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DECEMBER 1962

HELICOPTER AIRSPEED SYSTEM INVESTIGATIONS

PROJECT ENGINEER:
JOHN R. MELTON
2/LT TC

PROJECT PILOT:
JOHN A. JOHNSTON
CAPTAIN TC

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preface

This report is a technical note which presents the results of a test or investigation that was limited in scope. Its purpose is to increase the fund of information available on a particular subject and is not intended to be considered as a complete or all-inclusive work.

This report has been reviewed and approved

RICHARD J. KENNEDY, JR.
LT. COLONEL, TC
CHIEF

TABLE OF CONTENTS

PHOTO I - TEST AIRCRAFT H-23D "RAVEN"
equipped with test instrumentation including a telescoping airspeed boom.

I. GENERAL

A series of projects was conducted during the months of May through August, 1962, by the U.S. Army Aviation Test Activity, Test and Evaluation Command, at Edwards Air Force Base, California. The objectives of these projects were: (1) to obtain values of the error induced by the effect of volume balance in a pitot-static airspeed system while changing altitude at various airspeeds and rates; (2) to determine the effect of airspeed boom length upon airspeed position error caused by the pressure field surrounding the aircraft; and (3) to determine if an airspeed system volume balanced at a selected altitude would require rebalancing at other altitudes.

The aircraft used to conduct these tests was an H-23D "Raven," modified by installation of test instrumentation. The installation is described in Appendix III.

II. AIRSPEED ERROR WHILE CHANGING ALTITUDE DUE TO VOLUME UNBALANCE

A. Summary

The airspeed errors observed during these tests ranged in value from 0.25 knots IAS to a maximum of 2.5 knots IAS. The test data were obtained during descents at rates between approximately 150 fpm and 2700 fpm at airspeeds of 40 knots IAS, 60 knots IAS, and 70 knots IAS. Values of airspeed errors under these conditions were obtained while controlling rate of descent through 4000 feet and 6000 feet pressure altitudes. By this method, the effects of three parameters: rate of descent, airspeed, and altitude, could be evaluated.

The most significant factor contributing to airspeed error proved to be the rate of descent. With an increasing rate of descent, a larger airspeed error was observed. At 70 knots IAS, the error increased from 1.1 knots at 800 feet per minute rate of descent to 2.25 knots at 2500 feet per minute.

An airspeed error was also affected by airspeed, but to a lesser degree than by rate of descent. At a higher airspeed and the same rate of descent, a slightly higher airspeed error was observed; i.e., approximately 0.25 knots larger error at 70 knots IAS than at 40 knots IAS.

The effect of altitude upon the airspeed error; that is, the difference between the results at 4000 feet pressure altitude and the results at 6000 feet pressure altitude, was very small. At 40 knots IAS, the airspeed error at a particular rate of descent at 4000 feet pressure altitude is a uniform 0.15 knots IAS higher than at 6000 feet pressure altitude. At 60 knots IAS, this difference in error ranged from 0.08 knots IAS to 0.18 knots IAS higher at 4000 feet pressure altitude than at 6000 feet pressure altitude. At 70 knots IAS, there was negligible difference in the airspeed error at 4000 feet pressure altitude and 6000 feet pressure altitude.

B. Introduction

An airspeed indicator measures pitot-static pressure differential. When changing altitude or airspeed, however, the pressure in the static system may change at a different rate than that in the pitot system, causing an erroneous reading. For example, in a descent, the pressures at the pitot and static sensing ports will increase. To equilibrate, a small amount of air will flow into each system. The rate of change of pressure in each system will then be a function of each system's volume and internal flow characteristics as well as the aircraft's rate of descent. Since the volume and flow characteristics of each system will probably be different, an airspeed error will be induced while changing altitude.

In a typical pitot-static instrument system, the pitot sensing port is connected to only the airspeed indicator, while the static sensing port is connected not only to the airspeed indicator but also to the altimeter and rate of climb indicator. The static pressure system generally has considerably more volume than the pitot pressure system. This means that the existing pressure in the static system will change more slowly than the existing pressure in the pitot system when the aircraft is changing altitude or airspeed. In a descent then, the static pressure will increase more slowly than the pitot pressure, giving an erroneously high airspeed reading.

C. Test Procedure

The airspeed errors were determined by establishing a variety of rates of descent through pressure altitudes of 4000 and 6000 feet and indicated airspeeds of 40, 60, and 70 knots.

Following this series, spot checks were made at a variety of climb rates at 4000 and 6000 feet pressure altitude and 40 knots IAS.

The smallest possible airspeed lag error occurs when only an airspeed indicator is connected to both the pitot and static lines. An airspeed indicator was connected in this manner and provided a standard indicated airspeed system for comparison. To measure the error caused by the extra volume of an altimeter and rate of climb indicator, a valve was installed so that these instruments could be connected to or separated from the static pressure line. By observing the indicated airspeed in a stabilized descent both with and without the altimeter and rate of climb indicator connected to the static pressure line, the error caused by the addition of these instruments could be measured.

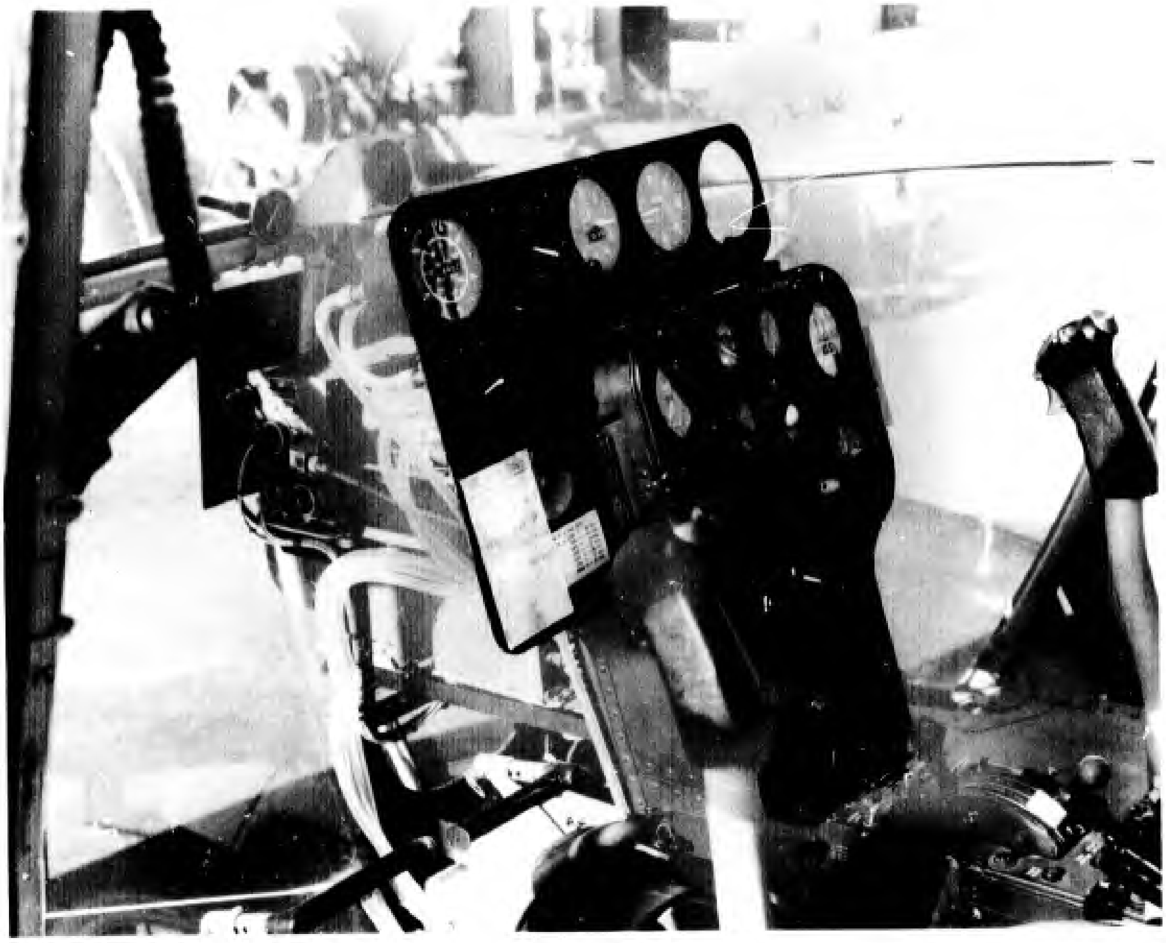


PHOTO II – COCKPIT ARRANGEMENT OF TEST AIRCRAFT
Test instrumentation is installed in auxiliary panel above and to left of standard instrument group.

D. Test Results

The results of these tests revealed that the error induced in an airspeed system during a descent is primarily dependent upon the rate of descent and is affected to a lesser degree by airspeed. The error is only slightly affected by altitude in the 4000 to 6000 foot pressure altitude range.

The results of these tests are presented in Figures 1 and 2, Appendix II. The airspeed error, Δ V (indicated), is the result of subtracting the airspeed readings taken with an altimeter and rate of climb connected to the static system from the readings taken with the altimeter rate of climb separated from the static system.

The Δ V values were the result of

readings taken at the same flight conditions and plotted against rate of descent.

At all airspeeds, the largest airspeed lag error is experienced at the highest rate of descent and the smallest airspeed error at the lowest rate of descent. The range of airspeed error in an autorotational descent through 6000 feet pressure altitude, for the airspeed range tested, was 1.25 knots IAS at 40 knots indicated airspeed to 2.5 knots IAS at 70 knots indicated airspeed.

The variation of airspeed lag error with airspeed was much less pronounced than with rate of descent. Throughout the range of rates of descent tested, the airspeed error was approximately 0.25 knots IAS higher for 70 knots IAS than for 40 knots IAS as can be seen in Figures 1 and 2.

With the test pitot-static system, the numerical value of the airspeed error in a climb was from 0.13 knots IAS to 0.38 knots IAS less than the error in a descent at the same flight conditions. The error was, of course, in the opposite direction. Due to the limited climb performance of the H-23, only a small range of rates of climb could be investigated.

The fact that the numerical value of the airspeed error is different in a climb and descent for similar flight conditions is partially explained by the fact that airflow resistance in the pitot-static system is different for flow in opposite directions.

E. Conclusions

The numerical values obtained during this test program are directly applicable only to the particular pitot-static system tested. Each system will have its own particular internal airflow characteristics and the relationship between the pitot and the static internal airflow characteristics will also be different from one system to another. The trends, therefore, discussed in this report are more significant than the numerical results.

For the purpose of the scope of these tests, it is further concluded that:

1. The airspeed error induced in a pitot-static system by the rate of climb indicator and altimeter in descending flight is primarily dependent upon and nearly proportional to the rate of descent.

2. The airspeed lag error (2.5 knots or less) is only slightly affected by airspeed in the 40 knot to 70 knot range.

3. The airspeed lag error is not significantly affected by altitude in the 4000 feet to 6000 feet pressure altitude range tested.

