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HIGH-ALTITUDE MANEUVER CONTROL TESTS IN THE NSWC HYPERVELOCITY WIND TUNNEL

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STRATEGIC SYSTEMS DEPARTMENT

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Wind Tunnel	Bi-Conic	Pressure Distribution

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Pressure tests were conducted in the NSWC Hypervelocity Wind Tunnel to obtain aerodynamic data on high-performance missile interceptor vehicles which are controlled by large jets issuing laterally from the vehicle. Reproduction of flight Reynolds number was of particular interest since for large lateral jets extensive separation can occur. The tests were conducted at nominal Reynolds numbers of 2.0, 3.2 and 4.6 million per ft., and at Mach numbers of 10 and 14. Model orientations included angles of attack from -15 to +15 deg. and vehicle roll angles of 0, 45 and 90 deg.

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FOREWORD

Pressure tests were conducted in the NSWC Hypervelocity Wind Tunnel to obtain aerodynamic data on high-performance missile interceptor vehicles which are controlled by large jets issuing laterally from the vehicle. Reproduction of flight Reynolds number was of particular interest since for large lateral jets extensive separation can occur. The tests were conducted at nominal Reynolds numbers of 2.0, 3.2 and 4.6 million per ft., and at Mach numbers of 10 and 14. Model orientations included angles of attack from -15 to +15 deg. and vehicle roll angles of 0, 45 and 90 deg.

The tests provided surface pressure distributions over the 112 orifice locations and shock shapes and separation data from the Schlieren photographs. These results constitute an advancement over past tests and provide guidance in calculating jet penetration, jet spreading and separation on high-altitude maneuver control vehicles and in calculating 3-D effects of jets on surface pressure.

Special acknowledgements are extended to the McDonnell Douglas Astronautics Company, in particular to Donald W. Harvey for his assistance as the program coordinator, and to the Hypervelocity Tunnel personnel, in particular E. Bruce Watts, J. P. Johnson, Mary Ellen Falusi, Stephen Cothran, Jr., and Mark M. Opeka, for their tireless efforts to make this a successful test program.

Approved by:

C.A. Fisher

C. A. FISHER, Head
Weapon Dynamics Division

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INTRODUCTION

High-altitude homing missile interceptor vehicles can be controlled by lateral propulsive impulse. The technical uncertainties associated with large lateral jets in a high-altitude maneuver control system include jet-induced pressure field effects on the body lift forces, the body center of pressure, and laminar boundary layer separation over the optical sensor window.

Tests were sponsored by the Ballistic Missile Defense Advanced Technology Center under contract (Contract no. DASG60-81-C-0068) to the McDonnell Douglas Astronautics Company West, to obtain experimental data on the effects experienced by high-altitude homing interceptors.[1] The data will be used to develop analytical models to aid in the design of interceptor maneuver control systems and to develop novel approaches to the technical uncertainties typical of the systems. Photographic and pressure tests of the interceptor vehicle were conducted by the Naval Surface Weapons Center in the Hypervelocity Wind Tunnel. Mach numbers and Reynolds numbers of the flight regime were fully duplicated for the high-altitude environment under consideration. The experimental surface pressure fields and the flow fields obtained photographically during the test can be geometrically scaled up to flight vehicle size, including boundary layer separation. The present test was planned as an extension of previous work, while increasing the jet size by more than one hundred percent.

The biconic blunt-nose body tested housed a lateral thruster designed to accept six interchangeable nozzle contour blocks and housed 112 pressure transducers to service the surface pressure orifices distributed over the entire model.

The test matrix included twenty-five runs. Variation from run to run included the following: Mach number, Reynolds number, jet blowing parameter, nozzle geometry and model bank angle. Two runs were conducted at Mach 10 and 14 to study blockage effects due to the jet and model. The model was not pitched during these two runs, however, the jet pressures were varied from zero up to the maximum allowed pressure. The remainder of the runs involved pitch sweeps from as much as -15 deg. to + 15 deg.

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[1]Harvey, D. W., "High Altitude Maneuver Control Preliminary Test Plan," McDonnell Douglas Astronautics Company West, Aug 1981.

