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RESEARCH MEMORANDUM

INITIAL FLUTTER TESTS IN THE LANGLEY TRANSONIC BLOWDOWN TUNNEL AND COMPARISON WITH FREE-FLIGHT FLUTTER RESULTS

By William J. Bursnall

Langley Aeronautical Laboratory
Langley Field, Va.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

January 7, 1953

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SUMMARY

An experimental wing-flutter investigation was conducted in the Langley transonic blowdown tunnel, which is equipped with a slotted test section, in order to determine the correlation between transonic-wind-tunnel and free-fall flight flutter results. Flutter was obtained in this variable density wind tunnel at several Mach numbers between 0.84 and 1.16 for two unswept, rectangular wings of aspect ratio 7.38 at 0° angle of attack. These wings were scaled models of wings which had fluttered in the course of free-fall flight tests. The very good agreement between the wind-tunnel and flight flutter test results shows the feasibility of conducting flutter tests in transonic wind tunnels with slotted test sections.

INTRODUCTION

The increased use of high-speed aircraft incorporating relatively flexible structural components has resulted in an urgent need for information permitting the prediction of the flutter characteristics of such airplanes. Analytical methods based on purely theoretical considerations, however, are not sufficiently developed at the present time to permit a satisfactory solution of the flutter problem of aircraft having low aspect ratio and swept wings flying in the transonic speed range. Thus, it has been necessary to resort to experimental methods to supplement the results of theoretical investigations.

The experimental information available at present dealing with flutter in the transonic range has been obtained primarily by the free-fall, or bomb-drop, technique and the rocket technique. Although the information obtained by these methods has proved to be of great interest, it has also become apparent that a great deal more information on flutter in the transonic speed range is needed. With the development of the transonic, slotted-throat wind tunnel, it seemed possible that some of these tunnels might be used for flutter work.

The Langley transonic blowdown tunnel, incorporating a slotted test section, is rather uniquely suited for flutter research. In this tunnel, it is possible to set and maintain any prescribed value of the Mach number between 0.80 and 1.45 during a test and independently vary the stagnation pressure from about 30 to 75 pounds per square inch. Since the occurrence of flutter, for a given model, depends upon air density as well as velocity and Mach number, this tunnel operating technique permits the attainment of flutter on a particular model at any number of Mach numbers between 0.80 and 1.45 simply by varying the tunnel pressure.

Before beginning a general investigation of flutter in the transonic blowdown tunnel, however, it seemed necessary first to determine whether flutter results obtained in a transonic tunnel would check those obtained in free air. Flutter tests at 0° angle of attack and in the Mach number range between 0.84 and 1.16 have accordingly been made in the transonic blowdown tunnel of a model which was geometrically and dynamically similar to one for which flutter data obtained by the bomb-drop technique were available (ref. 1). The results of this investigation are presented and analyzed herein.

SYMBOLS

A	aspect ratio including body intercept
A_g	aspect ratio of one wing panel, $l/2b$
a	nondimensional wing-elastic-axis position measured from midchord, positive rearward, $2x_o - 1$
$a + x_\alpha$	nondimensional wing-center-of-gravity position measured from midchord, positive rearward, $2x_1 - 1$
b	semichord of wing, ft
f_h	first bending natural frequency, cps
f_t	first torsion natural frequency, cps
f_α	uncoupled first torsion frequency relative to elastic axis, cps

I_α	polar moment of inertia of wing section about elastic axis, slug-ft^2/ft
l	length of wing panel, ft
M	Mach number
m	mass of wing per unit length, slugs/ft
r_α	nondimensional radius of gyration of wing section about elastic axis, $\sqrt{I_\alpha/mb^2}$
V	velocity, fps
x_o	distance of elastic axis of wing section behind leading edge, fraction of chord
x_1	distance of center of gravity of wing section behind leading edge, fraction of chord
Λ	sweep angle, deg
λ	taper ratio, Tip chord/Root chord
μ	ratio of mass of wing to mass of a cylinder of air of a diameter equal to chord of wing, both taken for equal length along span, $m/\pi\rho b^2$
ρ	air density, slugs/cu ft
ω	circular frequency, $2\pi f$, radians/sec

Subscripts:

e	experimental values
R	calculated values based on two-dimensional incompressible-flow theory with account taken of mode shape, (two degrees of freedom)
stnd	based on standard sea-level conditions

APPARATUS AND TESTS

Wind Tunnel

All tests of the present investigation were conducted in the Langley transonic blowdown tunnel. This tunnel is equipped with a slotted transonic test section permitting the operation of the tunnel through and above sonic speed. A plan view of the tunnel, with model installed, and a cross section of the octagonal test section are shown in figure 1.

Air flow through the tunnel is controlled by the simultaneous operation of three plug valves which regulate the flow of air from a high-pressure reservoir. By placing an orifice downstream of the test section it is possible to choke the tunnel at the orifice as well as at the minimum section upstream of the test section. For this condition, a constant test-section Mach number is maintained while varying the stagnation pressure. The test-section velocity, however, will vary with the decreasing temperature (resulting from expansion of air in the high pressure reservoir) as the test run progresses. The stagnation temperature at the start of flutter in the present tests ranged between approximately -80° F and $+15^{\circ}$ F, depending on the initial tunnel conditions and the length of the test run. Orifices of various areas permit constant Mach number tests in the range from 0.80 to approximately 1.45. It has been found that the stagnation pressure for orifice choke is approximately 30 pounds per square inch for all orifices. The tunnel, however, may be operated at stagnation pressures up to 75 pounds per square inch permitting variation of the test-section density and speed for a given Mach number. Valve closing and consequently cessation of the test run is accomplished in approximately 1/2 second.

Model and Support System

A 0.279-scale model of the bomb and wing tested in the investigation reported in reference 1 was constructed for the wind-tunnel tests (fig. 2). The wing (fig. 3) was unswept and untapered in plan form with a 16.4-inch span and aspect ratio of 7.38. The wing was tapered in thickness ratio from an NACA 65A004 airfoil section at the root to an NACA 65A002 airfoil section at the tip.