III. EFFECT OF ALTITUDE UPON PITOT-STATIC SYSTEM BALANCE

A. Summary

The airspeed system was balanced at 7000 feet pressure altitude by varying the volume of the pitot system. The system balance was then checked at 3000 feet pressure altitude and found to be balanced.

B. Introduction

As described in Part II,B of this report, an error will normally be seen in the airspeed indicator while changing altitude. This is due to the fact that it usually takes longer for the pressure to change in the static system than in the pitot system due to the larger volume of the static system. It is common practice to attempt to eliminate this error by the addition of extra volume in the pitot system. This extra volume is normally termed "a balance can." If the effect of the volume of the "balance can" is the same as that of the static instruments, the airspeed system is termed "balanced." The objective of this test was to determine the effect of altitude on system balance.

C. Test Procedure

To determine the effect of altitude upon airspeed system balance, the following test was performed.

A variable volume "balance can" was incorporated in the pitot system as shown in Appendix III. By turning a hand wheel on the "balance can," the volume of the pitot system could be changed. At one particular volume setting, the effect of the "balance can" volume upon the pitot system airflow characteristics will be the same as the effect of the altimeter and rate of climb indicator on the static system airflow characteristics.

A particular volume setting was determined by establishing a rate of descent and airspeed at a 7000 feet pressure altitude with only the airspeed indicator connected to the pitot-static system. When stabilized in this flight condition, both the static instruments and the "balance can" were connected to the pitot-static system. By adjusting the "balance can" to the correct volume, no change in indicated airspeed could be observed.

Using the volume that was determined for system balance at 7000 feet pressure altitude, the procedure was repeated descending through 3000 feet pressure altitude at the same indicated airspeed and rate of descent.

D. Test Results

The pitot-static system was balanced as described at 40 knots indicated airspeed and 1570 feet per minute rate of descent. Five descents through 7000 feet pressure altitude were required to determine the correct"balance can" volume.

The balance of the system was rechecked three times at 40 knots indicated airspeed and 1500 feet per minute rate of descent through 3000 feet pressure altitude.

The airspeed system was found to have balanced at 3000 feet pressure altitude.

E. Conclusions

In the 3000-7000 feet pressure altitude range tested, an airspeed system volume balanced at one altitude will remain in balance at all altitudes.

IV. EFFECT OF AIRSPEED BOOM LENGTH UPON AIRSPEED POSITION ERROR

A. Summary

The test helicopter was equipped with a telescoping airspeed boom, so that the pitot-static head could be moved closer or further from the aircraft to position it in different parts of the pressure field developed about the aircraft in forward flight.

At airspeeds above 40 knots IAS, the position error for a short boom was larger than for a long boom. Below 40 knots IAS, this correlation did not exist.

There was no significant difference observable in the consistency of results of the position error calibration. The short boom calibrations are as usable for flight test purposes as the longer boom calibrations.

B. Introduction

A pressure field is developed about an aircraft in forward flight, due to the compressibility of air. Since an airspeed indicator and altimeter are pressure sensing devices, their readings are affected by this pressure field. To accurately determine airspeed, the effect of the pressure field on a particular installation must be determined. With different airspeed boom lengths, the pressure sensing element is located in a different part of this pressure field, thus giving a different error in the instrument readings.

On flight test aircraft, it is necessary to accurately determine airspeed. The effect of the pressure field on the airspeed reading must therefore be found. Since the standard aircraft instrumentation usually has its static pressure sensing port on the surface of the aircraft where the effect of the pressure field is large and sometimes erratic, it is therefore necessary to install a boom with pressure sensing ports some distance in front of the aircraft. This test project had the objective of determining the effect of the length of the airspeed boom upon the airspeed system position error calibration.

C. Test Procedure

The test aircraft was equipped with a telescoping airspeed boom. The pressure sensing head could be located at nine different positions relative to the aircraft by lengthening or shortening the boom in one foot increments.

At each of the nine boom lengths, an airspeed position error calibration was performed utilizing the current ground speed course method. This method consisted of flying reciprocal headings over a known ground distance. At ten knot increments through the speed range of the aircraft, the time required to cover the known distance was recorded. Knowing the relationship between the true airspeed and the indicated airspeed, the position error may be calculated. For all airspeed calibrations the gross weight and C.G. were held as nearly constant as possible.

D. Test Results

The graphical results of the airspeed calibrations for each boom length are shown in Appendix II. Due to the inherent installation problems of helicopter pitot-static airspeed systems, unstable operation usually occurs in the 0-30 knot IAS range. The most significant data was collected in the 40 knot IAS to V_{max} range. In Figure 12 the maximum position error plotted is the maximum position error occuring above 30 knots IAS when the pressure field became more stabilized.

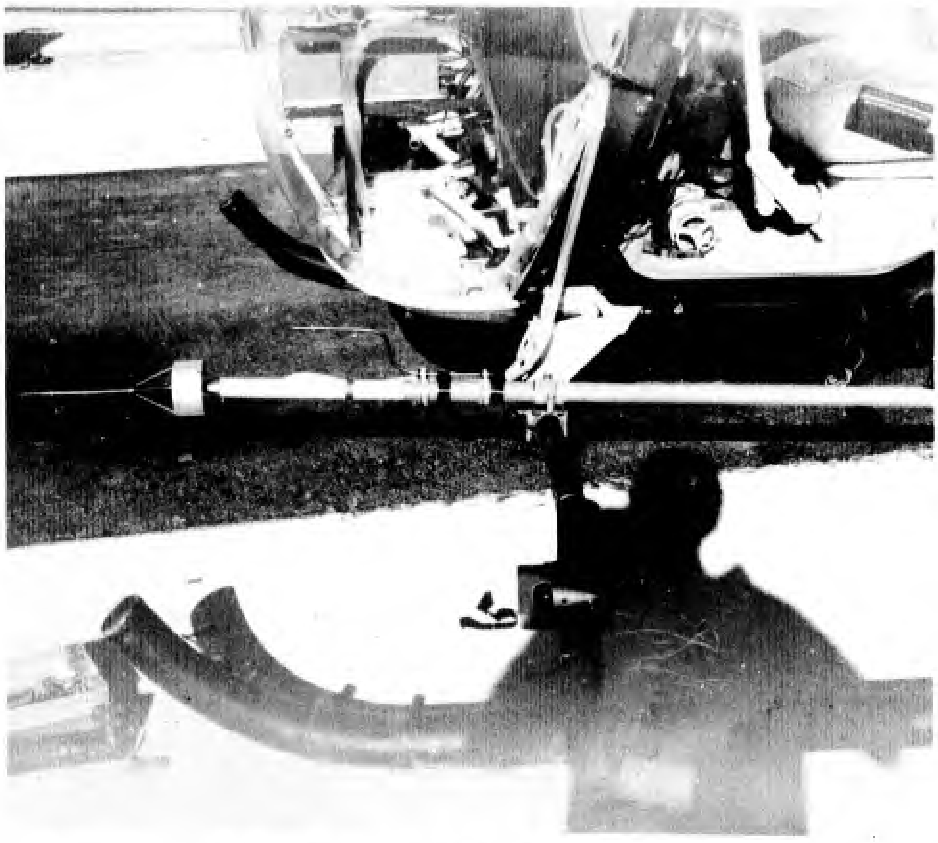


PHOTO III – TELESCOPING AIRSPEED BOOM IN ITS SHORTEST POSITION

The pressure field surrounding an aircraft is strongest nearer the aircraft, and as a result, a short boom would be expected to produce a higher position error than a long boom. Figure 12, Appendix II, shows the maximum values of these calibrations as plotted against boom length. The shorter boom, -7 in + 1 ft (see Figure 12, Appendix II), results in approximately twice as large an airspeed position error as the longest boom, -7 in + 8 ft.

The calibration on the shortest boom is not considered to be reliable because it was apparent while flying the calibration that the pitot-static head was located in an area of turbulence. The relative wind-seeking pitot-static head was in constant movement.

The results of this test revealed that extreme boom lengths are not necessary to obtain an accurate airspeed calibration. For flight test purposes, a long boom is desirable in order that the pilot may more conveniently fly a selected airspeed in which case a long boom is justified. In the general case, however, it is sufficient that the airspeed may be accurately calculated. This may be accomplished accurately with a moderate length boom and without resorting to the structural complexities required to support a long boom.

Since the primary effect of position error is in the static pressure system, the altimeter is similarly affected by the same position error as the airspeed indicator. The same arguments may be used concerning the altimeter position error as with the airspeed position error. In some cases, a long boom may be justified where very precise flying is required, but generally a short boom and an accurately determined position error correction factor will suffice.

E. Conclusions

1. Position error with a short boom is larger than position error with a long boom.

2. At boom lengths beyond two feet, there is negligible difference in the reliability of airspeed position error calibrations.

RECOMMENDATIONS

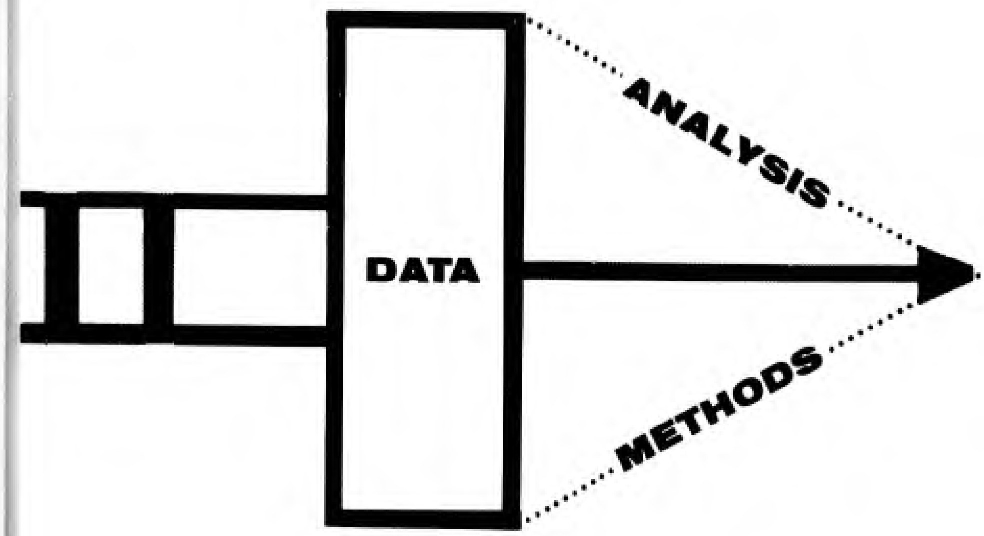
1. An airspeed boom on a light helicopter need not extend beyond two feet past the forward tip of the aircraft in order to obtain a reliable airspeed position error calibration.

2. A longer boom length may be necessary only when it is required that the indicated airspeed be slightly closer to the calibrated airspeed. This condition could possibly occur if a pilot were required to fly a series of accurate airspeeds with no time available for mental airspeed corrections between points.

3. Additional tests be performed to determine the optimum location for an airspeed boom on a light helicopter to obtain the most stable position error at low speed.

