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TECHNICAL NOTE

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FLIGHT CALIBRATIONS OF FUSELAGE STATIC-PRESSURE-VENT INSTALLATIONS FOR THREE TYPES OF TRANSPORTS

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SUMMARY

The flight calibrations of the copilot's fuselage static-pressure-vent installations for 16 different airplanes representing 3 types of military transports showed that, for the cruise condition, the sea-level values of the altitude error of the static-pressure source differed from the manufacturers' calibrations by as much as 30 feet for a two-engine piston-powered airplane, 105 feet for a four-engine piston-powered airplane, and 65 feet for a four-engine turbojet airplane. For the cruise speeds and operational altitudes of each type of airplane, the errors of the two-engine piston-powered airplanes and the four-engine turbojet airplanes appeared small with respect to the applicable vertical separation criteria; the errors of the four-engine piston-powered airplanes, on the other hand, were considered to be unduly large in comparison with the vertical separation criterion.

INTRODUCTION

During the past several years there has developed an increasing concern that the static-pressure errors of fuselage vent installations on different airplanes of the same type can vary by significant amounts. For a given Mach number and angle of attack, the static-pressure error of a fuselage vent can vary because of depression, protrusion, or misalinement of the vent tube or plate (ref. 1) or because of variations in fuselage surface contour (due to fuselage skin wrinkling, damage to the skin, or repairs following skin damage) in the vicinity of the vents (refs. 2 and 3).

In 1956 the static-pressure errors of a variety of civil and military aircraft were determined by means of the pacer-aircraft method (ref. 4). Although the tests disclosed large error differences among airplanes of the same type (as much as 200 feet at an altitude of 10,000 feet), the accuracy of the test method (±53 feet) was of the same order as many of the error differences reported.

In order to obtain more precise information on the differences in the static-pressure error of fuselage vent installations on different transports of the same type, the National Aeronautics and Space Administration has calibrated the service installations of four airplanes of one type and six each of two other types of military transports by means of a technique using a ground camera and a statoscope. This paper presents the results of these flight tests, which are discussed with respect to their significance relative to airplane-altitude separation criteria.

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SYMBOLS

Δp	static-pressure error, $\Delta p_m - \Delta p_c$, lb/sq ft
Δp_m	differential pressure measured by statoscope, lb/sq ft
Δp_c	calculated differential pressure, $-g\rho\Delta h$, lb/sq ft
g	acceleration due to gravity at test site, ft/sec^2
ρ	calculated air density at midpoint of height increment Δh, slugs/ft^3
Δh	height increase from position of statoscope with airplane on ground to position with airplane at test altitude, ft
ρ_s, p_s, T_s	density, pressure, and temperature, respectively, of standard atmosphere at altitude $\Delta h/2$
p	calculated pressure at altitude $\Delta h/2$, lb/sq ft
T	calculated temperature at altitude $\Delta h/2$, $^{\circ}$F abs
ΔH	pressure-altitude increment at sea level, equivalent to Δp, ft
ΔH_a	pressure-altitude increment at altitude, equivalent to Δp, ft
V_i	indicated airspeed corrected for instrument error, knots
M	Mach number

INSTRUMENTATION, TEST METHOD, AND DATA REDUCTION

Instrumentation

The measured pressure difference between the position of the airplane on the ground and its position at the test flight level was measured with an NASA recording statoscope (fig. 1(a)) which was connected to the static-pressure vent of the copilot's system. As shown in figure 1(b), the statoscope consists of a thermostatically controlled air chamber which was connected to the static-pressure source and which was sealed when the airplane was at rest on the ground immediately prior to the tests. In flight, the sensitive differential pressure capsule (fig. 1(b)) measured the difference between the pressure sensed by the static-pressure source and the pressure in the sealed air chamber. Thus, the statoscope measured the sum of (1) the actual pressure increment between the position of the airplane on the ground and at the flight level over the camera station and (2) the difference between the pressure sensed by the static-pressure source and the pressure at the flight level - that is, the static-pressure error. The approximate location on the fuselage of the static-pressure vents are shown in figure 2 for the three types of airplanes used in the investigation. The height of the wing-tip plane of the airplane in flight over the camera station was determined by photographing the airplane with a single-exposure, 5- by 5-inch aerial camera having its optic axis alined with the vertical (fig. 3). Details of the optical measurements and the technique used for these measurements are given in reference 5. The camera and the statoscope records were synchronized by means of a marker lamp in the statoscope which was manually actuated when a radio signal was transmitted by the camera operator at the instant he photographed the airplane.

Test Method

The calibrations of the fuselage static-pressure-vent installations in the present investigation were determined from constant-speed low-level flights over the ground camera station.

Immediately prior to the take-off for each test flight, the reference volume of air in the statoscope was sealed at a taxiway location, the elevation of which was known. The nine test runs were then made in as short a time period as possible to minimize the variation of the atmospheric pressure and the variation of the pressure in the air chamber due to instrument temperature variation during the test period. Immediately following the test runs, the airplane was returned to its initial position on the ground and a measure was obtained of any pressure difference between the pressure in the statoscope and the existing atmospheric

pressure at that time. For the tests where the statoscope reference pressure and/or the atmospheric pressure varied during the test period, corrections were applied to the differential pressure measured by the statoscope.

Data Reduction

The static-pressure error Δp was determined to be the difference between the pressure increment Δp_m measured by the statoscope and the calculated pressure increment Δp_c between the ground and flight test altitude.

This calculated pressure differential Δp_c was determined from the following relation:

$$\Delta p_c = -g\rho\Delta h$$

where g is the acceleration of gravity at the test site, ρ is the air density at the midpoint of the height increment Δh, and Δh is the height increase between the position of the statoscope with the airplane on the ground and with the airplane at the test flight level.

The density ρ was computed from the following equation:

$$\rho = \rho_s \frac{p}{p_s} \frac{T_s}{T}$$

where ρ_s, p_s, and T_s are the density, pressure, and temperature in the standard atmosphere at a height one-half of Δh, and p and T are the calculated values of the pressure and temperature at the same height. The calculated values of p and T were based on the values of pressure and temperature measured on the ground at the camera station and the assumption of a standard pressure and temperature gradient with altitude.

The value of Δh was determined from the elevation of the taxiway where the statoscope was sealed, the height of the wing-tip plane of the airplane above the taxiway, and the height of the wing-tip plane above the ground camera.

The static-pressure error, determined from the relation

$$\Delta p = \Delta p_m - \Delta p_c$$

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was converted to a pressure-altitude increment for sea-level conditions ΔH. From a consideration of the probable maximum (99.7 percent probability) value of each of the errors that determine the overall errors of Δp_m and Δp_c, the probable maximum error of ΔH was calculated to be ±12 feet. The possible errors in the measurement of Δp_m derive from (1) statoscope pressure cell and film reading error, (2) error in statoscope pressure due to synchronization error with small rate of change of pressure occurring, and (3) error due to temperature variation of reference air volume. The possible errors in the measurement of Δp_c derive from (1) camera-film reading error (2) and error in density ρ due to errors in temperature and pressure measurements and standard atmospheric gradient assumption.

