω = space frequency (rad/ft)

L = scale of turbulence (ft)

σ_w = RMS gust velocity (ft/sec)

The function $\phi(w)$ satisfies the relation:

$$\sigma_w^2 = \int_o^\infty \phi(w)dw.$$

Rigid-Body Admittance

To calculate the mechanical admittance of the system, the wing is subjected to a sinusoidal gust described by the real part of:

$$w_G = \bar{w}_G \, e^{ik(s-x^*)}$$

where k is the nondimensional frequency:

$$k \equiv \frac{wb}{U} = \Omega b.$$

The response of the wing is given by the solution of the equation of motion:

$$\frac{U^2}{b^2} M \, \xi''(s) = \int_{span} L_G \, dy + \int_{span} L_M \, dy$$

where:

M = airplane mass

ξ = normal coordinate

s = nondimensional time, $s = \frac{Ut}{b}$

L_G = lift due to gust

L_M = lift due to the motion

Substituting as a solution:

$$\xi = \bar{\xi} \, e^{iks}$$

and writing the expressions for L_G and L_M, we have:

$$\frac{\bar{\xi}}{\bar{w}_G} = \frac{b}{U} \frac{2K(k)}{k[2i\ C(k) - (2\lambda+1)k]}$$

where:

$\lambda = M/\pi\ \rho\ Sb$

ρ = density of the air (slugs/ft^3)

S = wing area (ft^2)

$C(k)$ = Theodorsen Function

$K(k) = C(k)[J_o(k) - i\ J_1(k)] + i\ J_1(k)$

The expression for $\frac{\bar{\xi}}{\bar{w}_G}$ is the admittance function with respect to vertical displacement.

Output

Let $\psi(w)$ be the power spectrum of the airplane response. Then

$$\overline{\xi^2} = \int_0^\infty \psi(w)dw$$

where $\sqrt{\overline{\xi^2}}$ is the root-mean-square displacement.

The following relation then holds:

$$\psi(w) = \left|\frac{\bar{\xi}}{\bar{w}_G}\right|^2 \phi(w)$$

Using polynomial approximations to the complex terms in the admittance expression, the power spectrum of the response becomes:

$$\psi(w) = \frac{b^2}{U^2} \frac{4|K(k)|^2}{k^2|2i\ C(k) - (2\lambda+1)k|^2} \phi(w)$$

$$= \frac{b^2}{U^2} \frac{4}{1+2\pi k} \frac{1}{k^2} \frac{1}{4+k^2(2\lambda+1)^2} \phi(w)$$

or

$$\psi(k) = \frac{b^2}{U^2} \frac{4}{1+2\pi k} \frac{1}{k^2} \frac{\frac{L}{\pi}\sigma_w^{\ 2}}{4+k^2(2\lambda+1)^2} \frac{1+\frac{8}{3}(1.339\ \frac{Lk}{b})^2}{[1+(1.339\ \frac{Lk}{b})^2]^{11/6}}$$

The mean-square displacement becomes:

$$\overline{\xi^2} = \int_0^\infty \frac{b}{U^2} \frac{4}{1+2\pi k} \frac{1}{k^2} \frac{\frac{L}{\pi}\sigma_w^{\ 2}}{4+k^2(2\lambda+1)^2} \frac{1+\frac{8}{3}(1.339\ \frac{Lk}{b})^2}{[1+(1.339\ \frac{Lk}{b})^2]^{11/6}}\, dk$$

$$= \frac{4Lb\sigma_w^{\ 2}}{\pi U^2} \int_0^\infty \frac{4}{1+2\pi k} \frac{1}{k^2} \frac{1}{4+k^2(2\lambda+1)^2} \frac{1+\frac{8}{3}(1.339\ \frac{Lk}{b})^2}{[1+(1.339\ \frac{Lk}{b})^2]^{11/6}}\, dk$$

Frequency Limits

The integrand in the above expression is singular at $k=0$; hence, a lower frequency limit (ε) on the power spectrum of the response is needed. This corresponds to an artificial high-pass filter. Since radar returns are garbled at very low frequencies (less than 5 Hz), this filtering is acceptable.

The mean-square displacement then becomes

$$\overline{\xi^2} = \frac{4Lb}{\pi} \left(\frac{\sigma_w}{U}\right)^2 \int_0^\infty \frac{4}{1+2\pi k} \frac{1}{k^2} \frac{1}{4+k^2(2\lambda+1)^2} \frac{1+\frac{8}{3}(1.339\ \frac{Lk}{b})^2}{[1+(1.339\ \frac{Lk}{b})^2]^{11/6}}\, dk$$

In practice, integration to a finite upper frequency limit suffices as the response becomes vanishingly small at the higher frequencies.

Results

The rigid body plunging responses of the three aircraft types (A,B,C: medium weight) were computed with the low frequency cutoff point as the parameter. The upper frequency cutoff was set at 100 Hz. Results are shown in Figure 32 for a root-mean-square gust velocity of 2 ft/sec. The mean-square displacement of the airplane as a whole is not large, unless one is willing to attempt to detect frequencies below 1 hertz, say.

5.4 Interpretation of the Aircraft Motion for RCS Work to Follow

The elastic motion of an aircraft in flight has been studied in a deterministic as well as a random approach. There is a question as to which way provides more information for identification.. The random approach of section 5.3 is not encouraging because of the small rigid body displacements which were found - approximately 0.01 inch r.m.s. when signals above 2 hz only are processed.* The deterministic approach for the sharp-edged gust does not give much more hope. Peak elastic response at the wingtip was only 0.12" for a type A aircraft, 0.75" for a type B, and 0.33" for a type C, all at medium weight. The dominant elastic motion in response to the sharp-edged gust is in the fundamental mode (corresponding to the lowest frequency).

A deterministic approach will be used in the following RCS work. The response of aircraft type A with wings fully forward will be the test case. The vertical motion of the aircraft will be assumed harmonic and consisting solely of the first elastic mode. The half-amplitude of motion of the wing tip will be taken as 0.3 cm (0.12"). This number is as large as can be reasonably inferred from Figure 29 and achieving it would require successive upward and downward gusts. The RCS calculations therefore study a motion which is an upper bound to what one could observe in typical turbulence cells occupying 5% of the earth's surface. Finally, it is noted that the type A aircraft has a relatively large fuselage displacement in the first mode as can be seen by comparing Figures 24a-c.

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* Even if this value of 0.01 inch rms were doubled or tripled at the wingtip due to inclusion of elastic effects, it would still be small.

VI. RADAR CROSS-SECTION (RCS) STUDIES

Structural motion and aeroelastic vibration of aircraft subject to gusts and turbulence have been discussed in earlier chapters. The present chapter theoretically studies the effects, if any, of these aerodynamically induced motion of the aircraft on its RCS.

6.1 General Considerations

Theoretical determination of the RCS of a complex body (with regard to its electromagnetic scattering) such as an aircraft is an extremely difficult boundary-value problem in electromagnetics. A plot of the RCS versus time for an aircraft in flight often appears as a noise-like fluctuation even when the nominal aspect of the aircraft is constant. Because of the uncertainties about the aspect (due to roll, pitch, etc.), the RCS is often best described statistically in terms of various distribution functions [15,16]. In general, however, these distributions point out that there is no simple solution to the RCS problem for all aircraft [17]. To avoid unnecessary complications we will avoid the statistical approach. Instead, we confine our attention to a deterministic study of the effects on the ambient RCS of an aircraft produced by some of its identifiable motion induced by air-turbulence, etc. The next section describes the simplified electromagnetic model of an aircraft, and how the static and dynamic cross sections are obtained.

6.2 Scattering Model

The fundamental assumption of the theoretical method for obtaining the RCS of an aircraft is that electromagnetic scattering by the aircraft may be assumed to be that due to a collection of independent scatterers which may be identified with the various components of the aircraft (e.g., fuselage, wing, etc.) [18]. Usually, this is possible at sufficiently high radar frequencies where the appropriate dimensions of the aircraft are large compared to the radar wavelength. To this end, the aircraft is considered to be an ensemble of components, each of which can be geometrically approximated by a simple shape in such a way that the RCS of the simple shape approximates the RCS of the component it models. Once the component scatterers are identified, their cross sections are obtained from known results. Each of the components is then replaced by a point scatterer located at its scattering (or phase) center, and having a scattering area equivalent to that of the component. Finally, the component cross sections are combined appropriately to estimate the RCS of the entire aircraft. This method has been found useful in the theoretical estimation of RCS of aircraft, missiles and the results have been found to be in fair agreement with measured values [19,20,21].

Let us assume that for a given combination of aircraft aspect angle, wavelength and polarization of the radar, N scattering components have been identified for which the radar cross sections are $\sigma_1, \sigma_2, \ldots, \sigma_N$. One of the methods of combination of these cross sections involves the relative phase angles between the scattered fields from the N scatterers. This leads to the following expression, denoted by σ_p (cross section by relative phase), for the RCS of the entire aircraft:

$$\sigma_p = \left| \sum_{j=1}^{N} (\sigma_j)^{1/2} \exp(i\beta_j) \right|^2, \tag{6.1}$$

where σ_j is the crosssection of the j-th component and β_j is the relative phase angle associated with the radar return from the j-th component. The magnitudes of β_j's are determined by selecting a reference point (or origin) on the aircraft and obtaining the phase angle of the return from each component from its distance from the origin. For this purpose consider a rectangular coordinate system (x,y,z) with origin at 0 which also serves as the origin of a spherical polar coordinate system (r,θ,ϕ) with its polar axis oriented along the z-direction. Let the aircraft be oriented horizontally (in the y-z plane) with its nose aligned along the z-axis and its center of the fuselage located at the origin 0, as shown in Figure 33. Let the coordinates of the j-th scattering center be (x_j, y_j, z_j). Under these assumptions it can be shown that in the radar direction (θ_o, ϕ_o) the phase angle β_j appropriate for the j-th scatterer is given by:

$$\beta_j = 2k[x_j \sin\theta_o \cos\phi_o + y_j \sin\theta_o \sin\phi_o + z_j \cos\theta_o] \qquad (6.2)$$

where $k = \frac{2\pi}{\lambda}$, λ being the wavelength of the radar waves. Observe that for $\phi_o = 0$, i.e., in the x-z plane of Figure 35, Equation (6.2) reduces to

$$\beta_j = 2k[x_j \sin\theta_o + z_j \cos\theta_o] \qquad (6.3)$$

which indicates that β_j is independent of the y-coordinate of the scattering center. Also, note that in the ideal case when all the returns combine in phase one obtains the maximum RCS of the aircraft as:

$$\sigma_{p_{max}} = \left| \sum_{j=1}^{N} (\sigma_j)^{1/2} \right|^2 \qquad (6.4)$$

It is evident that in the above approach one must know the distances of the scattering centers from the chosen origin. These can be estimated either from the aircraft drawings or from their scale models. However, Equation (6.2) or (6.3) indicates that the phase angle β_j depends directly upon the ratios x_j/λ, etc. Therefore, for a large aircraft at small wavelengths it may be impossible to obtain these distances from the drawings or models with sufficient accuracy [18, 19].

As an alternative to the relative phase method, there exists another method often referred to as the random phase method [19]. This method is based upon the assumption that many different β_j's are randomly distributed, then upon averaging over β_j we obtain the average RCS, denoted by σ', as

$$\sigma' = \sum_{j=1}^{N} \sigma_j \tag{6.5}$$

The deviations of the observed RCS from the average crosssection σ' are characterized by employing the concept of RMS spread, denoted by S. This measure of the probable variations in cross section due to the relative phase effects leads to the bounds ($\sigma' \pm S$) for the observed total RCS where

$$S^2 = \left(\sum_{j=1}^{N} \sigma_j\right)^2 - \sum_{j=1}^{N} \sigma_j^{\,2}. \tag{6.6}$$

It is evident from the above discussion that the random phase method gives estimates of the amount by which the cross section might deviate from the average value because of the phase effects. On the other hand, the relative phase method of combination not only estimates the amount by which the cross-section deviates from the average value but also the locations (in aspect or wavelength) of the relative peaks and nulls in the RCS.

So far, it has been assumed that the aircraft is static and hence, the RCS values obtained from the above expressions will be independent of time and, thus, will be referred to as the static RCS. In the next section we describe the method of obtaining the RCS of an aircraft undergoing vibratory motion induced by air turbulence.

6.3 Dynamic RCS

In earlier chapters we have obtained the frequencies and modes of vibration of the aircraft caused by air turbulence. It was found that the vertical displacement of the aircraft was significant; the mode shape and frequencies of these displacements were obtained numerically. We shall assume that the scattering centers of the aircraft experience similar kinds of vertical motion in time. Thus, for any mode of vibration, the x-coordinate of the scattering center will vary in time according to that mode shape and at a frequency corresponding to that modal frequency. It should be noted that the modal shapes obtained from free vibration considerations (see Chapter III) must be scaled properly to obtain the scattering center displacements.

In accordance with our earlier notation, let the x-coordinate of the j-th scattering center (associated with the wing) undergoing the i-th mode of vibration will be denoted by

$$x_j = \eta_j \cos(\omega_i t + \alpha_j), \tag{6.7}$$

where,

$$\eta_j = [f_1(y_j)^2 + f_2(y_j)^2]^{1/2} \tag{6.8}$$

$$\tan \alpha_j = \frac{f_2(y_j)}{f_1(y_j)} \; . \tag{6.9}$$

The quantities $f_1(y_j)$, $f_2(y_j)$ and α_j identify the shape of the induced motion of the scattering center caused by the i-th mode of radian frequency ω_i. Vertical motions of the scattering centers associated with the fuselage and other components are included in a similar manner.

$$\sigma_p(\theta_o,\phi_o, t) = \left| \sum_{j=1}^{N} (\sigma_j)^{1/2} \exp(i\beta_j) \right|^2, \tag{6.10}$$

with

$$\beta_j = 2k[\eta_j \sin\theta_o \cos\phi_o \cos(\omega_i t + \alpha_j) + y_j \sin\theta_o \sin\phi_o$$

$$+ z_j \cos\theta_o], \tag{6.11}$$

where the dependence of time and radar direction are shown explicitly in σ_p. If the radar is located in the vertical x-z plane, $\phi_o = 0$, the RCS expression reduces to:

$$\sigma_p(\theta_o, t) = \left| \sum_{j=1}^{N} (\sigma_j)^{1/2} \exp i\, 2k[\eta_j \sin\theta_o \cos(\omega_i t + \alpha_j) + z_j \cos\theta_o] \right|^2 \tag{6.12}$$

Note that with this model, the nose-on (θ_o=0) RCS of the aircraft is unaffected by the vibration.

6.4 Scattering Model of the Aircraft

Aerodynamic studies, discussed in earlier chapters, indicated that the vibration mode of type A aircraft is reasonably invariant to airspeed, fuel load and wing sweep. For this reason we have chosen to study the RCS of the aircraft model F-111 which belongs to type A. The component scatters and the location of the corresponding scattering centers are obtained by studying a 1/72-scale model of the aircraft. The orientation of the aircraft is as shown in Figure 33 and it will be assumed that the radar is located in the x-z plane. Dominant scattering components are identified from a study of the scale model. The approximate geometrical shapes and the corresponding theoretical expressions for their cross sections are as follows [18-21]:

(i) The nose of the aircraft is approximated by a section of a conducting paraboloid. This component will contribute (σ_1) in the range $0 \leq \theta \leq 74°$. It's contribution is obtained from:

$$\sigma_1(\theta_o) = \frac{\pi}{4} \sec^4 \theta_o. \tag{6.13}$$

(ii) The main body of the fuselage is approximated by a conductive circular cylinder of length L = 19.72 m and radius a = 1.08 m. This will contribute in the range $74° \leq \theta_o \leq 140°$. It's contribution ($\sigma_2$) is determined by using the following expressions:

$$\sigma_2(\theta_o) = \frac{\lambda\ a \sin\theta_o}{8\pi^2 \cos^2\theta_o}, \quad 74° \leq \theta_o \leq 85° \tag{6.14a}$$

$$\sigma_2(\pi/2) = \frac{2\pi\ L^2\ a}{\lambda}, \quad \text{for } 95° \leq \theta_o \leq 140° \tag{6.14b}$$

(iii) Each of the two wings is approximated by a conducting rectangular plate oriented in the y-z plane. Each plate has dimensions W and L in the y and z directions, respectively. The combined contribution (σ_3) from the two wings is obtained from:

$$\sigma_3(\theta_o) = 2\ \frac{4\pi\ W^2 L^2}{\lambda^2} \sin^2\theta_o \frac{\sin^2(kL\cos\theta_o)}{(kL\cos\theta_o)^2} \tag{6.15}$$

with L = 3.6 m and W = 7.92 m.

(iv) Each of the two tail fins is approximated by a rectangular metal plate oriented in the y-z plane. This combined contribution (σ_4) is obtained from (6.15) with L = 4.32m and W = 1.44m.

(v) The two engine ducts in the front are approximated by circular cavities each having a diameter a = 1.5 m. The scale model indicated that the opening of each duct is one-fourth of the complete circular area, πa^2. The contributions (σ_5) of the two engine ducts is obtained from:

$$\sigma_5(\theta_o) = 0.05(ka)^3\ 2\ \frac{\sin^2(2ka\sin\theta_o)}{(2ka\sin\theta_o)^2} \tag{6.16}$$

$$\text{for } 0 \leq \theta_o \leq 84°.$$

(vi) The two exhaust ducts, located in the rear of the aircraft, are modelled by circular cavities each having a radius a = 0.5 m. Their combined contribution (σ_6) is obtained from:

$$\sigma_6(\theta_o) = 2 \times 0.4(ka)^3\lambda^2 \frac{\sin^2(2ka \sin \theta_o)}{(2ka \sin \theta_o)^2} \qquad (6.17)$$

$$\text{for } 135° \leq \theta_o \leq 180°$$

Note that all linear dimensions are expressed in meters and the calculated cross sections are obtained in square meters. Also note that shapes and dimensions of the scattering components are assumed such that their scattering cross-sections are polarization independent.

The z-coordinates of the scattering centers of the above components are: z_1 = 9.36 m, z_2 = 0.0, z_3 = - 1.08 m, z_4 = -7.2 m, z_5 = + 1.08 m and z_6 = - 9.36 m.

Even after identification of the various scattering components, proper care must be taken to combine them for a given aspect angle. This is because the geometry may be such that at some aspect angle parts or all of the scattering from a component may be shadowed by other(s). To avoid the complications due to shadowing effects, we have restricted ourselves to the determination of RCS in the x-z plane and proper considerations have been given in obtaining the cross section expressions given above.

6.5 Numerical Results

Figures 34(a) and 34(b) show the static average, RMS and relative phase crosssections, σ', $\sigma' \pm s$ and σ_p, respectively, versus aspect angle θ_o for the aircraft obtained at λ = 3 and 30 cm. Of course the average cross section σ' stays within the RMS bounds $\sigma' \pm s$ at all aspect angles. Over most of the range σ_p also stays within the RMS bounds; at some aspect angles, σ_p moves out of the RMS bounds. These results are in general agreement with results discussed elsewhere [18, 19] .

The dynamic cross sections are obtained for the fundamental vibration mode at $w_i/2\pi$ = 6 hz and a wing tip half-deflection η = 0.3 cm.

Two sets of dynamic RCS have been obtained: one referred to as the tip-scattering center assumes that the scattering center of the wing is located at its tip which would experience the maximum displacement due to the induced vibration; the second set of results, referred to as the mid-wing scattering center, assumes that the scattering center of the wing is located at its center. Figures 35(a-e) and 36(a-e) show the dynamic RCS as a function of time and for selected values of the aspect angle θ_o. Observe that the results are shown over a complete cycle of vibrations at the dominant mode. Notice that at some aspect angles, the RCS values at $\lambda = 3$ cm are lower than those at $\lambda = 30$ cm; this is due to the fact that those aspect angles are located at or near the nulls in the RCS pattern at $\lambda = 3$ cm. Generally, the dynamic results in Figures 35 and 36 indicate that the aircraft vibration induces some kind of fluctuation (or modulation) in its RCS. At most of the aspect angles, the total deviations in the dynamic RCS values from the corresponding static values appear to be more at $\lambda = 3$ cm than those at $\lambda = 30$ cm. The fluctuations appear to be large at selected aspect angles. As a function of time, the modulation in the RCS for $\lambda = 30$ cm appears to occur at the frequency of vibration of the aircraft. For $\lambda = 3$ cm, the modulation also contains a component of the vibrating frequency at all aspect angles; in addition, at longer aspect angles ($\theta_o = 35°$, and $45°$) there exist quite strong third harmonic components. The location of the scattering center of the wing at its center or tip does not appear to change the general nature of the results.

Observe from Figure 35(e) that at $\theta_o = 45°$, as a function of time, the total deviation in the dynamic RCS, at $\lambda = 3$ cm is about 8 dB. However, the static results shown in Figure 34(a) indicate that near $\theta_o = 45°$, the RCS varies quite strongly with θ_o. If it is assumed that aspect angle varies about

$\pm$ 2° due to reasons other than the induced aircraft vibration, then careful study of Figure 34(a) near $\theta_o \simeq 45°$ indicates that this may cause about 1 to 5 dB variations. This implies that under such conditions vibration-induced modulation would produce about 3 to 7 dB deviations in the dynamic RCS. Perhaps it should be mentioned that low frequency variations in the observed dynamic RCS of aircraft have been reported in [20,22]. Our results here tend to indicate that these may occur as low frequency variations in the RCS of the aircraft due to its vibration induced by air turbulence.

In our scattering model we have neglected the effects of shadowing by the individual scattering components and of the incident polarization. The accuracy of the assumed model is satisfactory for the static case [18] and should be acceptable for rough estimation of the general effects in the dynamic case. Further study is required to obtain the detailed nature of the effects of aircraft vibration on its RCS.

VII. CONCLUSIONS AND RECOMMENDATIONS FOR FURTHER STUDY

7.1 Conclusions

Three fighter aircraft have been analyzed for aeroelastic response to gusts and turbulence. The study included the effect of operating conditions on the modal frequencies and shapes, as well as determination of relative amplitude response of elastic modes.

It was found that airspeed had a moderate effect on frequencies and modes for all three aircraft. Fuel and armament loads had a large effect, particularly when carried on the wings (as opposed to fuselage). The one aircraft with a swing-wing had dominant fuselage bending at the lower frequencies; these modes were changed only moderately by the wing position.

Perhaps a more critical issue than the invariance of the modes is whether the modes are excited sufficiently by gusts and turbulence to allow observation. Only five percent of the atmosphere contains turbulent patches with a gust greater than 2 ft/sec. recorded at least every ten seconds. A five-mode simulation of the symmetric wing bending problem was carried out for each aircraft, using a 2 ft/sec. sharp-edged gust. Each of the aircraft responded with a total elastic wing tip deflection of 3/4-inch or less. The bulk of this response was in the first mode, and response in higher modes was small.

The only situation which has any promise for identification is the fundamental (fuselage bending) mode for the Type A fighter/bomber. The frequency of this case can be mimicked by smaller fighters carrying sufficient stores. Therefore, for success, some unique characteristic of the mode shape, such as the large tail motion, needs to be exploited.

After identifying its dominant scattering components, a theoretical scattering center model has been obtained to calculate the RCS of a type A aircraft. Average RCS and relative phase RCS of the static aircraft have been determined as functions of the radar aspect angle, and for λ = 3 and 30 cm. It has been assumed that the radar is ground-based, and the RCS calculations have been performed in a vertical plane such that the shadowing effects on RCS are minimum. Dynamic RCS of the aircraft has been obtained by assuming that the appropriate scattering centers experience vertical displacements (in time) produced by the motion of the aircraft undergoing its fundamental mode of vibration induced by air turbulence. At some selected values of aspect angles, we have determined the dynamic RCS over a complete time period of the fundamental mode.

With the assumed scattering model, the dynamic RCS of the aircraft in the nose-on direction appears to be independent of the aircraft vibration in the vertical plane. In other directions (aspect angles), the RCS values appear to contain amplitude modulations at the fundamental and the third harmonic of the frequency of the fundamental mode of vibration of the aircraft. Although these modulations are generally found for both λ = 30 and 3 cm, those of the latter wavelength appear significant enough to be observable.

The significant finding of the study is that the motion of the aircraft induced by air turbulence seems to produce low frequency amplitude modulation of its ambient RCS. From the considerations of the maximum vertical displacements due to turbulence suffered by the wings (or wing-tips) of a Type A aircraft, it appears that such modulations may be observable with a 3 cm ground-based radar system. At the completion of the present study, it is not clear whether such observations could be used to identify the aircraft. Further investigation is needed for this purpose.

7.2 Recommendations

The present investigation should be considered as a preliminary study of the general problem of identifying an aircraft by its RCS modulations induced by airframe vibration. Although some of the results of the present study are found to be significant from this point of view, they are not complete and well understood. Therefore, to ascertain the potentialities and practical realizability of this method of aircraft identification, the following studies are recommended.

(i) Obtain the RCS vs. time for a given aircraft in flight.

(ii) Obtain experimentally the RCS modulations for a model aircraft undergoing a motion simulating that of the fundamental mode of vibration.

(iii) Investigate the implications of the results obtained in (i) and (ii) with regard to the identification of the aircraft.

7.3 Unresolved Points

Some unresolved technical points include:

(i) Should the identification be based primarily on random or deterministic concepts? In the present study, deterministic ideas have dominated.

(ii) If a random approach is taken, are the newer, non-Gaussian turbulence models [23] more appropriate than the von Karman isotropic turbulence model? Although more accurate, the newer theory will probably not be worth the computational effort.

(iii) How much effort does the longitudinal rigid body pitching mode have? The so-called short longitudinal mode has frequencies of the order of one hertz and is felt not to couple into the problem.

(iv) Will the active control systems of the future couple with gust and turbulence response? Both the Rockwell B-1 and the latest versions of the Lockheed 1011 have active systems which suppress elastic modes but may introduce frequencies peculiar to the control system.

7.4 Acknowledgements

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IX. TABLES

Table 1

Aircraft Weights

The airplane cases studied follow. In all cases, additional loading is simulated by inertial effects alone.

Type A: 2 different wing-sweep configurations, 3 internally-loaded cases

A) 16 deg. l.e. sweep

1--1: dry airplane ('light')
gross weight = 44,507 lb.
1--2: dry wing, 23,327 lb. fuselage fuel ('medium')
gross weight = 71,834 lb.
1--3: 5,468 lb. wing fuel, 23,327 lb. fuselage fuel ('heavy')
gross weight = 77,302 lb.

B) 72.5 deg. l.e. sweep

1--4: dry airplane ('light')
gross weight = 44,507 lb.
1--5: dry wing, 23,327 lb. fuselage fuel ('medium')
gross weight = 71,834 lb.
1--6: 5,468 lb. wing fuel, 27,327 lb. fuselage fuel ('heavy')
gross weight: 77,302 lb.

Type B: 3 externally-loaded cases

4--1: clean airplane, unloaded pylons ('light')
gross weight = 37,704 lb.
4--2: 2 loaded pylons per semi-span ('medium')
total of 4 loaded I/B pylons per airplane (14 M117GP bombs)
gross weight = 49,068 lb.
4--3: loaded pylons per semi-span ('heavy')
total CF 4 loaded I/B and 4 loaded O/B pylons per airplane
(I/B: 14 M117GP bombs) plus
(O/B: 2 MK81 bombs and 2 LAU 32 A/A FFAR rocket
gross weight = 51,752 lb.

Type C: 3 externally-loaded cases

5--1: clean airplane, all stations unloaded ('light')
gross weight = 15,265 lb.
5--2: I/B and tip stations loaded ('medium')
I/B pylon: 50% 150 G. fuel tank
Tip station: AIM-98 'sidewinder' missile & launch
gross weight = 17,258 lb.
5--3: O/B ANC tip stations loaded ('heavy')
I/B pylon: full 275 G. fuel tank
O/B pylon: BLU-27/B(F)
tip station: AIM-98 'sidewinder' missile & launch
gross weight = 21,908 lb.

