Naval Test Pilot School Flight Test Manual Chapter 2 (Pitot Static System Performance) for Fixed Wing Performance

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CHAPTER 2

PITOT STATIC SYSTEM PERFORMANCE

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EQUATIONS

$$V_e = \sqrt{\frac{2q}{\rho_{ssl}}} = \sqrt{\frac{\sigma\, 2q}{\rho_a}} = \sqrt{\sigma}\, V_T$$ (Eq 2.13) 2.11

$$V_{e_{Test}} = V_{e_{Std}}$$ (Eq 2.14) 2.12

$$V_T^2 = \frac{2\gamma}{\gamma - 1} \frac{P_a}{\rho_a} \left[\left(\frac{P_T - P_a}{P_a} + 1 \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right]$$ (Eq 2.15) 2.13

$$V_T = \sqrt{\frac{2\gamma}{\gamma - 1} \frac{P_a}{\rho_a} \left[\left(\frac{q_c}{P_a} + 1 \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right]}$$ (Eq 2.16) 2.13

$$q_c = q \left(1 + \frac{M^2}{4} + \frac{M^4}{40} + \frac{M^6}{1600} + \ldots \right)$$ (Eq 2.17) 2.13

$$V_c^2 = \frac{2\gamma}{\gamma - 1} \frac{P_{ssl}}{\rho_{ssl}} \left[\left(\frac{P_T - P_a}{P_{ssl}} + 1 \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right]$$ (Eq 2.18) 2.14

$$V_c = \sqrt{\frac{2\gamma}{\gamma - 1} \frac{P_{ssl}}{\rho_{ssl}} \left[\left(\frac{q_c}{P_{ssl}} + 1 \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right]}$$ (Eq 2.19) 2.14

$$V_c = f\left(P_T - P_a\right) = f\left(q_c\right)$$ (Eq 2.20) 2.14

$$V_{c_{Test}} = V_{c_{Std}}$$ (Eq 2.21) 2.14

$$\frac{P'_T}{P_a} = \left[\frac{\gamma + 1}{2} \left(\frac{V}{a} \right)^2 \right]^{\frac{\gamma}{\gamma - 1}} \left[\frac{1}{\frac{2\gamma}{\gamma + 1} \left(\frac{V}{a} \right)^2 - \frac{\gamma - 1}{\gamma + 1}} \right]^{\frac{1}{\gamma - 1}}$$ (Eq 2.22) 2.15

$$\frac{q_c}{P_{ssl}} = \left[1 + 0.2 \left(\frac{V_c}{a_{ssl}} \right)^2 \right]^{3.5} - 1$$

(For $V_c \le a_{ssl}$) (Eq 2.23) 2.15

$$\frac{q_c}{P_{ssl}} = \left[\frac{166.921 \left(\frac{V_c}{a_{ssl}} \right)^7}{\left[7 \left(\frac{V_c}{a_{ssl}} \right)^2 - 1 \right]^{2.5}} \right] - 1$$

(For $V_c \ge a_{ssl}$) (Eq 2.24) 2.15

$$V_e = \sqrt{\frac{2\gamma}{\gamma - 1} \frac{P_a}{\rho_{ssl}} \left[\left(\frac{q_c}{P_a} + 1 \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right]}$$

(Eq 2.25) 2.17

$$V_e = V_T \sqrt{\sigma}$$

(Eq 2.26) 2.17

$$M = \frac{V_T}{a} = \frac{V_T}{\sqrt{\gamma g_c R T}} = \frac{V_T}{\sqrt{\gamma \frac{P}{\rho}}}$$

(Eq 2.27) 2.17

$$M = \sqrt{\frac{2}{\gamma - 1} \left[\left(\frac{P_T - P_a}{P_a} + 1 \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right]}$$

(Eq 2.28) 2.17

$$\frac{P_T}{P_a} = \left(1 + \frac{\gamma - 1}{2} M^2 \right)^{\frac{\gamma}{\gamma - 1}}$$

(Eq 2.29) 2.18

$$\frac{q_c}{P_a} = \left(1 + 0.2 M^2 \right)^{3.5} - 1$$

for $M < 1$ (Eq 2.30) 2.18

$$\frac{q_c}{P_a} = \left[\frac{166.921 M^7}{\left(7M^2 - 1 \right)^{2.5}} \right] - 1$$

for $M > 1$ (Eq 2.31) 2.18

$M = f\left(P_T - P_a, P_a\right) = f\left(V_c, H_P\right)$	(Eq 2.32)	2.19
$M_{Test} = M$	(Eq 2.33)	2.19
$\Delta H_{P_{ic}} = H_{P_i} - H_{P_o}$	(Eq 2.34)	2.22
$\Delta V_{ic} = V_i - V_o$	(Eq 2.35)	2.22
$H_{P_i} = H_{P_o} + \Delta H_{P_{ic}}$	(Eq 2.36)	2.22
$V_i = V_o + \Delta V_{ic}$	(Eq 2.37)	2.22
$\Delta P = P_s - P_a$	(Eq 2.38)	2.27
$\Delta V_{pos} = V_c - V_i$	(Eq 2.39)	2.27
$\Delta H_{pos} = H_{P_c} - H_{P_i}$	(Eq 2.40)	2.27
$\Delta M_{pos} = M - M_i$	(Eq 2.41)	2.27
$\frac{P_s}{P_a} = f_1\left(M, \alpha, \beta, R_e\right)$	(Eq 2.42)	2.28
$\frac{P_s}{P_a} = f_2\left(M, \alpha\right)$	(Eq 2.43)	2.28
$\frac{\Delta P}{q_c} = f_3\left(M, \alpha\right)$	(Eq 2.44)	2.28
$\frac{\Delta P}{q_c} = f_4\left(M\right)$ (High speed)	(Eq 2.45)	2.28
$\frac{\Delta P}{q_c} = f_5\left(C_L\right)$ (Low speed)	(Eq 2.46)	2.28
$\frac{\Delta P}{q_{c_i}} = f_6\left(M_i\right)$ (High speed)	(Eq 2.47)	2.29

$\frac{\Delta P}{q_c} = f_7 \left(W, V_c \right)$ (Low speed)	(Eq 2.48)	2.29
$V_{c_W} = V_{c_{Test}} \sqrt{\frac{W_{Std}}{W_{Test}}}$	(Eq 2.49)	2.30
$\frac{\Delta P}{q_c} = f_8 \left(V_{c_W} \right)$ (Low speed)	(Eq 2.50)	2.30
$V_{i_W} = V_{i_{Test}} \sqrt{\frac{W_{Std}}{W_{Test}}}$	(Eq 2.51)	2.30
$\frac{\Delta P}{q_{c_i}} = f_9 \left(V_{i_W} \right)$ (Low speed)	(Eq 2.52)	2.30
$\frac{T_T}{T} = 1 + \frac{\gamma - 1}{2} M^2$	(Eq 2.53)	2.32
$\frac{T_T}{T} = 1 + \frac{\gamma - 1}{2} \frac{V_T^2}{\gamma g_c R T}$	(Eq 2.54)	2.32
$\frac{T_T}{T} = 1 + \frac{K_T \left(\gamma - 1 \right)}{2} M^2$	(Eq 2.55)	2.33
$\frac{T_T}{T} = 1 + \frac{K_T \left(\gamma - 1 \right)}{2} \frac{V_T^2}{\gamma g_c R T}$	(Eq 2.56)	2.33
$\frac{T_T}{T_a} = \frac{T_i}{T_a} = 1 + \frac{K_T M^2}{5}$	(Eq 2.57)	2.33
$T_T = T_i = T_a + \frac{K_T V_T^2}{7592}$	(Eq 2.58)	2.33
$T_i = T_o + \Delta T_{ic}$	(Eq 2.59)	2.35

$$K_T = \left(\frac{T_i\ (^\circ K)}{T_a\ (^\circ K)} - 1 \right) \frac{5}{M^2}$$ (Eq 2.60) 2.35

$$V_{G_1} = 3600 \left(\frac{D}{\Delta t_1} \right)$$ (Eq 2.61) 2.50

$$V_{G_2} = 3600 \left(\frac{D}{\Delta t_2} \right)$$ (Eq 2.62) 2.50

$$V_T = \frac{V_{G_1} + V_{G_2}}{2}$$ (Eq 2.63) 2.50

$$\rho_a = \frac{P_a}{g_c\ R\ T_{a_{ref}}\ (^\circ K)}$$ (Eq 2.64) 2.50

$$\sigma = \frac{\rho_a}{\rho_{ssl}}$$ (Eq 2.65) 2.51

$$V_c = V_e - \Delta V_c$$ (Eq 2.66) 2.51

$$M = \frac{V_T}{38.9678 \sqrt{T_{a_{ref}}\ (^\circ K)}}$$ (Eq 2.67) 2.51

$$q_c = P_{ssl} \left\{ \left[1 + 0.2 \left(\frac{V_c}{a_{ssl}} \right)^2 \right]^{3.5} - 1 \right\}$$ (Eq 2.68) 2.51

$$q_{c_i} = P_{ssl} \left\{ \left[1 + 0.2 \left(\frac{V_i}{a_{ssl}} \right)^2 \right]^{3.5} - 1 \right\}$$ (Eq 2.69) 2.51

$$\Delta P = q_c - q_{c_i}$$ (Eq 2.70) 2.51

$$V_{i_W} = V_i \sqrt{\frac{W_{Std}}{W_{Test}}}$$ (Eq 2.71) 2.51

$$H_{P_{i_{ref}}} = H_{P_{o_{ref}}} + \Delta H_{P_{ic_{ref}}}$$ (Eq 2.72) 2.54

$$H_{P_i} = \frac{T_{ssl}}{a_{ssl}} \left[1 - \left(\frac{P_s}{P_{ssl}} \right)^{\frac{1}{\left(\frac{g_{ssl}}{g_c\ a_{ssl}\ R} \right)}} \right]$$ (Eq 2.73) 2.54

$$H_{P_{i_{ref}}} = \frac{T_{ssl}}{a_{ssl}} \left[1 - \left(\frac{P_a}{P_{ssl}} \right)^{\frac{1}{\left(\frac{g_{ssl}}{g_c\ a_{ssl}\ R} \right)}} \right]$$ (Eq 2.74) 2.55

$$\Delta h = d \tan\theta$$ (Eq 2.75) 2.57

$$\Delta h = L_{a/c} \frac{y}{x}$$ (Eq 2.76) 2.57

$$H_{P_c} = H_{P_{c_{twr}}} + \Delta h \frac{T_{Std}\ (^\circ K)}{T_{Test}\ (^\circ K)}$$ (Eq 2.77) 2.57

$$P_s = P_{ssl} \left(1 - 6.8755856 \times 10^{-6}\ H_{P_i} \right)^{5.255863}$$ (Eq 2.78) 2.57

$$P_a = P_{ssl} \left(1 - 6.8755856 \times 10^{-6}\ H_{P_c} \right)^{5.255863}$$ (Eq 2.79) 2.58

$$\text{Curve slope} = K_T \frac{\gamma - 1}{\gamma} T_a = 0.2\ K_T\ T_a\ (^\circ K)\ \text{(High speed)}$$ (Eq 2.80) 2.60

$$\text{Curve slope} = K_T \frac{0.2\ T_a\ (^\circ K)}{a_{ssl}^2}\ \text{(Low speed)}$$ (Eq 2.81) 2.60

$$K_T = \frac{\text{slope}}{0.2\ T_a\ (^\circ K)} \text{ (High speed)}$$ (Eq 2.82) 2.60

$$K_T = \frac{\text{slope}\ a^2_{ssl}}{0.2\ T_a\ (^\circ K)} \text{ (Low speed)}$$ (Eq 2.83) 2.61

$$M_i = \sqrt{\frac{2}{\gamma - 1}\left[\left(\frac{q_{c_i}}{P_s} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]}$$ (Eq 2.84) 2.62

$$\Delta P = \left(\frac{\Delta P}{q_{c_i}}\right) q_{c_i}$$ (Eq 2.85) 2.62

$$q_c = q_{c_i} + \Delta P$$ (Eq 2.86) 2.62

$$\Delta V_{pos} = V_c - V_{i_W}$$ (Eq 2.87) 2.63

$$P_a = P_s - \Delta P$$ (Eq 2.88) 2.63

$$H_{P_c} = \frac{T_{ssl}}{a_{ssl}}\left[1 - \left(\frac{P_a}{P_{ssl}}\right)^{\frac{1}{\left(\frac{g_{ssl}}{g_c\ a_{ssl}\ R}\right)}}\right]$$ (Eq 2.89) 2.63

CHAPTER 2

PITOT STATIC SYSTEM PERFORMANCE

2.1 INTRODUCTION

The initial step in any flight test is to measure the pressure and temperature of the atmosphere and the velocity of the vehicle at the particular time of the test. There are restrictions in what can be measured accurately, and there are inaccuracies within each measuring system. This phase of flight testing is very important. Performance data and most stability and control data are worthless if pitot static and temperature errors are not corrected. Consequently, calibration tests of the pitot static and temperature systems comprise the first flights in any test program.

This chapter presents a discussion of pitot static system performance testing. The theoretical aspects of these flight tests are included. Test methods and techniques applicable to aircraft pitot static testing are discussed in some detail. Data reduction techniques and some important factors in the analysis of the data are also included. Mission suitability factors are discussed. The chapter concludes with a glossary of terms used in these tests and the references which were used in constructing this chapter.

2.2 PURPOSE OF TEST

The purpose of pitot static system testing is to investigate the characteristics of the aircraft pressure sensing systems to achieve the following objectives:

1. Determine the airspeed and altimeter correction data required for flight test data reduction.
2. Determine the temperature recovery factor, K_T.
3. Evaluate mission suitability problem areas.
4. Evaluate the requirements of pertinent Military Specifications.

2.3 THEORY

2.3.1 THE ATMOSPHERE

The forces acting on an aircraft in flight are a function of the temperature, density, pressure, and viscosity of the fluid in which the vehicle is operating. Because of this, the flight test team needs a means for determining the atmospheric properties. Measurements reveal the atmospheric properties have a daily, seasonal, and geographic dependence; and are in a constant state of change. Solar radiation, water vapor, winds, clouds, turbulence, and human activity cause local variations in the atmosphere. The flight test team cannot control these natural variances, so a standard atmosphere was constructed to describe the static variation of the atmospheric properties. With this standard atmosphere, calculations are made of the standard properties. When variations from this standard occur, the variations are used as a method for calculating or predicting aircraft performance.

2.3.2 DIVISIONS OF THE ATMOSPHERE

The atmosphere is divided into four major divisions which are associated with physical characteristics. The division closest to the earth's surface is the troposphere. Its upper limit varies from approximately 28,000 feet and -46°C at the poles to 56,000 feet and -79°C at the equator. These temperatures vary daily and seasonally. In the troposphere, the temperature decreases with height. A large portion of the sun's radiation is transmitted to and absorbed by the earth's surface. The portion of the atmosphere next to the earth is heated from below by radiation from the earth's surface. This radiation in turn heats the rest of the troposphere. Practically all weather phenomenon are contained in this division.

The second major division of the atmosphere is the stratosphere. This layer extends from the troposphere outward to a distance of approximately 50 miles. The original definition of the stratosphere included constant temperature with height. Recent data show the temperature is constant at 216.66°K between about 7 and 14 miles, increases to approximately 270°K at 30 miles, and decreases to approximately 180°K at 50 miles. Since the temperature variation between 14 and 50 miles destroys one of the basic definitions of the stratosphere, some authors divide this area into two divisions: stratosphere, 7 to 14 miles, and mesosphere, 15 to 50 miles. The boundary between the troposphere and the stratosphere is the tropopause.

The third major division, the ionosphere, extends from approximately 50 miles to 300 miles. Large numbers of free ions are present in this layer, and a number of different electrical phenomenon take place in this division. The temperature increases with height to 1500˚K at 300 miles.

The fourth major division is the exosphere. It is the outermost layer of the atmosphere. It starts at 300 miles and is characterized by a large number of free ions. Molecular temperature increases with height.

2.3.3 STANDARD ATMOSPHERE

The physical characteristics of the atmosphere change daily and seasonally. Since aircraft performance is a function of the physical characteristics of the air mass through which it flies, performance varies as the air mass characteristics vary. Thus, standard air mass conditions are established so performance data has meaning when used for comparison purposes. In the case of the altimeter, the standard allows for design of an instrument for measuring altitude.

At the present time there are several established atmosphere standards. One commonly used is the Arnold Research and Development Center (ARDC) 1959 model atmosphere. A more recent one is the U.S. Standard Atmosphere, 1962. These standard atmospheres were developed to approximate the standard average day conditions at 40˚ to 45˚N latitude.

These two standard atmospheres are basically the same up to an altitude of approximately 66,000 feet. Both the 1959 ARDC and the 1962 U.S. Standard Atmosphere are defined to an upper limit of approximately 440 miles. At higher levels there are some marked differences between the 1959 and 1962 atmospheres. The standard atmosphere used by the U.S. Naval Test Pilot School (USNTPS) is the 1962 atmosphere. Appendix VI gives the 1962 atmosphere in tabular form.

The U.S. Standard Atmosphere, 1962 assumes:

1. The atmosphere is a perfect gas which obeys the equation of state:

$$P = \rho \, g_c \, R \, T \qquad \text{(Eq 2.1)}$$

2. The air is dry.
3. The standard sea level conditions:

a_{ssl}	Standard sea level speed of sound	661.483 kn
g_{ssl}	Standard sea level gravitational acceleration	32.174049 ft/s^2
P_{ssl}	Standard sea level pressure	2116.217 psf
		29.9212 inHg
ρ_{ssl}	Standard sea level air density	0.0023769 $slugs/ft^3$
T_{ssl}	Standard sea level temperature	15˚C or 288.15˚K.

4. The gravitational field decreases with altitude.
5. Hydrostatic equilibrium exists such that:

$$dP_a = - \rho \, g \, dh \qquad \text{(Eq 2.2)}$$

6. Vertical displacement is measured in geopotential feet. Geopotential is a measure of the gravitational potential energy of a unit mass at a point relative to mean sea level and is defined in differential form by the equation:

$$g_{ssl} \, dH = g \, dh \qquad \text{(Eq 2.3)}$$

Where:

g	Gravitational acceleration (Varies with altitude)	ft/s
g_c	Conversion constant	32.17 $lb_m/slug$
g_{ssl}	Standard sea level gravitational acceleration	32.174049 ft/s^2
H	Geopotential (At the point)	ft
h	Tapeline altitude	ft
P	Pressure	psf

P_a	Ambient pressure	psf
R	Engineering gas constant for air	96.93 ft-lb_f/lb_m - ˚K
ρ	Air density	slug/ft^3
T	Temperature	˚K.

