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RESEARCH MEMORANDUM

PRELIMINARY RESULTS FROM FLIGHT MEASUREMENTS IN GRADUAL-TURN MANEUVERS OF THE WING LOADS AND THE DISTRIBUTION OF LOAD AMONG THE COMPONENTS OF A BOEING B-47A AIRPLANE

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SUMMARY

Results are presented of a preliminary analysis of some strain-gage measurements of the loads carried by the wing and horizontal tail of a Boeing B-47A airplane during gradual-turn maneuvers for altitudes varying from 15,000 feet to 35,000 feet and covering a Mach number range from 0.44 to 0.80.

The additional air-load center of pressure on the 35° sweptback, high-aspect-ratio, flexible wing of the test airplane exhibted no significant tendency to change appreciably in a spanwise direction because of Mach number throughout the test range. The center of pressure, however, moved forward with increasing Mach number.

The distribution of load among the components of the Boeing B-47A airplane followed the pattern which was typical of that found to exist on several airplanes tested previously; that is, the various components carried approximately a constant proportion of the total airplane additional lift throughout the Mach number range covered and the fraction of the wing-fuselage normal-force coefficient carried by the exposed wing was approximately the same magnitude as the ratio of the area of the exposed wing to the area of the total wing.

INTRODUCTION

The National Advisory Committee for Aeronautics is currently conducting a flight investigation with a Boeing B-47A airplane. The aims

of the flight program are to obtain information on the effects of aeroelasticity on quasi-static loads, dynamic stability, dynamic loads, and handling qualities of a large flexible airplane having sweptback wing and tail surfaces.

Although the flight tests are still in progress, sufficient data have been obtained to indicate the approximate magnitude of the Mach number and dynamic-pressure effects on the distribution of the additional load. This report is the result of a preliminary examination of some of the loads measurements obtained by simplified data-reduction procedures and is subject to any modification that might be warranted when a detailed analysis is made at the completion of the flight investigation.

Data were obtained in gradual-turn maneuvers made at altitudes from 15,000 feet to 35,000 feet covering the Mach number range from 0.44 to 0.80.

SYMBOLS

a	speed of sound, ft/sec
V	true airspeed, ft/sec
M	Mach number, V/a
q	dynamic pressure, lb/sq ft
R_1	left-wing inboard shear and moment reference axis (see fig. 3)
R_2	left-wing inboard torque reference axis (see fig. 3)
R_3	left-wing outboard shear and moment reference axis (see fig. 3)
L	aerodynamic load on left wing outboard of reference axis
L_T	total aerodynamic load on horizontal tail outboard of gage locations
$\bar{c}_1$	mean aerodynamic chord of wing
$\bar{c}_2$	mean aerodynamic chord of wing outboard of reference axis R_1
S	total wing area, including portion covered by fuselage (wing-fuselage area), sq ft

S_1 twice the area of wing outboard of reference axis R_1, sq ft

W airplane gross weight, lb

n airplane normal load factor at airplane center of gravity, g units

C_{N_A} airplane normal-force coefficient, nW/qS

C_{N_T} horizontal-tail normal-force coefficient, L_T/qS

$C_{N_{R_1}}$ normal-force coefficient of left-wing air load outboard of reference axis R_1, L_{R_1}/qS

$C_{N_{R_3}}$ normal-force coefficient of left-wing air load outboard of reference axis R_3, L_{R_3}/qS

$C_{N_{WF}}$ normal-force coefficient of wing-fuselage combination, $C_{N_A} - C_{N_T}$

APPARATUS

Airplane.- The Boeing B-47A airplane, as illustrated by the photograph of figure 1 and the three-view drawing shown in figure 2, is a jet-propelled medium bomber powered by six General Electric J-47-GE-23 turbojet engines, each having a sea-level thrust rating of 5,800 pounds at 100 percent rotational speed. The pertinent dimensions and characteristics of the airplane are listed in table I.

The airplane had two spanwise rows of vortex generators, approximately two-thirds the span of the aileron in length, mounted on the upper-wing surface close to the front spar. (See fig. 3.) The yaw damper, although part of the standard equipment, was not used during the present tests.

Instrumentation.- The instrumentation installed in the test airplane relevant to this report consisted of standard NACA recording instruments to measure airspeed, altitude, and normal acceleration at the airplane center of gravity. The airspeed head was mounted on a boom extending forward of the nose of the fuselage equivalent to a distance of approximately 0.8 of the maximum diameter of the fuselage and the airspeed system was calibrated in flight.

Strain gages pertinent to this paper were located on the left wing and horizontal tail (see figs. 2 and 3) to measure structural shear,

moment, and torque for the load acting outboard of axis R_1, shear and moment for the load acting outboard of axis R_3, and the horizontal-tail load. A strain-gage-calibration procedure similar to that outlined in reference 1 was used to combine the primary strain-gage bridges and to obtain equations for structural loads in terms of the gage outputs as recorded on 18-channel oscillographs.

A 1/10-second time pulse was used to correlate the records of all recording instruments.

TESTS, RESULTS, AND DISCUSSION

In order to expedite the presentation of some preliminary results the maneuvers analyzed for this report consisted only of wind-up turns that initiated from trimmed steady flight and were gradually tightened in such a manner that the wing loading was maintained essentially symmetrical and the acceleration essentially uniform over the component parts of the airplane.

The test conditions covered a range of load factor from 1 g to 2.5g or stall buffeting whichever occurred first, a range of Mach number from 0.44 to 0.80, a range of dynamic pressure from 130 lb/ft^2 to 460 lb/ft^2, and a range of altitude from 15,000 feet to 35,000 feet. The airplane take-off gross weight was approximately 125,000 pounds and the landing weight was approximately 100,000 pounds. The center of gravity was varied between 14 and 27 percent of the mean aerodynamic chord by loading the fuel in selected quantities in the five fuselage tanks and then programming fuel consumption from the tanks while in flight. During any particular test run, the center-of-gravity position varied less than 1 percent because of fuel consumption.

Some inconsistencies in the levels of the measured strain-gage data were noted and there were indications that these effects resulted from wing and tail structure temperatures varying from flight to flight with changes in altitudes during each flight. At the present time, sufficient data have not been obtained to ascertain completely the magnitude of these temperature-induced effects on the strain-gage outputs. For this reason, only additional air-load results are presented since these results can be expressed as the rate of change of one load with respect to another during the short period of a maneuver and do not require correction for temperature effects. These results are presented in terms of wing load and center of pressure and percent of total airplane load acting on the wing and tail.

Wing-load center of pressure.- The rates of change of moment and torque with shear locate the spanwise and chordwise positions, respectively,

of the center of pressure from the pertinent reference stations. Typical variations of the torque and moment with shear for the air load acting outboard of axis R_1 are shown in figure 4 for a maneuver made at $M = 0.60$ at 20,000 feet. It can be seen from the linearity of the plots that the center-of-pressure location of the additional air load remained unchanged throughout the maneuver. The variation of moment with shear for the air load acting outboard of reference axis R_3 is shown in figure 5 for the same maneuver.

The center-of-pressure positions were determined from curves similar to those shown in figure 4 for each maneuver and the range of values obtained is indicated by the envelope drawn on the left-wing plan form in figure 6. It can be seen from this envelope that the additional air-load center of pressure had a 50-inch spanwise range, a 17-inch chordwise range, and encompassed the 25-percent $\bar{c}_2$ point. The changes in the center-of-pressure location can be attributed to the effects of various parameters, such as Mach number, dynamic pressure, and airplane weight, which varied from run to run. No attempt was made to isolate the individual effect on the center of pressure of any of the parameters mentioned owing to the relatively few maneuvers analyzed. The center-of-pressure values have, however, been plotted against Mach number and are presented in figures 7 and 8.