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AIRPLANES AND TEST PROGRAM

The airplanes used in the present program included four of a two-engine piston-powered transport (type A), six of a four-engine piston-powered transport (type B), and six of a four-engine turbojet-powered transport (type C). Information on the size, test weight, age, and service life of each of the airplanes is given in table I. For the tests, each airplane was flown in the cruise condition at three airspeeds: one near the minimum speed, one near the maximum (placard) speed for level flight, and one at an intermediate speed. The specific speeds for each type of airplane are given in table I. The sequence of the test speeds was low, medium, and high with the sequence being repeated twice. Each test run was conducted at a constant indicated airspeed and at a constant indicated altitude of approximately 500 feet over the camera station.

The indicated airspeed at which each test run was performed was determined from the pilot's airspeed indicator, which was calibrated to determine its scale error at the test airspeed.

RESULTS AND DISCUSSION

The calibrations of the fuselage static-pressure-vent installations for types A, B, and C airplanes are presented in figures 4(a), 4(b), and 4(c), respectively. For each airplane the static-pressure error is presented in terms of the corresponding altitude error at sea level ΔH and is plotted as a function of indicated airspeed V_i. For most of the calibrations the deviation of the data points about the least-squares straight-line fairing is usually within the estimated accuracy of ±12 feet noted previously. In some tests fewer than three data points

at the same test speed of a given airplane may be shown because of deviations in the path of the airplane over the camera station with a resulting loss of suitable photographic records.

The faired calibrations shown in figure 4 are compared with the manufacturer's sea-level calibration data in figure 5. The spread of the measured altitude errors of the airplanes of each type is indicative of the precision with which the fuselage vent installations of these airplanes can be duplicated and of the effects of changes in the installation and the surrounding fuselage area that can result from service operation. The deviations of the measured errors from the manufacturers' calibrations indicate the magnitudes of the residual errors.

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For the four type A airplanes (fig. 5(a)) the maximum spread in the altitude error determined in the present tests was about 20 feet at a speed of 140 knots and about 10 feet at a speed of 190 knots. The differences between these four type A airplane calibrations and the manufacturer's calibration ranged from about 15 feet to 30 feet over the speed range from 140 knots to 190 knots.

For the six type B airplanes (fig. 5(b)) the maximum spread of altitude errors was about 75 feet and 145 feet at speeds of 140 knots and 240 knots, respectively. The maximum differences between the manufacturer's calibration curve and the altitude errors of the present tests were -25 feet to 50 feet at the lowest test speed of 140 knots and about -40 feet to 105 feet at the highest test speed of 240 knots. The large spread in the measured errors of these airplanes is probably due to the fact that the vents are located at a position only 0.5 fuselage diameter aft of the nose (fig. 2), a position where, because of a large rate of change and point of inflection in the longitudinal pressure distribution in this region, the measured pressure is very sensitive to the small differences in location of the vents and to variations in fuselage skin contour in the vicinity of the vents.

For the six type C airplanes (fig. 5(c)) the maximum spread in the altitude errors ranges from about 20 feet at 240 knots to about 65 feet at 340 knots. The maximum differences between the manufacturer's calibration and the calibrations for the present tests were about -5 feet to -25 feet at the lowest test speed of 240 knots and about 0 feet to 65 feet at the highest speed of 340 knots. Because of the low age and service life of these airplanes (table I), the spread of the measured errors provides an indication of present-day manufacturing tolerances for duplicating fuselage vent installations (vent assembly and surrounding skin contours).

APPLICATION OF RESULTS TO CRUISE ALTITUDES

In order to permit ready conversion of sea-level values of the altitude errors ΔH to the corresponding errors at altitude ΔH_a, the variation of $\Delta H_a/\Delta H$ with altitude is given in figure 6 for altitudes up to about 50,000 feet.

An indication of the significance of the sea-level values of the altitude errors of the airplanes used in the present investigation is given for each of the three types of airplanes at their cruise speeds and altitudes. This extrapolation does not take into account the effect of small changes in airplane angle of attack with altitude.

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For the type A airplane the normal cruise altitude is about 10,000 feet and the normal indicated airspeed for cruise at this altitude is 190 knots. The largest difference between the measured errors and the manufacturer's calibration at this speed is 25 feet (airplane A-1 in fig. 5(a)). The error at 10,000 feet that would correspond to the sea-level value would be about 35 feet (25×1.35, the value of $\Delta H_a/\Delta H$ for 10,000 feet from fig. 6). The minimum vertical separation limit for operation under combined visual and instrument flight rules at 10,000 feet is 500 feet; if it is assumed that, for safe operation, aircraft should remain within one-half of the separation minimum, the 35-foot residual error from the static-pressure source is satisfactorily small for operation within a ±250-foot bracket.

For the type B airplanes, the normal cruise speed is 220 knots at the cruise altitude of 18,000 feet. The maximum variation of the measured errors from the manufacturer's value at this speed is 95 feet (airplane B-5 in fig. 5(b)) and the corresponding error at 18,000 feet is about 165 feet. Since the vertical separation limits at 18,000 feet are the same as those at 10,000 feet, the assumed safe operational bracket of ±250 feet is the same. The residual error of 165 feet for the type B airplanes would appear to represent an unduly large portion of this bracket.

For the type C airplanes, the cruise altitude range is between 30,000 and 40,000 feet. At an altitude of 30,000 feet, the calibration of this type of airplane, as determined by the manufacturer, changes direction at a Mach number of 0.85 and thereafter increases at a rapid rate with increasing speed. For this reason the calibration data for the present tests can be evaluated only for the Mach number range below the reversal point. At this Mach number, the indicated airspeed at 35,000 feet is 291 knots; the maximum deviation of the measured errors from the manufacturer's calibration at this speed is 31 feet (airplane C-3 in fig. 5(c)) and the corresponding error at 35,000 feet is about 105 feet.

Since the minimum vertical separation limit for combined visual and instrument flight rules at altitudes above 30,000 feet is 1,000 feet, the 105-foot residual error would appear to be sufficiently small for safe operation within 500 feet, which is one-half of the separation limit. However, whether the residual errors would remain within acceptable limits at $M > 0.85$ cannot, of course, be stated from the sea-level data of the present tests.

CONCLUDING REMARKS

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The low altitude calibrations of the fuselage static-pressure-vent installations of four airplanes of a two-engine piston-powered transport, six airplanes of a four-engine piston-powered transport, and six airplanes of a four-engine turbojet transport showed that, for the cruise condition, the sea-level values of the altitude error of the static-pressure source differed from the manufacturer's calibrations by as much as 30 feet for the two-engine piston transports, 105 feet for the four-engine piston transports, and 65 feet for the four-engine turbojet transports.

