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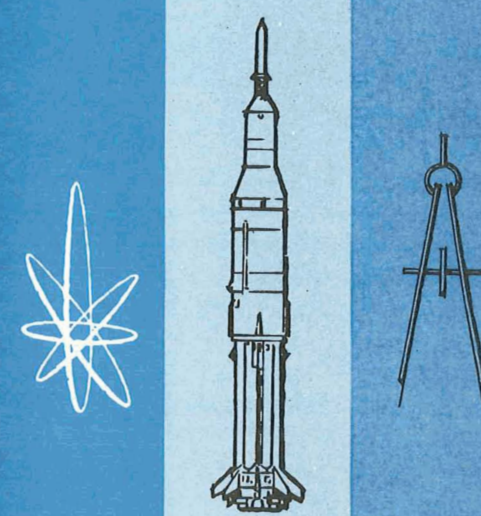
CORRECTION OF FOUR-PERCENT SATURN V MODEL PROTUBERANCE TEST DATA



SPACE DIVISION CHRYSLER CORPORATION

HUNTSVILLE OPERATIONS

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CORRECTION OF FOUR-PERCENT SATURN V
MODEL PROTUBERANCE TEST DATA

By

George F. McCanless, Jr.

April 21, 1970

FOREWORD

This report was prepared by the Aero-Space Mechanics Branch, Structures and Mechanics Engineering Department, Huntsville Operations, Chrysler Corporation. The work was authorized by NASA Contract NAS8-30517 which was issued by the Unsteady Aerodynamics Branch, Aerodynamics Division, Aero-Astrodynamics Laboratory, Marshall Space Flight Center in Huntsville, Alabama. The purpose of this study is to correct Saturn V wind tunnel data and to determine methods of extrapolating this data to full scale.

ABSTRACT

Tests measuring wind tunnel background pressure fluctuations were conducted in the AEDC 16 ft transonic wind tunnel. The objective of these tests was to obtain data that can be used to correct Saturn V test data obtained in this facility. The model and instrumentation were fabricated and installed by MSFC personnel. The data reduction is being performed by Chrysler personnel. This report describes the preliminary data reduction which is confined to the amplitudes of the pressure fluctions. Future data reduction will determine the frequency characteristics of these fluctuations.

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1.0 INTRODUCTION

Prior to this decade the unsteady forces that act on aircraft, missiles, and space vehicles were generally ignored. This was permissible because of the large safety factors that were previously in use. These large safety factors were dictated by the general lack of refinement in the methods of aerospace engineering. However, ever increasing precision is being required in the methods of aerospace engineering. This is caused by the following factors:

- The large cost of the aircraft, missiles, and space vehicles that are built today precludes the earlier procedures of trial and error.
- The competitive nature of the aerospace field requires improvements in performance. This in turn requires that all variables in the system be optimized.
- Commercial transports now carry large numbers of people and space vehicles are now manned. These considerations demand steady improvements in safety.

The increases in design precision imply that the unsteady aerodynamic forces must be established early in the development program. This establishes a basis for designing the structure so that it will withstand the dynamic loading and yet not be excessively heavy. The need for unsteady aerodynamic data is accented by the fact that the structures that are currently being built are larger than those that were fabricated several years ago. The skin of these structures is about the same thickness as those used with the smaller aircraft, missiles, and space vehicles. This combination results in flexible structures that are highly responsive to unsteady aerodynamic excitations.

The unsteady aerodynamic phenomena that must be determined in order to design aircraft, missiles, and space vehicles are briefly described below:

- <u>Boundary Layer Turbulence</u>

  This phenomena can cause fatigue failures and can saturate control sensors. The noise generated by turbulence, especially jet turbulence, can cause discomfort or injury to man. Reducing the turbulent flow area by extending the laminar flow regime will also reduce drag.

- <u>Panel Flutter</u>

  Panel flutter can result in both direct structural failure and fatigue failures.

- <u>Wing or Fin Flutter</u>

  This type of flutter can also result in direct structural failure and fatigue failure.

- High Angle-of-Attack Buffeting

  This mode of buffeting can result in structural failure.

- Transonic Buffeting

  Transonic buffeting can cause any of the problems mentioned above.

- Ground Wind Oscillations

  Ground winds cause structures to oscillate. This problem is particularly acute with missiles and space vehicles which can be blown over.

The unsteady aerodynamic loads must be established by experimental testing, because theoretical procedures have not been perfected that are adequate for establishing the vehicle design requirements. Wind tunnel tests have been found to be generally the most satisfactory means of determining the fluctuating pressure environment. However, virtually all wind tunnels were designed before the time that the need for unsteady aerodynamic testing was recognized. Thus, little or no attempt was made to minimize the background pressure fluctuations that are inherent in fluid flow processes.

Several investigators have measured fluctuations in subsonic, transonic, and supersonic wind tunnels. Mahinder S. Uberoi conducted a study in a subsonic wind tunnel on the behavior of turbulence as it passes from the stilling chamber through the throat of the tunnel. This study is described in Reference 1. The wind tunnel had the blower downstream of the test section and used an open return. Virtually all of the data was obtained with hot wire anemometers. Based on his study, Uberoi states, "Turbulent velocity measurements show that in absolute magnitudes, the longitudinal component decreases and the lateral component increases as the flow accelerates through the contraction." He found that turbulence from the blower was being propagated through the open loop and into the test section. This was eliminated, or stabilized, by installing a honeycomb in the stilling chamber. It was also found that sound waves generated by the blower were being radiated up stream to the test section. This conclusion was based on the fact that a correlation coefficient of 0.9 was computed from the measurements of velocity fluctuations across the test section. Uberoi extends his results by stating, "For supersonic nozzles, elementary considerations show that the effects of increase in the mean speed and decrease in density are both beneficial in reducing the flow irregularities."

Tests were conducted by Mark V. Morkovin in the continuous supersonic wind tunnel at Johns Hopkins University. These tests are described in Reference 2. All tests were at Mach 1.76. He considers three fluctuations modes:

- "Sound mode (variation of pressure, density, and temperature).

- Entropy mode (variation of entropy, density, and temperatures).

- Vorticity mode (Variation of the sinusoidal component of the velocity field which is known as turbulence at incompressible speeds)."

He states that, "The entropy and vorticity modes are essentially convected along streamlines so that in a supersonic tunnel they must be traceable... to conditions in the stilling chamber." He further states that, "The sound disturbances can travel across streamlines so that they come from the settling chamber and from the boundaries of the test section."

Morkovin then classified the sound fluctuations originating at the wall into four types:

a. "Radiation from nascent turbulence...

b. Radiation from developed turbulent boundary layers.

c. Diffraction and scattering of otherwise steady pressure gradients and shock waves (as generated by nozzle contours unintended waviness or roughness, models, supports, etc.) through the turbulent boundary layer.

d. Radiation from unsteady wall vibrations caused by pressure fluctuations in the boundary layer or by the loads on the diffuser associated with the unsteadiness of the terminal shock wave."

From hot wire anomometer data he concluded that the ratio of the rms pressure fluctuations to the free stream static pressure is 0.2 to 0.4. This converts to a rms pressure coefficient of 0.00092 to 0.00185. "For a given wall geometry this sound of category (c) is likely to decrease with Mach number [while that of category (b) may possibly increase]." He concludes that the fluctuations are not convected from the stilling chamber. For this to be significant, he states that the fluctuations in the test section would have to be 114 db, whereas in normal operation the noise level in the stilling chamber of this continuous flow tunnel was in the range of 70 to 80 db. However, during the starting process (before sonic conditions), large sound levels were present in the stilling chamber which originated in the diffuser. He also concludes that magnitude of sound category (d) is unlikely to reach the intensity of the sound of category (c), i.e., 120-130 db.

J. S. Murphy (References 3 and 4) conducted an early study of the pressure fluctuations in the Douglas Trisonic One-Foot Tunnel. This is a blowdown facility with a Mach number range of 0.2 to 1.8. Tests were conducted over this Mach number range using microphone, hot wire anemometers, and strain gauge dynamic pressure transducers. It was concluded that the primary cause of the pressure fluctuations in the stilling chamber is a high-intensity sound field that originated in the neighborhood of the control valve. A sound-absorbent muffler was designed, built, and installed in the stilling chamber. At Mach 1.0 it reduced the value of the rms pressure coefficient in the test section from 0.058 to 0.022. The reduction in the ratio of the stilling chamber rms static pressure to stagnation pressure is from 0.025 to 0.005. This proved that a large portion of the fluctuations in the test section are caused by fluctuations either in the entrance to the stilling chamber or upstream of it.

John Laufer conducted a study in 1961 of the fluctuation levels in the 18 x 20 in. supersonic wind tunnel at the Jet Propulsion Laboratory. Reference 5 describes this study. This is a closed circuit, continuous wind tunnel with solid walls in the test section. The tests were conducted over the Mach number range of 1.6 to 5.0. Virtually all data was obtained with a hot wire anemometer. Velocity fluctuations measured in this manner were used to compute the test section static pressure fluctuations. This resulted in a computed value of the rms pressure coefficient of 0.0009 at a Mach number of 1.6. It was concluded that the source of the fluctuations is the turbulent boundary layer on the test section walls. Laufer states that the Reynolds number of the tunnel was lowered to the point that the boundary layer on the walls was laminar. This caused the fluctuation level in the test section to drop by an order of magnitude.