In figure 7 the spanwise center of pressure of the additional load outboard of axis R_1 and that outboard of axis R_3 are plotted against Mach number. The ordinate is percent of span outboard of axis R_1. Mean lines are drawn through the respective sets of points at the 45-percent- and 73-percent-span stations. There is no significant tendency indicated that the load shifted appreciably with changing Mach number.

Although any attempt at detailed correlation with theory for the B-47A configuration is not warranted for these data, theoretical centers of pressure are, for general comparison, shown in figure 7 for a rigid-wing-alone, incompressible, additional load distribution as obtained from reference 2.

In figure 8 the chordwise centers of pressure, in percent $\bar{c}_2$, for the loads outboard of axis R_1 are plotted against Mach number and a mean line drawn through the plotted values. A gradual forward or destabilizing shift with increasing Mach number of about 3 percent $\bar{c}_2$ from the quarter-chord position is apparent over the Mach number range.

Component loads.- The fraction of the total-airplane normal-force coefficient C_{N_A} carried by the wings and tail was determined from the slopes of the respective curves for the variation of the component normal-force coefficients with C_{N_A}. Although the left-wing loads only were

measured, the right-wing loads were assumed to be equal; an assumption justified by the type of maneuver analyzed.

Component-force coefficients, $C_{N_{R_1}}$, $C_{N_{R_2}}$, and C_{N_T}, for the wing- and tail-load variation with C_{N_A} are presented in figures 9 and 10 for a typical run made at $M = 0.60$ and 20,000 feet altitude. As in the case of the plots used in determining the centers of pressure (figs. 4 and 5), the variations shown in figures 9 and 10 are linear so that the fraction of the total load carried by each component remained constant throughout the C_{N_A} range of the maneuver. Similar plots were made for all the runs and the quantities $2\left(dC_{N_{R_1}}/dC_{N_A}\right)$, $2\left(dC_{N_{R_2}}/dC_{N_A}\right)$, and dC_{N_T}/dC_{N_A} are plotted against Mach number in figure 11. The fraction of C_{N_A} carried by the wing did not change appreciably with Mach number. Once again no attempt was made to isolate the effects of the parameters which varied from run to run. Approximately 84 percent of the C_{N_A} was carried by the wing area outboard of axis R_1 plus the corresponding area on the right wing whereas the wing area outboard of axis R_3 plus the corresponding area on the right wing carried approximately 28 percent of C_{N_A}.

The quantity $\frac{dC_{N_T}}{dC_{N_A}}$ varies from approximately zero for the maneuvers made with the center of gravity at 14 percent $\bar{c}_1$ to 5 percent of the total airplane normal-force coefficient for maneuvers made with the center of gravity at 27 percent $\bar{c}_1$. A slight increase in $\frac{dC_{N_T}}{dC_{N_A}}$ with increasing Mach number, which is consistent with the forward or destabilizing shift in chordwise center of pressure on the wing, can be noted for the forward and normal center-of-gravity points shown in figure 11.

As found in previous tests with several other airplanes of differing configuration and rigidity, (refs. 3 and 4), the fraction of the wing-fuselage load carried by the wing of the test airplane (0.86) was approximately equal to the area ratio $\frac{S_1}{S}$. The normal-force coefficient of the wing-fuselage combination $C_{N_{WF}}$ was determined by subtracting C_{N_T} from C_{N_A}. As a matter of interest the variation with Mach number of the fractions of the wing-fuselage load carried by the wing of the North American B-45A, Douglas D-558-II, and test airplane are shown in figure 12. A table listing pertinent physical characteristics of the airplanes is included in the figure.

CONCLUDING REMARKS

Within the scope and limitations of a preliminary analysis of the measured additional air loads on a Boeing B-47A airplane in gradual-turn maneuvers, it was indicated that:

1. The chordwise center of pressure on the wing showed a gradual forward movement with increasing Mach number and there was no significant trend noted that indicated an appreciable shift of the load in a spanwise direction due to Mach number.

2. The proportion of the total airplane normal-force coefficient carried by the wing remained approximately constant up to the maximum test Mach number of 0.80.

3. The wings carried approximately 0.86 of the total-airplane-less-tail normal-force coefficient whereas the ratio of the exposed-wing area to the wing-fuselage area was approximately 0.87.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., January 24, 1955.

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2. DeYoung, John, and Harper, Charles W.: Theoretical Symmetric Span Loading at Subsonic Speeds for Wings Having Arbitrary Plan Form. NACA Rep. 921, 1948.

3. Harper, Paul W.: Wing and Fuselage Loads Measured in Flight on the North American B-45 and F-82 Airplanes. NACA RM L52L09, 1953.

4. Mayer, John P., and Valentine, George M.: Flight Measurements With the Douglas D-558-II (BuAero No. 37974) Research Airplane. Measurements of the Distribution of the Aerodynamic Load Among the Wing, Fuselage, and Horizontal Tail at Mach Numbers up to 0.87. NACA RM L50J13, 1951.

TABLE I

DIMENSIONS AND CHARACTERISTICS OF THE BOEING B-47A AIRPLANE

Wing:	
Span, ft	116.0
Area, sq ft	1428.0
Aspect ratio	9.43
Taper ratio	0.42
Thickness ratio	0.12
Mean aerodynamic chord, in.	155.9
Sweep at 25-percent chord, deg	35.0
Root chord, in.	208.0
Tip chord, in.	87.0
Airfoil section	BAC 145
Incidence (root and tip), deg	2° 45'
Dihedral, deg	0
Horizontal Tail:	
Span, ft	33.0
Area, sq ft	268
Aspect ratio	4.06
Taper ratio	0.42
Thickness ratio	0.10
Mean aerodynamic chord, in.	102.9
Sweep at 25-percent chord, deg	33.0
Root chord, in.	137.0
Tip chord, in.	58.0
Incidence	-0.15'
Airfoil section	BAC 100
Vertical Tail:	
Span, ft	18.9
Area (including dorsal), sq ft	230.0
Aspect ratio	1.55
Taper ratio	0.34
Thickness ratio	0.10
Mean aerodynamic chord, in.	158.4
Sweep at 25-percent chord, deg	35.0
Root chord (at water level 200), in.	216.0
Tip chord, in.	74
Airfoil section	BAC 100

Power plant:

Six General Electric J-47-GE-23 turbojet engines with a sea-level military thrust rating of 5,800 pounds at 100 percent rotational speed