Extrapolation of the low-altitude errors to the cruise altitudes for the three types of airplanes to show their significance relative to airplane separation standards indicated that for the two-engine piston transport the maximum difference between the measured errors and the manufacturer's calibration would correspond to an error of 35 feet for the normal cruise speed and altitude (10,000 feet) of this airplane; this 35-foot error is small with respect to 250 feet, which is one-half of the minimum vertical separation limit of 500 feet for combined visual and instrument flight rules at this altitude. For the four-engine piston transport, the largest variation of the measured errors from the manufacturer's calibration would correspond to an error of about 165 feet at the normal cruise speed and altitude (18,000 feet) of this airplane; this error would appear to be unduly large with respect to 250 feet, which is one-half of the 500-foot minimum vertical separation limit. For the four-engine turbojet transport, the maximum variation of the measured errors from the manufacturer's calibration would correspond to an error of about 105 feet for a Mach number equal to or less than 0.85 at 35,000 feet; this error is small with respect to 500 feet, which is one-half of the 1,000-foot minimum vertical separation limit at this altitude.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., March 28, 1962.

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5. Gracey, William, Jewel, Joseph W., Jr., and Carpenter, Gene T.: Measurement of the Errors of Service Altimeter Installations During Landing-Approach and Take-Off Operations. NASA TN D-463, 1960.

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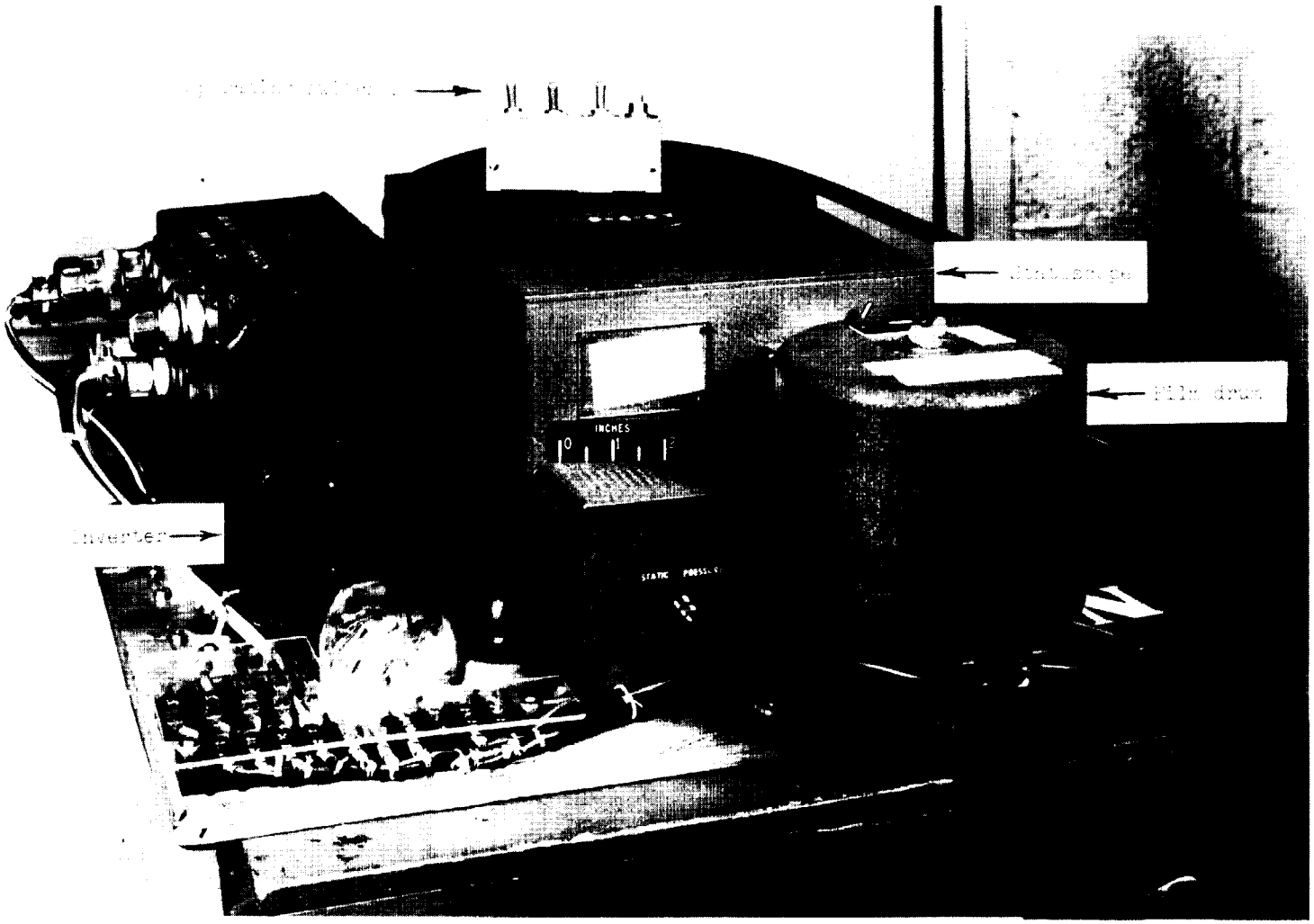
TABLE I