Each point in the atmosphere has a definite geopotential, since g is a function of latitude and altitude. Geopotential is equivalent to the work done in elevating a unit mass from sea level to a tapeline altitude expressed in feet. For most purposes, errors introduced by letting h = H in the troposphere are insignificant. Making this assumption, there is slightly more than a 2% error at 400,000 feet.

7. Temperature variation with geopotential is expressed as a series of straight line segments:

a. The temperature lapse rate (a) in the troposphere (sea level to 36,089 geopotential feet) is 0.0019812˚C/geopotential feet.

b. The temperature above 36,089 geopotential feet and below 65,600 geopotential feet is constant -56.50˚C.

2.3.3.1 STANDARD ATMOSPHERE EQUATIONS

From the basic assumptions for the standard atmosphere listed above, the relationships for temperature, pressure, and density as functions of geopotential are derived.

Below 36,089 geopotential feet, the equations for the standard atmosphere are:

$$\theta = \frac{T_a}{T_{ssl}} = \left(1 - 6.8755856 \times 10^{-6} H\right) \qquad \text{(Eq 2.4)}$$

$$\delta = \frac{P_a}{P_{ssl}} = \left(1 - 6.8755856 \times 10^{-6} H\right)^{5.255863} \qquad \text{(Eq 2.5)}$$

$$\sigma = \frac{\rho_a}{\rho_{ssl}} = \left(1 - 6.8755856 \times 10^{-6} H\right)^{4.255863} \qquad \text{(Eq 2.6)}$$

$$P_a = P_{ssl} \left(1 - 6.8755856 \times 10^{-6} H_P\right)^{5.255863} \qquad \text{(Eq 2.7)}$$

Above 36,089 geopotential feet and below 82,021 geopotential feet the equations for the standard atmosphere are:

$$T_a = -56.50^\circ C = 216.65^\circ K \qquad \text{(Eq 2.8)}$$

$$\delta = \frac{P_a}{P_{ssl}} = 0.223358\ e^{-4.80614 \times 10^{-5} (H - 36089)} \qquad \text{(Eq 2.9)}$$

$$\sigma = \frac{\rho_a}{\rho_{ssl}} = 0.297069\ e^{-4.80614 \times 10^{-5} (H - 36089)} \qquad \text{(Eq 2.10)}$$

$$P_a = P_{ssl} \left(0.223358\ e^{-4.80614 \times 10^{-5} \left(H_P - 36089\right)}\right) \qquad \text{(Eq 2.11)}$$

Where:

δ	Pressure ratio	
e	Base of natural logarithm	
H	Geopotential	ft
H_P	Pressure altitude	ft
P_a	Ambient pressure	psf
P_{ssl}	Standard sea level pressure	2116.217 psf
θ	Temperature ratio	
ρ_a	Ambient air density	slug/ft^3
ρ_{ssl}	Standard sea level air density	0.0023769 slug/ft^3
σ	Density ratio	
T_a	Ambient temperature	˚C or ˚K

T_{ssl}	Standard sea level temperature	15˚C or 288.15˚K.

2.3.3.2 ALTITUDE MEASUREMENT

With the establishment of a set of standards for the atmosphere, there are several different means to determine altitude above the ground. The means used defines the type of altitude. Tapeline altitude, or true altitude, is the linear distance above sea level and is determined by triangulation or radar.

A temperature altitude can be obtained by modifying a temperature gauge to read in feet for a corresponding temperature, determined from standard tables. However, since inversions and nonstandard lapse rates exist, and temperature changes daily, seasonally, and with latitude, such a technique is not useful.

If an instrument were available to measure density, the same type of technique could be employed, and density altitude could be determined.

If a highly sensitive accelerometer could be developed to measure gravitational acceleration, geopotential altitude could be measured. This device would give the correct reading in level, unaccelerated flight.

A practical fourth technique, is based on pressure measurement. A pressure gauge is used to sense the ambient pressure. Instead of reading pounds per square foot, it indicates the corresponding standard altitude for the pressure sensed. This altitude is pressure altitude, H_P, and is the parameter on which flight testing is based.

2.3.3.3 PRESSURE VARIATION WITH ALTITUDE

The pressure altitude technique is the basis for present day altimeters. The instrument only gives a true reading when the pressure at altitude is the same as standard day. In most cases, pressure altitude does not agree with the geopotential or tapeline altitude.

Most present day altimeters are designed to follow Eq 2.5. This equation is used to determine standard variation of pressure with altitude below the tropopause. An example of the variation described by Eq 2.5 is presented in figure 2.1.

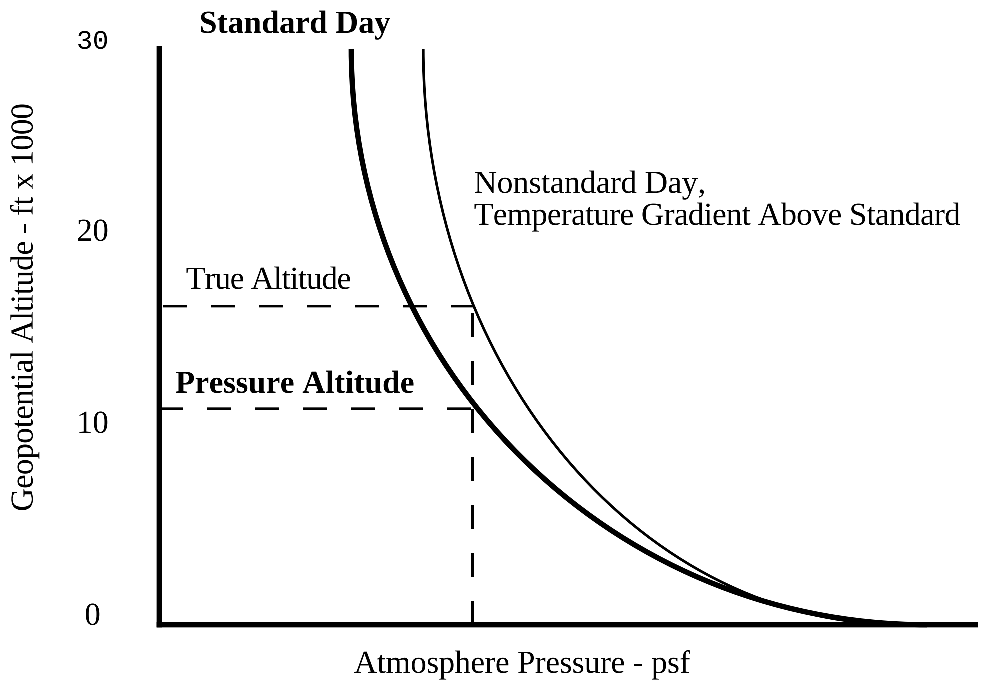


Figure 2.1
PRESSURE VARIATION WITH ALTITUDE

The altimeter presents the standard pressure variation in figure 2.1 as observed pressure altitude, H_{P_o}. If the pressure does not vary as described by this curve, the altimeter indication will be erroneous. The altimeter setting, a provision made in the construction of the altimeter, is used to adjust the scale reading up or down so the altimeter reads true elevation if the aircraft is on deck.

Figure 2.1 shows the pressure variation with altitude for a standard and non-standard day or test day. For every constant pressure (Figure 2.1), the slope of the test day curve is greater than the standard day curve. Thus, the test day temperature is warmer than the standard day temperature. This variance between true altitude and pressure altitude is important for climb performance. A technique is available to correct pressure altitude to true altitude.

The forces acting on an aircraft in flight are directly dependent upon air density. Density altitude is the independent variable which should be used for aircraft performance comparisons. However, density altitude is determined by pressure and temperature through the equation of state relationship. Therefore, pressure altitude is used as the independent variable with test day data corrected for non-standard temperature. This greatly facilitates flight testing since the test pilot can maintain a given pressure altitude regardless of the test day conditions. By applying a correction for non-standard temperature to flight test data, the data is corrected to a standard condition.

2.3.4 ALTIMETER SYSTEMS

Most altitude measurements are made with a sensitive absolute pressure gauge, an altimeter, scaled so a pressure decrease indicates an altitude increase in accordance with the U.S. Standard Atmosphere. If the altimeter setting is 29.92, the altimeter reads pressure altitude, H_P, whether in a standard or non-standard atmosphere. An altimeter setting other than 29.92 moves the scale so the altimeter indicates field elevation with the aircraft on deck. In this case, the altimeter indication is adjusted to show tapeline altitude at one elevation. In flight testing, 29.92 is used as the altimeter setting to read pressure altitude. Pressure altitude is not dependent on temperature. The only parameter which varies the altimeter indication is atmospheric pressure.

The altimeter is constructed and calibrated according to Eq 2.7 and 2.11 which define the standard atmosphere. The heart of the altimeter is an evacuated metal bellows which expands or contracts with changes in outside pressure. The bellows is connected to a series of gears and levers which cause a pointer to move. The whole mechanism is placed in an airtight case which is vented to a static source. The indicator reads the pressure supplied to the case. Altimeter construction is shown in figure 2.2. The altimeter senses the change in static pressure, P_S, through the static source.

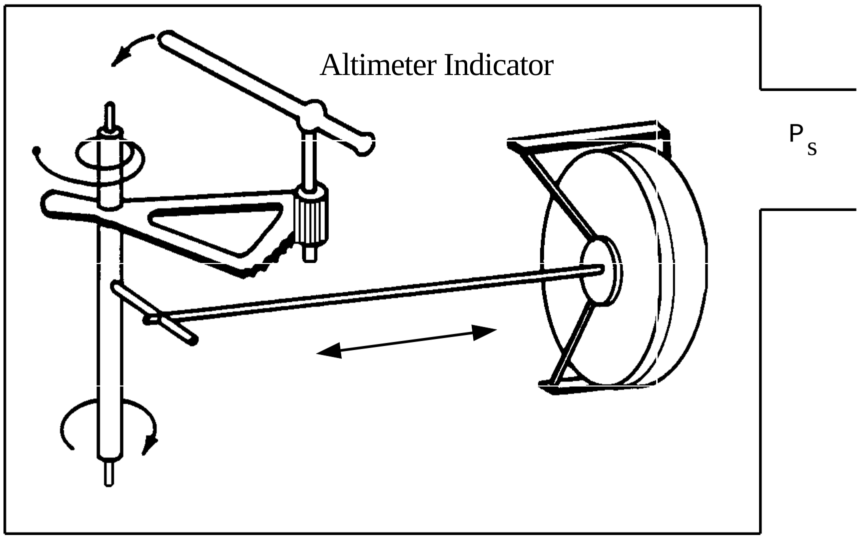


Figure 2.2
ALTIMETER SCHEMATIC

2.3.5 AIRSPEED SYSTEMS

Airspeed system theory was first developed with the assumption of incompressible flow. This assumption is only useful for low speeds of 250 knots or less at relatively low altitudes. Various concepts and nomenclature of incompressible flow are in use and provide a step toward understanding compressible flow relations.

2.3.5.1 INCOMPRESSIBLE AIRSPEED

True airspeed, in the incompressible case, is defined as:

$$V_T = \sqrt{\frac{2}{\rho_a}\left(P_T - P_a\right)} = \sqrt{\frac{2q}{\rho_a}} \qquad \text{(Eq 2.12)}$$

It is possible to use a pitot static system and build an airspeed indicator to conform to this equation. However, there are disadvantages:

1. Density requires measurement of ambient temperature, which is difficult in flight.

2. The instrument would be complex. In addition to the bellows in figure 2.3, ambient temperature and pressure would have to be measured, converted to density, and used to modify the output of the bellows.

3. Except for navigation, the instrument would not give the required pilot information. For landing, the aircraft is flown at a constant lift coefficient, C_L. Thus, the pilot would compute a different landing speed for each combination of weight, pressure altitude, and temperature.

4. Because of its complexity, the instrument would be inaccurate and difficult to calibrate.

Density is the variable which causes the problem in a true airspeed indicator. A solution is to assume a constant value for density. If ρ_a is replaced by ρ_{ssl} in Eq 2.12, the resultant velocity is termed equivalent airspeed, V_e:

$$V_e = \sqrt{\frac{2q}{\rho_{ssl}}} = \sqrt{\frac{\sigma\, 2q}{\rho_a}} = \sqrt{\sigma}\, V_T \tag{Eq 2.13}$$

A simple airspeed indicator could be built which measures the quantity (P_T - P_a). Such a system requires only the bellows system shown in figure 2.3 and has the following advantages:

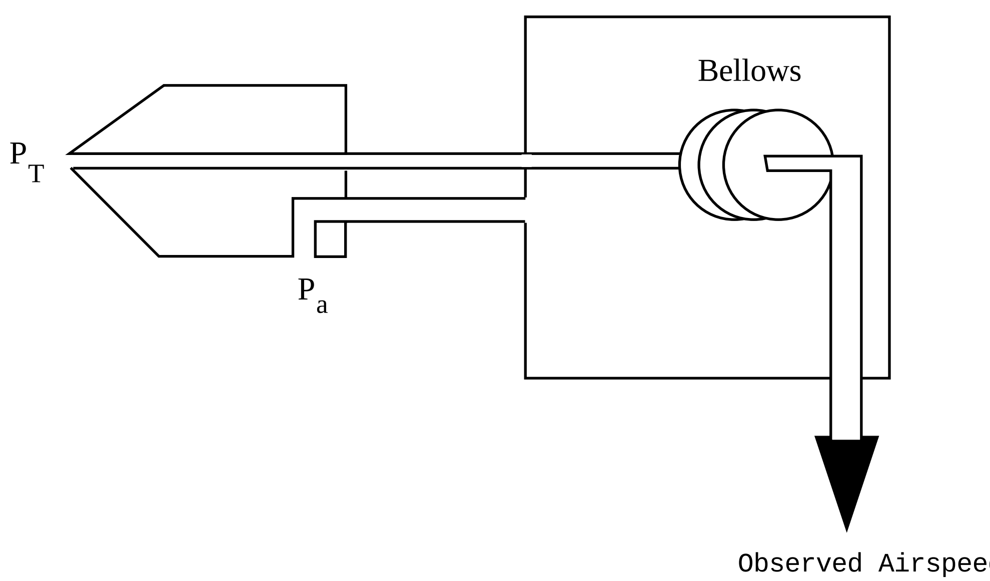


Figure 2.3
PITOT STATIC SYSTEM SCHEMATIC

1. Because of its simplicity, it has a high degree of accuracy.

2. The indicator is easy to calibrate and has only one error due to airspeed instrument correction (ΔV_{ic}).

3. The pilot can use V_e. In computing either landing or stall speed, the pilot only considers weight.

4. Since $V_e = f\ (P_T - P_a)$, it does not vary with temperature or density. Thus for a given value of $P_T - P_a$:

$$V_{e_{Test}} = V_{e_{Std}} \qquad \text{(Eq 2.14)}$$

Where:

P_a	Ambient pressure	psf
P_T	Total pressure	psf
q	Dynamic pressure	psf
ρ_a	Ambient air density	slug/ft^3
ρ_{ssl}	Standard sea level air density	0.0023769 slug/ft^3
σ	Density ratio	
V_e	Equivalent airspeed	ft/s
$V_{e_{Std}}$	Standard equivalent airspeed	ft/s
$V_{e_{Test}}$	Test equivalent airspeed	ft/s
V_T	True airspeed	ft/s.

V_e derived for the incompressible case was the airspeed primarily used before World War II. However, as aircraft speed and altitude capabilities increased, the error resulting from the assumption that density remains constant became significant. Airspeed indicators for today’s aircraft are built to consider compressibility.

2.3.5.2 COMPRESSIBLE TRUE AIRSPEED

The airspeed indicator operates on the principle of Bernoulli's compressible equation for isentropic flow in which airspeed is a function of the difference between total and static pressure. At subsonic speeds Bernoulli's equation is applicable, giving the following expression for V_T:

$$V_T^2 = \frac{2\gamma}{\gamma - 1}\frac{P_a}{\rho_a}\left[\left(\frac{P_T - P_a}{P_a} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right] \qquad \text{(Eq 2.15)}$$

Or:

$$V_T = \sqrt{\frac{2\gamma}{\gamma - 1}\frac{P_a}{\rho_a}\left[\left(\frac{q_c}{P_a} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]} \qquad \text{(Eq 2.16)}$$

Dynamic pressure, q, and impact pressure, q_c, are not the same. However, at low altitude and low speed they are approximately the same. The relationship between dynamic pressure and impact pressure converges as Mach becomes small as follows:

$$q_c = q\left(1 + \frac{M^2}{4} + \frac{M^4}{40} + \frac{M^6}{1600} + \dots\right) \qquad \text{(Eq 2.17)}$$

Where:

γ	Ratio of specific heats	
M	Mach number	
P_a	Ambient pressure	psf
P_T	Total pressure	psf
q	Dynamic pressure	psf
q_c	Impact pressure	psf
ρ_a	Ambient air density	slug/ft^3
V_T	True airspeed	ft/s.