4. Airspeed systems be properly volume balanced for all test and standard aircraft installations.

5. Additional tests be conducted on a higher performance helicopter to verify that altitude does not effect airspeed system balance.



APPENDIX I

A. AIRSPEED ERROR WHILE CHANGING ALTITUDE

During these tests there were four quantities requiring analysis before presentation of the results: (1) rate of descent; (2) altitude; (3) airspeed error; and (4) airspeed.

1. Indicated rate of descent was found by measuring the time required to pass between two altitudes corrected for instrument error.

$$R/D = \frac{H_{Pic(1)} - H_{Pic(2)}}{T} \cdot \frac{T_t}{T_s}$$

where:

R/D = Rate of Descent (ft/min)

$H_{Pic(1)}$ = Pressure Altitude at start of timing corrected for instrument error (ft)

$H_{Pic(2)}$ = Pressure Altitude at end of timing corrected for instrument error (ft)

T = Time of Descent (min)

2. Altitude of the test was the mean between $H_{Pic(1)}$ and $H_{Pic(2)}$. (Feet)

3. Airspeed error was the difference between the indicated airspeed, instrument corrected, with the altimeter and rate of climb indicator connected to the static system, minus the indicated airspeed, instrument corrected, with only the airspeed indicator connected to the static system.

Airspeed Error = V_{ic} (w/Alt and R/C) minus V_{ic} (w/airspeed ind. only).

Airspeed Error - knots

V_{ic} w/Alt. and R/C - knots

V_{ic} w/airspeed ind. only - knots

B. EFFECT OF ALTITUDE UPON AIRSPEED SYSTEM BALANCE

No data reduction required.

C. EFFECT OF BOOM LENGTH UPON POSITION ERROR

The airspeed position error for each boom length was determined by the Ground Speed Course Method as described in AFFTC-TN-59-22, "Flight Test Handbook." Due to the number of repetitous calculations required, the data was reduced using an IBM 1620 Computer.

$$\Delta V_{pc} = V_{cal} - V_{ic}$$

$$\Delta V_{pc} = V_t \sqrt{\sigma} - (V_i + \Delta V_{ic})$$

where:

ΔV_{pc} = Airspeed Position Error - knots

$V_t = \left(\frac{Dist}{t_1} + \frac{Dist}{t_2}\right)(.004[illegible]3)$ - knots

Dist = length of ground course - feet

t_1 = time required to traverse course - min

t_2 = time required to traverse course on reciprocal heading - min

$$\sigma = \frac{(9.631)(29.92)(1.0-0.000006875\ H_{Pic})^{5.256}}{FAT_{ic} + 273}$$

H_{Pic} = instrument corrected pressure altitude - feet

FAT_{ic} = instrument corrected free air temperature - degrees centagrade

V_i = indicated airspeed - knots

ΔV_{ic} = airspeed indicator error - knots

V_{ic} = airspeed corrected for instrument error - knots

APPENDIX II

GRAPHIC TEST RESULTS

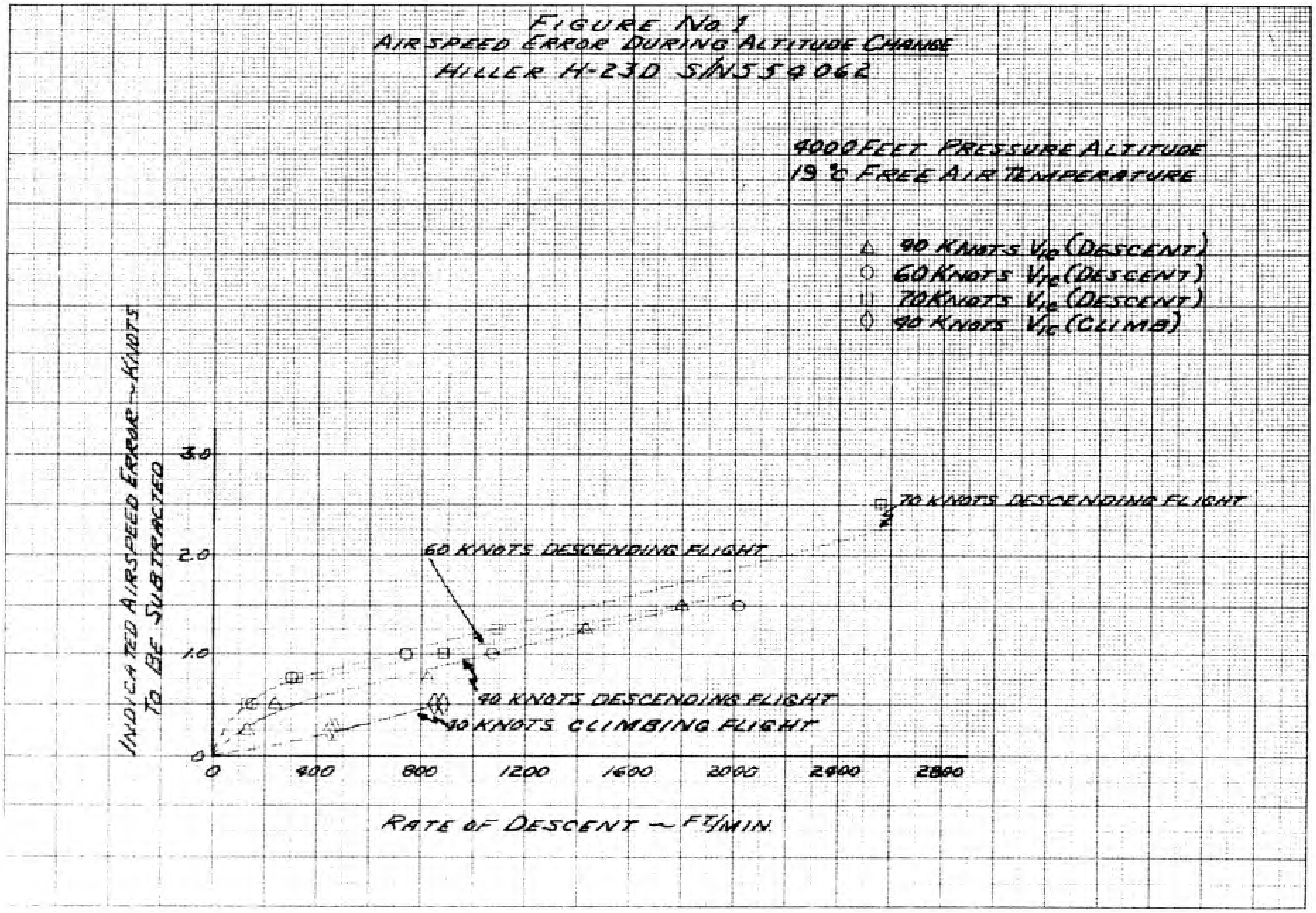


FIGURE No 1
AIRSPEED ERROR DURING ALTITUDE CHANGE
HILLER H-23D S/N 55 4062
4000 FEET PRESSURE ALTITUDE
19 °C FREE AIR TEMPERATURE
90 KNOTS V_{IC} (DESCENT)
60 KNOTS V_{IC} (DESCENT)
70 KNOTS V_{IC} (DESCENT)
90 KNOTS V_{IC} (CLIMB)
70 KNOTS DESCENDING FLIGHT
60 KNOTS DESCENDING FLIGHT
90 KNOTS DESCENDING FLIGHT
90 KNOTS CLIMBING FLIGHT
INDICATED AIRSPEED ERROR ~ KNOTS
TO BE SUBTRACTED
0
1.0
2.0
3.0
0
400
800
1200
1600
2000
2400
2800
RATE OF DESCENT ~ FT/MIN.

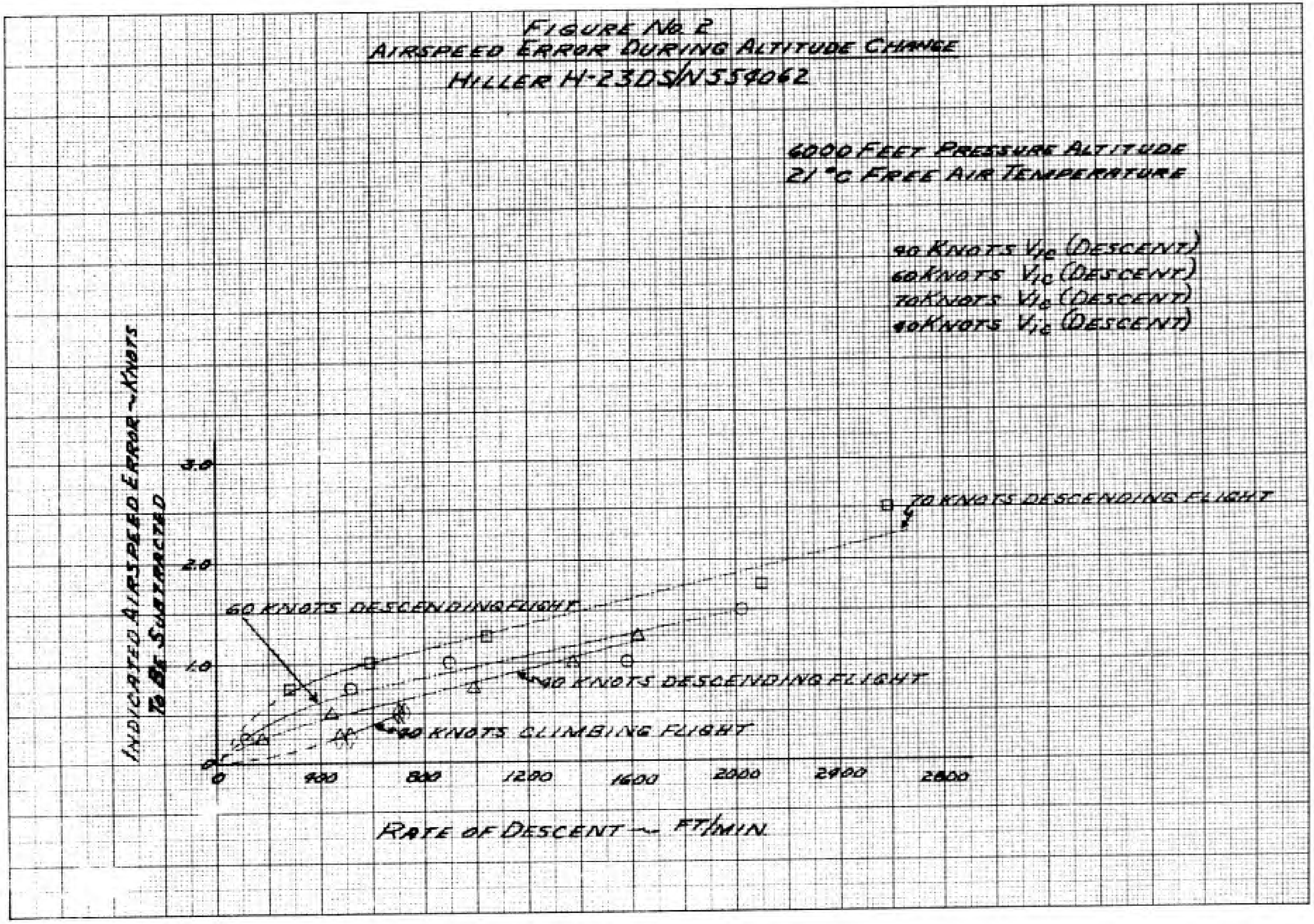


FIGURE No. 2
AIRSPEED ERROR DURING ALTITUDE CHANGE
HILLER H-23D S/N 554062
6000 FEET PRESSURE ALTITUDE
21 °C FREE AIR TEMPERATURE
90 KNOTS Vic (DESCENT)
60 KNOTS Vic (DESCENT)
70 KNOTS Vic (DESCENT)
40 KNOTS Vic (DESCENT)
70 KNOTS DESCENDING FLIGHT
60 KNOTS DESCENDING FLIGHT
90 KNOTS DESCENDING FLIGHT
90 KNOTS CLIMBING FLIGHT
INDICATED AIRSPEED ERROR ~ KNOTS
TO BE SUBTRACTED
3.0
2.0
1.0
0
400
800
1200
1600
2000
2400
2800
RATE OF DESCENT ~ FT/MIN