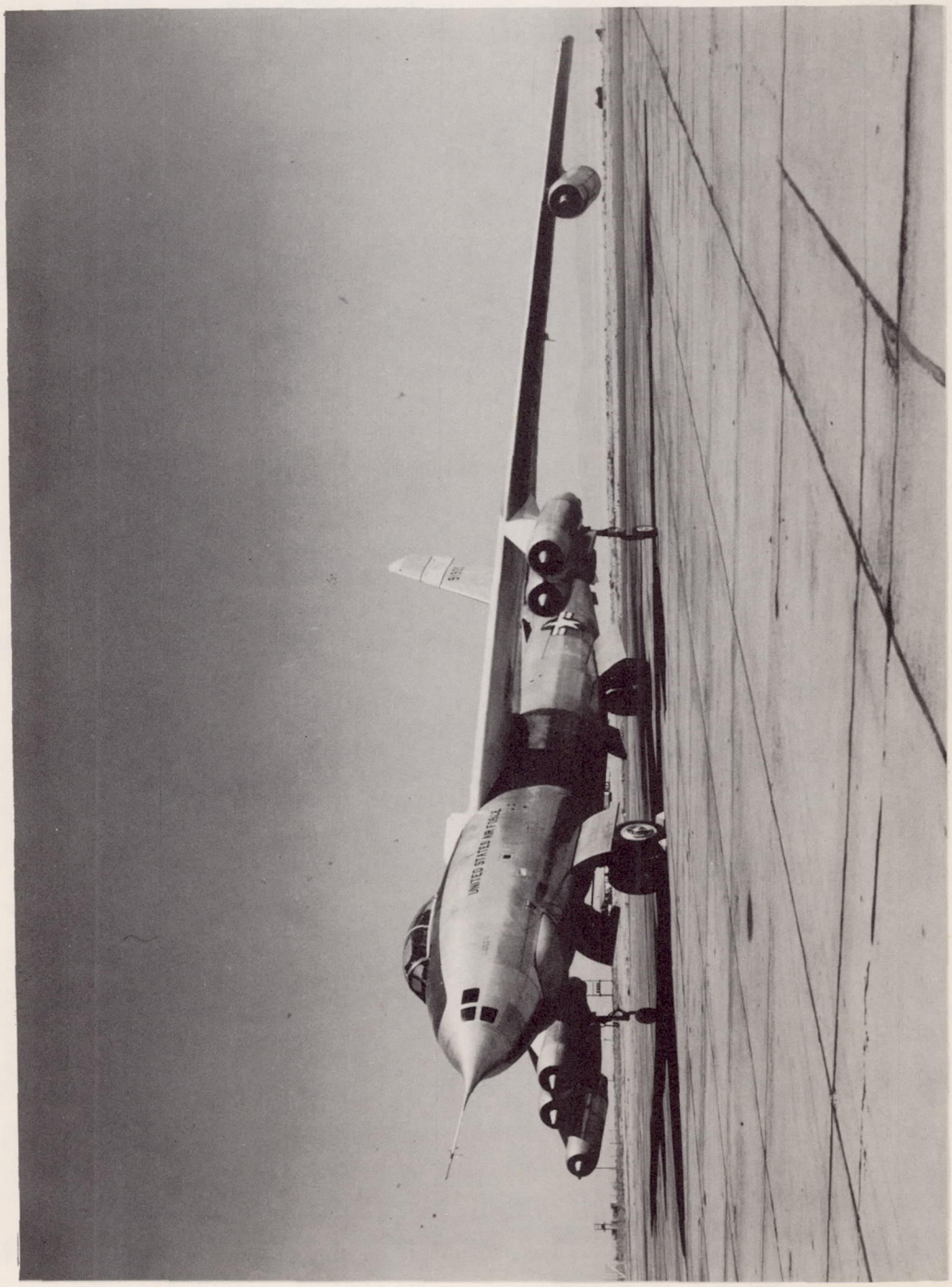

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Figure 1.- Photograph of test airplane.

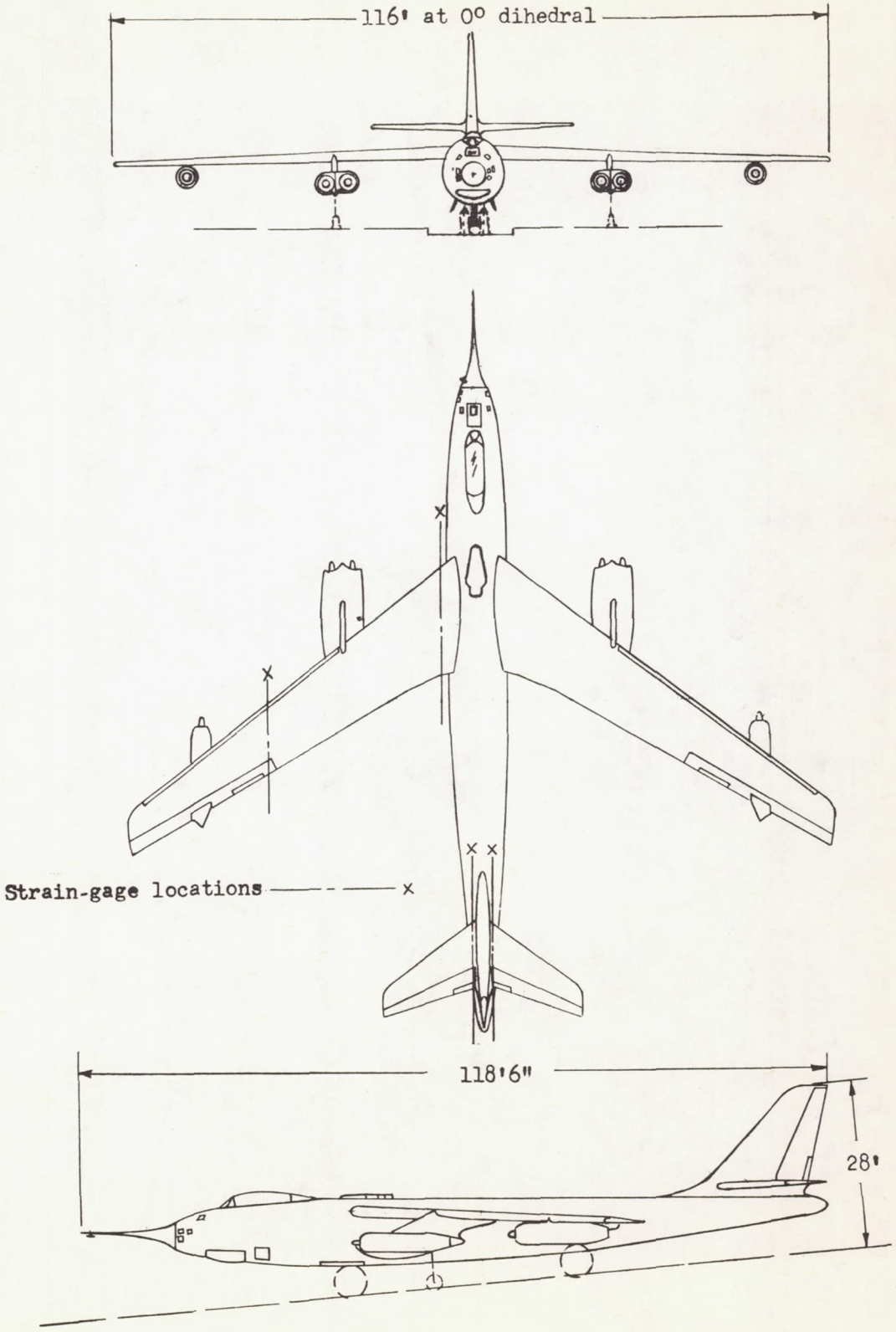

Figure 2.- Three-view drawing of the test airplane.