X. FIGURES

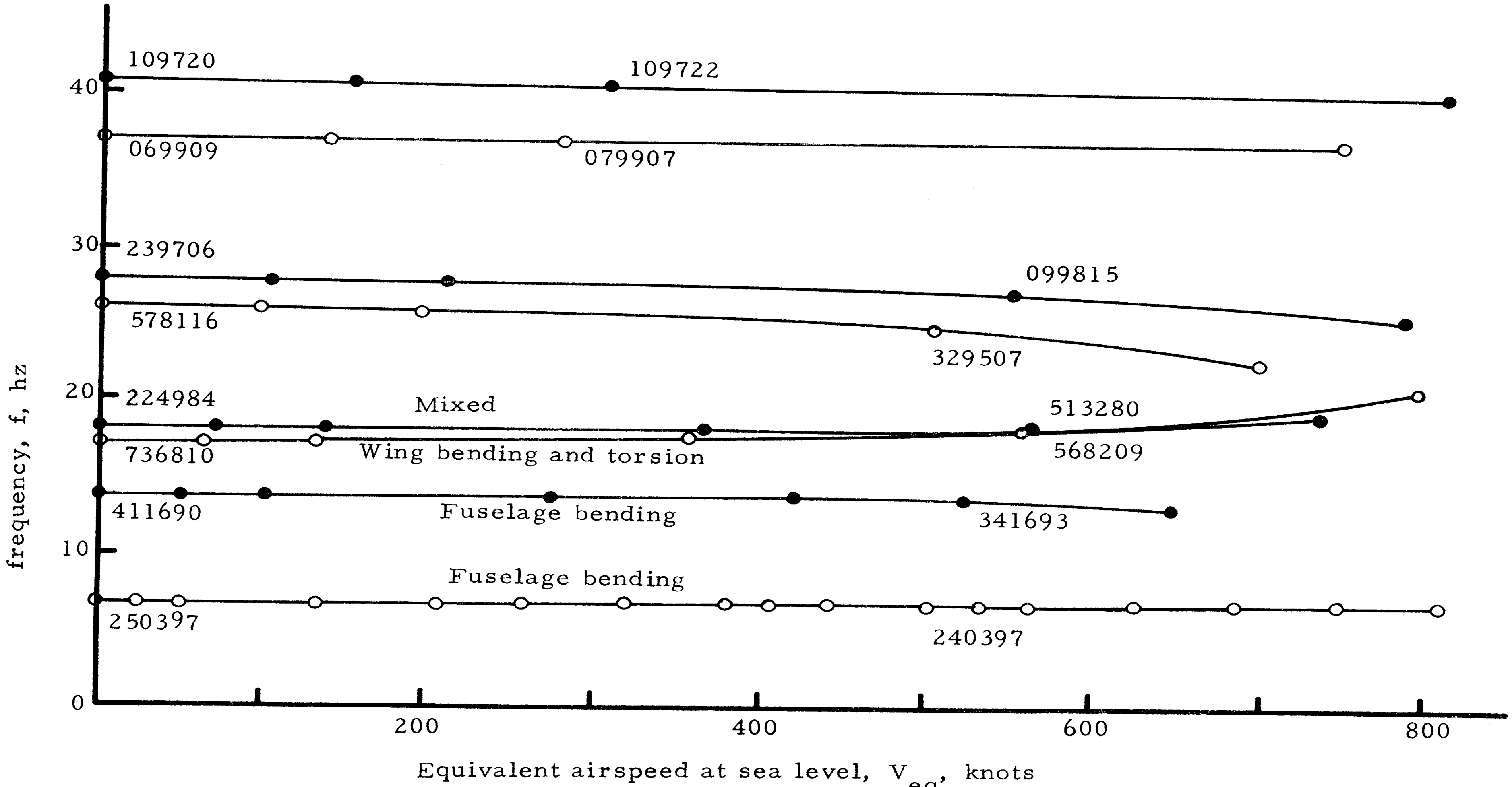


Fig. 1. Modal frequencies of aircraft type A in flight. Weight = 44, 507 lb. (Dry airplane, 16° l. e. sweep, symmetric modes.)

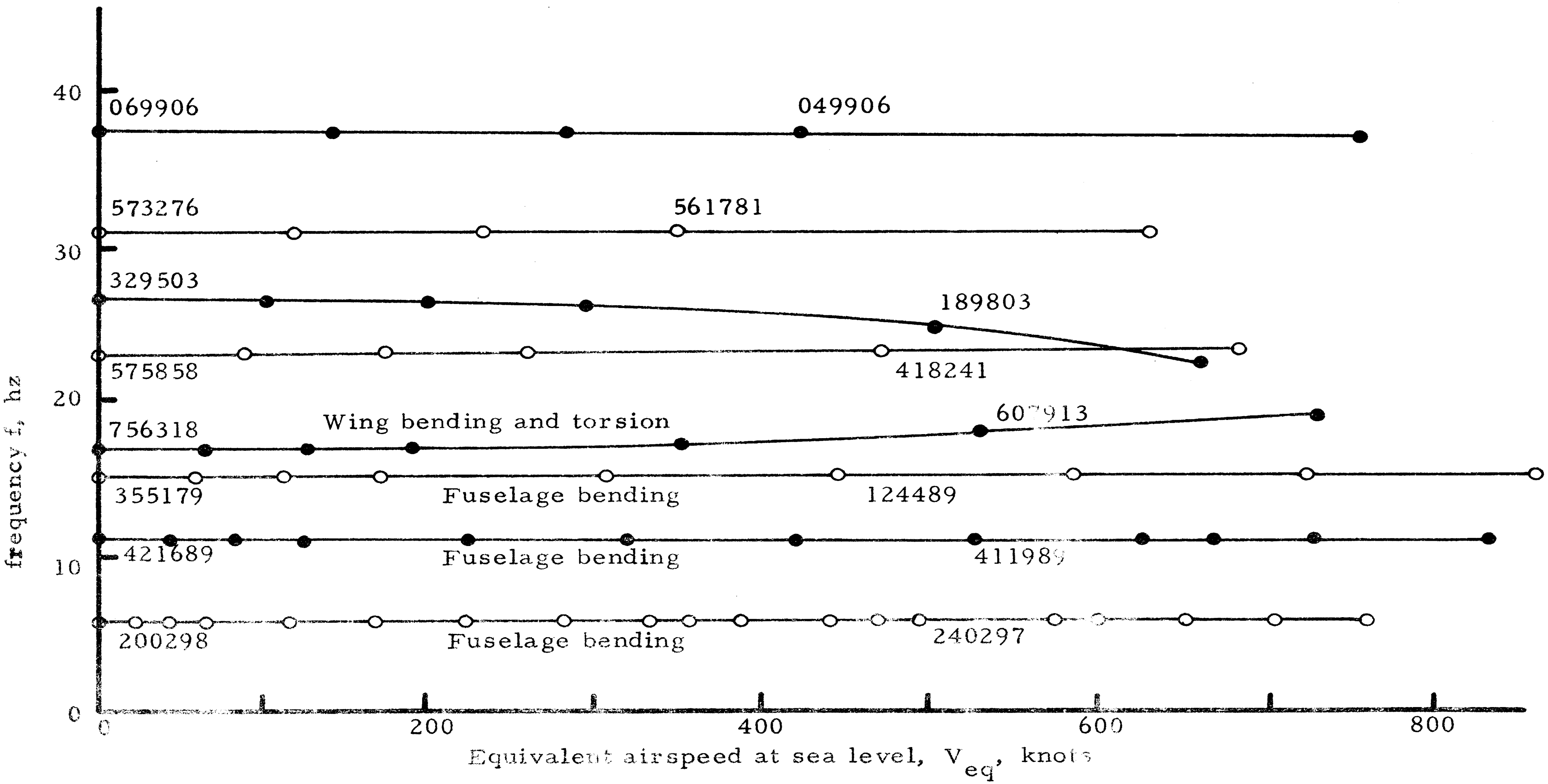


Fig. 2. Modal frequencies of aircraft type A in flight. Weight = 71,834 lb. (Dry wing, full fuselage fuel = 23,327 lb, 16° l.e. sweep, symmetrical modes.)

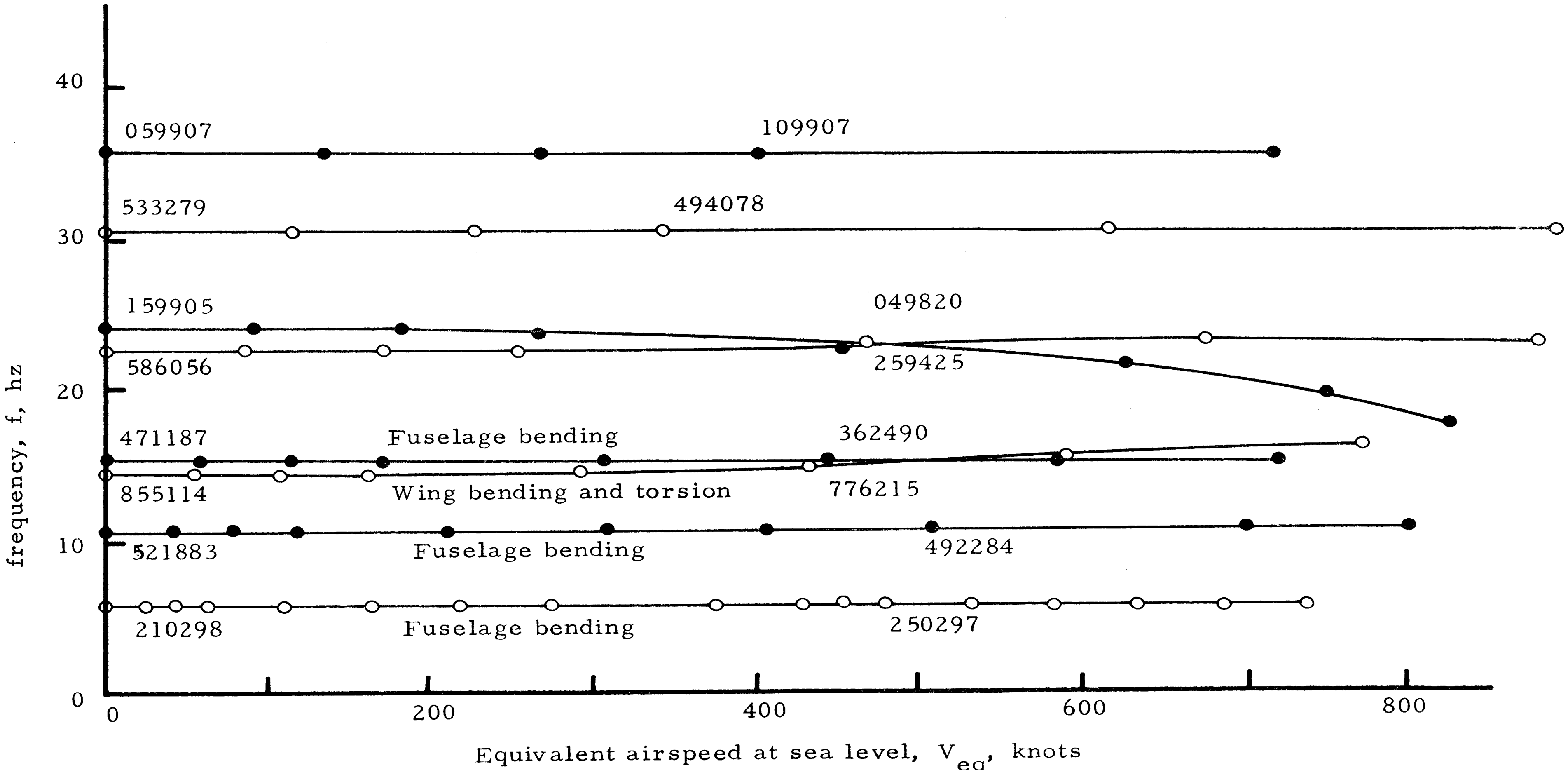


Fig. 3. Modal frequencies of aircraft type A in flight. Weight = 77, 302 lb. (Full fuel load, 16^{o} l. e. sweep, symmetric modes.)

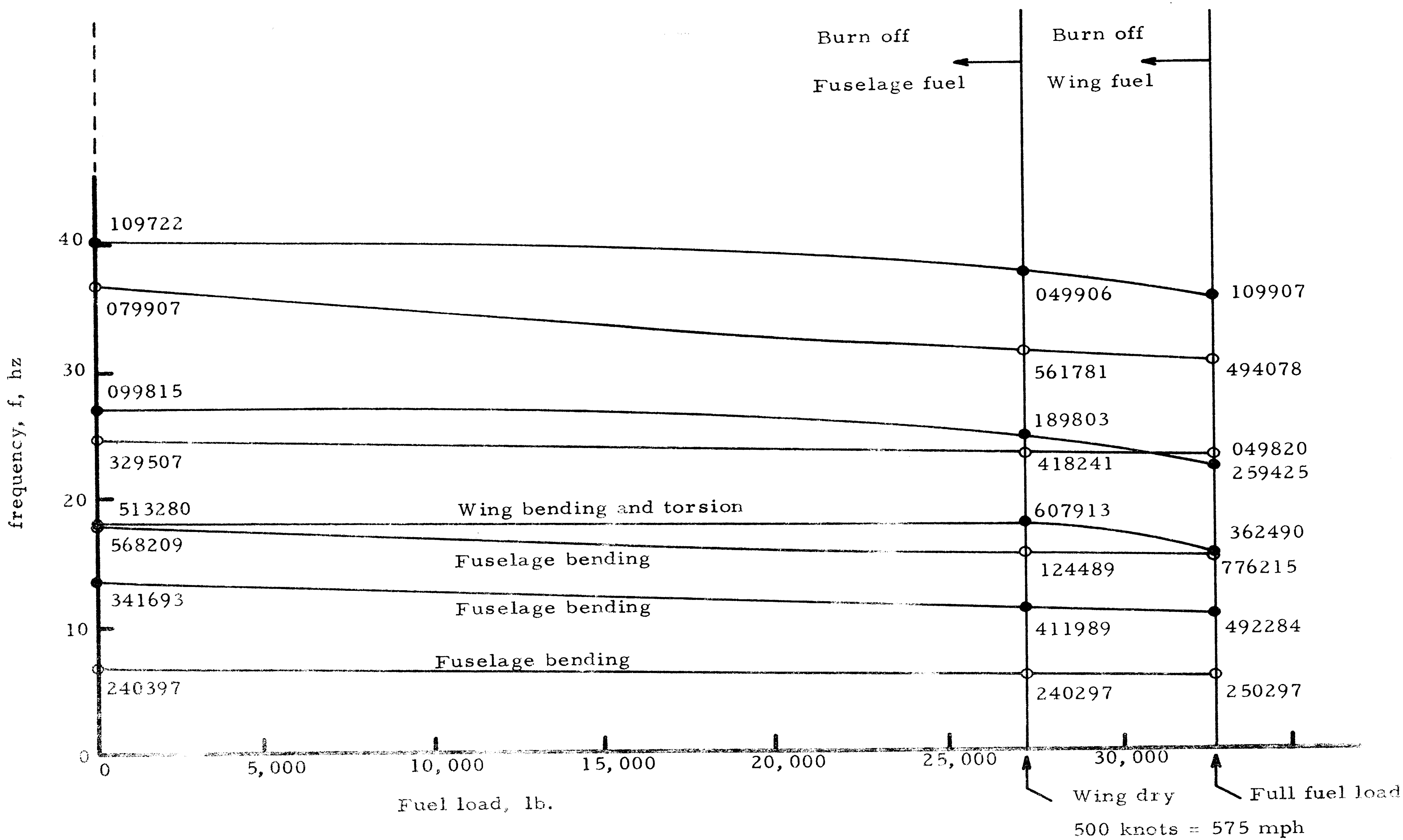


Fig. 4. Modal frequencies with varying fuel load for aircraft type A at 500 knots. (16° l.e. sweep, symmetric modes.)

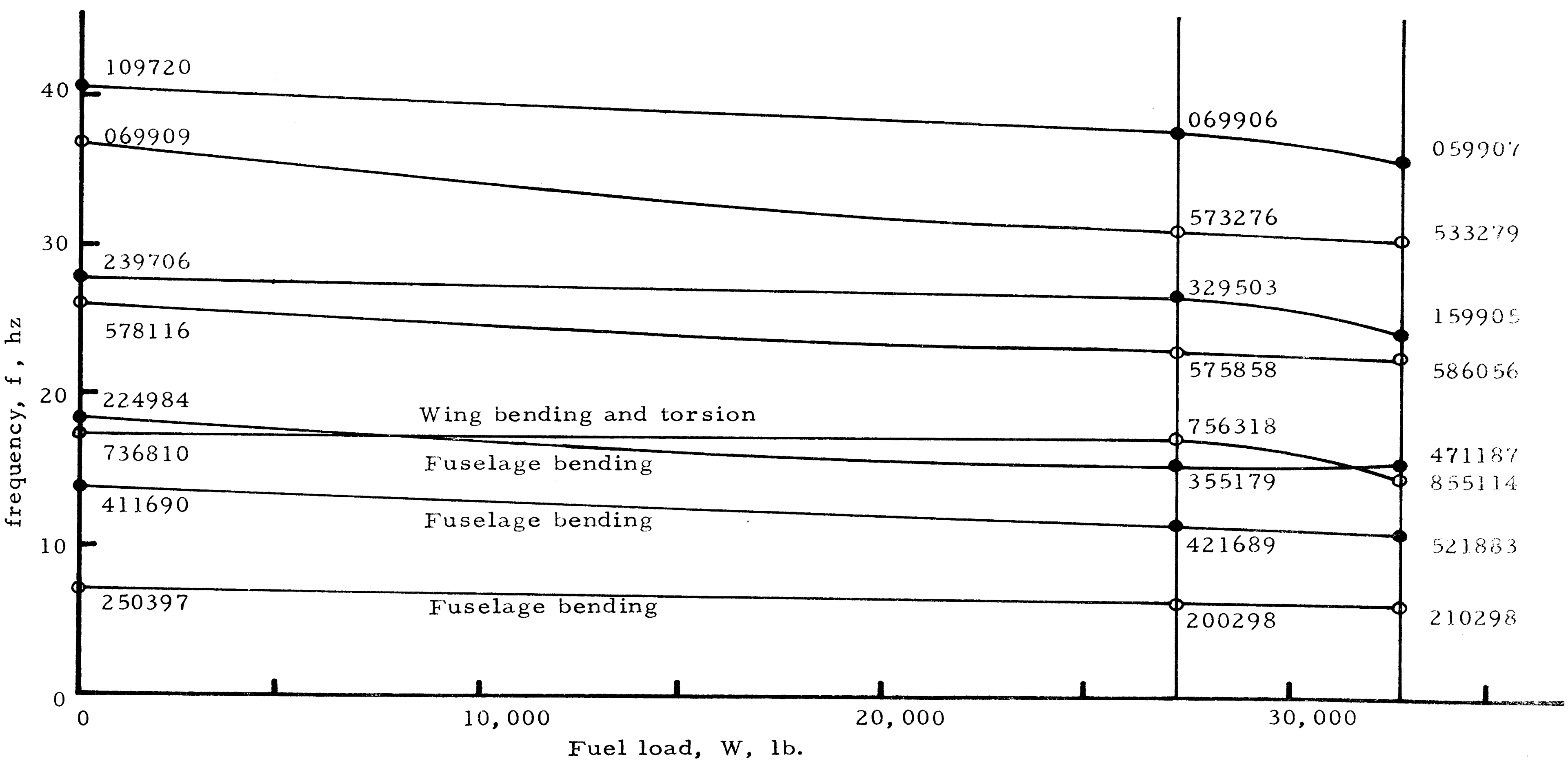


Fig. 5. Ground vibration frequencies for aircraft type A with varying fuel load. (16° l.e. sweep.)

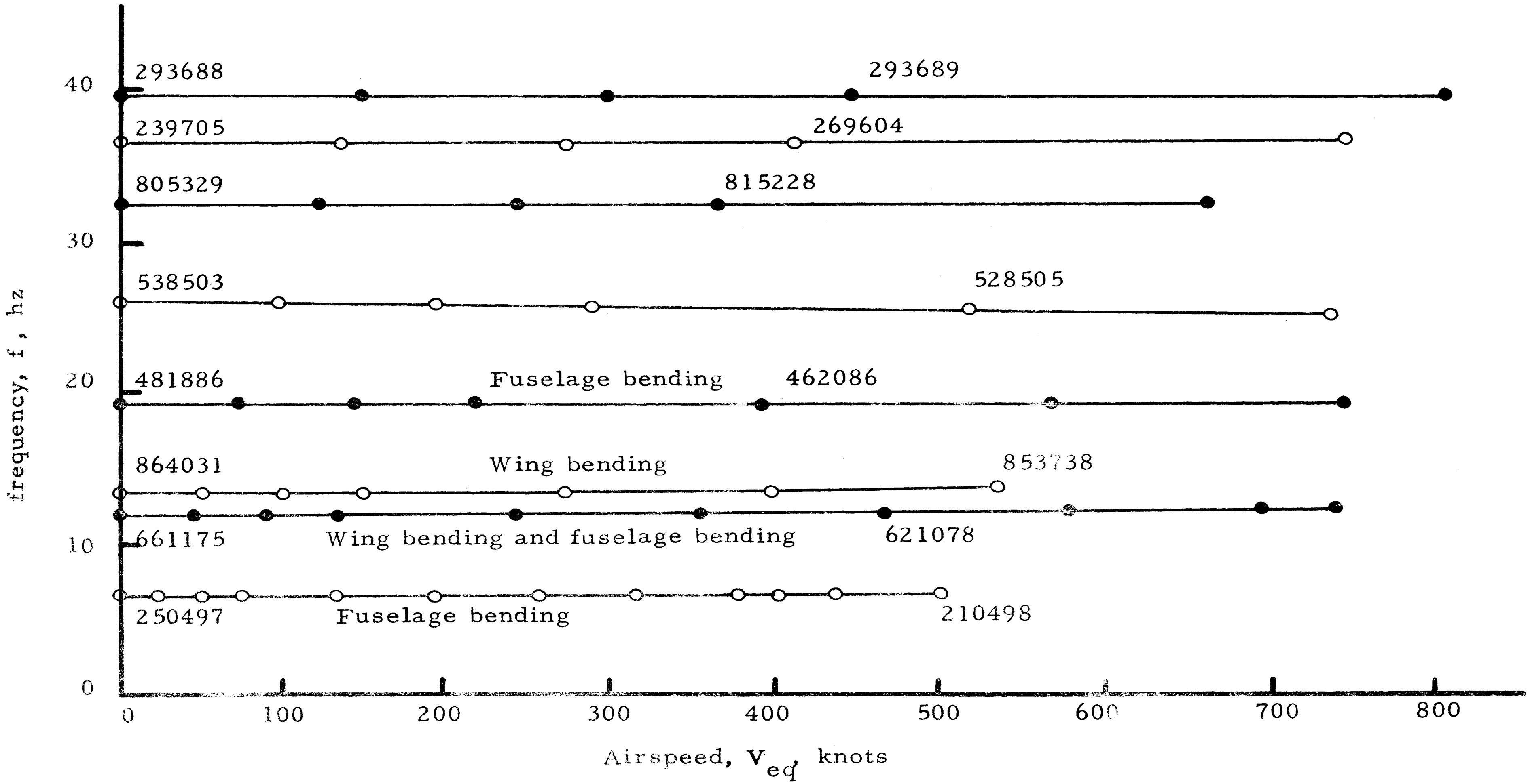


Fig. 6. Modal frequencies of aircraft type A in flight. Weight = 44, 507 lb. (Dry airplane, 72.5° l.e. sweep, symmetric modes.)

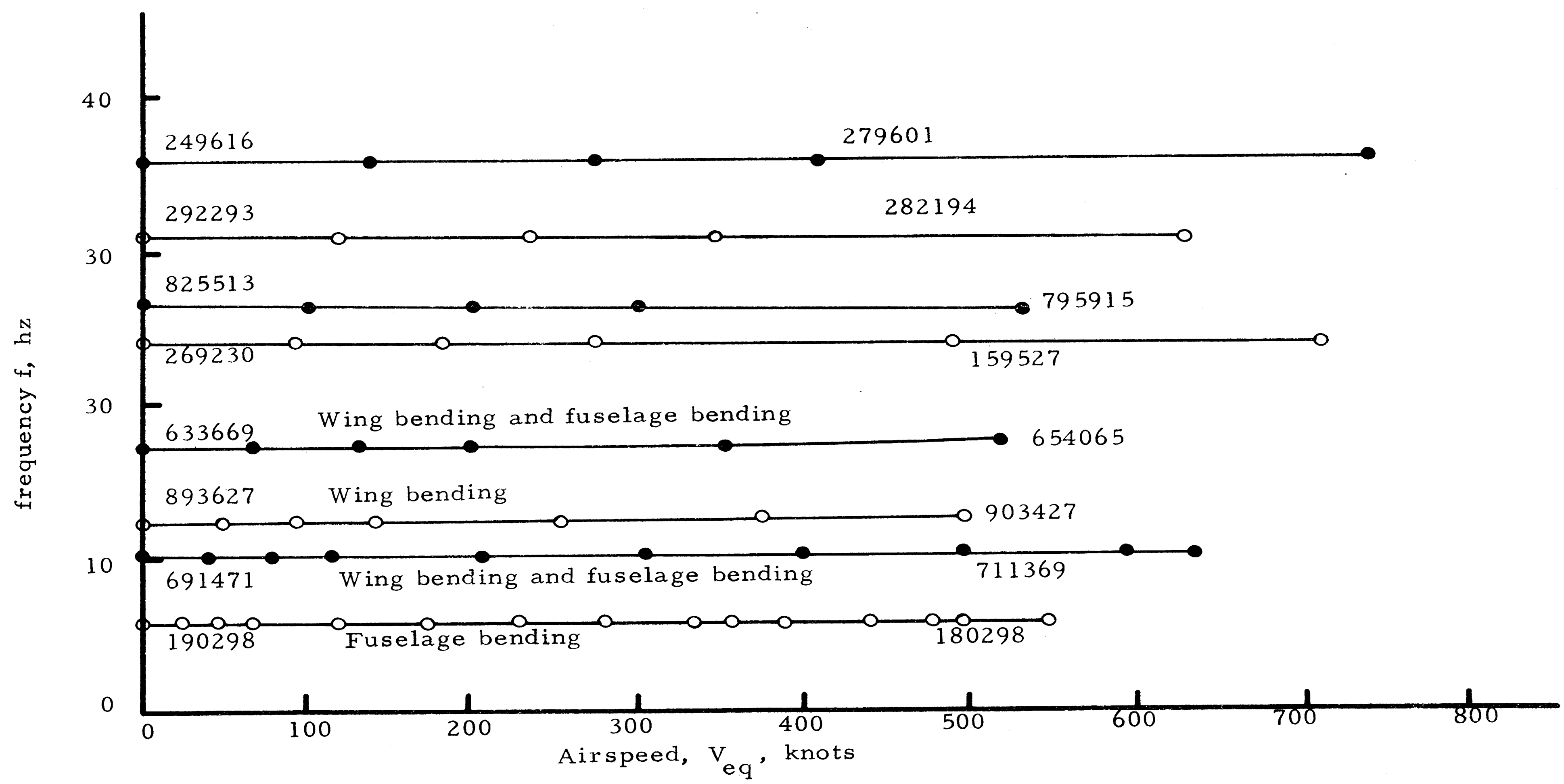


Fig. 7. Modal frequencies of aircraft type A in flight. Weight = 71, 834 lb. (Drying, full fuselage fuel = 23, 327 lb, 72.5^{o} l.e. sweep, symmetric modes.)

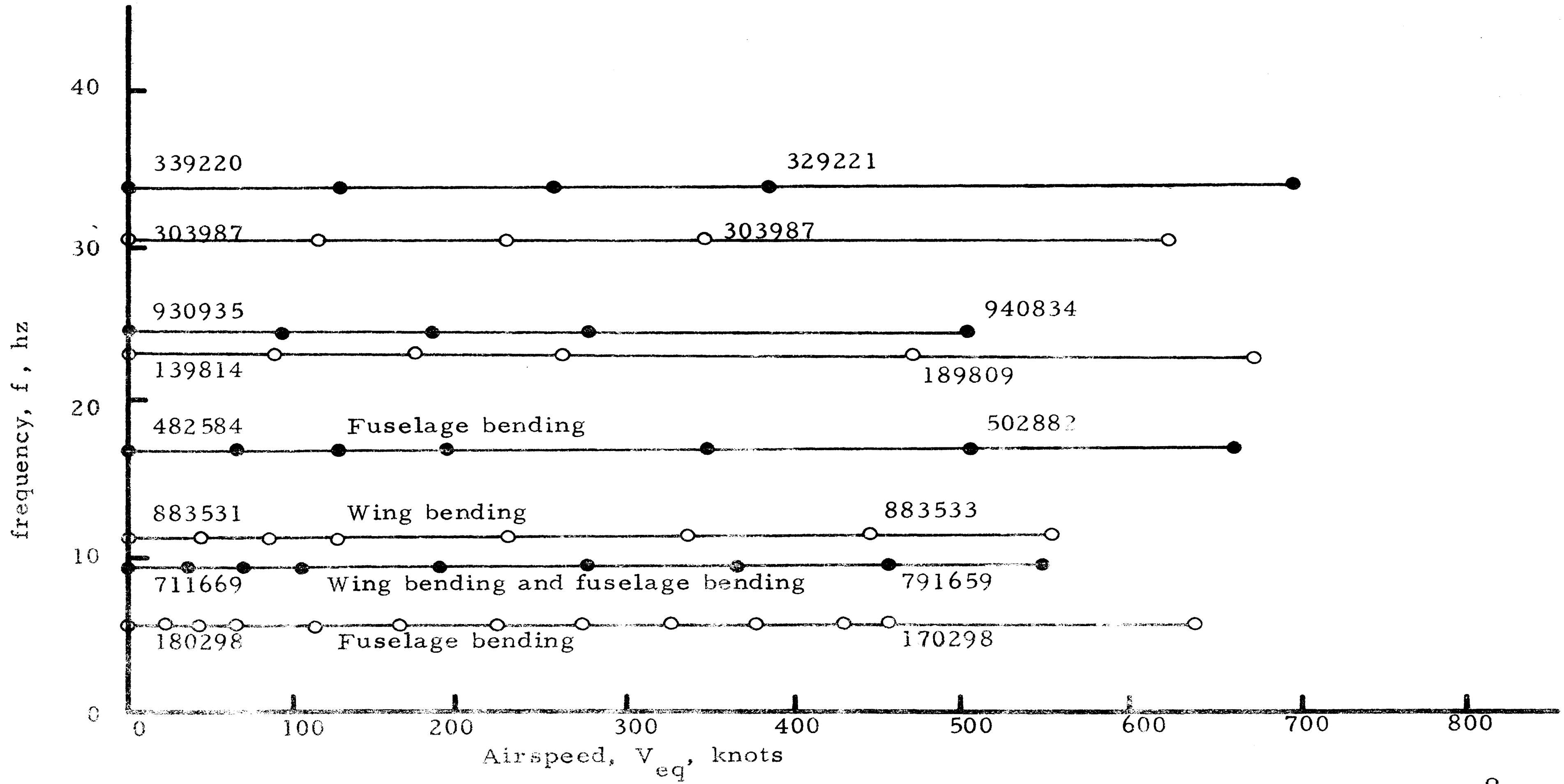


Fig. 8. Modal frequencies of aircraft type A in flight. Weight = 77,302 lb. (Fully fueled, 72.5° l.e. sweep, symmetric modes.)