P. A. Irani and K. Sridnor Iya (Reference 6) surveyed the general problem area of aerodynamic noise. Their objective was to establish a rational basis for reducing the noise level in the trisonic wind tunnel at the National Aeronautical Laboratory, Bangalore, India. Of special interest is their description of the noise reduction program conducted by R. Westley of the National Aeronautical Establishment, Ottawa, Canada. Westley's objective was to reduce the fluctuation level in the NAE 5 x 5 ft trisonic wind tunnel. This is a blowdown tunnel with a Mach number range of 0.2 to 4.5. A scale model of this tunnel was built with a 5 x 5 in. test section. Stilling chamber pressure fluctuations measuring ± 0.023 of the settling chamber static pressure were obtained. This converts to an approximate value of 0.016 for the ratio of the rms static pressure to the stagnation pressure. Dynamic pressure transducers were used to measure the test section pressure fluctuations. These measurements resulted in a rms pressure coefficient of 0.038. External micorphone measurements were made above the wind tunnel. The largest noise levels were measured near the control valve and near the diffuser shock wave. At a Mach number of 1.17, noise levels of approximately 110 db were measured at both locations.

J. S. Murphy, D. A. Bies, and W. W. Speaker (Reference 7) conducted studies of boundary layer noise in the previously described Douglas Trisonic One-Foot tunnel. A 26,000 cu ft tank was connected parallel to the 8,000 cu ft reservoir of the tunnel. This facilitated the operation of the tunnel by maintaining the reservoir pressure at the tunnel stagnation pressure. There was no choked flow through a control valve with its associated stilling chamber fluctuations. The stagnation pressure reduces slightly during tunnel operation. However, satisfactory test conditions of 15 sec were obtained. The authors state, "The modification of the blowdown wind tunnel, enabling operation with stagnation pressure equal to reservoir pressure, produced a facility which has satisfactory characteristics (low background noise level) to enable boundary layer noise to be measured over the Mach range $0.4 \leq M \leq 3.5$ in a single experimental arrangement." Unfortunately, no comparative data is given to show how much (if any) reduction is achieved in the pressure fluctuations in the test section.

Hartmut Bossel conducted a dynamic investigation, which is described in Reference 8, of the Hess 6 in. supersonic wind tunnel. This is a continuous, closed-cycle tunnel with a Mach number range of 1.8 to 2.8. Tests were conducted with dynamic pressure transducers in the test section. He found that in the test section "The mean fluctuation from the mean wall static pressure was about 0.3% at Re = 4 x 10^5/in." at Mach number 2.4. This corresponds to a rms pressure coefficient of approximately 0.00075. Spectrum analysis showed peaks at about 260 and 10,000 cps. Observations were made in the stilling chamber of the flow following the last screen. Here erratic jumps occurred in the flow direction of 15 degrees with a frequency of about 5 cps.

Modifications were made in the stilling chamber. The final screen was removed and a 3 in. thick honey comb screen was installed. The stilling chamber flow channel surfaces were smoothed. The static pressure fluctuations were no longer measurable. Hot wire measurements showed that the large low frequency disturbances disappeared. An additional modification was made to the tunnel. It consisted of removing a portion of the boundary layer by suction prior to the nozzle throat. Hot wire anemometer tests were then conducted. The suction was found to be beneficial at high Reynolds numbers and detrimental at low Reynolds numbers.

A sidewall calibration of the AEDC 16 ft transonic tunnel was conducted by the Martin Company during a 6 percent Titan III B Agena model test. This calibration is described in Reference 9. Two pressure microphones were located on the wind tunnel sidewalls. Both microphones were located upstream of the porous tunnel walls. The data from these microphones were presented as sound pressure level versus frequency. In addition, C. D. Riddle conducted cone calibration tests in the AEDC 16 ft transonic and 16 ft supersonic wind tunnels. A description of this calibration is given in Reference 10. The tests in the transonic tunnel covered the Mach number range of 0.6 to 1.4; supersonic tunnel data encompassed the Mach number range of 1.8 to 3.1. The calibration device consisted of a 10° apex angle cone. Two dynamic sensors (a transducer and a microphone) were located longitudinally adjacent to each other at three body stations. The rms pressure coefficient from these tests reached a maximum of 0.028 at Mach number 0.78. Below Mach number 0.70 and above Mach number 0.85 the rms pressure coefficient is less than 0.016.

Data from both of these tests were reduced in terms of power spectral densities by the authors. Reference 11 gives this reduced data. Examination of these power spectral densities reveals a large concentration of fluctuations at frequencies between 500 and 600 cps in the low transonic Mach number regime. This fluctuation concentration reaches a maximum at Mach number 0.75 and decreases both below and above this Mach number. Above the sonic Mach number the fluctuation concentration essentially disappears. The frequency composition remains virtually constant with varying Mach number. Further examination of the reduced data shows that a concentration of fluctuations occurs between approximately 1800 and 2500 cps. Both cone and sidewall calibration data indicate that this concentration of fluctuations is a function of a Mach number between Mach number 0.75 and Mach number 1.30.

Tests were conducted by J. A. B. Wills in a low speed (160 ft/sec max), open circuit, 15 x 10 in. cross section wind tunnel. He describes these tests in Reference 12. He theorized that, "The combination of rapidly-growing boundary layers and comparatively high speeds (in the sonic diffuser) produces intense low-frequency fluctuation which propagate back through the working section as sound waves." He operated the tunnel without the diffuser and observed that the low frequency fluctuations in the test section were greatly reduced. This substantiated his hypothesis.

J. M. Christophe and J. M. Loniewski conducted tests in the transonic test section of the S-2 wind tunnel of the Modane (France) ONERA test center. Reference 13 includes the results of these tests. This facility is a closed circuit, continuous wind tunnel with a 6 x 6 ft cross section. The transonic circuit covers a Mach number range of 0.2 to 1.3. Fluctuations occur in the test section between Mach number 0.62 and 0.91. The frequency of these fluctuations decreased as the Mach number increased from 0.56 to 0.8.

The objective of these tests was to establish the source of the 500-700 cps acoustic perturbations. It was found that there were no fluctuations in the stilling chamber. Changing the second throat had no effect on the test section fluctuation, and altering the plenum chamber volume had only a secondary effect on the fluctuations. The investigators found that, "Using a tape to cover completely the perforations of the upper and lower walls led to the elimination of the perturbating frequencies as evidenced simultaneously by the analyzer and by the change in noise from the wind tunnel." The fluctuations were not influenced by variations in the permeability (or porosity) of the lateral walls. The authors conclude that the fluctuations in this tunnel can be eliminated by setting the upper and lower walls between 0 and 0.05% permeability.

Chrysler Huntsville Operations conducted tests in the Marshall Space Flight Center 14 in. trisonic wind tunnel. These tests are described in Reference 14. This is a blowdown tunnel with interchangeable transonic and supersonic tests sections. The transonic test section has a Mach number range of 0.2 to 2.5, and the supersonic test section operates from Mach 2.75 to 5.0. The facility consists of a compressor, high pressure storage tank, control valve, stilling chamber with a heat exchanger test section, diffuser, and atmospheric exhaust tower. The transonic test section plenum is normally connected to vacuum tanks.

The experimental program conducted by Chrysler was unique in two ways. It made use of extensive instrumentation, including transducers located just down stream of the control valve, down stream of the stilling chamber, on the test section wall, on the test section calibration model, in the diffuser, atmospheric exhaust tower, and in the vacuum tanks. An accelerometer was located on the porous walls. The second unique factor was based on the fact that the Marshall Space Flight Center 14 x 14 in. trisonic wind tunnel can be operated in various configurations. This flexibility of operating modes is ideally suited to identifying the sources of test section fluctuations. Tests were conducted with both the transonic and supersonic test sections. These results give a comparison of their influence on fluctuation levels. Tests were conducted with the transonic test section using solid as well as porous walls. Tests were also conducted using various porosity settings. This indicated the effect of porosity on test section fluctuations. Tests were conducted with the high pressure system and the valve disconnected from the stilling chamber. In this configuration the tunnel was driven by the vacuum tanks. This gave an indication of the effects of the valve flow and its associated upstream turbulence on the test section fluctuations. Tests were conducted with the stilling chamber removed from the facility. Again the tunnel was powered by the vacuum tanks. This showed the effects of the stilling chamber on the fluctuations. These combinations of wind tunnel components show the influence of porous walls, upstream turbulence, and the diffuser flow on the test section fluctuations.

The results of this experimental program indicate that the largest fluctuations occur in the transonic regime. The largest component of test section noise consist of a fluctuation concentration that varies from 6,000 to 12,000 cps, depending on the particular operating conditions of the wind tunnel. This fluctuation is generated by the porous walls. The upstream turbulence apparently has a strong influence on the generation of these fluctuations. This 6,000 to 12,000 cps fluctuations has its counter part in the 16 ft transonic wind tunnel at AEDC and the 5 ft trisonic wind tunnel at ONERA in France. The amplitude of the overall fluctuation level in the MSFC 14 in. transonic test section compares favorably with that measured in the AEDC wind tunnel and with that in the Douglas 1 ft trisonic tunnel and in the NAE 5 ft transonic tunnel.

2.0 WIND TUNNEL TEST FACILITY AND SCHEDULE

2.1 TEST FACILITY

The AEDC 16 ft transonic wind tunnel can operate from Mach numbers of 0.5 to 1.6. The Mach number is continuously variable over this range. This tunnel is equipped with fixed porosity walls. The porosity is 6.0% of the wall area. Removable plates are provided for viewing of the model under test conditions. Stagnation pressures up to 28 psi can be achieved under most test conditions. This will provide Reynolds numbers of up to 8.4 million under most test conditions. Additional information concerning the AEDC 16 ft transonic tunnel can be found in Reference 15.

