Test of a Probe Used to Sense Altitude Through Measurement of Pressure

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Abstract

A STATIC pressure probe was tested to determine the feasibility of using the probe as an integral part of a missile nose to sense missile altitude. Experiments were conducted at Mach 2.0 and at Mach 1.51. At Mach 2.0, the static pressure probe will perform within altitude specifications of 7600 m ± 600 m at angles of attack ranging from −8 to +8 deg. At Mach 2.0, within an angle of attack ranging from 0 to 6 deg, the probe will measure freestream static pressure within 4%; a 4% error in measurement is equivalent to an altitude error of 275 m. The missile nose shock will remain downstream of the probe pressure ports for flight Mach numbers above 1.5.

Contents

The ZEPPO rocket is an air-launched system designed to deliver an expendable pulse power communications jammer at long range. To achieve this objective, the system is loft-launched at angles up to 45 deg. At the proper point in the trajectory, a deployment sequence is initiated by the fuse. At the end of this sequence, the expendable jammer is suspended and operated from a parachute. The original deployment system consisted of: 1) a timer fuse in the nose, 2) an expulsion charge, 3) a drogue parachute, 4) a main parachute, and 5) an expendable electronic jammer.

The optimum point for deployment initiation is 7600 m ± 600 m. Deployment at this altitude allows maximum payload time aloft and, therefore, maximum operation time. System testing has demonstrated that there is difficulty in attaining the optimum deployment altitude with the original timer fuse. To remedy this situation, a fusing system initiated by a pressure-sensitive switch was selected. A static pressure probe, which was based on a design by Pinckney,[1] is extended from the missile nose. This synoptic discusses the analysis and testing of the probe.

Static Pressure Probe Design

Pinckney[1] discusses the design of a short static pressure probe that is relatively insensitive to angle of attack. The general probe design, which is illustrated in Fig. 1, consists of a conical nose followed by a parabolic tangent ogive fairing the nose into a second, truncated cone. The second cone mates with the cylindrical afterbody of the probe.

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Static Pressure Distribution

Using the supersonic small perturbation theory developed by Gawain and Schonberger,[2] the static pressure distribution along the probe was calculated for $M_\infty = 1.4$, 1.6, 1.8, and 2.0. The corresponding locations for the local pressure to be equal to the freestream pressure, P_∞, are $X/D = 1.93$, 2.02, 2.10, and 2.18, respectively. D is probe diameter; see Fig. 1. Hence, as Mach number increases from 1.4 to 2.0, the location where P/P_∞ equals unity shifts aft by $\Delta X/D = 0.25$. Due to the range of flight Mach numbers, a design compromise was made that minimizes the pressure error over the range of M_∞, rather than for one specific M_∞. A graph of

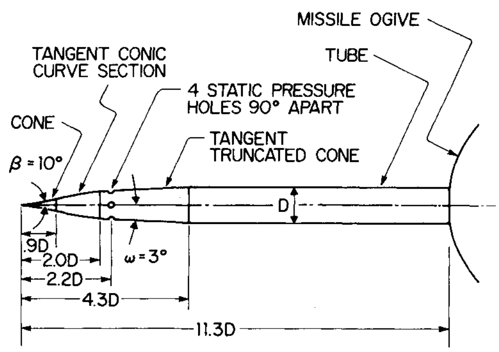

Fig. 1 Geometry of static pressure probe, adapted from Pinckney.[1]

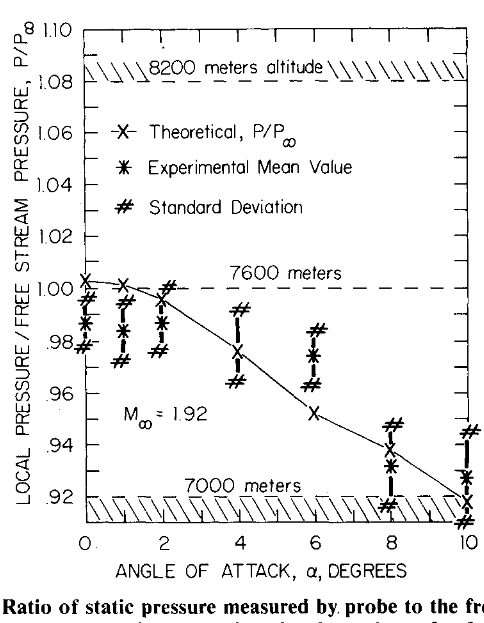

Fig. 2 Ratio of static pressure measured by probe to the freestream pressure, P/P_∞, as a function of angle of attack, α, for freestream Mach number $M_\infty = 1.92$. Activation altitudes corresponding to P/P_∞ values are shown. Cross-hatch indicates boundary of allowable values.

