NASA Contractor Report 4289

Exhaust Environment Measurements of a Turbofan Engine Equipped With an Afterburner and 2D Nozzle

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Exhaust Environment Measurements of a Turbofan Engine Equipped With an Afterburner and 2D Nozzle

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TABLE OF CONTENTS

1.0 SUMMARY

A test to measure the acoustic noise and static pressure environment on structure exposed to engine exhaust flow was conducted at the NASA Lewis Research Center (LeRC) engine test facilities. The engine was an F100 derivative, serial number XD11-12, with a two-dimensional convergent-divergent (2D/CD) non-flight-weight demonstrator nozzle. Testing was conducted in an altitude test chamber of the Propulsion Systems Laboratory (PSL) which allows for static testing at simulated altitude for both intermediate and augmented engine power settings. A highly instrumented, water cooled flat panel was placed behind the 2D/CD nozzle, and tests were conducted at simulated Mach number and altitude flight conditions with the engine at military (MIL) or maximum afterburner (MAX A/B) power setting. The panel instrumentation consisted of acoustic pressure microphones, thermocouples, and static pressure pickups. Considering Mach number and altitude conditions, panel positions, and engine power settings, a total of 39 test points were requested by MCAIR. The Mach number 0.8 and 24,000 feet test condition was required by NASA and P&W. All of the intermediate power test conditions were obtained, but only about half of the augmented test conditions were achieved due to engine and nozzle flap liner problems.

On site octave band spectrum analyses were performed for all of the data. The data appear reasonable and valid in comparison with

limited measurements from a similar test at McDonnell Aircraft Co. (MCAIR). The most significant trend observed during the test is the reduction in overall sound pressure levels with increasing altitude for all power settings tested. Substantial pressure level across the entire frequency spectrum indicates that the exhaust environment may excite structural resonances as high as 10,000 Hz.

The test was a cooperative effort involving McDonnell Aircraft Co. (MCAIR), Air Force Wright Research & Development Center (WRDC), NASA Lewis Research Center (LeRC), and Pratt & Whitney (P&W).

2.0 INTRODUCTION

A test to measure the acoustic noise and static pressure environment on structure exposed to engine exhaust flow was conducted from 27 April through 6 May 1986 at the LeRC engine test facilities. The engine used was a turbofan equipped with an afterburner and two-dimensional nozzle. Specifically, it was an F100 derivative, serial number XD11-12, with a 2D/CD non-flight-weight demonstrator exhaust nozzle.

Both augmented and non-augmented engine modes were used. The exposed structure for the test was a highly instrumented water cooled flat panel. Measurements were obtained on the panel surface which was placed behind the 2D/CD nozzle at three positions (grazing and two positions away from grazing) relative to the exhaust flow. The panel was instrumented with microphones, static pressure ports, and thermocouples. Acoustic data were then analyzed to obtain sound pressure level, power spectral density (PSD), cross power spectral density (CSD), and coherence (COH). These results will be used for design to predict vibratory structural response. Laboratory tests can then be performed for preliminary qualification of aircraft structure exposed to engine exhaust flow.

Measurements were obtained at eight simulated flight conditions for non-augmented operation and at two flight conditions for maximum augmentation. Mach number ranged from 0.8 to 1.83, and altitude ranged from 15,000 to 40,000 feet. The test was a cooperative effort involving MCAIR, WRDC, LeRC, and P&W.

3.0 TEST FACILITY AND HARDWARE

A P&WA F-100 derivative engine with the 2D/CD demonstrator nozzle was installed in an LeRC Propulsion Systems Lab (PSL) altitude engine test cell. MCAIR designed and fabricated an instrumented test panel and support system to measure the exhaust environment generated by the engine. Overall test approval was granted by WRDC.

3.1 PROPULSION SYSTEMS LAB ALTITUDE TEST CELL

Testing was conducted in PSL test chamber 3, a ground level high altitude engine test facility. Figure 1 is a photograph of the general test layout with all hardware, including the instrumented panel, installed. Recent modifications added the capability to run thrust reversing. This included ducts (not used for this test) which turned the reversed flow back downstream into the conical exhaust collector. MCAIR designed their hardware to interface with the conical collector.

This pressurized facility allowed for static testing at simulated altitude for both intermediate and augmented engine power settings. A forward bulkhead separated the inlet plenum from the test chamber. Conditioned air, at the desired inlet pressure and temperature, flowed from the plenum through a bellmouth and duct to the engine. Engine exhaust was captured by a collector which

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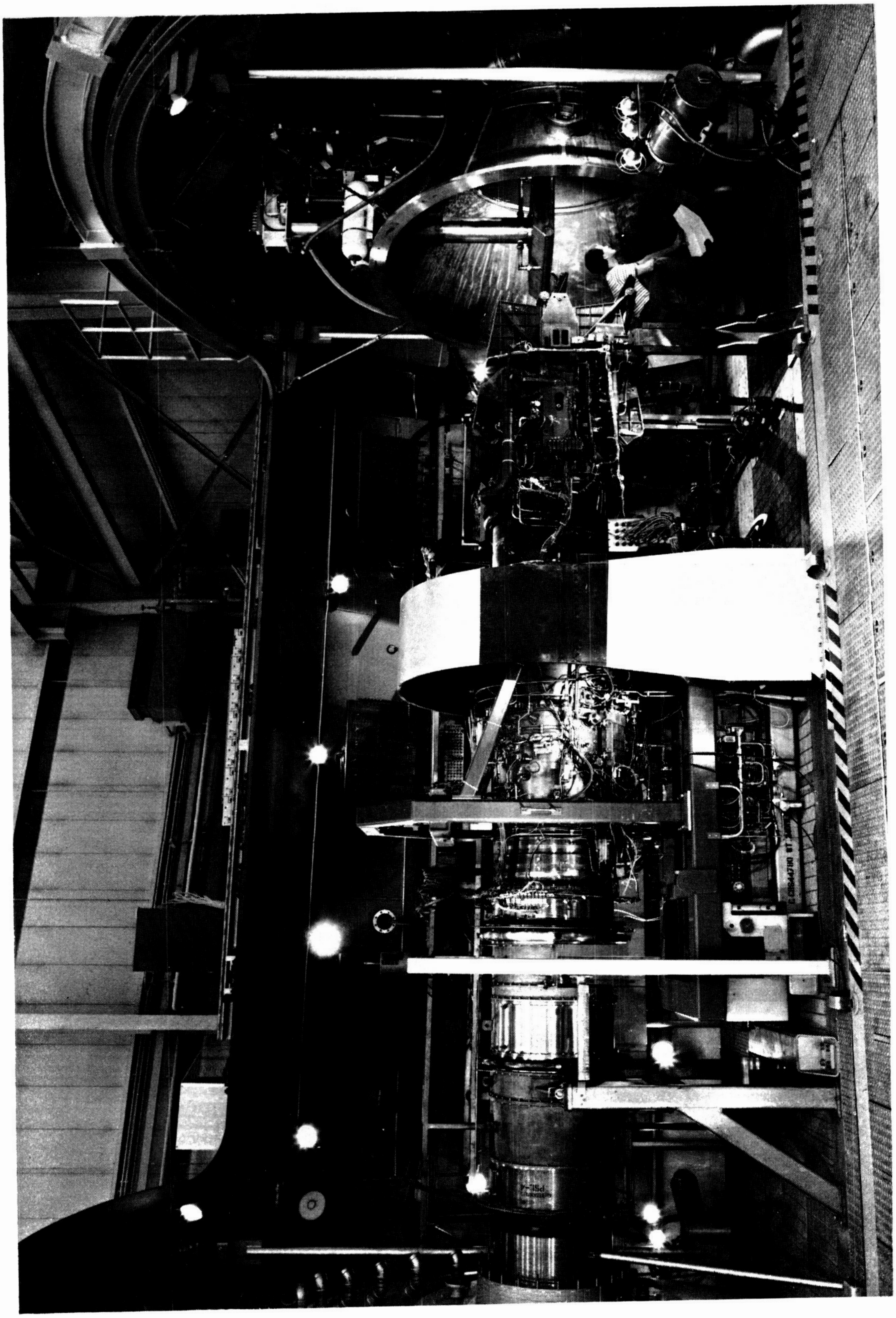

Figure 1. General Test Layout

extends through the test chamber rear bulkhead. Altitude was simulated by evacuating the test chamber to the desired pressure. A cover plate, not shown in Figure 1, was mounted on the conical exhaust collector section. This plate had a rectangular opening for the engine exhaust flow. It also provided a shield for the test panel in the stowed position. The tunnel operating capabilities, shown in Figure 2, cover an altitude range of about 10,000 to 55,000 feet. One primary limitation for this installation is 190 ^{o}F engine face total temperature. This limit was imposed to protect some temporary hardware located in the inlet plenum section. Minimum altitudes for maximum and minimum augmentation are also indicated.

3.2 ENGINE AND NOZZLE

The propulsion system consisted of an F100 derivative, SN XD11-12, low bypass turbofan engine with thrust augmentation and a 2D/CD demonstrator nozzle. This nozzle operates in either a conventional, vectored, or reversing thrust mode. All the MCAIR testing was performed with the nozzle operating in the conventional forward-flight mode. The nozzle flaps were lined with a high temperature material.

3.3 PANEL AND INSTRUMENTATION

MCAIR designed and built the highly instrumented, water cooled flat panel displayed in Figure 3. The welded, stainless steel,