Since the flight-test wings, which were made of dural, fluttered at values of air density which were relatively low compared to those obtained in the tunnel, it was necessary to construct the wind-tunnel wings of solid steel in order to maintain similar values of the mass-ratio parameter μ. The mass density of steel is about three times that of dural. The Reynolds number of the wind-tunnel model, therefore, will be about the same as that of the flight model if flutter occurs at the same values of velocity and mass-ratio parameter μ in the two cases. Because the ratios of the moduli of elasticity to material density are about the same for steel and dural, the pertinent natural frequencies of the two models in still air differed only by the scale factor. The center-of-gravity and elastic-axis positions of the flight and wind-tunnel wings were approximately the same because of homogeneous construction and geometric similarity. Two model wings were employed in the investigation. The physical properties of these wings are given in table I.

The wing was rigidly mounted in the 3-inch-diameter cylindrical body (fig. 3) by means of close fitting filler blocks and two rows of $\frac{3}{8}$-inch bolts spaced at $1\frac{5}{8}$-inch intervals. The scaled ogival nose of the wind-tunnel body (fig. 3) could not be used throughout the Mach number range of the tests. Because of the presence of tunnel walls, the bow shock waves formed just ahead of the nose are reflected back onto the model at certain Mach numbers. In order to eliminate this condition, some of the tests were conducted with a fuselage nose which extended into the subsonic flow region of the tunnel entrance cone where it was supported by guy wires. The extension section added $94\frac{1}{2}$ inches to the over-all length of the body. Both the original and extended-nose configurations are shown in figure 1.

The back end of the test body was supported in the tunnel by a sting attached to the angle-of-attack mechanism. The fuselage with the original nose weighed approximately 70 pounds and, acting as a cantilever beam, had a natural frequency of the order of 30 cycles per second. The nose extension added 106 pounds to the weight of the system, but no measurements were made of the natural frequency of this system. It is believed, however, that the large value of the ratio of flutter frequency to support system natural frequency (flutter frequency ranging from approximately 105 to 166 cycles per second) prevented any significant coupling between the support system and the wings. Normal acceleration and rate-of-roll measurements during the free-flight bomb tests showed that the flight wings could also be considered as attached to an essentially rigid body.

Tests and Measuring Equipment

The test program consisted of the determination of the flutter speed and flutter frequency of the wings at 0° angle of attack for various Mach numbers in the range between 0.84 and 1.16. Although the transonic blowdown tunnel is capable of operating at Mach numbers up to 1.45, flutter was not obtained above $M = 1.16$ because the elastic and mass properties of the wing were such that prohibitively high stagnation pressures would have been required for flutter at higher Mach numbers. The test procedure followed in obtaining flutter at any particular Mach number was to increase the stagnation pressure above that for tunnel choke until the model was seen to flutter by an observer looking through a porthole in the side of the tunnel. The tunnel was then stopped immediately in an effort to prevent destruction of the model.

The models were instrumented with electrical strain gages on the surfaces of the wings near the root as shown in figure 3. The gages were so oriented that their output gave a time history of the frequency and amplitude of both the bending and the torsion oscillations of the wing. In order to permit the determination of tunnel conditions corresponding to the start of flutter, measurements throughout a test were simultaneously recorded of the wing oscillations, tunnel stagnation pressure and temperature, and test-section static pressure by means of a multichannel recording galvanometer. A sample test record is shown in figure 4. The start of flutter is clearly indicated by the rapid build-up of the oscillations to a large-amplitude sinusoidal form. Although there are some random oscillations of relatively small amplitudes prior to the start of flutter, the distinction between these oscillations and those corresponding to flutter is quite sharp.

RESULTS AND DISCUSSION

In the two bomb-drop tests reported in reference 1, flutter of two similar wings was attained at different Mach numbers by starting the drops at two different altitudes. Both panels of one wing fluttered at a Mach number of 0.85 whereas, in the other drop, one panel fluttered at $M = 1.03$ and the other at $M = 1.07$. Scaled models of the flight wings fluttered in the present wind-tunnel investigation at several Mach numbers in the range between approximately 0.84 and 1.16. The results of the wind-tunnel tests are summarized in table II.

The wind-tunnel and flight flutter results are compared in figure 5(a) on the basis of the speed ratio V_e/V_R as a function of Mach number, where V_e is the experimental flutter speed and V_R is the calculated reference flutter speed. The reference flutter speeds used in reference 1, and for the present tests, are determined from calculations which are based on two-dimensional incompressible-flow theory, and include the mode shapes of the first bending and first torsion uncoupled modes (refs. 2 and 3). For the present tests, the mode shapes used were the uncoupled modes of a uniform cantilever beam as given in reference 3. The use of a reference flutter speed as the basis of comparison of the flight and wind-tunnel tests is convenient because of the different temperatures encountered in the two test methods. The velocities in flight and in the wind tunnel may be quite different for the same Mach number, so that a direct comparison of the experimental flutter speeds would not be possible.

An inspection of figure 5(a) shows that the correlation between flight and wind-tunnel flutter test results is entirely satisfactory both in the vicinity of $M = 0.85$ and near $M = 1.05$. All the test points, with the exception of the two at $M = 0.84$, were obtained with one model (wing 2). Only one test was completed with wing 1 due to a structural failure of one panel. In view of the consistency of the results, no distinction is made in the presentation of the results either between the two wings or between the panels of each wing. Furthermore, no distinction is made in the figure between tests with and without the fuselage extension because the only apparent effect of the extension was to change slightly the Mach number attainable with a given orifice and stagnation pressure. An additional comparison of wind-tunnel and flight flutter test results is presented in figure 5(b) in terms of the ratio ω_e/ω_α as a function of Mach number where ω_e is the experimental flutter frequency and ω_α is the uncoupled first torsion natural frequency. Here again the correlation is very satisfactory. The very good agreement between tunnel and flight flutter-speed ratios and flutter-frequency ratios shown in figure 5 indicates that reliable transonic flutter data may be obtained in wind tunnels equipped with slotted transonic test sections.

The results of the present investigation are further compared (fig. 6) with available data obtained in wind-tunnel, bomb-drop, and rocket tests of similar unswept, rectangular wings (ref. 4). The faired curve from reference 4, however, is for wings with a thickness ratio of 9 percent. The various physical parameters of all the wings considered are within the following limits: The wings have semispan aspect ratios ranging from 2 to 3.5, center-of-gravity locations between 43.7 and 49.6 percent chord, elastic axes between 30 and 50 percent chord, and wing-density parameters μ from 30 to 85.