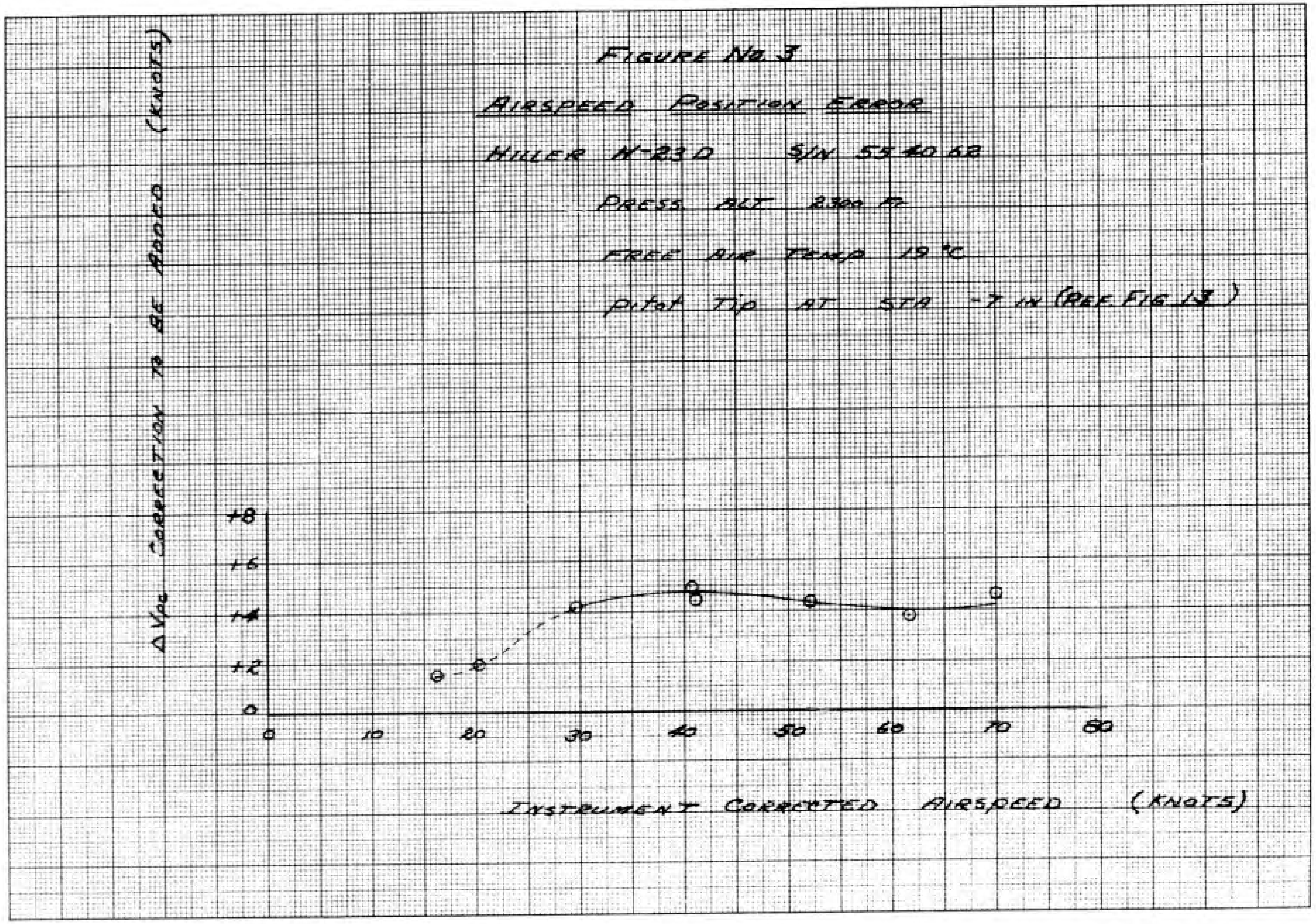


FIGURE NO 3
AIRSPEED POSITION ERROR
HILLER H-23D S/N 55 40 62
PRESS. ALT 2300 FT
FREE AIR TEMP 19 °C
pitot Tip AT STA -7 IN (REF FIG 13)
ΔVpc CORRECTION TO BE ADDED (KNOTS)
INSTRUMENT CORRECTED AIRSPEED (KNOTS)
0
+2
+4
+6
+8
0
10
20
30
40
50
60
70
80

FIGURE No 4

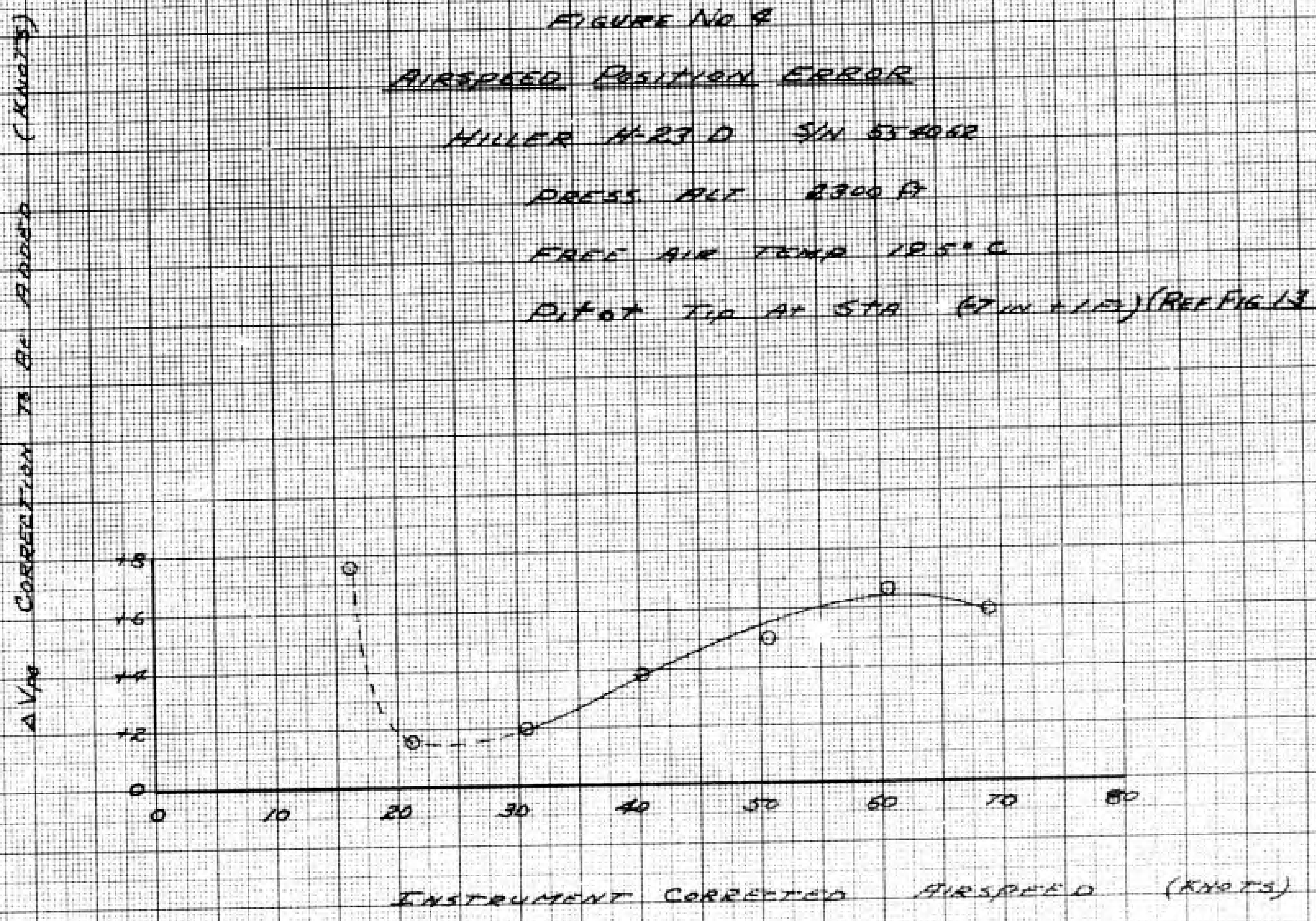
AIRSPEED POSITION ERROR

HILLER H-23 D S/N 53-4062

PRESS. ALT 2300 Ft

FREE AIR TEMP 12.5° C

PITOT TIP AT STA (67 IN + 1 Ft) (REF FIG 13)