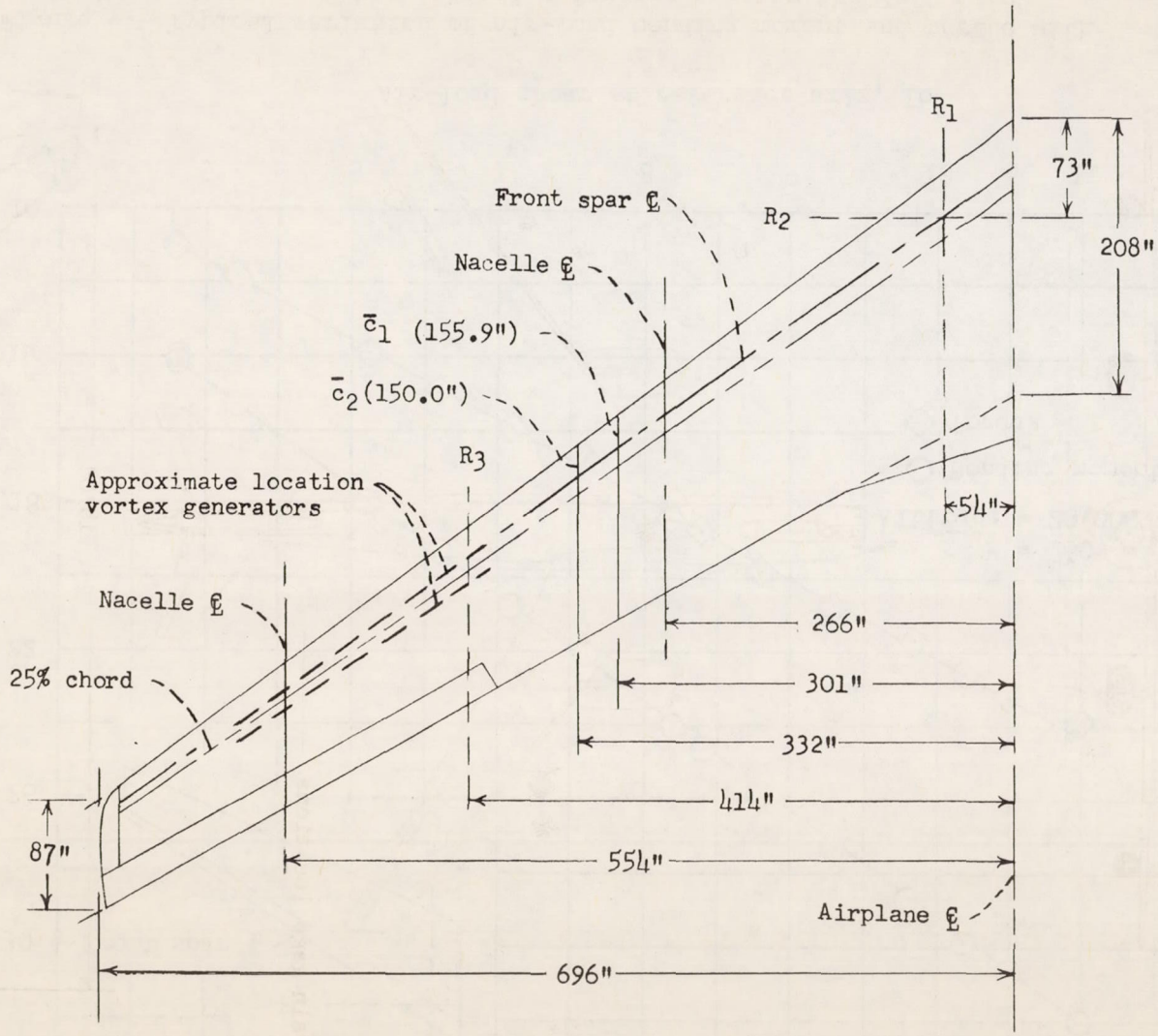

Figure 3.- Sketch of left wing showing pertinent dimensions and reference-axes locations.

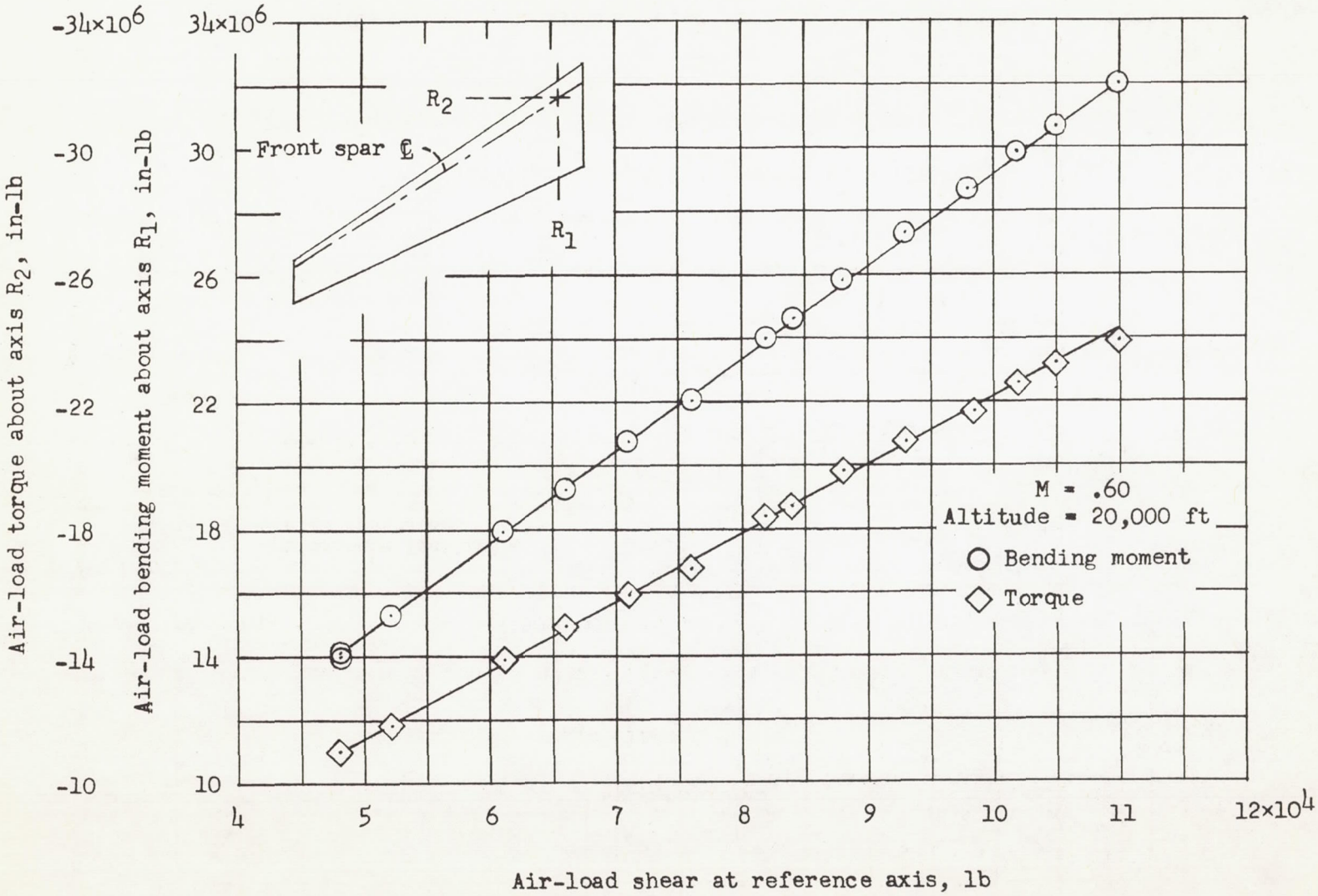

Figure 4.- Typical variation of air-load bending moment and torque with air-load shear at the inboard reference axes on the left wing.