The primary observation to be made concerning the comparison of the data of figure 6 is the separation of the results of the present investigation and of reference 1 from the faired curve at Mach numbers above approximately 0.9. It was surmised on the basis of the test results in reference 1 that the region in which V_e/V_R increases rapidly with M, defined as the region around $M = 0.9$ for the 9-percent-thick wings (ref. 4), might be moved to a higher Mach number range for thinner wings. This idea is clearly substantiated by the results of the present wind-tunnel tests which show that the flutter-speed ratio does not begin to increase rapidly until very close to $M = 1.0$ for the wings tested.

In the course of obtaining the data presented in figure 5, no wing failure due to flutter occurred. The flutter of the wings was such that the amplitude of the oscillations built up to a certain magnitude and then remained relatively constant throughout the test until tunnel operation ceased. This condition was also experienced in the flight tests in which the wings fluttered without failure. In an effort to bring about destructive flutter, one test was made in which the tunnel stagnation pressure was increased substantially above that required to initiate flutter. A failure occurred as a wrapping of the wings around the fuselage (fig. 7), but this failure was not of the type usually associated with flutter. The test records showed that there was no substantial increase in the oscillation amplitude (double amplitude was approximately 0.6 of the semispan). Although the wing panels were still attached to the fuselage at the end of the test, small cracks had appeared at the root of each panel. The value of V_e/V_R for panel failure is shown in figure 8 in relation to the initial flutter condition and the faired curve of figure 6. For the present case, at least, the aerodynamic and structural characteristics of the wings tested were such that the occurrence of flutter was not followed immediately by destructive instability.

CONCLUDING REMARKS

An experimental wing-flutter investigation was conducted in the Langley transonic blowdown tunnel, which is equipped with a slotted test section, in order to determine the correlation between transonic-wind-tunnel and flight flutter results. Flutter was obtained in the wind tunnel at several Mach numbers between 0.84 and 1.16 for two unswept, rectangular wings of aspect ratio 7.38 at 0° angle of attack. These wings were scaled models of wings which had fluttered in the course of free-fall flight tests. The very good agreement between the wind-tunnel and flight flutter test results shows the feasibility of conducting flutter tests in transonic wind tunnels with slotted test sections.

Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va.

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1. Lauten, William T., Jr., and Nelson, Herbert C.: Results of Two Free-Fall Experiments on Flutter of Thin Unswept Wings in the Transonic Speed Range. NACA RM L51C08, 1951.

2. Smilg, Benjamin, and Wasserman, Lee S.: Application of Three-Dimensional Flutter Theory to Aircraft Structures. ACTR No. 4798, Materiel Div., Army Air Corps, July 9, 1942.

3. Barmby, J. G., Cunningham, H. J., and Garrick, I. E.: Study of Effects of Sweep on the Flutter of Cantilever Wings. NACA Rep. 1014, 1951. (Supersedes NACA TN 2121.)

4. Lauten, William T., Jr., and Barmby, J. G.: Continuation of Wing Flutter Investigation in the Transonic Range and Presentation of a Limited Summary of Flutter Data. NACA RM L9B25b, 1949.

TABLE I

WING PARAMETERS

Parameter	Wing 1		Wing 2	
	Left	Right	Left	Right
Root section	NACA 65A004		NACA 65A004	
Tip section	NACA 65A002		NACA 65A002	
Span, in.	16.4		16.4	
A	7.38		7.38	
Λ, deg	0		0	
λ	1		1	
A_g	3.01	3.01	3.01	3.01
l, ft	0.558	0.558	0.558	0.558
b, ft	0.0928	0.0928	0.0926	0.0927
x_1	0.445	0.445	0.445	0.445
x_o	0.432	0.405	0.409	0.423
a	-0.136	-0.190	-0.182	-0.154
$a + x_\alpha$	-0.110	-0.110	-0.110	-0.110
r_α^2	0.1961	0.2018	0.2010	0.1973
f_h, cps	60.3	60.8	58.8	59.3
f_t, cps	365	365	377	372
f_α, cps	364	359	372	370
$(\omega_h/\omega_\alpha)^2$	0.0274	0.0287	0.0251	0.0257
μ_{stnd} at $0.7l$	154	154	154	154

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TABLE II

EXPERIMENTAL AND CALCULATED RESULTS AT START OF FLUTTER

M_e	V_e, fps	ρ_e, slugs/cu ft	μ_e (a)	ω_e, radians/sec	V_R, fps	ω_R, radians/sec	V_e/V_R	Wing	Run	Nose
0.839	836.5	0.00578	56.02	776	777.0	1059	1.077	1-L	1	Ogive
.839	836.5	.00578	56.02	776	771.6	1065	1.084	1-R	1	↓
.990	1010.3	.00391	83.16	666	964.0	1052	1.048	2-L	2	
.990	1010.3	.00391	83.16	659	952.3	1039	1.061	2-R	2	
1.035	1077.0	.00431	75.45	666	919.9	1061	1.171	2-L	3	
1.020	1056.5	.00434	74.78	678	906.2	1049	1.166	2-R	3	
1.056	1067.1	.00482	67.47	771	872.8	1073	1.223	2-L	4	
1.066	1064.9	.00454	71.48	648	886.8	1056	1.201	2-R	4	
1.099	1020.4	.00607	53.57	848	782.5	1096	1.304	2-L	5	Ext.
1.105	1016.6	.00628	51.68	848	759.9	1086	1.338	2-R	5	↓
1.116	1092.5	.00600	54.20	843	785.5	1101	1.391	2-L	6	
1.087	996.8	.00649	50.00	922	748.7	1090	1.331	2-R	6	
1.146	1035.4	.00709	45.87	1040	727.2	1115	1.424	2-L	7	
1.157	1057.6	.00669	48.50	985	738.1	1095	1.433	2-R	7	
.956	937.8	.00457	71.13	669	894.3	1073	1.049	2-L	8	
.956	937.8	.00457	71.13	678	884.1	1053	1.061	2-R	8	
[b].904	872.4	.00673	48.30	----	745.5	1106	1.170	2-L	8	
[b].904	872.4	.00673	48.30	----	736.6	1095	1.184	2-R	8	

[a]Referred to station at 0.7l
[b]Conditions at wing failure

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[redacted]