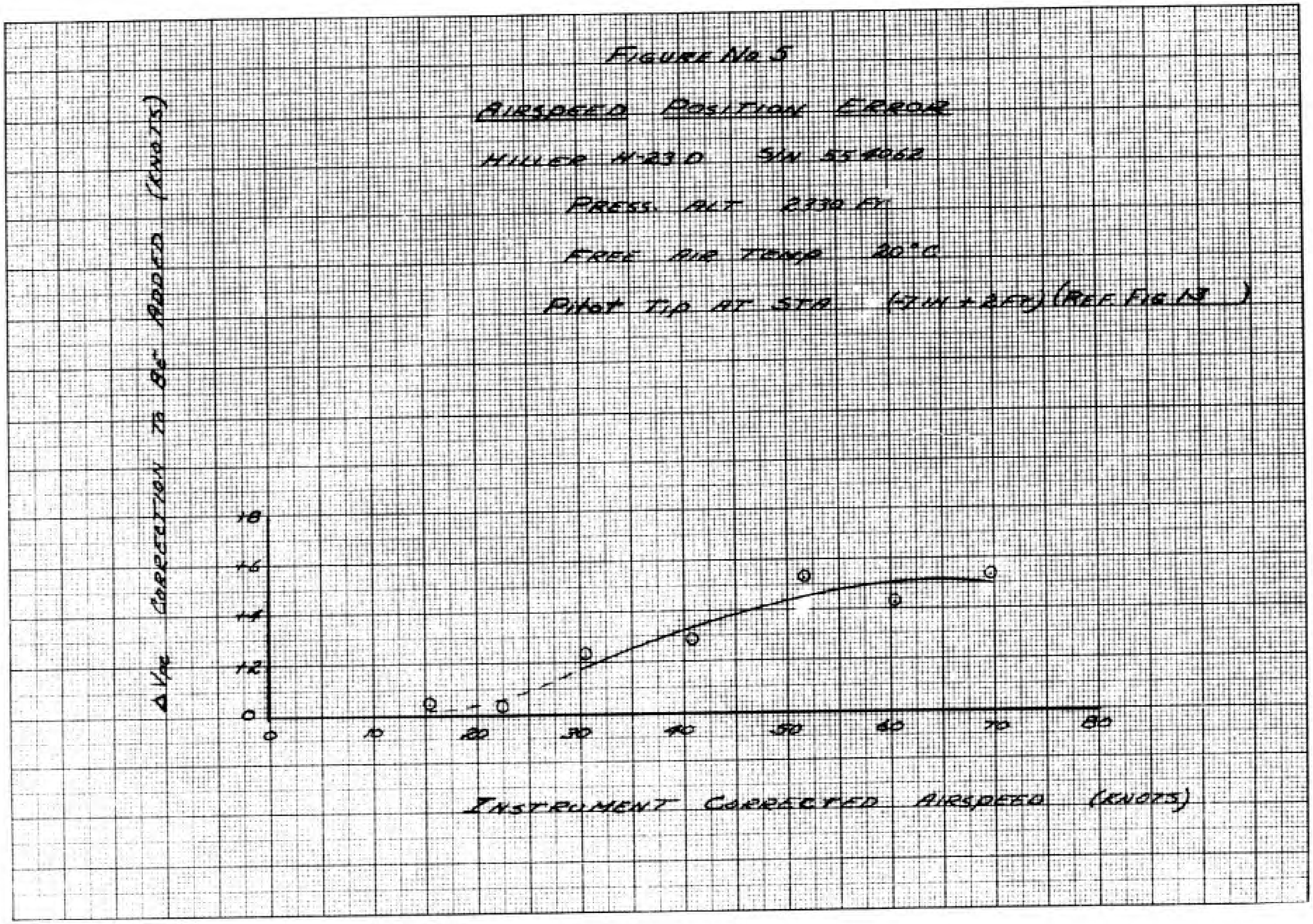


FIGURE No 5
AIRSPEED POSITION ERROR
HILLER H-23D S/N 55-4062
PRESS. ALT 2330 FT
FREE AIR TEMP 20°C
Pitot Tip AT STA (7 in + 2 FT) (REF FIG 13)
ΔVpe CORRECTION TO BE ADDED (KNOTS)
0
+2
+4
+6
+8
0
10
20
30
40
50
60
70
80
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

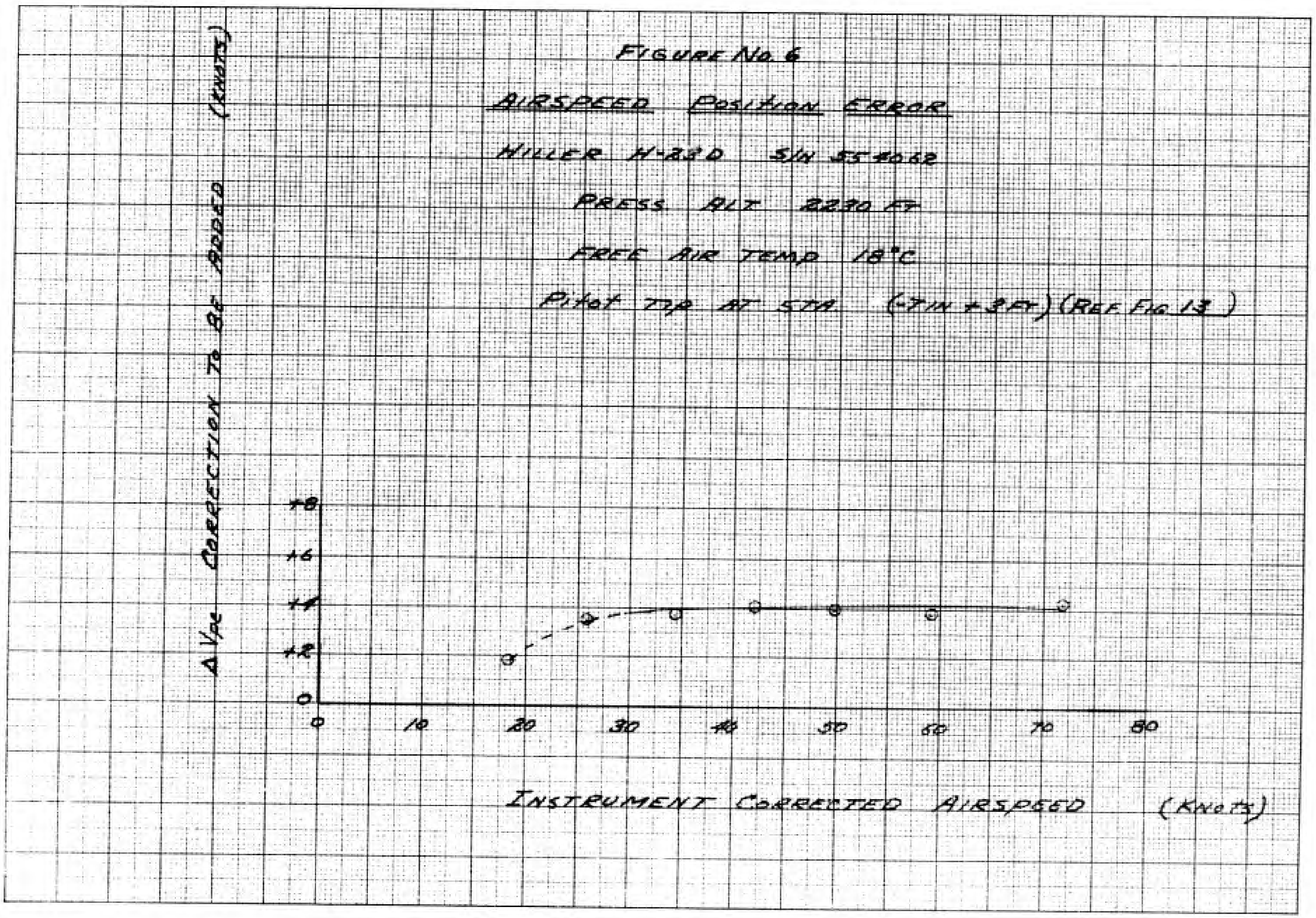


FIGURE No. 6
AIRSPEED POSITION ERROR
HILLER H-23D S/N 55-4062
PRESS ALT 2230 FT
FREE AIR TEMP 18°C
Pitot Tip AT STA (-7 IN + 3 FT) (REF. FIG. 13)
Δ Vpe CORRECTION TO BE ADDED (KNOTS)
+8
+6
+4
+2
0
0
10
20
30
40
50
60
70
80
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

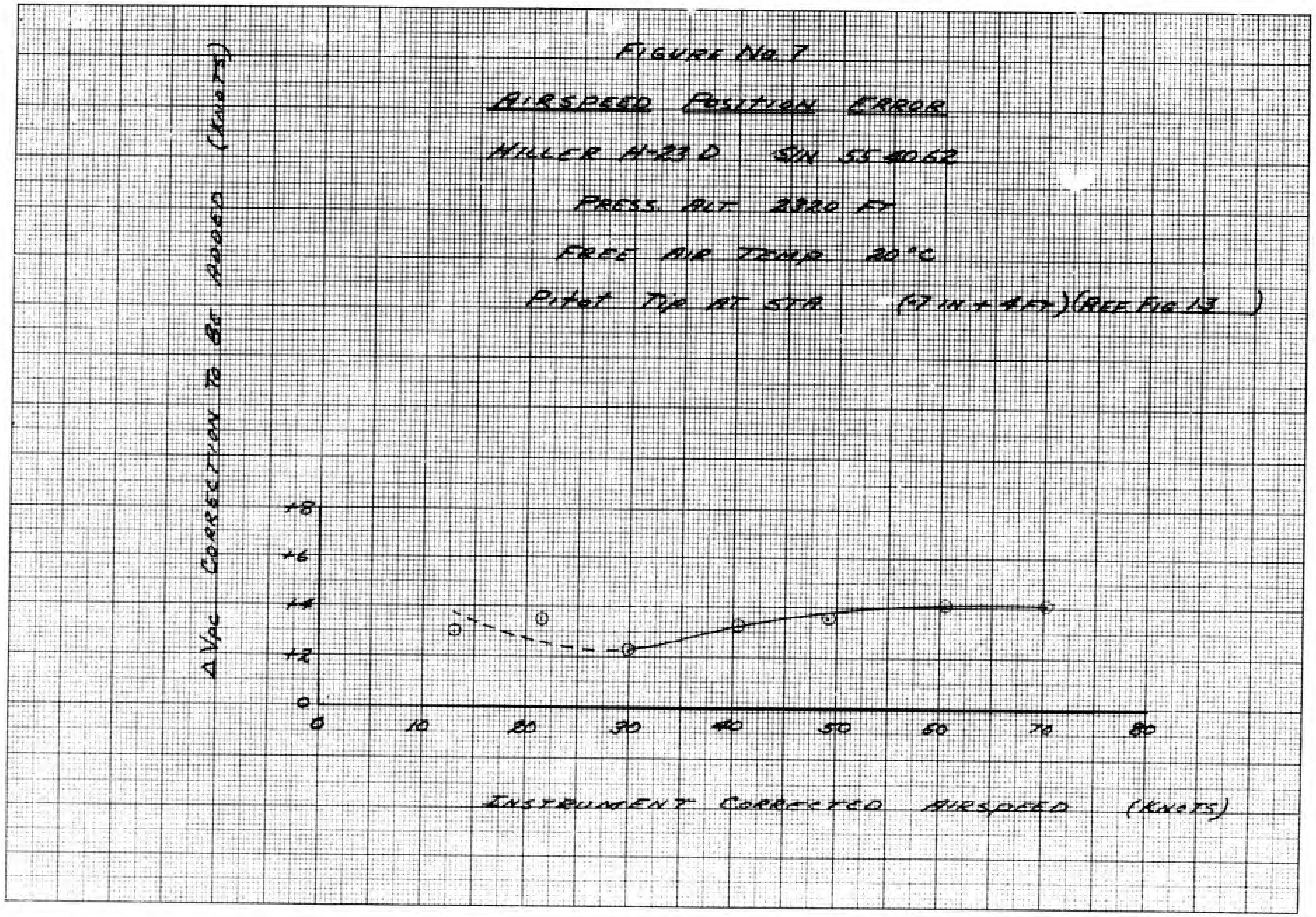


FIGURE No. 7
AIRSPEED POSITION ERROR
HILLER H-23 D S/N 55 4062
PRESS. ALT. 2320 FT
FREE AIR TEMP. 20°C
Pitot Tip AT STA. (-7 IN + 4 FT) (REF FIG 13)
ΔVpc CORRECTION TO BE ADDED (KNOTS)
+8
+6
+4
+2
0
0
10
20
30
40
50
60
70
80
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