FACILITY

The Naval Surface Weapons Center Hypervelocity Wind Tunnel, shown in Figure 1, has a five ft. diameter test section and operates at Mach numbers of 10 and 14 with Reynolds numbers up to 5 million per ft. The large test cell can accommodate models up to six ft. in length and eighteen in. in diameter. For the present test program the large test cell allowed for testing of a model with a large jet plume and pitching over a thirty deg. angle range. The facility has useful run times of 15 sec. or less depending on the desired Reynolds number.[2]

In the operation of the Hypervelocity Wind Tunnel a fixed quantity of nitrogen is heated in the tunnel heater vessel prior to each run. The blowdown operation of the tunnel begins with the bursting of a set of metal diaphrams located just upstream of the nozzle inlet. Control valves open and allow cold gas from several storage vessels to flow into the tunnel heater vessel in order to maintain a constant pressure during the period of data aquisition. Tunnel supply pressure and temperature build up gradually to the desired running levels over a start-up period of about 0.6 sec., as during the high Reynolds number operation, illustrated in Figure 2. The start-up period is followed by about 0.7 sec. of useful run time with nearly constant tunnel supply conditions.

The tunnel supply temperature begins to fall when all the hot gas has been expelled from the heater vessel, and the control valves are closed at this time. This ends the period of useful tunnel operation and begins the shutdown sequence.

Data recording is started after the tunnel supply pressure drops at the end of the run, as seen in Figure 2. During this test program, data were recorded at a rate of 100 samples per sec.

TEST HARDWARE

The geometry of the biconic blunt-nose model used in the test program is illustrated in Figure 3 and a photograph of the model is shown in Figure 4. The thruster was located in the solid aluminum centerbody and contained interchangeable nozzle blocks. Nitrogen was supplied to the thruster through the sting. The thruster supply pressure was monitored by pressure transducers in the sting and in the thrust plenum. A total of 112 surface pressure orifices were located at twenty-two stations along the model as shown in Figure 5.

The model geometry was invariable except for the thruster nozzle. Six interchangeable nozzle blocks of differing throat diameters and area ratios were provided to satisfy test requirements. These nozzle blocks are shown in Figure 6. The centerbody section was designed to interface with the model

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[2]Hill, J. A. F., Wardlaw, A. B., Jr., Pronchick, S. W., and Holmes, J. E., "Verification Tests in Mach 14 Nozzle of the Hypervelocity Tunnel at NSWC (White Oak Lab)," AIAA Paper 77-150, Jan 1977.