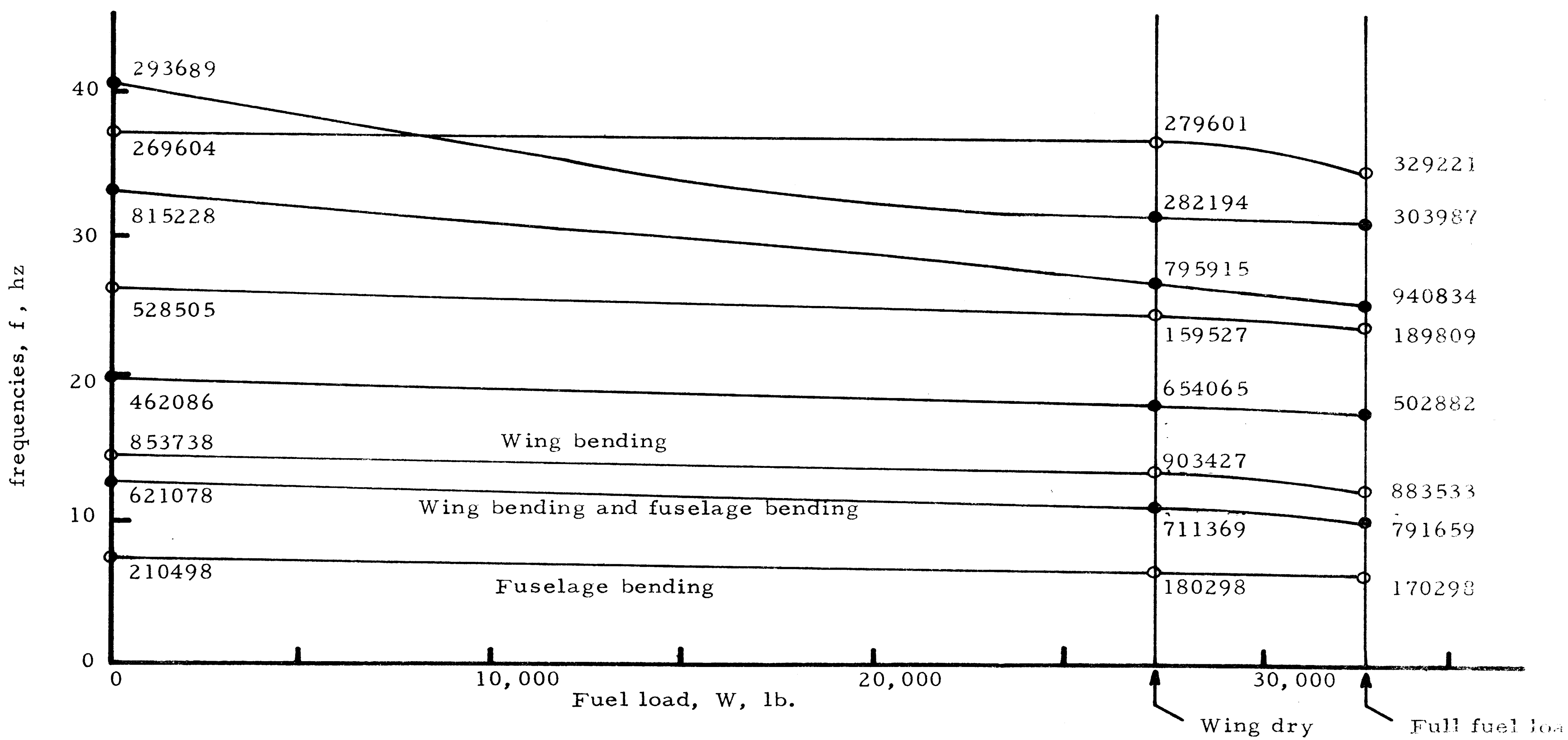


Fig. 9. Modal frequencies for aircraft type A with varying fuel load at 500 knots. (72.5° l.e. sweep, symmetric modes.)

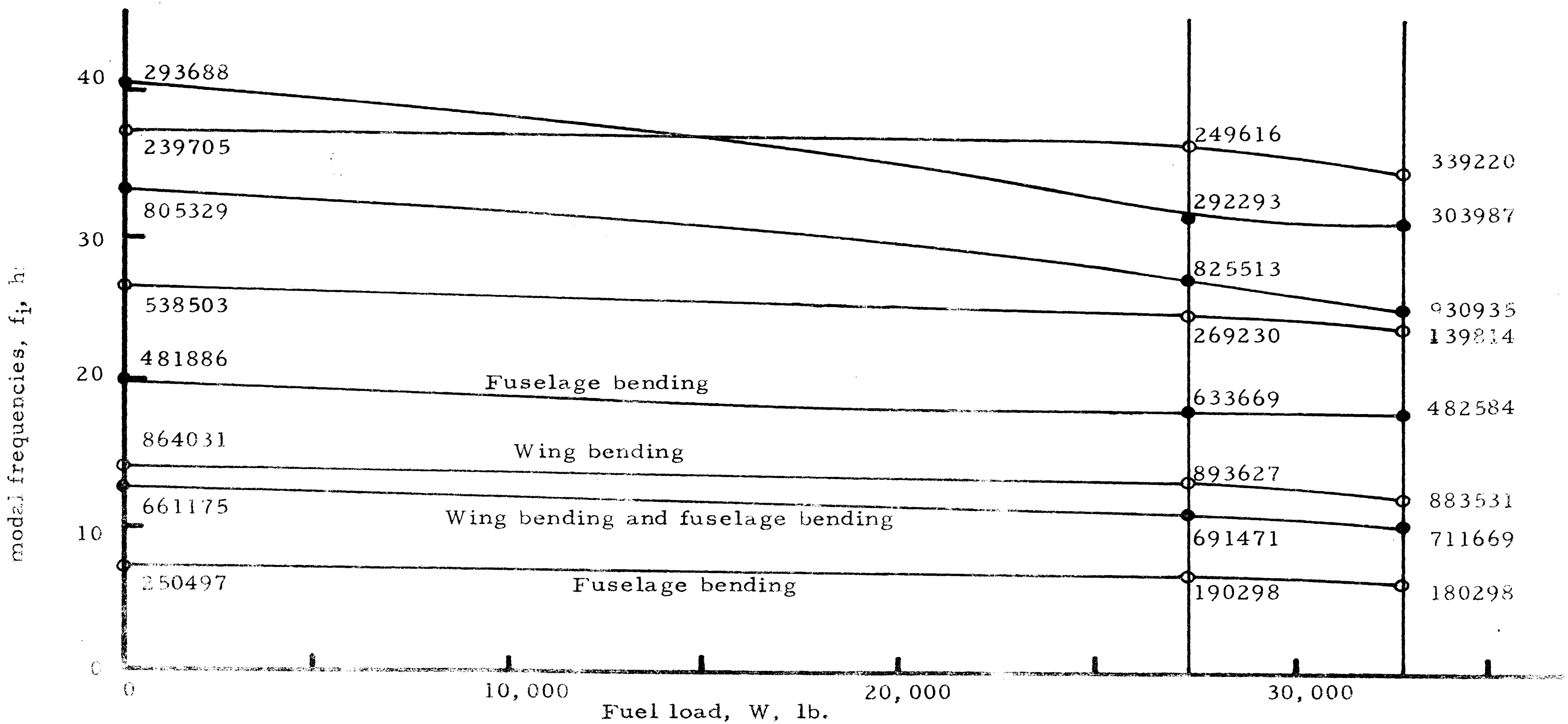


Fig. 10. Ground vibration frequencies for aircraft type A with varying fuel load. (72.5° l.e. sweep, symmetric modes.)

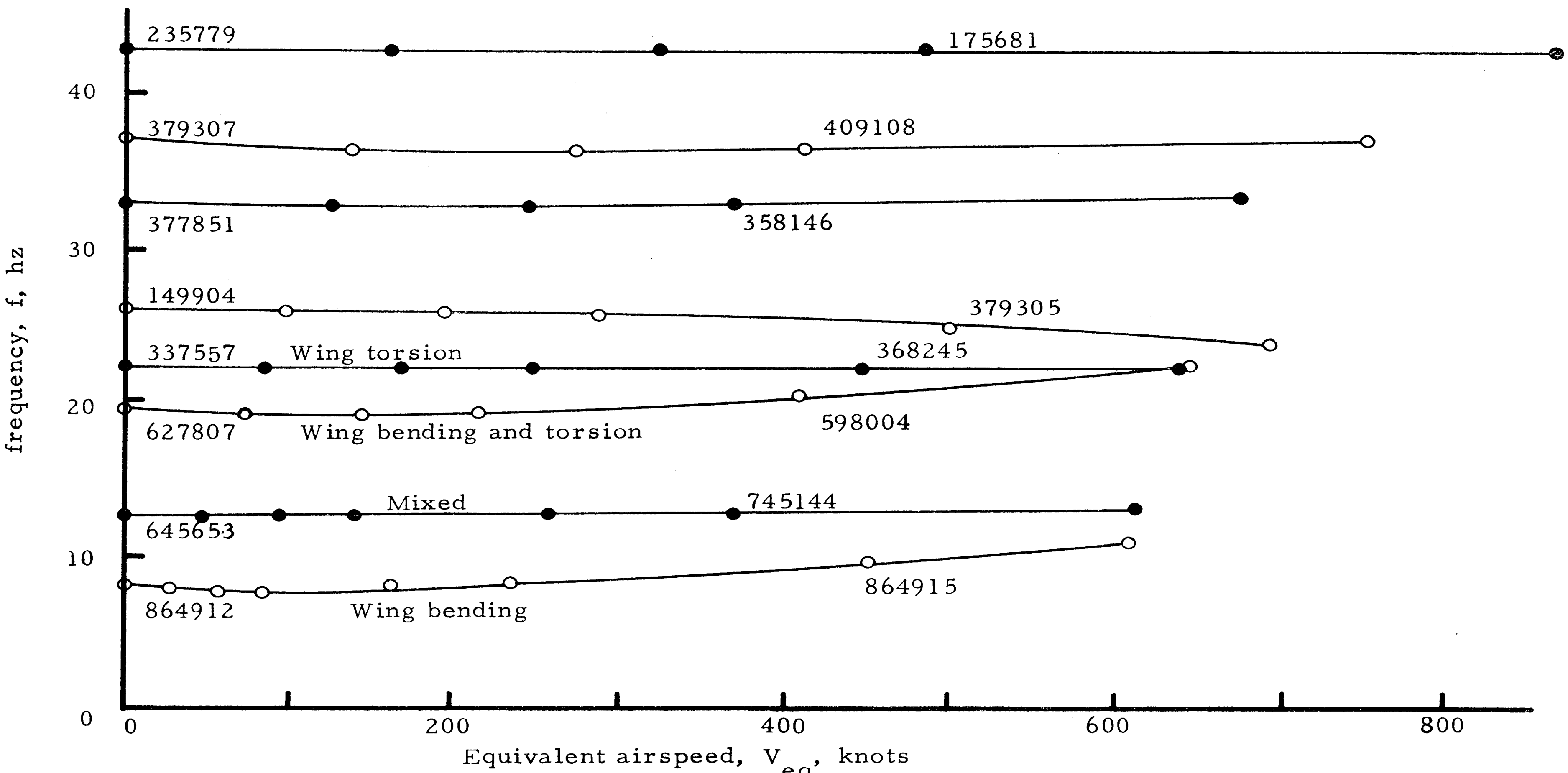


Fig. 11. Modal frequencies of aircraft type B in flight. Weight = 37,704 lb. (No pylons, partially fueled, symmetric modes.)

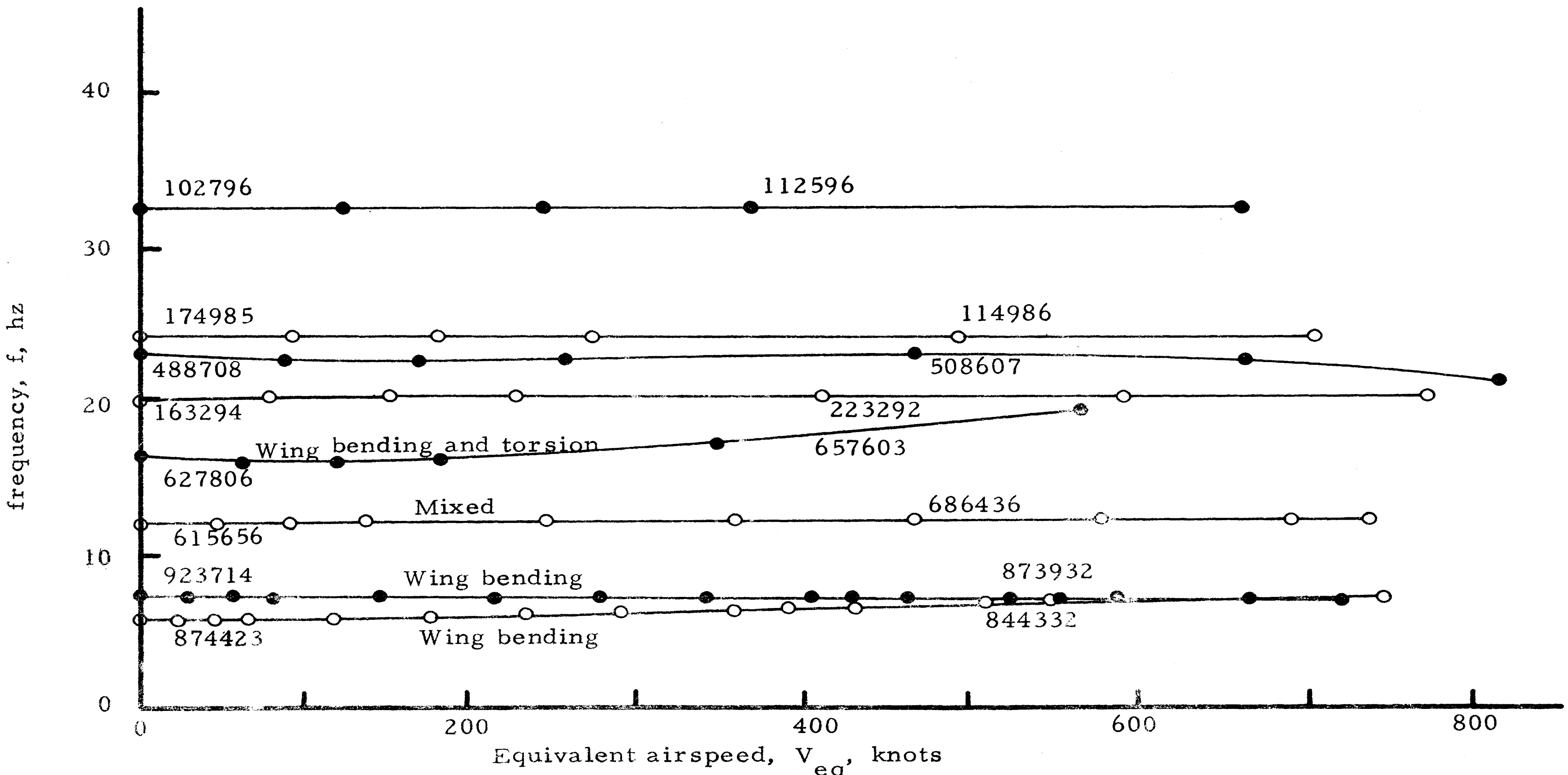


Fig. 12. Modal frequencies of aircraft type B in flight. Weight = 49,068 lb. (Four pylons per wing with two MER racks, 11,364 lb. armament, symmetric modes.)

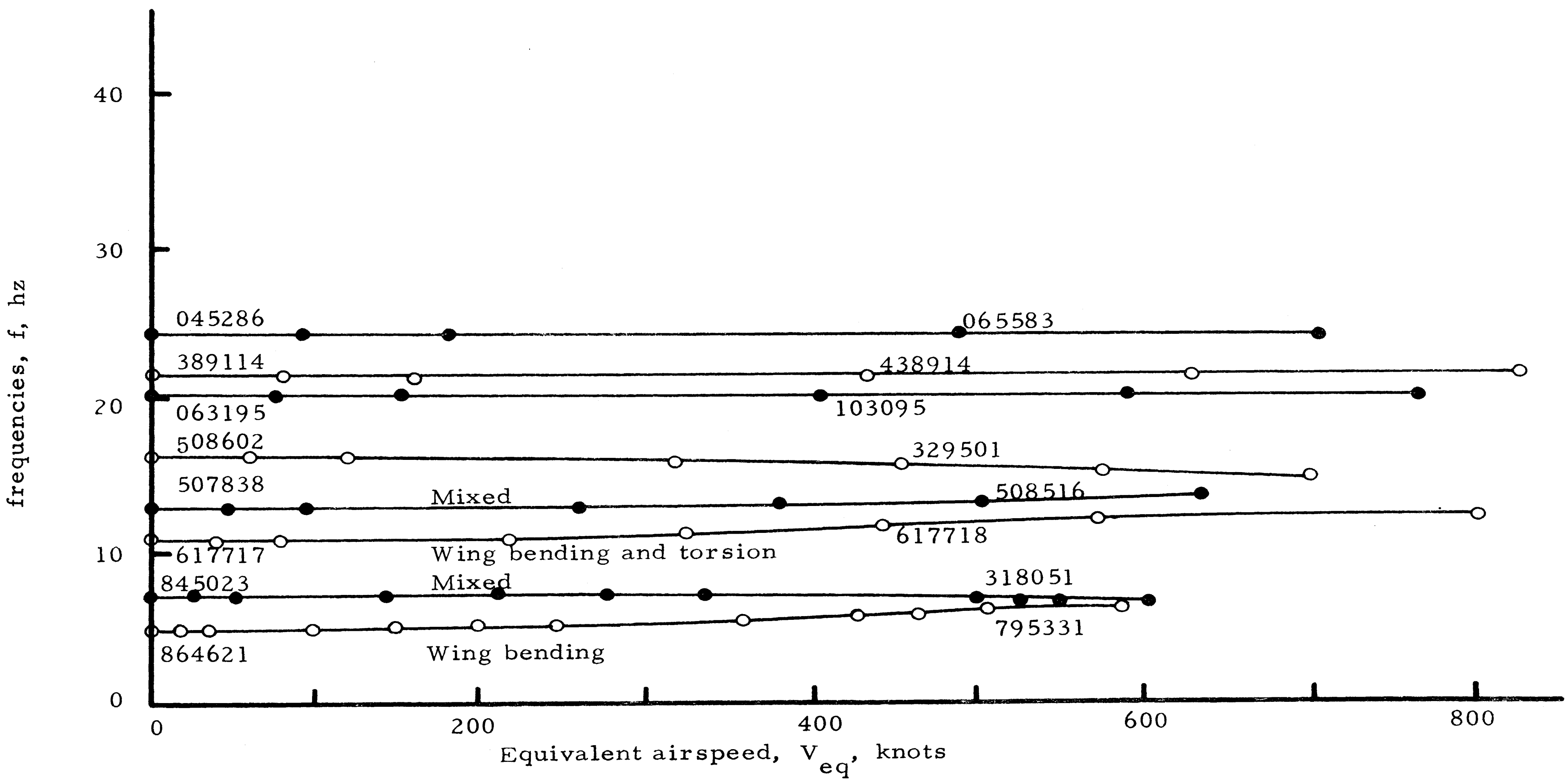
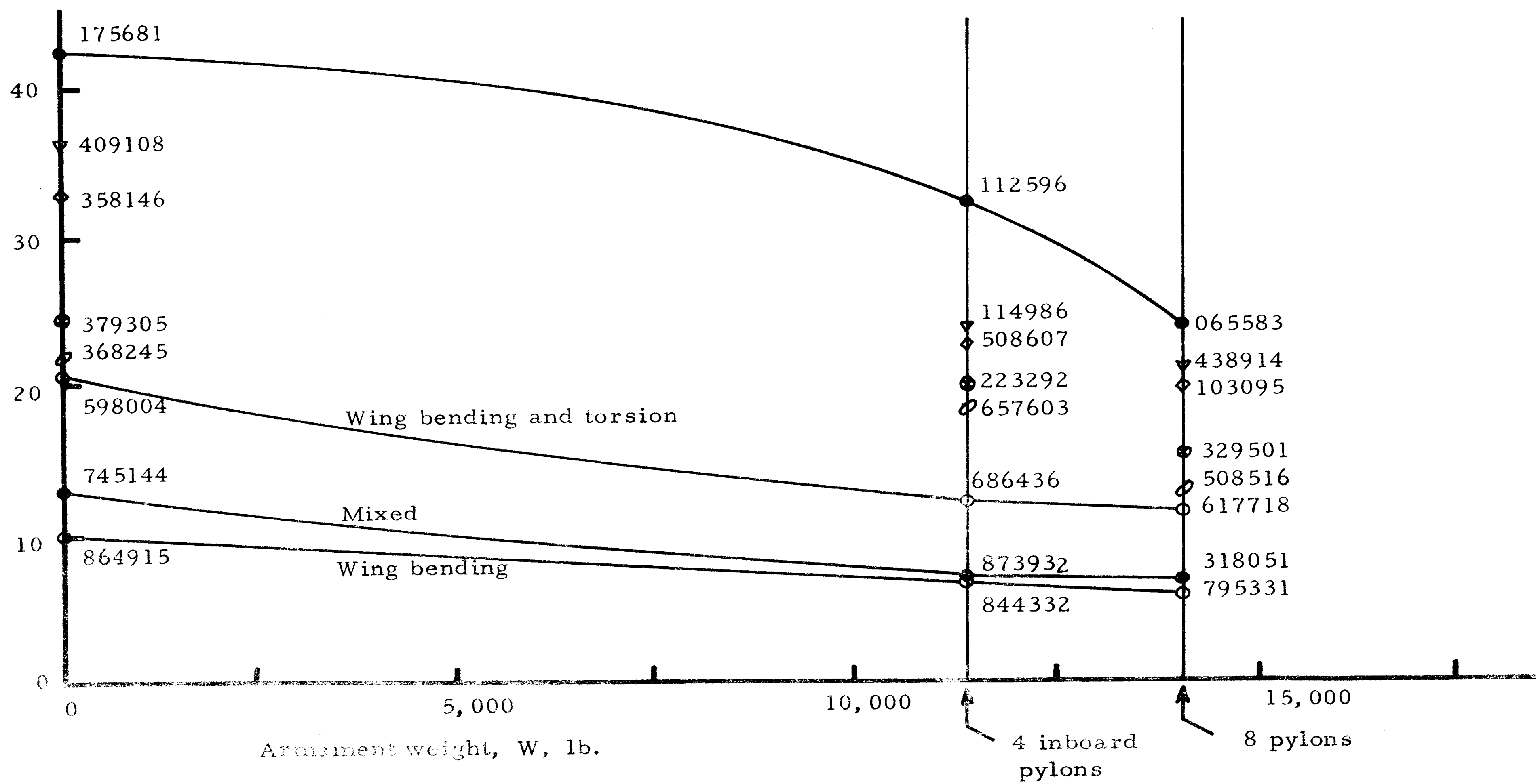


Fig. 13. Modal frequencies of aircraft type B in flight. Weight = 51, 752 lb. (Four pylons with two MER per wing, 14, 048 lb. armament, symmetric modes.)



500 knots (Symmetric modes.)

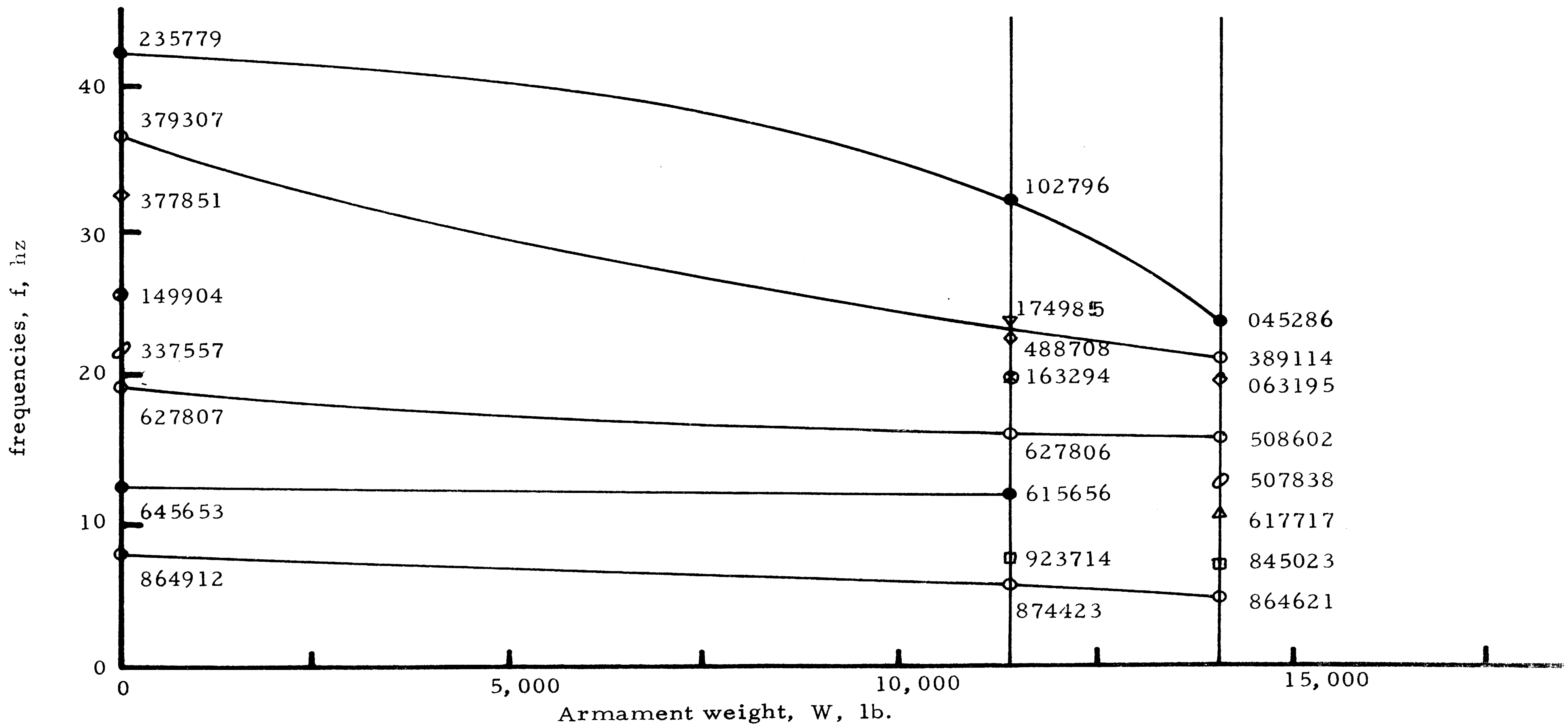


Fig. 15 Ground vibration frequencies for aircraft type B with varying armament. (Symmetric modes.)