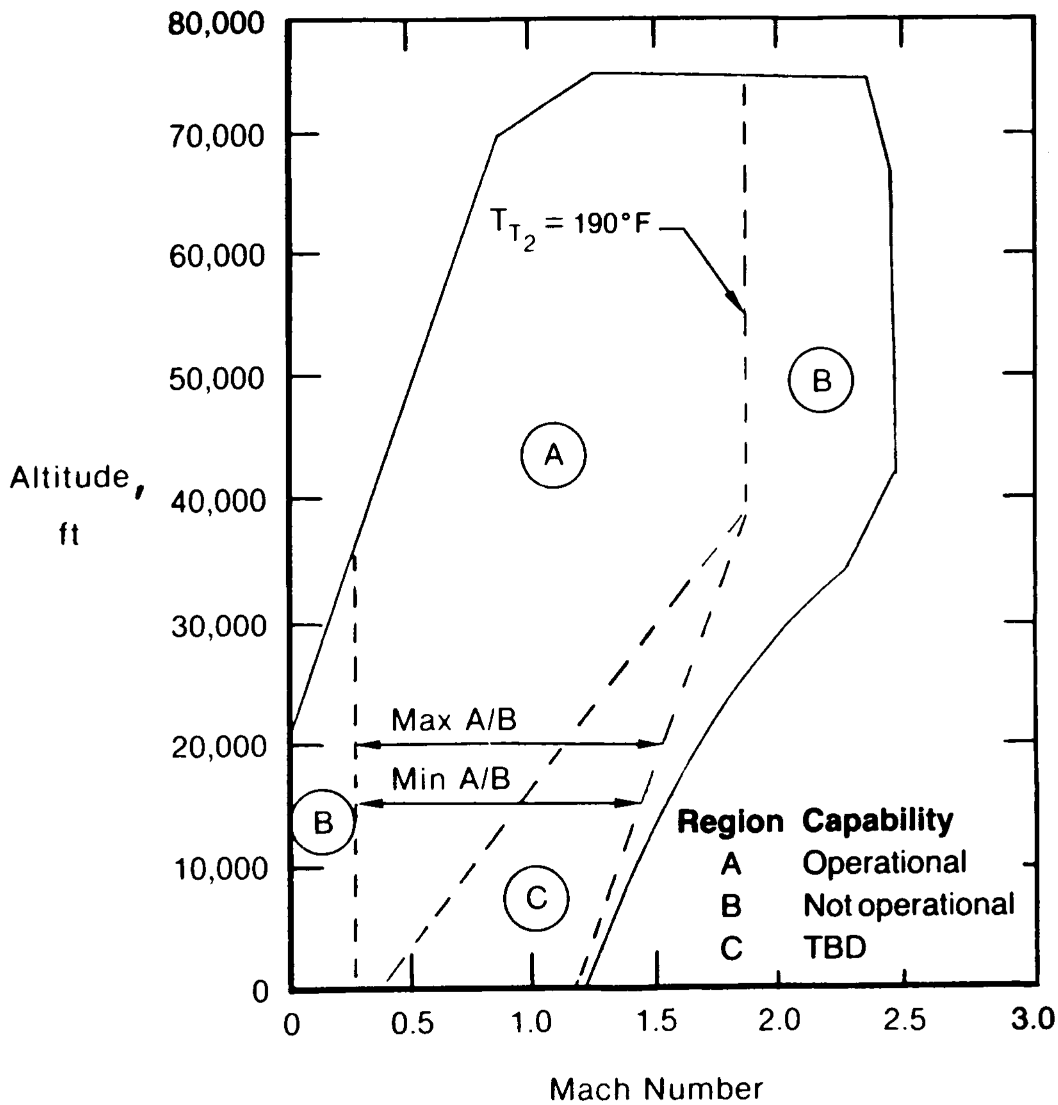

Figure 2. Engine Test Facility Operating Capability

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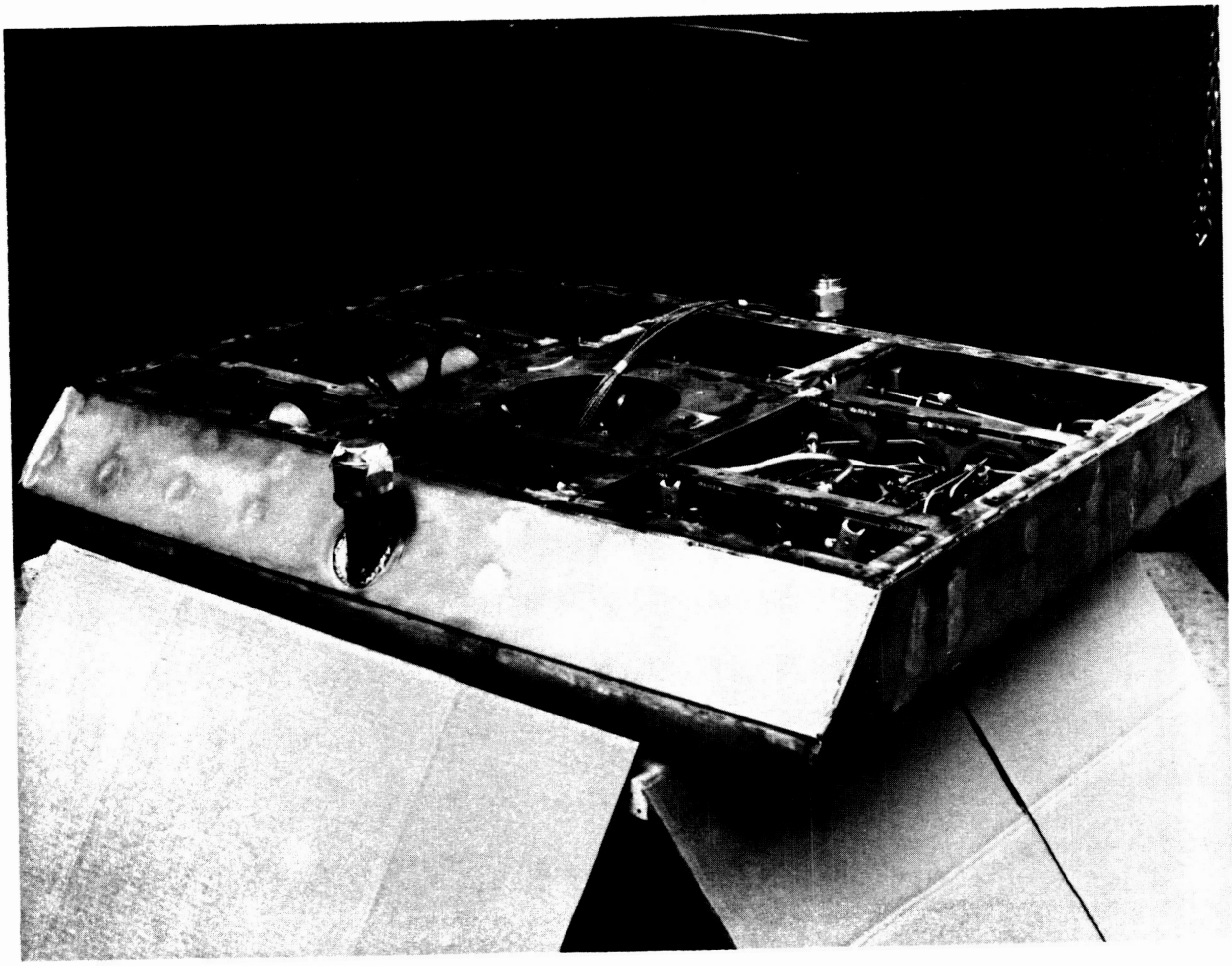
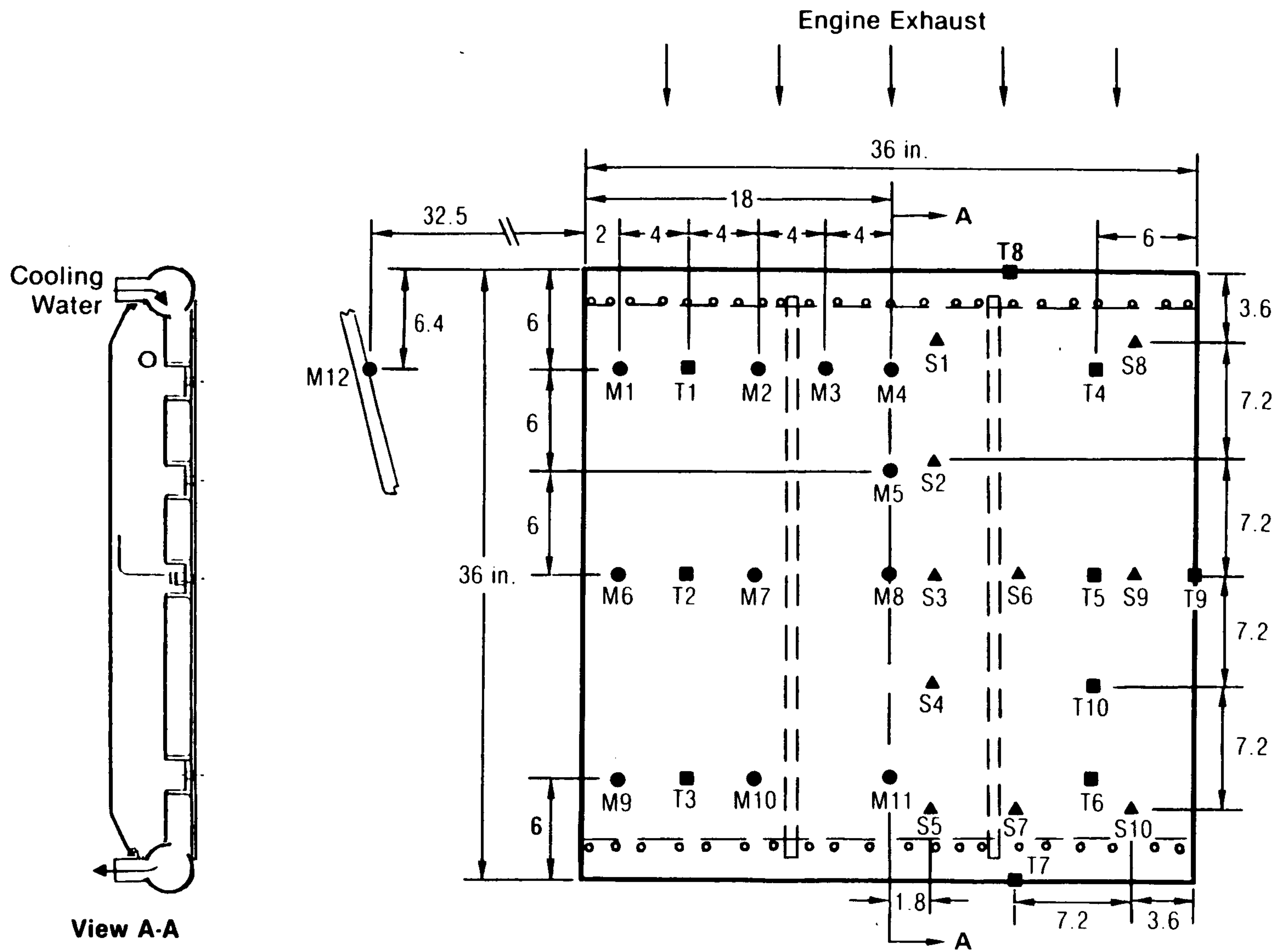


Figure 3. MCAIR Instrumented Test Panel

sandwich construction had internal recessed instrumentation installations for the transducer assemblies. The panel instrumentation consisted of 11 acoustic pressure microphones, 10 thermocouples, and 10 static pressure ports arranged as in Figure 4. Each microphone (PCB Piezotronics Model 112A21) was mounted in a water jacket (PCB Model 64A02) positioned with the diaphragm flush with the outer surface. The high temperature transducer leads were wrapped with aluminum foil for added protection. A twelfth microphone was mounted on the conical exhaust collector wall to provide a reference for any acoustic corrections required due to the effects of the enclosure. Before and after each test day, microphone calibrations were checked with a broadband random noise source (150 dB overall) generated by a portable acoustic driver with a rubber horn attachment. By placing the horn over each microphone, a seal was formed to eliminate external noise. The static pressure ports were connected to the NASA data system with stainless steel tubing. Thermocouple wires were tack welded to the plate inner surface and routed to the NASA data system. Figures 5 and 6 illustrate typical instrumentation installations.

3.4 PANEL SUPPORT AND POSITIONING STRUCTURE

Figure 7 is a schematic diagram of the test facilities used to measure the exhaust environment. The panel was connected to a shaft that travelled vertically through a guide cylinder on dry film lubricated sleeve bearings placed inside each end. The shaft was



● 12 acoustic pressure microphones (M1 – M12)
■ 10 thermocouples (T1 – T10) with T7 – T10 on panel backside
▲ 10 static pressure pickups (S1 – S10)

Figure 4. Test Panel Instrumentation Locations

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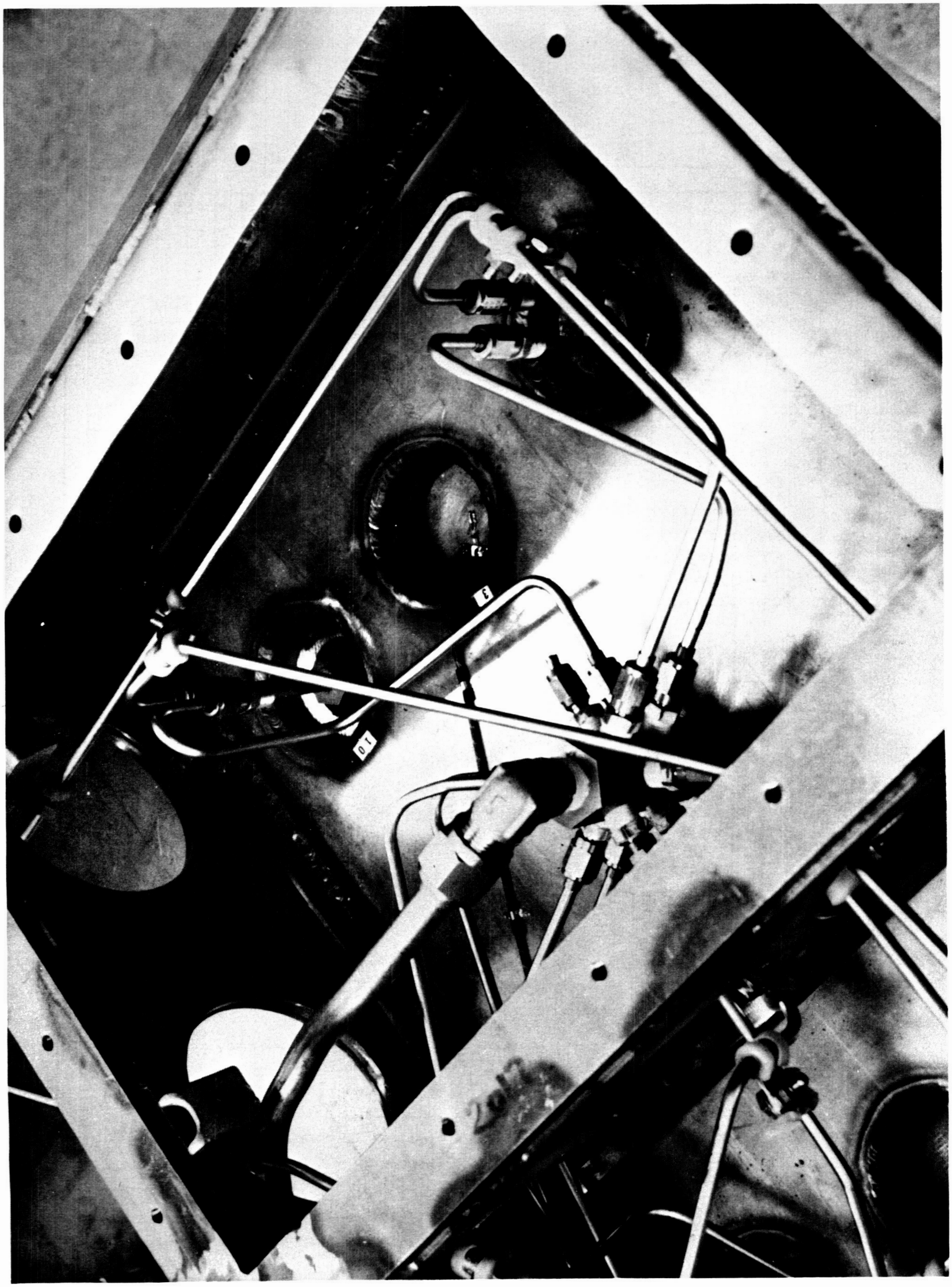

Figure 5. Test Panel Microphone and Thermocouple Installation (Typical)

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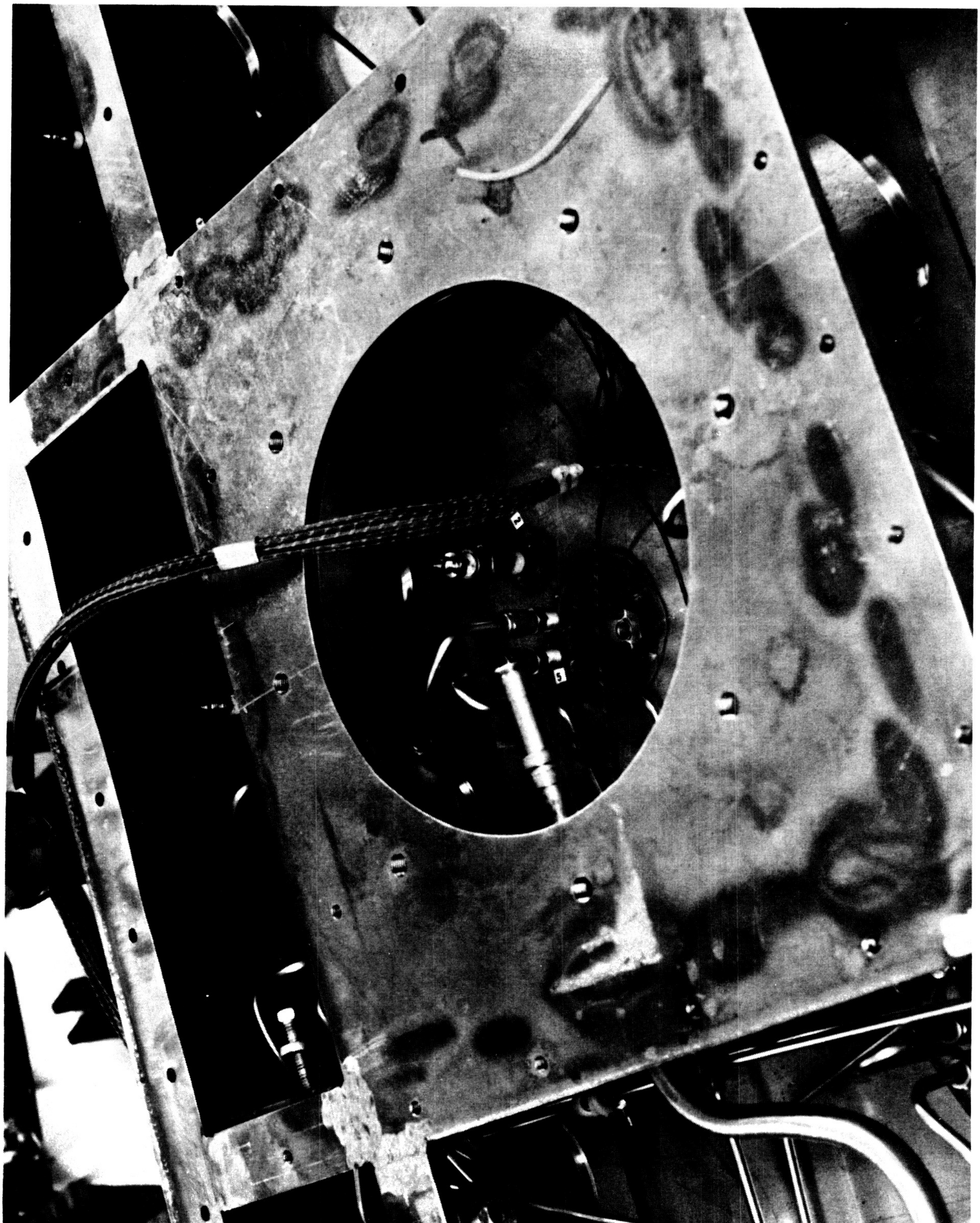


Figure 6. Test Panel Microphone and Static Pressure Port Installation (Typical)

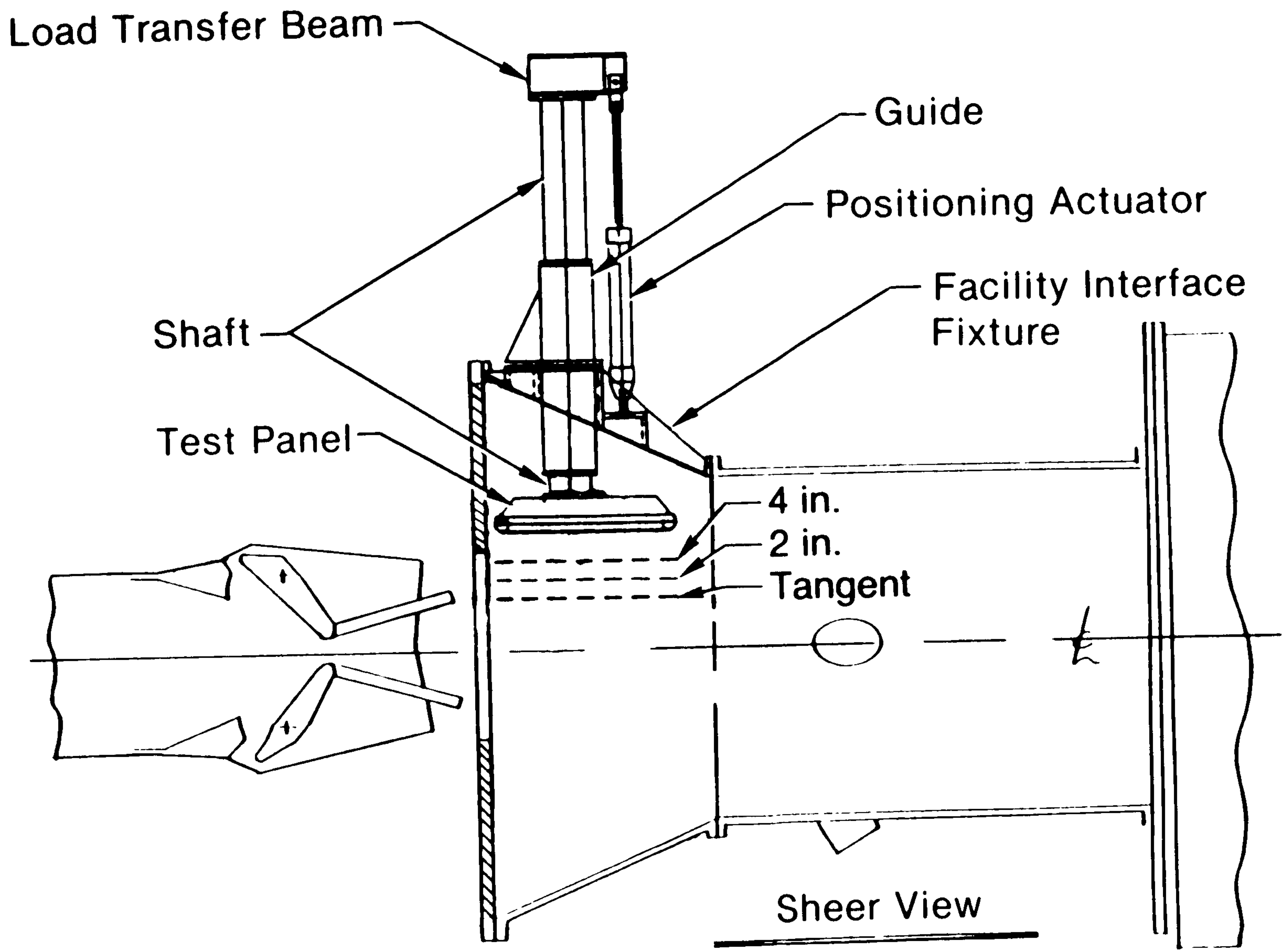


Figure 7. Schematic of Engine Test Facility

keyed to the guide to restrict rotation. Cooling water, shop air, transducer wires, and tubing were routed through the shaft core. A load transfer beam connected the shaft top to the ram end of a positioning actuator and also supported various incidental items not shown. The lower ends of the actuator and guide were bolted to a facility interface fixture. Panel vertical motion was hydraulically powered and a low pressure accumulator acted as the actuator reservoir. A charged high pressure accumulator was included for emergency retraction. The system electronics enabled accurate panel placement by using a positioning transducer within a feedback loop.