2.3.5.3 CALIBRATED AIRSPEED

The compressible flow true airspeed equation (Eq 2.16) has the same disadvantages as the incompressible flow true airspeed case. Additionally, a bellows would have to be added to measure P_a. The simple pitot static system in figure 2.3 only measures P_T - P_a. To modify Eq 2.16 for measuring the quantity P_T - P_a, both ρ_a and P_a are replaced by the constant ρ_{ssl} and P_{ssl}. The resulting airspeed is defined as calibrated airspeed, V_c:

$$V_c^2 = \frac{2\gamma}{\gamma - 1}\frac{P_{ssl}}{\rho_{ssl}}\left[\left(\frac{P_T - P_a}{P_{ssl}} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right] \qquad \text{(Eq 2.18)}$$

Or:

$$V_c = \sqrt{\frac{2\gamma}{\gamma - 1}\frac{P_{ssl}}{\rho_{ssl}}\left[\left(\frac{q_c}{P_{ssl}} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]} \qquad \text{(Eq 2.19)}$$

Or:

$$V_c = f\left(P_T - P_a\right) = f\left(q_c\right) \qquad \text{(Eq 2.20)}$$

An instrument designed to follow Eq 2.19 has the following advantages:

1. The indicator is simple, accurate, and easy to calibrate.
2. V_c is useful to the pilot. The quantity V_c is analogous to V_e in the incompressible case, since at low airspeeds and moderate altitudes $V_e \cong V_c$. The aircraft stall speed, landing speed, and handling characteristics are proportional to calibrated airspeed for a given gross weight.
3. Since temperature or density is not present in the equation for calibrated airspeed, a given value of $(P_T - P_a)$ has the same significance on all days and:

$$V_{c_{Test}} = V_{c_{Std}} \qquad \text{(Eq 2.21)}$$

Eq 2.19 is limited to subsonic flow. If the flow is supersonic, it must pass through a shock wave in order to slow to stagnation conditions. There is a loss of total pressure when the flow passes through the shock wave. Thus, the indicator does not measure the total pressure of the supersonic flow. The solution for supersonic flight is derived by considering a normal shock compression in front of the total pressure tube and an isentropic compression in the subsonic region aft of the shock. The normal shock assumption is good since the pitot tube has a small frontal area. Consequently, the radius of the shock in front of the hole may be considered infinite. The resulting equation is known as the Rayleigh Supersonic Pitot Equation. It relates the total pressure behind the shock P_T' to the free stream ambient pressure P_a and free stream Mach:

$$\frac{P'_T}{P_a} = \left[\frac{\gamma + 1}{2}\left(\frac{V}{a}\right)^2\right]^{\frac{\gamma}{\gamma - 1}} \left[\frac{1}{\frac{2\gamma}{\gamma + 1}\left(\frac{V}{a}\right)^2 - \frac{\gamma - 1}{\gamma + 1}}\right]^{\frac{1}{\gamma - 1}} \quad \text{(Eq 2.22)}$$

Eq 2.22 is used to calculate the ratio of dynamic pressure to standard sea level pressure for super and subsonic flow. The resulting calibrated airspeed equations are as follows:

$$\frac{q_c}{P_{ssl}} = \left[1 + 0.2\left(\frac{V_c}{a_{ssl}}\right)^2\right]^{3.5} - 1 \quad \text{(For } V_c \leq a_{ssl}\text{)} \quad \text{(Eq 2.23)}$$

Or:

$$\frac{q_c}{P_{ssl}} = \left[\frac{166.921\left(\frac{V_c}{a_{ssl}}\right)^7}{\left[7\left(\frac{V_c}{a_{ssl}}\right)^2 - 1\right]^{2.5}}\right] - 1 \quad \text{(For } V_c \geq a_{ssl}\text{)} \quad \text{(Eq 2.24)}$$

Where:

a	Speed of sound	ft/s or kn
a_{ssl}	Standard sea level speed of sound	661.483 kn
γ	Ratio of specific heats	
P_a	Ambient pressure	psf
P_{ssl}	Standard sea level pressure	2116.217 psf
P_T	Total pressure	psf
P_T'	Total pressure at total pressure source	psf
q_c	Impact pressure	psf

ρ_{ssl}	Standard sea level air density	0.0023769 slug/ft^3
V	Velocity	ft/s
V_c	Calibrated airspeed	ft/s
$V_{c_{Std}}$	Standard calibrated airspeed	ft/s
$V_{c_{Test}}$	Test calibrated airspeed	ft/s.

Airspeed indicators are constructed and calibrated according to Eq 2.23 and 2.24. In operation, the airspeed indicator is similar to the altimeter, but instead of being evacuated, the inside of the capsule is connected to the total pressure source, and the case to the static pressure source. The instrument then senses total pressure (P_T) within the capsule and static pressure (P_s) outside it as shown in figure 2.4.

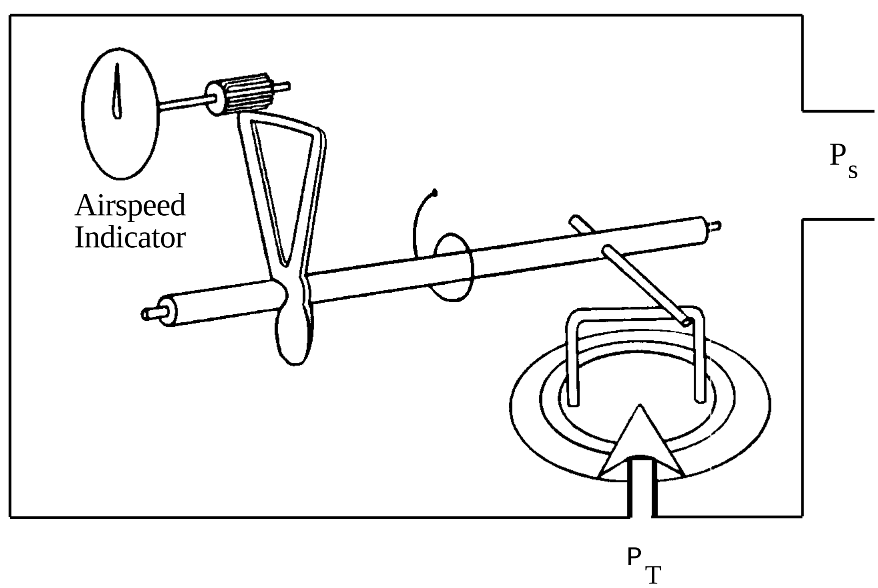


Figure 2.4
AIRSPEED SCHEMATIC

2.3.5.4 EQUIVALENT AIRSPEED

Equivalent airspeed (V_e) was derived from incompressible flow theory and has no real meaning for compressible flow. However, V_e is an important parameter in analyzing certain performance and stability and control parameters since they are functions of equivalent airspeed. The definition of equivalent airspeed is:

$$V_e = \sqrt{\frac{2\gamma}{\gamma - 1}\frac{P_a}{\rho_{ssl}}\left[\left(\frac{q_c}{P_a} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]} \qquad \text{(Eq 2.25)}$$

$$V_e = V_T\sqrt{\sigma} \qquad \text{(Eq 2.26)}$$

Where:

γ	Ratio of specific heats	
P_a	Ambient pressure	psf
q_c	Impact pressure	psf
ρ_{ssl}	Standard sea level air density	0.0023769 slugs/ft^3
σ	Density ratio	
V_e	Equivalent airspeed	ft/s
V_T	True airspeed	ft/s.

2.3.6 MACHMETERS

Mach or Mach number, M, is defined as the ratio of the true airspeed to the local atmospheric speed of sound.

$$M = \frac{V_T}{a} = \frac{V_T}{\sqrt{\gamma\, g_c\, R\, T}} = \frac{V_T}{\sqrt{\gamma \frac{P}{\rho}}} \qquad \text{(Eq 2.27)}$$

Substituting this relationship in the equation for V_T yields:

$$M = \sqrt{\frac{2}{\gamma - 1}\left[\left(\frac{P_T - P_a}{P_a} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]} \qquad \text{(Eq 2.28)}$$

Or:

$$\frac{P_T}{P_a} = \left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{\gamma}{\gamma - 1}} \qquad \text{(Eq 2.29)}$$

This equation, which relates Mach to the free stream total and ambient pressures, is good for supersonic as well as subsonic flight. However, P_T' rather than P_T is measured in supersonic flight. By using the Rayleigh pitot equation and substituting for the constants, we obtain the following expressions:

$$\frac{q_c}{P_a} = \left(1 + 0.2\ M^2\right)^{3.5} - 1 \quad \text{for } M < 1 \qquad \text{(Eq 2.30)}$$

$$\frac{q_c}{P_a} = \left[\frac{166.921\ M^7}{\left(7M^2 - 1\right)^{2.5}}\right] - 1 \quad \text{for } M > 1 \qquad \text{(Eq 2.31)}$$

The Machmeter is essentially a combination altimeter and airspeed indicator designed to solve these equations. An altimeter capsule and an airspeed capsule simultaneously supply inputs to a series of gears and levers to produce the indicated Mach. A Machmeter schematic is presented in figure 2.5. Since the construction of the Machmeter requires two bellows, one for impact pressure (q_c) and another for ambient pressure (P_a), the meter is complex, difficult to calibrate, and inaccurate. As a result, the Machmeter is not used in flight test work except as a reference instrument.

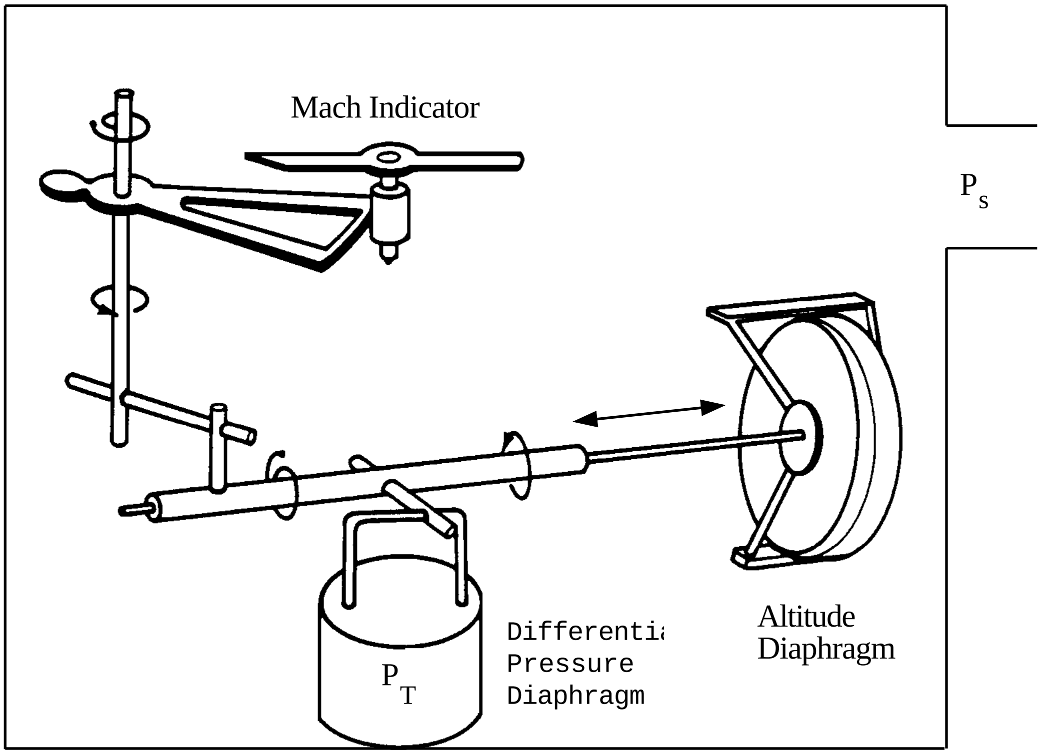


Figure 2.5
MACHMETER SCHEMATIC

Of importance in flight test is the fact:

$$M = f\left(P_T - P_a, P_a\right) = f\left(V_c, H_P\right) \tag{Eq 2.32}$$

As a result, Mach is independent of temperature, and flying at a given pressure altitude (H_P) and calibrated airspeed (V_c), the Mach on the test day equals Mach on a standard day. Since many aerodynamic effects are functions of Mach, particularly in jet engine-airframe performance analysis, this fact plays a major role in flight testing.

$$M_{Test} = M \tag{Eq 2.33}$$

Where:

a	Speed of sound	ft/s or kn
g_c	Conversion constant	32.17 lb_m/slug

γ	Ratio of specific heats	
H_P	Pressure altitude	ft
M	Mach number	
M_{Test}	Test Mach number	
P	Pressure	psf
P_a	Ambient pressure	psf
P_T	Total pressure	psf
q_c	Impact pressure	psf
R	Engineering gas constant for air	96.93 ft-lb_f/lb_m-˚K
ρ	Air density	slug/ft^3
T	Temperature	˚K
V_c	Calibrated airspeed	ft/s
V_T	True airspeed	ft/s.

2.3.7 ERRORS AND CALIBRATION

The altimeter, airspeed, Mach indicator, and vertical rate of climb indicators are universal flight instruments which require total and/or static pressure inputs to function. The indicated values of these instruments are often incorrect because of the effects of three general categories of errors: instrument errors, lag errors, and position errors.

Several corrections are applied to the observed pressure altitude and airspeed indicator readings (H_{P_o}, V_o) before calibrated pressure altitude and calibrated airspeed (H_{P_c}, V_c) are determined. The observed readings must be corrected for instrument error, lag error, and position error.

2.3.7.1 INSTRUMENT ERROR

The altimeter and airspeed indicator are sensitive to pressure and pressure differential respectively, and the dials are calibrated to read altitude and airspeed according to Eq 2.7, 2.11 and 2.23, and 2.24. Perfecting an instrument which represents such nonlinear functions under all flight conditions is not possible. As a result, an error exists called instrument error. Instrument error is the result of several factors:

1. Scale error and manufacturing discrepancies due to an imperfect mechanization of the controlling equations.
2. Magnetic Fields.
3. Temperature changes.
4. Friction.
5. Inertia.
6. Hysteresis.

The instrument calibration of an altimeter and airspeed indicator for instrument error is conducted in an instrument laboratory. A known pressure or pressure differential is applied to the instrument. The instrument error is determined as the difference between this known pressure and the observed instrument reading. As an instrument wears, its calibration changes. Therefore, an instrument is calibrated periodically. The repeatability of the instrument is determined from the instrument calibration history and must be good for a meaningful instrument calibration.

Data are taken in both directions so the hysteresis is determined. An instrument with a large hysteresis is rejected, since accounting for this effect in flight is difficult. An instrument vibrator can be of some assistance in reducing instrument error. Additionally, the instruments are calibrated in a static situation. The hysteresis under a dynamic situation may be different, but calibrating instruments for such conditions is not feasible.

When the readings of two pressure altimeters are used to determine the error in a pressure sensing system, a precautionary check of calibration correlations is advisable. A problem arises from the fact that two calibrated instruments placed side by side with their readings corrected by use of calibration charts do not always provide the same calibrated value. Tests such as the tower fly-by, or the trailing source, require an altimeter to provide a reference pressure altitude. These tests require placing the reference altimeter next to the aircraft altimeter prior to and after each flight. Each altimeter reading should be recorded and, if after calibration corrections are applied, a discrepancy still exists between the two readings, the discrepancy should be incorporated in the data reduction.

Instrument corrections ($\Delta H_{P_{ic}}$, ΔV_{ic}) are determined as the differences between the indicated values (H_{P_i}, V_i) and the observed values (H_{P_o}, V_o):

$$\Delta H_{P_{ic}} = H_{P_i} - H_{P_o} \qquad \text{(Eq 2.34)}$$

$$\Delta V_{ic} = V_i - V_o \qquad \text{(Eq 2.35)}$$

To correct the observed values:

$$H_{P_i} = H_{P_o} + \Delta H_{P_{ic}} \qquad \text{(Eq 2.36)}$$

$$V_i = V_o + \Delta V_{ic} \qquad \text{(Eq 2.37)}$$

Where:

$\Delta H_{P_{ic}}$	Altimeter instrument correction	ft
ΔV_{ic}	Airspeed instrument correction	kn
H_{P_i}	Indicated pressure altitude	ft
H_{P_o}	Observed pressure altitude	ft
V_i	Indicated airspeed	kn
V_o	Observed airspeed	kn.

2.3.7.2 PRESSURE LAG ERROR

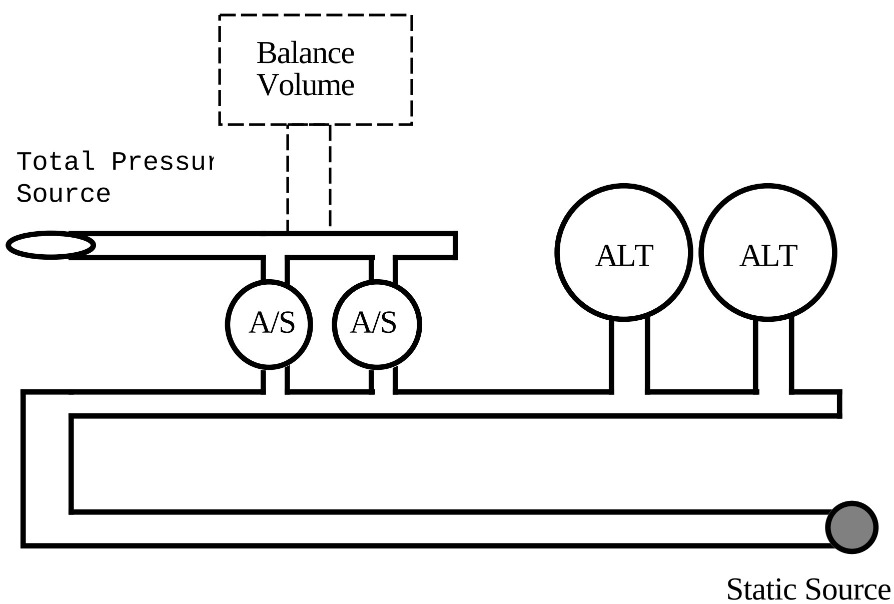
The presence of lag error in pressure measurements is associated generally with climbing/descending or accelerating/decelerating flight and is more evident in static systems. When changing ambient pressures are involved, as in climbing and descending flight, the speed of pressure propagation and the pressure drop associated with flow through a tube introduces lag between the indicated and actual pressure. The pressure lag error is basically a result of:

1. Pressure drop in the tubing due to viscous friction.
2. Inertia of the air mass in the tubing.
3. Volume of the system.
4. Instrument inertia and viscous and kinetic friction.
5. The finite speed of pressure propagation.

Over a small pressure range the pressure lag is small and can be determined as a constant. Once a lag error constant is determined, a correction can be applied. Another approach, which is suitable for flight testing, is to balance the pressure systems by equalizing their volumes. Balancing minimizes or removes lag error as a factor in airspeed data reduction for flight at a constant dynamic pressure.

2.3.7.2.1 LAG CONSTANT TEST

The pitot static pressure systems of a given aircraft supply pressures to a number of different instruments and require different lengths of tubing for pressure transmission. The volume of the instrument cases plus the volume in the tubing, when added together for each pressure system, results in a volume mismatch between systems. Figure 2.6 illustrates a configuration where both the length of tubing and total instrument case volumes are unequal. If an increment of pressure is applied simultaneously across the total and static sources of figure 2.6, the two systems require different lengths of time to stabilize at the new pressure level and a momentary error in indicated airspeed results.



System	Length of 3/16 Inch Inside Diameter Tube	Total Volume of Instrument Cases
Static	18 ft	370×10^{-4} ft^3
Pitot	6 ft	20×10^{-4} ft^3

Figure 2.6

ANALYSIS OF PITOT AND STATIC SYSTEMS CONSTRUCTION

The lag error constant (λ) represents the time (assuming a first order dynamic response) required for the pressure of each system to reach a value equal to 63.2 percent of the applied pressure increment as shown in figure 2.7(a). This test is accomplished on the ground by applying a suction sufficient to develop a change in pressure altitude equal to 500 feet or an indicated airspeed of 100 knots. Removal of the suction and timing the pressure drop to 184 feet or 37 knots results in the determination of λ_S, the static pressure lag error constant (Figures 2.7(b) and 2.7(c)). If a positive pressure is applied to the total pressure pickup (drain holes closed) to produce a 100 knot indication, the total pressure lag error constant (λ_T) can be determined by measuring the time required for the indicator to drop to 37 knots when the pressure is removed. Generally the λ_T will be much smaller than the λ_S because of the smaller volume of the airspeed instrument case.