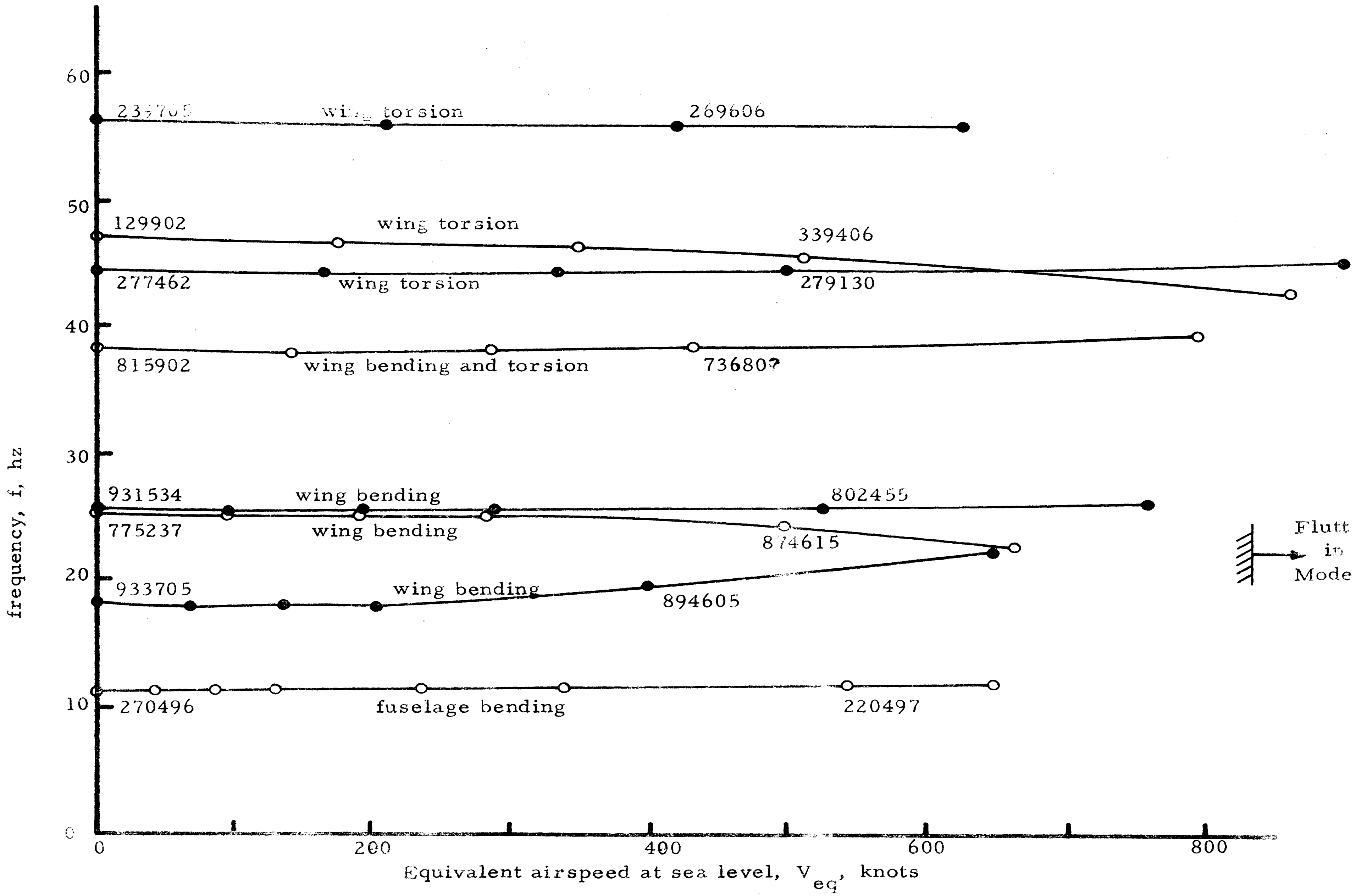


Fig. 16 Modal frequencies for aircraft type C in flight. Weight = 15,265 lb. Internal fuel only (light). (Symmetric modes. Strip theory.)

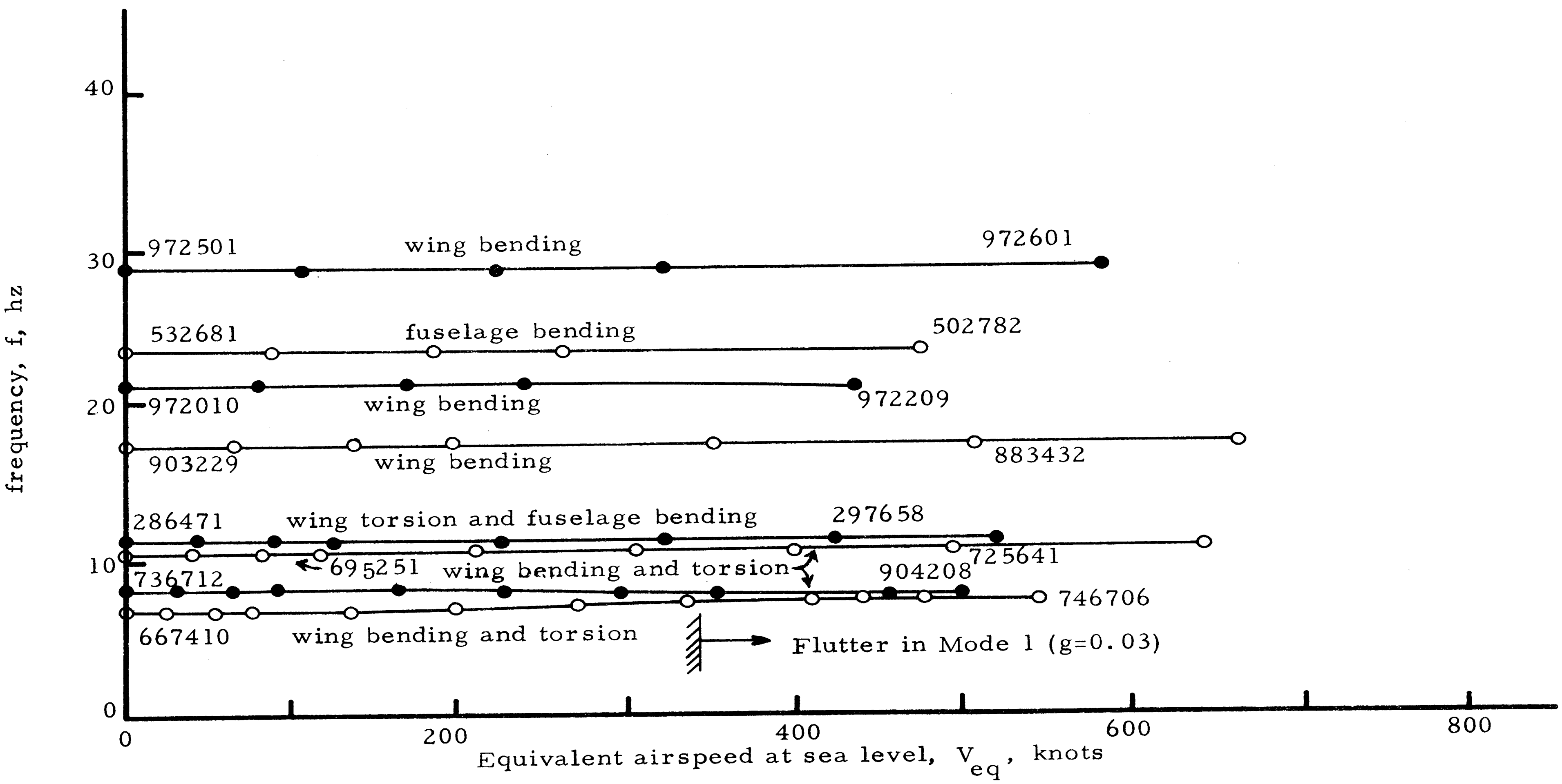


Fig. 13 Modal frequencies for aircraft type C in flight. Weight = 17,258 lb. (Symmetric modes.)

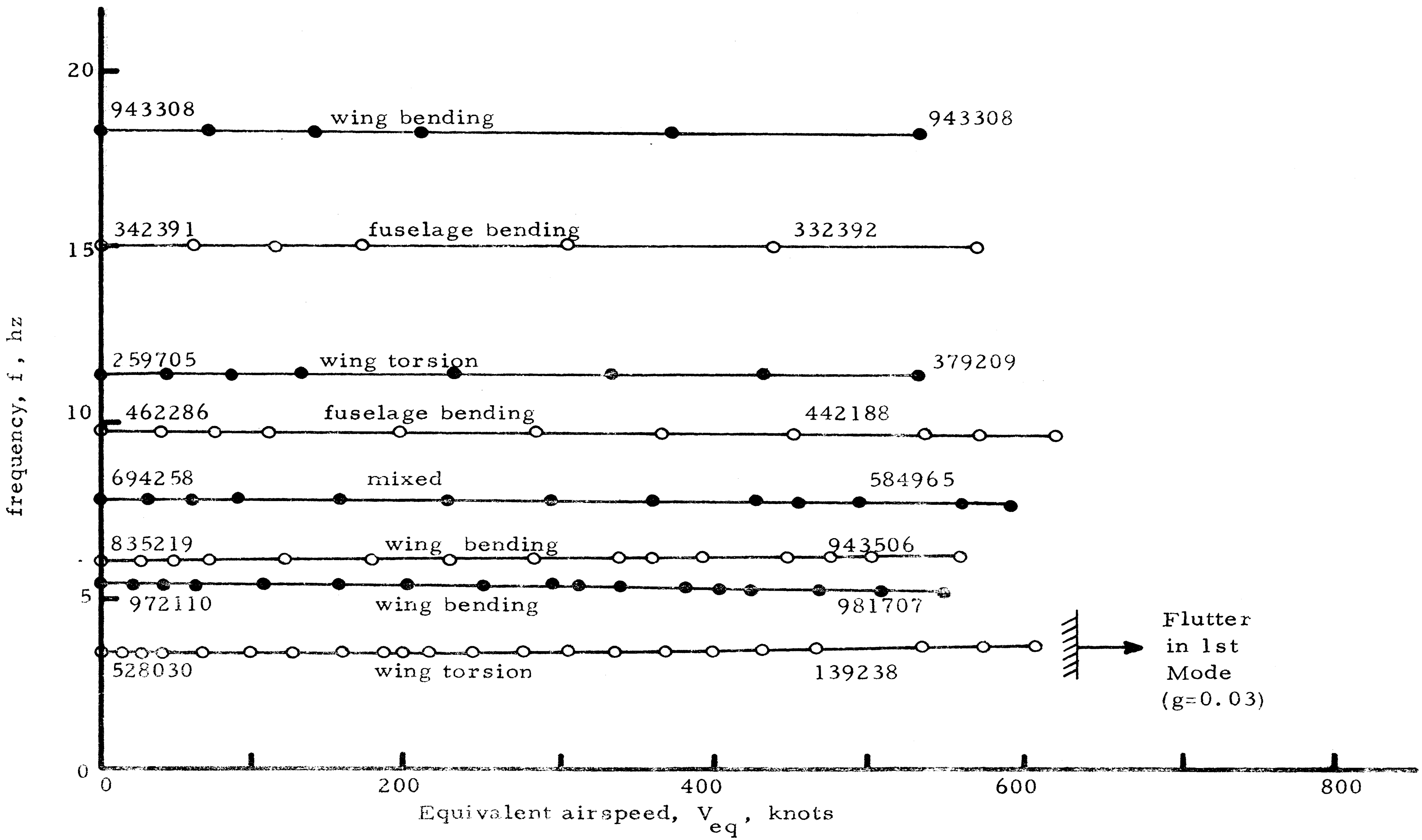


Fig. 18 Modal frequencies for aircraft type C in flight. Weight = 21,908 lb. (Symmetric modes.)

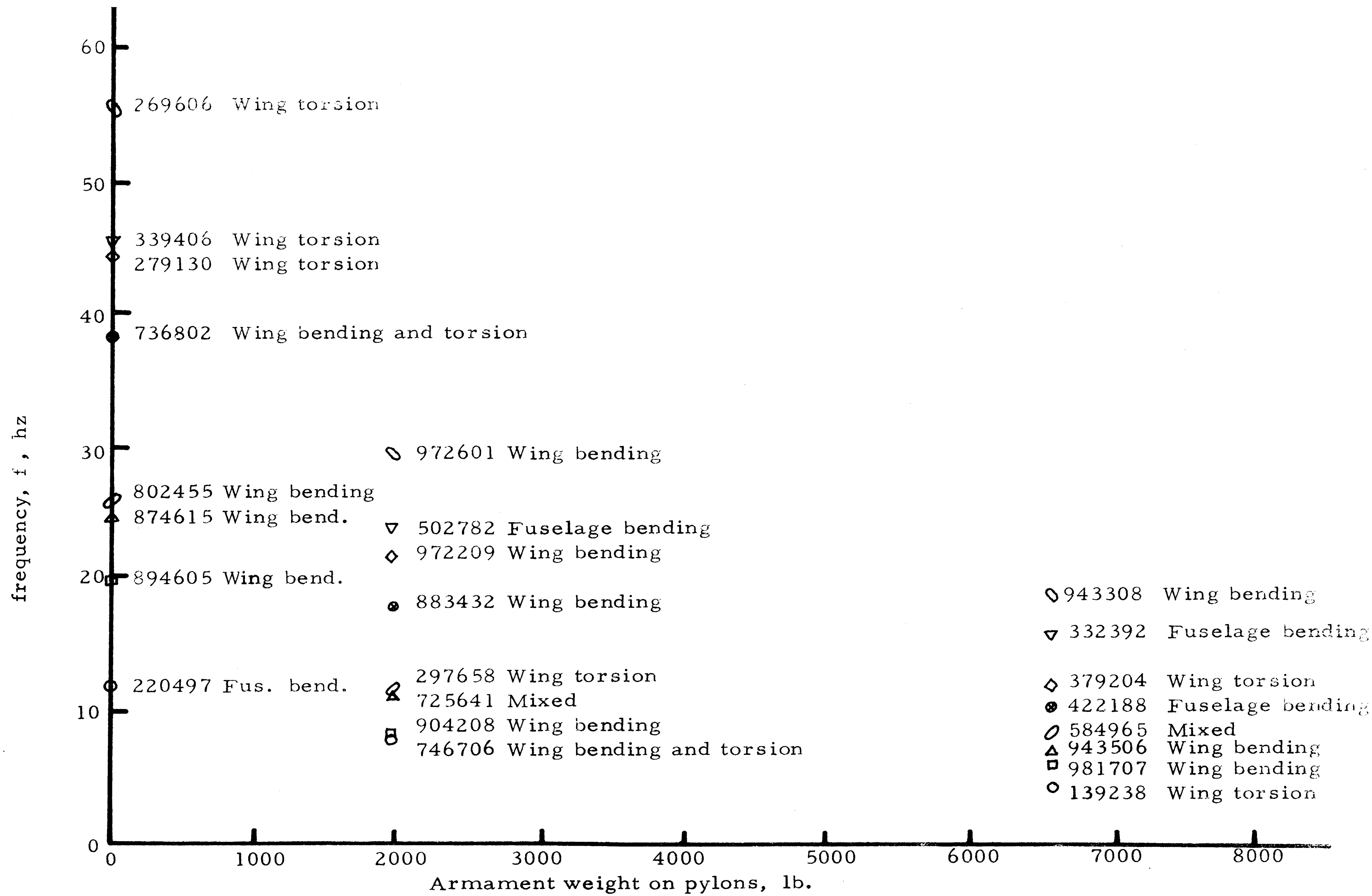


Fig. 19 Modal frequencies of airplane type C in flight at 500 knots. (Symmetric modes.)

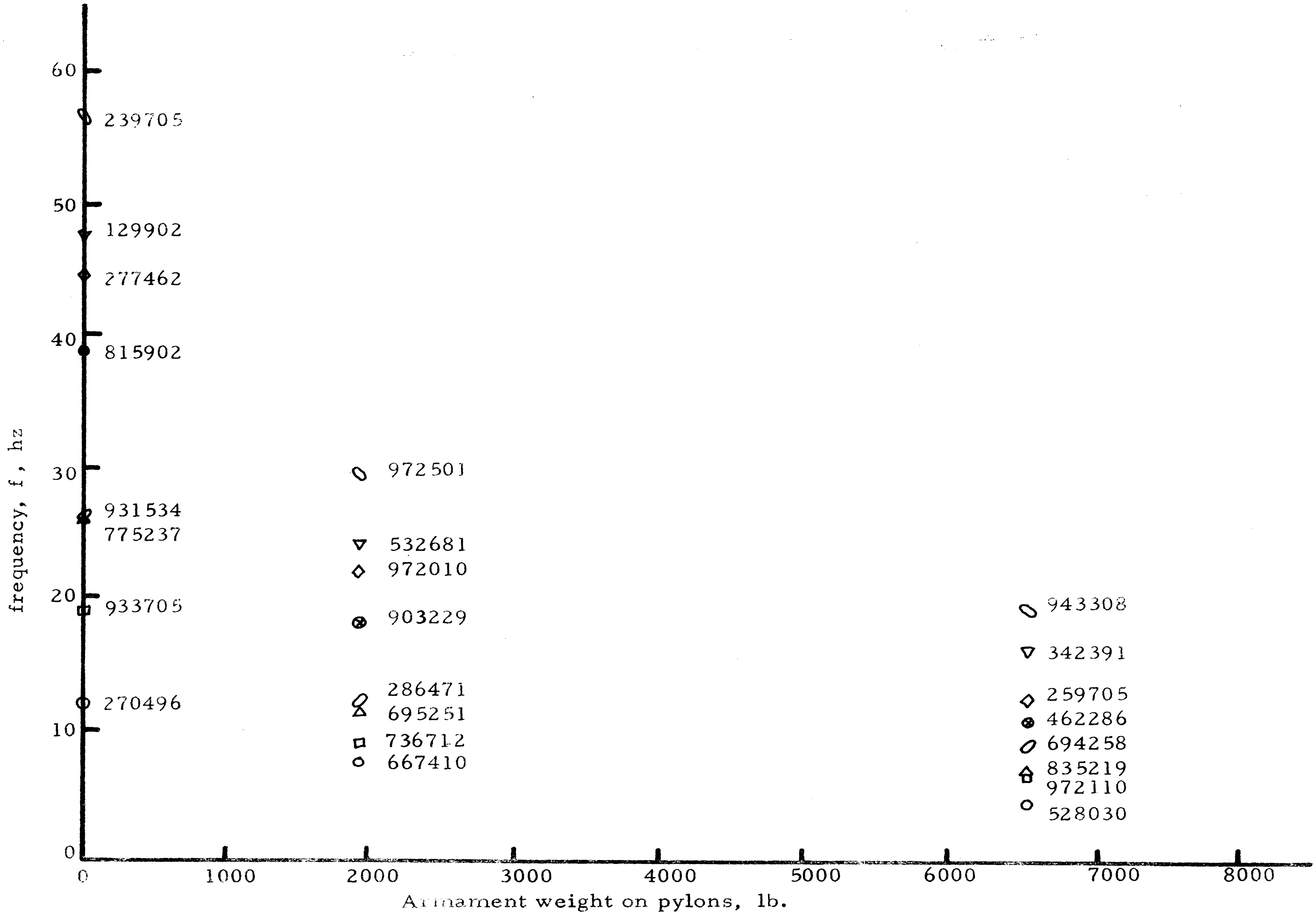


Fig. 20 Ground vibration frequencies for aircraft type C with varying fuel and armament load.

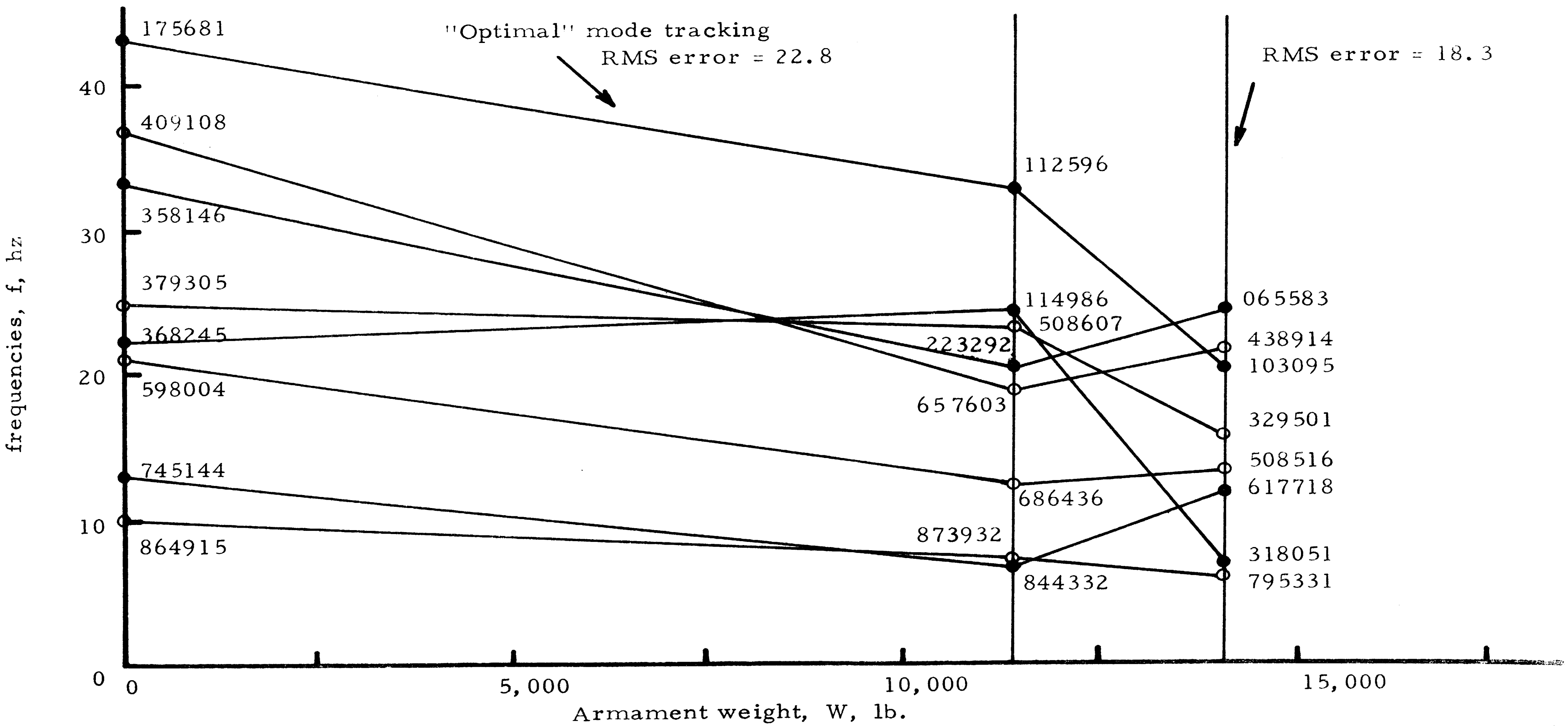


Fig. 21 Automated mode tracking attempt for aircraft type B at 500 knots. (Symmetric modes. A rework of data in Figure 14.)

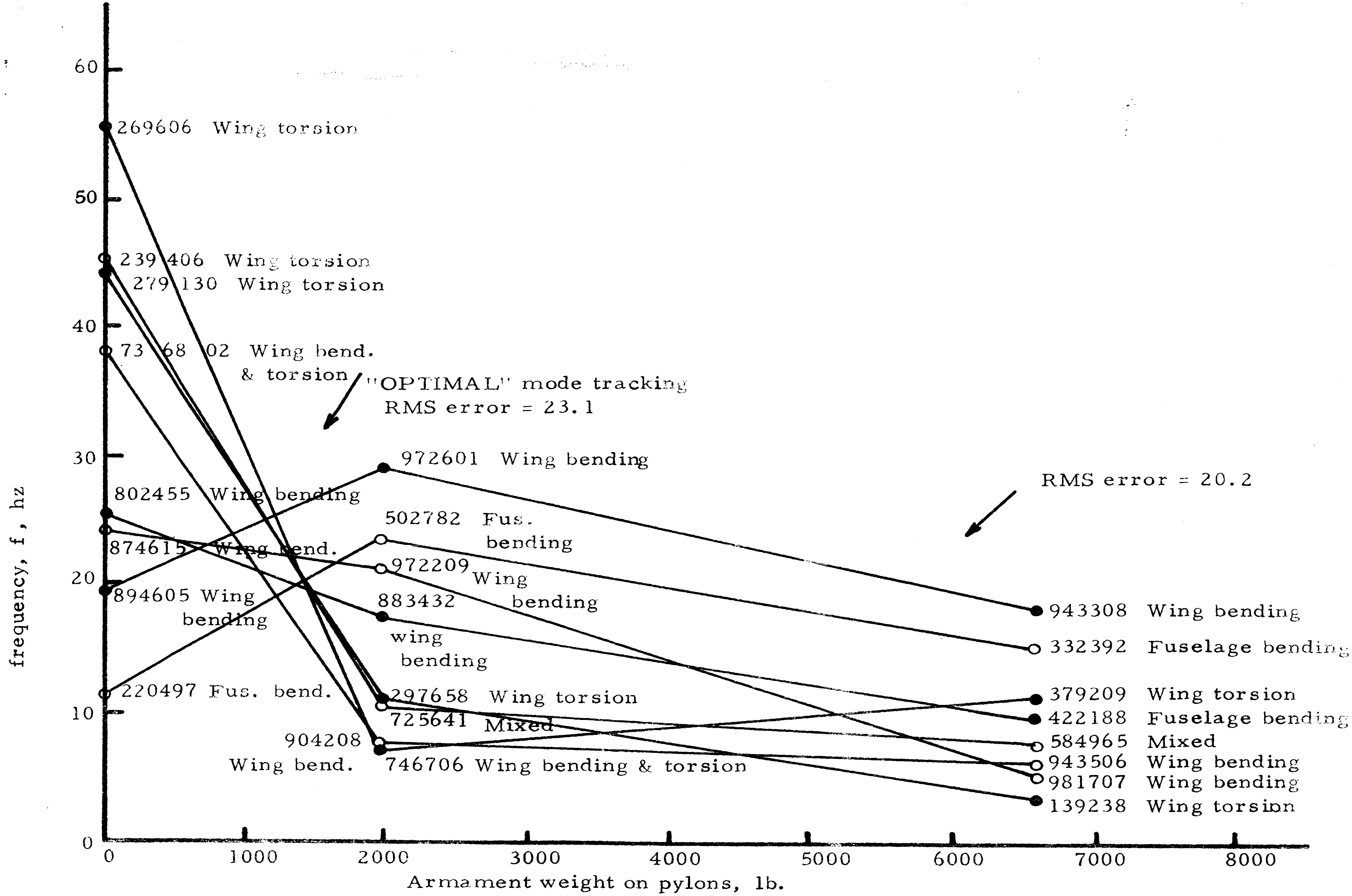


Fig. 22 Modal frequencies of airplane type C in flight at 500 knots. (Mode tracking "experiment" with frequency data. Symmetric modes.)

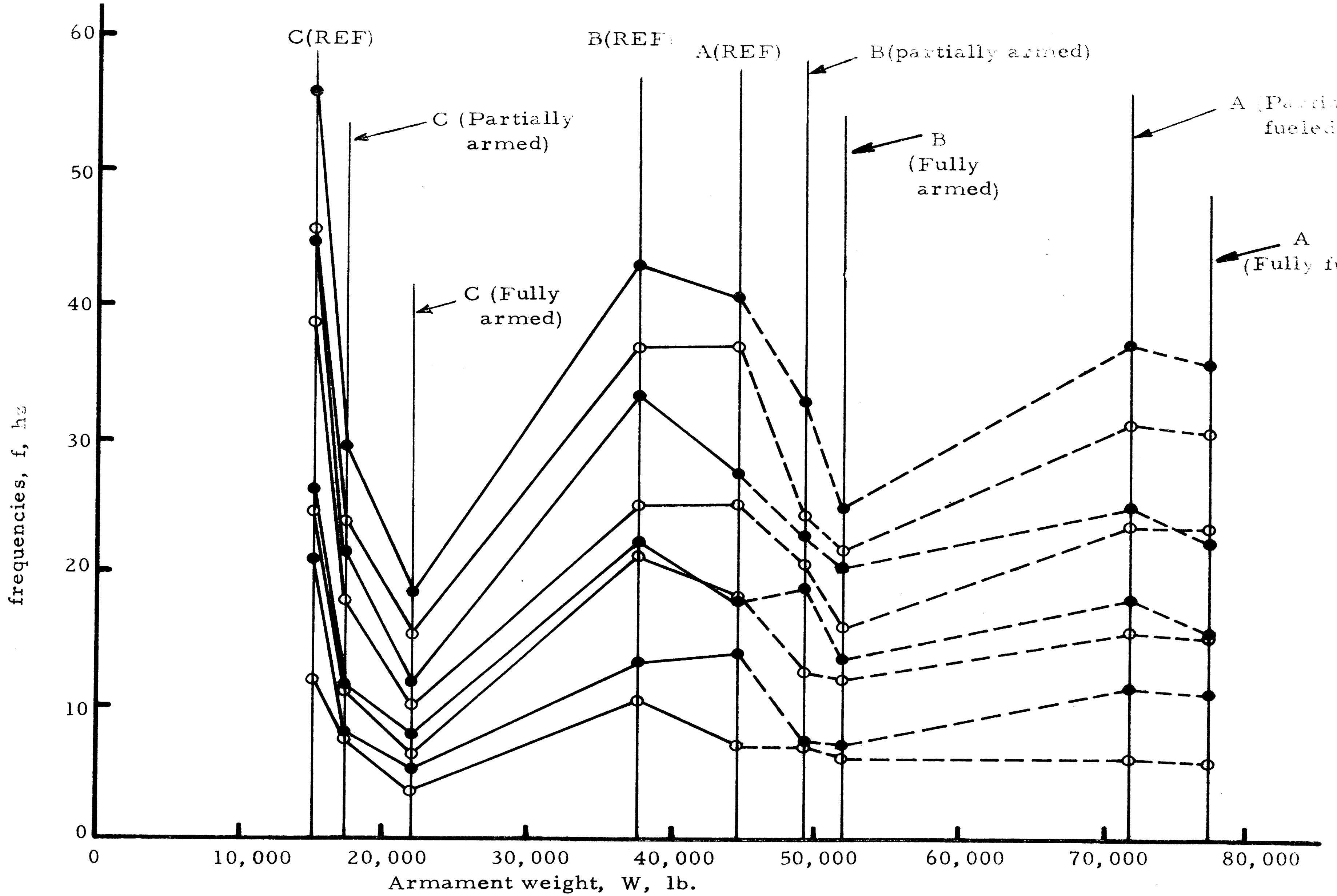


Fig. 23. Composite relation between modal frequencies and total aircraft weight at 500 knots. (Reference aircraft have no external stores.) (Symmetric modes.)