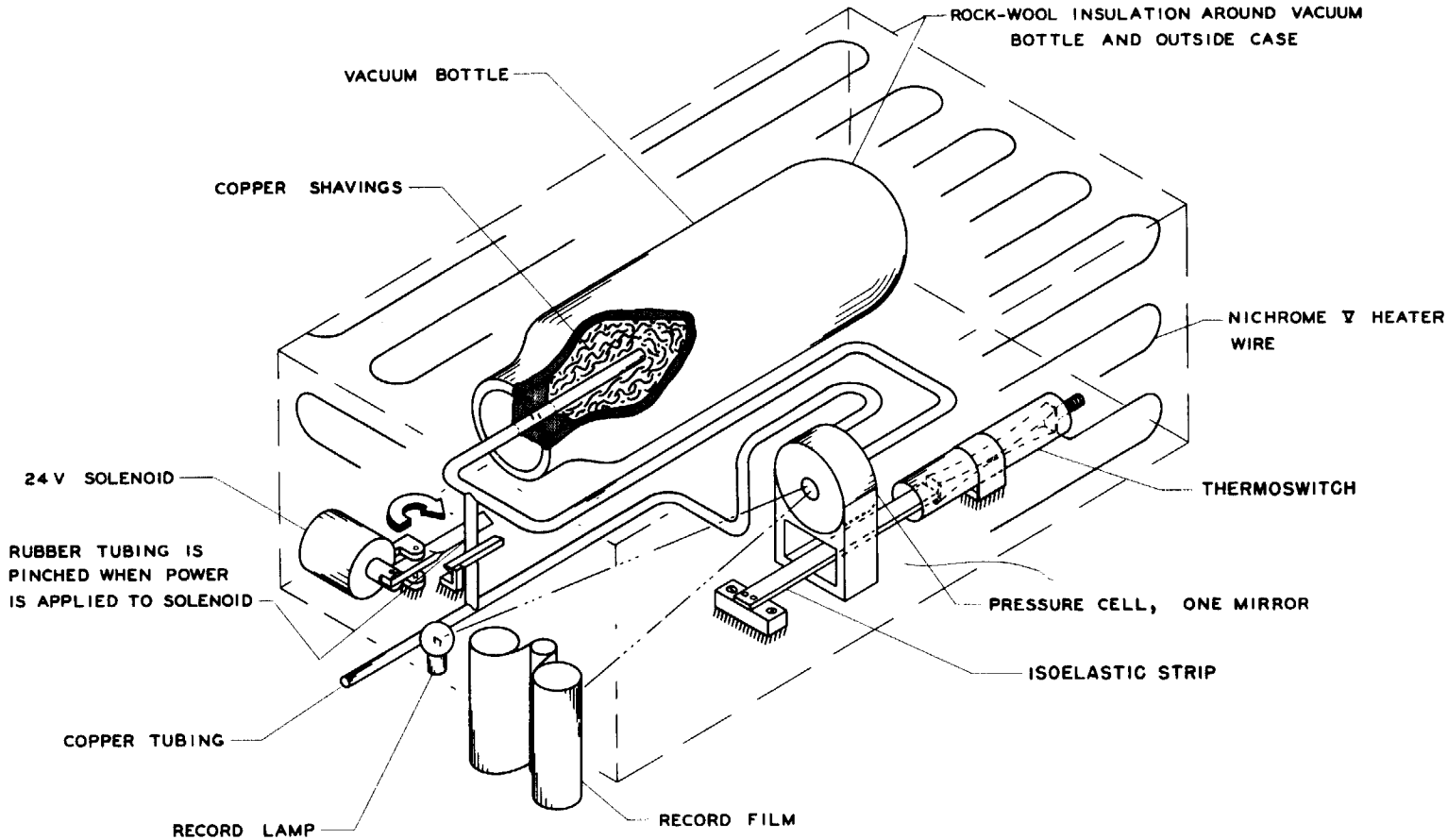
GENERAL DATA FOR AIRPLANES USED IN FLIGHT TESTS

Airplane type	Wing span, ft	Number of engines	Type engine	Nominal test speeds, knots	Identification number	Approximate test take-off weight, lb	Airplane delivery date	Flight operation prior to tests, hr
A	91.8	2	Piston	140, 170, 190	A-1	38,500	Mar. 1954	8,307
					A-2	38,000	May 1954	8,946
					A-3	40,000	July 1954	9,151
					A-4	38,000	Oct. 1954	8,884
B	117.5	4	Piston	140, 190, 240	B-1	83,000	Aug. 1952	8,979
					B-2	85,000	Aug. 1952	9,785
					B-3	85,500	Oct. 1952	12,111
					B-4	90,000	Nov. 1952	10,761
					B-5	87,000	May 1955	11,488
					B-6	85,000	June 1955	11,716
C	130.8	4	Turbojet	240, 290, 340	C-1	193,000	Aug. 1961	390
					C-2	193,000	Aug. 1961	372
					C-3	193,000	Sept. 1961	224
					C-4	193,000	Nov. 1961	91
					C-5	195,000	Nov. 1961	85
					C-6	195,000	Nov. 1961	30



(a) Auxiliary equipment. L-62-456.1

Figure 1.- Statoscope and auxiliary equipment.



NASA STATOSCOPE

(b) Sketch showing details of statoscope construction. L-60961.1

Figure 1.- Concluded.

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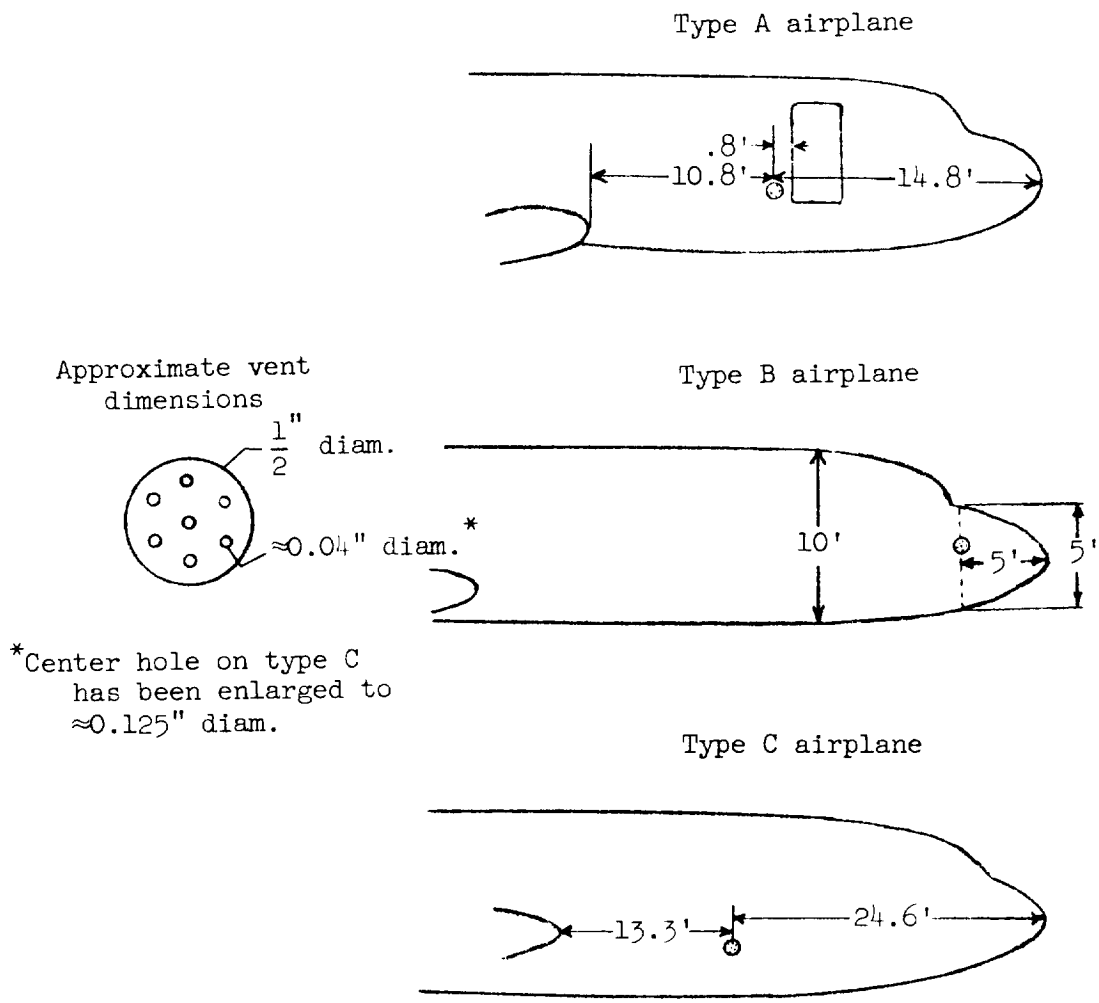

Note: Static-pressure vents are located on both sides of fuselage and teed together

Figure 2.- Sketch showing size and approximate location of copilot's static-pressure vent in relation to major sources of interference.