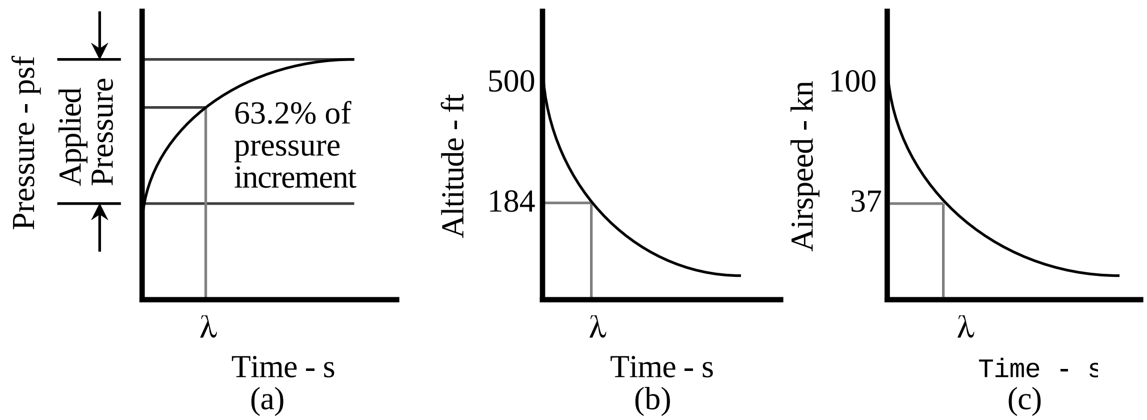


Figure 2.7

PITOT STATIC SYSTEM LAG ERROR CONSTANT

2.3.7.2.2 SYSTEM BALANCING

The practical approach to lag error testing is to determine if a serious lag error exists, and to eliminate it where possible. To test for airspeed system balance, a small increment of pressure (0.1 inch water) is applied simultaneously to both the pitot and static systems. If the airspeed indicator does not fluctuate, the combined systems are balanced and no lag error exists in indicated airspeed data because the lag constants are matched. Movement of the airspeed pointer indicates additional volume is required in one of the systems. The addition of a balance volume (Figure 2.6) generally provides satisfactory airspeed indications. Balancing does not help the lag in the altimeter, as this difficulty is due to the length of the static system tubing. For instrumentation purposes, lag can be eliminated from the altimeter by remotely locating a static pressure recorder at the static port. The use of balanced airspeed systems and remote static pressure sensors is useful for flight testing.

2.3.7.3 POSITION ERROR

Determination of the pressure altitude and calibrated airspeed at which an aircraft is operating is dependent upon the measurement of free stream total pressure, P_T, and free stream ambient pressure, P_a, by the aircraft pitot static system. Generally, the pressures registered by the pitot static system differ from free stream pressures as a result of:

1. The existence of other than free stream pressures at the pressure source.
2. Error in the local pressure at the source caused by the pressure sensors.

The resulting error is called position error. In the general case, position error may result from errors at both the total and static pressure sources.

2.3.7.3.1 TOTAL PRESSURE ERROR

As an aircraft moves through the air, a static pressure disturbance is generated in the air, producing a static pressure field around the aircraft. At subsonic speeds, the flow perturbations due to the aircraft static pressure field are nearly isentropic and do not affect the total pressure. As long as the total pressure source is not located behind a propeller, in the wing wake, in a boundary layer, or in a region of localized supersonic flow, the pressure errors due to the position of the total pressure source are usually negligible. Normally, the total pressure source can be located to avoid total pressure error.

An aircraft capable of supersonic speeds should be equipped with a noseboom pitot static system so the total pressure source is located ahead of any shock waves formed by the aircraft. A noseboom is essential, since correcting for total pressure errors which result when oblique shock waves exist ahead of the pickup is difficult. The shock wave due to the pickup itself is considered in the calibration equation.

Failure of the total pressure sensor to register the local pressure may result from the shape of the pitot static head, inclination to the flow due to angle of attack, α, or sideslip angle, β, or a combination of both. Pitot static tubes are designed in varied shapes. Some are suitable only for relatively low speeds while others are designed to operate in supersonic flight. If a proper design is selected and the pitot tube is not damaged, there should be no error in total pressure due to the shape of the probe. Errors in total pressure caused by the angle of incidence of a probe to the relative wind are negligible for most flight conditions. Commonly used probes produce no significant errors at angles of attack or sideslip up to approximately 20˚. With proper placement, design, and good leak checks of the pitot probe, zero total pressure error is assumed.

2.3.7.3.2 STATIC PRESSURE ERROR

The static pressure field surrounding an aircraft in flight is a function of speed and altitude as well as the secondary parameters, angle of attack, Mach, and Reynold's number. Finding a location for the static pressure source where free stream ambient pressure is sensed under all flight conditions is seldom possible. Therefore, an error generally exists in the measurement of the static pressure due to the position of the static pressure source.

At subsonic speeds, finding some location on the fuselage where the static pressure error is small under all flight conditions is often possible. Aircraft limited to subsonic speeds are instrumented with a flush static pressure ports in such a location.

On supersonic aircraft a noseboom installation is advantageous for measuring static pressure. At supersonic speeds, when the bow wave is located downstream of the static pressure sources, there is no error due to the aircraft pressure field. Any error which may exist is a result of the probe itself. Empirical data suggests free stream static pressure is sensed if the static ports are located more than 8 to 10 tube diameters behind the nose of the pitot static tube and 4 to 6 diameters in front of the shoulder of the pitot tube.

In addition to the static pressure error introduced by the position of the static pressure sources in the pressure field of the aircraft, there may be error in sensing the local static pressure due to flow inclination. Error due to sideslip is minimized by locating flush mounted static ports on opposite sides of the fuselage. For nosebooms, circumferential location of the static pressure ports reduces the adverse effect of sideslip and angle of attack.

2.3.7.3.3 DEFINITION OF POSITION ERROR

The pressure error at the static source has the symbol, ΔP, and is defined as:

$$\Delta P = P_s - P_a \tag{Eq 2.38}$$

The errors associated with ΔP are the position errors. Airspeed position error, ΔV_{pos}, is:

$$\Delta V_{pos} = V_c - V_i \tag{Eq 2.39}$$

Altimeter position error, ΔH_{pos}, is:

$$\Delta H_{pos} = H_{P_c} - H_{P_i} \tag{Eq 2.40}$$

Mach position error, ΔM_{pos}, is:

$$\Delta M_{pos} = M - M_i \tag{Eq 2.41}$$

Where:

ΔH_{pos}	Altimeter position error	ft
ΔM_{pos}	Mach position error	
ΔP	Static pressure error	psf
ΔV_{pos}	Airspeed position error	kn
H_{P_c}	Calibrated pressure altitude	ft
H_{P_i}	Indicated pressure altitude	ft
M	Mach number	
M_i	Indicated Mach number	
P_a	Ambient pressure	psf
P_s	Static pressure	psf
V_c	Calibrated airspeed	kn
V_i	Indicated airspeed	kn.

The definitions result in the position error having the same sign as ΔP. If P_s is greater than P_a, the airspeed indicator indicates a lower than actual value. Therefore, ΔP and ΔV_{pos} are positive in order to correct V_i to V_c. The correction is similar for ΔH_{pos} and ΔM_{pos}.

2.3.7.3.4 STATIC PRESSURE ERROR COEFFICIENT

Dimensional analysis shows the relation of static pressure (P_s) at any point in an aircraft pressure field to the free stream ambient pressure (P_a) depends on Mach (M), angle of attack (α), sideslip angle (β), and Reynold's number (R_e):

$$\frac{P_s}{P_a} = f_1\left(M, \alpha, \beta, R_e\right) \qquad \text{(Eq 2.42)}$$

Reynold's number effects are negligible as the static source is not located in a thick boundary layer, and small sideslip angles are assumed. The relation simplifies to:

$$\frac{P_s}{P_a} = f_2\left(M, \alpha\right) \qquad \text{(Eq 2.43)}$$

This equation can be generalized as follows:

$$\frac{\Delta P}{q_c} = f_3\left(M, \alpha\right) \qquad \text{(Eq 2.44)}$$

The term $\frac{\Delta P}{q_c}$ is the static pressure error coefficient and is used in position error data reduction. Position error data presented as $\frac{\Delta P}{q_c}$ define a single curve for all altitudes.

For flight test purposes the static pressure error coefficient is approximated as:

$$\frac{\Delta P}{q_c} = f_4\left(M\right) \text{ (High speed)} \qquad \text{(Eq 2.45)}$$

$$\frac{\Delta P}{q_c} = f_5\left(C_L\right) \text{ (Low speed)} \qquad \text{(Eq 2.46)}$$

For the high speed case, the indicated relationship is:

$$\frac{\Delta P}{q_{c_i}} = f_6\left(M_i\right) \text{ (High speed)} \tag{Eq 2.47}$$

The high speed indicated static pressure error coefficient is presented as a function of indicated Mach number in figure 2.8.

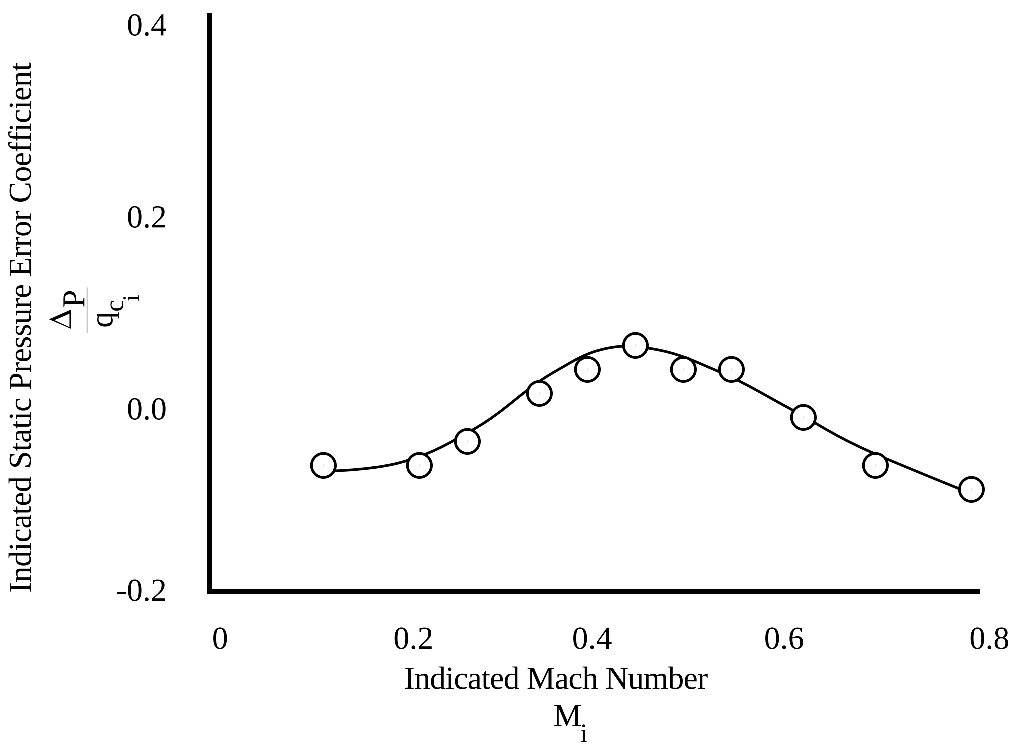


Figure 2.8
HIGH SPEED INDICATED STATIC PRESSURE ERROR COEFFICIENT

For the low speed case, where $C_L = f\ (W,\ n_z,\ V_e)$; and assuming $n_z = 1$ and $V_e \cong V_c$ then:

$$\frac{\Delta P}{q_c} = f_7\left(W, V_c\right) \text{ (Low speed)} \tag{Eq 2.48}$$

The number of independent variables is reduced by relating test weight, W_{Test}, to standard weight, W_{Std}, as follows:

$$V_{c_W} = V_{c_{Test}} \sqrt{\frac{W_{Std}}{W_{Test}}} \quad \text{(Eq 2.49)}$$

Therefore, the expression for the static pressure error coefficient is:

$$\frac{\Delta P}{q_c} = f_8 \left(V_{c_W} \right) \text{ (Low speed)} \quad \text{(Eq 2.50)}$$

For the indicated variables, the low speed relationships are:

$$V_{i_W} = V_{i_{Test}} \sqrt{\frac{W_{Std}}{W_{Test}}} \quad \text{(Eq 2.51)}$$

$$\frac{\Delta P}{q_{c_i}} = f_9 \left(V_{i_W} \right) \text{ (Low speed)} \quad \text{(Eq 2.52)}$$

Where:

α	Angle of attack	deg
β	Sideslip angle	deg
C_L	Lift coefficient	
ΔP	Static pressure error	psf
$\frac{\Delta P}{q_c}$	Static pressure error coefficient	
$\frac{\Delta P}{q_{c_i}}$	Indicated static pressure error coefficient	
M	Mach number	
M_i	Indicated Mach number	
n_z	Normal acceleration	g
P_a	Ambient pressure	psf
P_s	Static pressure	psf
q_c	Impact pressure	psf
q_{c_i}	Indicated impact pressure	psf
R_e	Reynold's number	
V_c	Calibrated airspeed	kn

$V_{c_{Test}}$	Test calibrated airspeed	kn
V_{c_W}	Calibrated airspeed corrected to standard weight	kn
V_e	Equivalent airspeed	kn
$V_{i_{Test}}$	Test indicated airspeed	kn
V_{i_W}	Indicated airspeed corrected to standard weight	kn
W	Weight	lb
W_{Std}	Standard weight	lb
W_{Test}	Test weight	lb.

The low speed indicated static pressure error coefficient is presented as a function of indicated airspeed corrected to standard weight in figure 2.9.

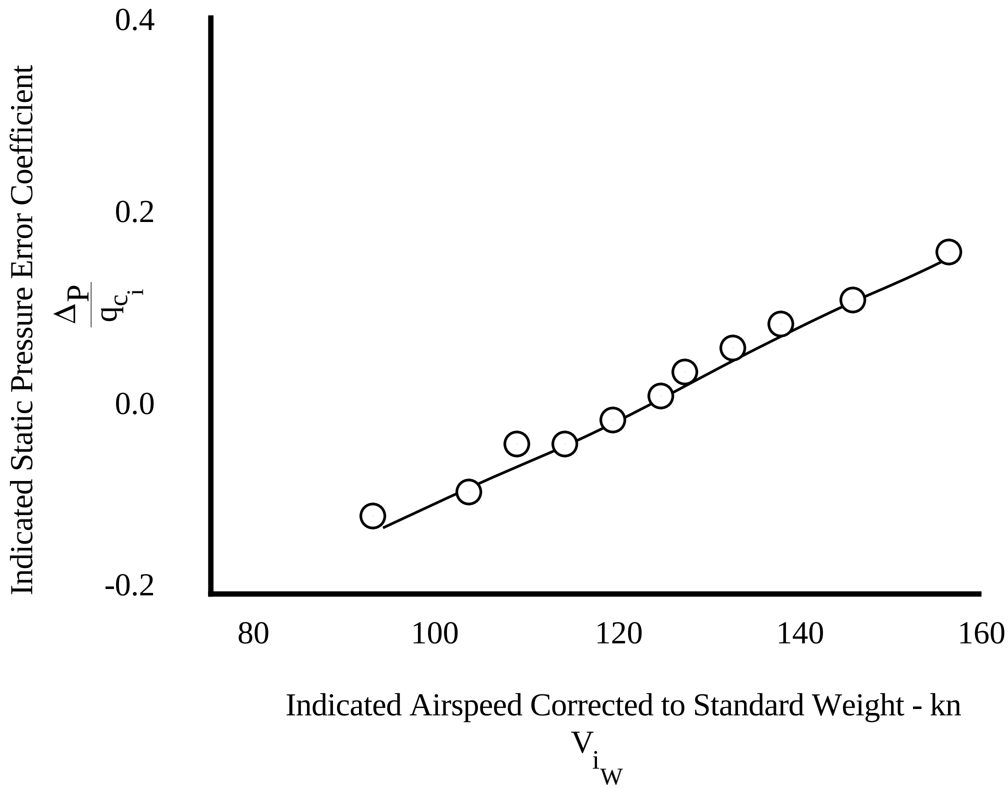


Figure 2.9
LOW SPEED INDICATED STATIC PRESSURE ERROR COEFFICIENT

2.3.8 PITOT TUBE DESIGN

The part of the total pressure not sensed through the pitot tube is referred to as pressure defect, and is a function of angle of attack. However, pressure defect is also a function of Mach number and orifice diameter. As explained in reference 4, the total pressure defect increases as the angle of attack or sideslip angle increases from zero; decreases as Mach number increases subsonically; and decreases as the ratio of orifice diameter to tube outside diameter increases. In general, if the ratio of orifice diameter to tube diameter is equal to one, the total pressure defect is zero up to angles of attack of 25 degrees. As the diameter ratio decreases to 0.74, the defect is still insignificant. But as the ratio of diameters decreases to 0.3, there is approximately a 5 percent total pressure defect at 15 degrees angle of attack, 12 percent at 20 degrees, and 22 percent at 25 degrees. For given values of orifice diameter and tube diameter, with an elongated nose shape, the elongation is equivalent to an effective increase in the ratio of diameters and the magnitude of the total pressure defect will be less than is indicated above for a hemispherical head. These pitot tube design guidelines are general rules for accurate sensing of total pressure. All systems must be evaluated in flight test, but departure from these proven design parameters should prompt particular interest.

2.3.9 FREE AIR TEMPERATURE MEASUREMENT

Knowledge of ambient temperature in flight is essential for true airspeed measurement. Accurate temperature measurement is needed for engine control systems, fire control systems, and weapon release computations.

From the equations derived for flow stagnation conditions, total temperature, T_T, is expressed as:

$$\frac{T_T}{T} = 1 + \frac{\gamma - 1}{2} M^2 \qquad \text{(Eq 2.53)}$$

Expressed in terms of true airspeed:

$$\frac{T_T}{T} = 1 + \frac{\gamma - 1}{2} \frac{V_T^2}{\gamma\, g_c\, R\, T} \qquad \text{(Eq 2.54)}$$

These temperature relations assume adiabatic flow or no addition or loss of heat while bringing the flow to stagnation. Isentropic flow is not required. Therefore, Eq 2.53 and 2.54 are valid for supersonic and subsonic flows. If the flow is not perfectly adiabatic, a temperature recovery factor, K_T, is used to modify the kinetic term as follows:

$$\frac{T_T}{T} = 1 + \frac{K_T(\gamma - 1)}{2} M^2 \qquad \text{(Eq 2.55)}$$

$$\frac{T_T}{T} = 1 + \frac{K_T(\gamma - 1)}{2} \frac{V_T^2}{\gamma g_c R T} \qquad \text{(Eq 2.56)}$$

If the subscripts are changed for the case of an aircraft and the appropriate constants are used:

$$\frac{T_T}{T_a} = \frac{T_i}{T_a} = 1 + \frac{K_T M^2}{5} \qquad \text{(Eq 2.57)}$$

$$T_T = T_i = T_a + \frac{K_T V_T^2}{7592} \qquad \text{(Eq 2.58)}$$

Where:

g_c	Conversion constant	32.17 lb_m/slug
γ	Ratio of specific heats	
K_T	Temperature recovery factor	
M	Mach number	
R	Engineering gas constant for air	96.93 ft-lb_f/lb_m-˚K
T	Temperature	˚K
T_a	Ambient temperature	˚K
T_i	Indicated temperature	˚K
T_T	Total temperature	˚K
V_T	True airspeed	kn.