Figure 24(a). Fundamental mode of aircraft type A with wing fully forward (16° l.e. sweep). 494 knots at sea level. Medium weight = 71,834 lb. Vertical displacement of wing. f_1 = 5.98 hz

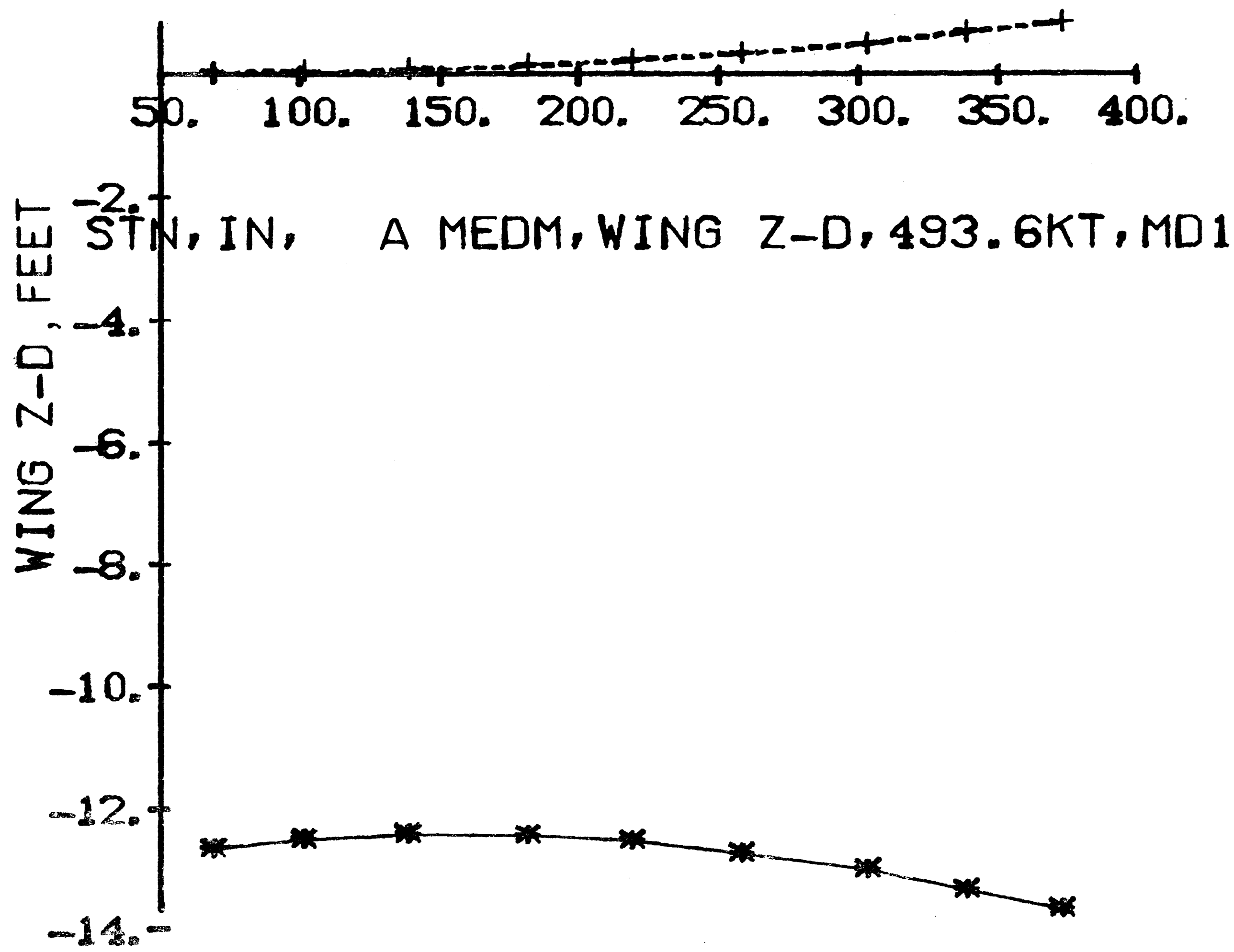


Figure 24(b). (cont.) Torsional rotation of wing.

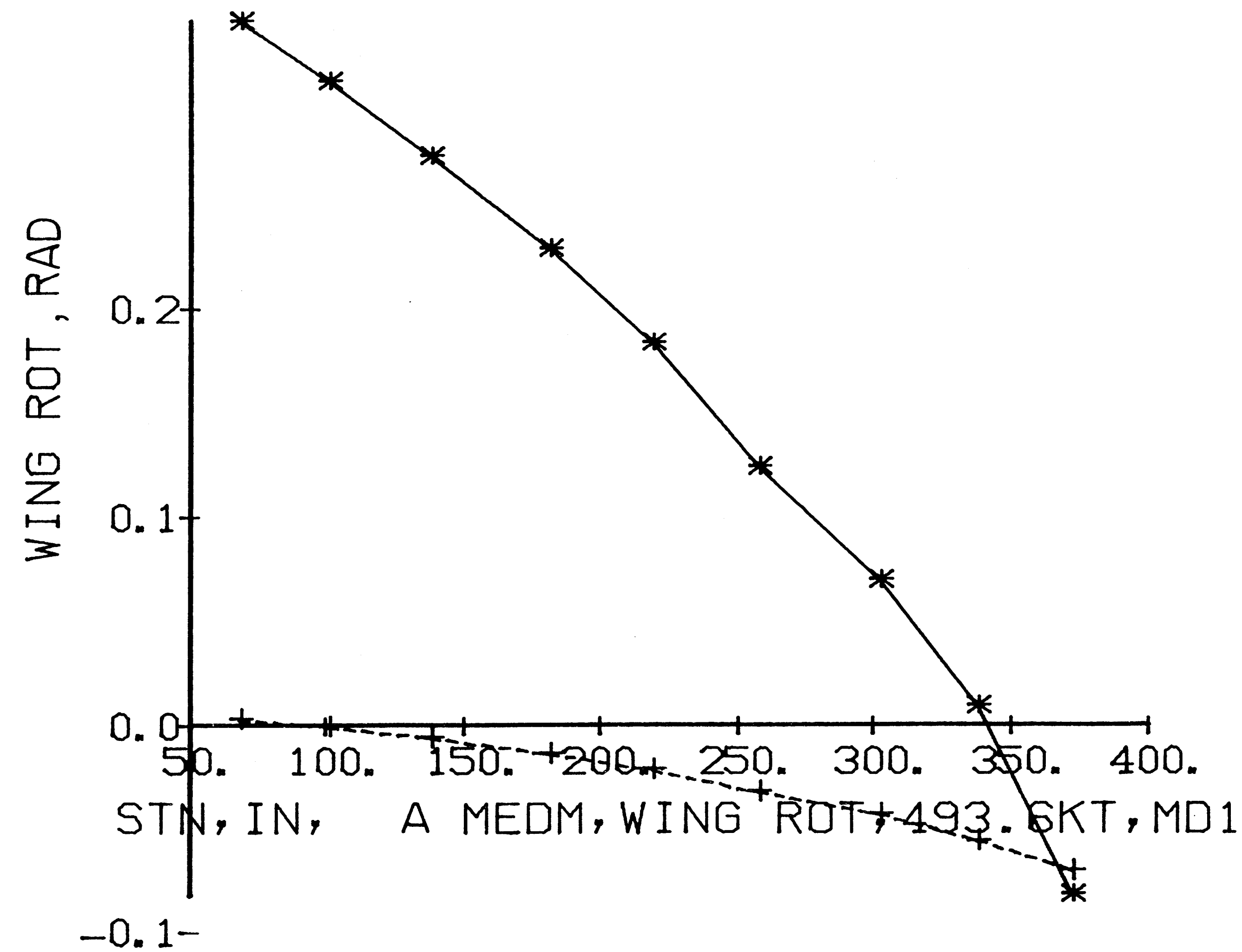


Figure 24(c). (cont.) Vertical displacement of fuselage.

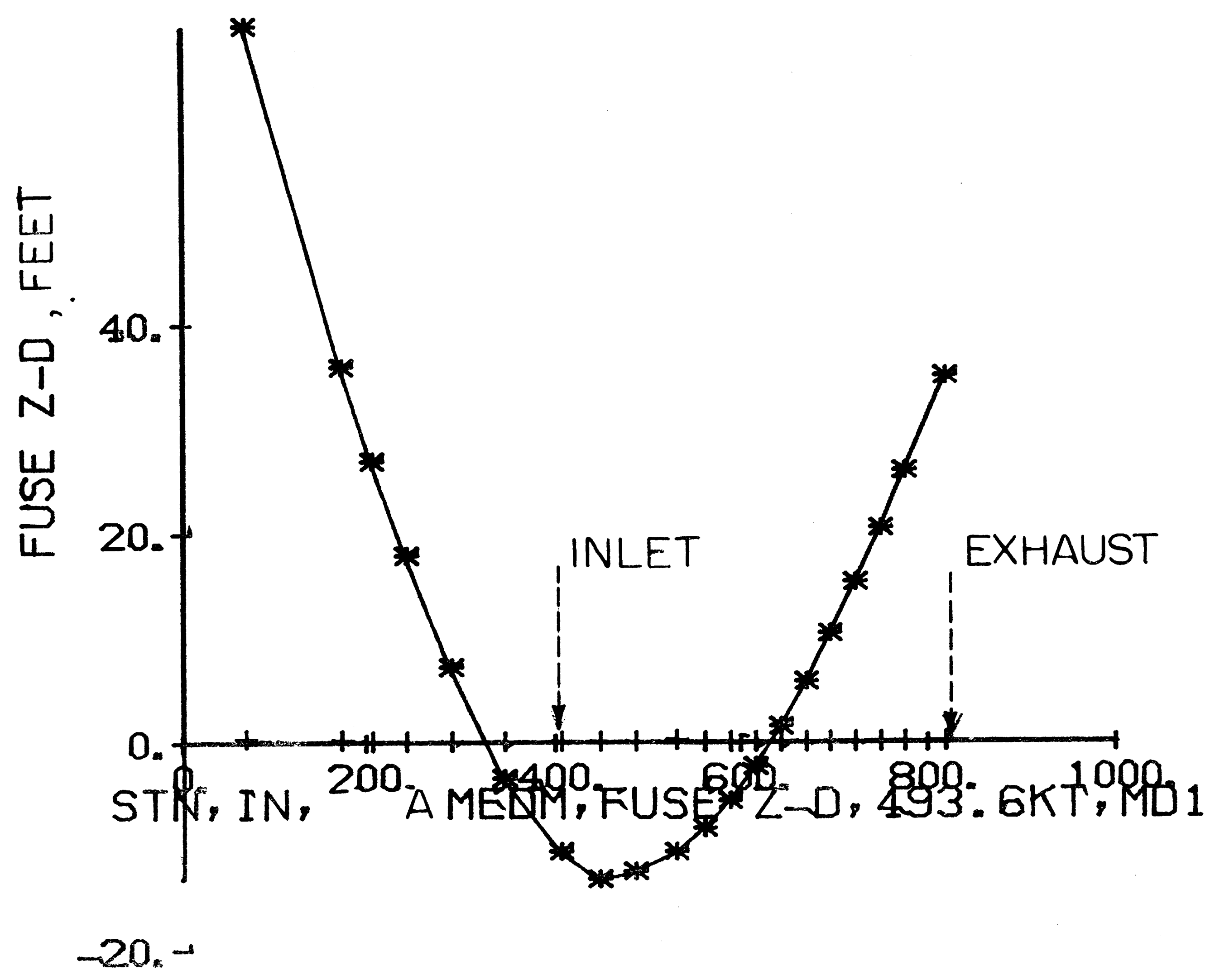


Figure 25(a). Fundamental mode of aircraft type A with wing fully swept (72.5° l.e. sweep). 498 knots at sea level. Medium weight = 71,834 lb. Vertical displacement of wing. $f_1 = 6.03$ hz.

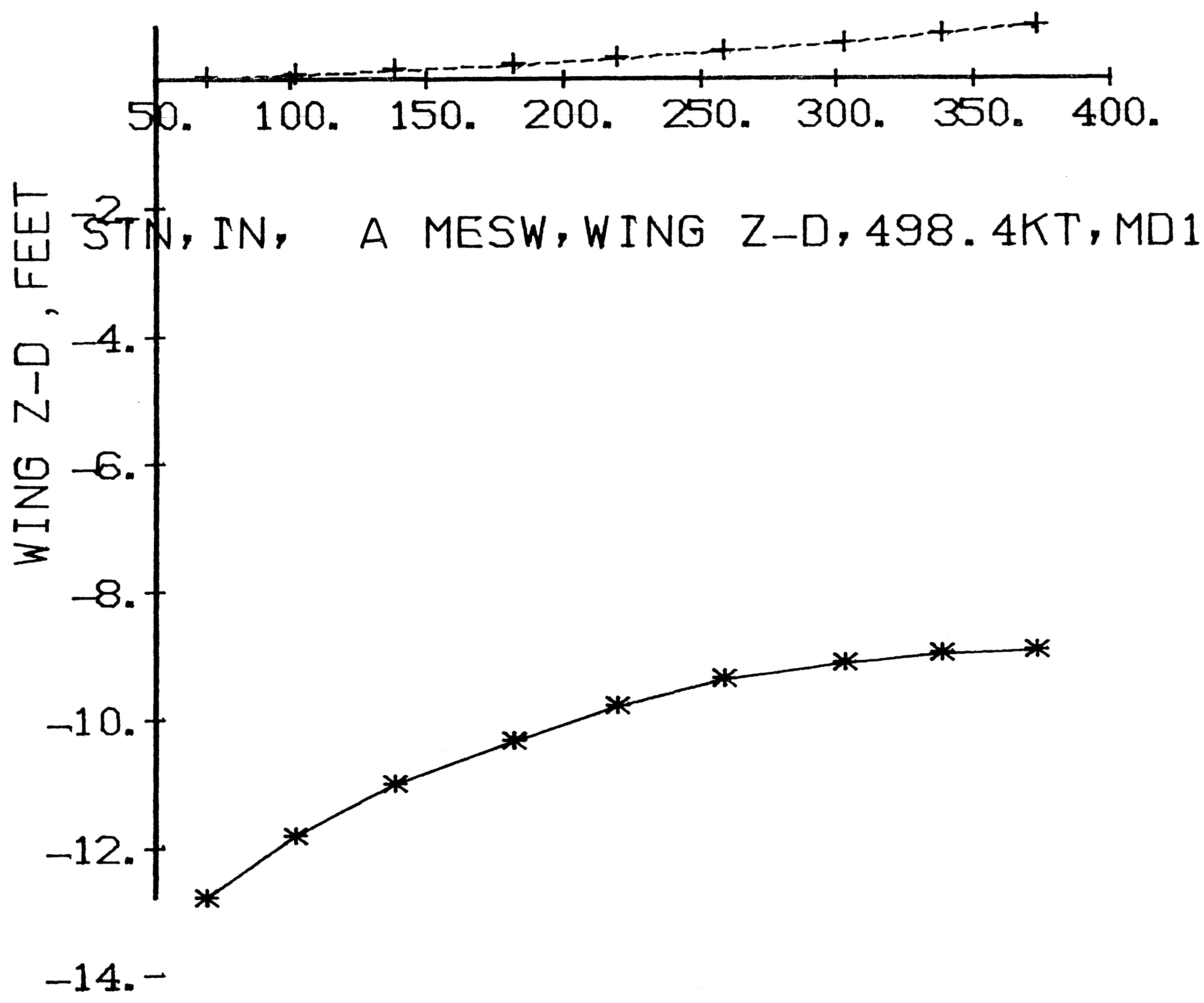


Figure 25(b). (cont.) Torsional rotation of wing.

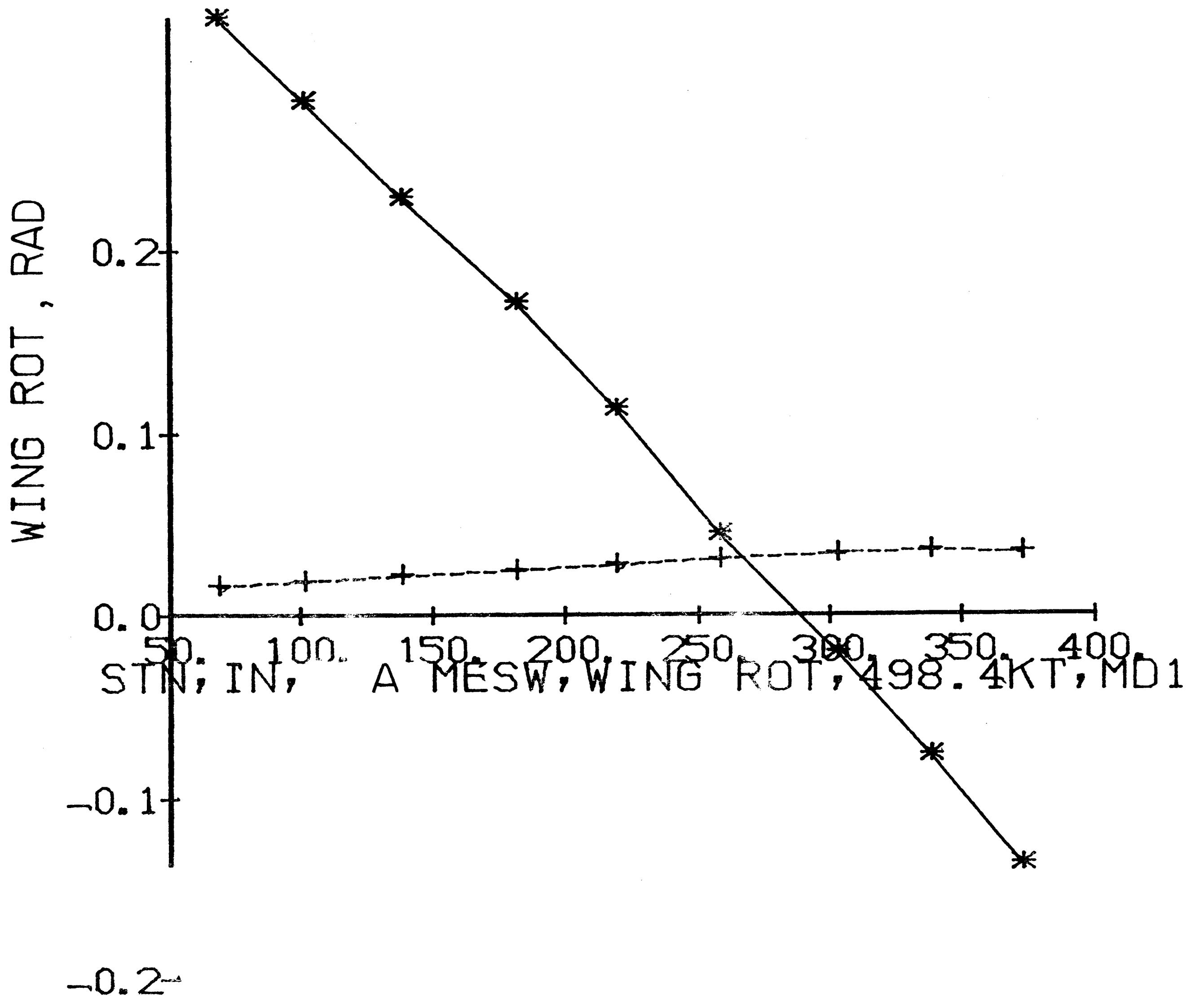


Figure 25(c). (cont.) Vertical displacement of fuselage.

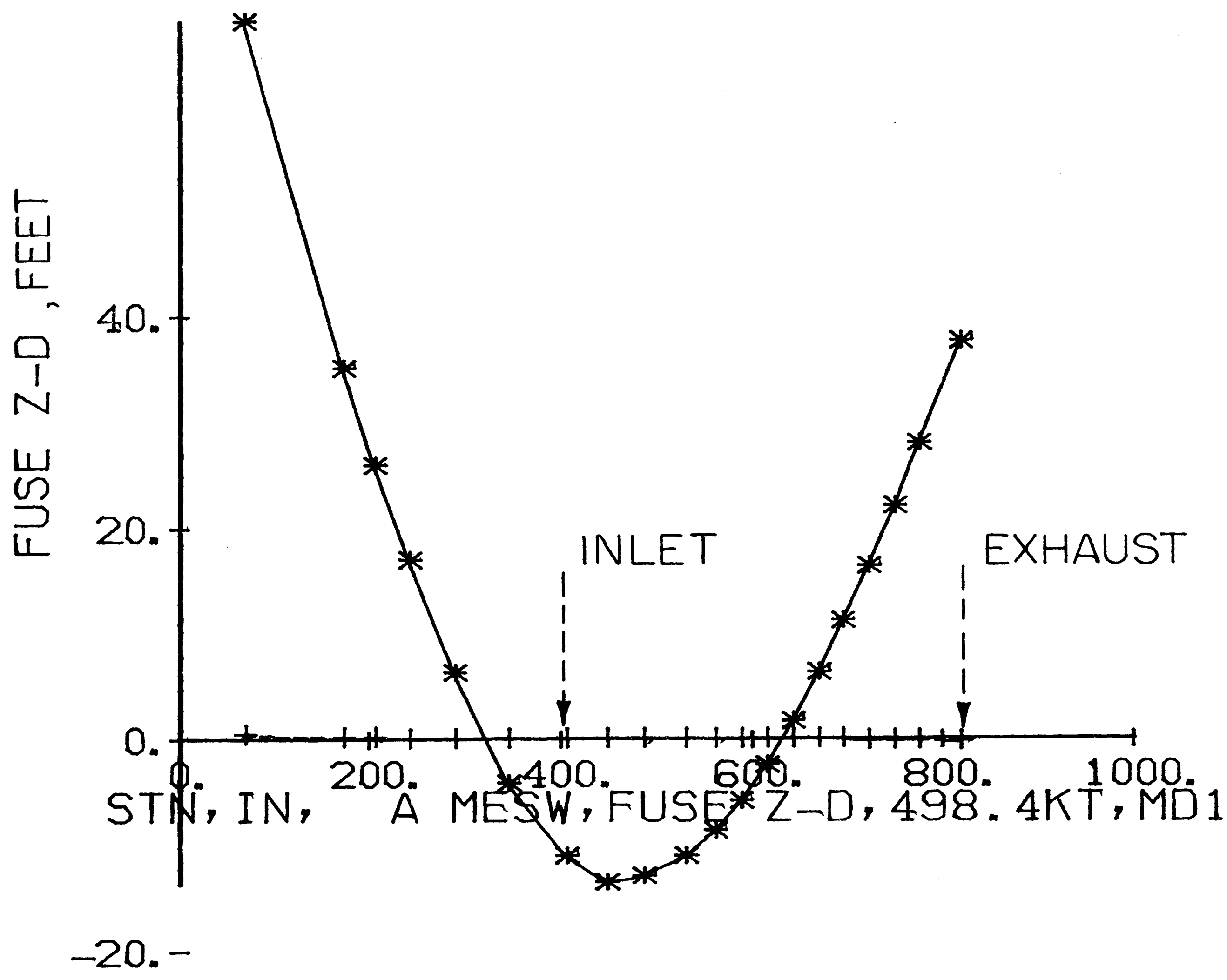


Figure 26(a). Fundamental mode of aircraft type B. 533.5 knots at sea level. Medium weight = 49,068 lb. Vertical displacement of wing. f_1 = 6.70 hz.

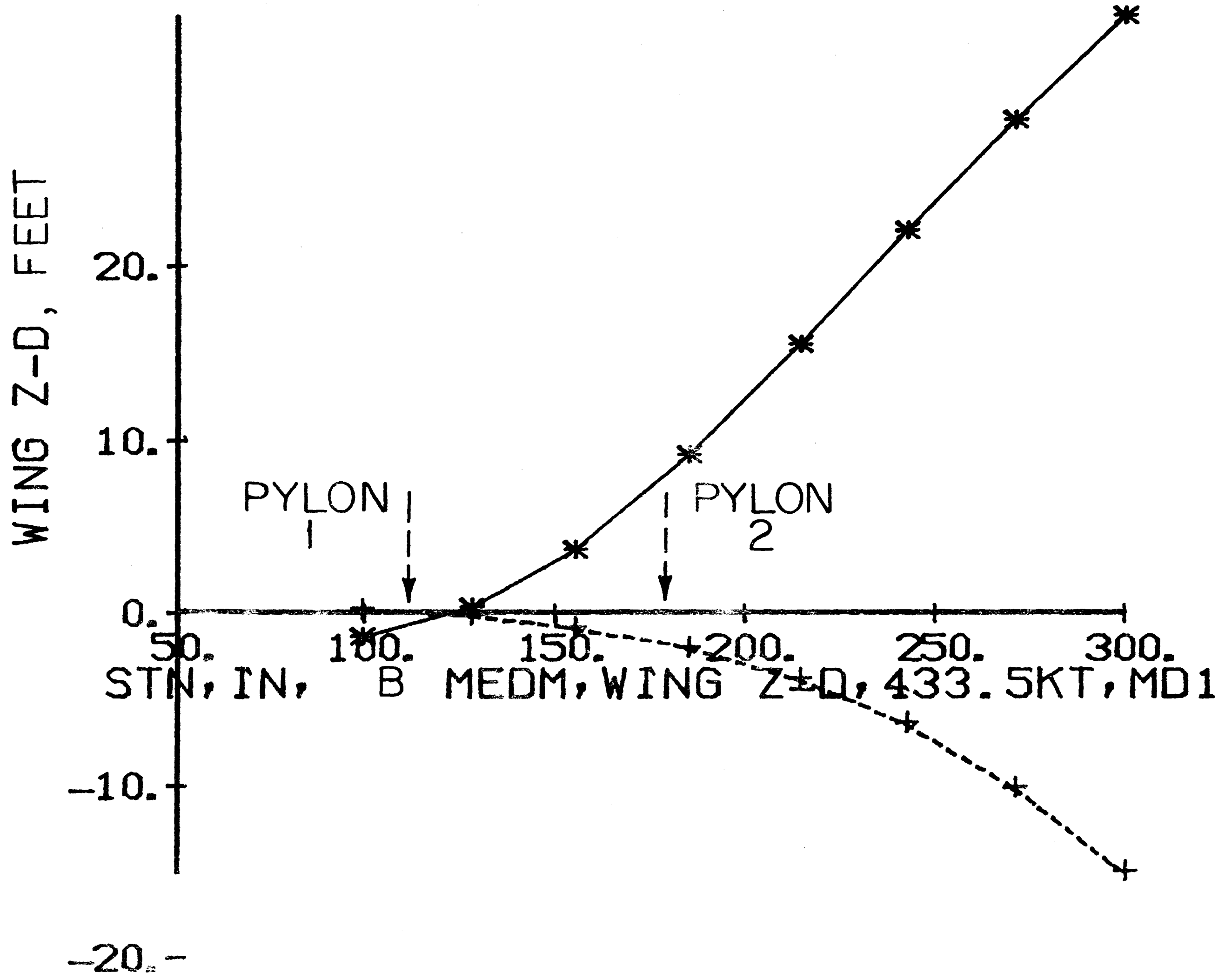


Figure 26(b). (cont.) Torsional rotation of wing.