P/P_∞ which appears in Tillotson,[3] clearly shows that minimum difference in pressure occurs at $X/D \simeq 2.2$. The difference in P/P_∞ due to M_∞ is less than four %.

Error in Altitude Due to Error in Pressure Measurement

Pressure as a function of altitude is given by

$$P = P_0 e^{-\xi h} \tag{1}$$

where $\xi = (7.6 \text{ km})^{-1}$ and P_0 is the pressure at sea level. Logarithmic differentiation of Eq. (1) yields

$$dh = -\frac{1}{\xi} \cdot \frac{dP}{P} \tag{2}$$

With $1/\xi = 7.62$ km and $dP/P = 0.04$, the value of dh is 300 m (984 f). This error is well within the tolerance specified for probe performance.

Mach 2.0 Wind Tunnel Tests

Wind tunnel tests were conducted in the Naval Postgraduate School 10×10 cm (4×4 in.) supersonic wind tunnel using the existing Mach 2.0 nozzle blocks. (The Mach number of 2 is nominal; the actual Mach number is 1.92.) Utilizing a one-fifth scale model and regulating wind tunnel stagnation pressure, an actual Reynolds number of $Re = 1.3 \times 10^5$ was attainable, which compares favorably with the Reynolds number for the full-scale probe at 7600 m (viz., $Re = 4.3 \times 10^5$). The reference length for the calculation of the Reynolds number for the model is 0.020 m. Twenty-three wind tunnel tests were conducted at $M_\infty = 1.92$ with $\alpha = 0$, 1, 2, 4, 6, 8, and 10 deg.

Mach 1.51 Wind Tunnel Tests

Results from this series of tests are discussed in Ref. 4.

Mach 1.4 Wind Tunnel Tests

Tests were attempted utilizing the existing Mach 1.4 nozzle blocks. Supersonic flow could not be established in the test section. For Mach 1.4, the wind tunnel is extremely sensitive to blockage.

Model Design

A one-fifth scale model of the static pressure probe and missile nose was used; see Fig. 1. Since a complete model could not be mounted in the tunnel, the missile nose was truncated at 40 % of the nose length. Maximum model size was dictated by maximum allowed wind tunnel blockage (approximately 6.4%) at Mach 1.4. The allowed wind tunnel blockage limited the combined frontal area of the model and mounting system.

The ogive,which represents the missile nose, is 30.5 mm from the missile nose. The four probe pressure holes, which are 90 deg apart and 0.34 mm in diam., are located 3.37 mm from the probe conical nose tip. The four pressure ports enter a common manifold. The parabolic tangent ogive and second truncated cone start at 1.33 mm and 3.11 mm, respectively, from the probe nose tip.

Experimental Results

Figure 2 is a plot of the data obtained in the series of experiments. The solid curve is the value P/P_∞ predicted using the computer code of Ref. 2. The mean value of the measured pressure is shown as an asterisk, and the ends of the error bar for a data point are denoted by the symbol #. The error bar is one standard deviation above and below the mean value.

Three altitudes are drawn in Fig. 2 at pressures corresponding to 7000, 7600, and 8200 m. If the pressure ratio were the value shown on the ordinate, the fuse would activate at the altitudes shown. The altitude interval from 7000 to 8200m is an acceptable range. For an angle of attack of 6 deg or less, the probe yields pressure ratios well within the allowable error band. At an angle of attack of 8 deg, a small probability exists that the fuse will activate at a low altitude.

Conclusions

The following conclusions result from the static probe study.

1) The static pressure probe of Ref. 1, with $\beta = 10$ deg and $\omega = 3$ deg, will, as an integral part of the missile nose, perform within altitude specifications of 7600 ± 600 m at Mach 2.0 at angles of attack from −8 to +8 deg.

2) At Mach 2.0, within an angle of attack range from 0 to 6 deg, the probe will measure freestream static pressure within 4%. A 4% error in measurement is equivalent to an altitude error of 275 m.

3) The shock wave due to the missile ogive nose remains downstream of the probe pressure ports for flight Mach numbers above 1.5.

4) Since the theory predicted probe performance within 2.6% at Mach 2.0, and since the theory predicted satisfactory performance at Mach 1.4, the static pressure probe should perform within specifications at Mach 1.4.

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