2.2 TEST SCHEDULE

The schedule of the test is shown in Figure 1. The AEDC 16 ft tunnel test schedule is organized to yield as much comparative data between the AEDC 16 ft tunnel and the MSFC 14 in. tunnel as possible. Wherever possible, both unit Reynolds number (Reynolds number per foot) and local Reynolds number were matched between the AEDC and MSFC test schedules. Some test conditions are included that match those used by other investigators who have conducted acoustic tests in this tunnel. Test points that match some of this test data were also included. The test is also arranged to provide information concerning the interrelationship between pressure fluctuations in various sections of the wind tunnel. The effects of Mach number, stagnation pressure, stagnation temperature, tunnel diffuser, tunnel compressor, and scaling on the background pressure fluctuation were investigated.

Run No.	Mach No.	Stagnation Pres.(psi)	Stagnation Temp.(°R)	Plenum Suction	Purpose	Similar Test In MSFC Tunnel
1	0.60	8.23	585	Standard	Match Local	Yes
2	0.65				Reynolds No.	
3	0.70					
4	0.75					
5	0.80					
6	0.90					
7	1.00					
8	1.05					
9	1.10					
10	1.30					
11	1.20	11.10			Match AEDC	
12	1.10				Cone Tests	
13	1.00					
14	0.90					
15	0.80					
16	0.75					
17	0.60					
18	0.60	14.70			Pressure	Yes
19	0.65				Effects	
20	0.70					Yes
21	0.75					
22	0.80					
23	0.85					Yes
24	0.90					
25	0.95					
26	0.80-1.05-0.52				Mach Sweep	
27	0.75	22.00	560		Temperature	
28	0.90				Effects	

FIGURE 1. AEDC 16 FT. WIND TUNNEL TEST SCHEDULE

Run No.	Mach No.	Stagnation Pres.(psi)	Stagnation Temp. (°R)	Plenum Suction	Purpose	Similar Test In MSFC Tunnel
29	1.10	22.00	560	Standard	Temperature	
30	1.10		585		Effects	
31	1.10					
32	1.10					
33	1.10				Match Unit	Yes
34	1.05				Reynolds No.	Yes
35	1.00					Yes
36	0.95					Yes
37	0.90					Yes
38	0.85					Yes
39	0.80					Yes
40	0.75					
41	0.70					
42	0.65					
43	0.60					Yes
44	0.75			Variations	Plenum Suction	
45	0.75				Effects	
46	0.75					
47	0.90					
48	0.90					
49	0.90					
50	0.80	28.00		Standard	Pressure	Yes
51	0.75				Effects	
52	0.70					
53	0.60					Yes
54	0.75	22.00	610		Temperature	
55	0.90				Effects	
56	1.10					

FIGURE 1. AEDC 16 FT. WIND TUNNEL TEST SCHEDULE

3.0 INSTRUMENTATION

Three types of instrumentation were chosen for measuring the fluctuations in static pressure in the AEDC 16 ft transonic wind tunnel. A cone calibration device was fabricated. The instrumentation was installed on it and on the side walls of this wind tunnel.

3.1 INSTRUMENTATION FOR FLUCTUATIONS IN STATIC PRESSURE

All fluctuating pressures are recorded by three types of pressure transducers. These transducers are:

- Schaevitz - Bytrex Corp., Model HFD-25
- Kulite Corp., Model CPL-070-4
- Kistler Instrument Corp., Model 601L

All acoustic transducers were calibrated using a 1000 cps signal from a Photocon Research Products, Model PC 125, calibrator. Both the Schaevitz-Bytrex Corp., Model HFD transducer, and the Kulite Corp., Model CPL-070-4, transducer are strain gauge transducers. A part of the strain gauge is located outside the transducer as a compensation module. The Tektronic, Inc., Model RM 122, low level amplifiers are used to amplify the output of both these transducers. The Kistler Instrument Corp., Model 601L, transducer is a Quartz crystal transducer. The Kistler Instrument Corp., Model 553, charge amplifier is used to amplify the output signal of the Kistler transducers. The amplified transducer output is then input to a Data Control Systems, Inc., Model GOV-4, voltage controlled oscillator which converts the output to a FM signal. The FM signal is then recorded on one of the nine channels of a Consolidated Electrodynamics Corp., Model VR-3600, tape system. Each of the nine channels has a $\pm$ 40 KC range and a FM separation of 80 KC. A monitor station is provided between the amplifier and the voltage controlled oscillators. A Ballantine Laboratories, Inc., Model 320A, true rms voltmeter and a Tektronic, Inc., Model 502, oscilloscope are provided at the tunnel monitor station.

3.2 DYNAMIC CALIBRATION CONE FOR THE AEDC 16 FT TRANSONIC TUNNEL

In Reference 11 it was shown that several types of calibration devices have been used in wind tunnel acoustic testing. A brief evaluation of each type of calibration device is presented in Reference 11. It was shown in this evaluation that the most acceptable pressure fluctuation data can be obtained from a combination of calibration devices. This combination was shown to be a slender cone with flat surfaces for mounting instrumentation and sidewall mounted instrumentation.

The AEDC dynamic calibration cone is geometrically similar to the MSFC dynamic calibration device. The instrumentation that was installed was capable of measuring fluctuating pressures in the same frequency range as measured in the MSFC 14 in. tunnel. Figure 2 is a scaled drawing of this calibration

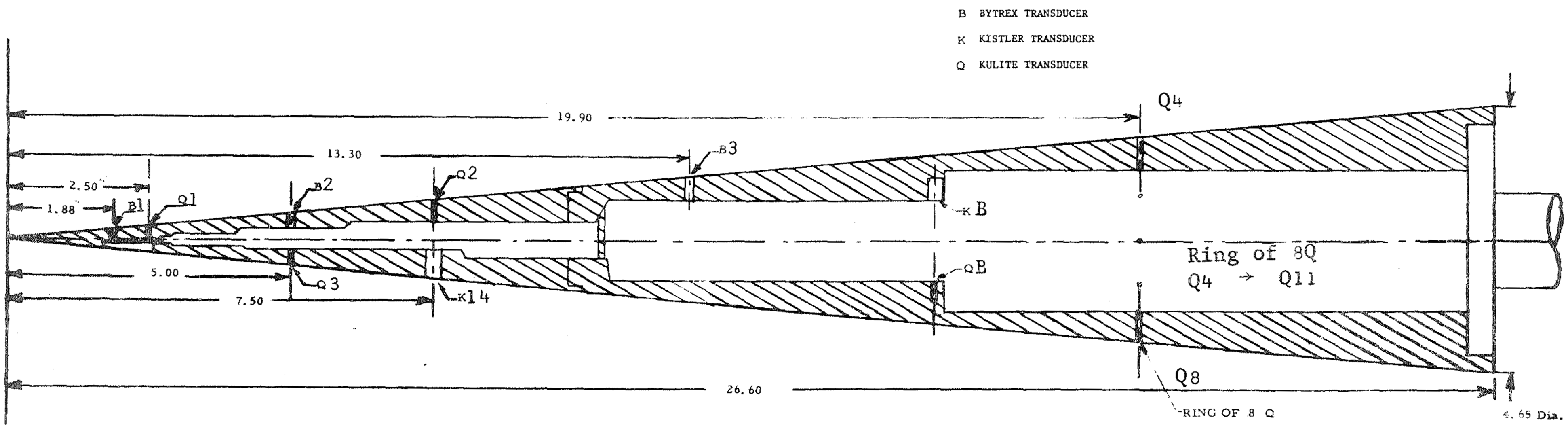

FIGURE 2. DYNAMIC CALIBRATION CONE FOR AEDC 16 FT WIND TUNNEL

device. As can be seen, three different types of transducers were used. The instrumentation location and type are shown in Figure 2. The Bytrex and Kolite transducers require venting. The area and shape of the venting cavity is identical with that of the MSFC 14 in. dynamic calibration cone. The flat surfaces of the cone were mounted facing the upper and lower walls of the tunnel. A ring of transducers is provided at dimensionless model station 0.75 to determine the ring correlation.

3.3 DYNAMIC CALIBRATION SIDEWALL MOUNTED INSTRUMENTATION FOR THE AEDC 16 FT TUNNEL

Wall mounted transducers were used in the AEDC tests to determine the sources of fluctuating pressures and the interdependance of the fluctuating pressures in various sections of the wind tunnel. The locations and designations of the sidewall transducers is shown in Figure 3. Kistler transducers were used for these measurements.

3.4 DATA REDUCTION INSTRUMENTATION

Partial data reduction was conducted. The schedule of the transducer connections to the tape recorder is given in Figure 4. The equipment used in this data reduction was similar to that used in the data acquisition. Tapes were played on a Consolidated Electrodynamics Corp., Model VR-3600, tape system through output voltage controlled oscillators. The output was monitored and rms voltages recorded using a Ballantine Laboratories, Inc., Model 320A, true rms voltmeter. A Tektronic, Inc., Model 502, oscilloscope was also used and a tape monitor.