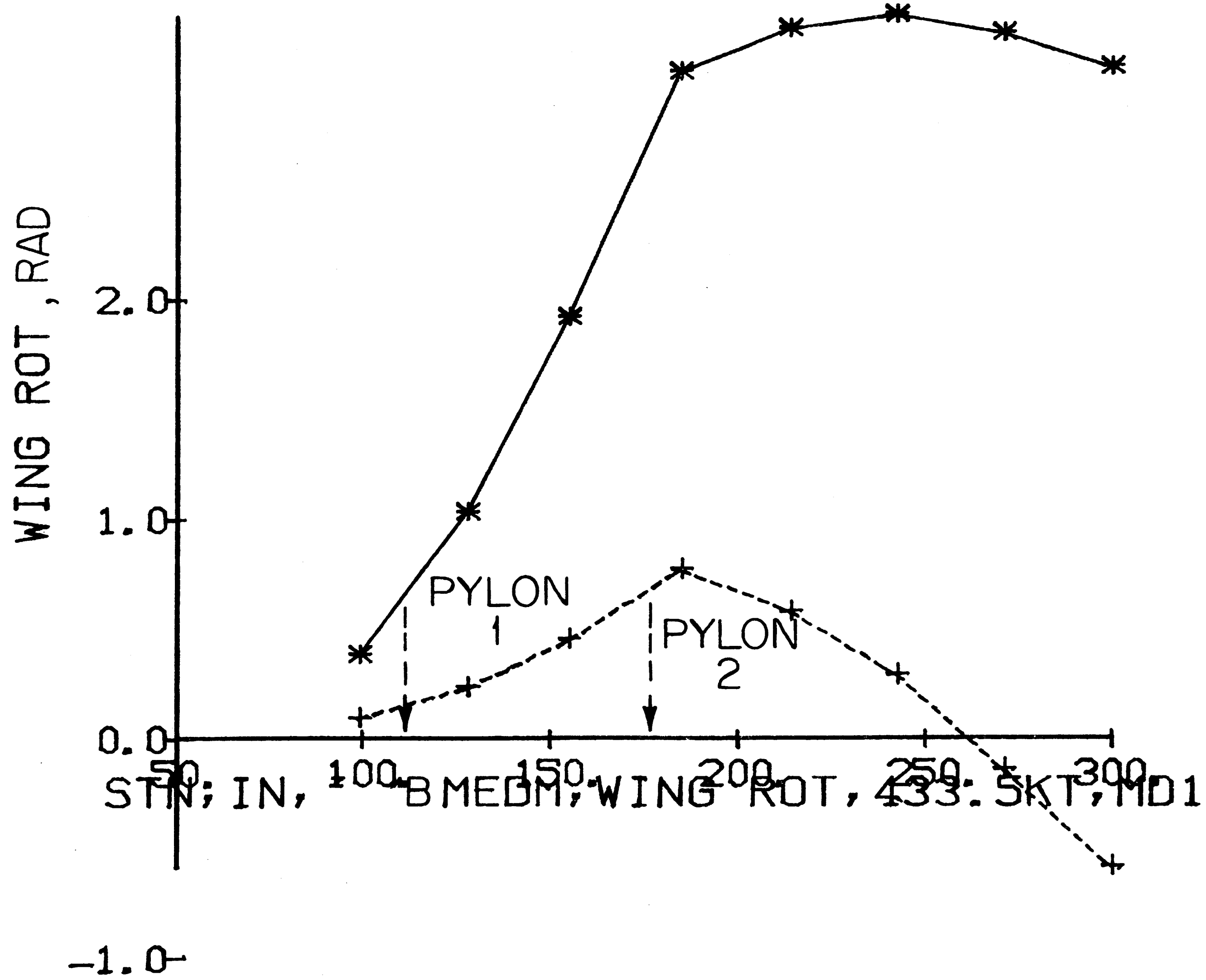
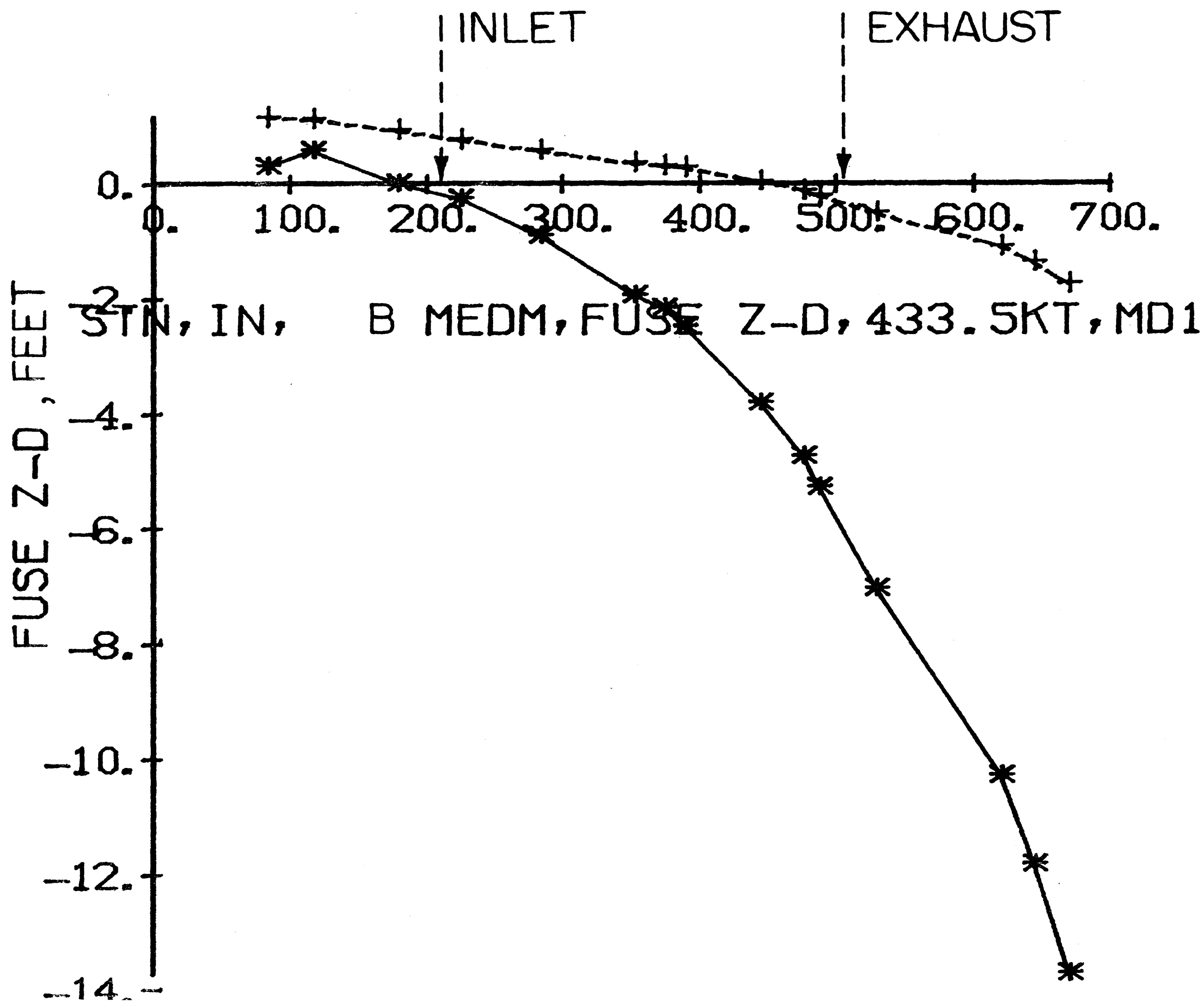


Figure 26(c). (cont.) Vertical displacement of fuselage.



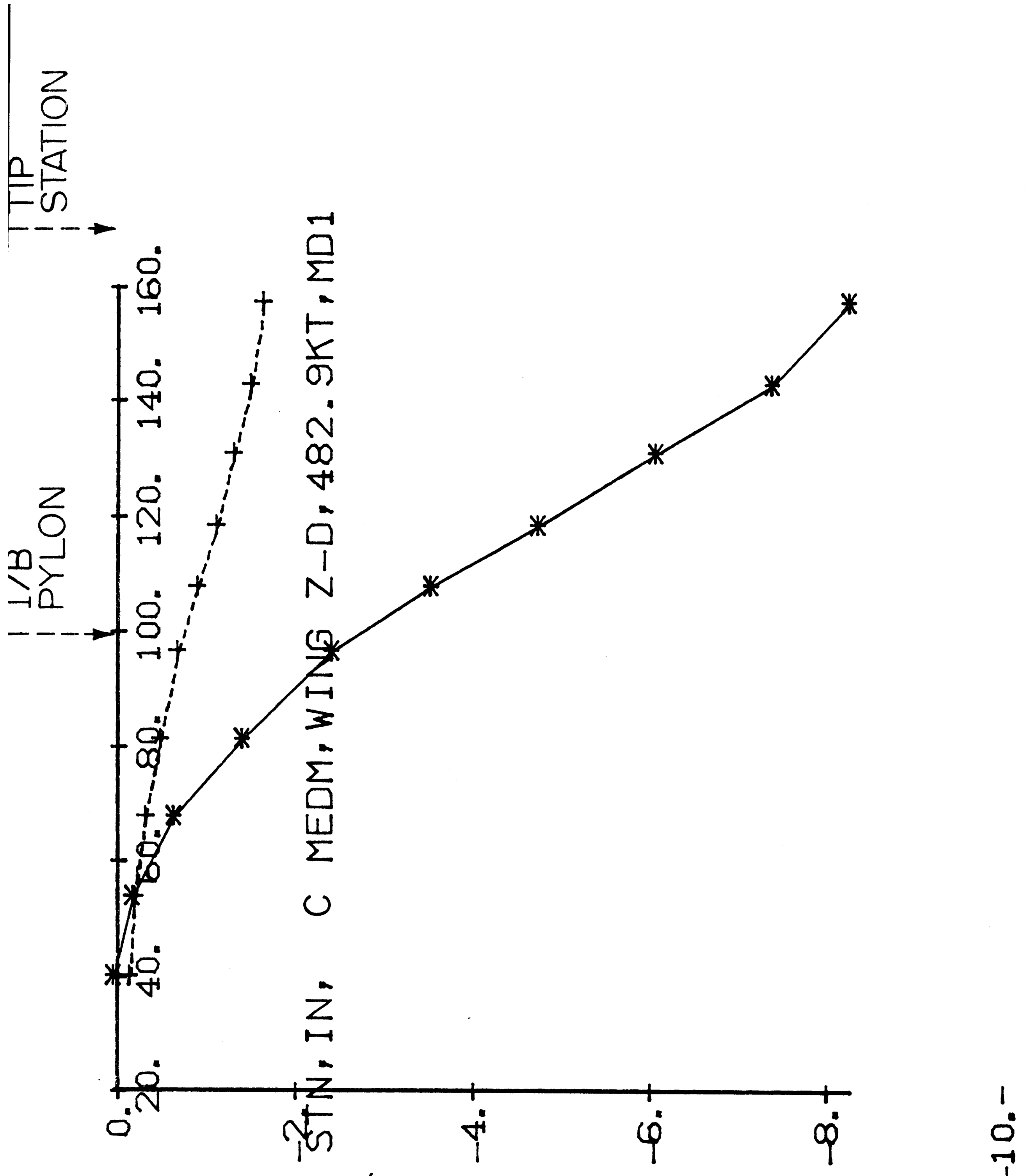


Figure 27(a). Fundamental mode of aircraft type C. 483 knots at sea level. Medium weight = 17,258 lb. Vertical displacement of wing. f_1 = 7.46 hz.

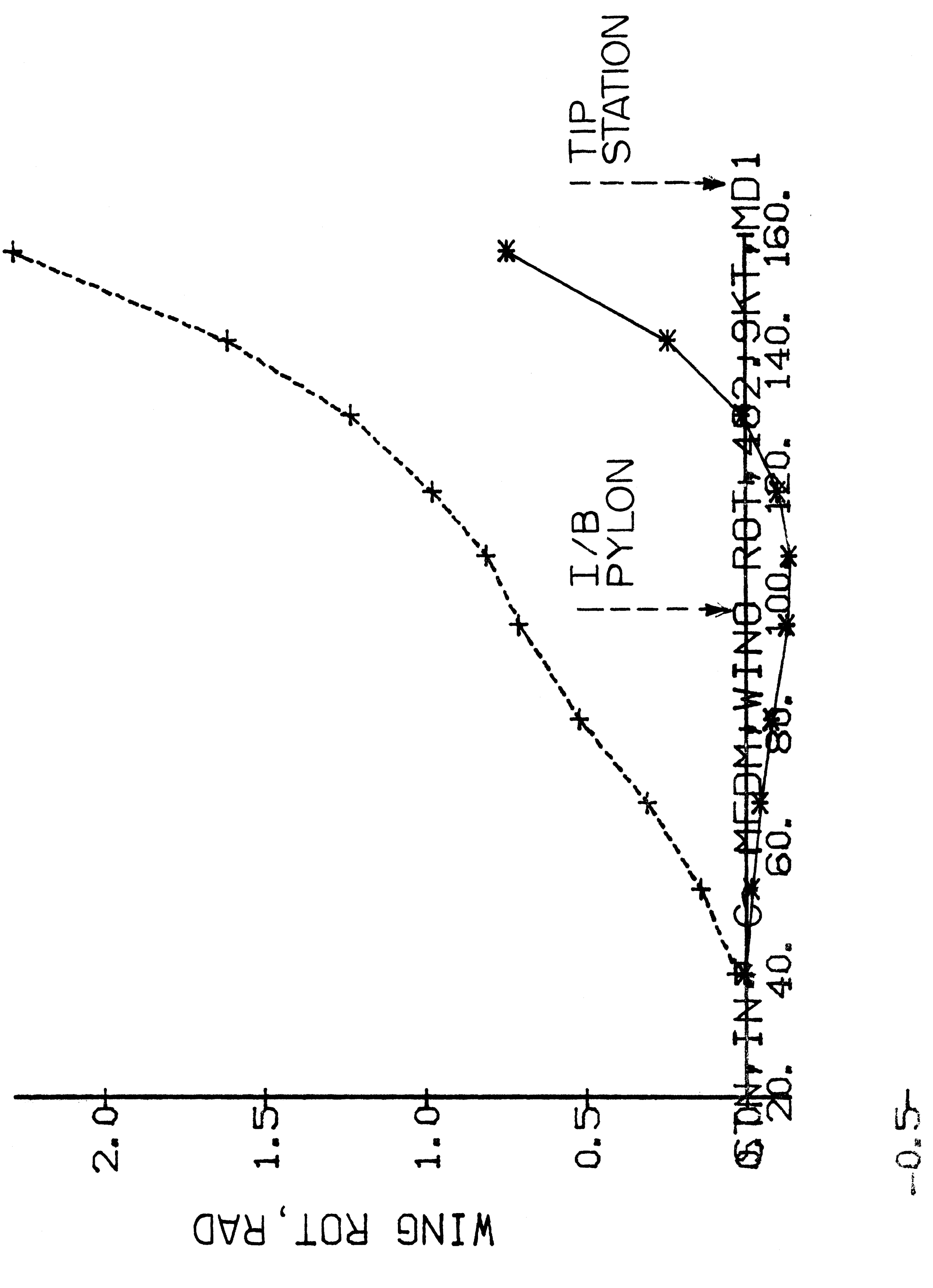


Figure 27(b). (cont.) Torsional rotation of wing.

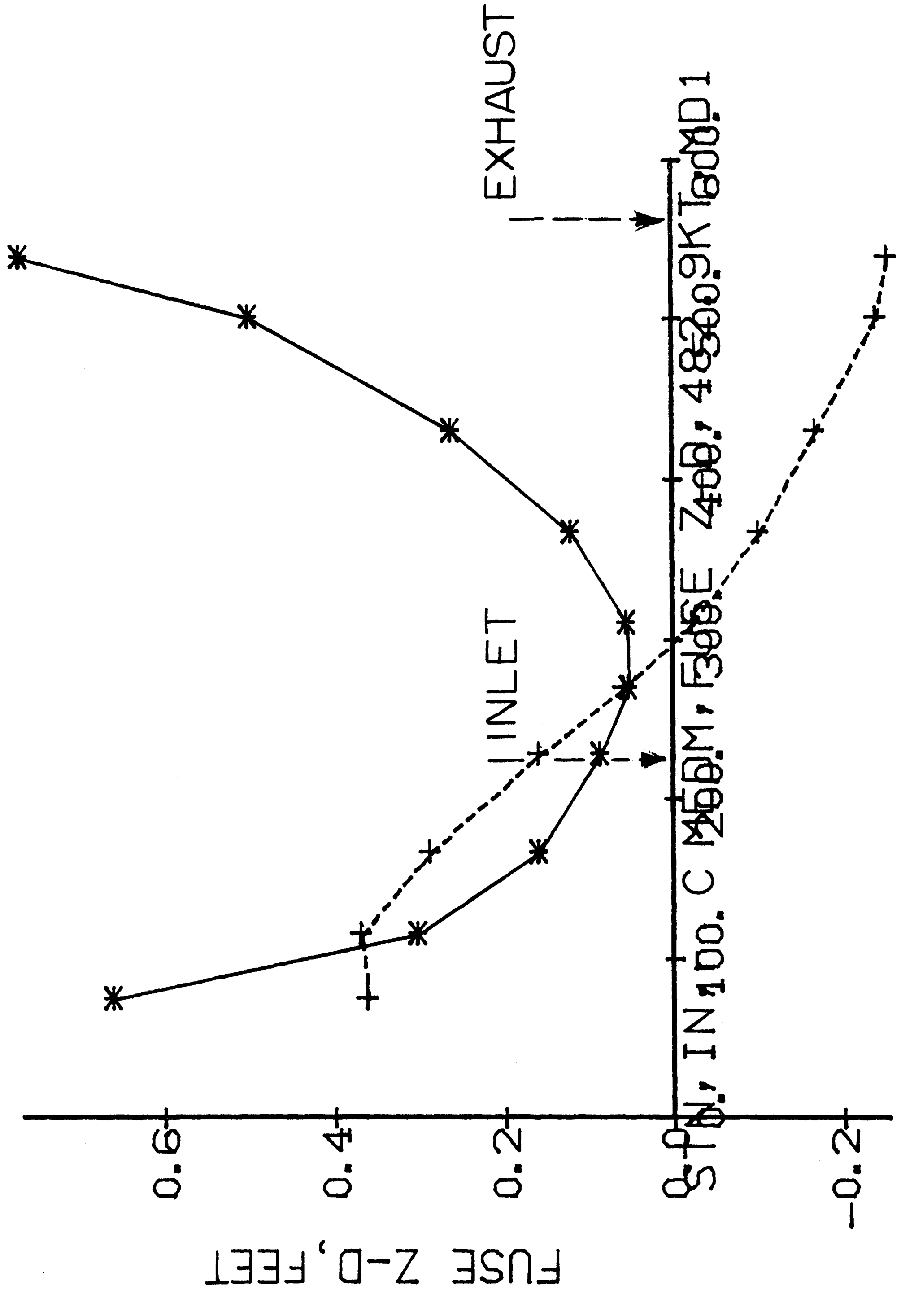


Figure 27(c). (cont.) Vertical displacement of fuselage.

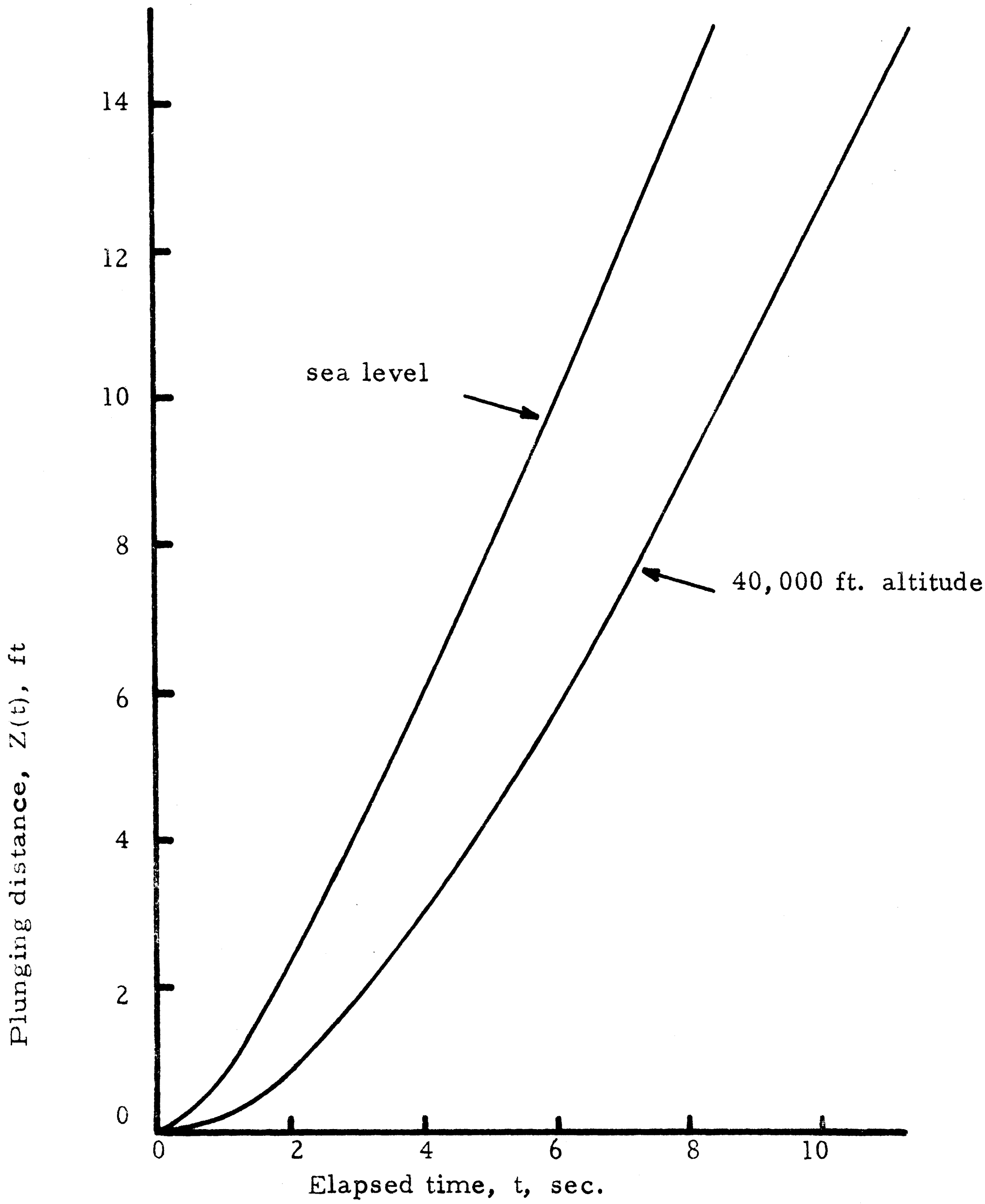


Fig. 28. Sharp-edged gust response for rigid type B aircraft. Gust vertical velocity = 2 ft/sec. Aircraft weight = 54,611 lb.

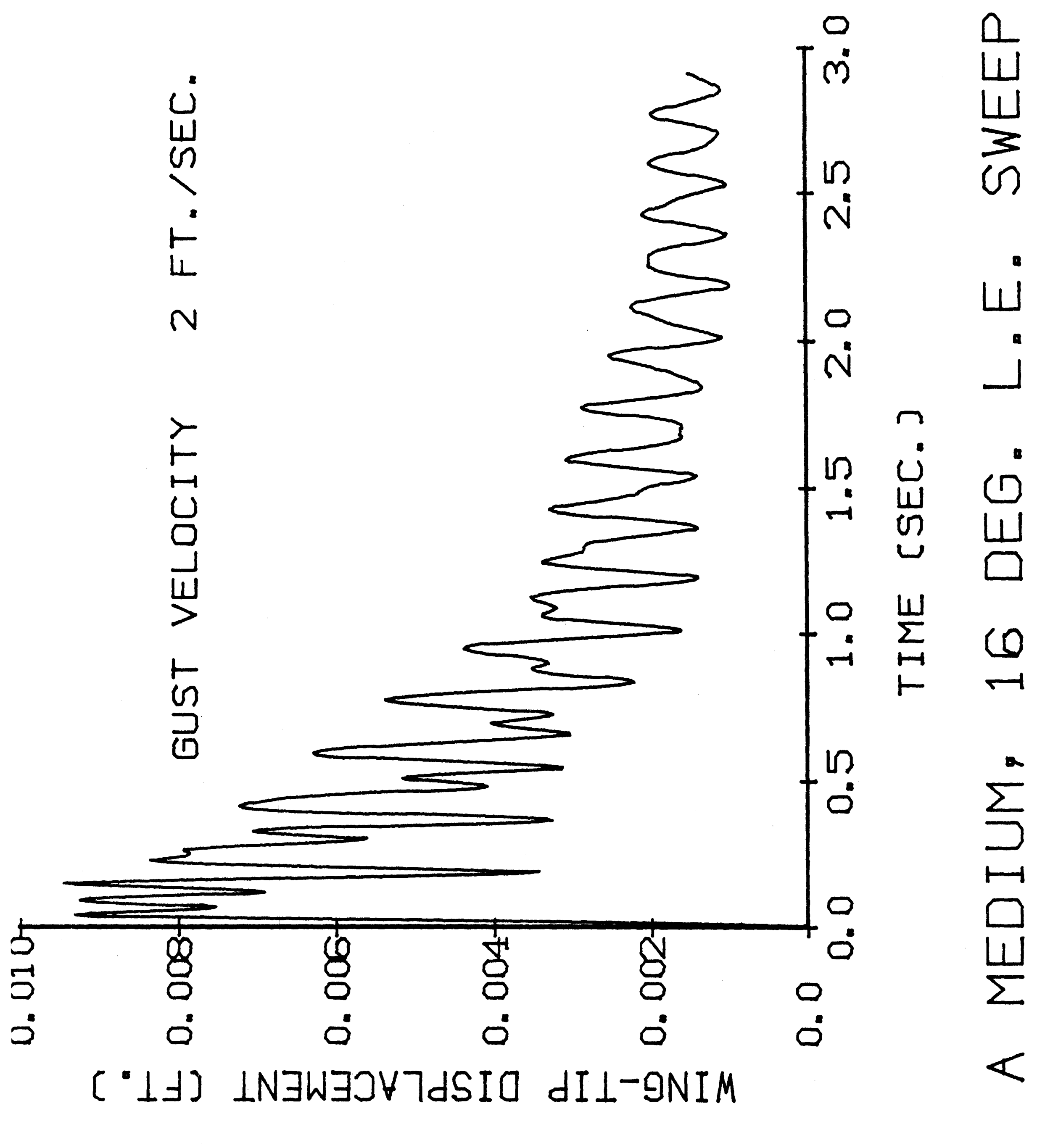


Figure 29. Elastic response to sharp-edged gust, aircraft type A.

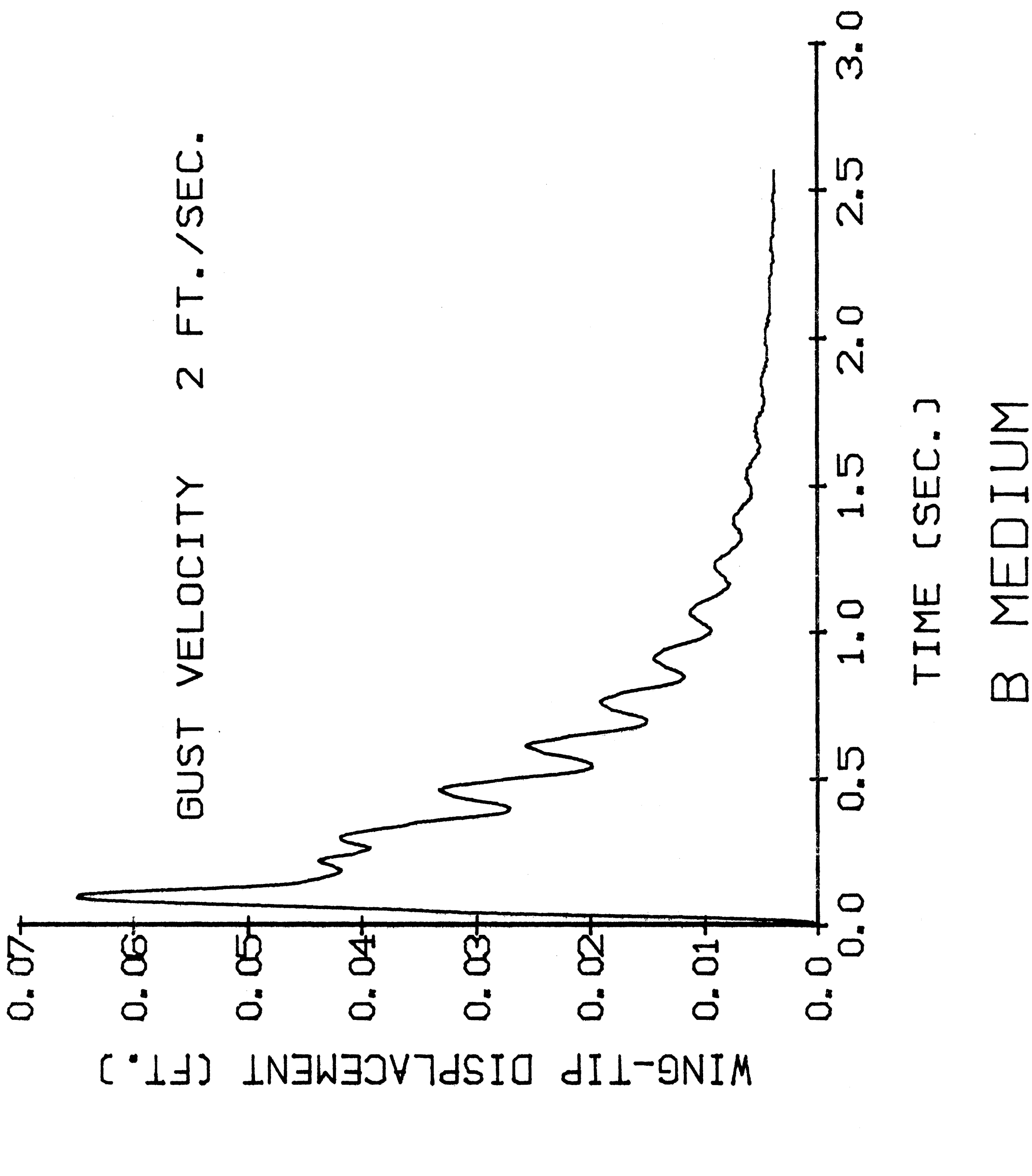


Figure 30. Elastic response to sharp-edged gust, aircraft type B.