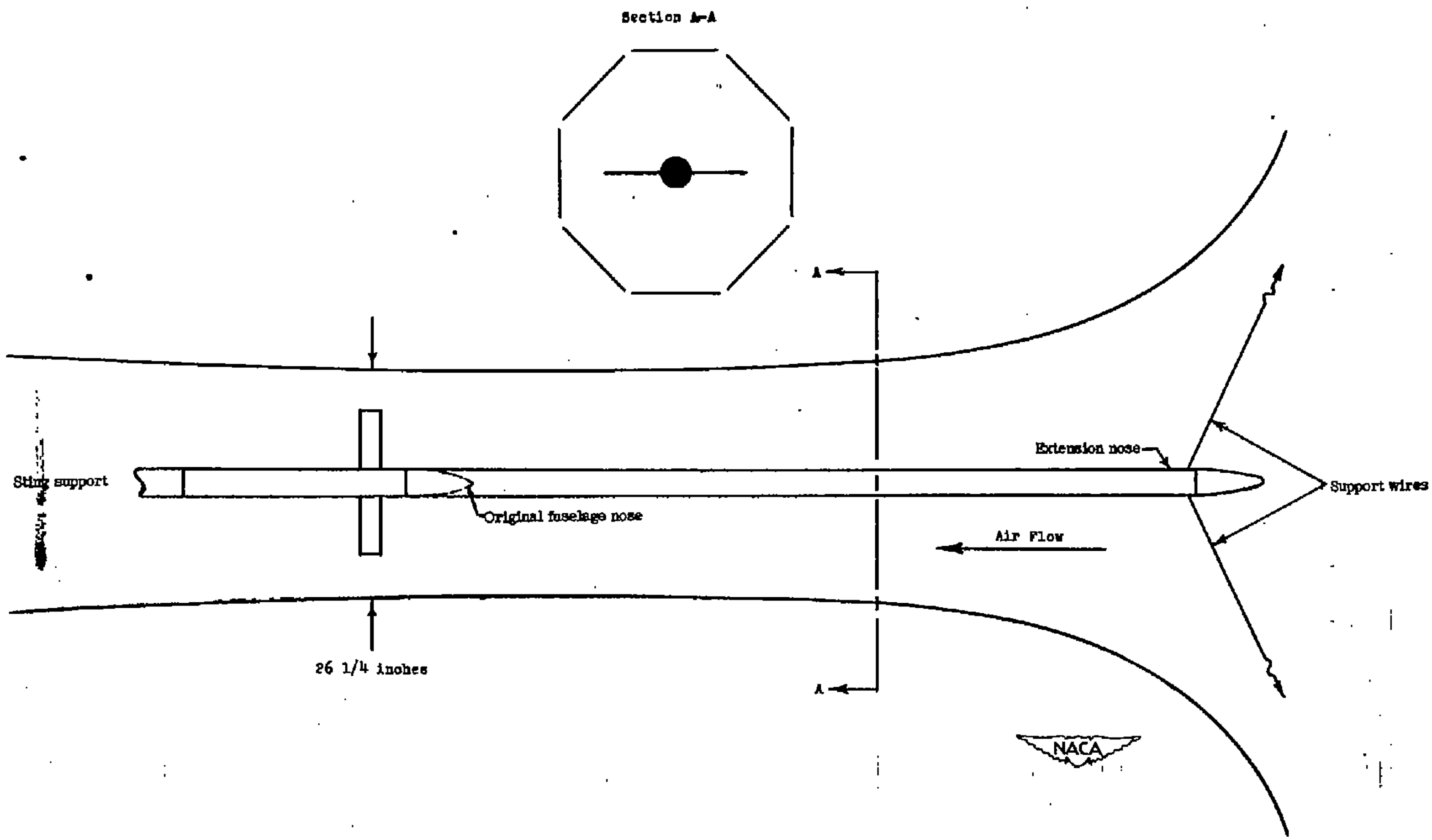


Figure 1.- Plan view of Langley transonic blowdown tunnel with flutter model installed.