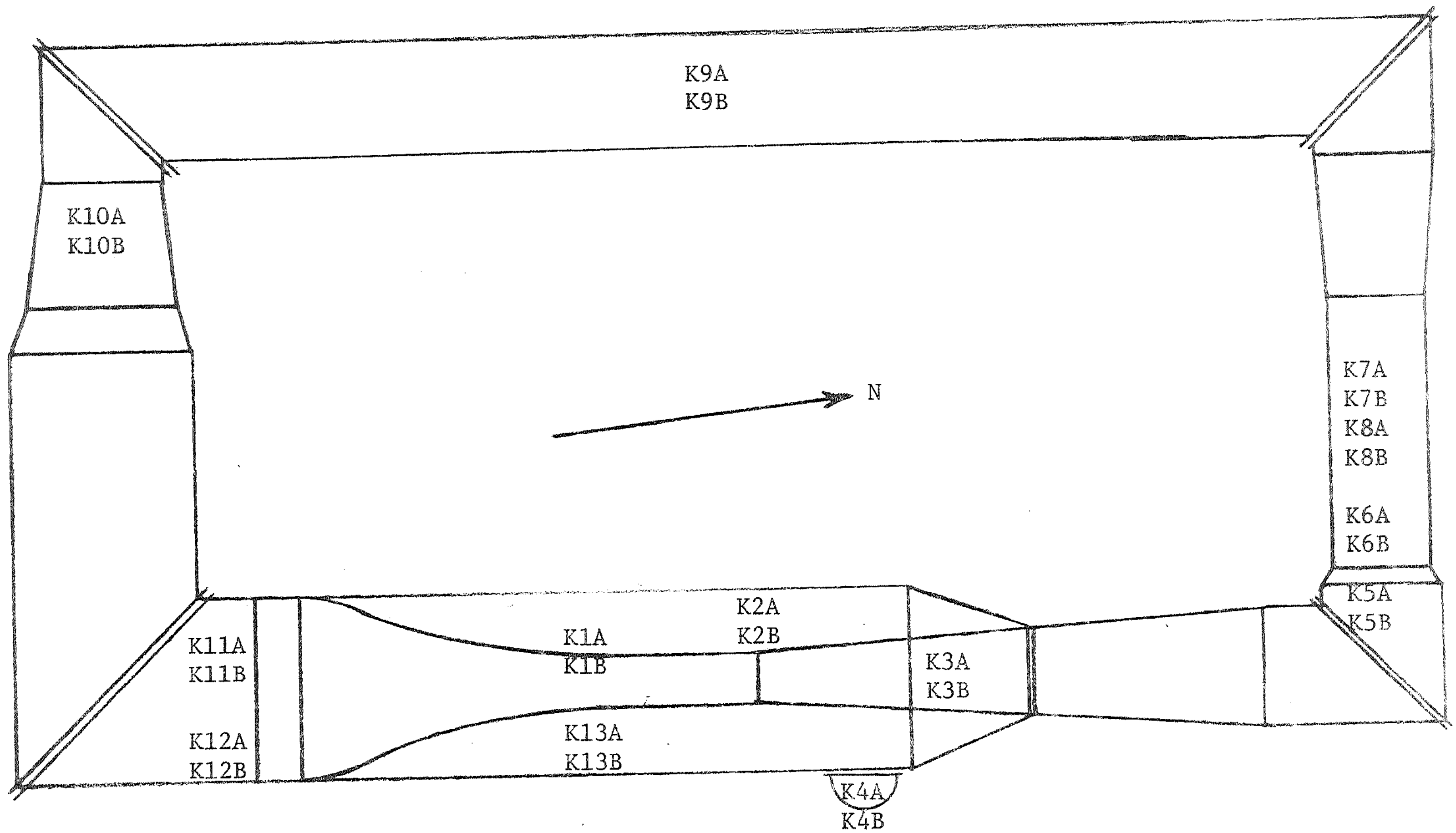

FIGURE 3. LOCATION OF SIDEWALL TRANSDUCERS IN AEDC 16 FT WIND TUNNEL

Transducer	VCO-Tape Ch.	Transducer	VCO-Tape Ch.	Transducer	VCO-Tape Ch.
K1A	1-2	K10A	4-4	B3	7-6
K1B	2-2	K10B	5-4	Q4	8-6
K2A	3-2	K11A	6-4	QB	9-6
K2B	4-2	K11B	7-4	Q3	1-9
K3A	5-2	K12A	8-4	K1A	2-9
K3B	6-2	K12B	9-4	K2A	3-9
K4A	7-2	K12A	1-5	Q4	4-9
K4B	8-2	K13A	2-5	Q8	5-9
K5A	9-2	K13B	3-5	B2	6-9
K4A	1-3	K1A	4-5	Q2	7-9
K5A	2-3	K1B	5-5	Q4	8-9
K5B	3-3	K2A	6-5	QB	9-9
K6A	4-3	K2B	7-5	KB	1-10
K6B	5-3	K1A	8-5	K14	2-10
K7A	6-3	K2A	9-5	Q5	3-10
K7B	7-3	K1A	1-6	Q6	4-10
K8A	8-3	K2A	2-6	Q7	5-10
K8B	9-3	B1	3-6	Q8	6-10
K8A	1-4	Q1	4-6	Q9	7-10
K9A	2-4	B2	5-6	Q10	8-10
K9B	3-4	Q2	6-6	Q11	9-10

FIGURE 4. TRANSDUCER CONNECTIONS WITH TAPE RECORDER

4.0 RESULTS

The data have been reduced by passing the transducer signals from the tape recorder through a root mean square voltmeter. The "wind off" fluctuations were found to be large compared with data recorded during the tests. These "wind off" fluctuations consist of noise in the instrumentation system that appears to be largely concentrated in the 60 cycle frequency regime.

The test data were corrected to account for this instrumentation noise. This correction consisted of subtracting one third of the measured values of the "wind off" fluctuation from the measured values of pressure fluctuations obtained during the tests.

Figures 5 and 6 show the corrected and uncorrected fluctuation values that were measured with a Kulite transducer on the calibration cone. The corrected data are more compact. It shows that the corrected fluctuation coefficients range from 0.008 to 0.024.

Figure 7 gives a comparison between the corrected data from this transducer with data from a buried Kulite transducer located in the calibration cone. The data from the external transducer are three to four times as great as those obtained from the buried transducer.

Figures 8 and 9 show the corrected and uncorrected fluctuation values that were measured with a Kistler transducer located on the porous walls of the test section. The corrected fluctuation coefficients range from 0.009 to 0.040. These values are about twenty five percent larger than those obtained on the calibration cone.

Figure 10 provides a comparison of this wall mounted transducer with a corresponding buried transducer. In this case the fluctuations from the wall mounted transducer are approximately one half of those obtained from the buried transducer.

Figure 11 describes the fluctuations measured in the plenum chamber. The fluctuation coefficients range from 0.004 to 0.016. This is about two thirds of the amplitude of the fluctuations measured on the calibration cone and about half of the values measured at the porous walls.

Figure 12 indicates the comparison of the plenum chamber transducer and the transducer buried in the plenum chamber. As with the sidewall case, the plenum chamber fluctuations are approximately one half of those obtained from the buried transducer.

Figure 13 yields the pressure fluctuations measured with a Bytrex transducer located on the tip of the calibration cone. Values of the fluctuation coefficient of 0.06 to 0.020 were determined. These measurements are about two thirds of those obtained on the calibration cone by the Kulite transducer Q4.