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Figure 3.- Aerial-type camera used to measure geometric height of airplane. Thermometer (in cage) and sensitive altimeter (for barometric-pressure deviations) also shown.

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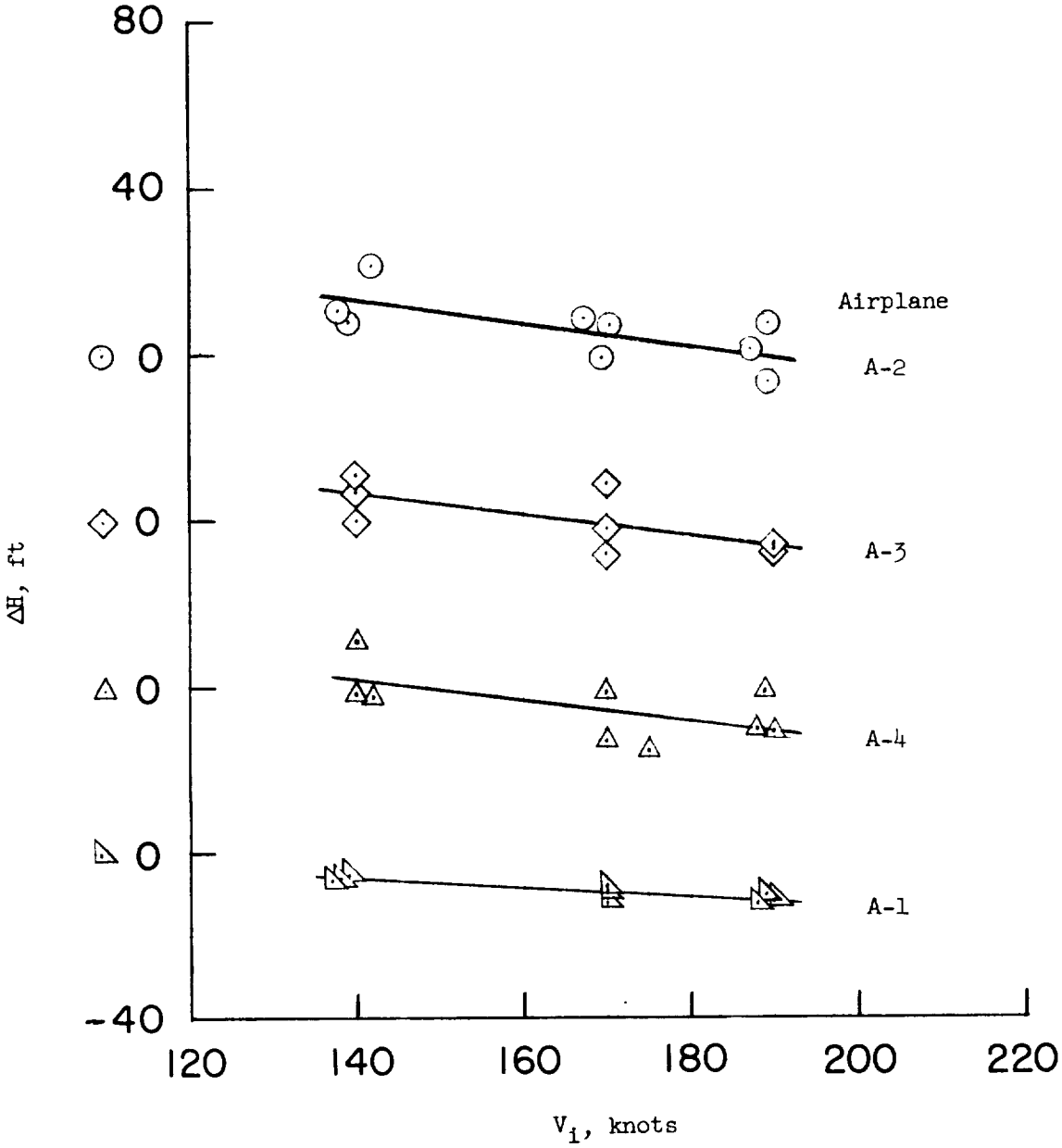
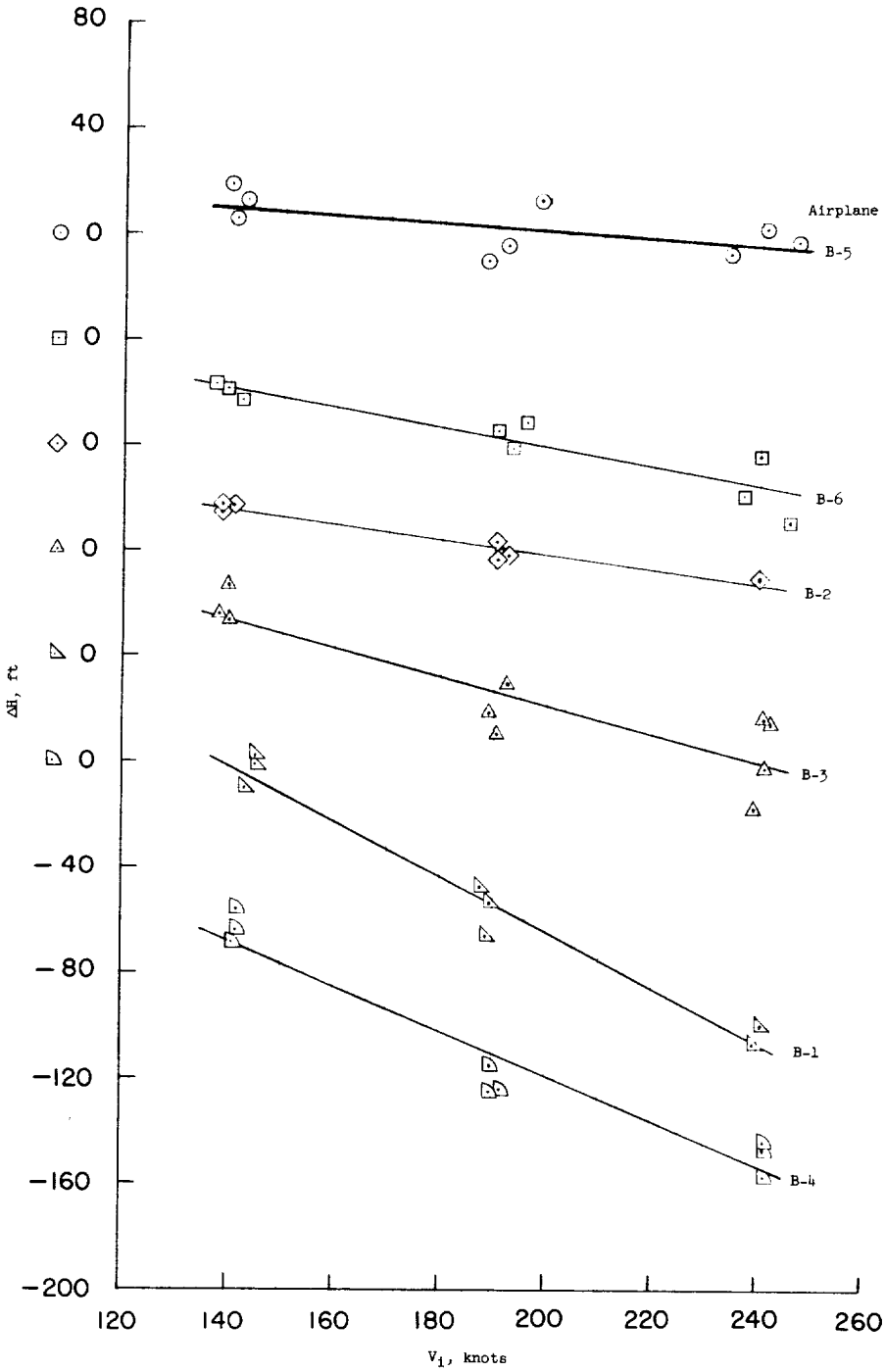