The temperature recovery factor, K_T, indicates how closely the total temperature sensor observes the total temperature. The value of K_T varies from 0.7 to 1.0. For test systems a range of 0.95 to 1.0 is common. There are a number of errors possible in a temperature indicating system. In certain installations, these may cause the recovery factor to vary with airspeed. Generally, the recovery factor is a constant value. The following are the more significant errors:

1. Resistance - Temperature Calibration. Generally, building a resistance temperature sensing element which exactly matches the prescribed resistance - temperature curve is not possible. A full calibration of each probe is made, and the instrument correction, ΔT_{ic}, applied to the data.

2. Conduction Error. A clear separation between recovery errors and errors caused by heat flow from the temperature sensing element to the surrounding structure is difficult to make. This error can be reduced by insulating the probe. Data shows this error is small.

3. Radiation Error. When the total temperature is relatively high, heat is radiated from the sensing element, resulting in a reduced temperature indication. This effect is increased at very high altitude. Radiation error is usually negligible for well designed sensors when Mach is less than 3.0 and altitude is below 40,000 feet.

4. Time Constant. The time constant is defined as the time required for a certain percentage of the response to an instantaneous change in temperature to be indicated on the instrument. When the temperature is not changing or is changing at an extremely slow rate, the time constant introduces no error. Practical application of a time constant in flight is extremely difficult because of the rate of change of temperature with respect to time. The practical solution is to use steady state testing.

2.3.9.1 TEMPERATURE RECOVERY FACTOR

The temperature recovery system has two errors which must be accounted for, instrument correction, ΔT_{ic}, and temperature recovery factor, K_T. Although ΔT_{ic} is called instrument correction, it accounts for many system errors collectively from the indicator to the temperature probe. The ΔT_{ic} correction is obtained under controlled laboratory conditions.

The temperature recovery factor, K_T, measures the temperature recovery process adiabatically. A value of 1.0 for K_T is ideal, but values greater than 1.0 are observed when heat is added to the sensors by conduction (hot material around the sensor) or radiation (exposure to direct sunlight). The test conditions must be selected to minimize this type of interference.

Normally, temperature probe calibration can be done simultaneously with pitot static calibration. Indicated temperature, instrument correction, aircraft true Mach, and an accurate ambient temperature are the necessary data. The ambient temperature is obtained from a reference source such as a pacer aircraft, weather balloon, or tower thermometer. Accurate ambient temperature may be difficult to obtain on a tower fly-by test because of steep temperature gradients near the surface.

The temperature recovery factor at a given Mach may be computed as follows:

$$T_i = T_o + \Delta T_{ic} \tag{Eq 2.59}$$

$$K_T = \left(\frac{T_i\,(^\circ K)}{T_a\,(^\circ K)} - 1 \right) \frac{5}{M^2} \tag{Eq 2.60}$$

Where:

ΔT_{ic}	Temperature instrument correction	˚C
K_T	Temperature recovery factor	
M	Mach number	
T_a	Ambient temperature	˚C or ˚K
T_i	Indicated temperature	˚C or ˚K
T_o	Observed temperature	˚C.

2.4 TEST METHODS AND TECHNIQUES

The objective of pitot static calibration test is to determine position error in the form of the static pressure error coefficient. From the static pressure error coefficient, ΔV_{pos} and ΔH_{pos} are determined. The test is designed to produce an accurate calibrated pressure altitude (H_{P_c}), calibrated velocity (V_c), or Mach (M), for the test aircraft. Position error is sensitive to Mach, configuration, and perhaps angle of attack depending upon the type of static source. Choose the test method to take advantage of the capability of the instrumentation. Altimeter position error (ΔH_{pos}) is usually evaluated because H_{P_c} is fairly easy to determine, and the error can be read more accurately on the altimeter.

The test methods for calibrating pitot and static systems are numerous and often a test is known by several different titles within the aviation industry. Often, more than one system requires calibration, such as separate pilot, copilot, and flight test systems. Understanding the particular system plumbing is important for calibrating the required systems. The most common calibration techniques are presented and discussed briefly. Do not overlook individual instrument calibration in these tests. Leak check pitot static systems prior to calibration test programs.

One important part of planning for any flight test is the data card. Organize the card to assist the crew during the flight and emphasize the most important flight parameters. Match the inputs for a computer data reduction program to the order of test parameters. H_{P_o} is read first because it is the critical parameter, and the other parameters are listed in order of decreasing sensitivity. The tower operator’s data card includes the tower elevation and the same run numbers with columns for theodolite reading, time, temperature, and tower pressure altitude. The time entry allows correlation between tower and flight data points. Include space on both cards for repeated or additional data points.

There are a few considerations for pilot technique during pitot static calibration flights. During stabilized points, fly the aircraft in coordinated flight, with the altitude and angle of attack held steady. Pitch bobbling or sideslip induce error, so resist making last second corrections. A slight climb or descent may cause the pilot to read the wrong altitude, particularly if there is any delay in reading the instrument. When evaluating altimeter position error, read the altimeter first. A slight error in the airspeed reading will not have much effect.

2.4.1 MEASURED COURSE

The measured course method is an airspeed calibration which requires flying the aircraft over a course of known length to determine true airspeed (V_T) from time and distance data. Calibrated airspeed, calculated from true airspeed, is compared to the indicated airspeed to obtain the airspeed position error. The conversion of true airspeed to calibrated airspeed requires accurate ambient temperature data. The validity of this test method is predicated on several important parameters:

1. Accuracy of elapsed time determination.
2. Accuracy of course measurement and course length.
3. A constant airspeed over the course.
4. Wind conditions.
5. Accurate temperature data.

Measurement of elapsed time is important and is one of the first considerations when preparing for a test. Elapsed time can be measured with extremely accurate electronic devices. On the other end of the spectrum is the human observer with a stopwatch.

Flying a measured course requires considerable pilot effort to maintain a stabilized airspeed for a prolonged period of time in close proximity to the ground. The problems involved in this test are a function of the overall aircraft flying qualities and vary with different aircraft. Averaging or integrating airspeed fluctuations is not conducive to accurate results. The pilot must maintain flight with small airspeed variations for some finite period of time at a given airspeed. This period of time is generally short on the backside of the level flight power polar. An estimate of the maximum time which stable airspeeds can be maintained for the particular aircraft is made to establish the optimum course length for the different airspeeds to be evaluated.

Ideally, winds should be calm when using the measured course. Data taken with winds can be corrected, provided wind direction and speed are constant. Wind data is collected for each data point using calibrated sensitive equipment located close to the ground speed course. In order to determine the no wind curve, runs are made in both directions (reciprocal headings). All runs must be flown on the course heading, allowing the aircraft to drift with the wind as shown in figure 2.10.

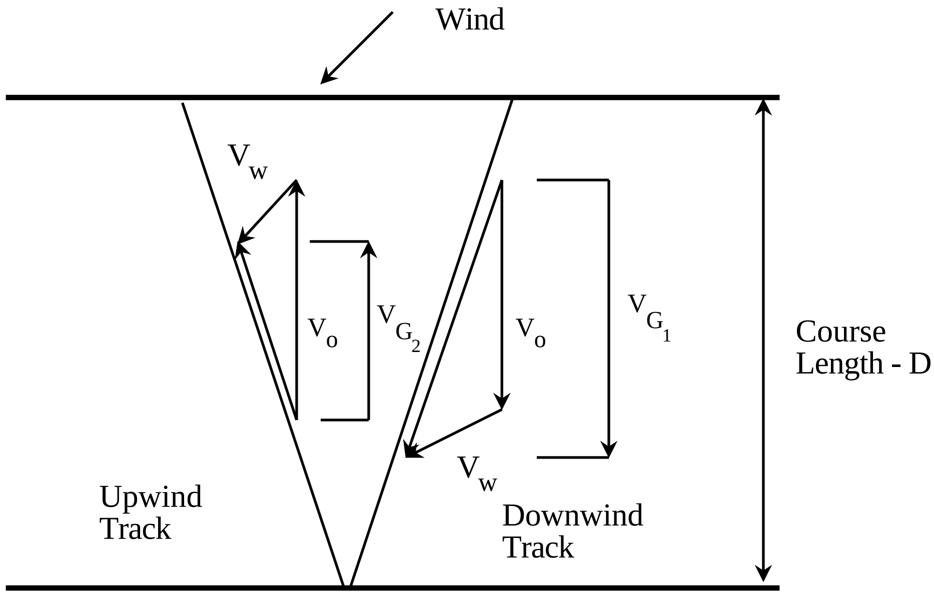


Figure 2.10
WIND EFFECT

True airspeed is determined by averaging the ground speeds. Calibrated airspeed is calculated using standard atmosphere relationships. This method of airspeed and altimeter system calibration is limited to level flight data point calibrations.

The speed course may vary in sophistication from low and slow along a runway or similarly marked course to high and fast when speed is computed by radar or optical tracking.

2.4.1.1 DATA REQUIRED

D, Δt, V_o, H_{P_o}, T_o, GW, $T_{a\ ref}$, $H_{P\ ref}$
Configuration
Wind data.

2.4.1.2 TEST CRITERIA

1. Coordinated, wings level flight.
2. Constant aircraft heading.
3. Constant airspeed.

4. Constant altitude.
5. Constant wind speed and direction.

2.4.1.3 DATA REQUIREMENTS

1. Stabilize 10 s prior to course start.
2. Record data during course length.
3. $V_o \pm 0.5$ kn.
4. $H_{P_o} \pm 20$ ft over course length.

2.4.1.4 SAFETY CONSIDERATIONS

Since these tests are conducted in close ground proximity, the flight crew must maintain frequent visual ground contact. The concentration required to fly accurate data points sometimes distracts the pilot from proper situational awareness. Often these tests are conducted over highly uniform surfaces (water or dry lake bed courses), producing significant depth perception hazards.

2.4.2 TRAILING SOURCE

Static pressure can be measured by suspending a static source on a cable and comparing the results directly with the static systems installed in the test aircraft. The trailing source static pressure is transmitted through tubes to the aircraft where it is converted to accurate pressure altitude by sensitive, calibrated instruments. Since the pressure from the source is transmitted through tubes to the aircraft for conversion to altitude, no error is introduced by trailing the source below the aircraft. The altimeter position error for a given flight condition can be determined directly by subtracting the trailing source altitude from the altitude indicated by the aircraft system. The trailing source cable should be a minimum of 2 wing spans in length, with as small an outside diameter as practical, and a rough exterior finish. The maximum speed of this test method may be limited to the speed at which the trailing source becomes unstable. Depending upon the frequency of the cable oscillation and the resultant maximum displacement of the towed source, large errors may be introduced into the towed source measurements. These errors are reflected as scatter. In addition to the errors induced by the tube oscillations, a fin stabilized source, once disturbed, tends to fly by itself and may move up into the downwash or wing vortices.

The test aircraft is stabilized prior to recording a data point; however, the trailing source may or may not be stable when the data is recorded. Therefore, a means must be provided to monitor the trailing source. When using a trailing source, record data when the aircraft and source are both stabilized in smooth air.

Since there is no method of predicting the point of instability of these trailing systems, monitor the trailing source continuously and plan the flight to accomplish moderate speed data points first and progress in a build-up fashion toward the higher speed points. Trailing source systems are known to exhibit instabilities at both very low speeds and high speeds. There are two main types of trailing sources, the trailing bomb and the trailing cone.

2.4.2.1 TRAILING BOMB

The aircraft static pressure, P_s, is compared directly with the ambient pressure, P_a, measured by a static source on a bomb shaped body suspended on a long length of pressure tubing below the aircraft. The trailing bomb, like the aircraft, may have a static source error. This error is determined by calibration in a wind tunnel.

The length of tubing required to place the bomb in a region where local static pressure approximates free stream pressure is at least twice the aircraft wing span. Since the bomb is below the aircraft, the static pressure is higher, but the pressure lapse in the tubing is the same as the free stream atmospheric pressure lapse. Thus, if the static source in the bomb is attached to an altimeter next to the aircraft, it indicates free stream pressure at altimeter level.

Accuracy depends upon the calibration of the bomb and the accuracy of the pressure gauge or altimeter used to read the trailing bomb’s static pressure. Stability of the bomb at speeds above 0.5 Mach must also be considered.

2.4.2.2 TRAILING CONE

With the trailing cone method, the aircraft’s static pressure, P_s, is compared to the ambient pressure, P_a, measured by a static source trailing behind the aircraft. A light weight cone is attached to the tube to stabilize it and keep the pressure tube taut.

The accuracy depends on the location of the static ports which should be at least six diameters ahead of the cone. The distance behind the aircraft is also important. The aircraft's pitot static instruments are calibrated with the trailing cone in place by tower fly-by or pace methods. These results are used to calibrate the cone installation. The cone can be used with good results as a calibration check of that aircraft's instruments or primary calibration of an aircraft of the same model.

2.4.2.3 DATA REQUIRED

V_o, H_{P_o}, $H_{P_{o\,ref}}$, T_o, GW, Configuration.

Note: Velocity and altitude data must be recorded for each system to be calibrated as well as the trailing source system (reference data).

2.4.2.4 TEST CRITERIA

1. Coordinated, wings level flight.
2. Constant aircraft heading.
3. Constant airspeed.
4. Constant altitude.
5. Steady indications on airspeed and altimeter systems.

2.4.2.5 DATA REQUIREMENTS

1. Stabilize 30 s prior to data record.
2. Record data for 15 s.
3. $V_o \pm 0.5$ kn.
4. $H_{P_o} \pm 10$ ft.

2.4.2.6 SAFETY CONSIDERATIONS

Considerable flight crew or flight and ground crew coordination is required to deploy and recover a trailing source system safely. Thorough planning and detailed pre-flight briefing are essential to ensure that each individual knows the proper procedure.

Trailing source instability stories are numerous. When these devices will exhibit unstable tendencies is difficult to predict. Factors such as probe design, cable length, airspeed, aircraft vibration levels, and atmospheric turbulence influence the onset of these instabilities. Monitor trailing source devices at all times. Chase aircraft normally accomplish this function. Under most circumstances, the onset of the instability is of sufficiently low frequency and amplitude so that corrective action can be taken. In the event the probe starts to exhibit unstable behavior, return to a flight condition which was previously satisfactory. If the instability grows to hazardous proportions, jettison the probe. Jettison devices vary in complexity and must be ground checked by the flight crew to ensure complete familiarity with procedures and proper operation.

2.4.3 TOWER FLY-BY

This method is a simple and excellent way to determine accurately static system error. A tall tower of known height is required as an observation point. The free stream static pressure can be established in any number of ways (such as a sensitive calibrated altimeter in the tower) and is recorded for each pass of the test aircraft. The test aircraft is flown down a predetermined track passing at a known distance (d) from the tower (Figure 2.11). Any deviation in the height of the aircraft above the tower (Δh) is determined by visual observation and simple geometry. The simplicity of this method allows a large number of accurate data points to be recorded quickly and inexpensively.

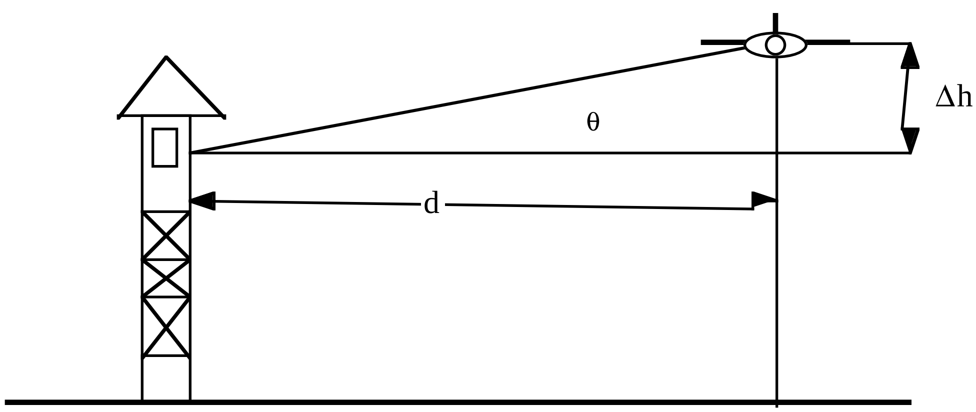


Figure 2.11
TOWER FLY-BY

It is important to ensure there is no false position error introduced during this test as a result of instrument calibration errors. Prior to the flight, with the test aircraft in a static condition, the reference instrument, used to establish the ambient pressure in the tower, is placed next to the aircraft test instrumentation. With both of these instruments in the same environment (and with their respective instrument corrections applied), the indications should be the same. If there is a discrepancy, the difference in readings is included in the data reduction.

The tower fly-by produces a fairly accurate calibrated pressure altitude, H_{P_c}, by triangulation. The aircraft is sighted through a theodolite and the readings are recorded along with tower pressure altitude on each pass.

An alternate method to determine the height of the aircraft above the tower (Δh) is to obtain a grid Polaroid photograph similar to the one in figure 2.12. The height of the aircraft above the tower is determined from the scaled length of the aircraft (x), scaled height of the aircraft above the tower (y), and the known length of the aircraft ($L_{a/c}$). Any convenient units of measure can be used for x and y. This photographic method of determining Δh has the advantage of not requiring the pilot to fly precisely over the predetermined track, thereby compensating for errors in (d) from figure 2.11.