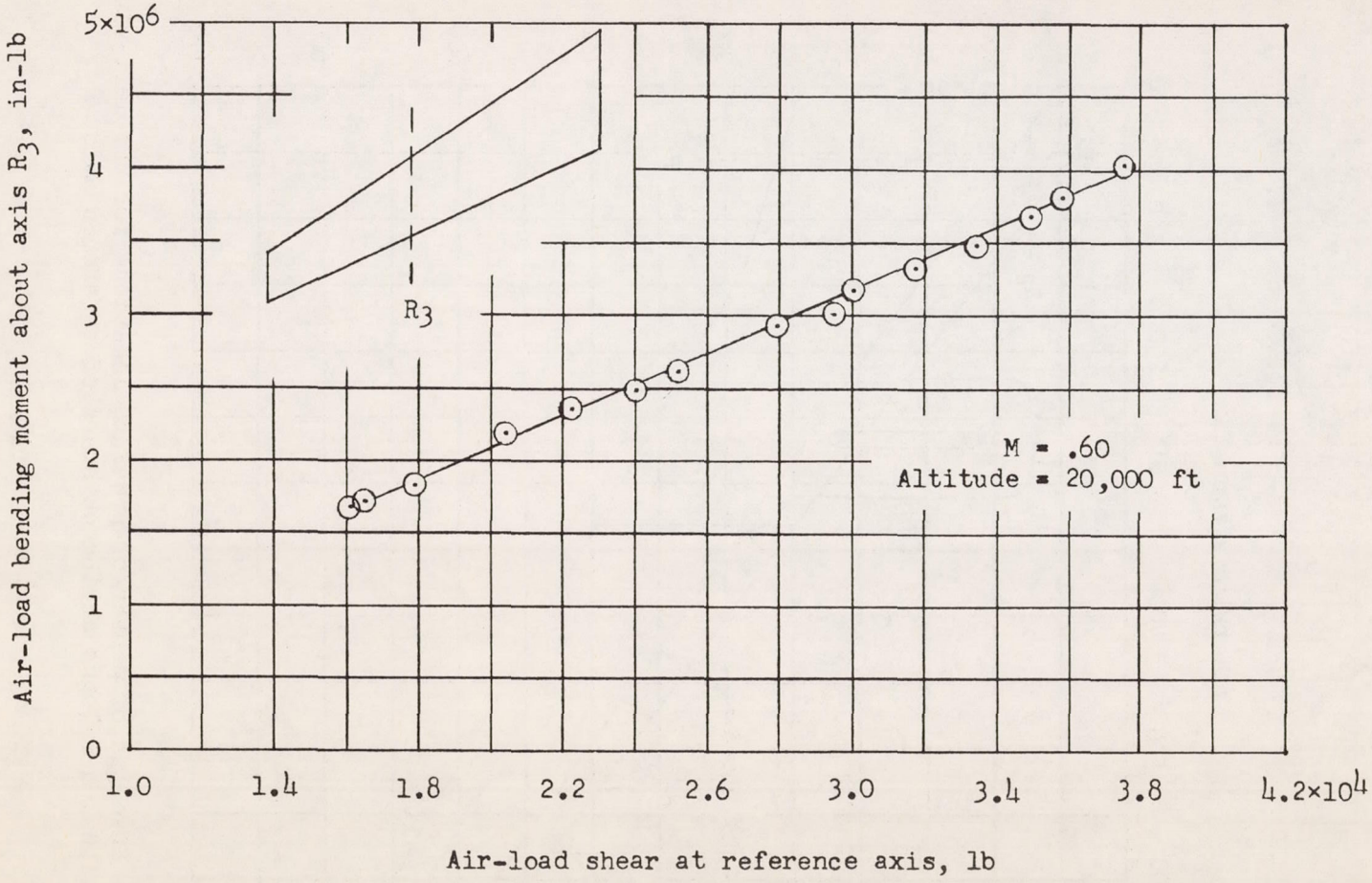

Figure 5.- Typical variation of air-load bending moment with air-load shear at the outboard reference axis on the left wing.

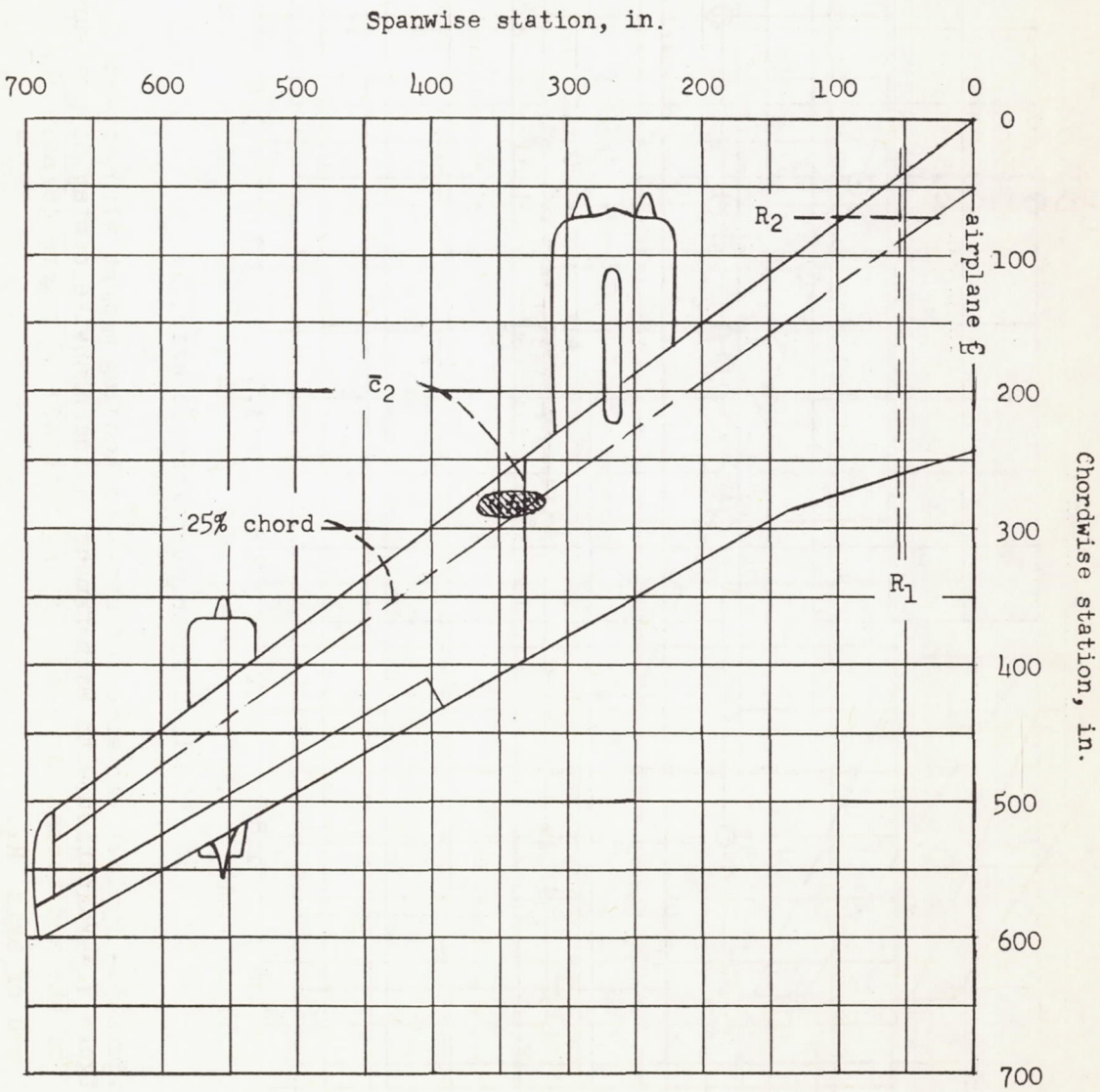

Figure 6.- Envelope of center-of-pressure locations for the left-wing additional air loads acting outboard of reference axis R_1.