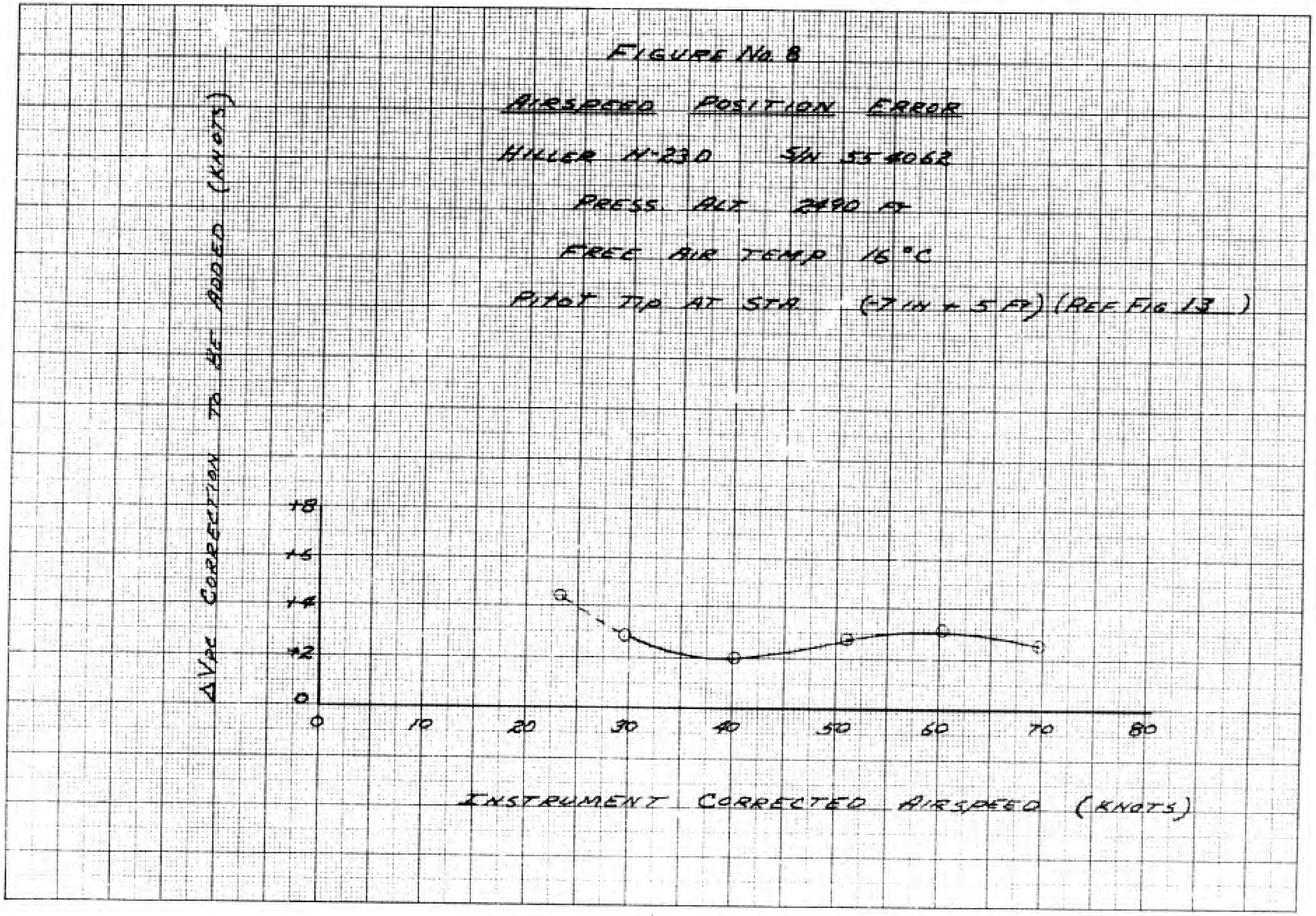


FIGURE No. 8
AIRSPEED POSITION ERROR
HILLER H-23D S/N 55-4062
PRESS. ALT 2490 FT
FREE AIR TEMP 16 °C
PITOT TIP AT STA (-7 IN + 5 FT) (REF FIG 13)
ΔVpc CORRECTION TO BE ADDED (KNOTS)
+8
+6
+4
+2
0
0
10
20
30
40
50
60
70
80
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

FIGURE No 9

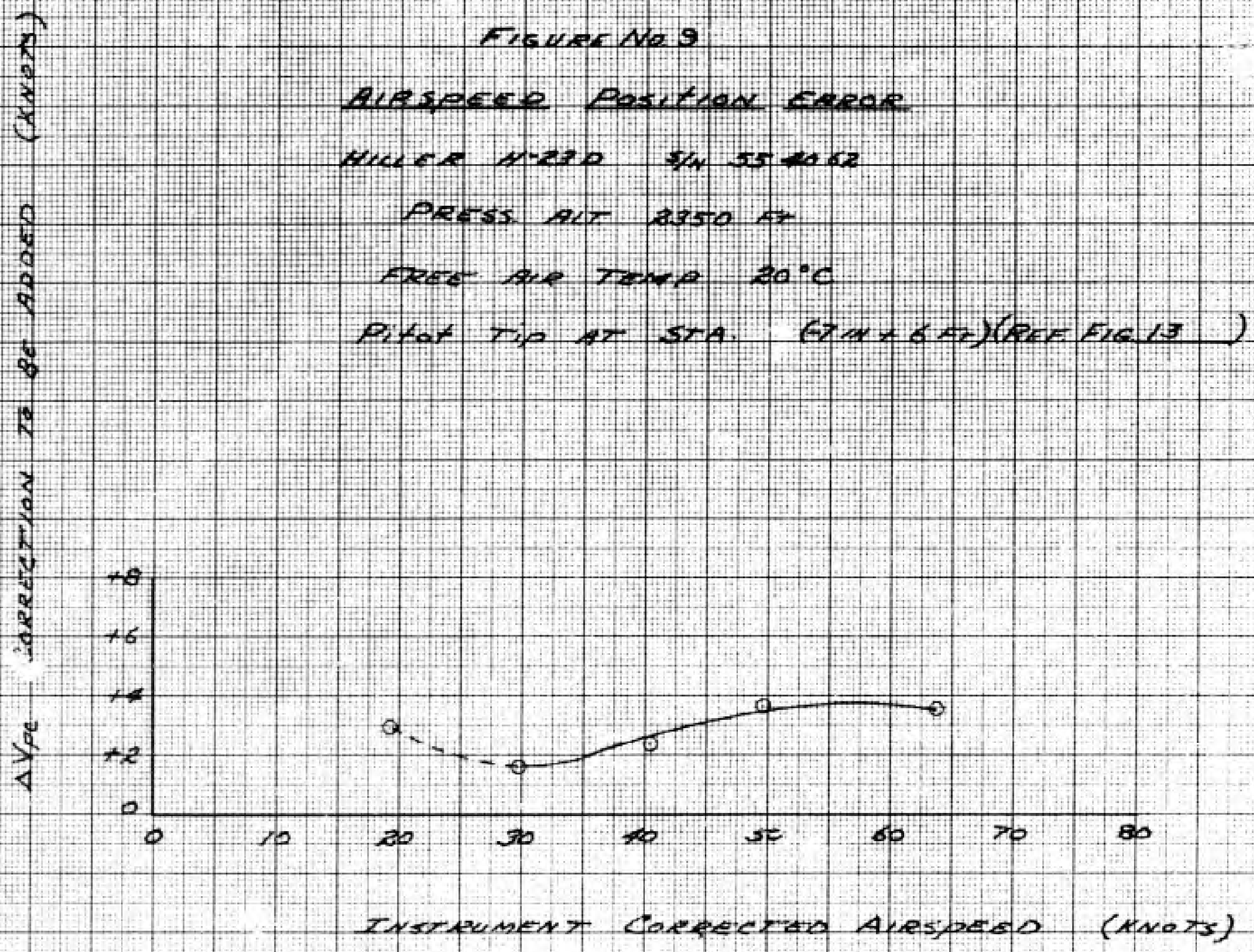
AIRSPEED POSITION ERROR

HILLER H-23D S/N 55-4062

PRESS ALT 2350 FT

FREE AIR TEMP 20°C

Pitot Tip AT STA. (67 IN + 6 FT)(REF FIG 13)