(a) Type A airplanes.

Figure 4.- Variation of copilot's static-pressure-system error with indicated airspeed for three types of transports.



(b) Type B airplanes.

Figure 4.- Continued.

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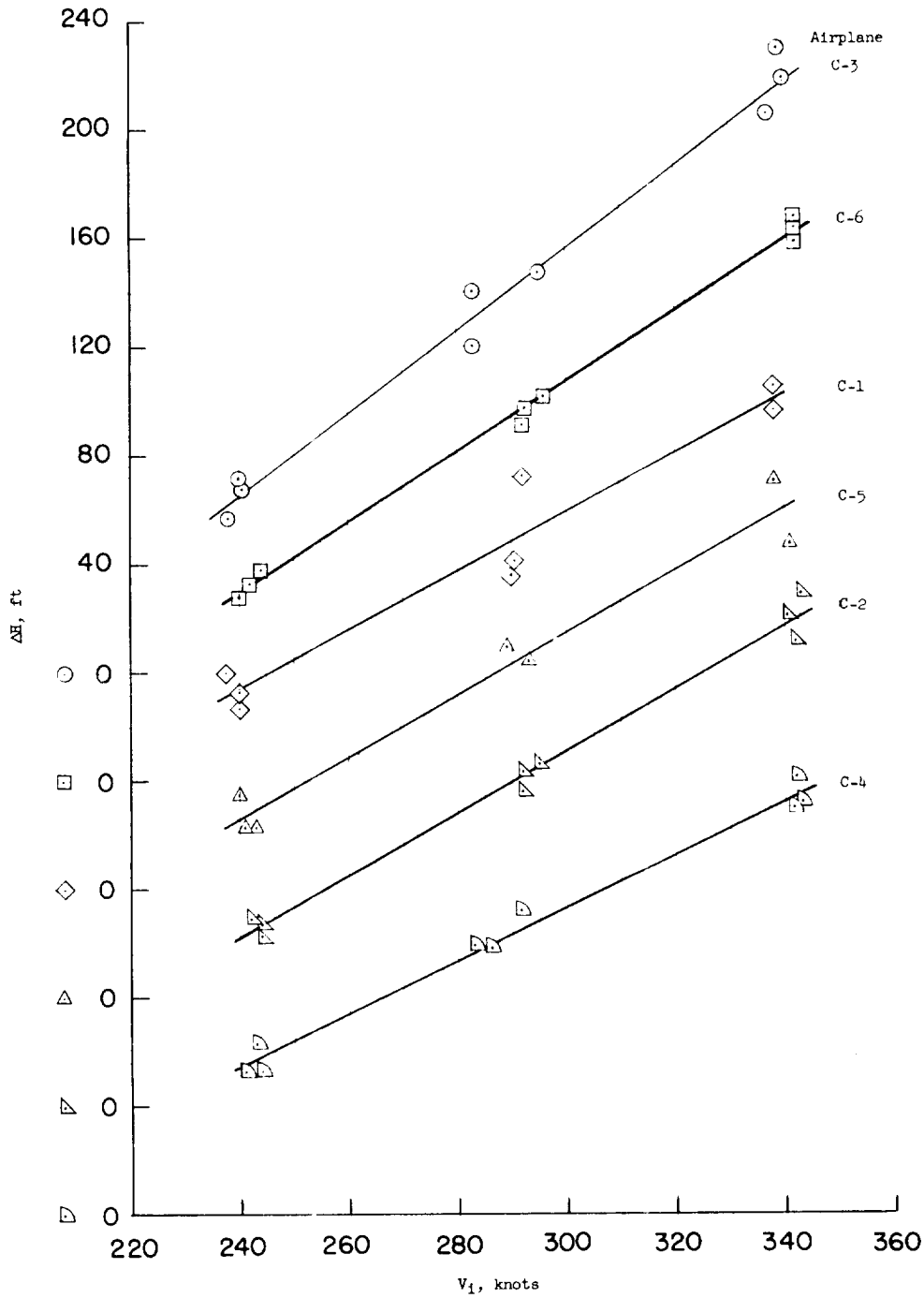
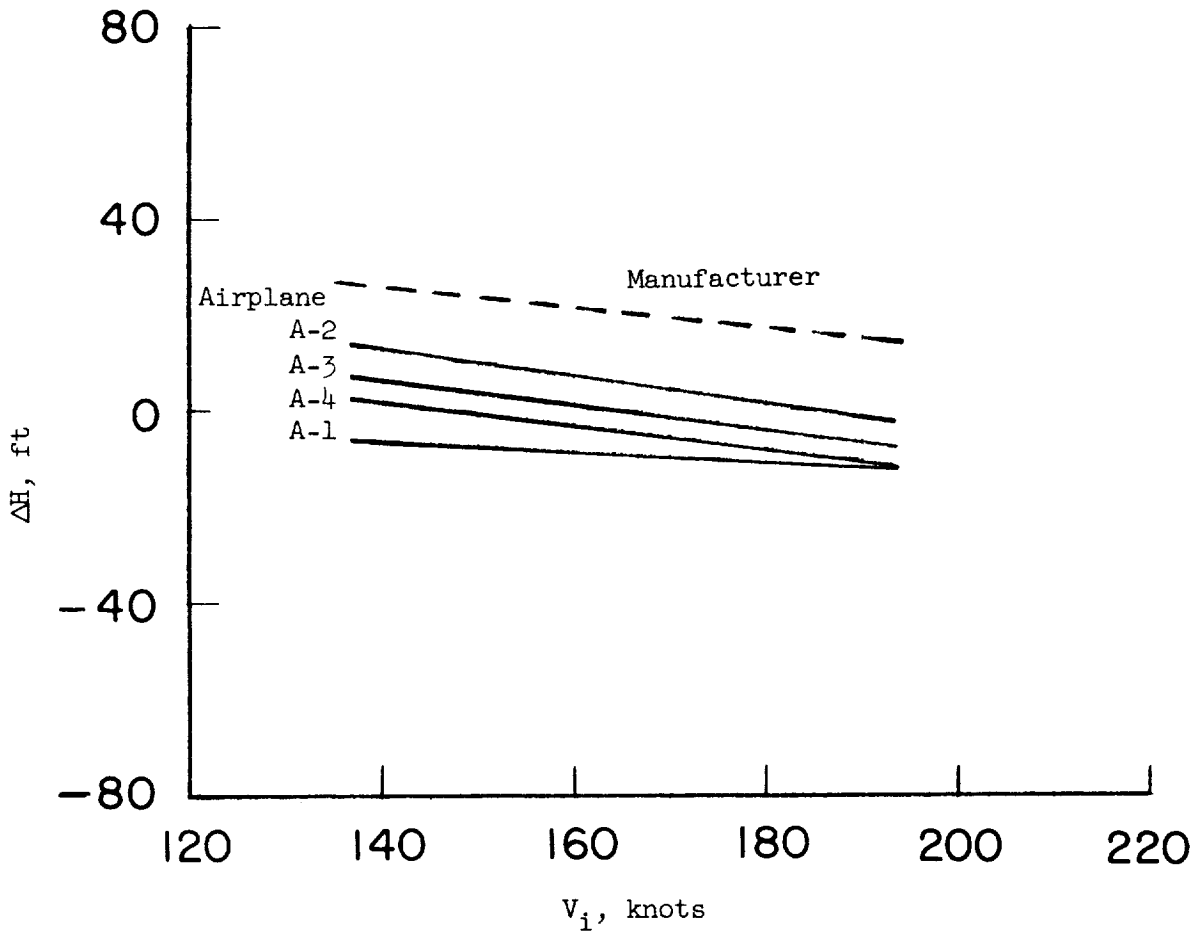