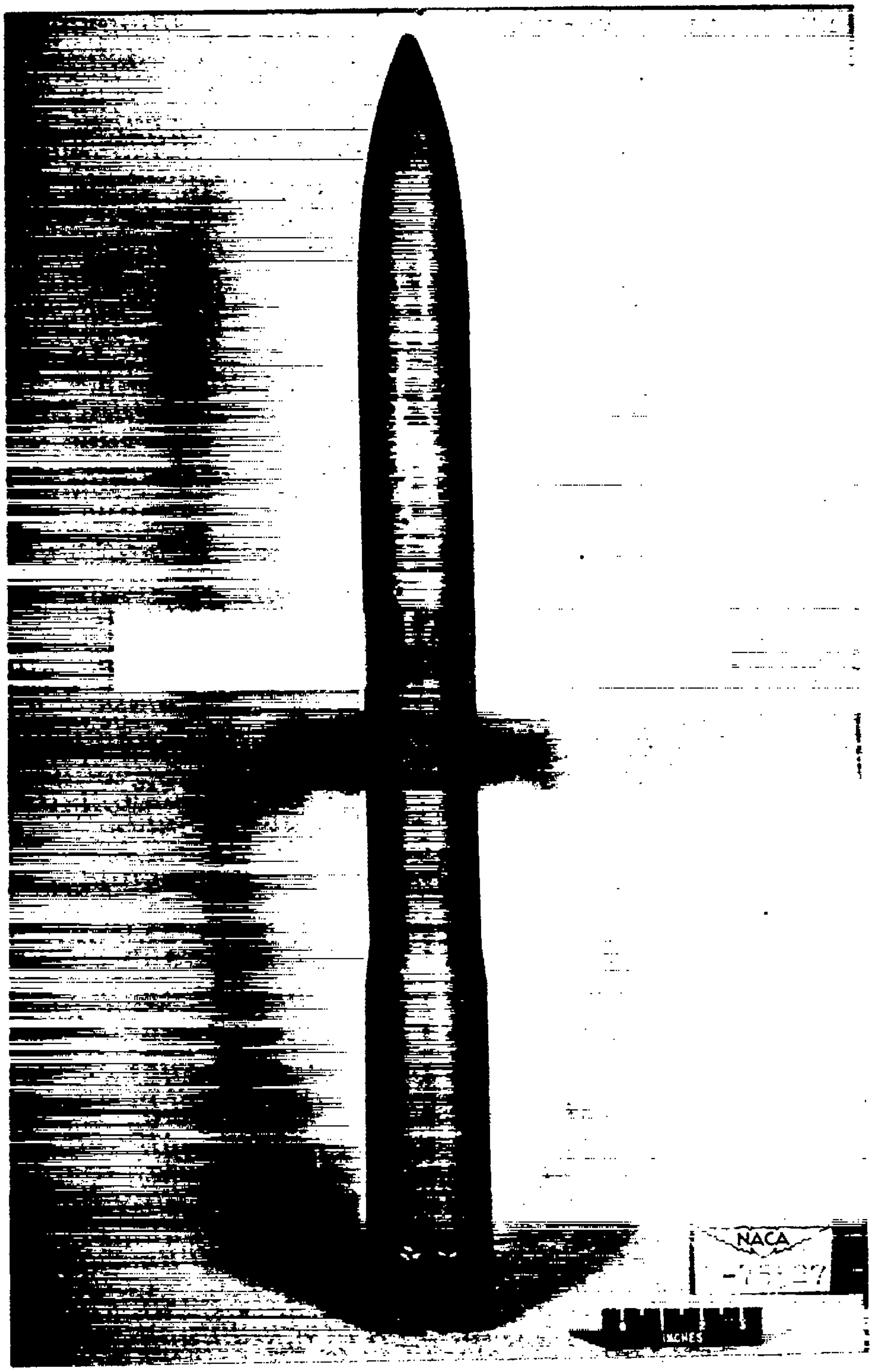


Figure 2.- Wind-tunnel flutter wing and fuselage.

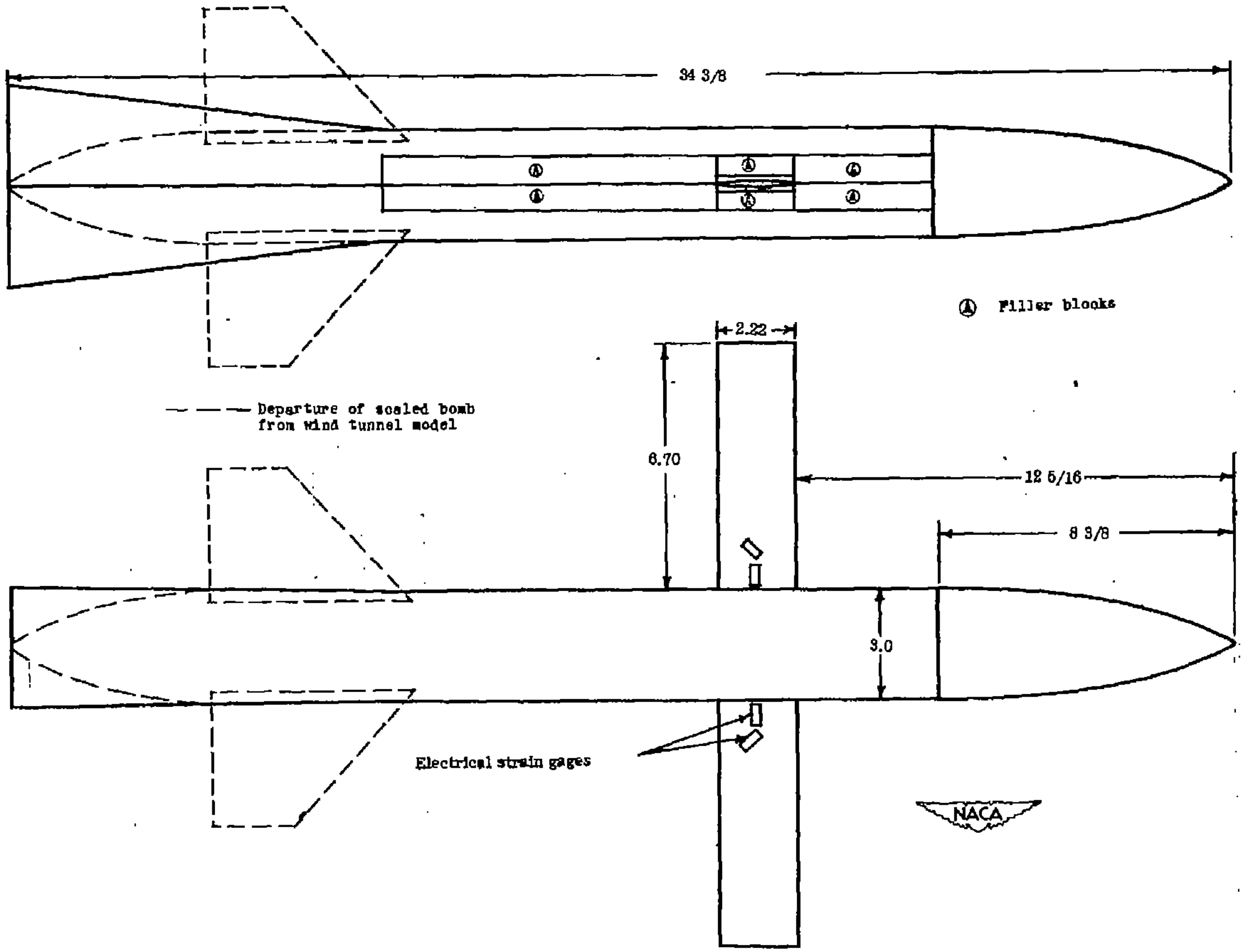


Figure 3.- Flutter-wing model and fuselage. All dimensions are in inches.