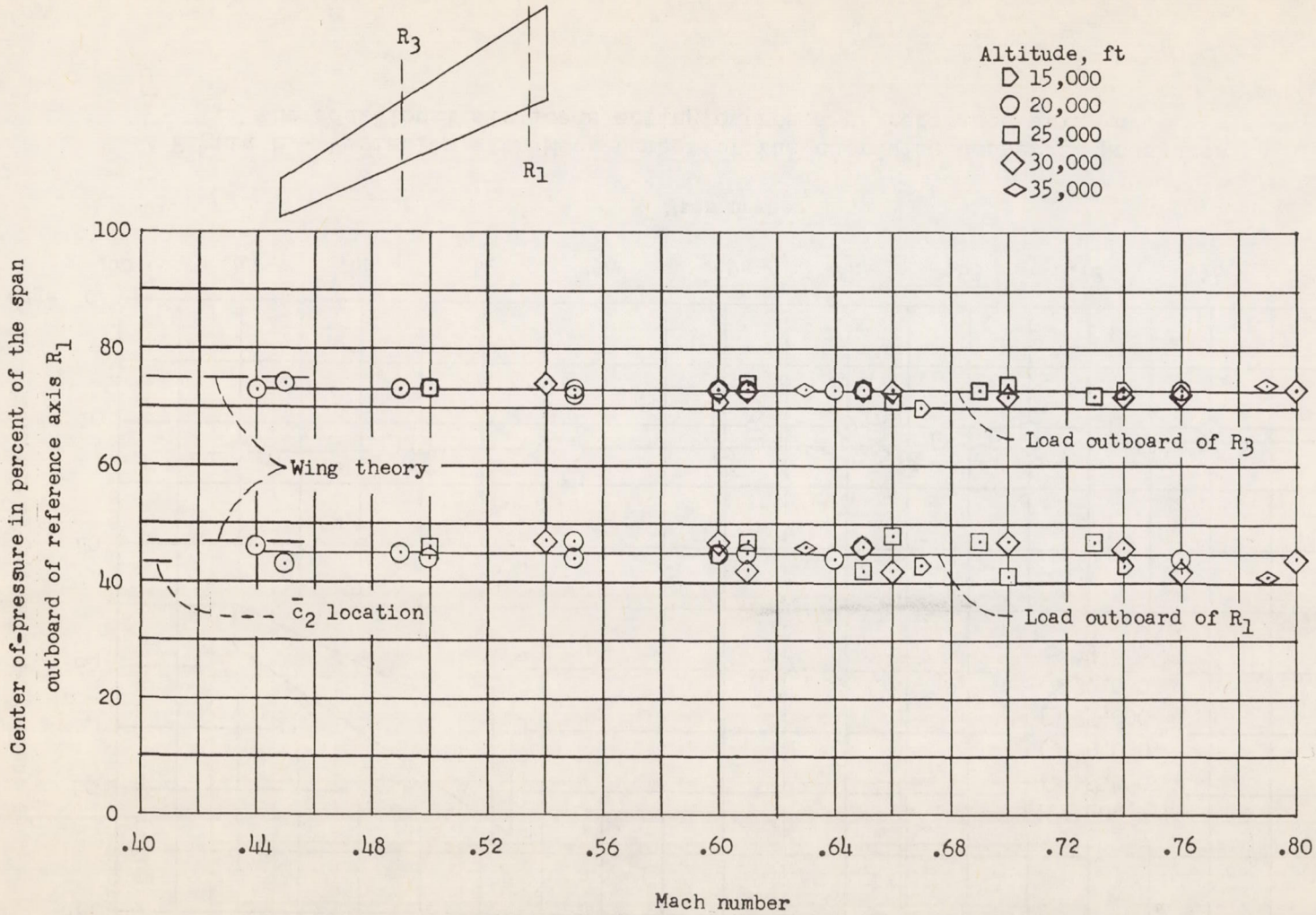

Figure 7.- Variation with Mach number of the spanwise center of pressure of the additional air loads acting outboard of reference axis R_1 and of axis R_3.

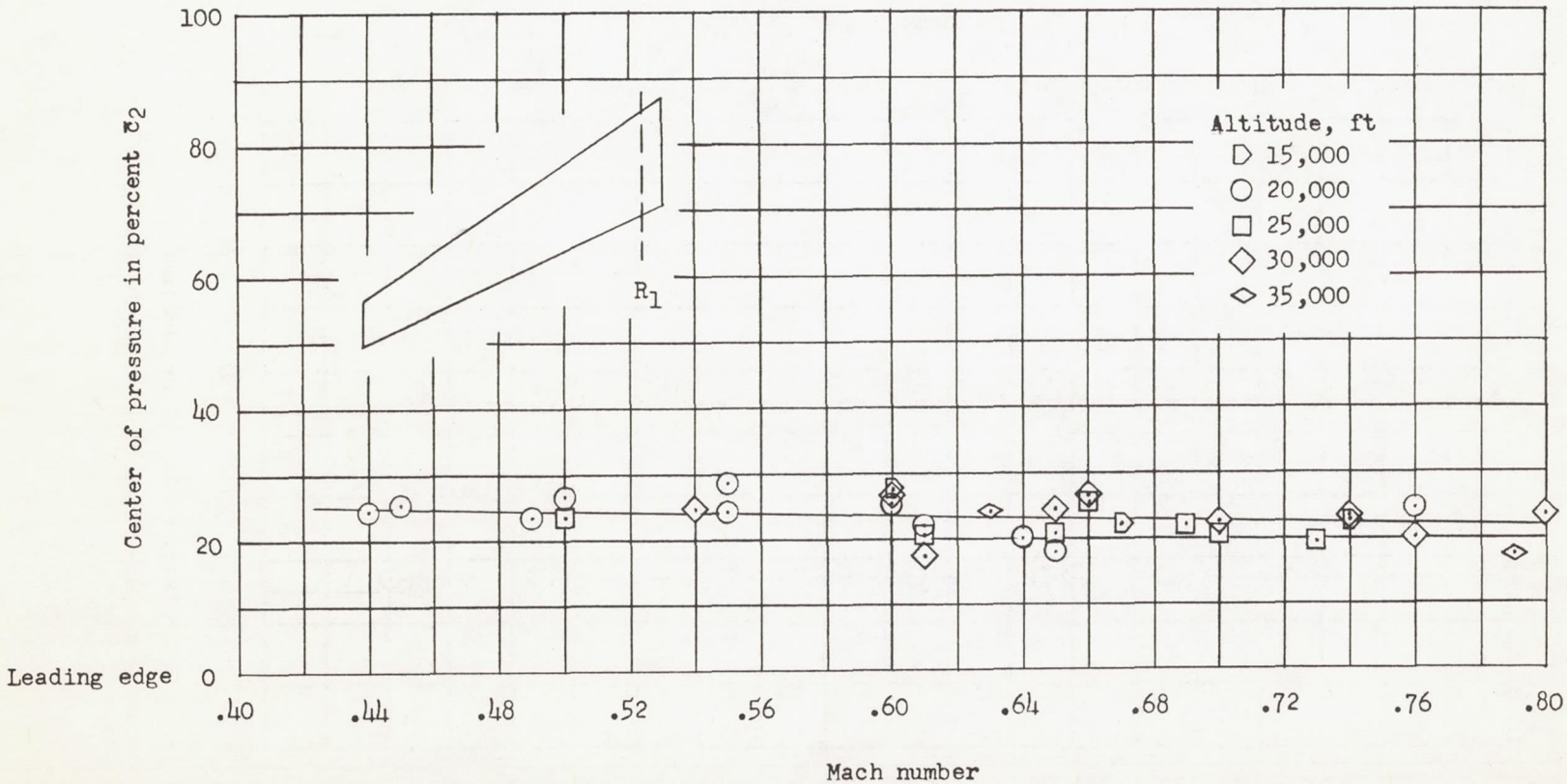

Figure 8.- Variation with Mach number of the chordwise center of pressures of the additional air loads acting outboard of reference axis R_1.