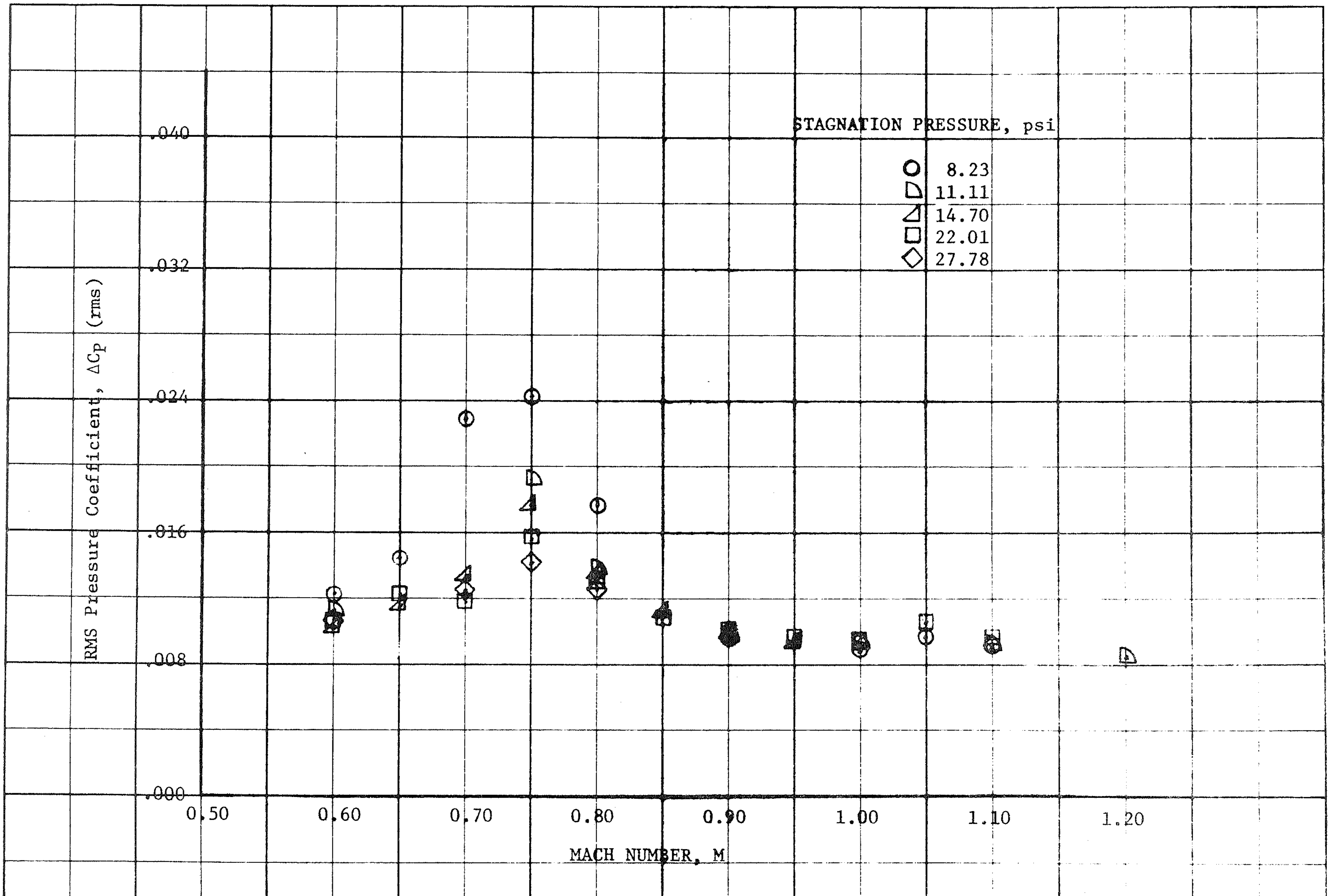

FIGURE 5. CORRECTED CONE PRESSURE FLUCTUATIONS, Q4

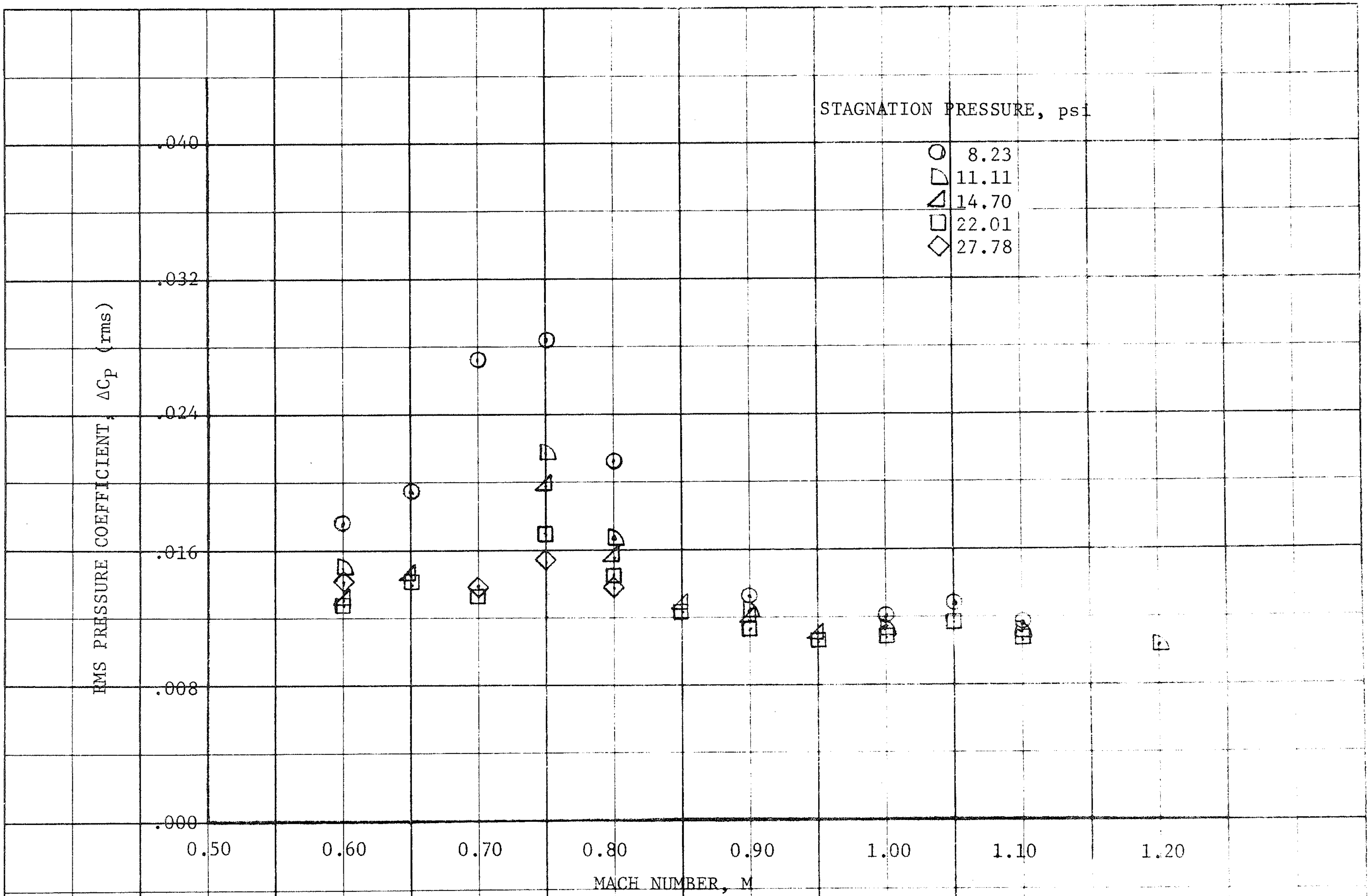

FIGURE 6. UNCORRECTED CONE PRESSURE FLUCTUATIONS, Q4

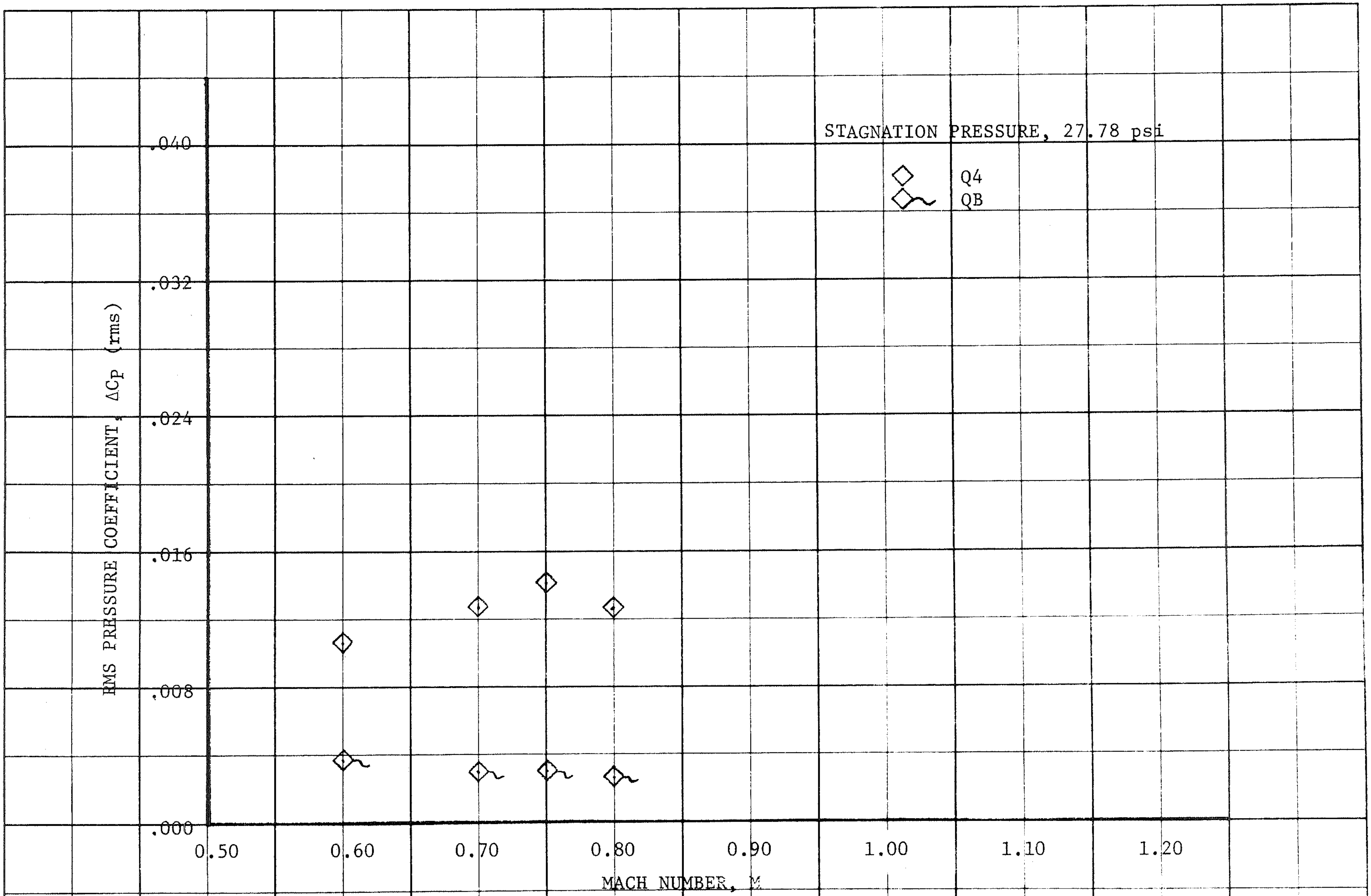

FIGURE 7. COMPARISON OF CORRECTED DATA FROM TRANSDUCER Q4 WITH CORRECTED DATA FROM THE BURIED TRANSDUCER, QB

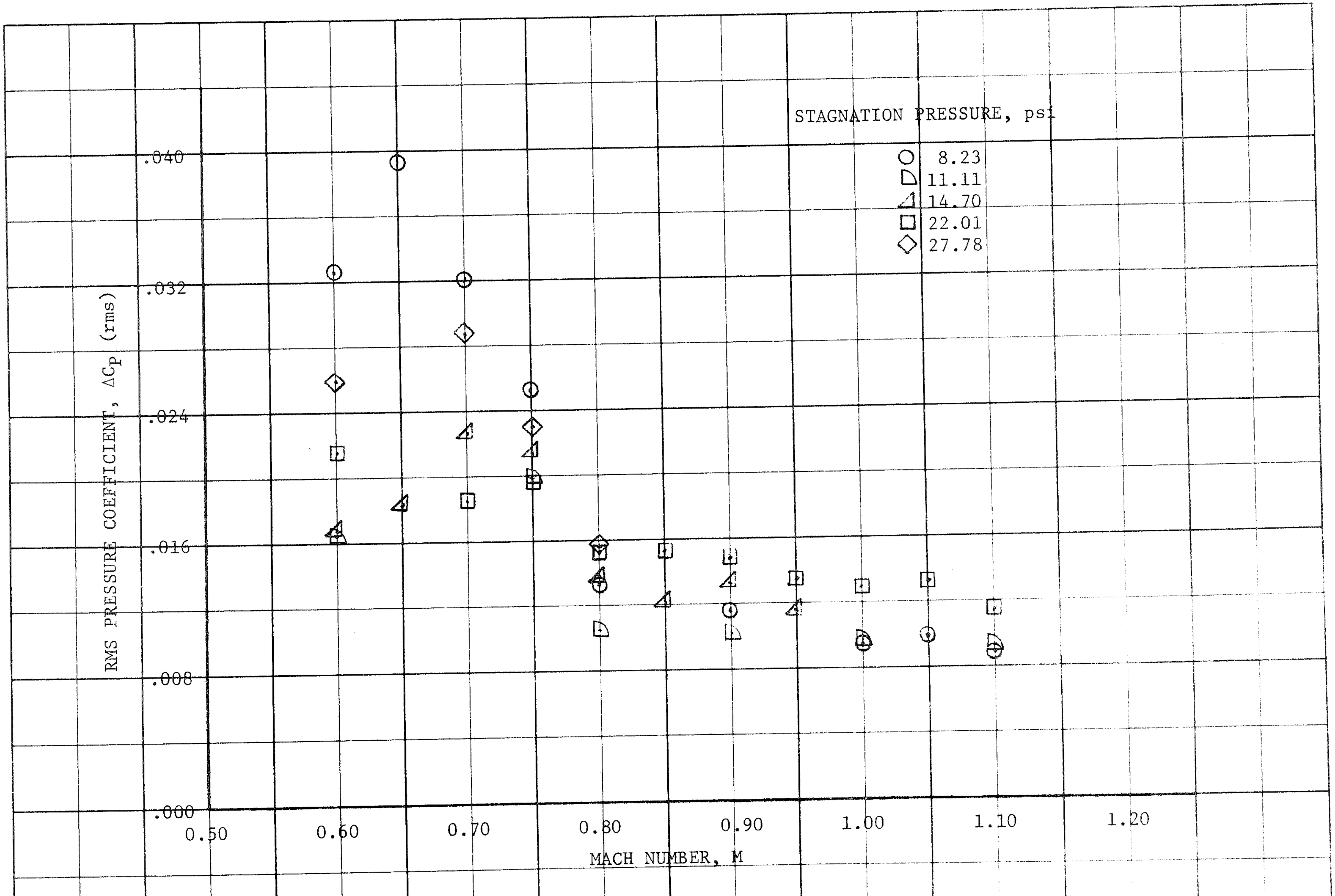

FIGURE 8. CORRECTED POROUS WALL FLUCTUATIONS, K1A