(c) Type C airplanes.

Figure 4.- Concluded.



(a) Type A airplanes.

Figure 5.- Comparison of the copilot's static-pressure-system error with the manufacturer's calibration for the three types of airplanes.

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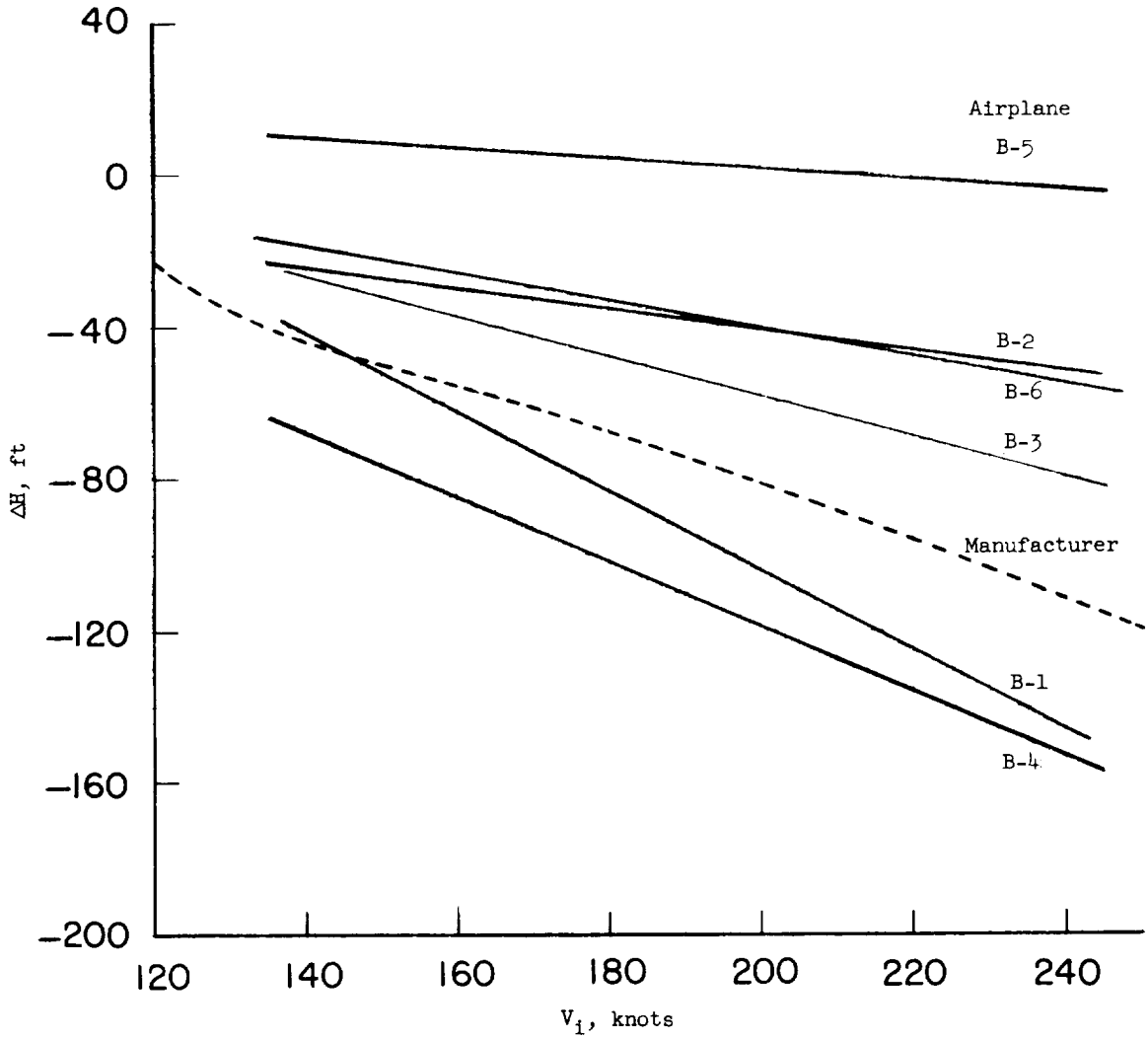
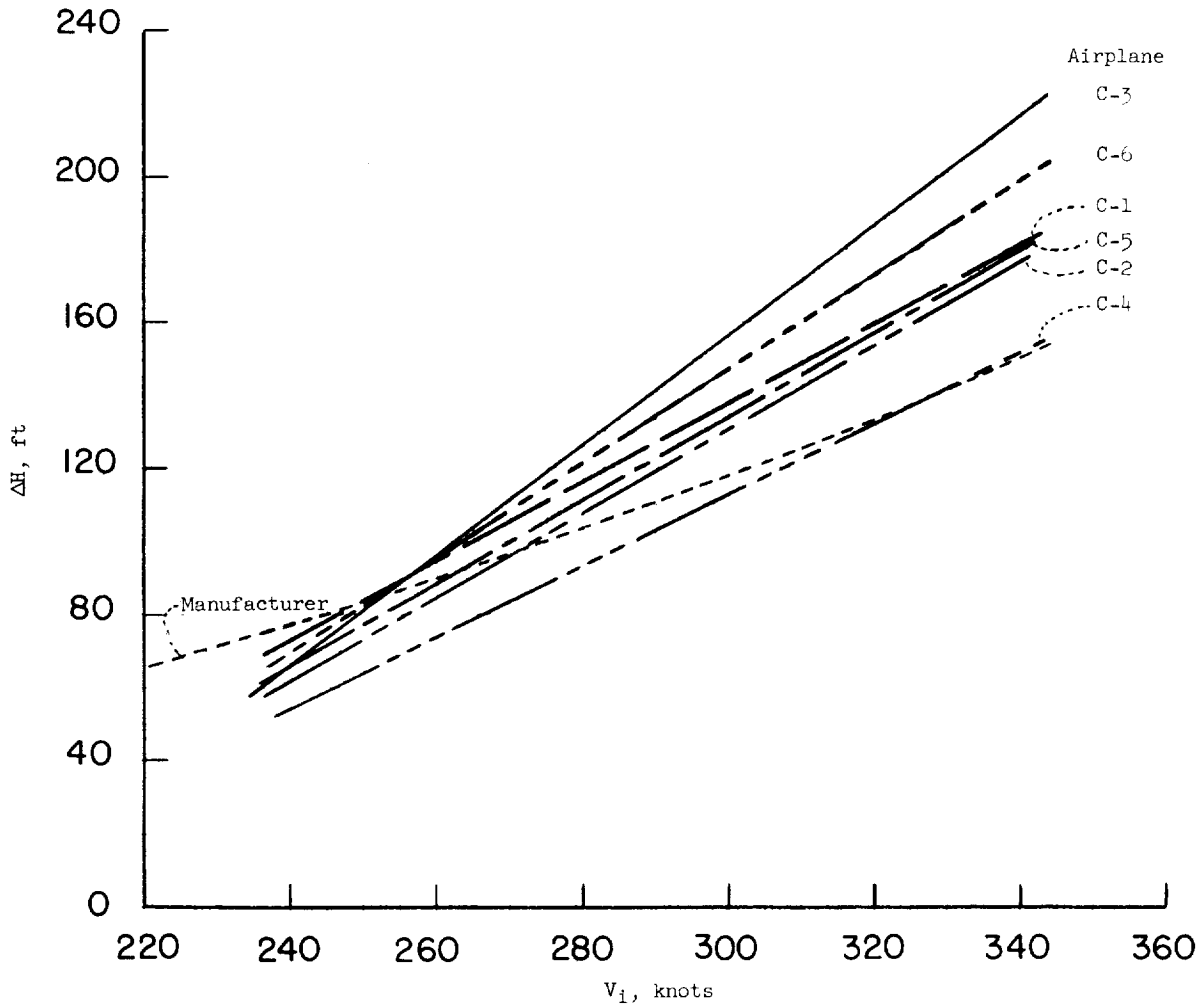