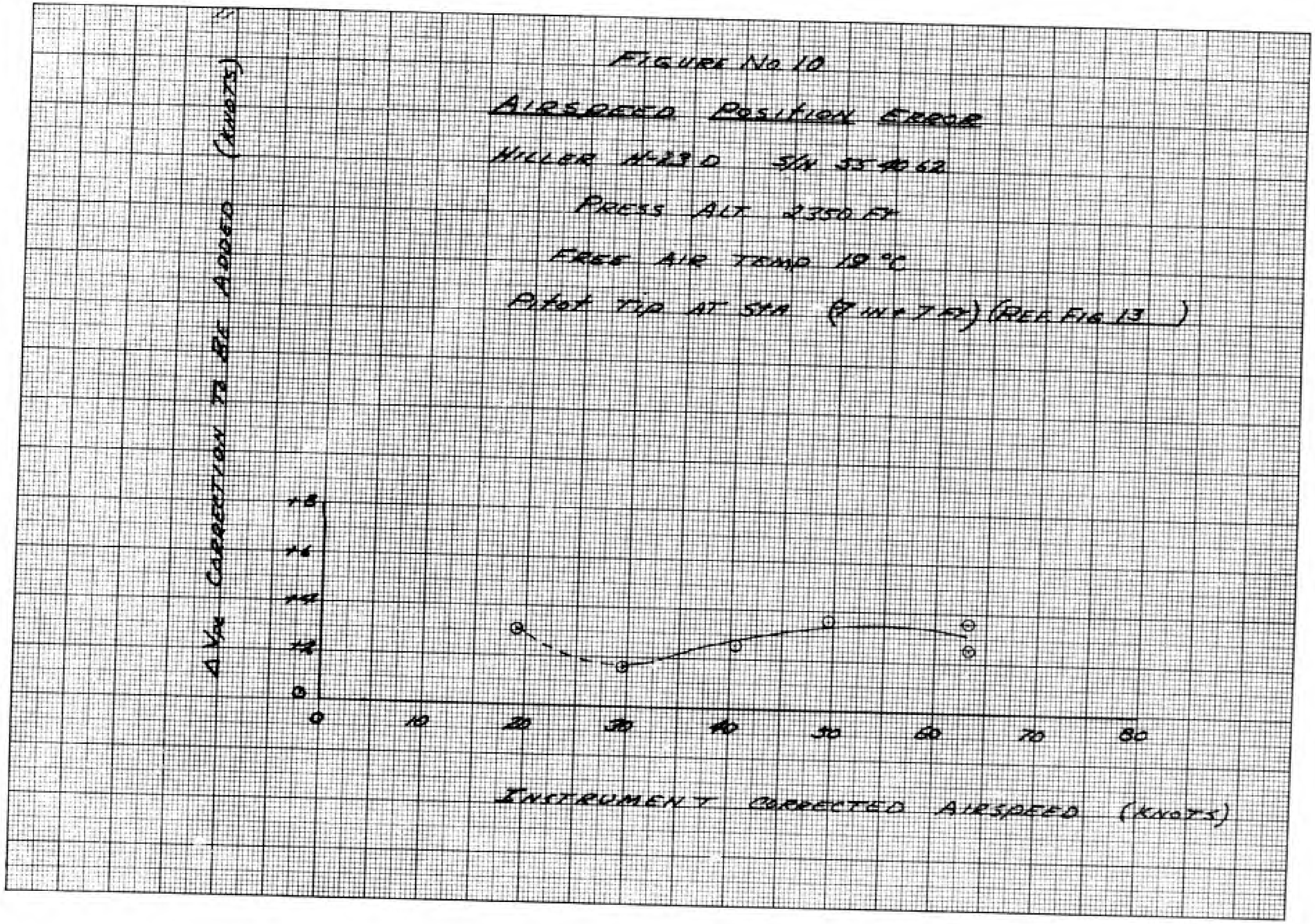


FIGURE No 10
AIRSPEED POSITION ERROR
HILLER H-23 D S/N 55 4062
PRESS ALT 2350 FT
FREE AIR TEMP 19 °C
PITOT TIP AT STA (7 IN + 7 FT) (REF FIG 13)
ΔVpe CORRECTION TO BE ADDED (KNOTS)
+8
+6
+4
+2
0
0
10
20
30
40
50
60
70
80
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

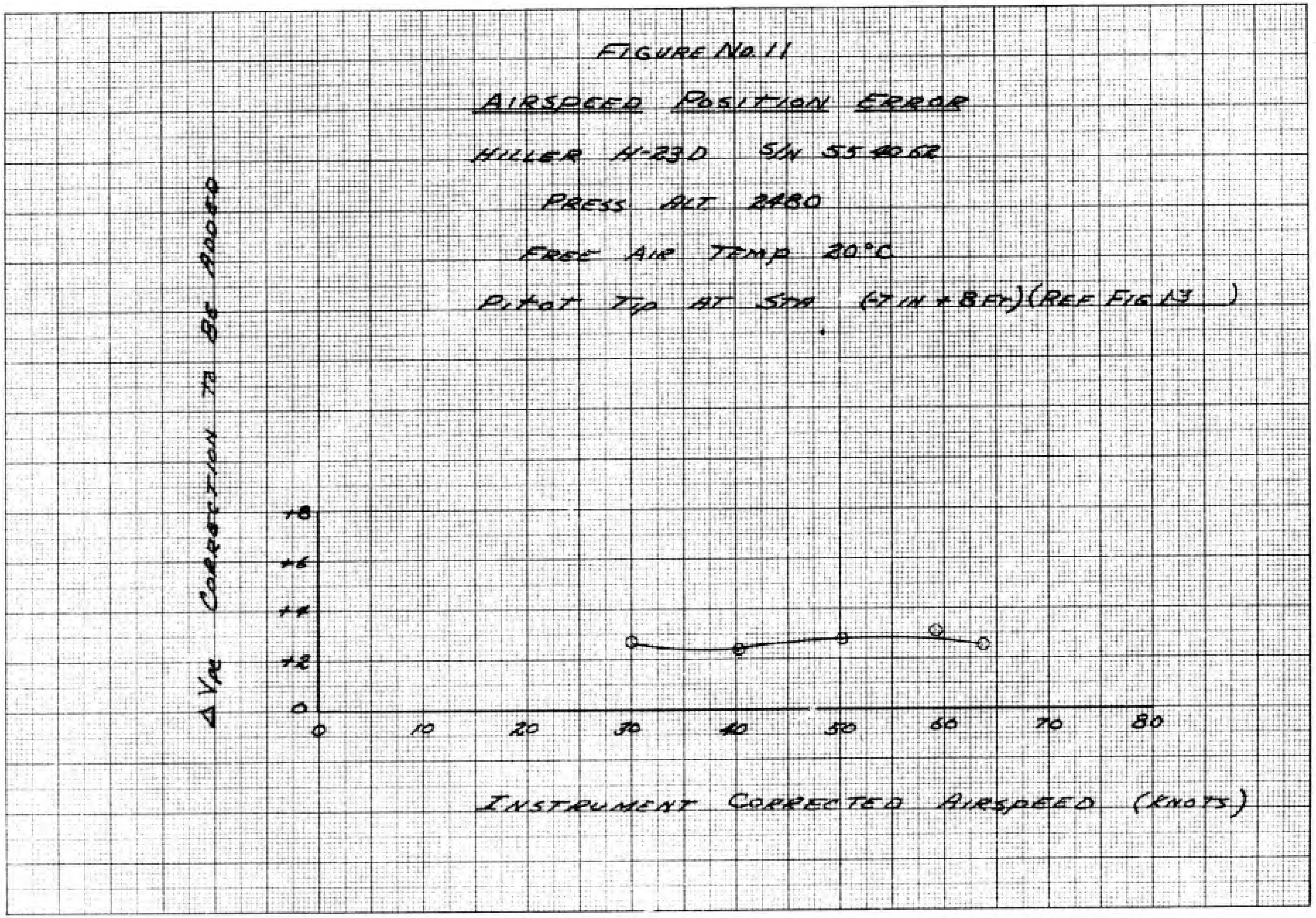


FIGURE No. 11
AIRSPEED POSITION ERROR
HILLER H-23D S/N 55 40 62
PRESS ALT 2480
FREE AIR TEMP 20°C
PITOT TIP AT STA (47 IN + 8 FT)(REF FIG 13)
ΔVpe CORRECTION TO BE ADDED
+8
+6
+4
+2
0
0
10
20
30
40
50
60
70
80
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

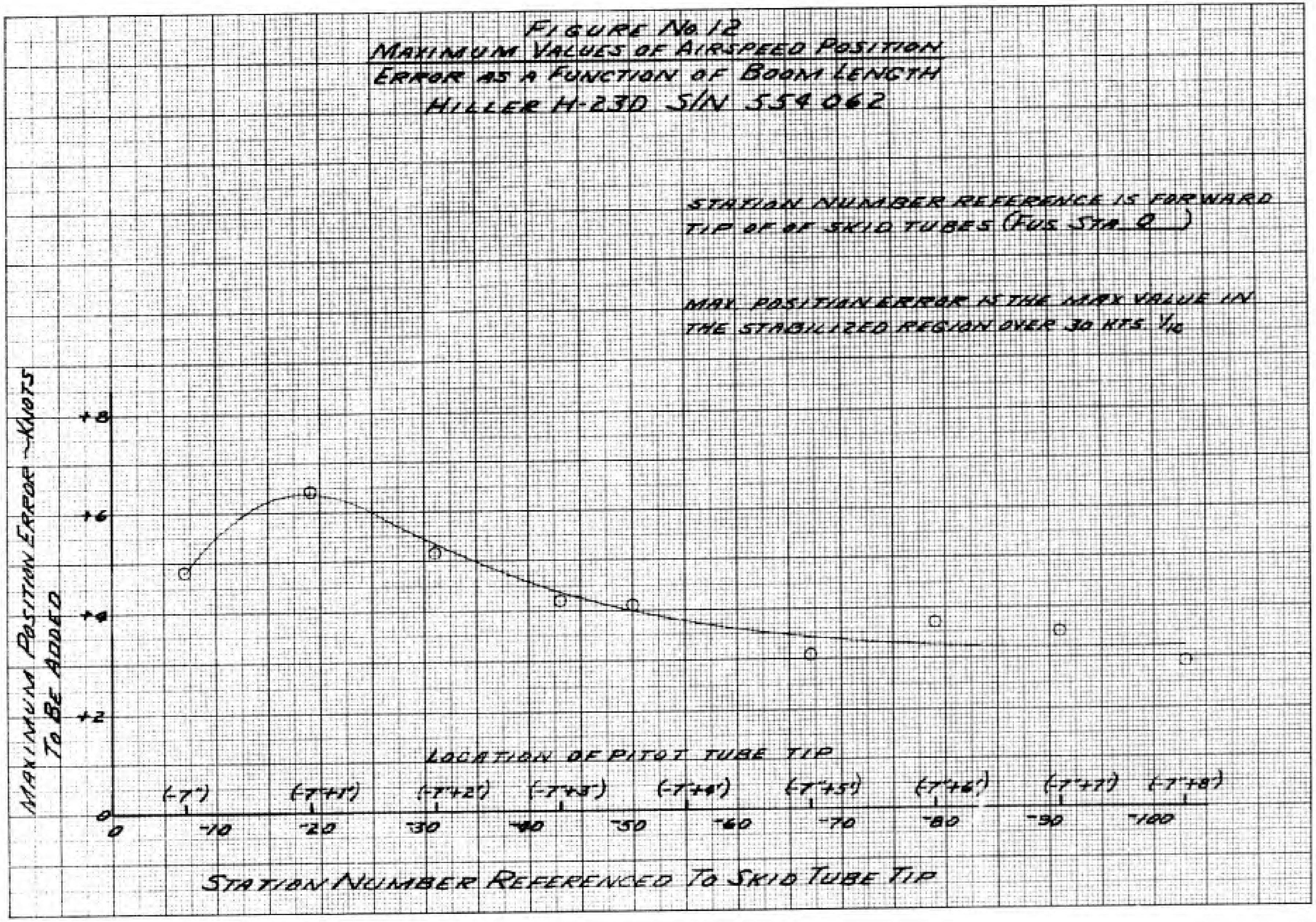


FIGURE No. 12
MAXIMUM VALUES OF AIRSPEED POSITION ERROR AS A FUNCTION OF BOOM LENGTH
HILLER H-23D S/N 554062

APPENDIX III

Instrumentation

The following calibrated instruments were installed in conjunction with an airspeed boom as shown in Figure A.

Sensitive Airspeed Indicator

Altimeter

Sensitive Rate of Climb Indicator

Pitot-Static Head

Variable Volume "Balance Can"

Necessary Plumbing and Pressure Switches

The plumbing arrangement for the test airspeed system followed conventional aircraft practice except for the following changes.

a. The variable volume "balance can" was plumbed so that it could be switched in and out of the total pressure side of the airspeed system. When not in use; i.e., switched out of the system, it was connected to an alternate pitot probe in order to keep the total pressure in the can reasonably well matched with the total pressure in the test airspeed system so that minimum time would be required for the readings to stabilize when the can was switched into the system.

b. A rate of climb indicator and an altimeter were plumbed through a pressure switching valve into the static side of the test airspeed system in a manner that allowed their internal volume to be connected or disconnected. When not connected to the test airspeed system, the altimeter and rate of climb indicator were vented into the cockpit.