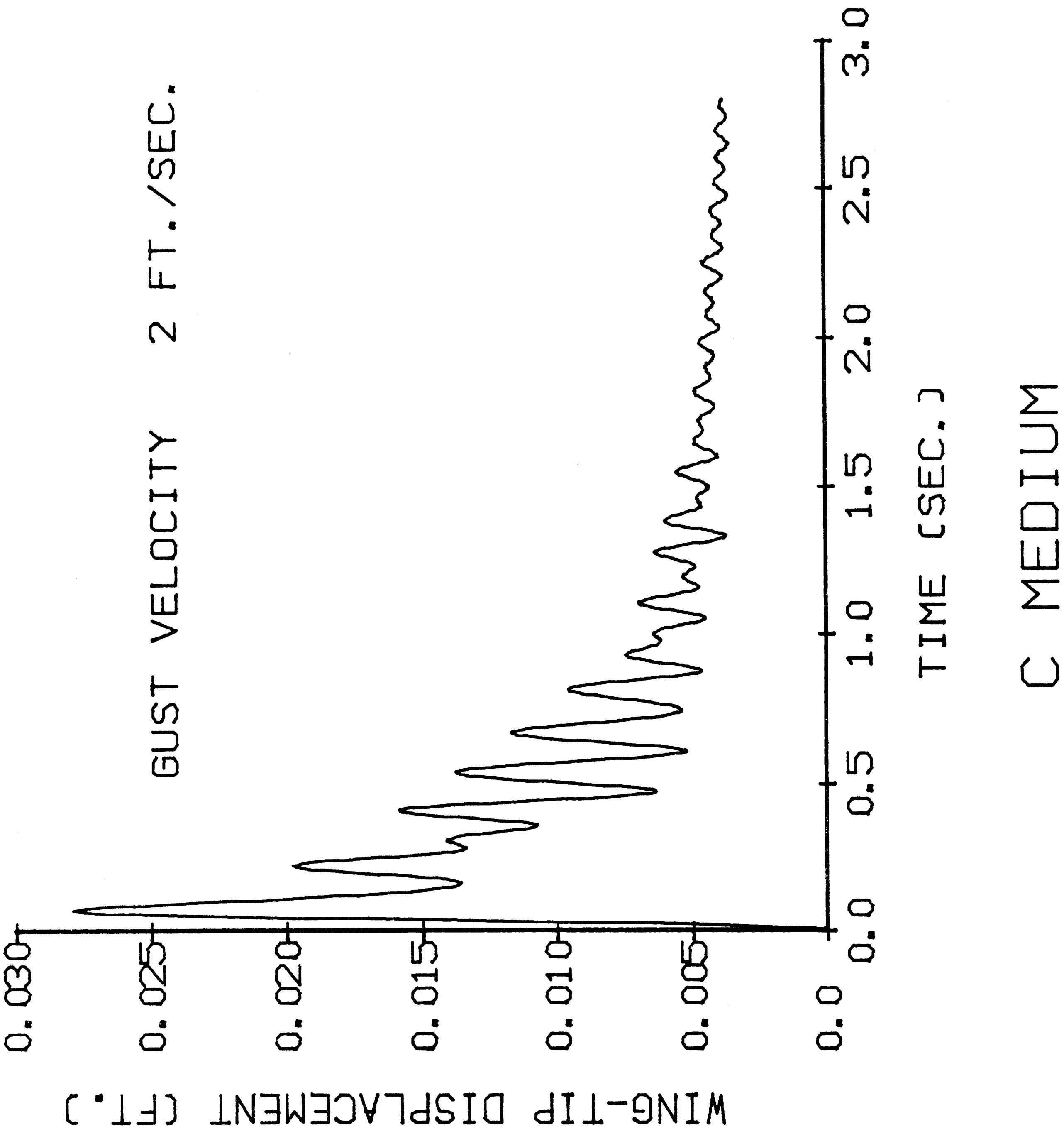


Figure 31. Elastic response to sharp-edged gust, aircraft type C.

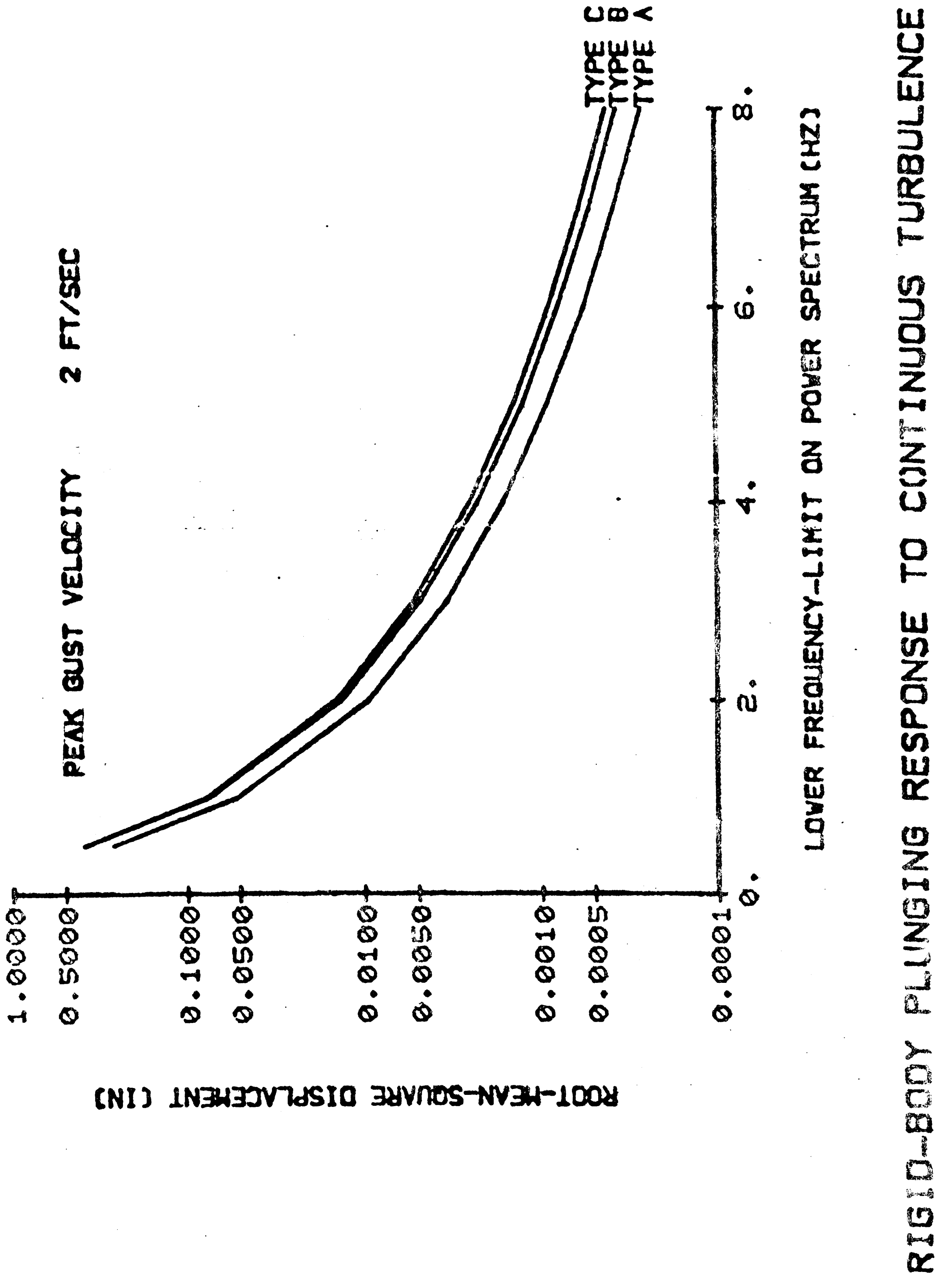


Figure 32. Rigid-body plunging response to continuous turbulence at 30,000 ft.

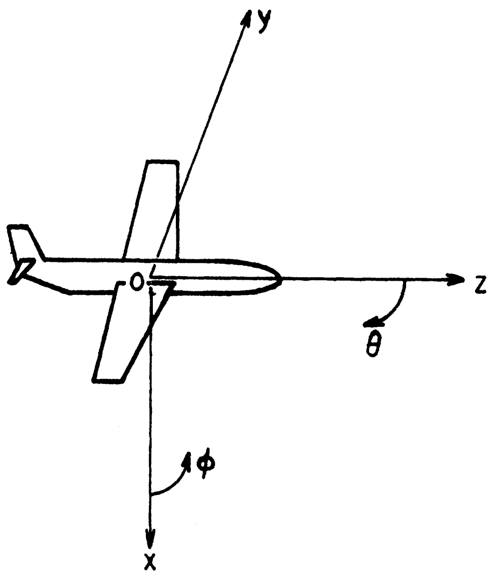


Figure 33. Aircraft coordinate system used for RCS calculation

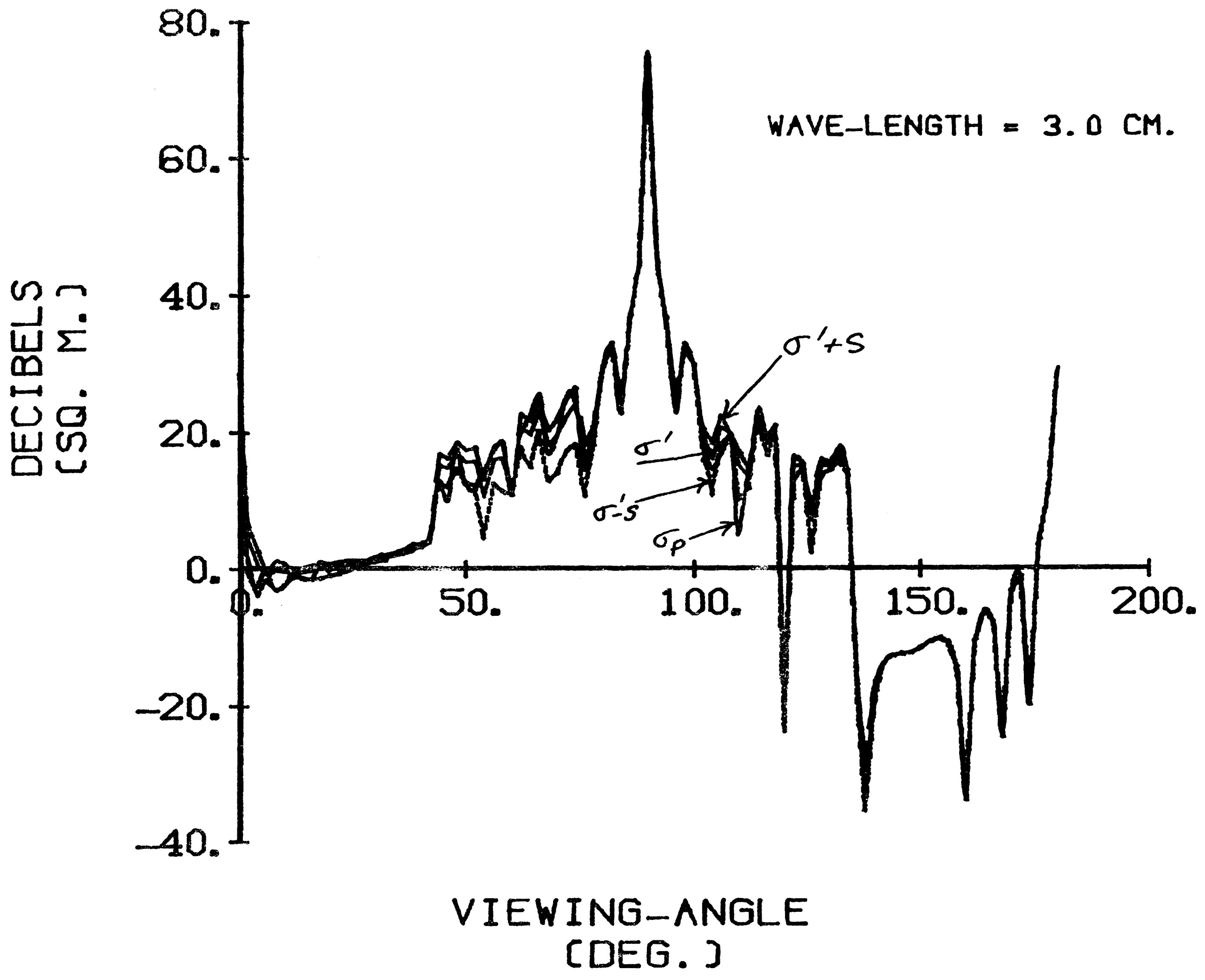


Figure 34(a).

STATIC RADAR-SCATTERING CROSS-SECTION

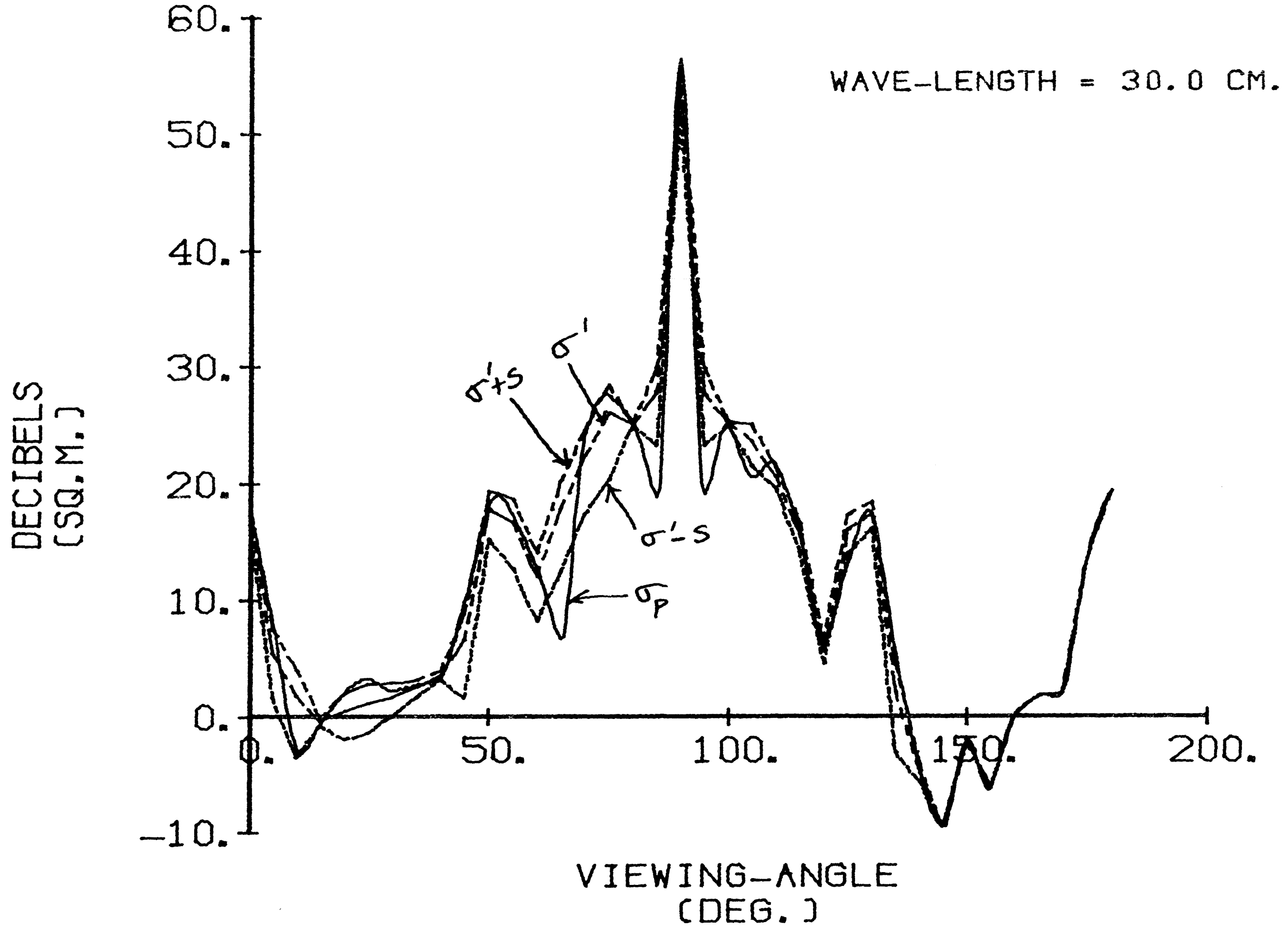


Figure 34(b).