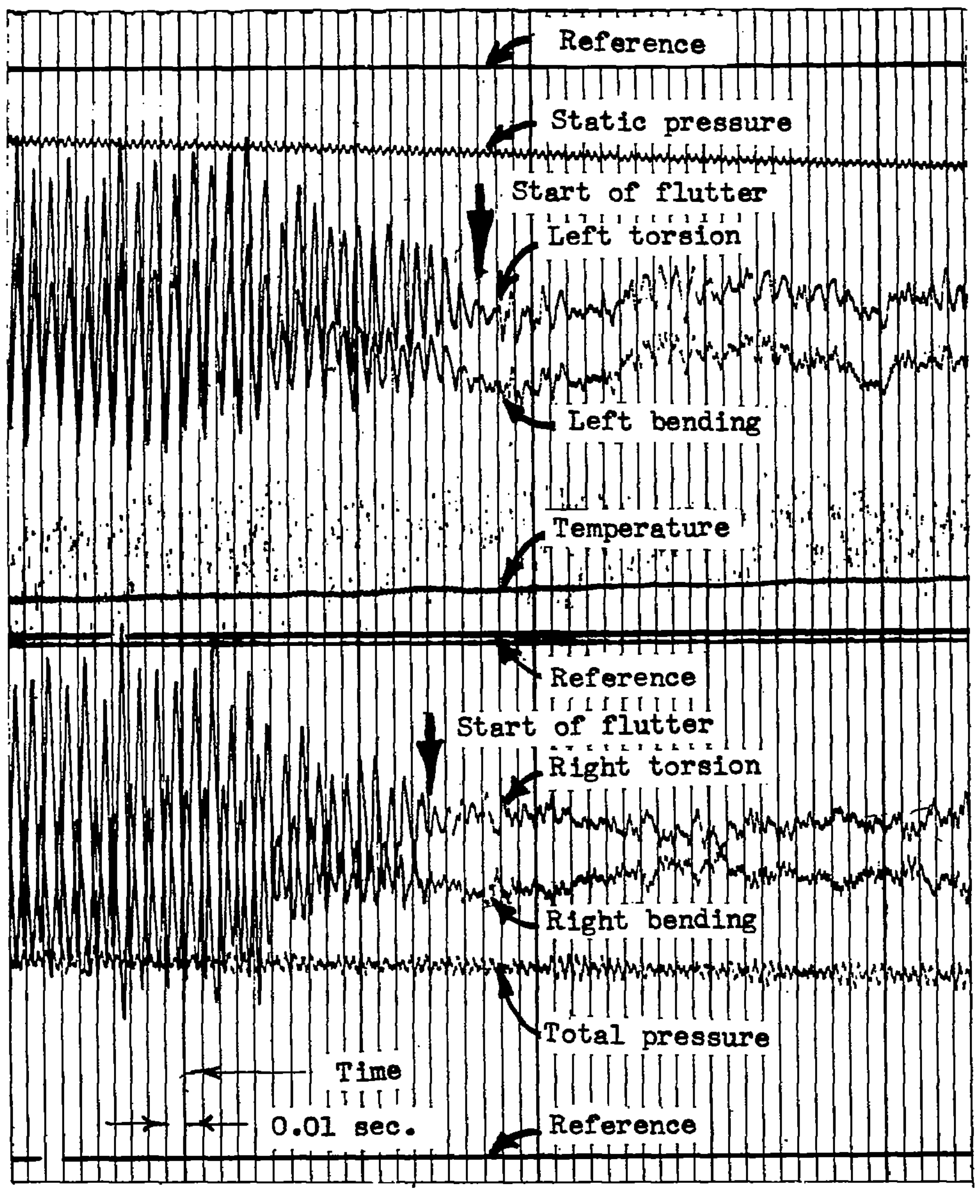
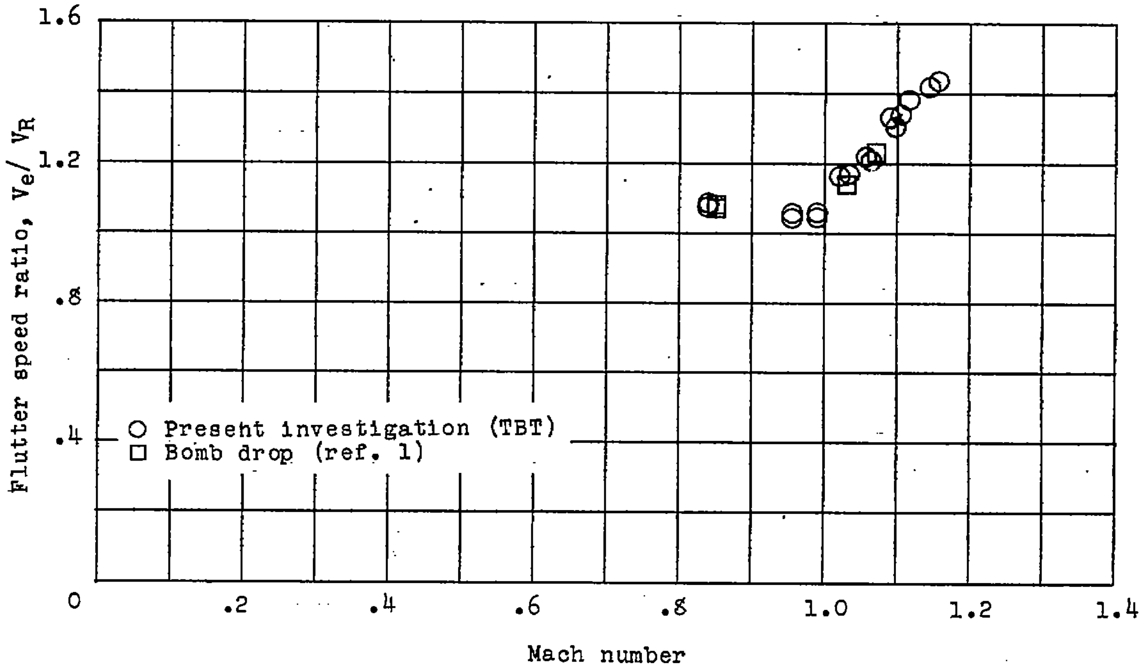
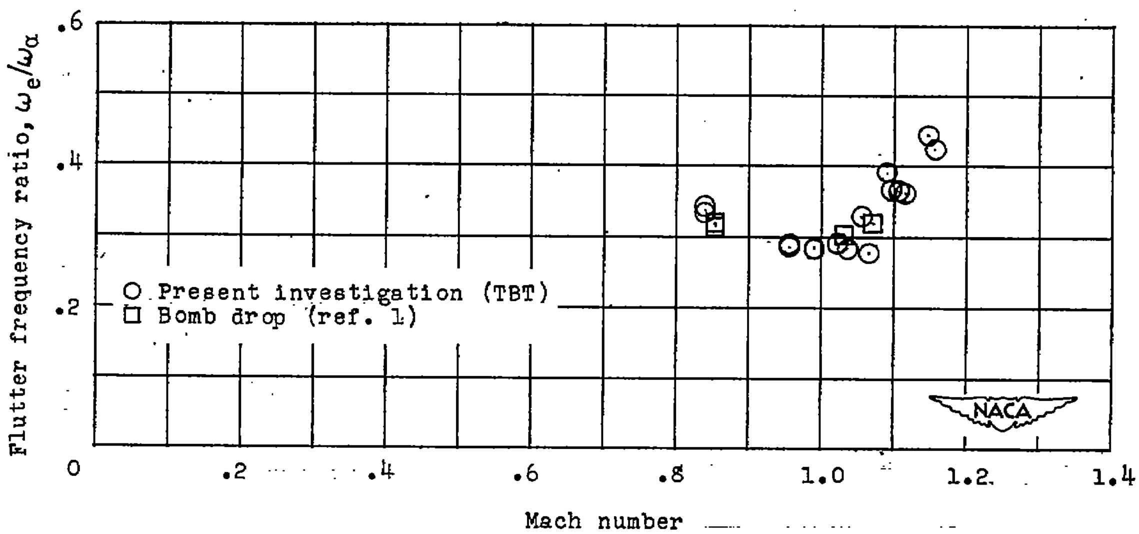


Figure 4.- Sample oscillograph record of flutter test near $M = 1.0$.