3.5 FACILITY AND HARDWARE INTEGRATION

The primary integration involved some modification to the conical exhaust collector. A hole was torch cut in the collector upper surface, and the interface adaptor fixture welded in position. From this base, the panel and positioning assembly were erected. Cooling tower and city water were used for cooling of the panel and microphones, respectively. The narrow waterjacket passages required clean city water.

4.0 DATA REQUIREMENTS AND TEST PROCEDURE

For each test condition, at least 30 seconds of data were taken at three panel positions; namely, tangential (grazing incidence), two inches, and four inches above tangential. Tangential location is defined as the intersection of the panel instrumented lower surface plane with the lower corner of the nozzle upper divergent flap trailing edge. Acoustic noise, static pressure, and associated temperature measurements were obtained for all test points. The instrumentation was considered adequate for a maximum temperature of 2,500 to 3,000 ^{o}F; static pressure of 30 psia; and overall sound pressure level (OASPL) of 180 dB for a frequency range of 25 to 10,000 Hz. Real time display of each transducer output was required to check proper operation and gain setting. Acoustic data were recorded on magnetic tape for later reduction. Static pressures and temperatures were sampled through the NASA data system and tabulated as average values. NASA furnished drawings and photographs of the test setup, and a test log which indicated simulated flight condition, engine power setting, and panel location.

After the hardware was assembled, engine-off checks were performed to calibrate the remote control mechanisms for the desired panel and nozzle flap relative positions. The nozzle was configured for engine power settings of idle or intermediate (MIL) and maximum afterburner (MAX A/B). Nozzle position for idle and MIL power is nominally the same at altitudes above sea level. MAX A/B nozzle position varies with altitude and engine face condition. A simple relationship between nozzle exit area/flap position and tangential panel location was derived. A closure panel was placed over the upstream opening of the conical exhaust collector. A rectangular opening, sized by the largest nozzle area ratio, allowed the exhaust flow to pass into the collector.

The test was conducted by NASA and MCAIR personnel from the control room adjacent to the cell. NASA directed the test, regulated the cell environment, and operated the engine/nozzle with assistance from a P&W representative. MCAIR chose the test conditions, operated the panel/positioner, monitored the instrumentation associated with their hardware, and recorded the acoustic data on magnetic tape.

5.0 TEST DISCUSSION

Testing was performed from 27 April through 6 May 1986 with exhaust measurements taken on 1 and 5 May. The test plan called for data to be acquired for the simulated flight conditions given in Figure 8. However, engine and nozzle flap liner problems prevented finishing the entire test program. The points are numbered in the order that they were attempted, and they were tested by holding the inlet temperature constant and changing the simulated altitude. NASA and P&WA required point 1 as a general data check each time the engine was tested.

Points 1 through 5 were run during the first test period. The MAX A/B power setting of point 5 could not be performed because the afterburner would not stay lit at this condition. This is puzzling since the flow conditions of points 3 or 4 should have been more stringent for the augmentor. A post test inspection revealed some isolated damage to the convergent section of the nozzle flap liners that required replacement. Also, the NASA data system showed that the exhaust flow was choked for the MAX A/B setting of point 3. This was corrected for successive testing by increasing the opening of the conical exhaust collector. The engine was retrimmed so the A/B would not blow out.

Points 1 and 3 were repeated during the second test period. Point 5 was again attempted, but the augmentor would still not stay

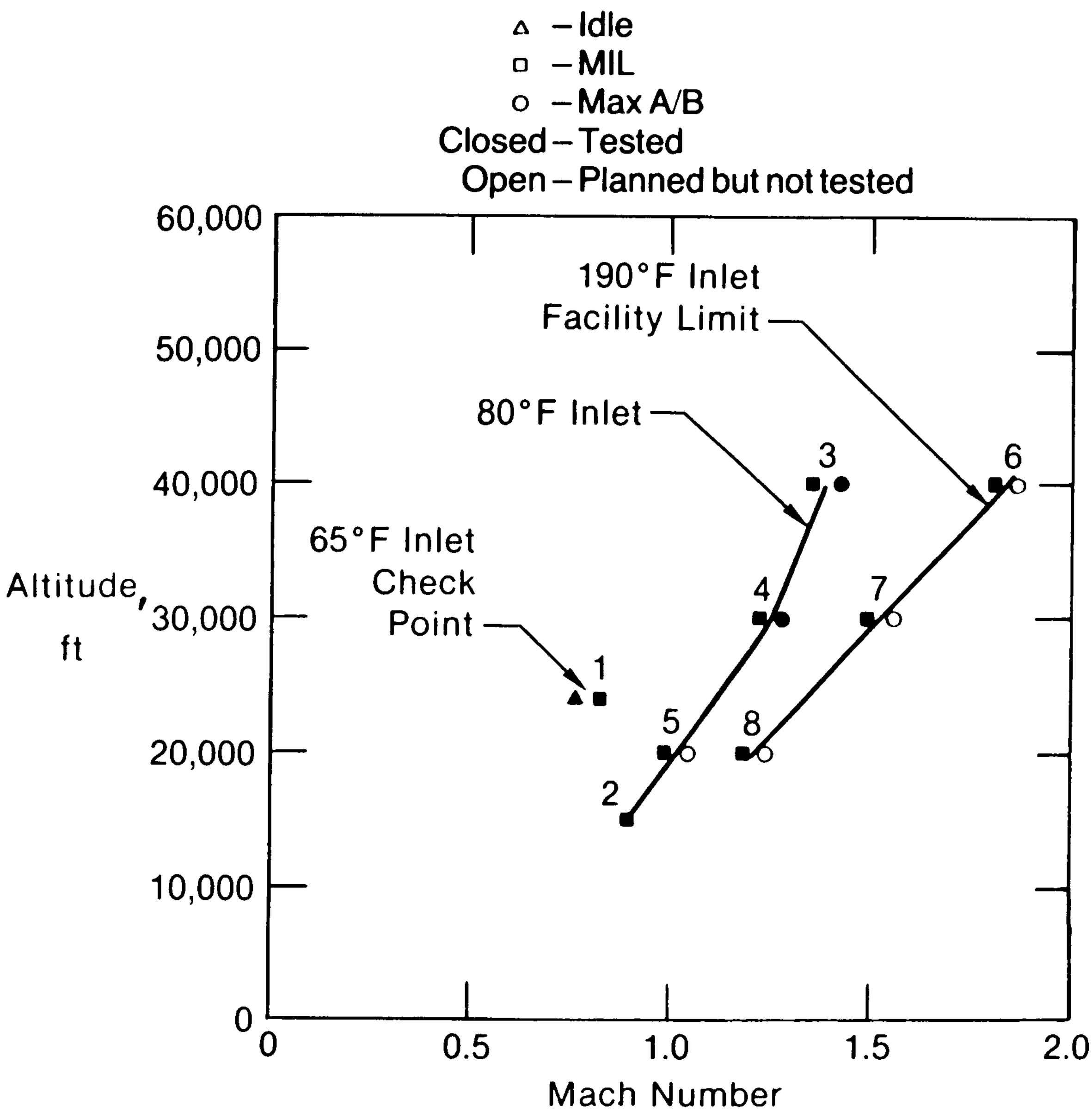


Figure 8. Engine Exhaust Test Points

lit. Testing was interrupted for facility and hardware inspection, and to retrim the engine once again. Convergent section nozzle flap liner damage was such that further augmented testing could not be accomplished during this test period. The MIL power setting for points 6 through 8 were completed without further complications.

Upon removal of the convergent section liners, more damage was discovered. After assessment of the damage and condition of available spares, P&W and NASA decided that the remaining augmented test points could not be completed.

6.0 TEST RESULTS

Acoustic fluctuating pressures, static pressures, and cooled panel temperatures were successfully measured for all simulated flight conditions. All instrumentation functioned properly throughout the test. The limited measurements, obtained during the earlier MCAIR sea level static test using an F100 engine (Ref. 1), indicated the data are reasonable and valid. Table I lists all conditions tested and the associated data which have been reduced by MCAIR, to date. Figure 9 depicts the instrumented panel position, relative to the nozzle flap, for typical idle or MIL and MAX A/B power settings. Figures 10 through 20 and Tables II through IV present a sampling of the acoustic data, most of which is for the Mach number 1.25 and 30,000 feet test condition (Test Point 4). This was the lowest altitude at which both dry power and augmented testing were completed. Table V lists static pressure and temperature data of Test Point 4 for both engine power settings and two panel positions.

6.1 ACOUSTIC PRESSURE

Test Point 4 octave band spectrum analyses from 25 to 10,000 Hz, Figures 10 through 13, are provided for microphones 2 and 7; panel positions graze (tangential) and four inches above graze; and MIL and MAX A/B engine power settings. Acoustic levels are defined in decibels (dB) referenced to 2.90×10^{-9} psia, the American National

TABLE I. TEST CONDITIONS AND DATA REDUCED

POINT	MACH NO.	ALT. (ft)	ENGINE POWER	ACOUSTIC DATA REDUCTION NO. OASPL	NO. OCT.	NO. PSD	NO. CSD	NO. COH	NO. STATIC PRESSURE	NO. COOLED TEMP.
1	0.80	24,000	IDLE	12	12				10	10
1	0.80	24,000	MIL	12	12				10	10
2	0.90	15,000	MIL	12	12				10	10
3	1.39	40,000	MIL	12	12				10	10
3	1.39	40,000	MAX	12	12				10	10
4	1.25	30,000	MIL	12	12	12	11*	11*	10	10
4	1.25	30,000	MAX	12	12	12	11*	11*	10	10
5	1.02	20,000	MIL	12	12				10	10
6	1.83	40,000	MIL	12	12				10	10
7	1.52	30,000	MIL	12	12				10	10
8	1.21	20,000	MIL	12	12				10	10

* Referenced to microphone 2.

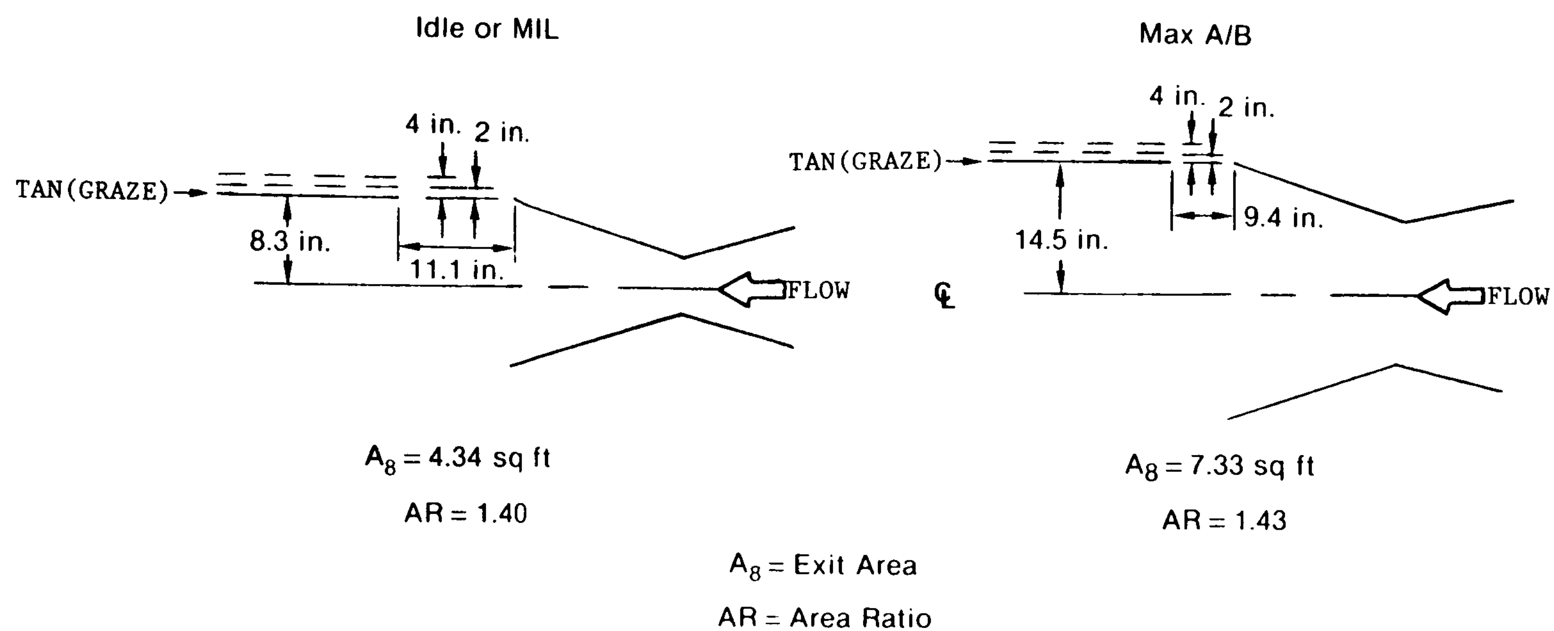
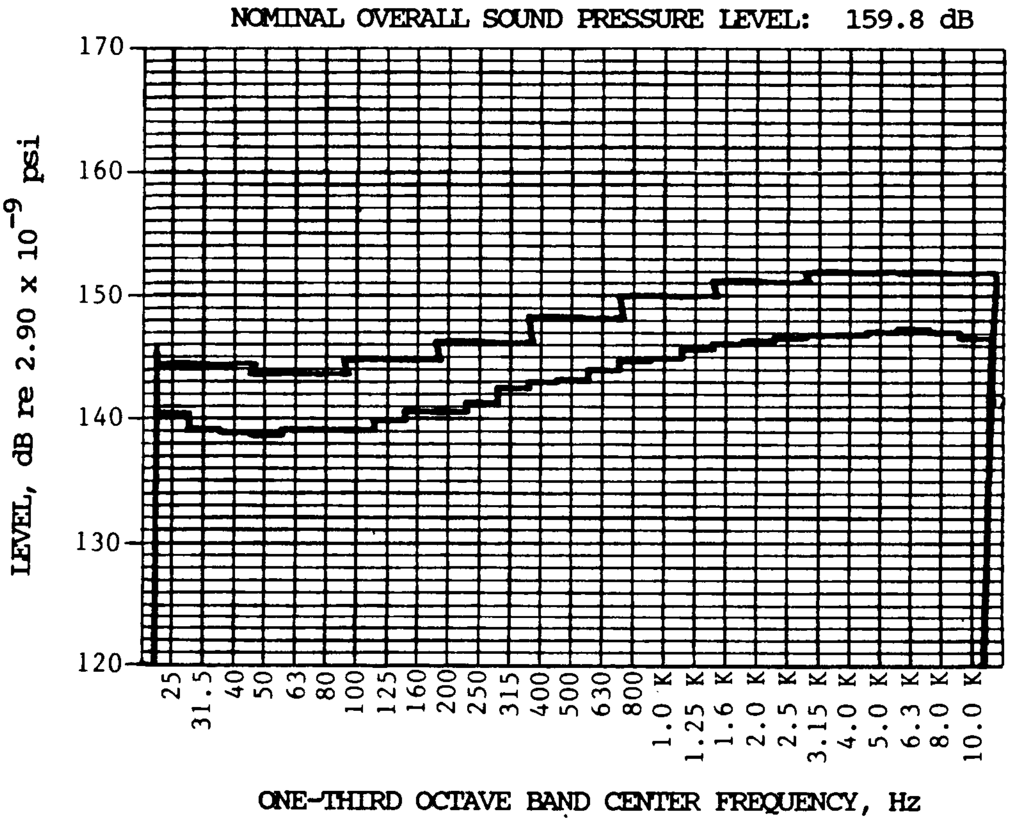


Figure 9. Test Panel Position Relative to Nozzle Flap

Standard. Figures 15 and 16 are the graze panel position power spectral density (PSD) plots from 25 to 2,000 Hz for the same transducers, power settings, and simulated flight condition. Similarly, the associated cross power spectral density (CSD), magnitude and phase, and coherence function (COH) plots are given in Figures 17 and 18 with microphone 2 as reference. The respective OASPL, a summation across the entire frequency range, is also included where applicable.