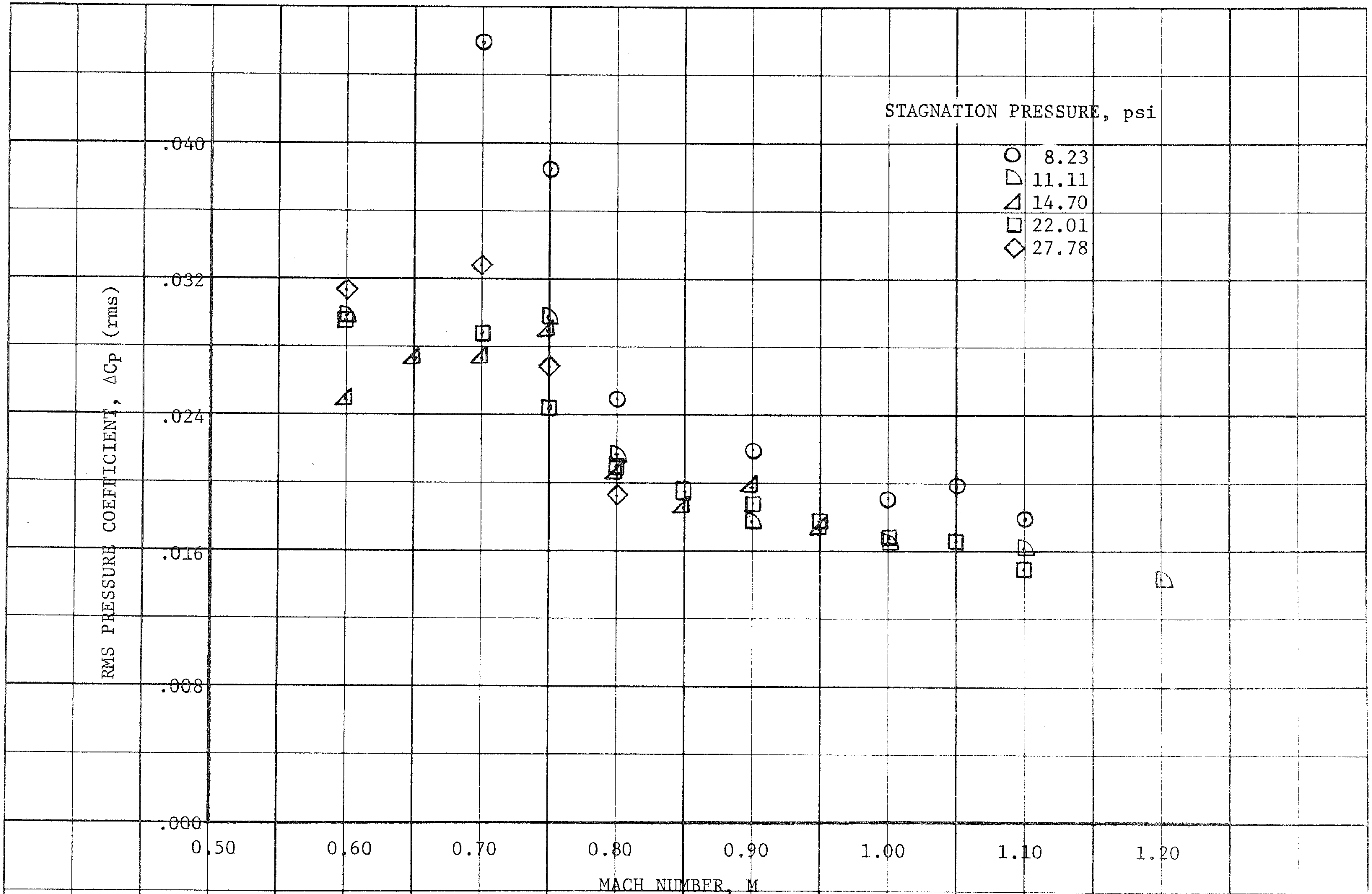

FIGURE 9. UNCORRECTED POROUS WALL FLUCTUATIONS, K1A

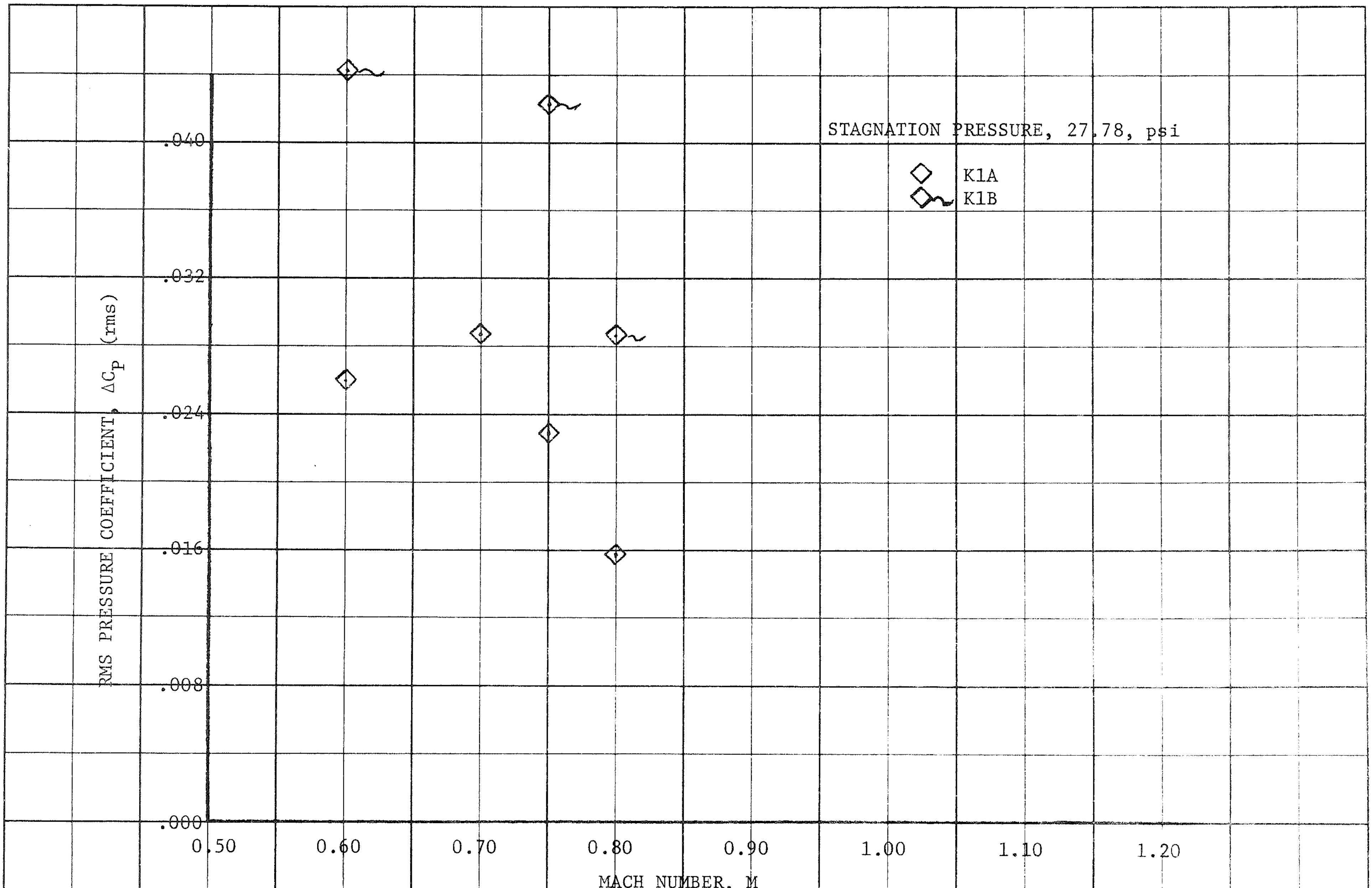

FIGURE 10. COMPARISON OF CORRECTED DATA FROM K1A WITH CORRECTED DATA FROM THE BURIED TRANSDUCER, K1B

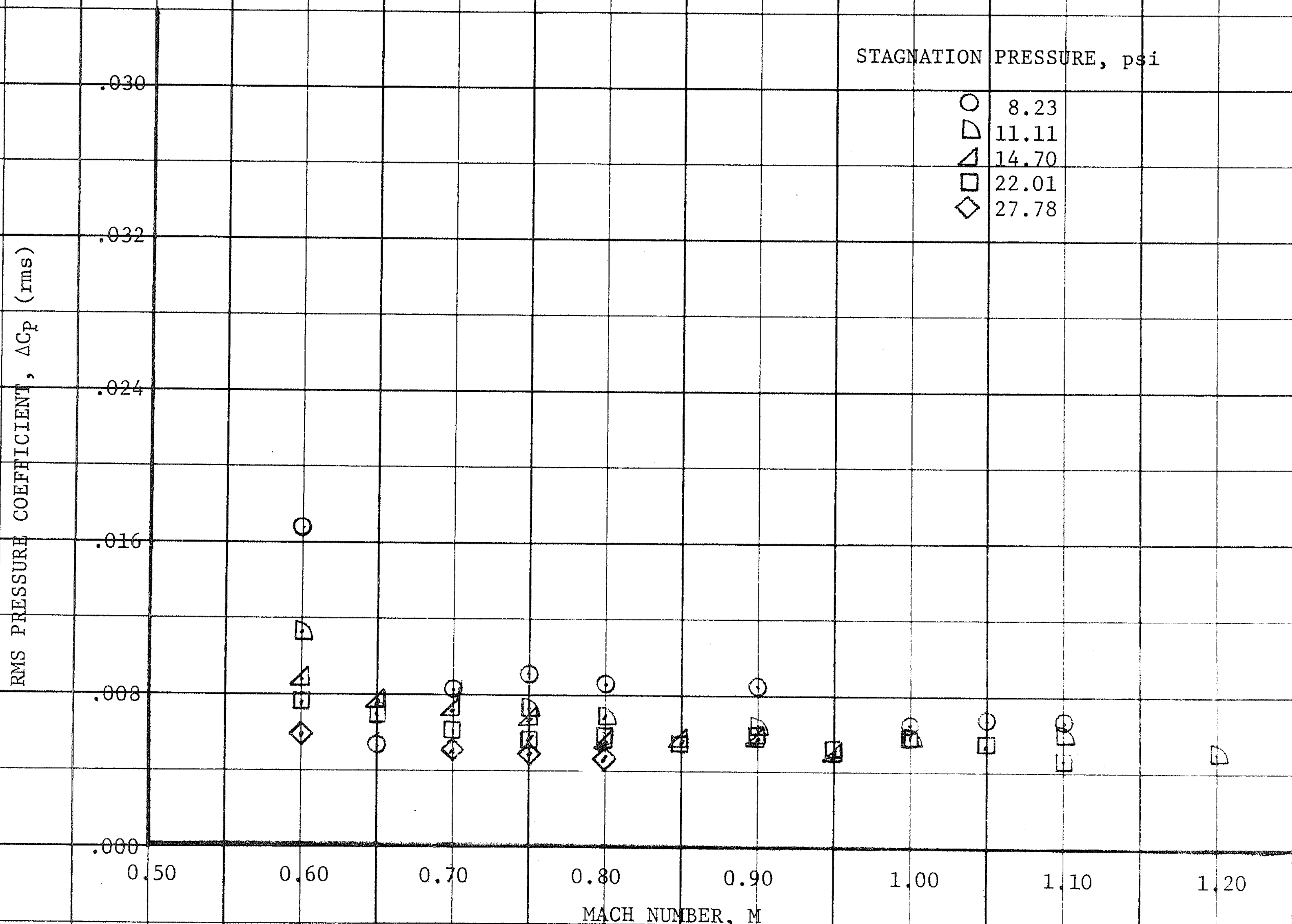

FIGURE 11. CORRECTED PLENUM CHAMBER FLUCTUATIONS, K2A

NO 4520-T 20X20 TO THE INCH

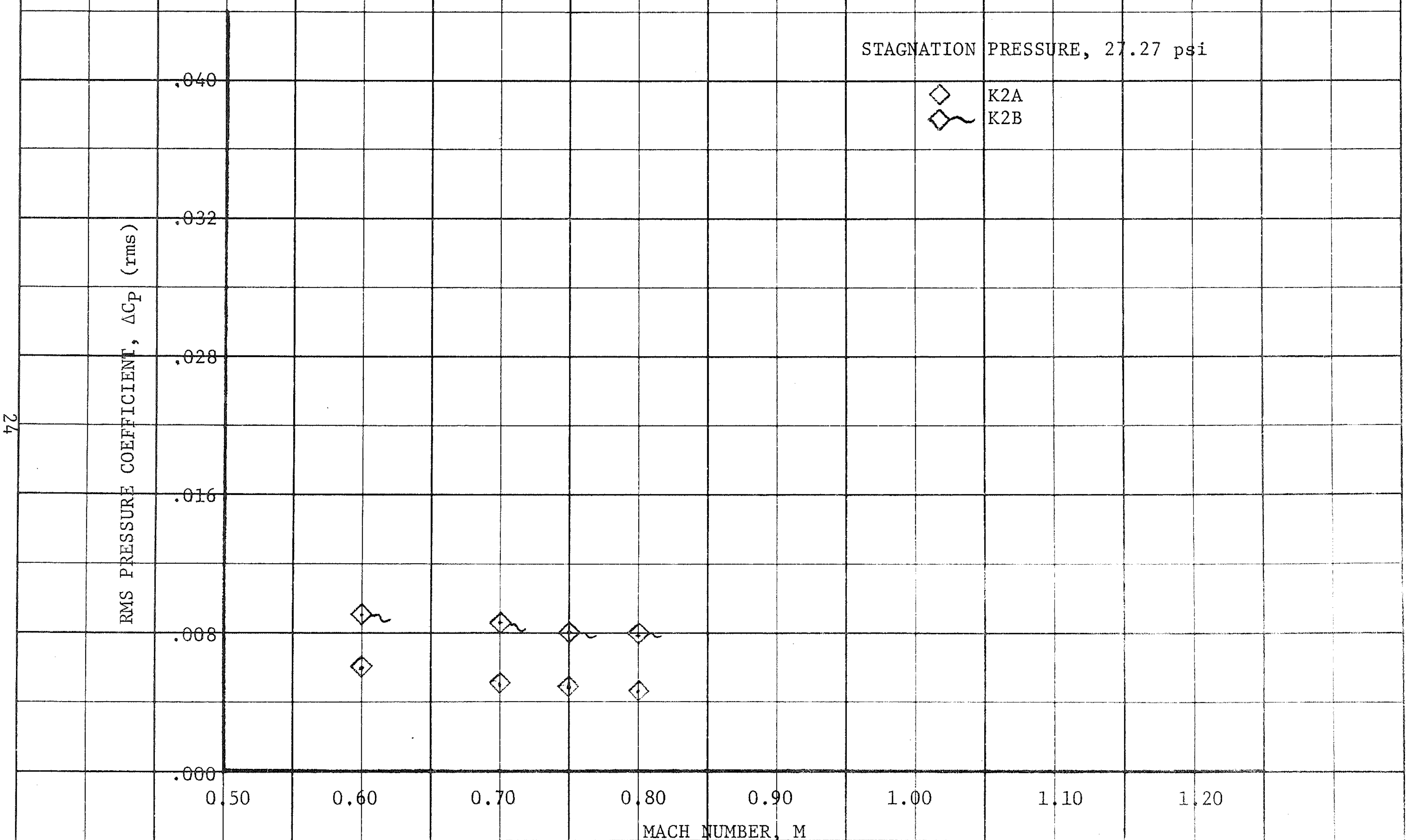