(b) Type B airplanes.

Figure 5.- Continued.



(c) Type C airplanes.

Figure 5.- Concluded.

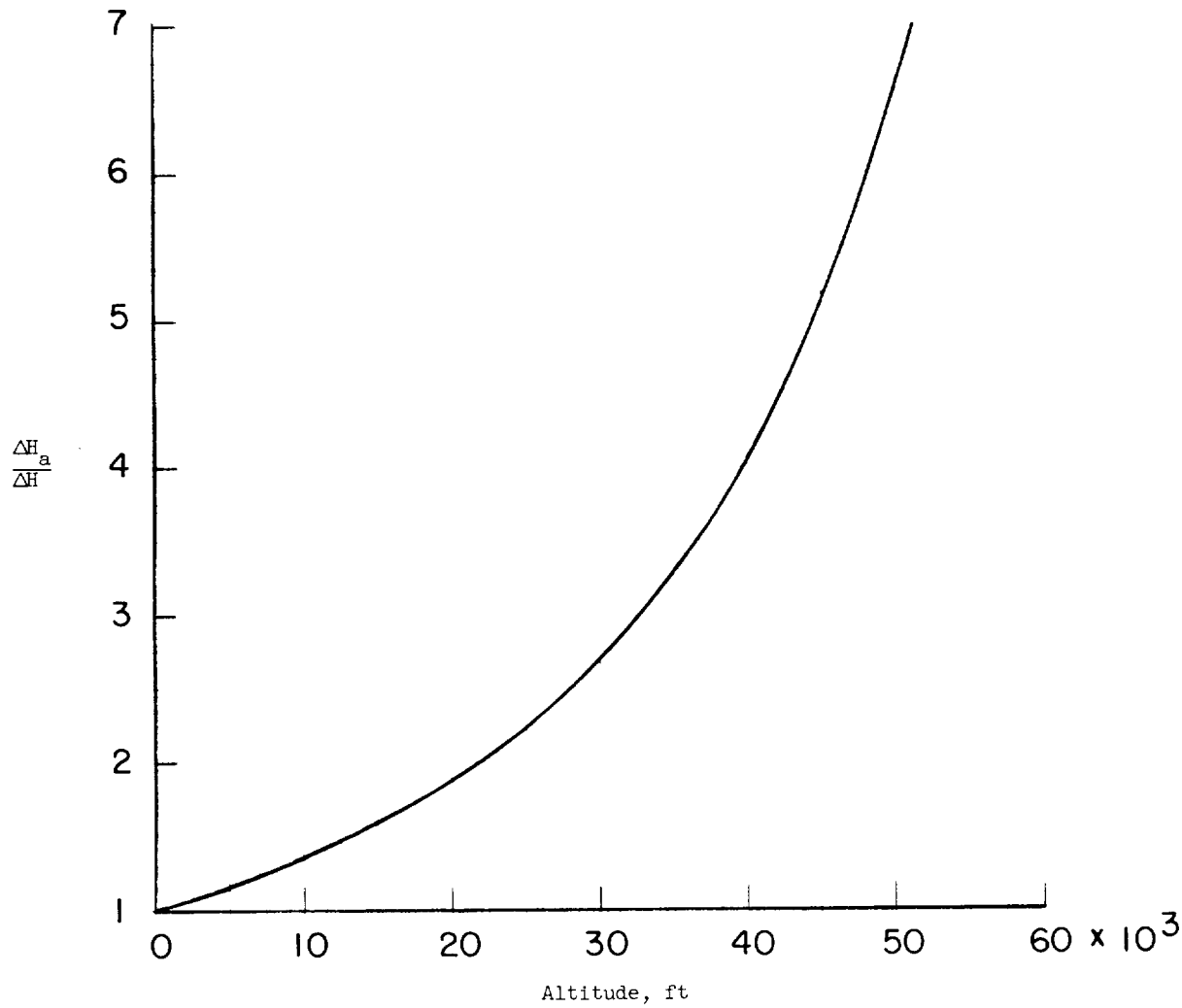


Figure 6.- Variation of $\frac{\Delta H_a}{\Delta H}$ with altitude for the standard atmosphere.

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