6.1.1 Octave Band Spectrum Analysis - For Figures 10 through 13, microphone 7 OASPL is approximately 3 dB above that of microphone 2; except for the MAX power and graze condition where it is about 9 dB below microphone 2 OASPL. This could be the result of variations in location and intensity of the complex shock structure within the jet exhaust. Figure 10 exhibits fairly flat continuous spectrums for the MIL power and graze configuration with about a 5 dB variation for either microphone over the 25 to 10,000 Hz frequency range. Microhone 7 is 10 dB above microphone 2 at frequencies below 100, and they are within 3 dB of each other at frequencies above 800 Hz. Nominal OASPL is 159.8 dB for microphone 2 and 163.0 dB for microphone 7.

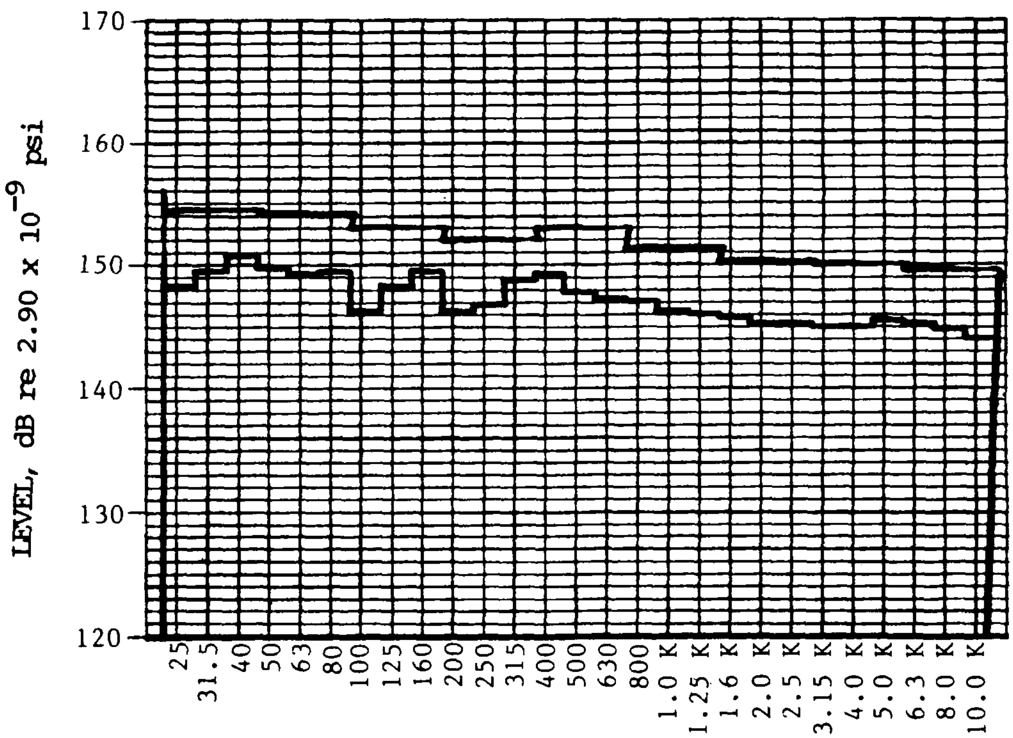
At MIL power and 4-inch panel position, Figure 11, the two spectrums again exhibit similar shape, and the OASPL of microphone 7 remains about 3 dB higher than that of microphone 2. There is about an 11 dB variation for both microphones over the 25 to 10,000 Hz



(a) Microphone 2

Figure 10. Sound Pressure Level; Mach 1.25; 30,000 ft; MIL Power; Graze Panel Position

NOMINAL OVERALL SOUND PRESSURE LEVEL: 163.0 dB



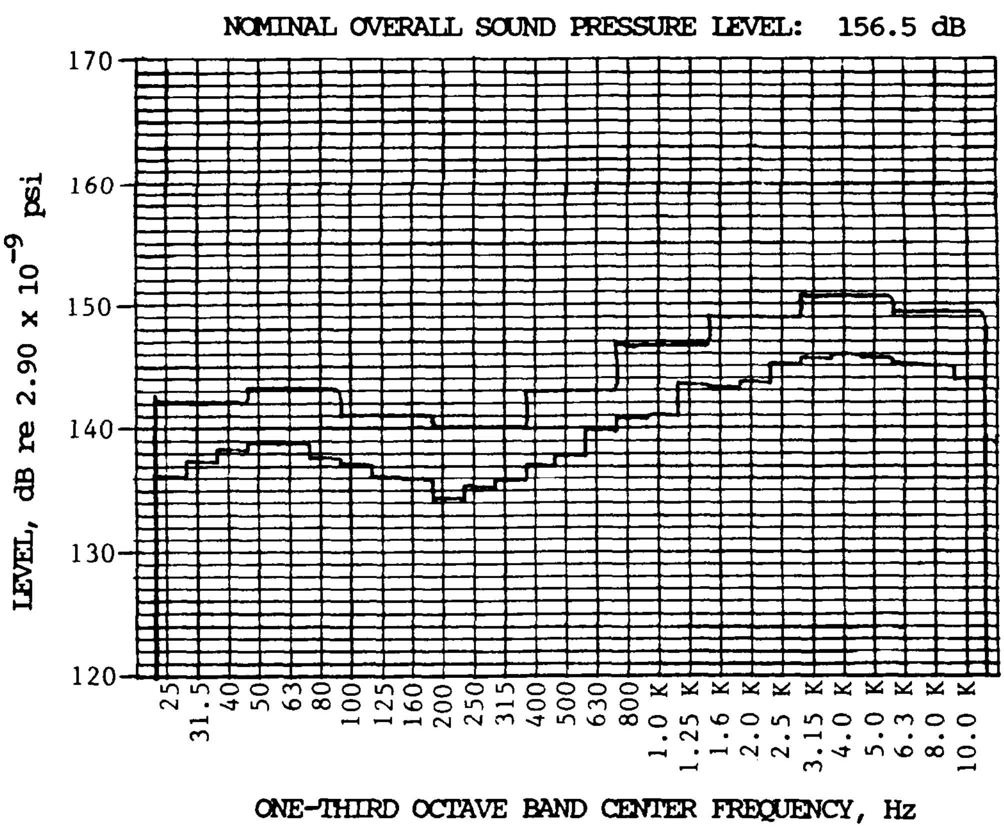
(b) Microphone 7

Figure 10. Concluded

frequency range. Compared to the graze spectrums of Figure 10, microphone 2 decreased as much as 6 dB from 125 to 1,000 Hz, while microphone 7 decreased up to 9 dB in the range below 500 Hz. Microphone 7 showed a small increase of sound pressure level (2 to 4 dB) above 2,000 Hz. OASPL decreased about 3 dB from Figure 10 to 156.5 dB and 160.0 dB for microphones 2 and 7, respectively.

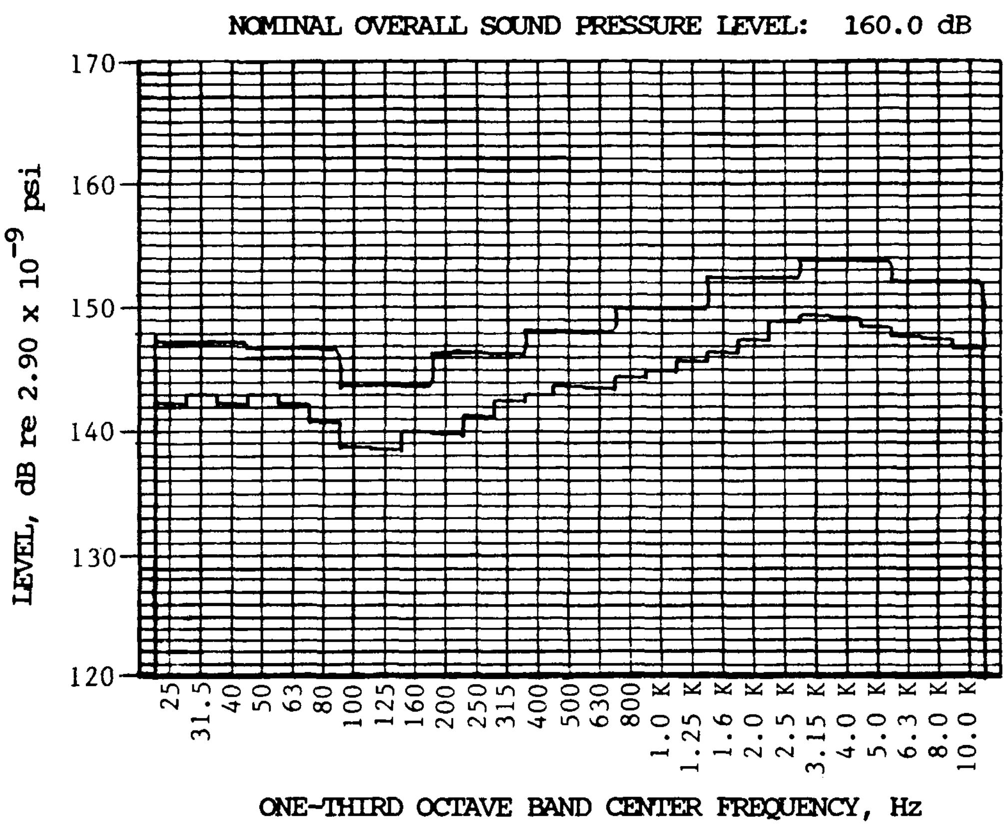
Similarly, the MAX power and graze panel position condition is presented in Figure 12. Microphones 2 and 7 continue to have basically the same spectrum shape. Both microphones have a 10 dB variation across the entire frequency range. The levels of microphone 7, which were generally higher than microphone 2 for MIL power, are now 5 to 10 dB lower. Compared with the MIL power and graze spectrum, Figure 10, microphone 2 levels increase more than 10 dB below 315 Hz and remain essentially unchanged above 315 Hz. The microphone 7 spectrum is well below the corresponding MIL power levels at all frequencies, and the decrease is as much as 16 dB between 160 and 400 Hz. OASPL is 163.2 dB for microphone 2 and 154.5 dB for microphone 7. This is a respective change of +4.4 dB and -8.5 dB relative to Figure 10.

For the MAX power and 4-inch panel position, Figure 13, microphones 2 and 7 have almost identical spectrum distributions and a 20 dB max-to-min variation. Microphone 2 levels vary from 1 to 7 dB below those of microphone 7. The OASPL is 162.0 dB for microphone 2 and 165.8 dB for microphone 7. Compared to Figure 12,



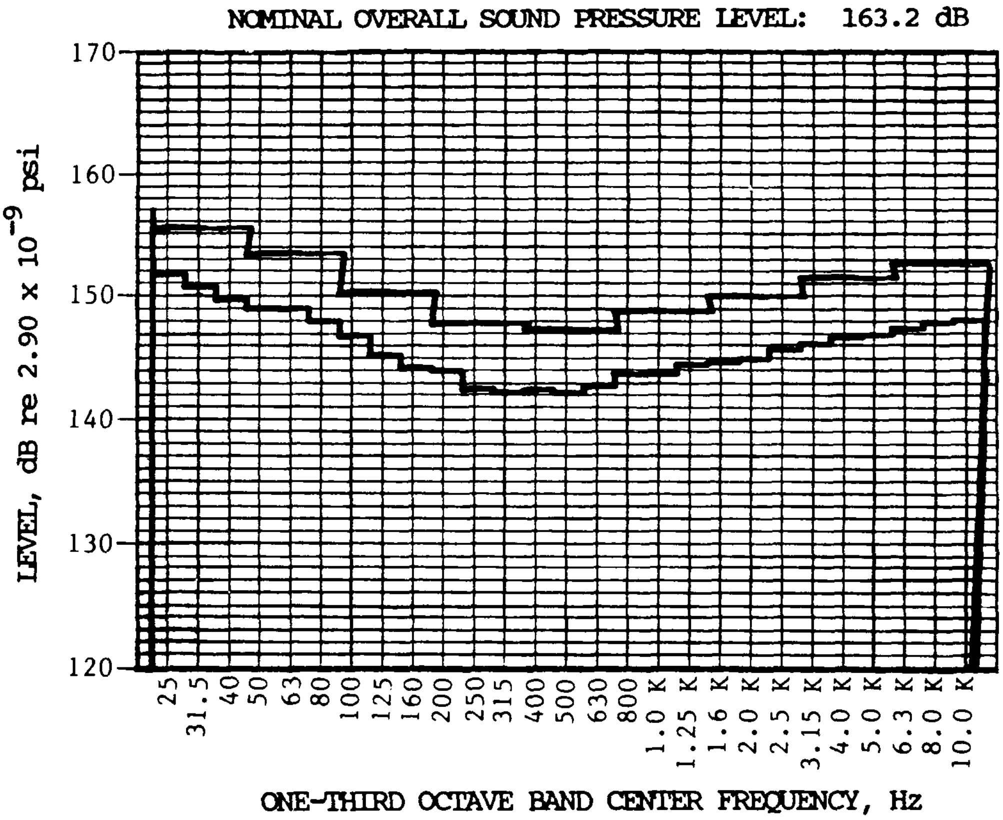
(a) Microphone 2

Figure 11. Sound Pressure Level; Mach 1.25; 30,000 ft; MIL Power; 4-inch Panel Position