FIGURE 12. COMPARISON OF CORRECTED FLUCTUATIONS FROM TRANSDUCER K2A WITH CORRECTED DATA FROM THE BURIED TRANSDUCER, K2B

NO 4520-T 20X20 TO THE INCH

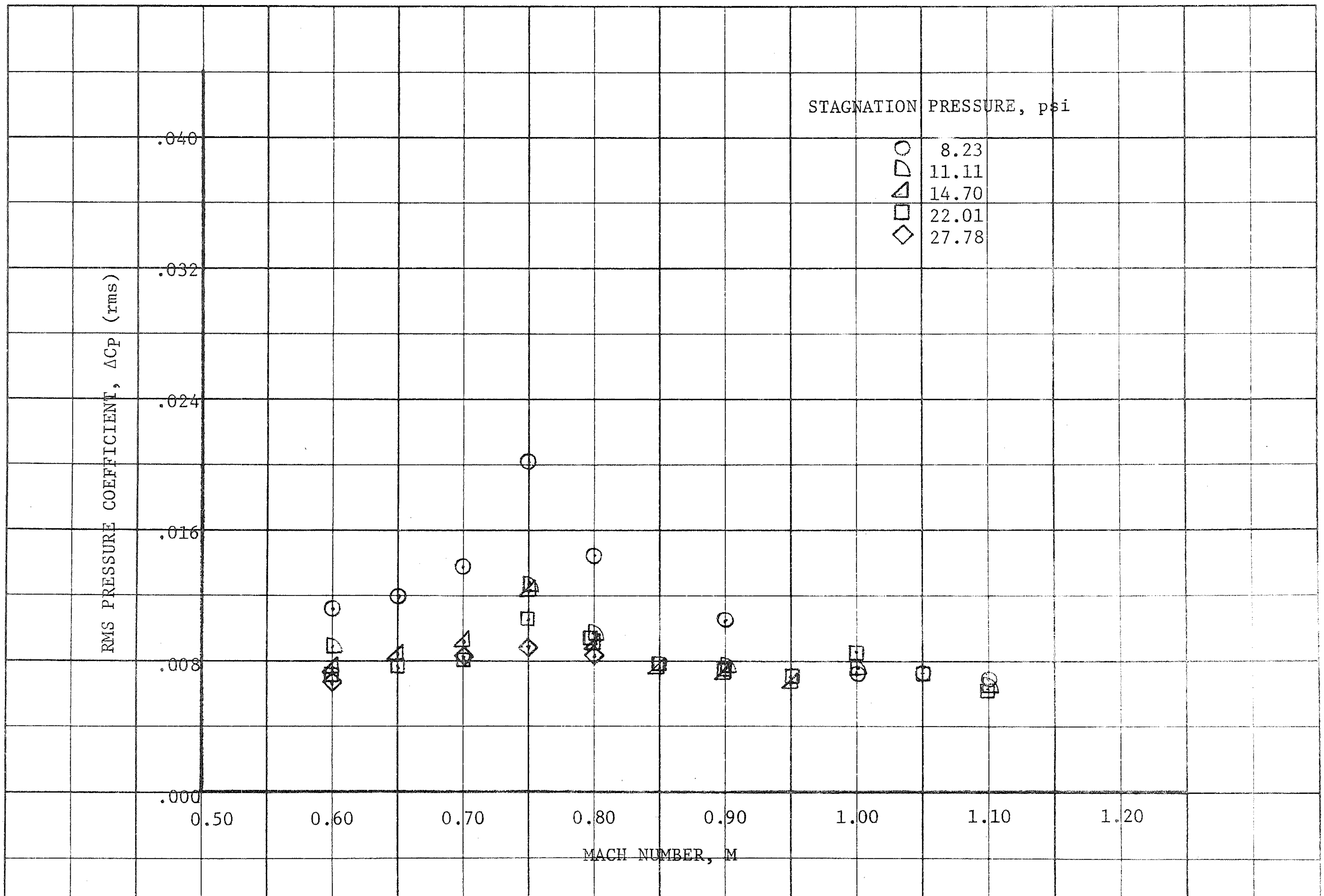

FIGURE 13. CORRECTED CONE PRESSURE FLUCTUATIONS, B1

Figure 14 shows the effect of stagnation pressure on the fluctuation coefficients obtained from the calibration cone at Mach 0.75. These fluctuations range from 0.014 to 