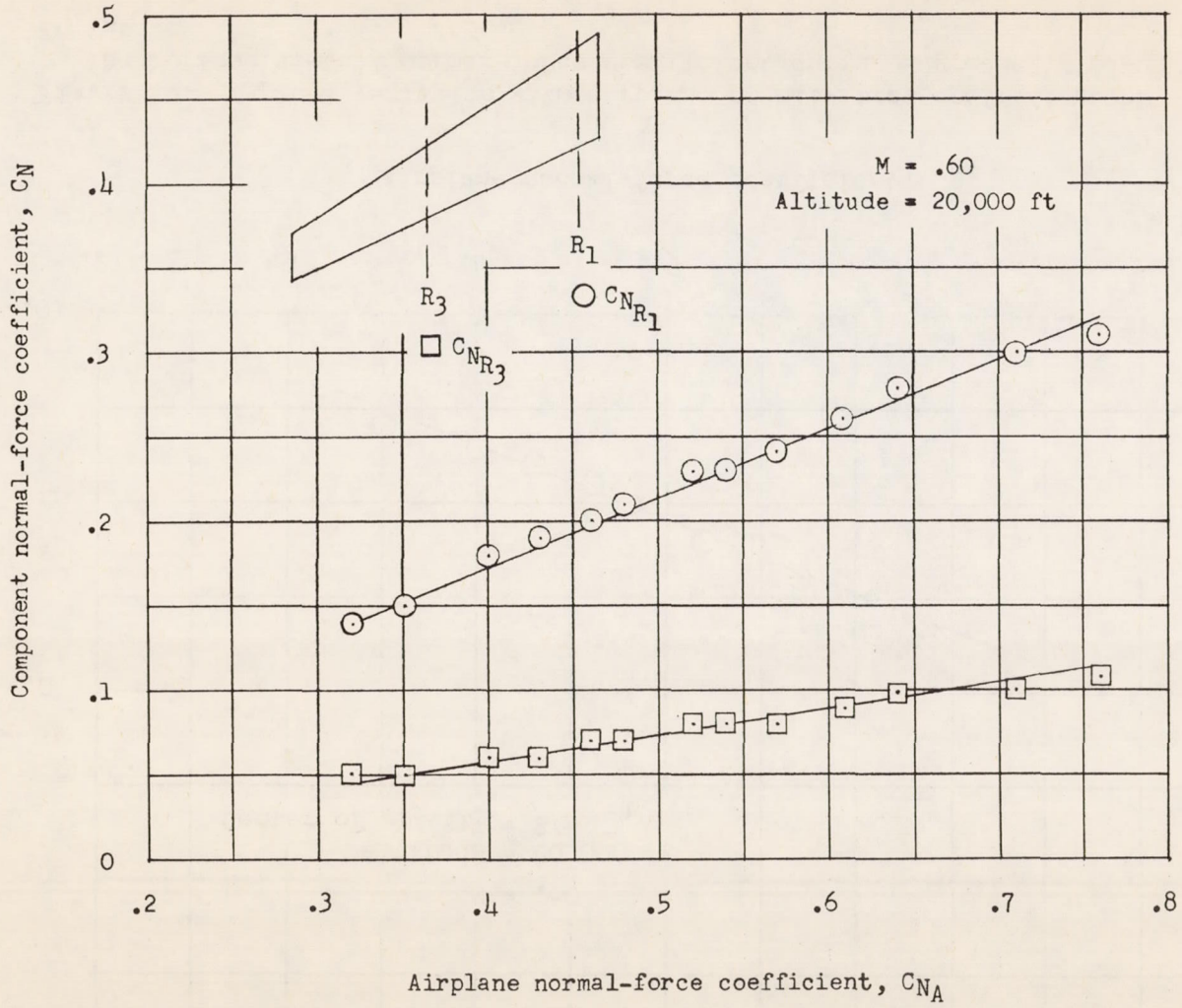

Figure 9.- Typical variation with airplane normal-force coefficient of the normal-force coefficients of the loads acting outboard of reference axis R_1 and of axis R_3.

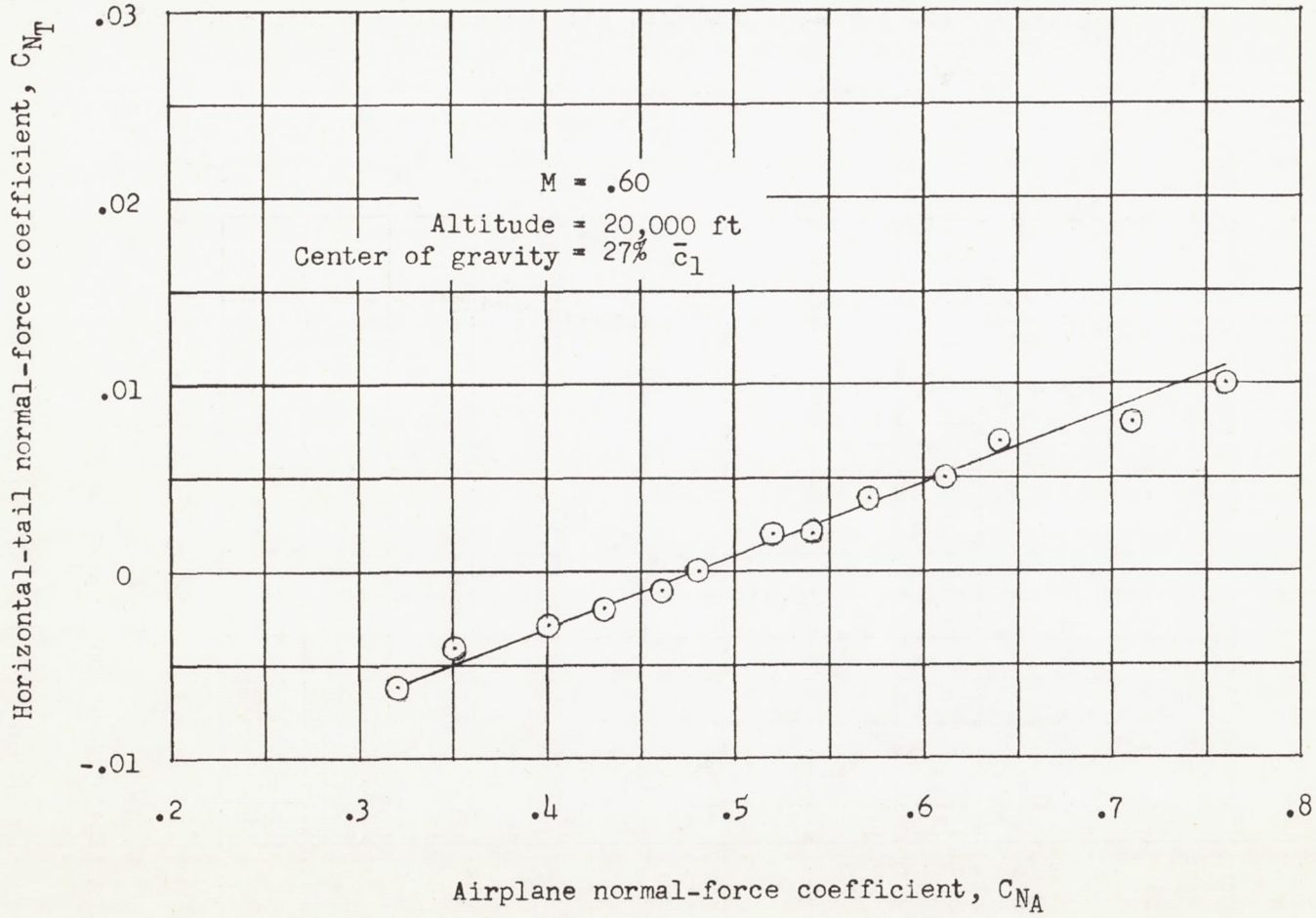

Figure 10.- Typical variation with airplane normal-force coefficient of the horizontal-tail normal-force coefficient during a gradual wind-up turn.