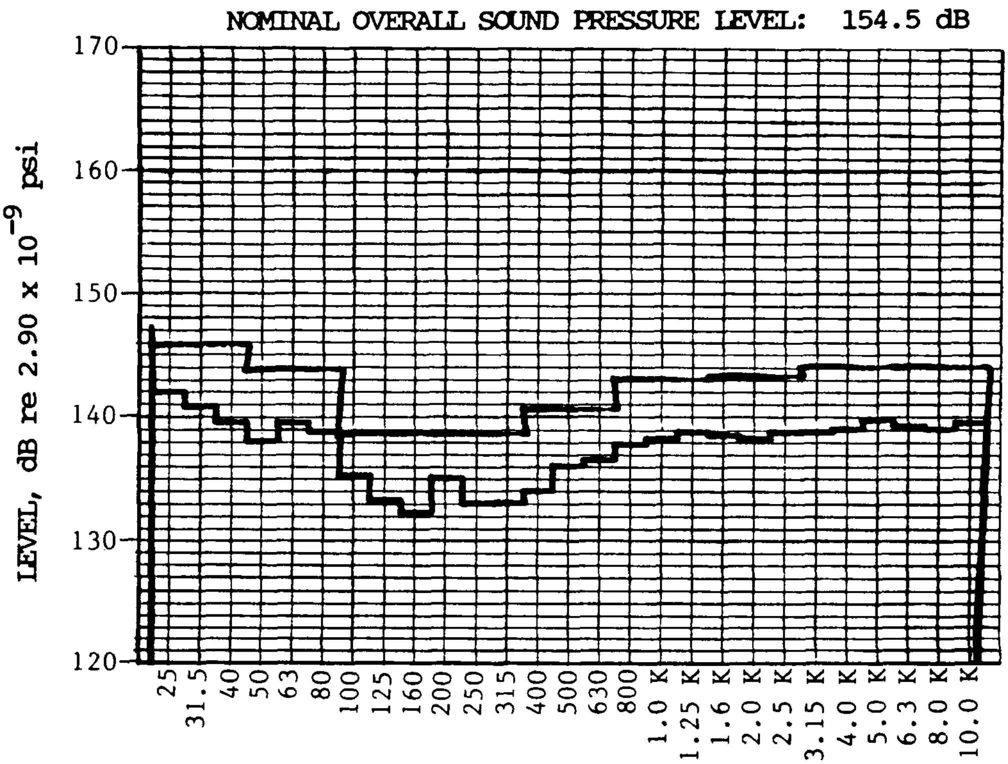
(b) Microphone 7

Figure 11. Concluded



(a) Microphone 2

Figure 12. Sound Pressure Level; Mach 1.25; 30,000 ft; MAX Power; Graze Panel Position



(b) Microphone 7

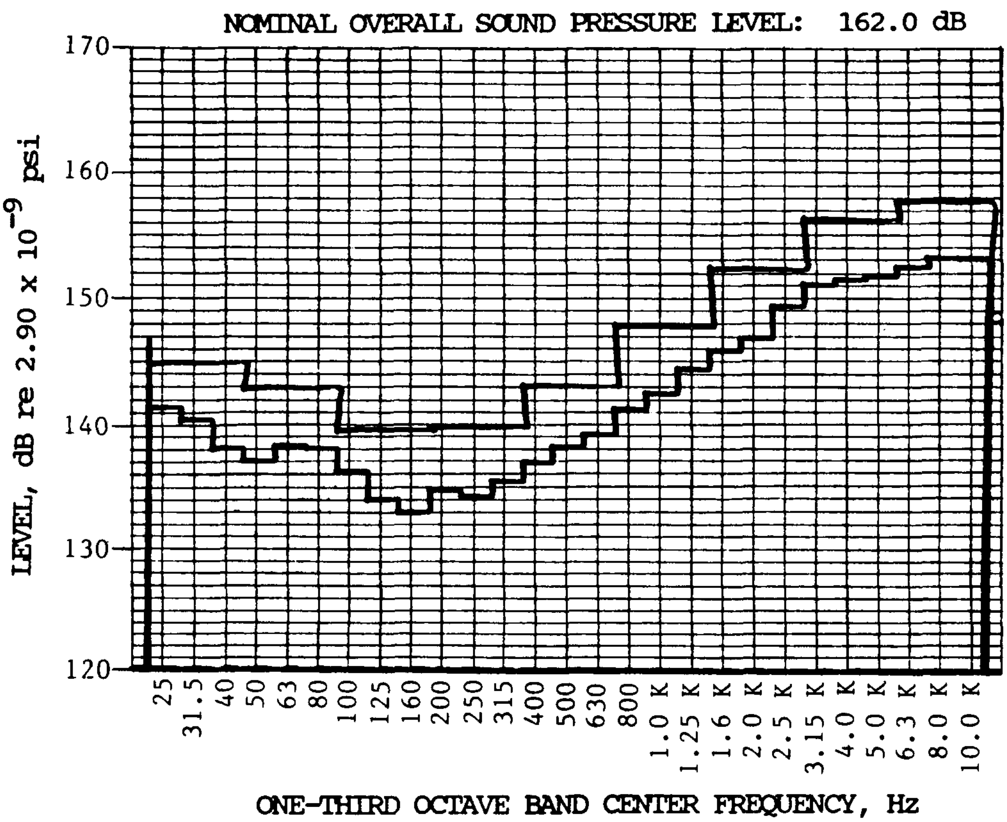
Figure 12. Concluded

moving the panel up 4 inches decreases microphone 2 levels below 1,000 Hz (up to 12 dB below 200 Hz) and increases them above 1,000 Hz (5 dB above 2,500 Hz). Microphone 7 is higher than Figure 12 (up to 15 dB) in all but one frequency band. Compared to Figure 11, an increase from MIL to MAX power results in a 6 to 8 dB level increase above 3150 Hz for both microphones.

Table II presents the nominal OASPL of microphones 2 and 7 for all the test conditions. The points are ordered, for the most part, in ascending Mach number and altitude. The corresponding engine power setting is indicated, and levels are listed for both the graze and 4-inch panel positions. The graze position measurements are presented in Figure 14 for the MIL power engine setting. Neglecting inlet Mach, temperature, and pressure effects, along with typical acoustic experimental scatter, the data show that OASPL decreases as altitude increases.

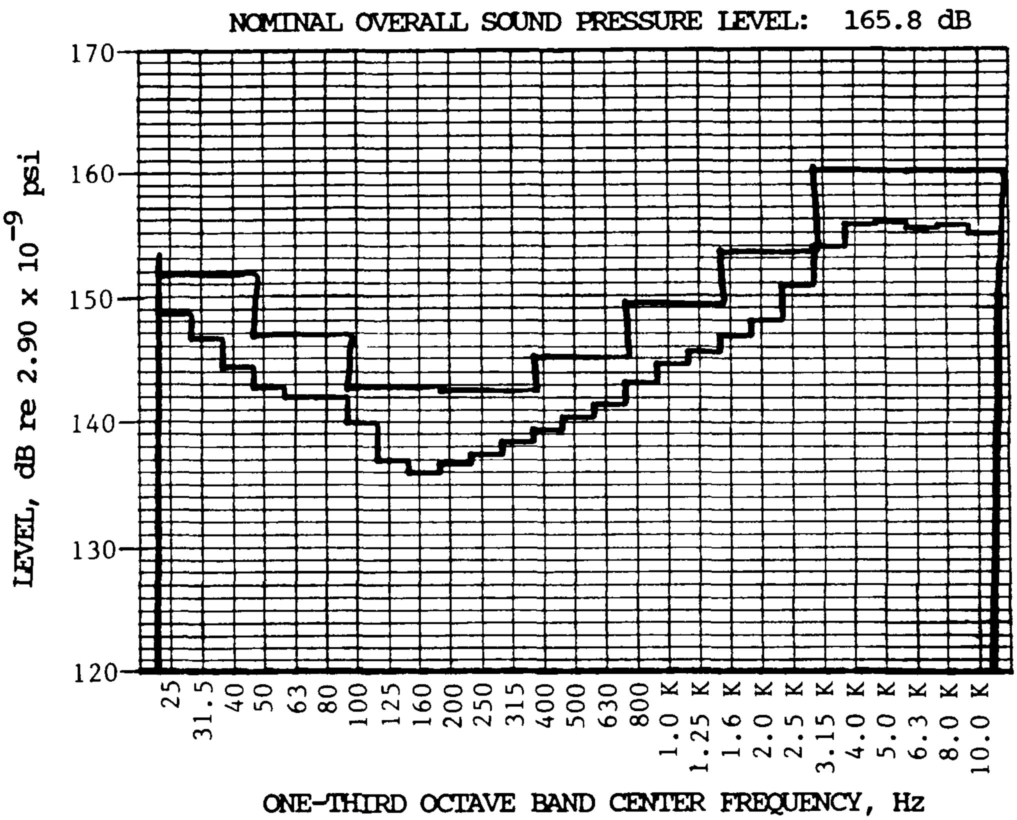
6.1.2 Power Spectral Density Analysis

Figures 15 and 16 are plots of the power spectral density (PSD) analysis for microphones 2 and 7 at the 1.25 Mach number; 30,000 feet altitude; and graze panel position test condition. These provide a sampling of data for both MIL and MAX engine power settings. All PSD analyses were computed for a 25 to 2,000 Hz frequency range. In agreement with the previous Octave Band Spectrum Analysis, no isolated frequency peaks are indicated.



(a) Microphone 2

Figure 13. Sound Pressure Level; Mach 1.25; 30,000 ft; MAX Power; 4-inch Panel Position



(b) Microphone 7

Figure 13. Concluded

TABLE II. OVERALL SOUND PRESSURE LEVEL (OASPL) FOR MICROPHONES 2 AND 7

POINT	MACH NO.	ALT. (ft)	ENGINE POWER	NOMINAL OASPL - dB re 2.90×10^{-9} psi (25 - 10,000 Hz) GRAZE MICROPHONE 2	GRAZE MICROPHONE 7	4-INCH MICROPHONE 2	4-INCH MICROPHONE 7
2	0.90	15,000	MIL	167.0	165.5	162.0	164.1
1	0.80	24,000	IDLE	149.5 148.5	148.2 147.6	151.0 149.5	152.4 151.0
1	0.80	24,000	MIL	163.0 163.3	160.0 161.5	158.5 158.6	161.8 161.5
5	1.02	20,000	MIL	163.5	169.0*	160.8	163.0
8	1.21	20,000	MIL	162.2	165.8	160.4	162.4
4	1.25	30,000	MIL	159.8	163.0	156.5	160.0
4	1.25	30,000	MAX	163.2	154.5	162.0	165.8
7	1.52	30,000	MIL	161.2	158.0	157.0	161.2
3	1.39	40,000	MIL	156.8	153.5	152.8	158.0
3	1.39 1.13	40,000 33,000	MAX MAX	158.8** 162.0**	150.8** 157.0**	159.7** 162.2**	163.5** 164.0**
6	1.83	40,000	MIL	161.0	165.2	158.3	162.5

* Data is questionable.
** Exhaust flow was choked.

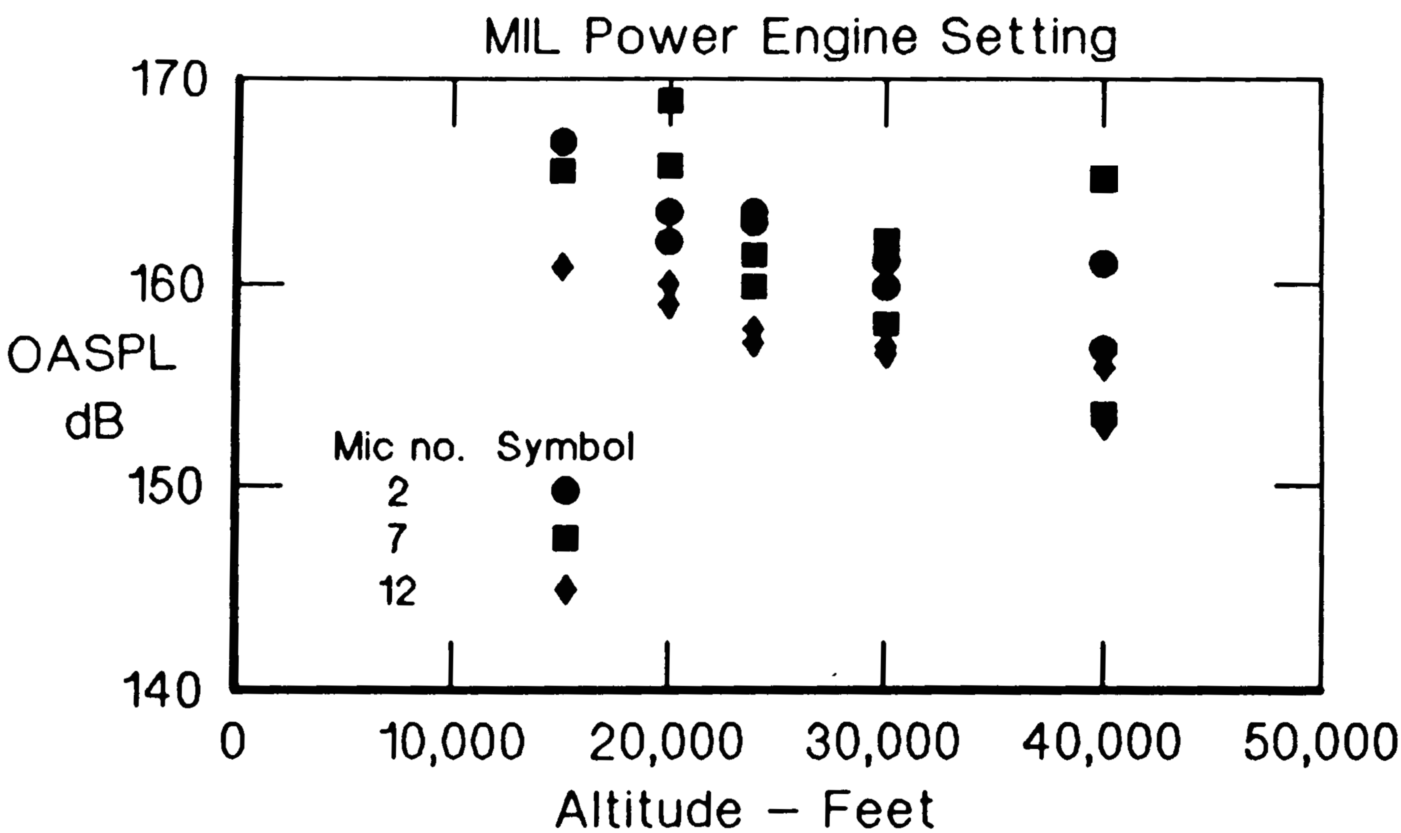


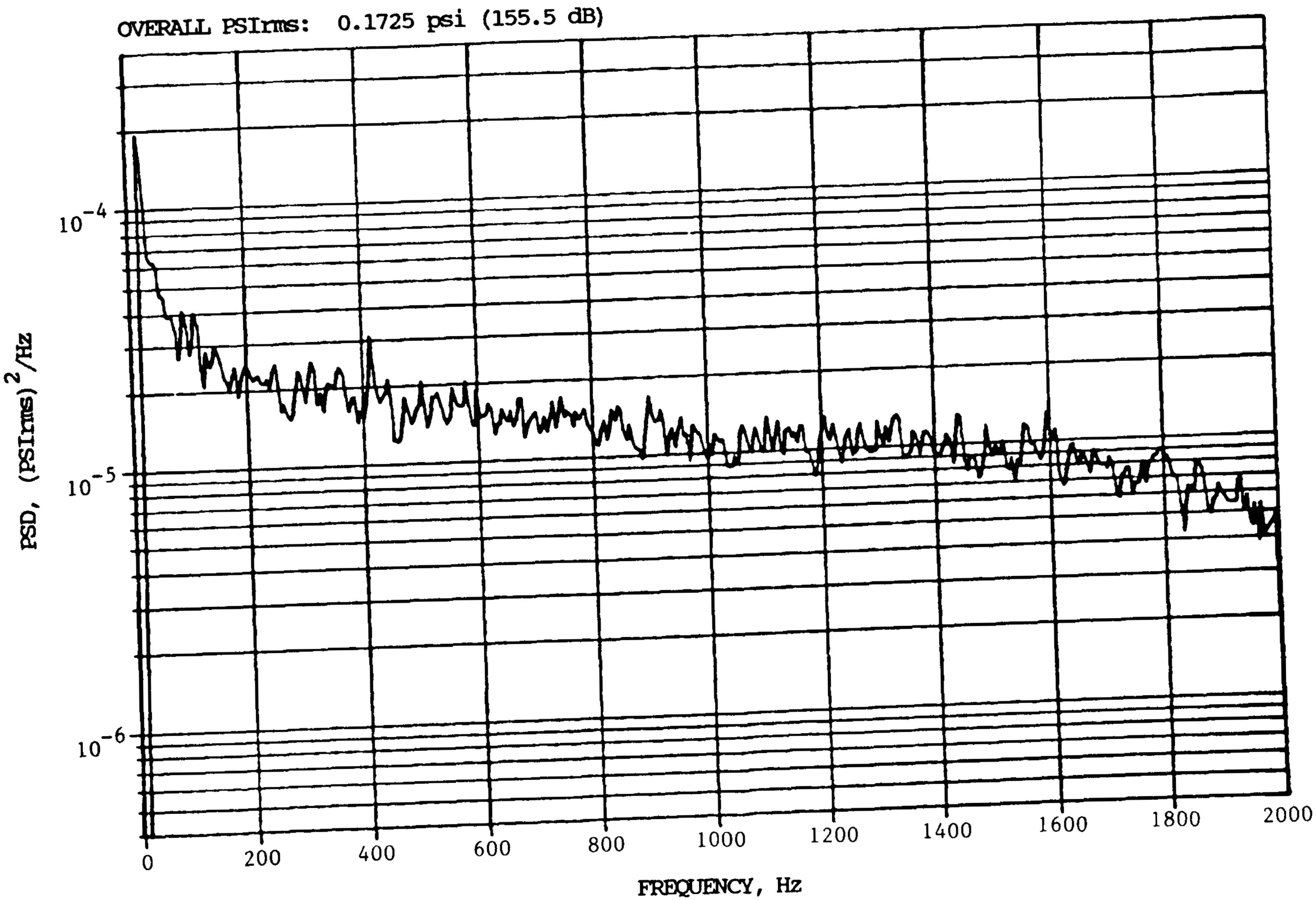
Figure 14. MIL Power Overall Sound Pressure Level (OASPL) vs Altitude; Graze Panel Position

Figure 15 provides data for the MIL power setting. The PSDs of microphones 2 and 7 are both fairly flat and exhibit no strong isolated frequency spikes. Both PSDs are approximately equal above 1,000 Hz. Below 1,000 Hz, microphone 7 levels are higher than microphone 2 which is consistent with the comparison of Figure 10 from the previous section. The overall rms pressure is 0.1725 psi (155.5 dB) and 0.3303 psi (161.1 dB) for microphones 2 and 7, respectively. For this same test condition, the corresponding OASPLs of Figure 10 are 159.8 dB and 163.0 dB from 25 to 10,000 Hz.