STATIC RADAR-SCATTERING CROSS-SECTION

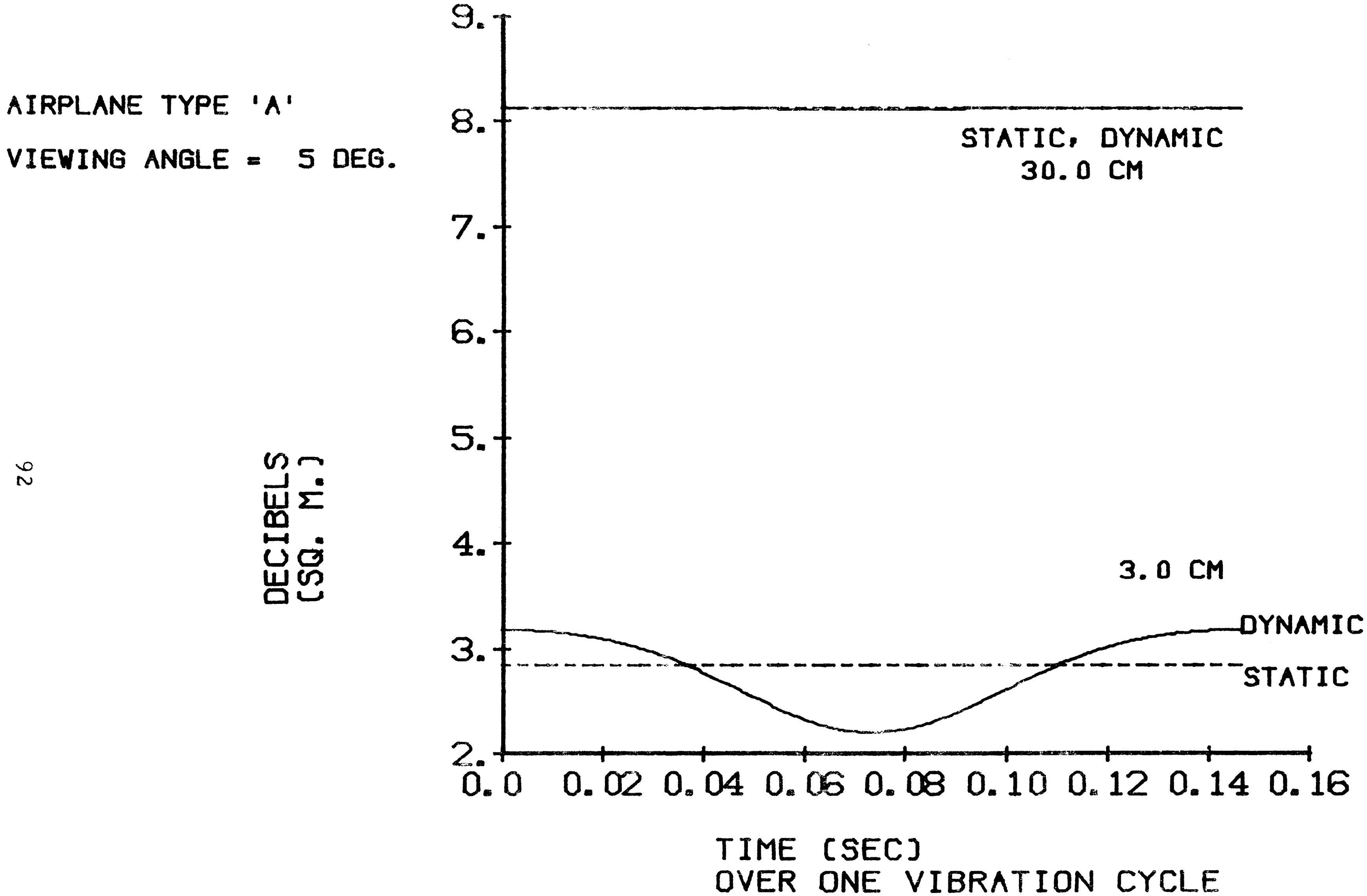


Figure 35(a). DYNAMIC RADAR-SCATTERING CROSS-SECTION

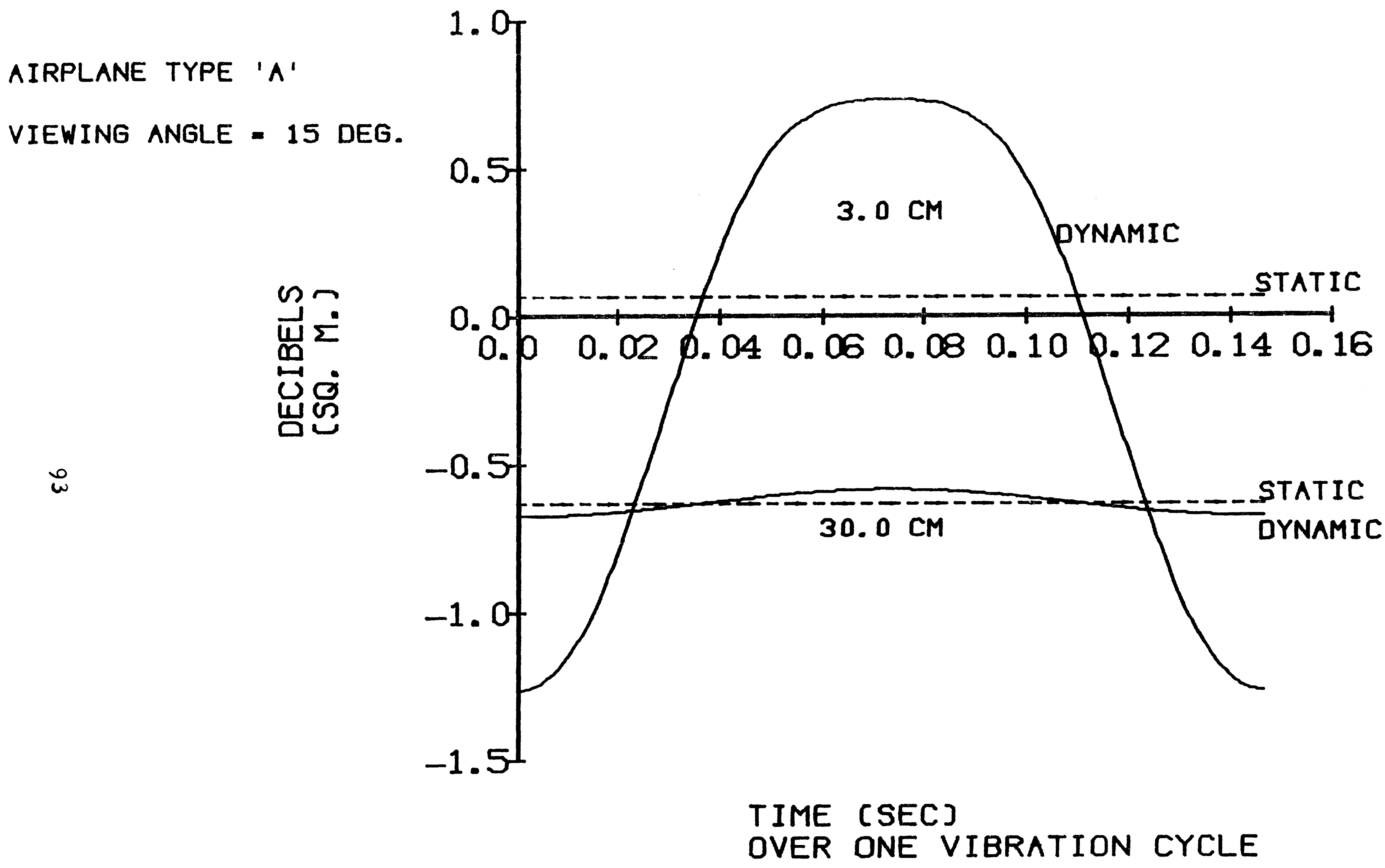


Figure 35(b). DYNAMIC RADAR–SCATTERING CROSS–SECTION

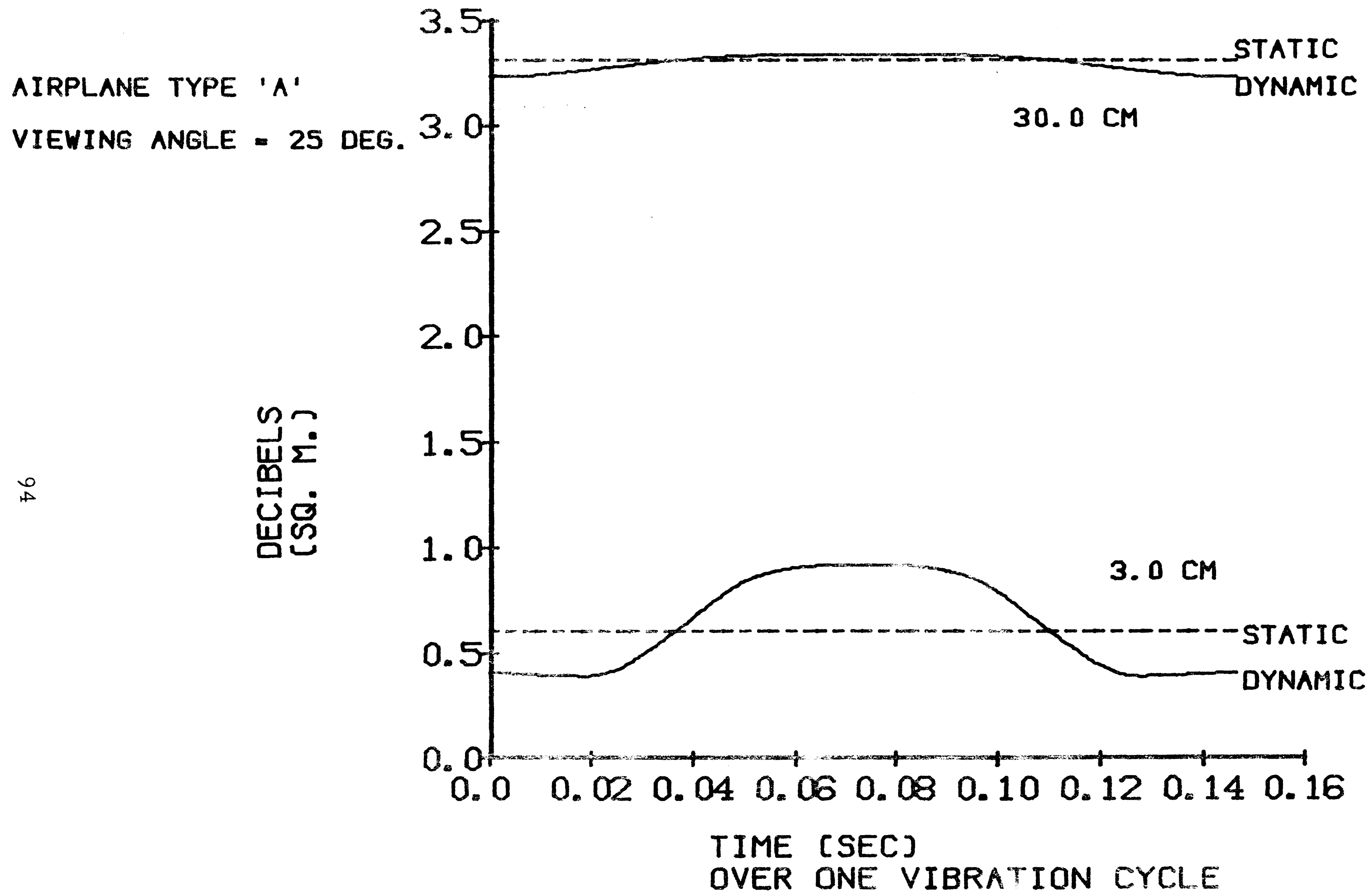


Figure 35(c). DYNAMIC RADAR-SCATTERING CROSS-SECTION

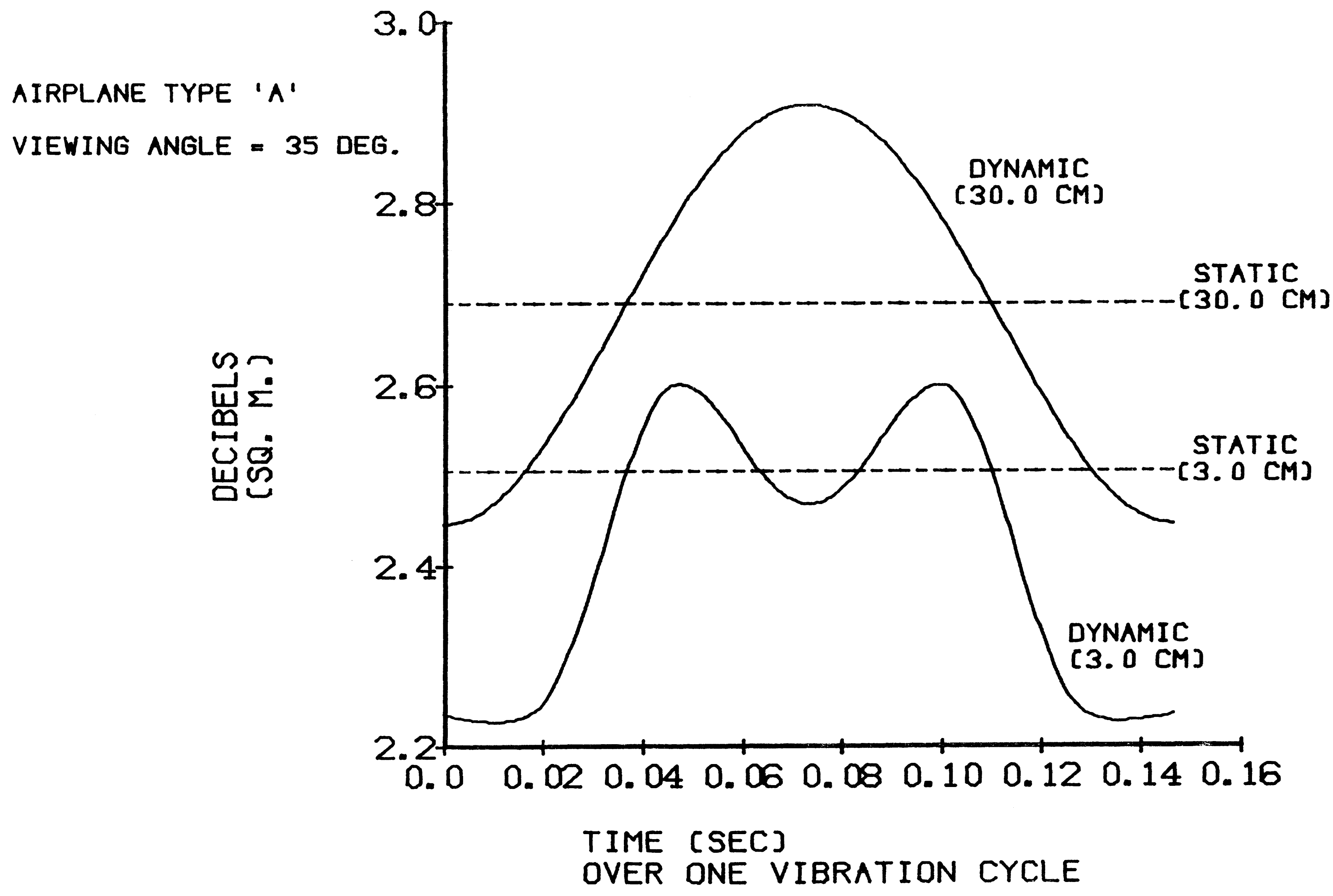


Figure 35(d). DYNAMIC RADAR-SCATTERING CROSS-SECTION

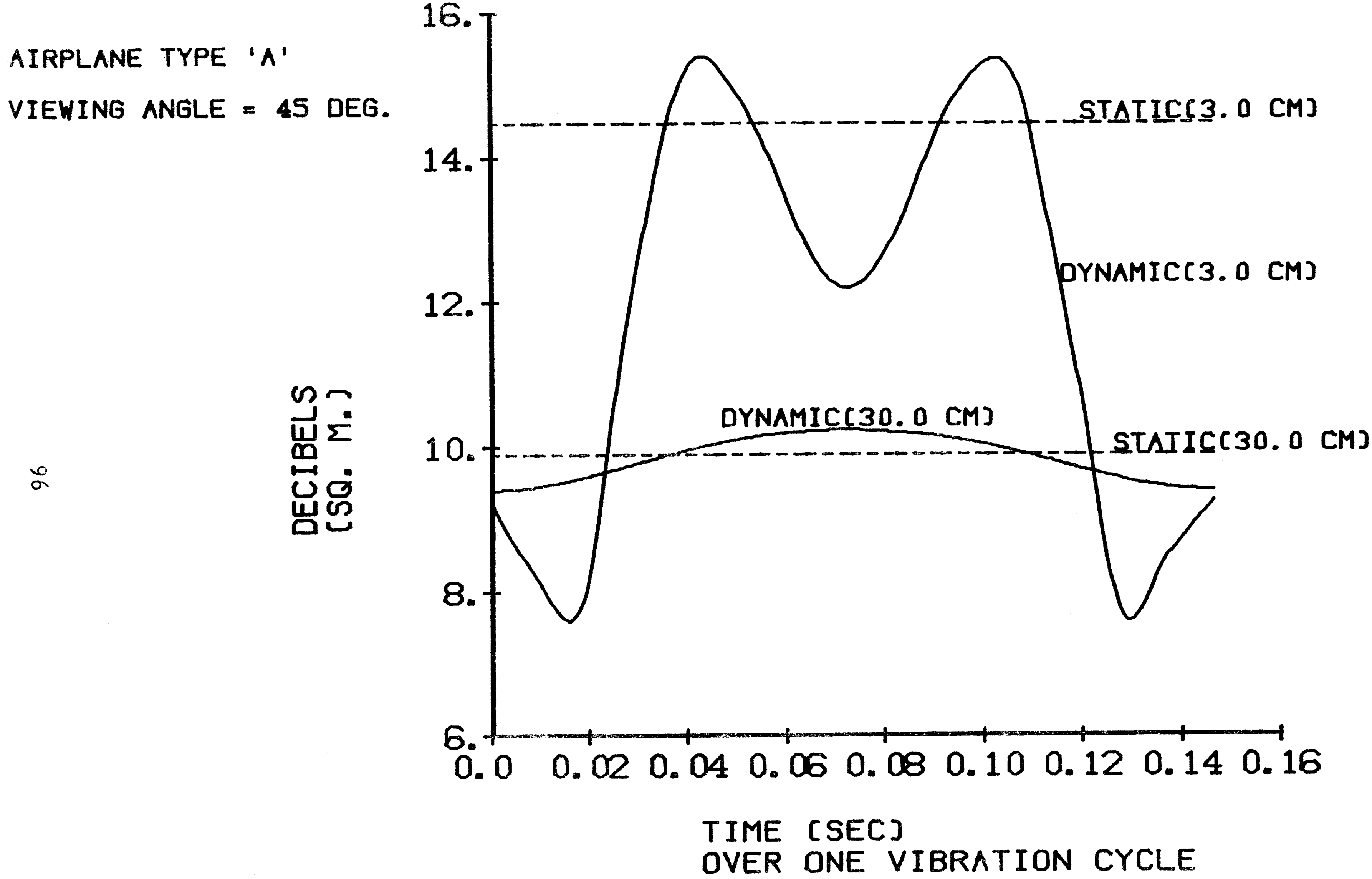


Figure 35 (e). DYNAMIC RADAR-SCATTERING CROSS-SECTION

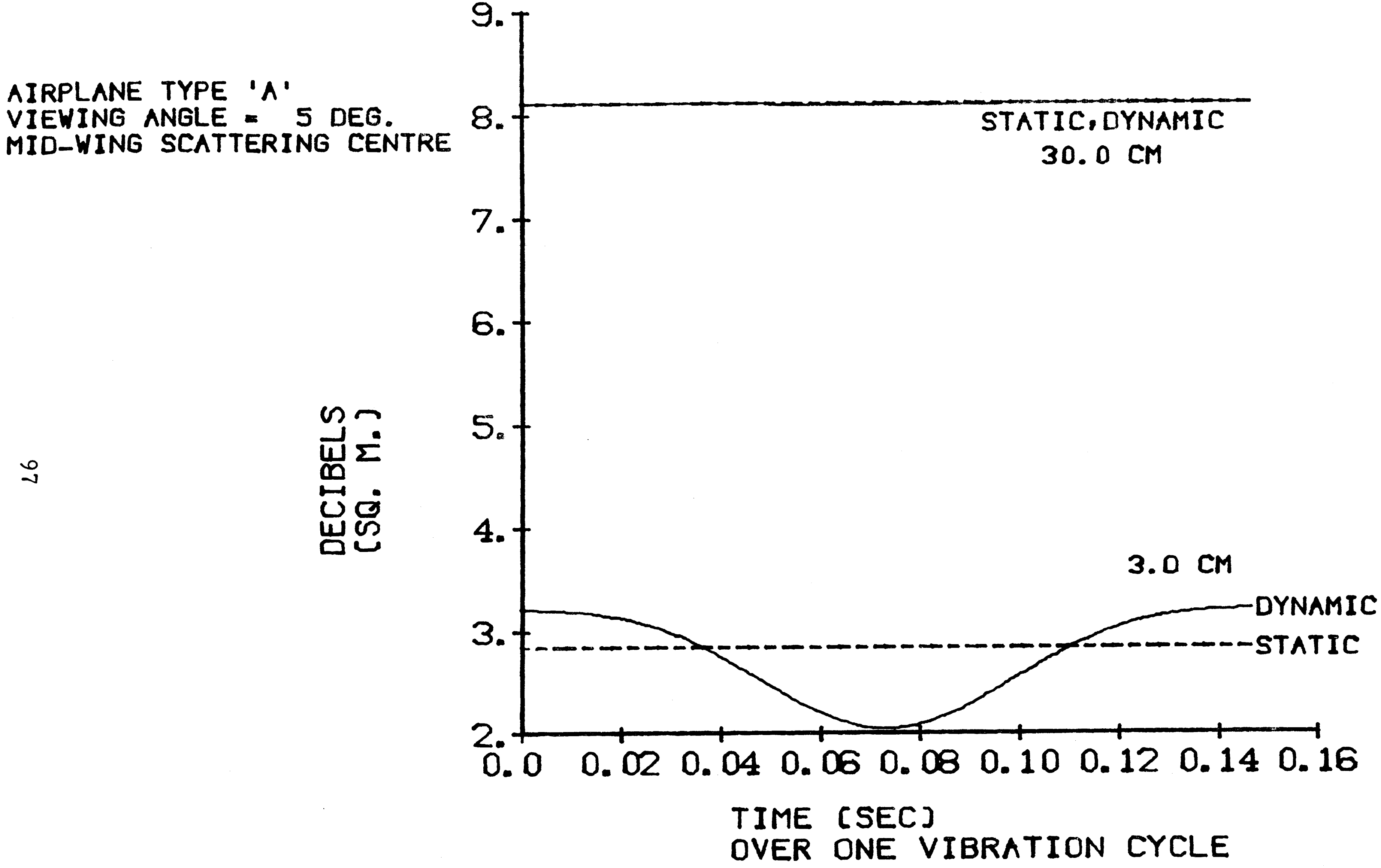


Figure 36(a). DYNAMIC RADAR-SCATTERING CROSS-SECTION

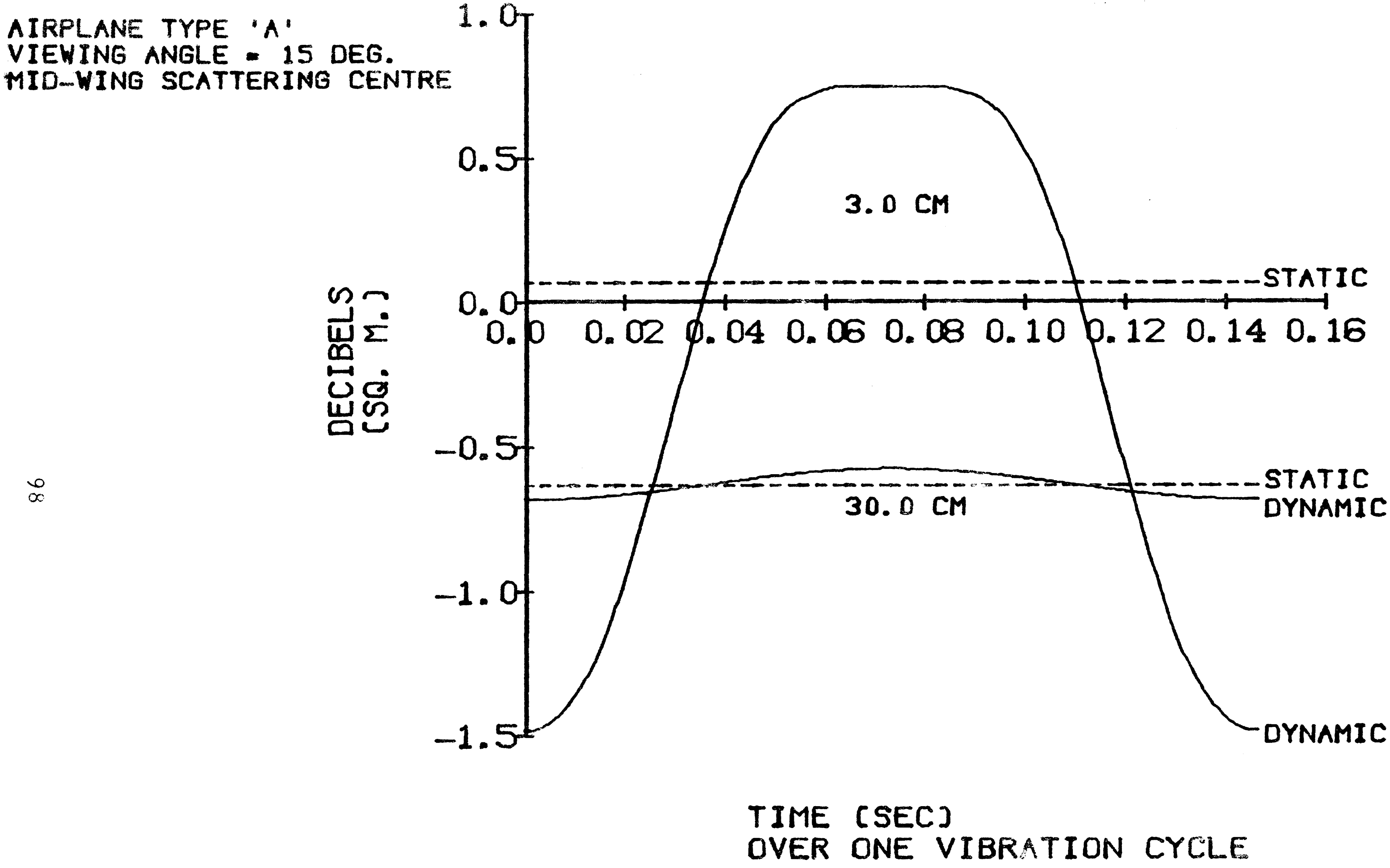


Figure 36(b) DYNAMIC RADAR-SCATTERING CROSS-SECTION

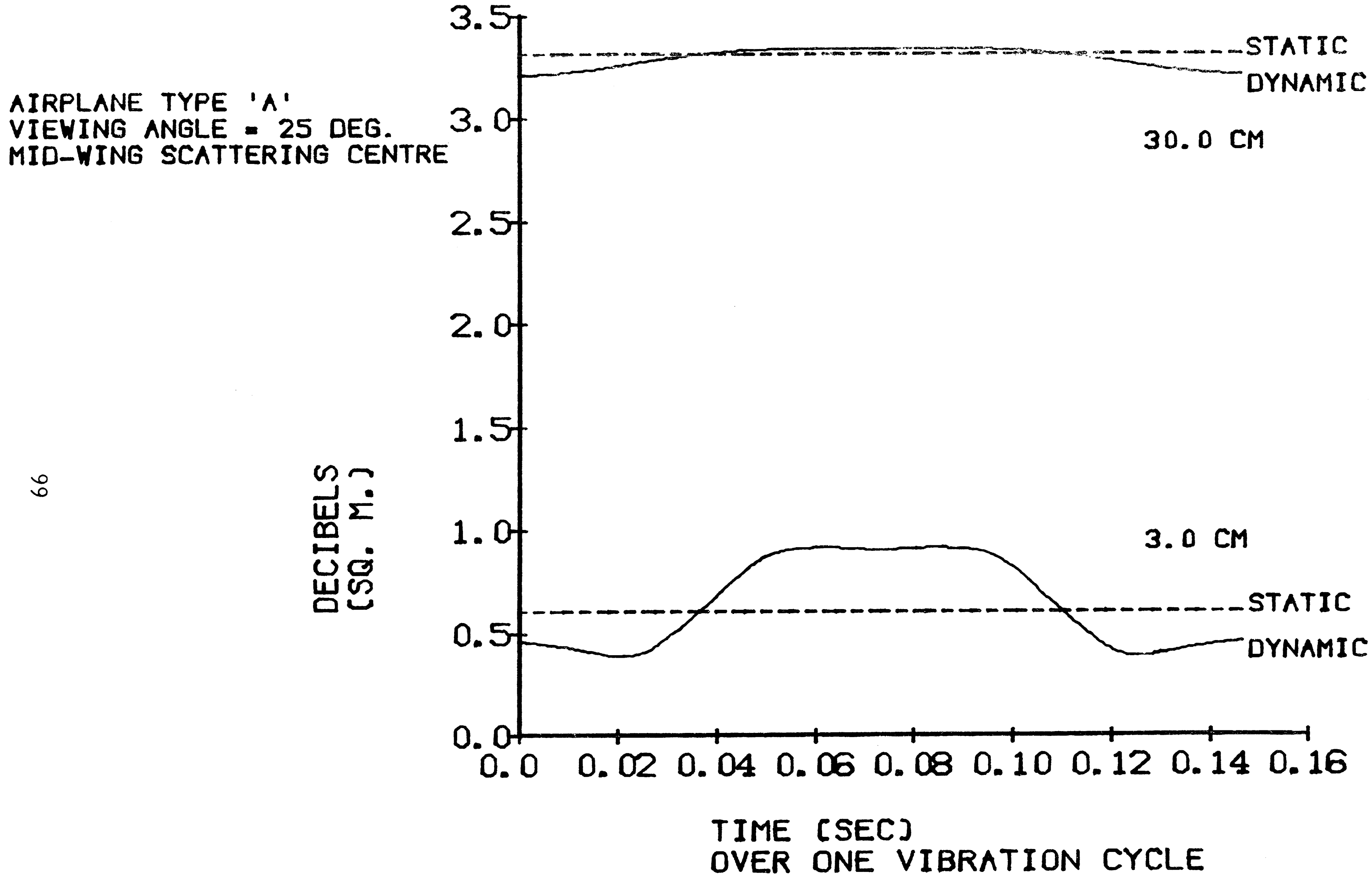


Figure 36(c). DYNAMIC RADAR-SCATTERING CROSS-SECTION

AIRPLANE TYPE 'A'
VIEWING ANGLE = 35 DEG.
MID-WING SCATTERING CENTRE

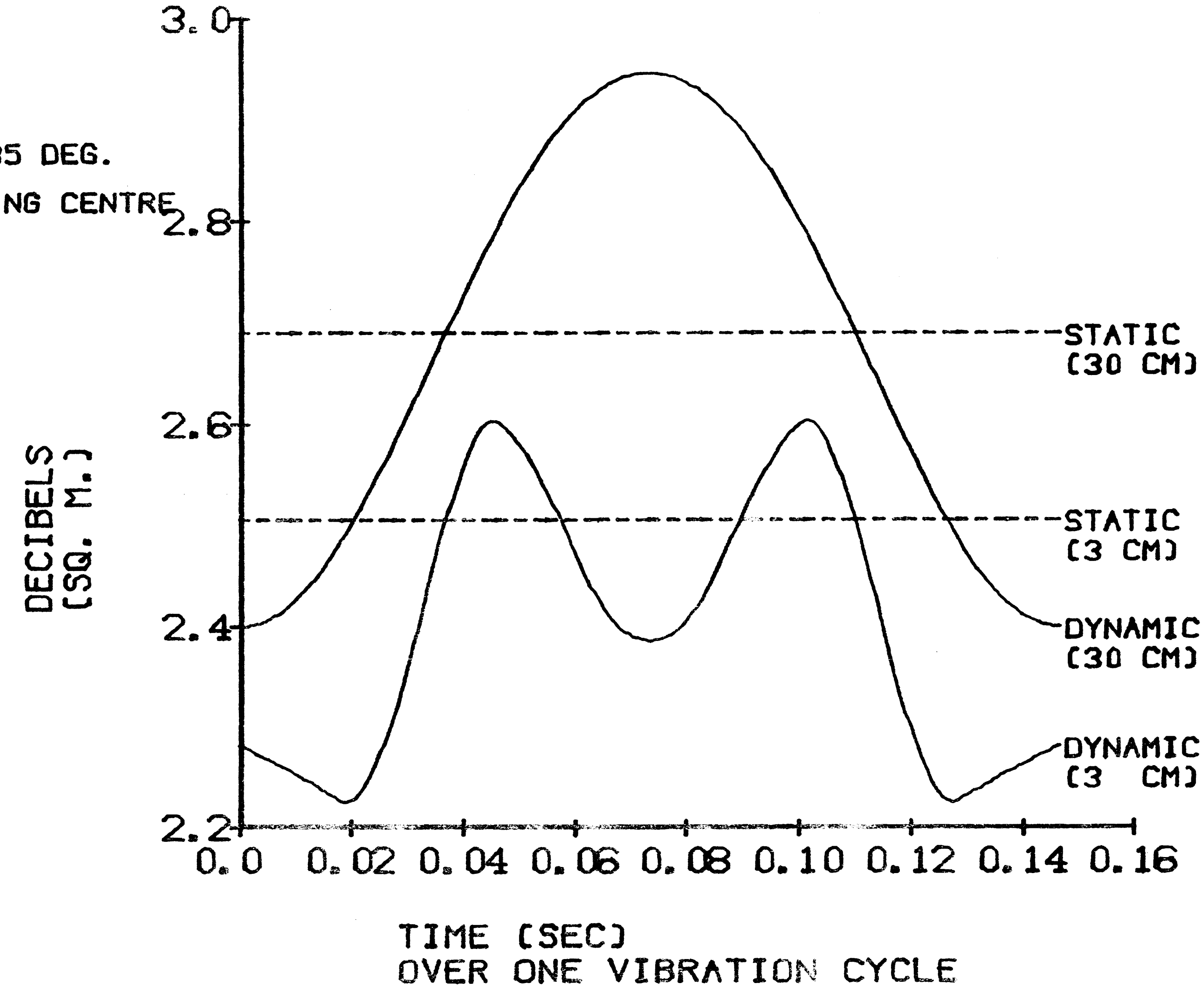


Figure 36(d). DYNAMIC RADAR-SCATTERING CROSS-SECTION

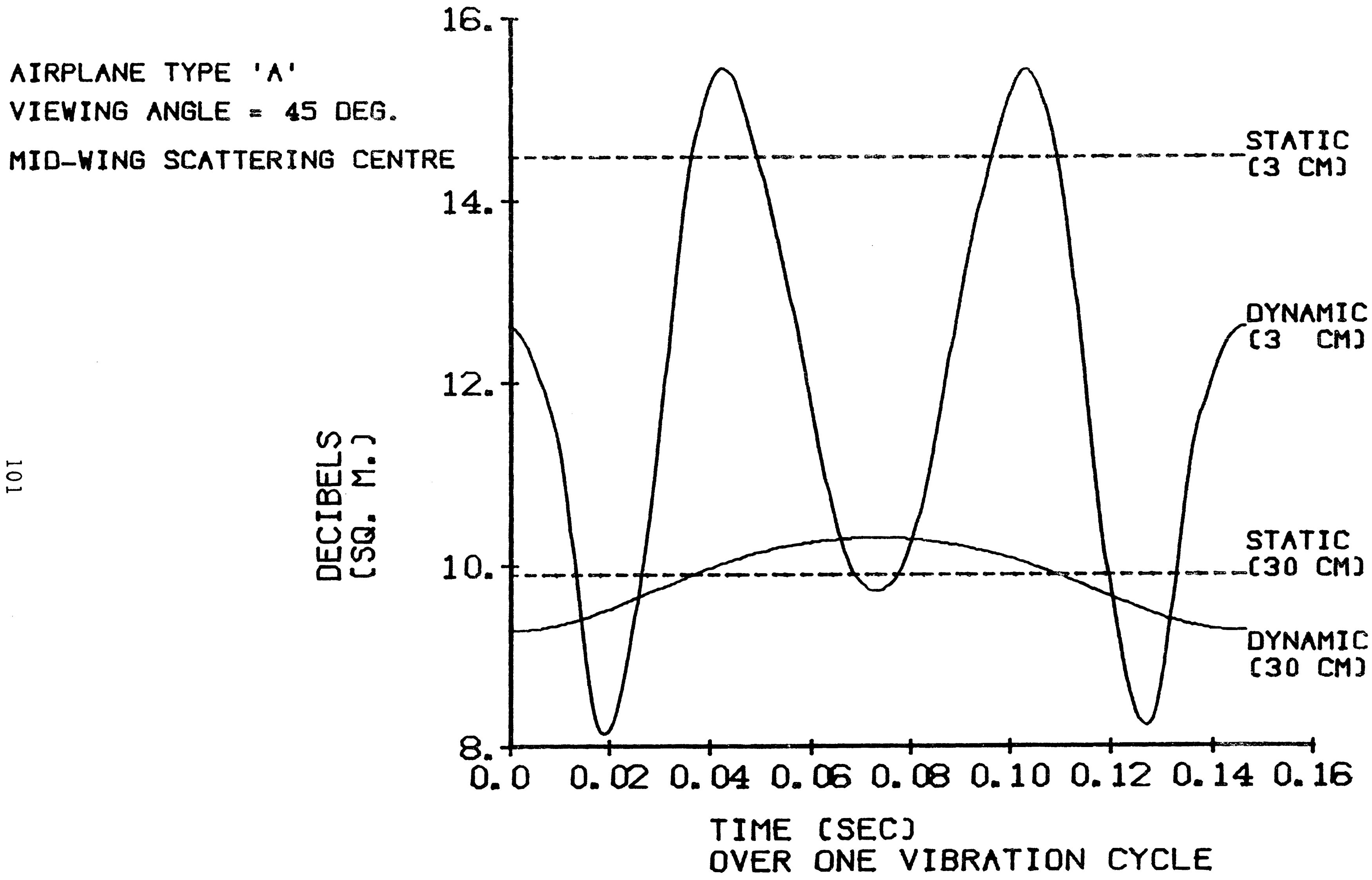


Figure 36(e). DYNAMIC RADAR-SCATTERING CROSS-SECTION

APPENDIX

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3) VIBRATION CHARACTERISTICS OF SPECIFIC AIRCRAFT SUCH AS F4, F-104 F-111
4) EFFECTS OF AGEING ON VIBRATION PROPERTIES OF AIRCRAFT

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SANJAY CORREA
10 OCTOBER 1977

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** WILL OVERPREDICT THE AIRCRAFT RESPONSE. NEVERTHELESS,
** THERE IS A SCALE FACTOR INVOLVED, I.E. THE LENGTH OF
** COHERENCE OVER THE WINGSPAN AND FOR FIGHTER
** AIRCRAFT THIS SCALE MAY BE LARGE ENOUGH TO ALLOW A
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