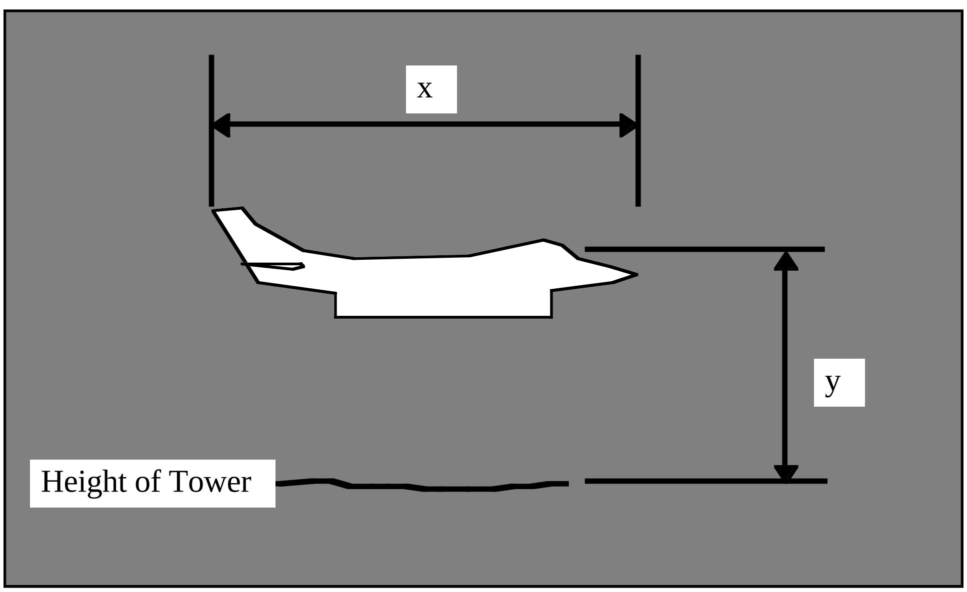


Figure 2.12
SAMPLE TOWER PHOTOGRAPH

The calibrated altitude of the aircraft is the sum of the pressure altitude of the theodolite at the time the point was flown plus the tapeline height above the tower corrected for nonstandard temperature.

Although the tower fly-by method is simple, accurate, and requires no sophisticated equipment, it has some disadvantages. It does not produce an accurate calibrated velocity, it is limited to subsonic flight, and angle of attack changes due to decreasing gross weight may affect the data. Angle of attack effects are most prevalent at low speeds, and all low speed points are flown as close to the same gross weight as possible. Make runs at least one wing span above the ground to remain out of ground effect.

2.4.3.1 DATA REQUIRED

V_o, H_{P_o}, $H_{P_{c\ twr}}$, T_o, $T_{a\ ref}$, GW, Configuration.

Note: Velocity and altitude data must be recorded for each system to be calibrated as well as tower reference pressure altitude, temperature, and aircraft geometric height data (d,θ).

2.4.3.2 TEST CRITERIA

1. Coordinated, wings level flight.
2. Constant heading and track over predetermined path.
3. Constant airspeed.
4. Constant altitude.

2.4.3.3 DATA REQUIREMENTS

1. Stabilize 15 s prior to abeam tower.
2. $V_o \pm 0.5$ kn.
3. $H_{P_o} \pm 10$ ft.
5. Ground track ± 1% of stand-off distance.

2.4.3.4 SAFETY CONSIDERATIONS

This test procedure requires considerable pilot concentration. Maintain situational awareness. Complete familiarity with normal and emergency aircraft procedures prevent excessive pilot distractions. Team work and well briefed/rehearsed data collection procedures also minimize distractions.

Most courses of this type are over highly uniform terrain such as water or dry lake beds, contributing to poor depth perception. A second crew member or ground safety observer can share backup altitude monitoring duties.

2.4.4 SPACE POSITIONING

Space positioning systems vary with respect to principle of operation. Automatic or manual optical tracking systems, radar tracking, and radio ranging systems fall into this category. A space positioning arrangement generally employs at least three tracking stations. These tracking stations track the test aircraft by radar lock or by a manual/visual sight arrangement. Depending upon the configuration, angular and linear displacements are recorded. Through a system of triangulation a computer solution of tapeline altitude and ground speed is obtained. The accuracy of the data can be increased by compensating for tracking errors developed as a result of the random drift away from a prearranged target point on the aircraft. Accuracy can be improved by using an on-board transponder to enhance the tracking process. Regardless of the tracking method, raw data is reduced using computer programs to provide position, velocity, and acceleration information. Normally the test aircraft is flown over a prearranged course to provide the station with a good target and the optimum tracking angles.

Depending on the accuracy desired and the existing wind conditions, balloons can be released and tracked to determine wind velocity and direction. Wind information can be fed into the solution for each data point and true airspeed determined. The true airspeed is used to determine calibrated airspeed and position error.

The use of space positioning systems requires detailed planning and coordination. Exact correlation between onboard and ground recording systems is essential. These systems are generally in high demand by programs competing for resource availability and priority. Due to the inherent complexities of hardware and software, this technique is expensive. The great value of this method is that large amounts of data can be obtained in a short time. Another aspect which makes these systems attractive is the wide variety of flight conditions which can be calibrated, such as climbs and descents. Accelerating and decelerating maneuvers can be time correlated if the data systems are synchronized.

The space positioning radar method is used primarily for calibrations at airspeeds unsuitable for tower fly-by or pacer techniques (i.e., transonic and supersonic speeds). The procedure requires an accurate radar-theodolite system and a pacer aircraft. If a pacer is unavailable, then the position error of the test aircraft must be known for one value of airspeed at the test altitude.

An important aspect of this method is the pressure survey required before the test calibration can be done. To do this, the pace aircraft flies at constant airspeed and altitude through the air mass to be used by the test aircraft. The radar continuously measures the pacer's tapeline altitude from start to finish of the survey. Since the altimeter position error of the pacer is known, the actual pressure altitude flown is known. The pressure altitude of the test aircraft is the tapeline difference between the test and pace aircraft corrected for non-standard temperature.

As soon as possible after completion of the pressure survey, the test aircraft follows the pacer aircraft through the air mass along the same ground track. A tracking beacon is required in the test aircraft, for both accurate radar ranging and to allow ground controllers to provide course corrections when necessary.

Because this method is used for transonic and supersonic portions of the calibration test, an accurate time correlation is necessary to properly relate radar data to aircraft instrumentation data.

2.4.4.1 DATA REQUIRED

V_o, H_{P_o}, T_o, GW, Configuration.

2.4.4.2 TEST CRITERIA

1. Coordinated, wings level flight.
2. Constant aircraft heading.
3. Constant airspeed.
4. Constant altitude.
5. Constant wind speed and direction.

2.4.4.3 DATA REQUIREMENTS

1. Stabilized 30 s prior to data record.
2. Record data for 15 s.
3. $V_o \pm 0.5$ kn.
4. $H_{P_o} \pm 10$ ft.

2.4.4.4 SAFETY CONSIDERATIONS

Knowledge of and adherence to normal and emergency operating procedures, limitations, and range safety requirements.

2.4.5 RADAR ALTIMETER

A calibrated radar altimeter can be used to determine altimeter position errors present in level flight at low altitude. A level runway or surface is required to establish a reference altitude for the test. A sensitive pressure altimeter is placed at the runway elevation during test runs. To determine position errors, the radar height is added to the runway pressure altitude and compared with the aircraft altimeter system indication. The aircraft altimeter may be used as the reference altimeter if the aircraft is landed on the runway reference point. The service system altimeter reading before and after the test runs can be used to establish a base pressure altitude to which the radar height is added. This method is recommended only for rough approximation of gross altimeter position errors. Flight above the level surface is conducted low enough to provide accurate height information with the radar altimeter, but not so low the pressure field around the aircraft is affected by ground proximity. A good guideline is generally one wing span.

2.4.5.1 DATA REQUIRED

Surface pressure altitude, V_o, H_{P_o}, T_o, GW, radar altitude. Configuration.

2.4.5.2 TEST CRITERIA

1. Coordinated, wings level flight.
2. Constant aircraft heading.
3. Constant airspeed.

4. Constant radar altimeter altitude.

2.4.5.3 DATA REQUIREMENTS

1. Stabilized flight for 15 s.
2. $V_o \pm 0.5$ kn.
3. $H_{P_o} \pm 10$ ft.
4. Radar altimeter altitude ± 3 ft.

2.4.5.4 SAFETY CONSIDERATIONS

The safety considerations relevant to this test are similar to those discussed for the measured course method.

2.4.6 PACED

The use of a pace calibration system offers many advantages and is used frequently in obtaining position error calibrations. Specially outfitted and calibrated pace aircraft are used for this purpose. These pace aircraft generally have expensive, specially designed, separate pitot static systems which are extensively calibrated. These calibrations are kept current and periodically cross-checked on speed courses or with space positioning equipment. These pace calibration aircraft systems, when properly maintained and documented, offer the tester a method of obtaining accurate position error information in a short period of time.

The test method requires precise formation flying. The pace aircraft can be lead or trail, depending on individual preference. The lead aircraft establishes the test point by stabilizing on the desired data point. The trail aircraft must stay close enough to the lead aircraft to detect minor relative speed differences but far enough away to prevent pitot static interference between aircraft systems and/or compromising flight safety. Experience shows good results are obtained by flying one wing span abeam, placing the two aircraft out of each others pressure field. Formation flight is used so the airspeed and altimeter instruments of both aircraft are at the same elevation.

Leak checks of all pitot static systems including the pace aircraft systems, is required prior to and following calibration flights. Many pace systems provide two separate calibration systems to further guarantee data accuracy.

The pacer aircraft provides the test aircraft with calibrated pressure altitude, H_{P_c}, and calibrated airspeed, V_c, at each test point. This method of calibration takes less flight time and can cover any altitude and airspeed as long as the two aircraft are compatible.

2.4.6.1 DATA REQUIRED

V_o, H_{P_o}, T_o, $V_{o\ ref}$, $H_{P_o\ ref}$, $T_{o\ ref}$, GW, Configuration.

Note: Velocity and altitude data must be recorded for each system to be calibrated as well as pace aircraft data.

2.4.6.2 TEST CRITERIA

1. Coordinated, wings level flight.
2. Constant heading.
3. Constant airspeed.
4. Constant altitude (or stable rate of change during climbs and descents).
5. No relative motion between the test and pace aircraft.

2.4.6.3 DATA REQUIREMENTS

1. Stabilize 30 s prior to data collection.
2. Record data for 15 s.
3. $V_o \pm 0.5$ kn.
4. $H_{P_o} \pm 10$ ft.

2.4.6.4 SAFETY CONSIDERATIONS

All the hazards of close formation flying are present with the pace calibration method. To reduce this risk, practice and teamwork are required. In addition to the workload present with formation flying, considerable attention must be directed toward accurately reading altimeter, airspeed, and other instruments. These increased risk factors may be mitigated by providing automatic data collection and/or an additional crew member to collect the necessary information while the pilot directs his attention to safe formation flight. Again, careful planning, a good mission briefing, and professional execution significantly decrease the inherent risks associated with these tests. Minimum safe calibration altitudes provide sufficient allowance for ejection/bailout. Be alert to the potential for mid-air collision and have a flight break-up procedure established. Minimum visual flight conditions are established in the planning phase (such as, 3 miles visibility and 1000 foot cloud clearance), and inadvertent instrument meteorological condition break-up procedures are established. Pilots of both aircraft must be aware of the minimum control airspeeds for their particular aircraft in each of the various configurations required during the test and some margin provided for formation maneuvering. As in all flight tests, normal and emergency operating procedures and limitations must be known. Upon completion of tests, a method of flight break-up or section recovery is used. At no time during these tests should there be a question in anyone's mind who is responsible for providing safe separation distance between aircraft.

2.5 DATA REDUCTION

2.5.1 MEASURED COURSE

The following equations are used for measured course data reduction:

$$V_i = V_o + \Delta V_{ic} \qquad \text{(Eq 2.37)}$$

$$V_{G_1} = 3600 \left(\frac{D}{\Delta t_1} \right) \qquad \text{(Eq 2.61)}$$

$$V_{G_2} = 3600 \left(\frac{D}{\Delta t_2} \right) \qquad \text{(Eq 2.62)}$$

$$V_T = \frac{V_{G_1} + V_{G_2}}{2} \qquad \text{(Eq 2.63)}$$

$$\rho_a = \frac{P_a}{g_c \, R \, T_{a_{ref}} \ (^\circ K)} \qquad \text{(Eq 2.64)}$$

$$\sigma = \frac{\rho_a}{\rho_{ssl}} \qquad \text{(Eq 2.65)}$$

$$V_e = V_T \sqrt{\sigma} \qquad \text{(Eq 2.26)}$$

$$V_c = V_e - \Delta V_c \qquad \text{(Eq 2.66)}$$

$$\Delta V_{pos} = V_c - V_i \qquad \text{(Eq 2.39)}$$

$$M = \frac{V_T}{38.9678 \sqrt{T_{a_{ref}} \ (^\circ K)}} \qquad \text{(Eq 2.67)}$$

$$T_i = T_o + \Delta T_{ic} \qquad \text{(Eq 2.59)}$$

$$K_T = \left(\frac{T_i \ (^\circ K)}{T_{a_{ref}} \ (^\circ K)} - 1 \right) \frac{5}{M^2} \qquad \text{(Eq 2.60)}$$

$$q_c = P_{ssl} \left\{ \left[1 + 0.2 \left(\frac{V_c}{a_{ssl}} \right)^2 \right]^{3.5} - 1 \right\} \qquad \text{(Eq 2.68)}$$

$$q_{c_i} = P_{ssl} \left\{ \left[1 + 0.2 \left(\frac{V_i}{a_{ssl}} \right)^2 \right]^{3.5} - 1 \right\} \qquad \text{(Eq 2.69)}$$

$$\Delta P = q_c - q_{c_i} \qquad \text{(Eq 2.70)}$$

$$V_{i_W} = V_i \sqrt{\frac{W_{Std}}{W_{Test}}} \qquad \text{(Eq 2.71)}$$

Where:

a_{ssl}	Standard sea level speed of sound	661.483 kn
D	Course length	nmi
ΔP	Static pressure error	psf
Δt	Elapsed time	s
ΔT_{ic}	Temperature instrument correction	˚C
ΔV_c	Compressibility correction	kn
ΔV_{ic}	Airspeed instrument correction	kn
ΔV_{pos}	Airspeed position error	kn
g_c	Conversion constant	32.17 lb_m/slug
K_T	Temperature recovery factor	
M	Mach number	
P_a	Ambient pressure	psf
P_{ssl}	Standard sea level pressure	2116.217 psf
q_c	Impact pressure	psf
q_{c_i}	Indicated impact pressure	psf
R	Engineering gas constant for air	96.93 ft-lb_f/lb_m ˚K
ρ_a	Ambient air density	slug/ft^3
ρ_{ssl}	Standard sea level air density	0.0023769 slugs/ft^3
σ	Density ratio	
T_a	Ambient temperature	˚C
$T_{a\ ref}$	Reference ambient temperature	˚C or ˚K
T_i	Indicated temperature	˚C
T_o	Observed temperature	˚C
V_c	Calibrated airspeed	kn
V_e	Equivalent airspeed	kn
V_G	Ground speed	kn
V_i	Indicated airspeed	kn
V_{i_W}	Indicated airspeed corrected to standard weight	kn
V_o	Observed airspeed	kn
V_T	True airspeed.	kn
W_{Std}	Standard weight	lb
W_{Test}	Test weight	lb.

From the observed and reference airspeed, altitude, and temperature data, compute V_{i_W}, ΔV_{pos}, K_T, $\frac{\Delta P}{q_{c_i}}$ as follows:

Step	Parameter	Notation	Formula	Units	Remarks
1	Observed airspeed	V_o		kn	
2	Airspeed instrument correction	ΔV_{ic}		kn	Lab calibration
3	Indicated airspeed	V_i	Eq 2.37	kn	
4	Elapsed time$_1$	Δt_1		s	One direction
5	Elapsed time$_2$	Δt_2		s	Reciprocal direction
6	Course length	D		nmi	Known distance
7	Ground speed$_1$	V_{G_1}	Eq 2.61	kn	One direction
8	Ground speed$_2$	V_{G_2}	Eq 2.62	kn	Reciprocal direction
9	True airspeed	V_T	Eq 2.63	kn	
10	Reference pressure altitude	$H_{P\ ref}$		ft	Reference altimeter at ground course, corrections applied
11	Ambient pressure	P_a		psf	From Appendix VI or calculated using 10
12	Ambient air density	ρ_a	Eq 2.64	slug/ft^3	
13	Density ratio	σ	Eq 2.65		
14	Equivalent airspeed	V_e	Eq 2.26	kn	
15	Compressibility correction	ΔVc		kn	From Appendix VII, or assumed 0 for low altitude/airspeed
16	Calibrated airspeed	V_c	Eq 2.66	kn	
17	Airspeed position error	ΔV_{pos}	Eq 2.39	kn	
18	Mach number	M	Eq 2.67		
19	Observed temperature	T_o		˚C	
20	Temperature instrument correction	ΔT_{ic}		˚C	Lab calibration

21	Indicated temperature	T_i	Eq 2.59	˚C
22	Temperature recovery factor	K_T	Eq 2.60	
23	Impact pressure	q_c	Eq 2.68	psf
24	Indicated impact pressure	q_{c_i}	Eq 2.69	psf
25	Static pressure error	ΔP	Eq 2.70	psf
26	Indicated airspeed corrected to standard weight	V_{i_W}	Eq 2.71	kn

Plot indicated static pressure error coefficient, $\frac{\Delta P}{q_{c_i}}$, as a function of indicated airspeed corrected to standard weight, V_{i_W}, as shown in figure 2.9.

Plot airspeed position error, ΔV_{pos}, versus indicated airspeed corrected to standard weight, V_{i_W}, as shown in figure 2.13.

2.5.2 TRAILING SOURCE/PACED

The following equations are used in the trailing source/paced data reduction:

$$H_{P_i} = H_{P_o} + \Delta H_{P_{ic}} \tag{Eq 2.36}$$

$$H_{P_{i_{ref}}} = H_{P_{o_{ref}}} + \Delta H_{P_{ic_{ref}}} \tag{Eq 2.72}$$

$$H_{P_i} = \frac{T_{ssl}}{a_{ssl}} \left[1 - \left(\frac{P_s}{P_{ssl}} \right)^{\frac{1}{\left(\frac{g_{ssl}}{g_c\, a_{ssl}\, R} \right)}} \right] \tag{Eq 2.73}$$

$$H_{P_{i_{ref}}} = \frac{T_{ssl}}{a_{ssl}} \left[1 - \left(\frac{P_a}{P_{ssl}} \right)^{\frac{1}{\left(\frac{g_{ssl}}{g_c \; a_{ssl} \; R} \right)}} \right] \qquad \text{(Eq 2.74)}$$

$$\Delta P = P_s - P_a \qquad \text{(Eq 2.38)}$$

$$V_i = V_o + \Delta V_{ic} \qquad \text{(Eq 2.37)}$$

$$q_{c_i} = P_{ssl} \left\{ \left[1 + 0.2 \left(\frac{V_i}{a_{ssl}} \right)^2 \right]^{3.5} - 1 \right\} \qquad \text{(Eq 2.69)}$$

$$V_{i_W} = V_i \sqrt{\frac{W_{Std}}{W_{Test}}} \qquad \text{(Eq 2.71)}$$

Where:

a_{ssl}	Standard sea level speed of sound	661.483 kn
a_{ssl}	Standard sea level temperature lapse rate	0.0019812 ˚K/ft
$\Delta H_{P_{ic}}$	Altimeter instrument correction	ft
$\Delta H_{P_{ic\ ref}}$	Reference altimeter instrument correction	ft
ΔP	Static pressure error	psf
ΔV_{ic}	Airspeed instrument correction	kn
g_c	Conversion constant	32.17 lb_m/slug
g_{ssl}	Standard sea level gravitational acceleration	32.174049 ft/s^2
H_{P_i}	Indicated pressure altitude	ft
$H_{P_{i\ ref}}$	Reference indicated pressure altitude	ft
H_{P_o}	Observed pressure altitude	ft
$H_{P_{o\ ref}}$	Reference observed pressure altitude	ft
P_a	Ambient pressure	psf

P_s	Static pressure	psf
P_{ssl}	Standard sea level pressure altitude	2116.217 psf
q_{c_i}	Indicated impact pressure	psf
R	Engineering gas constant for air	96.93 ft-lb_f/lb_m-˚K
T_{ssl}	Standard sea level temperature	15˚C or 288.15˚K
V_i	Indicated airspeed	kn
V_{i_W}	Indicated airspeed corrected to standard weight	kn
V_o	Observed airspeed	kn
W_{Std}	Standard weight	lb
W_{Test}	Test weight	lb.