MAX power data is presented in Figure 16. Compared to Figure 15, microphone 2 PSD is above the MIL power levels below 250 Hz and about equal above 250 Hz. Microphone 7 PSD levels are well below those of the corresponding MIL power levels for all frequencies. Again, these comparisons are also in agreement with the octave band spectrum analysis of Figure 12, and no strong power content is observed at any frequency. The overall rms pressure is 0.3015 psi (160.3 dB) for microphone 2 and 0.1725 psi (152.0 dB) for microphone 7. Figure 12 OASPLs are 163.2 dB and 154.5 dB.

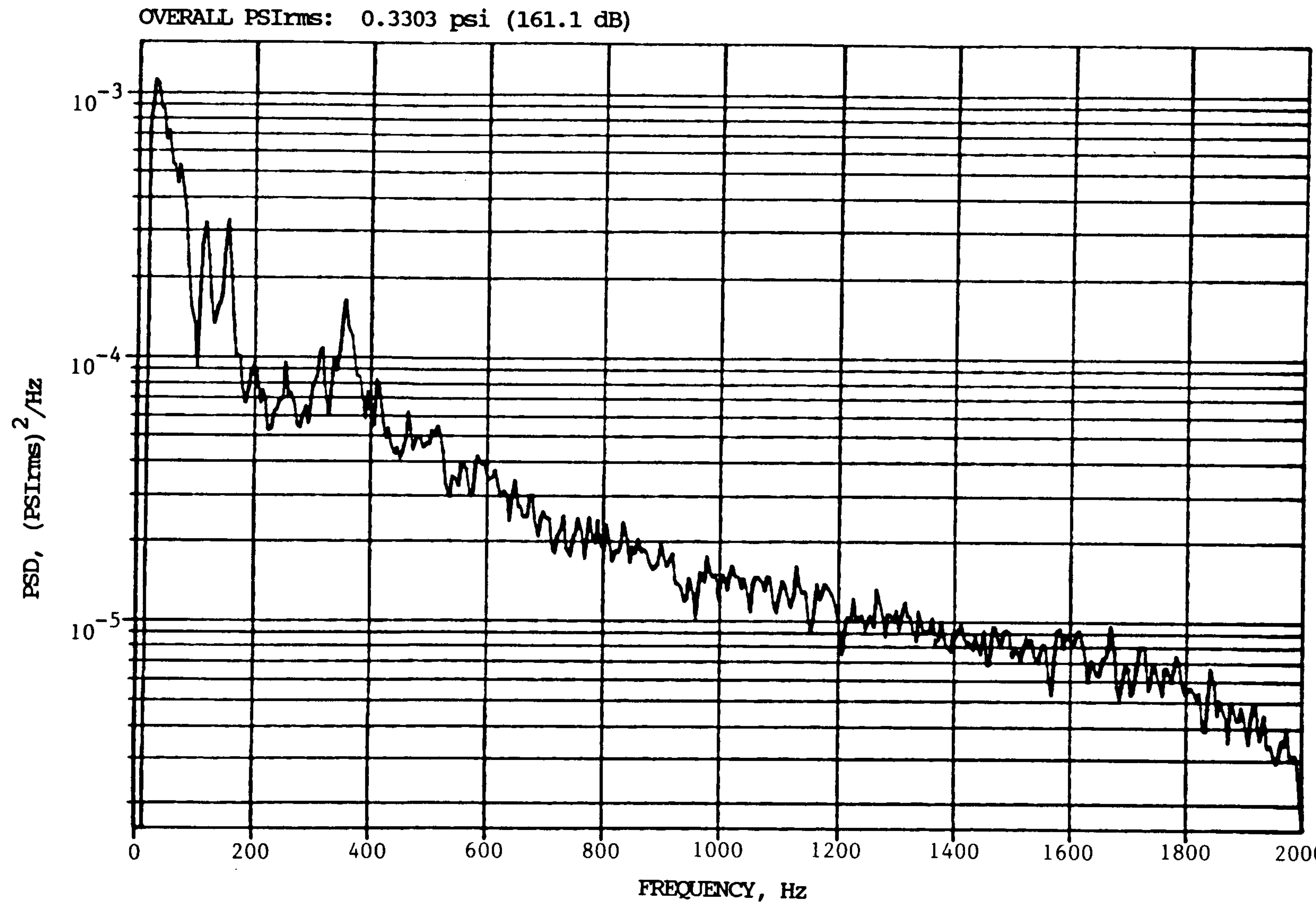
6.1.3 <u>Cross Power Spectral Density and Coherence Analysis</u>

At 1.25 Mach number and 30,000 feet altitude test condition, the microphone 7 signal is compared to microphone 2 for cross power spectral density (CSD) magnitude and phase, Figure 17, and coherence (COH), Figure 18. Both MIL and MAX power engine settings are given,



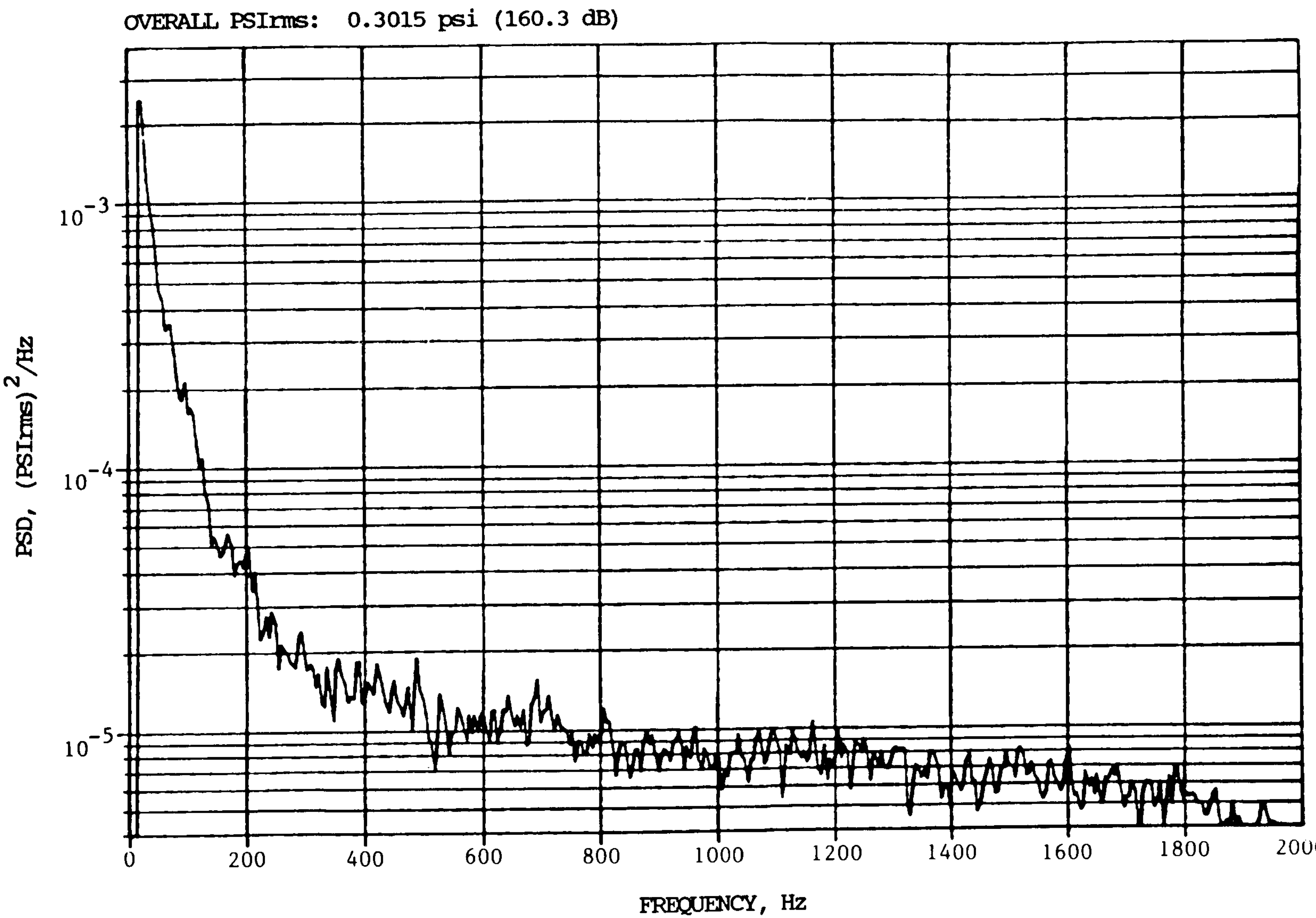
(a) Microphone 2

Figure 15. Power Spectral Density; Mach 1.25; 30,000 ft; MIL Power; Graze Panel Position



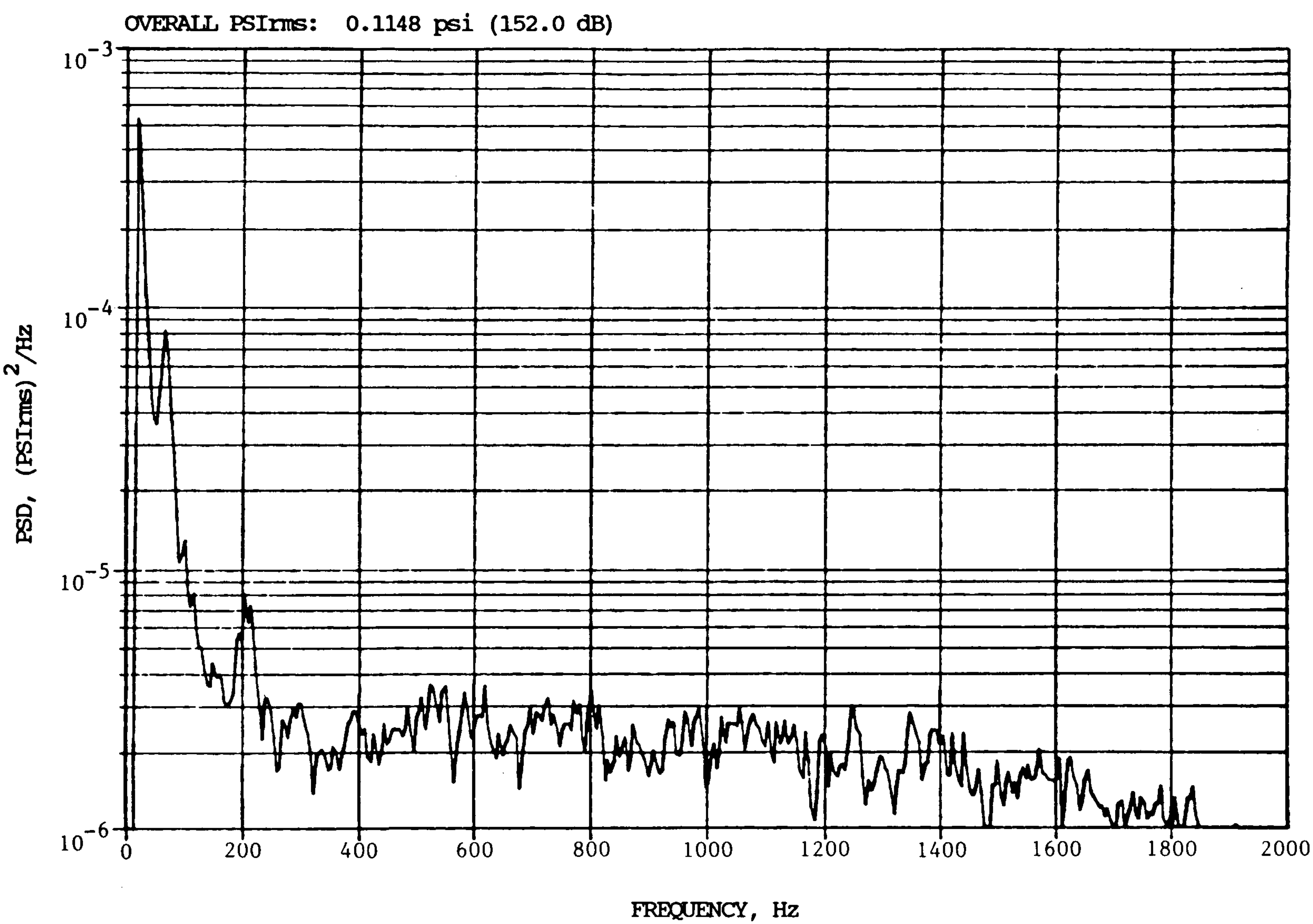
(b) Microphone 7

Figure 15. Concluded



(a) Microphone 2

Figure 16. Power Spectral Density; Mach 1.25; 30,000 ft; MAX Power; Graze Panel Position