0.028. Increasing stagnation pressure is seen to decrease the amplitude of the fluctuation coefficients.

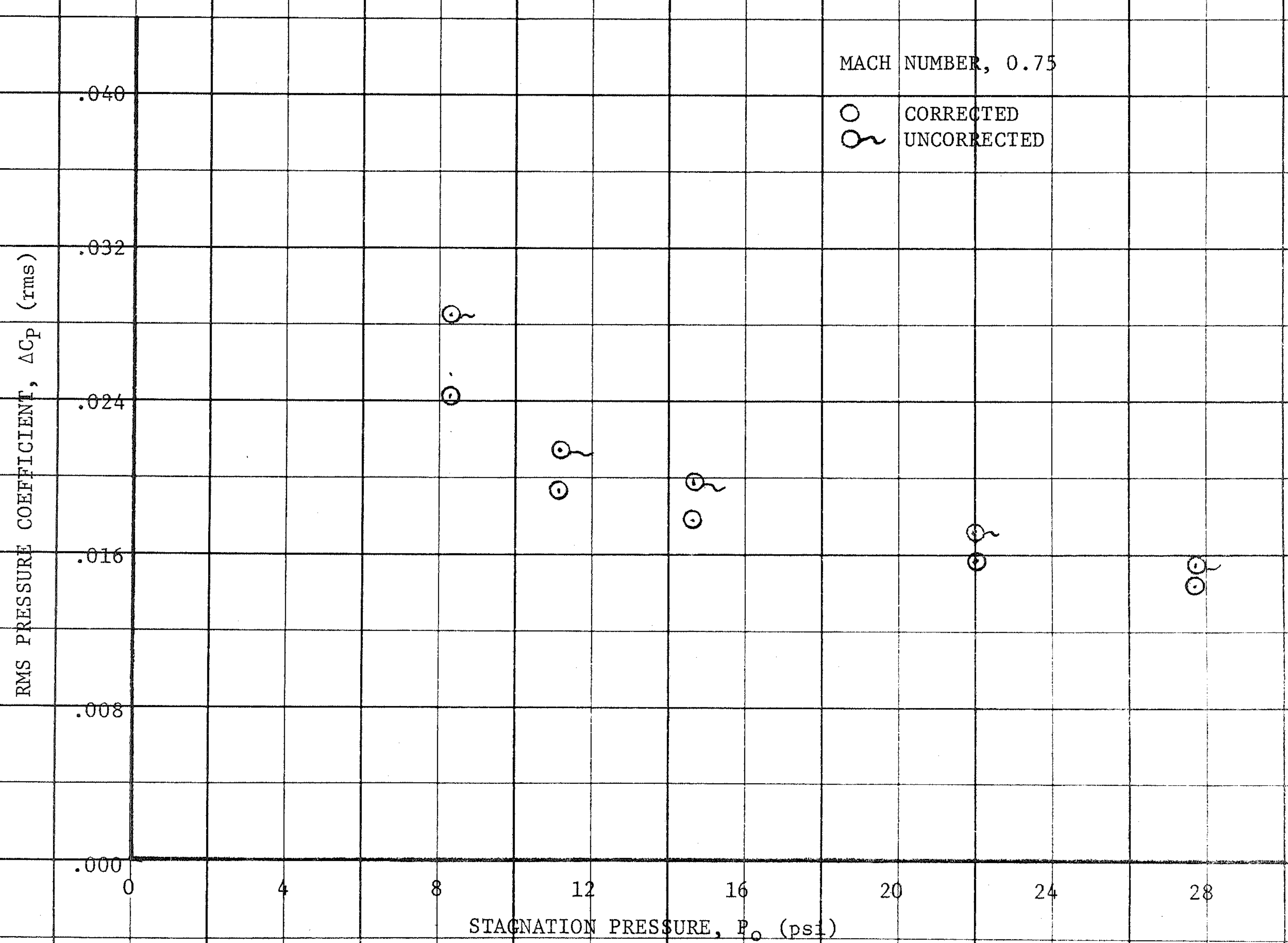

FIGURE 14. EFFECT OF STAGNATION PRESSURE ON CONE PRESSURE FLUCTUATIONS, Q4

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TECHNICAL REPORT HSM-R52-70

CORRECTION OF FOUR-PERCENT SATURN V
MODEL PROTUBERANCE TEST DATA

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