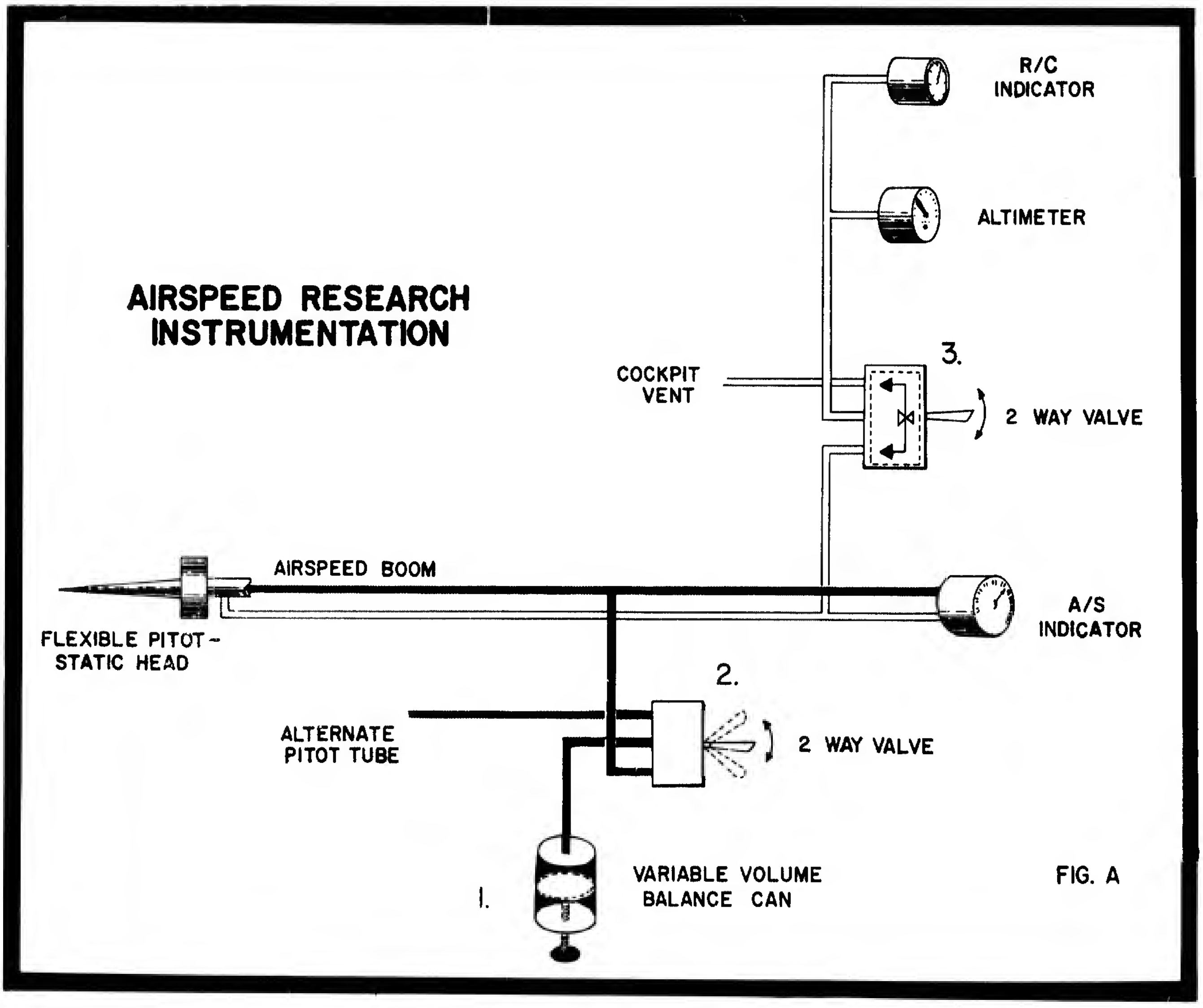
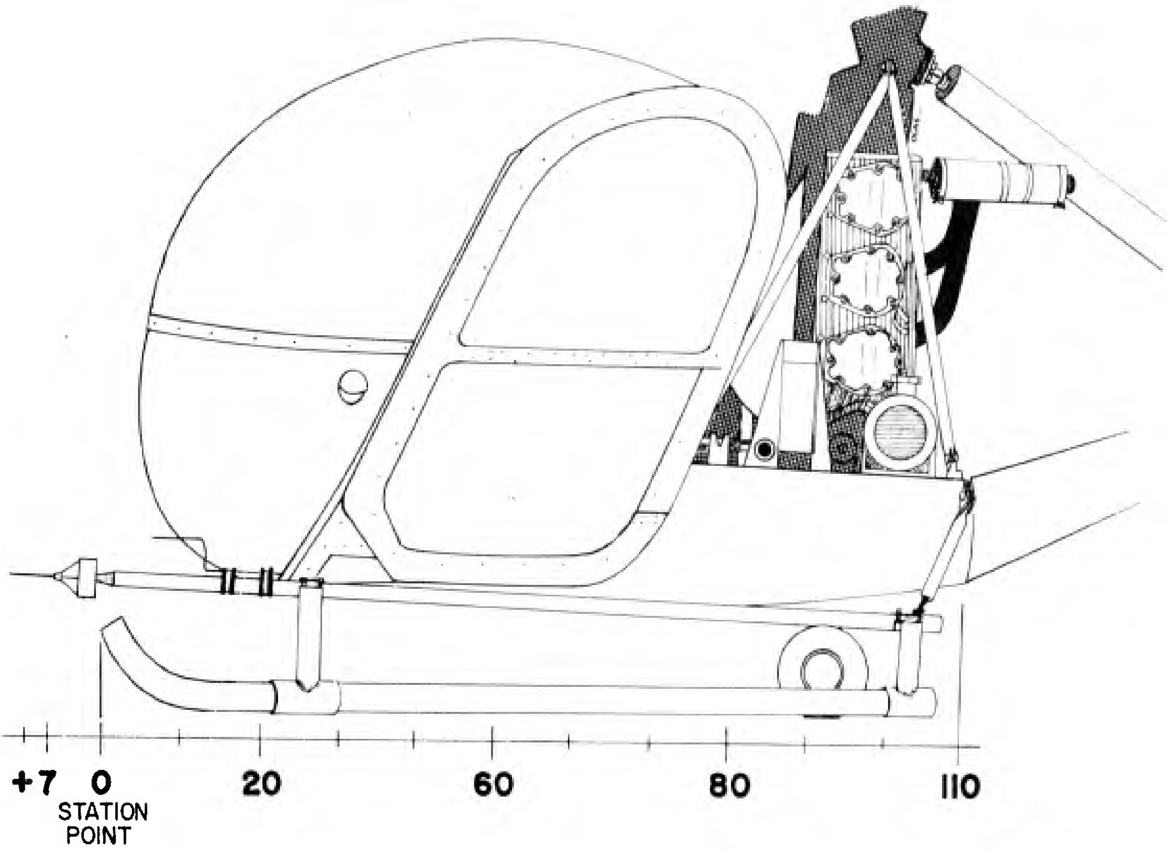
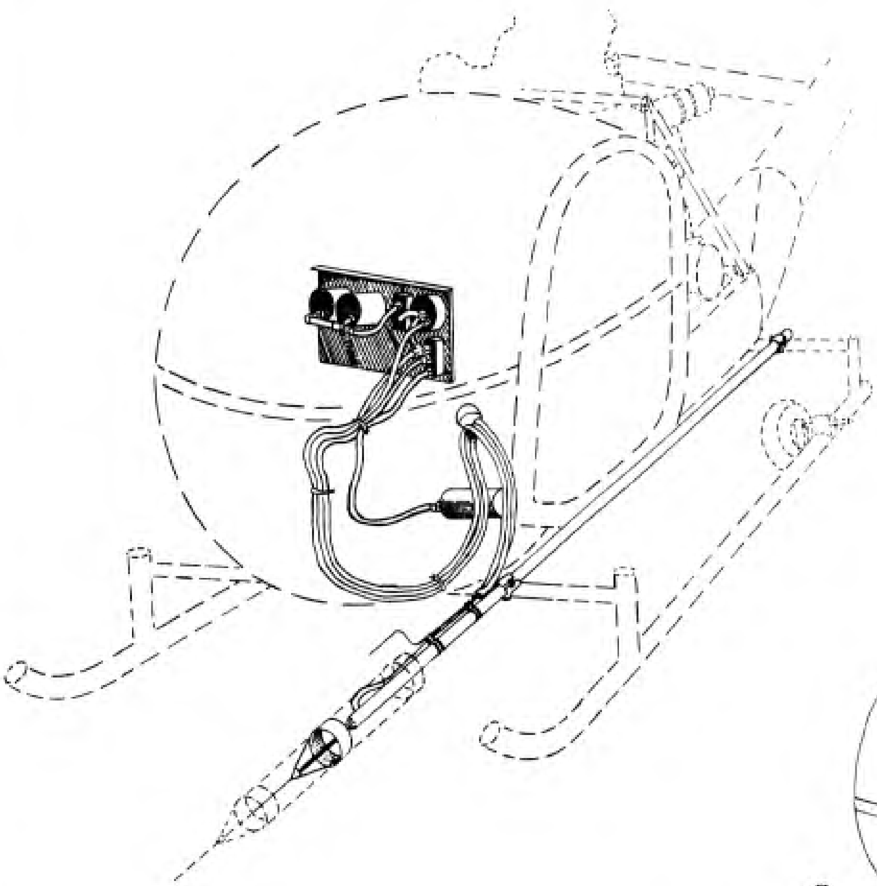
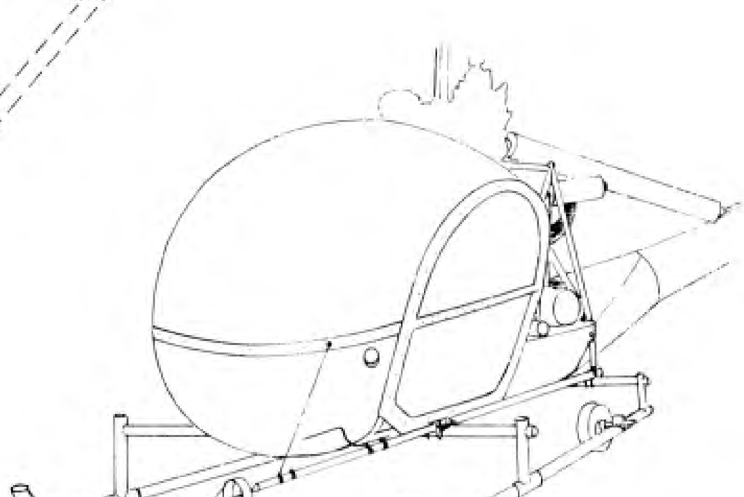


FIG. A







OH-23D RAVEN

FIG. 13

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13. ABSTRACT

This report is a technical note which presents the results of a test or investigation that was limited in scope. Its purpose is to increase the fund of information available on a particular subject and is not intended to be considered as a complete or all-inclusive work.

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