(b) Microphone 7

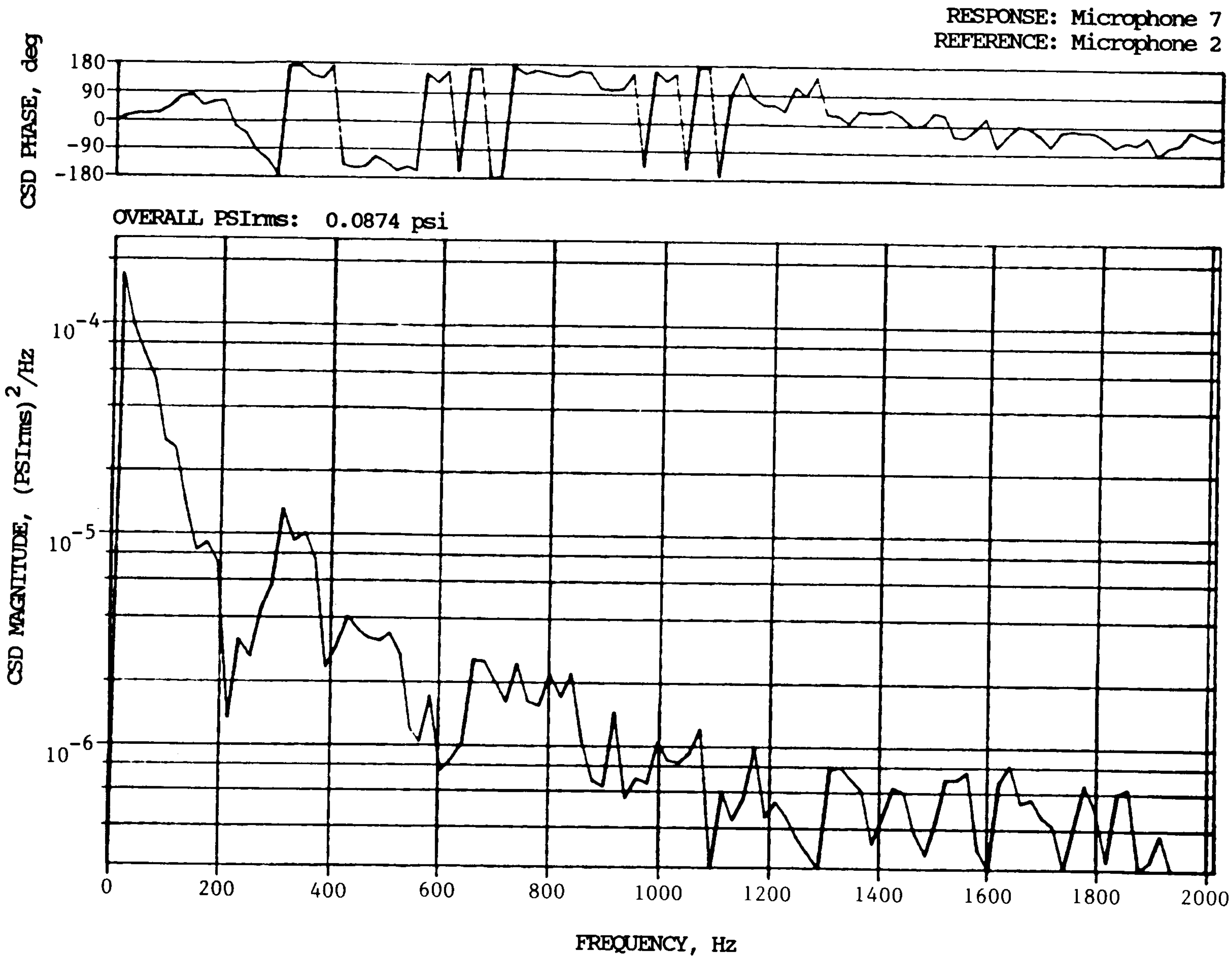
Figure 16. Concluded

and the panel is at the graze position. Little or no correlation is indicated between these two signals throughout the entire range of 25 to 2,000 Hz. Consistent with the previous analysis, MAX power CSD magnitudes are lower than MIL power below 1,000 Hz and about equal above 1,000 Hz.

6.1.4 Panel Distribution of Overall Sound Pressure Level

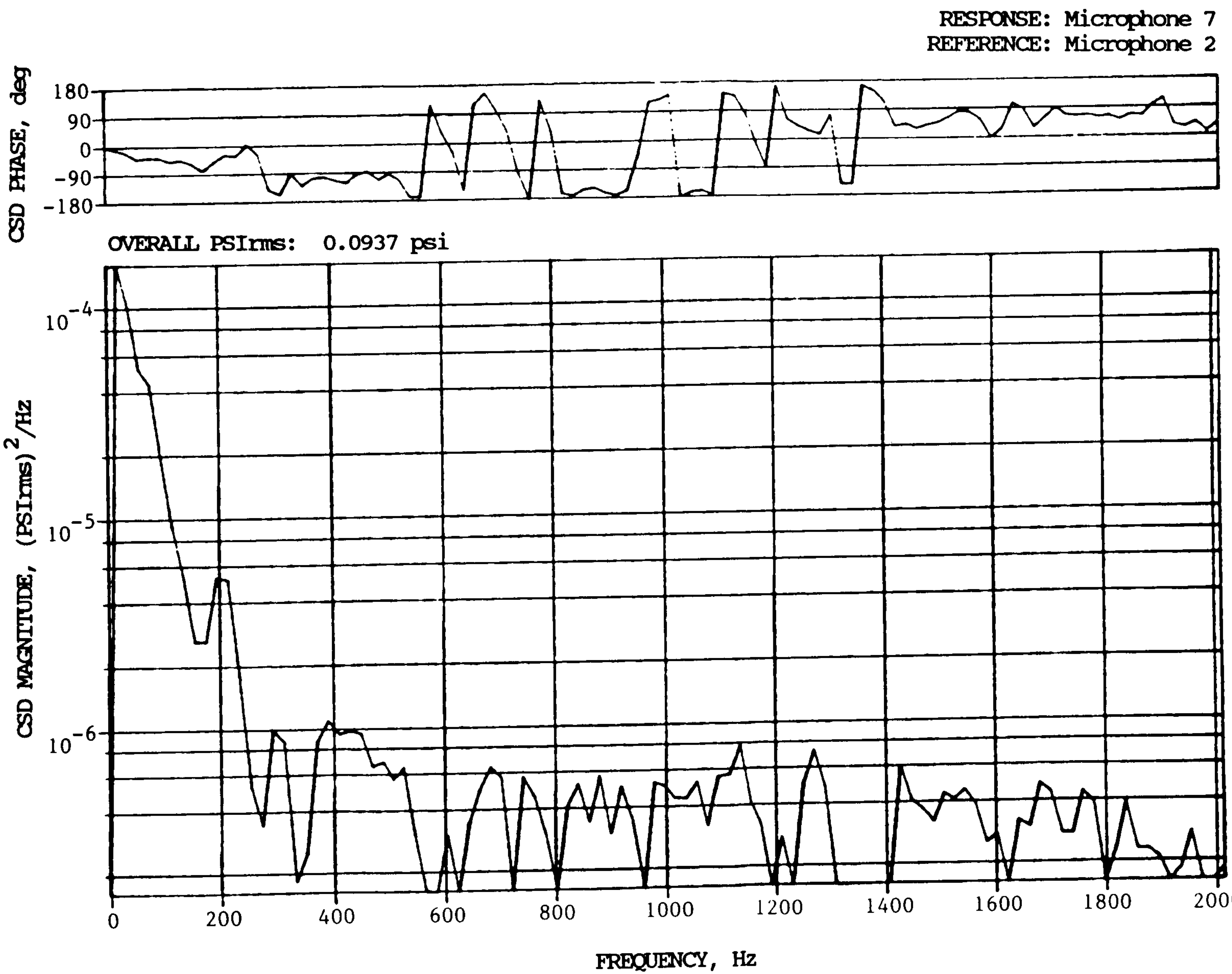
Table III lists the OASPL of all 12 microphones for the 1.25 Mach number and 30,000 feet altitude test condition with the panel at the graze position. Levels are tabulated for both the Octave Band and the PSD Analysis to compare OASPL with and without frequencies above 2,000 Hz. The Octave Band MAX level of microphone 9, 171.8 dB, is questionable because the difference between Octave Band and PSD analysis is 10 dB. The effect of frequencies above 2,000 Hz is no more than 4.5 dB for any other microphone at either engine power setting.

These results indicate that levels increase for the higher engine power setting except for microphones 7, 10, and 11. A better representation of this data may be as given in Figure 19 which also illustrates the relative locations of each microphone. Progressing downstream, MAX levels are higher than MIL levels until microphone 7, where the OASPL is approximately 9 dB below the MIL level. The same is true of microphones 10 and 11, and the levels of microphone 6 are the same for both engine power settings. This may indicate



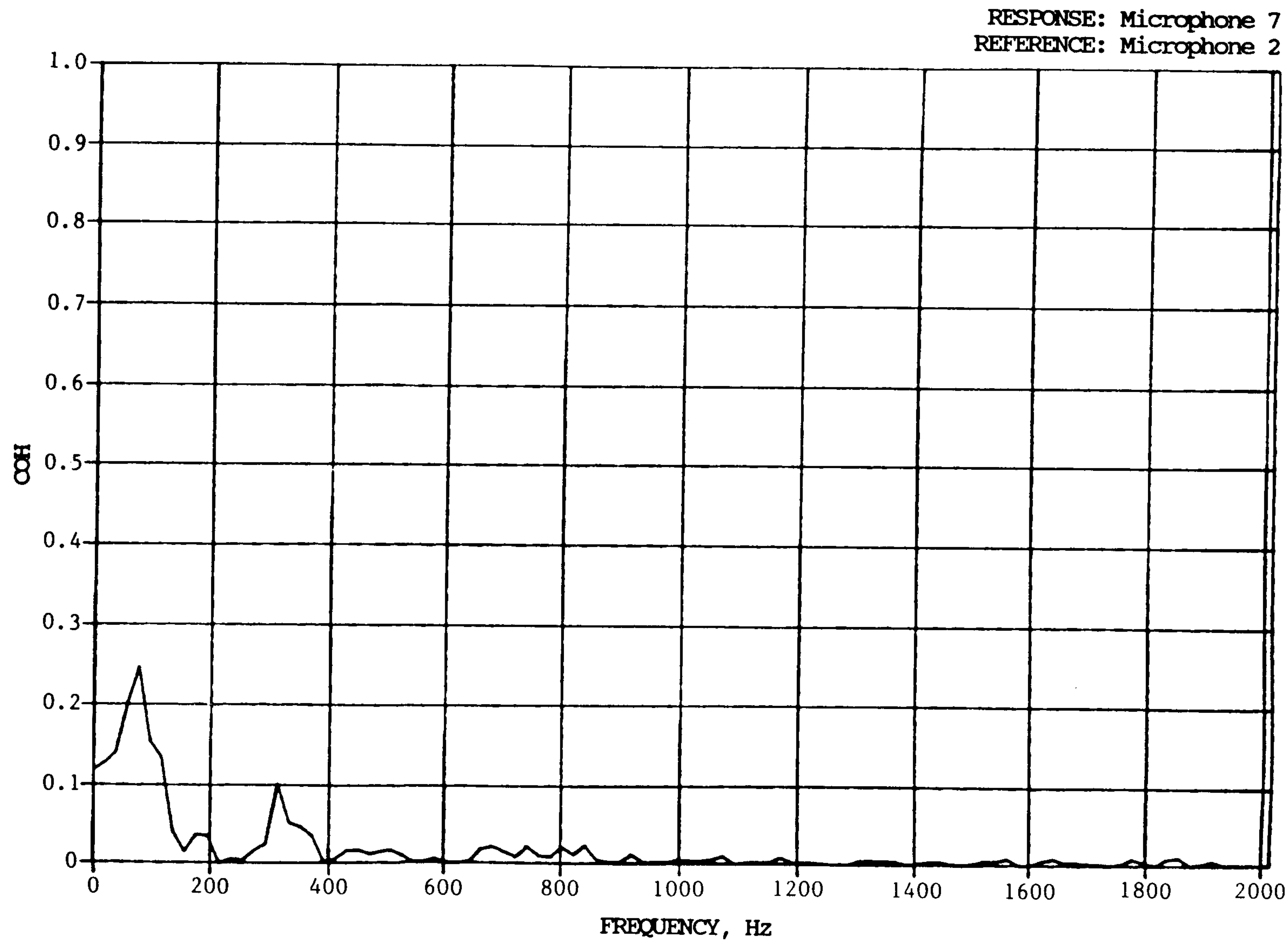
(a) MIL Power

Figure 17. Cross Power Spectral Density of Microphones 7 and 2; Mach 1.25; 30,000 ft; Graze Panel Position



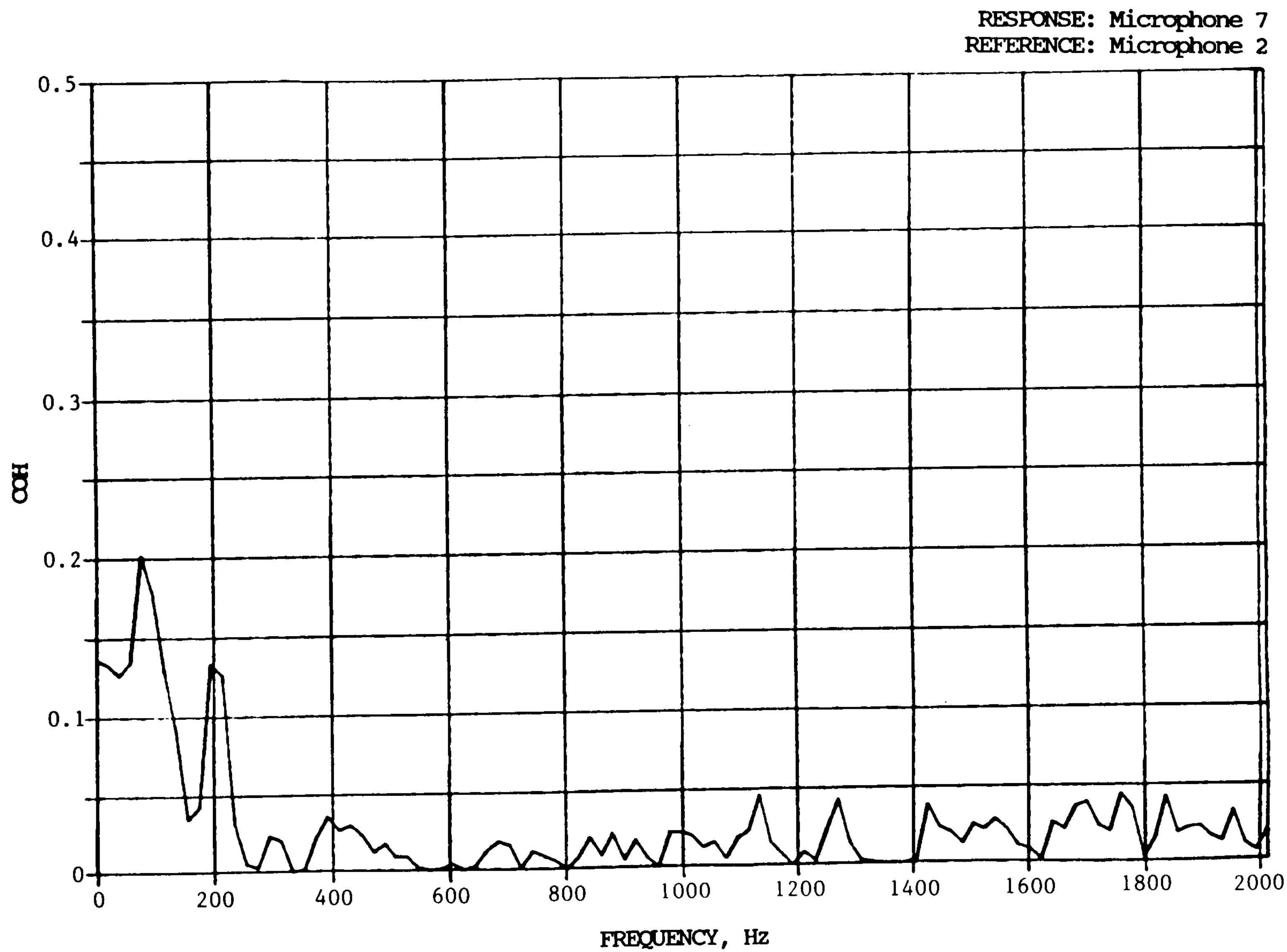
(b) MAX Power

Figure 17. Concluded



(a) MIL Power

Figure 18. Coherence of Microphones 7 and 2; Mach 1.25; 30,000 ft; Graze Panel Position