From the observed and reference airspeed and altitude data, compute V_{i_W} and $\frac{\Delta P}{q_{c_i}}$ as follows:

Step	Parameter	Notation	Formula	Units	Remarks
1	Observed pressure altitude	H_{P_o}		ft	
2	Altimeter instrument correction	$\Delta H_{P_{ic}}$		ft	Lab calibration
3	Indicated pressure altitude	H_{P_i}	Eq 2.36	ft	
4	Reference observed pressure altitude	$H_{P_{o\ ref}}$		ft	From reference source
5	Reference altimeter instrument correction	$\Delta H_{P_{ic\ ref}}$		ft	Lab calibration
6	Reference indicated pressure altitude	$H_{P_{i\ ref}}$	Eq 2.72	ft	
7	Static pressure	P_s	Eq 2.73	psf	Calculated, or from Appendix VI using H_{P_i}
8	Ambient pressure	P_a	Eq 2.74	psf	Calculated, or from Appendix VI using $H_{P_{i\ ref}}$
9	Static pressure error	ΔP	Eq 2.38	psf	
10	Observed airspeed	V_o		kn	

11	Airspeed instrument correction	ΔV_{ic}		kn	Lab calibration
12	Indicated airspeed	V_i	Eq 2.37	kn	
13	Indicated impact pressure	q_{c_i}	Eq 2.69	psf	
14	Standard weight	W_{Std}		lb	Specification weight or mission relevant weight
15	Test weight	W_{Test}		lb	
16	Indicated airspeed corrected to standard weight	V_{i_W}	Eq 2.71	kn	

Plot indicated static pressure error coefficient, $\frac{\Delta P}{q_{c_i}}$, as a function of indicated airspeed corrected to standard weight, V_{i_W}, as shown in figure 2.9.

2.5.3 TOWER FLY-BY

The following equations are used in the tower fly-by data reduction:

$$V_i = V_o + \Delta V_{ic} \qquad \text{(Eq 2.37)}$$

$$H_{P_i} = H_{P_o} + \Delta H_{P_{ic}} \qquad \text{(Eq 2.36)}$$

$$\Delta h = d \tan\theta \qquad \text{(Eq 2.75)}$$

$$\Delta h = L_{a/c} \frac{y}{x} \qquad \text{(Eq 2.76)}$$

$$H_{P_c} = H_{P_{c_{twr}}} + \Delta h \frac{T_{Std} \text{ (˚K)}}{T_{Test} \text{ (˚K)}} \qquad \text{(Eq 2.77)}$$

$$P_s = P_{ssl} \left(1 - 6.8755856 \times 10^{-6} H_{P_i}\right)^{5.255863} \qquad \text{(Eq 2.78)}$$

$$P_a = P_{ssl}\left(1 - 6.8755856 \times 10^{-6}\, H_{P_c}\right)^{5.255863} \qquad \text{(Eq 2.79)}$$

$$\Delta P = P_s - P_a \qquad \text{(Eq 2.38)}$$

$$q_{c_i} = P_{ssl}\left\{\left[1 + 0.2\left(\frac{V_i}{a_{ssl}}\right)^2\right]^{3.5} - 1\right\} \qquad \text{(Eq 2.69)}$$

$$V_{i_W} = V_i\sqrt{\frac{W_{Std}}{W_{Test}}} \qquad \text{(Eq 2.71)}$$

Where:

a_{ssl}	Standard sea level speed of sound	661.483 kn
d	Horizontal distance (Tower to aircraft)	ft
Δh	Aircraft height above tower	ft
$\Delta H_{P_{ic}}$	Altimeter instrument correction	ft
ΔP	Static pressure error	psf
ΔV_{ic}	Airspeed instrument correction	kn
H_{P_c}	Calibrated pressure altitude	ft
$H_{P_{c\,twr}}$	Tower calibrated pressure altitude	ft
H_{P_i}	Indicated pressure altitude	ft
H_{P_o}	Observed pressure altitude	ft
$L_{a/c}$	Length of aircraft	ft
P_a	Ambient pressure	psf
P_s	Static pressure	psf
P_{ssl}	Standard sea level pressure	2116.217 psf
θ	Angle (Aircraft above tower reference line)	deg
q_{c_i}	Indicated impact pressure	psf
T_{Std}	Standard temperature (At tower)	˚K
T_{Test}	Test temperature (At tower)	˚K
V_i	Indicated airspeed	kn
V_{i_W}	Indicated airspeed corrected to standard weight	kn

V_o	Observed airspeed	kn
W_{Std}	Standard weight	lb
W_{Test}	Test weight	lb
x	Scaled length of aircraft	
y	Scaled height of aircraft above tower.	

From the observed airspeed, pressure altitude, and tower data, compute $\frac{\Delta P}{q_{c_i}}$ and V_{i_W} as follows:

Step	Parameter	Notation	Formula	Units	Remarks
1	Observed airspeed	V_o		kn	
2	Airspeed instrument correction	ΔV_{ic}		kn	Lab calibration
3	Indicated airspeed	V_i	Eq 2.37	kn	
4	Observed pressure altitude	H_{P_o}		ft	
5	Altimeter instrument correction	$\Delta H_{P_{ic}}$		ft	Lab calibration
6	Indicated pressure altitude	H_{P_i}	Eq 2.36	ft	
7	Horizontal distance	d		ft	Tower to aircraft
8	Angle	θ		deg	Aircraft above tower reference line
9 a	Aircraft height above tower	Δh	Eq 2.75	ft	Tapeline altitude
9 b	Aircraft height above tower	Δh	Eq 2.76	ft	Tapeline altitude
10	Tower calibrated pressure altitude	$H_{P_{c\ twr}}$		ft	Tower calibrated altitude, corrections applied
11	Standard temperature	T_{Std}		˚K	Standard temperature at tower altitude
12	Test temperature	T_{Test}		˚K	Test temperature at tower attitude

13	Calibrated pressure altitude	H_{P_c}	Eq 2.77	ft	Tower calibrated pressure altitude plus temperature corrected tapeline altitude
14	Static pressure	P_s	Eq 2.78	psf	
15	Ambient pressure	P_a	Eq 2.79	psf	
16	Static pressure error	ΔP	Eq 2.38	psf	
17	Indicated impact pressure	q_{c_i}	Eq 2.69	psf	
18	Standard weight	W_{Std}		lb	
19	Test weight	W_{Test}		lb	
20	Indicated airspeed corrected to standard weight	V_{i_W}	Eq 2.71	kn	

Plot indicated static pressure error coefficient, $\frac{\Delta P}{q_{c_i}}$ as a function of indicated airspeed corrected to standard weight, V_{i_W}, as shown in figure 2.9.

2.5.4 TEMPERATURE RECOVERY FACTOR

Temperature recovery factor, K_T, can be determined as presented in Section 2.5.1 using Eq 2.60 when a reference ambient temperature is available or by using the following equations and method:

$$T_i = T_o + \Delta T_{ic} \tag{Eq 2.59}$$

$$\text{Curve slope} = K_T \frac{\gamma - 1}{\gamma} T_a = 0.2\, K_T\, T_a\ (^\circ K)\ \text{(High speed)} \tag{Eq 2.80}$$

$$\text{Curve slope} = K_T \frac{0.2\, T_a\ (^\circ K)}{a_{ssl}^2}\ \text{(Low speed)} \tag{Eq 2.81}$$

$$K_T = \frac{\text{slope}}{0.2\, T_a\ (^\circ K)}\ \text{(High speed)} \tag{Eq 2.82}$$

$$K_T = \frac{\text{slope } a_{ssl}^2}{0.2\ T_a\ (^\circ K)}\ \text{(Low speed)} \tag{Eq 2.83}$$

Where:

a_{ssl}	Standard sea level speed of sound	661.483 kn
ΔT_{ic}	Temperature instrument correction	˚C or ˚K
γ	Ratio of specific heats	
K_T	Temperature recovery factor	
T_a	Ambient temperature	˚C or ˚K.
T_i	Indicated temperature	˚C or ˚K
T_o	Observed temperature	˚C or ˚K.

Step	Parameter	Notation	Formula	Units	Remarks
1	Observed temperature	T_o		˚C or ˚K	
2	Temperature instrument correction	ΔT_{ic}		˚C or ˚K	Lab calibration
3	Indicated temperature	T_i	Eq 2.59	˚C or ˚K	
4	Mach	M			From tables for M versus V_c, H_{P_c}
5	Plot T_i versus M^2				High speed
5 a	Plot T_i versus V_c^2/δ				Low speed
6	Curve slope		Eq 2.80		High speed
6 a	Curve slope		Eq 2.81		Low speed
7	Ambient tempeature	T_a		˚K	T_a is the curve intercept
8	Temperature recovery factor	K_T	Eq 2.82		High speed
8 a	Temperature recovery factor	K_T	Eq 2.83		Low speed

2.6 DATA ANALYSIS

Once the indicated static pressure error coefficient, $\frac{\Delta P}{q_{c_i}}$ as a function of indicated airspeed corrected to standard weight, V_{i_W}, or M is determined by one of the test methods, airspeed position error, ΔV_{pos}, and altimeter position error, ΔH_{pos}, as a function of indicated airspeed corrected to standard weight, V_{i_W}, or M_i can be determined. The following equations are used:

$$P_s = P_{ssl}\left(1 - 6.8755856 \times 10^{-6} H_{P_i}\right)^{5.255863} \quad \text{(Eq 2.78)}$$

$$q_{c_i} = P_{ssl}\left\{\left[1 + 0.2\left(\frac{V_i}{a_{ssl}}\right)^2\right]^{3.5} - 1\right\} \quad \text{(Eq 2.69)}$$

$$M_i = \sqrt{\frac{2}{\gamma - 1}\left[\left(\frac{q_{c_i}}{P_s} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]} \quad \text{(Eq 2.84)}$$

$$V_{i_W} = V_i\sqrt{\frac{W_{Std}}{W_{Test}}} \quad \text{(Eq 2.71)}$$

$$\Delta P = \left(\frac{\Delta P}{q_{c_i}}\right) q_{c_i} \quad \text{(Eq 2.85)}$$

$$q_c = q_{c_i} + \Delta P \quad \text{(Eq 2.86)}$$

$$V_c = \sqrt{\frac{2}{\gamma - 1}\frac{P_{ssl}}{\rho_{ssl}}\left[\left(\frac{q_c}{P_{ssl}} + 1\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]} \quad \text{(Eq 2.19)}$$

$$\Delta V_{pos} = V_c - V_{i_W} \qquad \text{(Eq 2.87)}$$

$$P_a = P_s - \Delta P \qquad \text{(Eq 2.88)}$$

$$H_{P_c} = \frac{T_{ssl}}{a_{ssl}} \left[1 - \left(\frac{P_a}{P_{ssl}} \right)^{\frac{1}{\left(\frac{g_{ssl}}{g_c \, a_{ssl} \, R} \right)}} \right] \qquad \text{(Eq 2.89)}$$

$$\Delta H_{pos} = H_{P_c} - H_{P_i} \qquad \text{(Eq 2.40)}$$

Where:

a_{ssl}	Standard sea level speed of sound	661.483 kn
a_{ssl}	Standard sea level temperature lapse rate	0.0019812 ˚K/ft
ΔH_{pos}	Altimeter position error	ft
ΔP	Static pressure error	psf
ΔV_{pos}	Airspeed position error	kn
γ	Ratio of specific heats	
g_c	Conversion constant	32.17 lb_m/slug
g_{ssl}	Standard sea level gravitational acceleration	32.174049 ft/s^2
H_{P_c}	Calibrated pressure altitude	ft
H_{P_i}	Indicated pressure altitude	ft
M_i	Indicated Mach	
P_a	Ambient pressure	psf
P_s	Static pressure	psf
P_{ssl}	Standard sea level pressure	2116.217 psf
q_c	Impact pressure	psf
q_{c_i}	Indicated impact pressure	psf
R	Engineering gas constant for air	96.93 ft-lb_f/lb_m-˚K

ρ_{ssl}	Standard sea level air density	0.0023769 slug/ft^3
T_{ssl}	Standard sea level temperature	288.15 ˚K
V_c	Calibrated airspeed	kn
V_i	Indicated airspeed	kn
V_{i_W}	Indicated airspeed corrected to standard weight	kn
W_{Std}	Standard weight	lb
W_{Test}	Test weight	lb.

Airspeed position error, ΔV_{pos}, and altimeter position error, ΔH_{pos}, as a function of indicated Mach, M_i, or indicated airspeed, V_i, are determined from indicated static pressure error coefficient, $\frac{\Delta P}{q_{c_i}}$, as follows:

Step	Parameter	Notation	Formula	Units	Remarks
1	Indicated pressure altitude	H_{P_i}		ft	Select H_{P_i} of interest
2	Static pressure	P_s	Eq 2.78	psf	
3	Indicated airspeed	V_i		kn	Select V_i of interest
4	Indicated impact pressure	q_{c_i}	Eq 2.69	psf	
5	Indicated Mach	M_i	Eq 2.84		High speed case
5 a	Indicated airspeed corrected to standard weight	V_{i_W}	Eq 2.71	kn	Low speed case
6	Static pressure error coefficient	$\frac{\Delta P}{q_{c_i}}$			From figure 2.8, High speed case
6 a	Static pressure error coefficient	$\frac{\Delta P}{q_{c_i}}$			From figure 2.9, Low speed case
7	Static pressure error	ΔP	Eq 2.85	psf	
8	Impact pressure	q_c	Eq 2.86	psf	
9	Calibrated airspeed	V_c	Eq 2.19	kn	
10	Airspeed position error	ΔV_{pos}	Eq 2.87	kn	
11	Ambient pressure	P_a	Eq 2.88	psf	

12	Calibrated pressure altitude	H_{P_c}	Eq 2.89	ft	
13	Altimeter position error	ΔH_{pos}	Eq 2.40	ft	
14	Return to 3 and vary V_i				Repeat for a series of V_i at same H_{P_i}
15	Return to 1 and choose new H_{P_i}				Repeat for a series of V_i for new H_{P_i}

Plot ΔV_{pos} versus M_i (high speed case) or V_i (low speed case) and ΔH_{pos} versus M_i or V_i as shown in figures 2.13 and 2.14.

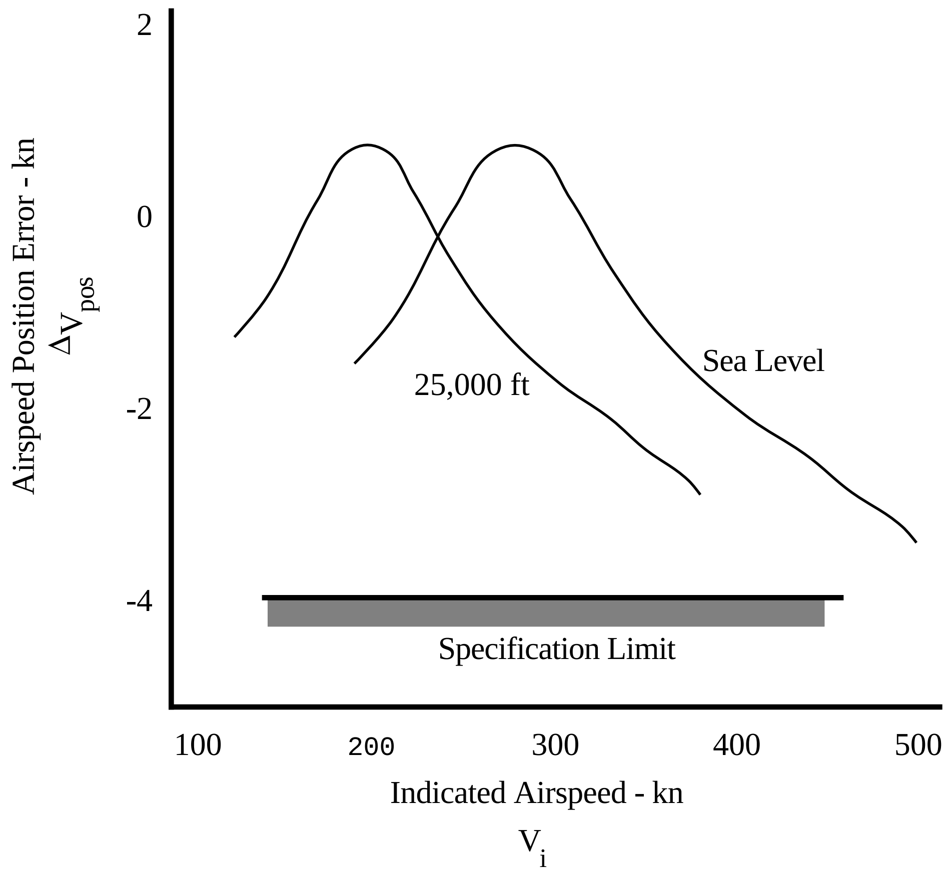


Figure 2.13
AIRSPEED POSITION ERROR

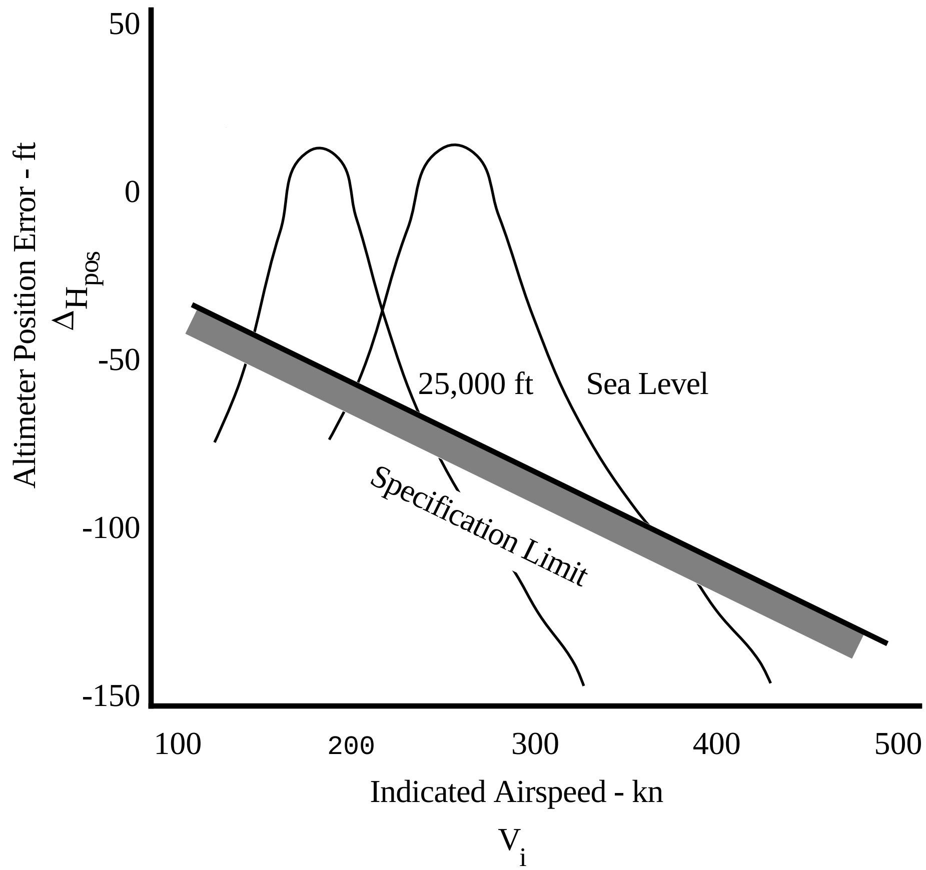


Figure 2.14
ALTIMETER POSITION ERROR

Analysis of pitot static system performance is done graphically. Superposition of the specification limits (Paragraph 2.8) on the data plots presented above identifies problems with system performance. Alternate configurations are evaluated when applicable. For each airspeed and altimeter calibration curve, be alert for discontinuities, large errors, and trends. Discontinuities, as in any primary flight instrument presentation, are not desirable.