(a) Correlation of flutter-speed ratio.



(b) Correlation of flutter-frequency ratio.

Figure 5.- Correlation of wind-tunnel and flight flutter characteristics.

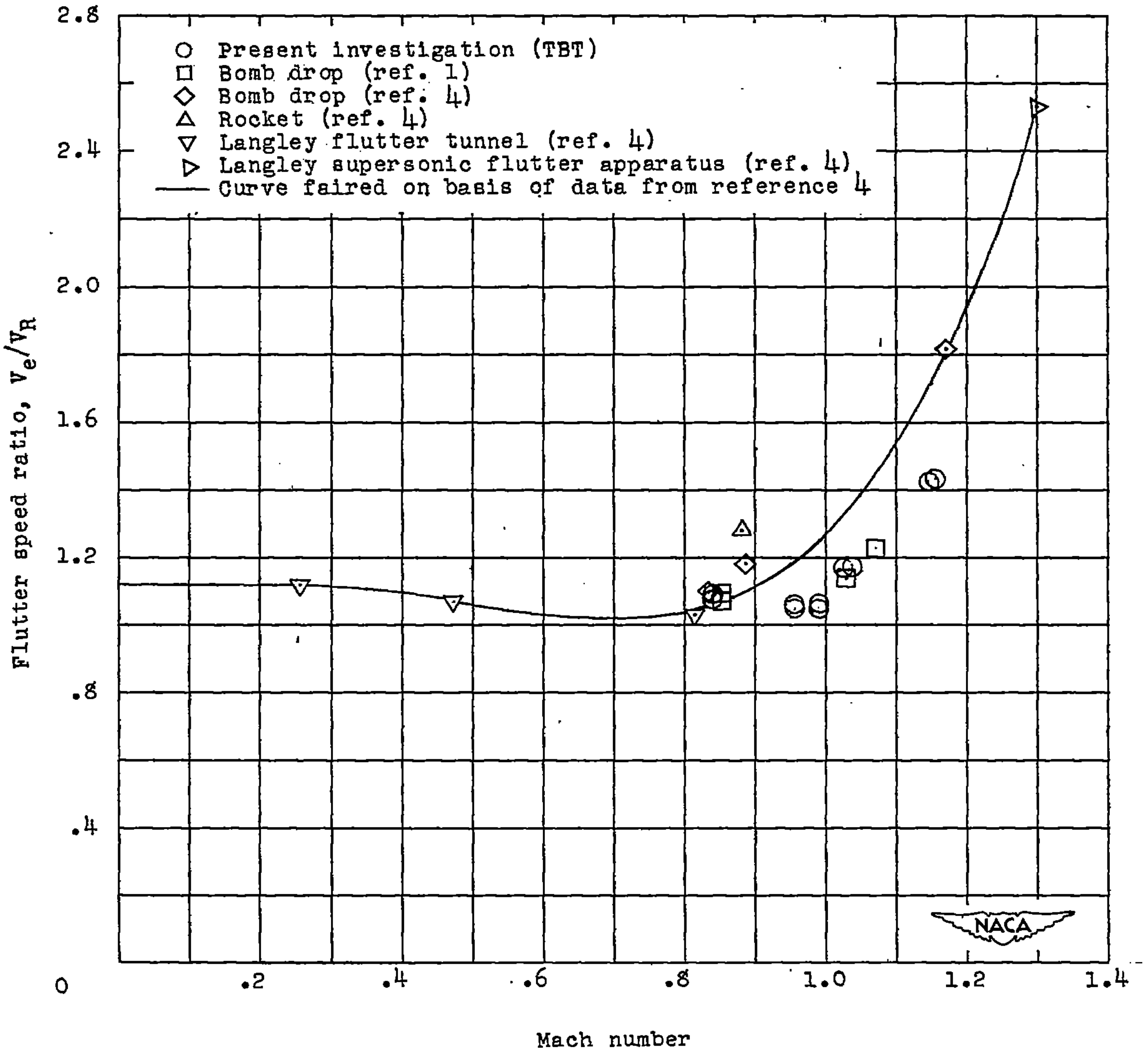
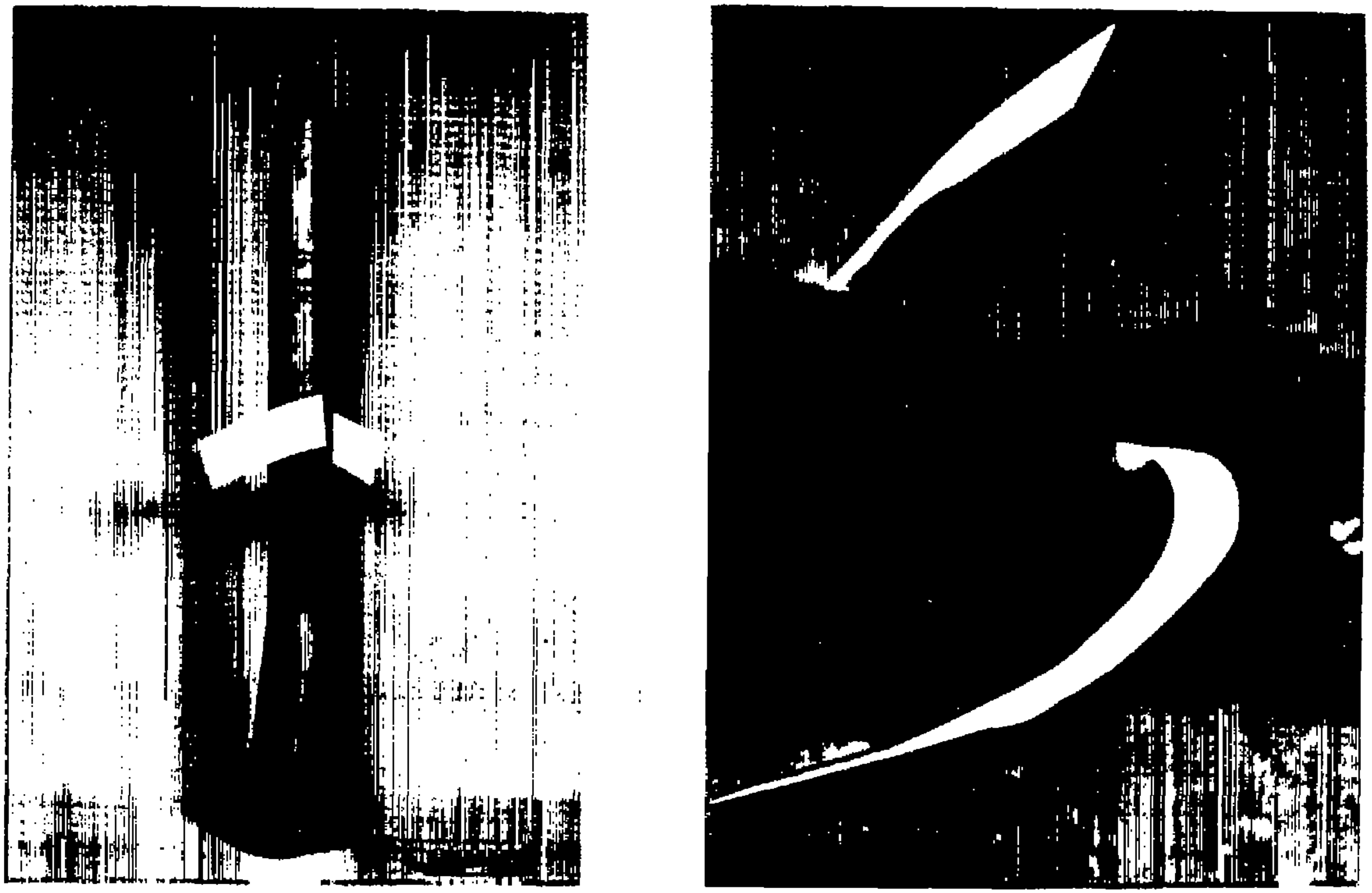