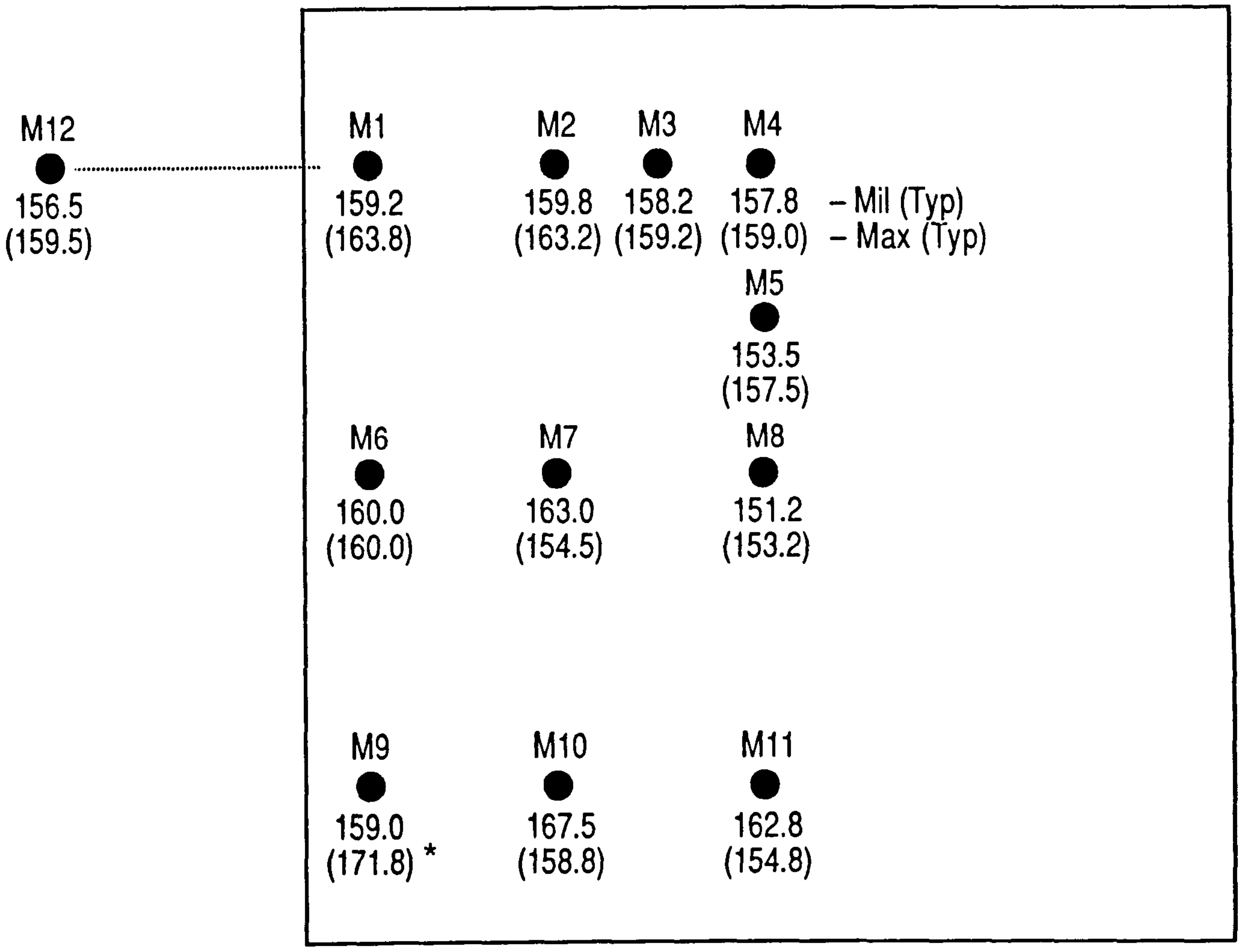
(b) MAX Power

Figure 18. Concluded

TABLE III. OVERALL SOUND PRESSURE LEVEL (OASPL); MACH 1.25; 30,000 ft; GRAZE PANEL POSITION

	NOMINAL OASPL - dB re 2.90×10^{-9} psi			
	OCTAVE BAND ANALYSIS (25 - 10,000 Hz)		PSD ANALYSIS (25 - 2,000 Hz)	
MICROPHONE	MIL POWER	MAX POWER	MIL POWER	MAX POWER
M1	159.2	163.8	155.7	159.3
M2	159.8	163.2	155.5	160.3
M3	158.2	159.2	154.9	156.1
M4	157.8	159.0	155.1	155.7
M5	153.5	157.5	149.9	154.0
M6	160.0	160.0	157.3	155.6
M7	163.0	154.5	161.1	152.0
M8	151.2	153.2	148.5	150.7
M9	159.0	171.8*	156.5	161.8
M10	167.5	158.8	164.7	155.9
M11	162.8	154.8	160.4	150.5
M12	156.5	159.5	154.2	155.2

* Data is questionable.



* Data is questionable.

Figure 19. Test Panel Showing Overall Sound Pressure Level (OASPL) for Each Microphone; Mach 1.25; 30,000 ft; Graze Panel Position

the location of a shock ahead of microphone 7 which is influencing microphones 7, 10, 11, and possibly 6.

Table IV and Figure 20 present the same data for the 1.39 Mach number; 40,000 feet altitude; and graze panel position test condition, except no PSD analysis has been performed. Similarly, these results indicate that levels increase for the higher engine power setting except for microphones 7, 9, and 10. MAX levels are higher than MIL levels until microphone 7, where the OASPL is 2.7 dB below the MIL level. The reduction for microphones 9 and 10 is 2.0 and 4.2 dB, respectively. This could indicate that the shock ahead of microphone 7 is in a slightly different location than for the previous test condition.

6.2 STATIC PRESSURE AND TEMPERATURE

All corresponding static pressure and temperature data for the Mach 1.25 and 30,000 feet altitude test condition described above are tabulated in Table V. These are average values which were recorded through the NASA data system. As noted, data system problems prevented acquiring static pressures for the graze panel position at MIL power. The spatial locations of each pressure port and thermocouple are provided in Figure 4.

The MAX power and graze panel position measurements show that static pressure decreases from about 4.5 psia (leading edge) to 3.1

TABLE IV. OVERALL SOUND PRESSURE LEVEL (OASPL); MACH 1.39; 40,000 ft; GRAZE PANEL POSITION

MICROPHONE	NOMINAL OASPL - dB re 2.90×10^{-9} psi (25 - 10,000 Hz)	
	MIL POWER	MAX POWER
M1	152.0	156.0
M2	156.8	158.8
M3	155.5	155.8
M4	154.8	157.2
M5	151.2	156.0
M6	150.0	151.2
M7	153.5	150.8
M8	147.5	151.5
M9	148.2	146.2
M10	151.0	146.8
M11	146.5	149.0
M12	153.0	157.0

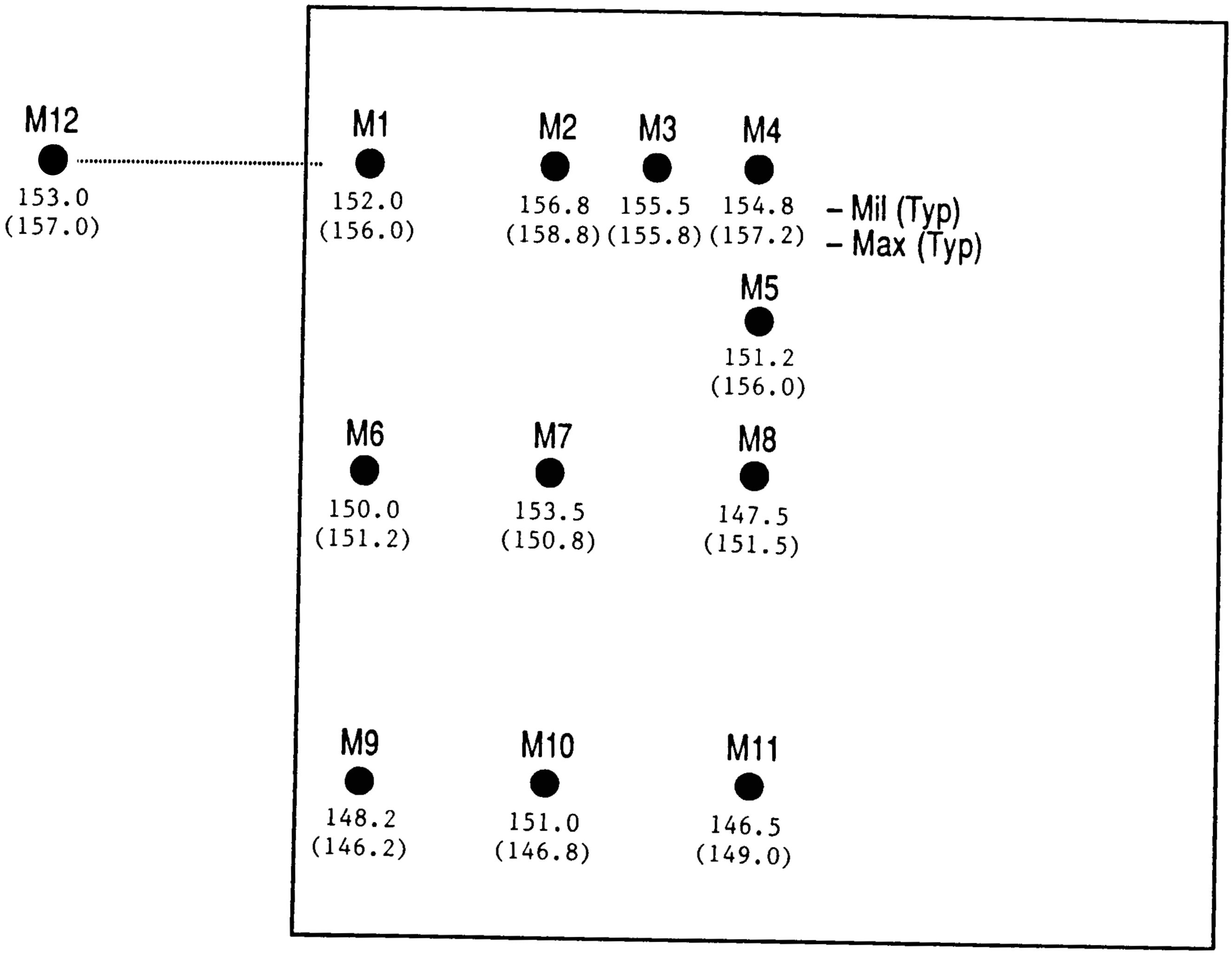


Figure 20. Test Panel Showing Overall Sound Pressure Level (OASPL) for Each Microphone; Mach 1.39; 40,000 ft; Graze Panel Position

TABLE V. STATIC PRESSURE AND TEMPERATURE DATA; MACH 1.25; 30,000 ft

	MIL POWER		MAX POWER	
NAME	GRAZE (16.60-in STROKE)	4-INCH (12.60-in STROKE)	GRAZE (10.26-in STROKE)	4-INCH (6.63-in STROKE)
PANEL STATIC PRESSURE, psia				
S1	Data System Problems, Data Unavailable	4.00	4.15	3.74
S2		4.12	3.22	3.53
S3		4.21	2.58	3.30
S4		4.28	2.56	3.10
S5		4.29	2.71	3.56
S6		4.19	3.38	3.35
S7		4.32	2.13	3.38
S8		4.07	4.80	3.81
S9		4.15	3.25	3.56
S10		4.34	2.54	3.19
PANEL WATER COOLED TEMPERATURE, ^oR (^oF)				
T1	650 (190)	562 (102)	1031 (571)	618 (158)
T2	613 (153)	560 (100)	660 (200)	598 (138)
T3	685 (225)	598 (138)	925 (465)	655 (195)
T4	673 (213)	569 (109)	1024 (564)	624 (164)
T5	709 (249)	587 (127)	978 (518)	658 (198)
T6	735 (275)	618 (158)	915 (455)	690 (230)
T7	655 (195)	609 (149)	774 (314)	695 (235)
T8	625 (165)	623 (163)	741 (281)	686 (226)
T9	639 (179)	616 (156)	865 (405)	709 (249)
T10	629 (169)	617 (157)	794 (334)	697 (237)

Test Cell Conditions:	
Altitude, ft	29900
Flight Mach Number	1.26
Engine Inlet Temperature, ^oR (^oF)	551 (91)
Engine Inlet Pressure, psia	11.33
Nozzle Back Pressure, psia	4.39

Engine Conditions:	MIL Power	MAX Power
N1 Rotor Speed, rpm	9610	9230
N2 Rotor Speed, rpm	12770	12660
Nozzle Pressure Ratio	5.92	5.82
Nozzle Area Ratio	1.40	1.44
Nozzle Throat Area, ft^2	3.15	5.13
Nozzle Exit Area, ft^2	4.42	7.37

psia (center) and 2.5 psia (trailing edge). Moving the panel up 4 inches tends to evenly distribute the pressure. Levels are approximately 4.2 psia for MIL and 3.5 psia for MAX engine power setting.

Note that the temperatures are water cooled values and therefore, not an actual measurement of the exhaust flow. In general, temperatures fluctuated considerably for the MAX power and graze condition, and were fairly evenly distributed for the other three conditions.

7.0 CONCLUSIONS

Considering Mach number and altitude conditions, panel positions, and engine power settings, a total of 39 test points were requested by MCAIR. The 0.8 Mach number and 24,000 feet test condition was required by NASA and P&W. All of the intermediate power test conditions were obtained, but only 44% of the augmented test conditions were achieved. Thus, 69% of the requested test was completed. Engine and nozzle flap liner problems were responsible for terminating the test prior to completion.

On site octave band spectrum analyses were performed for all of the data to gain confidence that it was correct. As expected, a significant trend observed during the test was the reduction in overall sound pressure level with increasing altitude for all power settings tested. The limited measurements obtained during a preliminary test at MCAIR using an F100 engine indicated the data are reasonable and valid. Reference 1 contains a good summary of this preliminary test. Acoustic data consistency was shown between the octave band and power spectral density analyses. Substantial pressure level across the entire frequency spectrum indicates that the exhaust environment may excite structural resonances as high as 10,000 Hz.

REFERENCES

1. Brase, L. O., "Near Field Exhaust Environment Measurements of a Full Scale Afterburning Jet Engine with Two-Dimensional Nozzle," AIAA Paper No. 88-0182, January 1988.

BIBLIOGRAPHY

Strong, D. R., Siddon, T. E., and Chu, W. T., "Pressure Fluctuations on a Flat Plate with Oblique Jet Impingement," NASA CR-839, July 1967.

Grosche, F. R. (D. G. Randall, transl.), "On the Generation of Sound Resulting From the Passage of a Turbulent Air Jet Over a Flat Plate of Finite Dimensions," Library Trans. No. 1460, Brit. R.A.E., October 1970.

Hayden, R. E., "Noise From Interaction of Flow With Rigid Surfaces: A Review of Current Status of Prediction Techniques," NASA CR-2126, 1972.

Clark, L. R., and Yu, J. C., "Effects of Geometry and Jet Velocity on Noise Associated with an Upper-Surface-Blowing Model," NASA TN D-8386, March 1977.

Hoch, R. G., "Some Thoughts on the Effects of Flight on Jet Noise as Observed in Actual Flight and in Wind Tunnels," Journal of Sound & Vibration, October 1980, pp. 379-389.

Michalke, A., "Prediction of Jet Noise in Flight from Static Tests," Journal of Sound & Vibration, December 1979, pp. 341-367.

Larson, R. S., McColgan, C. J., "Jet Noise Source Modification Due to Forward Flight," AIAA Journal, March 1978, pp. 225-232.

Seiner, J. M., "Advances in High Speed Jet Aeroacoustics," AIAA Paper No. 84-2275, October 1984.

Hariharan, S. I., "Numerical Solutions of Acoustic Wave Propagation Problems Using Euler Computations," AIAA Paper No. 84-2290, October 1984.

Bryce, W. D., "The Prediction of Static-to-Flight Changes in Jet Noise," AIAA Paper No. 84-2358, October 1984.

Buckley, R. and Morfey, C. L., "Scaling Laws for Jet Mixing Noise in Simulated Flight," AIAA Paper No. 84-2360, October 1984.

Lavin, S. P., Ho, P. Y., and Chamberlin, R., "Measurements and Predictions of Energy Efficient Engine Noise," AIAA Paper No. 84-2284, October 1984.

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16. Abstract

A test to measure the acoustic noise and static pressure environment on a structure exposed to engine exhaust flow was conducted at the NASA Lewis Research Center Propulsion Systems Laboratory by using an F100 derivative engine with a two-dimensional convergent-divergent (2D/CD) non-flight-weight demonstrator nozzle. A highly instrumented, water-cooled flat panel was placed behind the 2D/CD nozzle and tests were conducted at simulated Mach/altitude flight conditions with the engine at military or maximum-afterburner power setting. The panel instrumentation consisted of acoustic pressure microphones, thermocouples, and static-pressure pickups. Measurements indicated that exhaust environment may excite structural resonances up to 10000 HZ and that overall sound pressure levels decrease with increasing altitude.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
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