Unbalanced pitot and static systems, large lag errors, indicator oscillations, and poor system performance at extreme flight conditions frequently are identified first with pilot qualitative comments. Unfavorable qualitative comments or unexplained difficulty in performing the airspeed calibration tasks can be an indication of pitot static system problems. Collection of quantitative data provides the opportunity to analyze qualitatively system performance.

2.7 MISSION SUITABILITY

The flight calibration of aircraft pitot and static systems is a very necessary and important test. A large measure of the accuracy of all performance and flying qualities evaluations depends on the validity of these calibration tests. Errors in the pitot static pressure systems may have serious implications considered with extreme speeds (high/low), maneuvering flight, and instrument flight rules (IFR) missions. The seriousness of the problem is compounded if pressure errors are transmitted to automatic flight control systems such as altitude retention systems or stability augmentation systems (SAS).

2.7.1 SCOPE OF TEST

The requirements of military specifications and the intended mission of the aircraft initially define the scope of the pitot and static system evaluation. The scope of the performance and flying qualities investigation may dictate an increase in the scope beyond that required above. This increase in scope may require flights at various external configurations and some testing may be required for calibration of the flight test instrumentation system alone. Generally, a flight check of the requirements of the specifications for maneuvering flight may be accomplished concurrently with the standard test.

The test pilot is responsible for observing and reporting undesirable characteristics in the pitot static system before they produce adverse results or degrade mission performance. When evaluating a pitot static system, the test team must consider the mission of the aircraft. Results of a pitot static system calibration attained at sea level may indicate satisfactory performance and specification compliance. However, operations at altitude may not be satisfactory.

2.8 SPECIFICATION COMPLIANCE

There are two military specifications which cover the requirements for pressure sensing systems in military aircraft. MIL-I-5072-1 covers all types of pitot static tube operated instrument systems (Figure 2.15) while MIL-I-6115A deals with instrument systems operated by a pitot tube and a flush static port (Figure 2.16). Both specifications describe in detail the requirements for construction and testing of these systems and state that one of the following methods of test will be used to determine "installation error" (position error):

1. "The speed course method."
2. "The suspended head or trailing tube method."
3. "The altimeter method."
4. "Pacer airplane method."

Federal Aviation Regulations (FAR) are directly applicable to ensure safe operation on the federal airways. These requirements are found in FAR Parts 27.1323 and 27.1335.

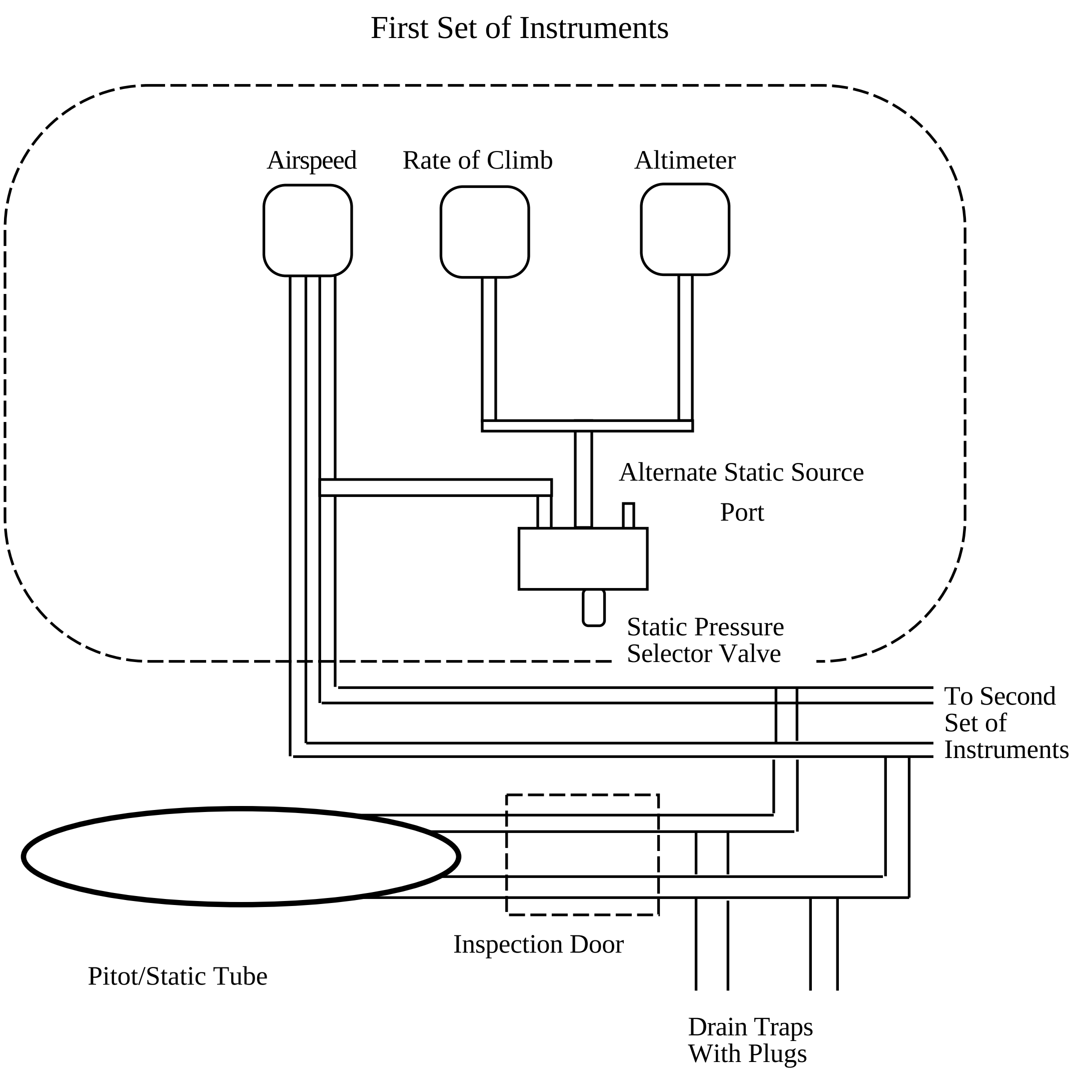


Figure 2.15
PITOT STATIC SYSTEM AS REFERRED TO IN MIL-I-5072-1

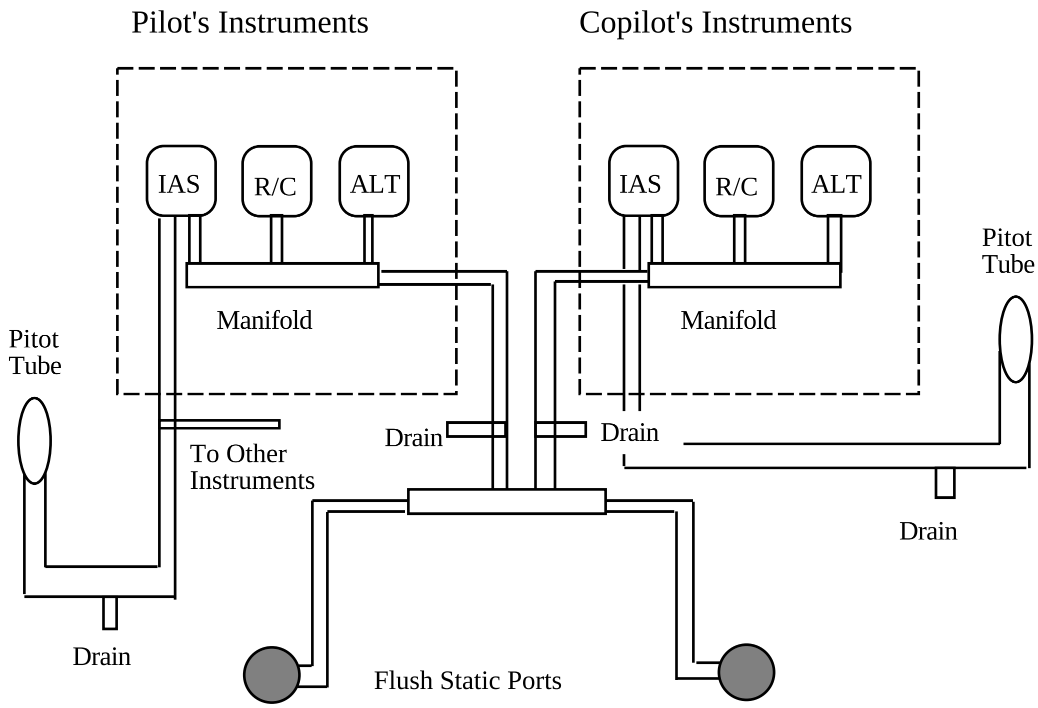


Figure 2.16

PITOT STATIC SYSTEM AS REFERRED TO IN MIL-I-6115A

2.8.1 TOLERANCES

Table 2.1, extracted from MIL-I-6115A, states the tolerance allowed on airspeed indicator and altimeter readings for five aircraft flight configurations, when corrected to standard sea level conditions.

Table 2.1

TOLERANCE ON AIRSPEED INDICATOR AND ALTIMETER READINGS

(Corrected to standard sea level condition 29.92 inHg and 15˚C)

Configuration	Speed Range	Gross Weight	Tolerances	
			Airspeed Indicator	Altimeter
Approach [1]	Stalling to 50 kn (58 mph) above stalling	Landing	± 4 kn ± 4.5 mph	25 ft per 100 KIAS
Approach [1]	Stalling to 50 kn (58 mph) above stalling	Normal	± 4 kn ± 4.5 mph	25 ft per 100 KIAS
Clean	Speed for maximum range to speed at normal rated power	Normal	± 1/2 % of indicated airspeed	25 ft per 100 KIAS
Clean	Stalling to maximum	Normal	± 4 kn ± 4.5 mph	25 ft per 100 KIAS
Clean	Stalling to maximum	Overload	± 4 kn ± 4.5 mph	25 ft per 100 KIAS
Dive	Maximum speed with brakes full open	Normal	± 6 kn ± 7 mph	50 ft per 100 KIAS

[1] The approach configuration shall include (in addition to wing flaps and landing gear down) such conditions as “canopy open”, “tail hook down”, etc., which may vary with or be peculiar to certain model airplanes.

2.8.2 MANEUVERS

There are four additional tests required by both military specifications which deal with the effect of maneuvers on pressure system operations. These tests are quite important and the descriptions of the test requirements are reprinted below as they appear in MIL-I-6115A.

2.8.2.1 PULLUP

“A rate of climb indicator shall be connected to the static pressure system of each pitot static tube (the pilot's and copilot's instruments may be used). The variation of static pressure during pullups from straight and level flight shall be determined at a safe altitude above the ground and at least three widely separated indicated airspeeds. During an abrupt “pullup” from level flight, the rate of climb indicator shall indicate “Up” without excessive hesitation and shall not indicate “Down” before it indicates “Up”.”

2.8.2.2 PUSHOVER

"A rate of climb indicator shall be connected to the static pressure system of each pitot static tube (the pilot's and copilot's instruments may be used). The variation of static pressure during pushover from straight and level flight shall be determined at a safe altitude above the ground and at least three widely separated indicated airspeeds. During an abrupt "pushover" from level flight, the rate of climb indicator shall indicate "Down" without excessive hesitation and shall not indicate "Up" before it indicates "Down"."

2.8.2.3 YAWING

"Sufficient maneuvering shall be done in flight to determine that the installation of the pitot static tube shall provide accurate static pressure to the flight instruments during yawing maneuvers of the airplane."

2.8.2.4 ROUGH AIR

"Sufficient maneuvering shall be done in flight to determine that the installation of the pitot static tube shall produce no objectionable instrument pointer oscillation in rough air. Pointer oscillation of the airspeed indicator shall not exceed 3 knots (4 mph)."

2.9 GLOSSARY

2.9.1 NOTATIONS

a	Speed of sound	kn
a	Temperature lapse rate	˚/ft
ARDC	Arnold Research and Development Center	
a_{ssl}	Standard sea level speed of sound	661.483 kn
a_{ssl}	Standard sea level temperature lapse rate	0.0019812 ˚K/ft
C_L	Lift coefficient	
D	Course length	nmi
d	Horizontal distance (Tower to aircraft)	ft
Δh	Aircraft height above tower	ft
$\Delta H_{P_{ic}}$	Altimeter instrument correction	ft

$\Delta H_{P_{ic\ ref}}$	Reference altimeter instrument correction	ft
ΔH_{pos}	Altimeter position error	ft
ΔM_{pos}	Mach position error	
ΔP	Static pressure error	psf
$\frac{\Delta P}{q_c}$	Static pressure error coefficient	
$\frac{\Delta P}{q_{c_i}}$	Indicated static pressure error coefficient	
Δt	Elapsed time	s
ΔT_{ic}	Temperature instrument correction	˚C
ΔV_c	Compressibility correction	kn
ΔV_{ic}	Airspeed instrument correction	kn
ΔV_{pos}	Airspeed position error	kn
e	Base of natural logarithm	
FAR	Federal Aviation Regulations	
g	Gravitational acceleration	ft/s^2
g_c	Conversion constant	32.17 lb_m/slug
g_{ssl}	Standard sea level gravitational acceleration	32.174049 ft/s^2
GW	Gross weight	lb
H	Geopotential	ft
h	Tapeline altitude	ft
H_P	Pressure altitude	ft
$H_{P\ ref}$	Reference pressure altitude	ft
H_{P_c}	Calibrated pressure altitude	ft
$H_{P_{c\ twr}}$	Tower calibrated pressure altitude	ft
H_{P_i}	Indicated pressure altitude	ft
$H_{P_{i\ ref}}$	Reference indicated pressure altitude	ft
H_{P_o}	Observed pressure altitude	ft
$H_{P_{o\ ref}}$	Reference observed pressure altitude	ft
IFR	Instrument flight rules	
K_T	Temperature recovery factor	
$L_{a/c}$	Length of aircraft	ft
M	Mach number	
M_i	Indicated Mach number	

M_{Test}	Test Mach number	
NACA	National Advisory Committee on Aeronautics	
n_z	Normal acceleration	g
P	Pressure	psf
P_a	Ambient pressure	psf
P_s	Static pressure	psf
P_{ssl}	Standard sea level pressure	2116.217 psf 29.9212 inHg
P_T	Total pressure	psf
P_T'	Total pressure at total pressure source	psf
q	Dynamic pressure	psf
q_c	Impact pressure	psf
q_{c_i}	Indicated impact pressure	psf
R	Engineering gas constant for air	96.93 ft-lb_f/lb_m-˚K
R_e	Reynold’s number	
SAS	Stability augmentation system	
T	Temperature	˚C or ˚K
T_a	Ambient temperature	˚C or ˚K
$T_{a\ ref}$	Reference ambient temperature	˚C or ˚K
T_i	Indicated temperature	˚C or ˚K
T_o	Observed temperature	˚C
T_{ssl}	Standard sea level temperature	15˚C or 288.15˚K
T_{Std}	Standard temperature (At tower)	˚K
T_T	Total temperature	˚K
T_{Test}	Test temperature (At tower)	˚K
USNTPS	U.S. Naval Test Pilot School	
V	Velocity	kn
V_c	Calibrated airspeed	kn
$V_{c_{Std}}$	Standard calibrated airspeed	kn
$V_{c_{Test}}$	Test calibrated airspeed	kn
V_{c_W}	Calibrated airspeed corrected to standard weight	kn
V_e	Equivalent airspeed	kn
$V_{e_{Std}}$	Standard equivalent airspeed	kn

$V_{e_{Test}}$	Test equivalent airspeed	kn
V_G	Ground speed	kn
V_i	Indicated airspeed	kn
$V_{i_{Test}}$	Test indicated airspeed	kn
V_{i_W}	Indicated airspeed corrected to standard weight	kn
V_o	Observed airspeed	kn
$V_{o\ ref}$	Reference observed airspeed	kn
V_T	True airspeed	kn
V_w	Wind velocity	kn
W	Weight	lb
W_{Std}	Standard weight	lb
W_{Test}	Test weight	lb
x	Scaled length of aircraft	
y	Scaled height of aircraft above tower	

2.9.2 GREEK SYMBOLS

α (alpha)	Angle of attack	deg
β (beta)	Sideslip angle	deg
δ (delta)	Pressure ratio	
γ (gamma)	Ratio of specific heats	
λ (lambda)	Lag error constant	
λ_s	Static pressure lag error constant	
λ_T	Total pressure lag error constant	
θ (theta)	Angle, Temperature ratio	deg
ρ (rho)	Air density	slug/ ft^3
ρ_a	Ambient air density	slug/ ft^3
ρ_{ssl}	Standard sea level air density	0.0023769 slug/ ft^3
σ (sigma)	Density ratio	

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