Figure 6.- Variation of flutter-speed ratio with Mach number of several similar wings as obtained in various flight and wind tunnel tests.



(a) Side view.

(b) One-quarter front view.

Figure 7.- Condition of wing after failure. L-77030

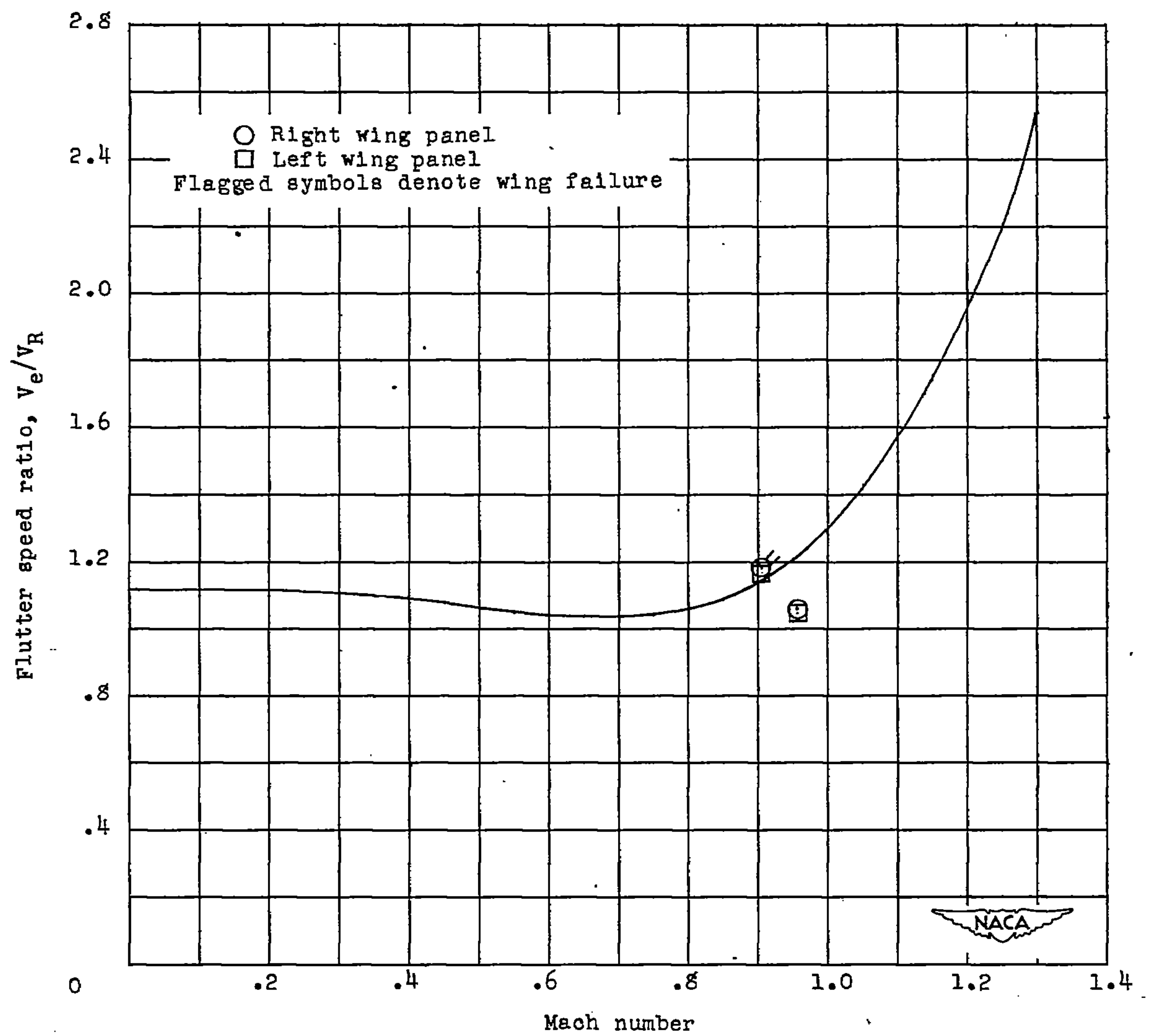


Figure 8.- Comparison of flutter-speed ratio at beginning of flutter and at wing failure.

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