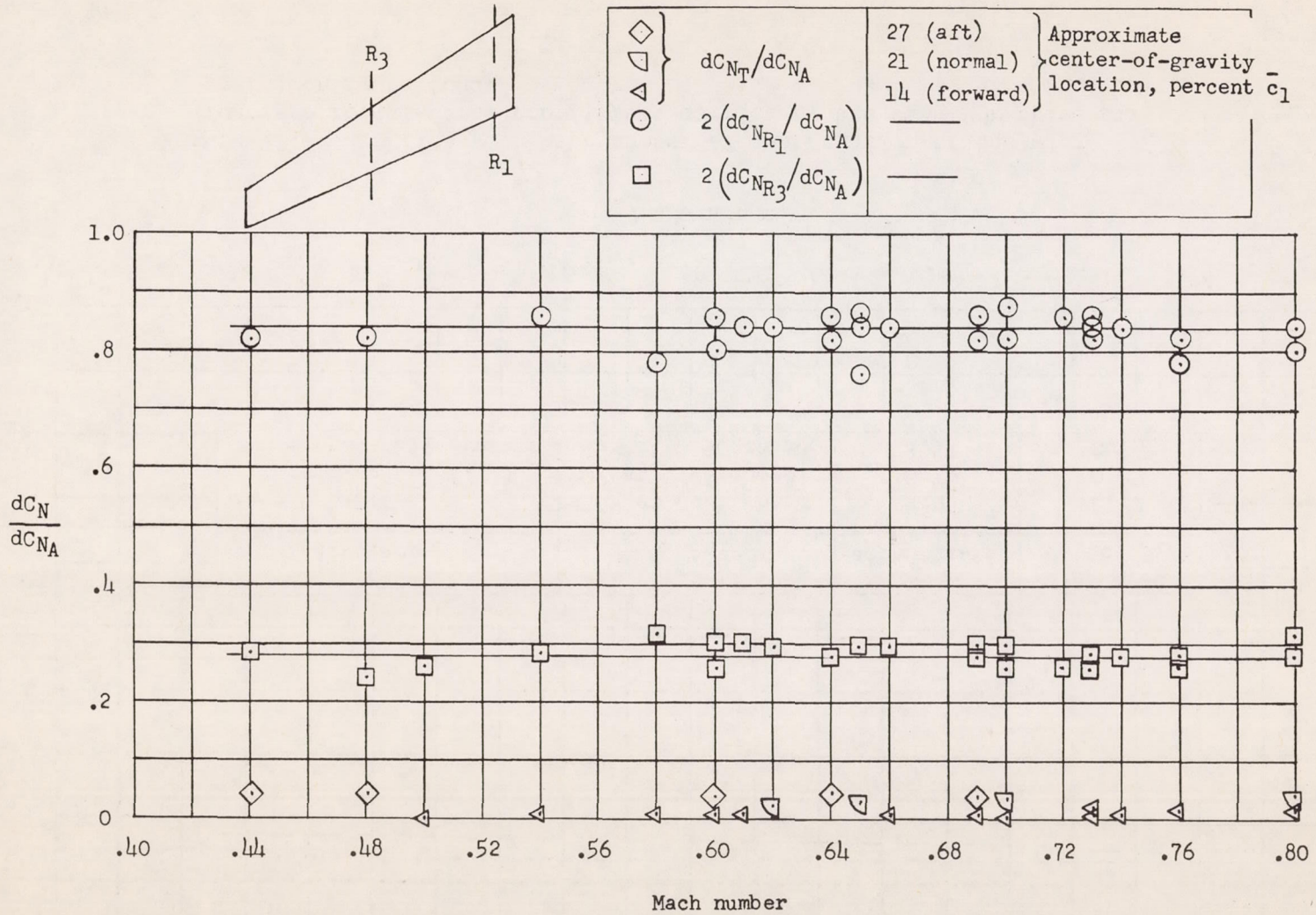

Figure 11.- Variation with Mach number of the fractions of the total airplane normal-force coefficient carried on the wing and tail.

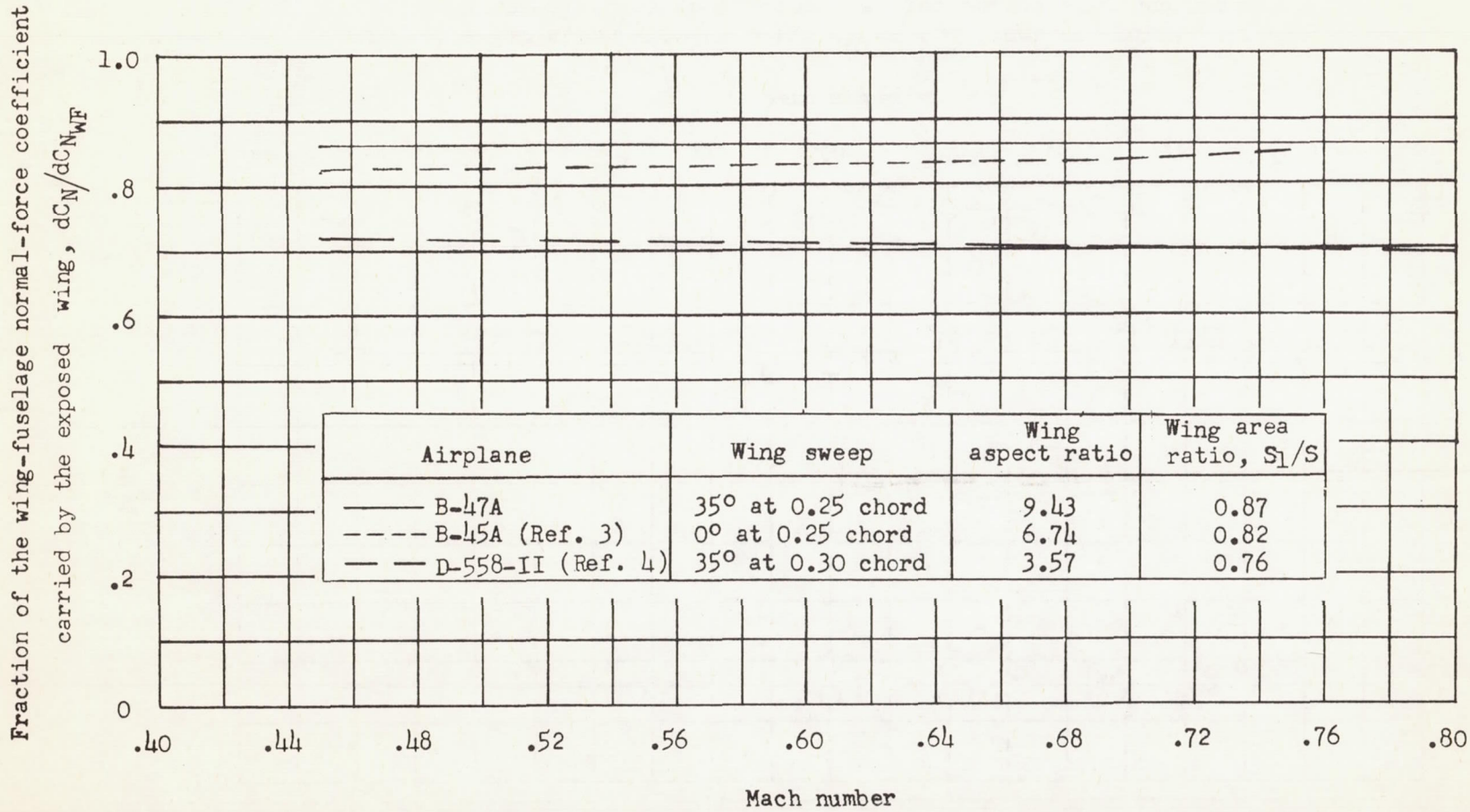

Figure 12.- Variation with Mach number of the fraction of the wing-fuselage normal-force coefficient carried by the wing for three aircraft configurations.