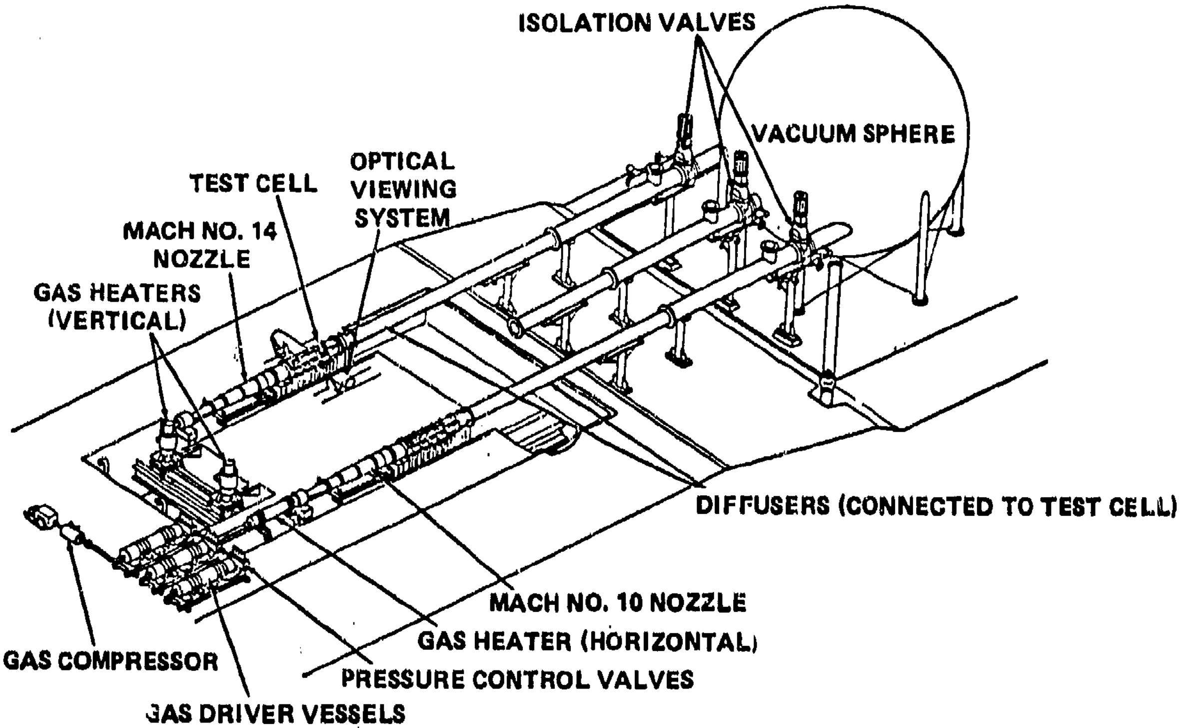

FIGURE 1. NSWC HYPERVELOCITY WIND TUNNEL

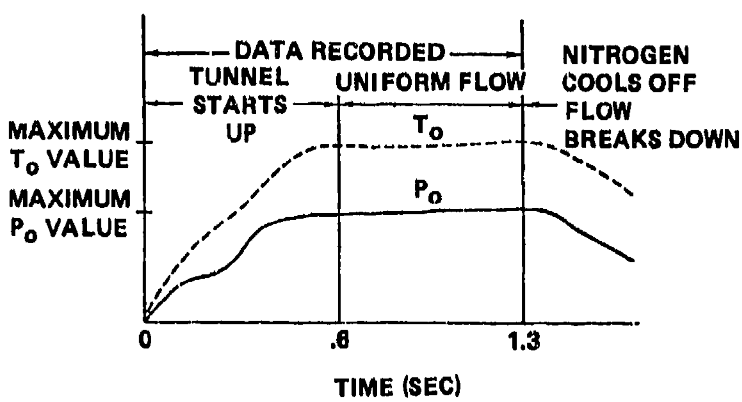

FIGURE 2. TYPICAL WIND TUNNEL SUPPLY CONDITIONS FOR HIGH REYNOLDS NUMBER SIMULATION

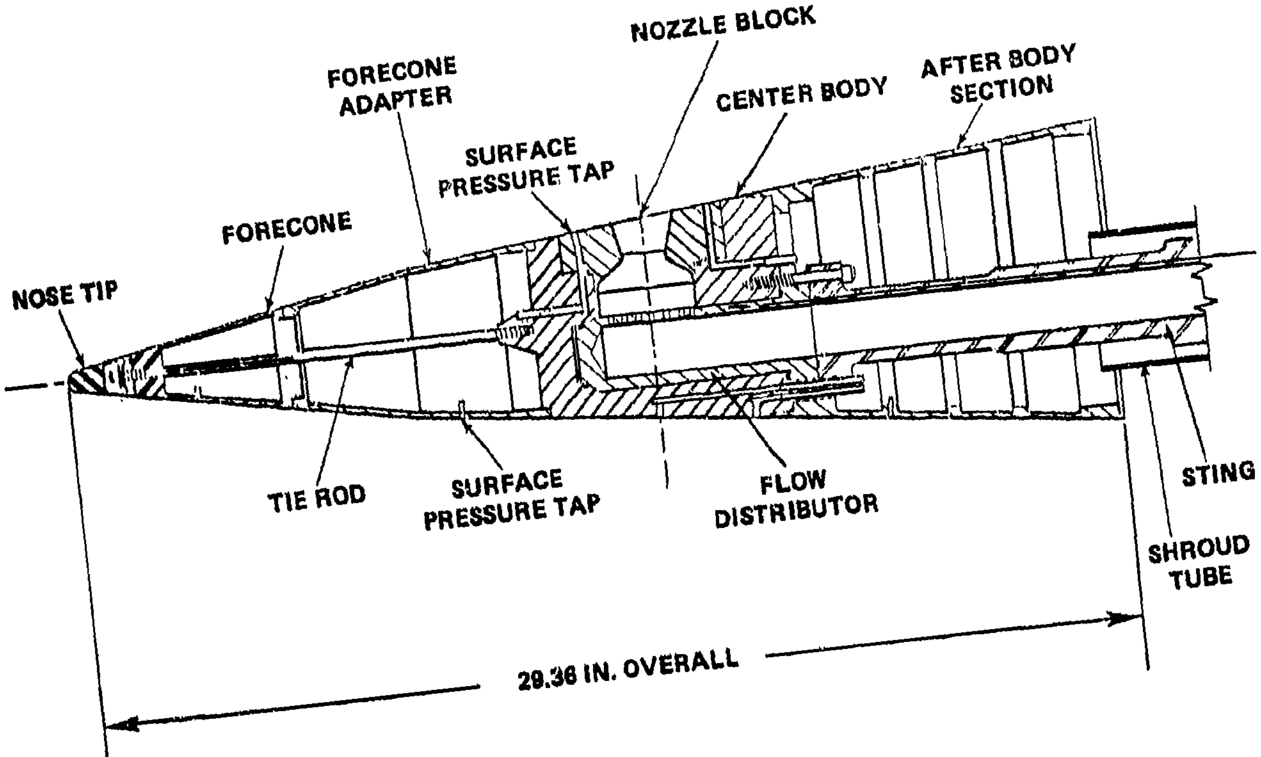

FIGURE 3. HIGH ALTITUDE MANEUVER CONTROL WIND TUNNEL MODEL

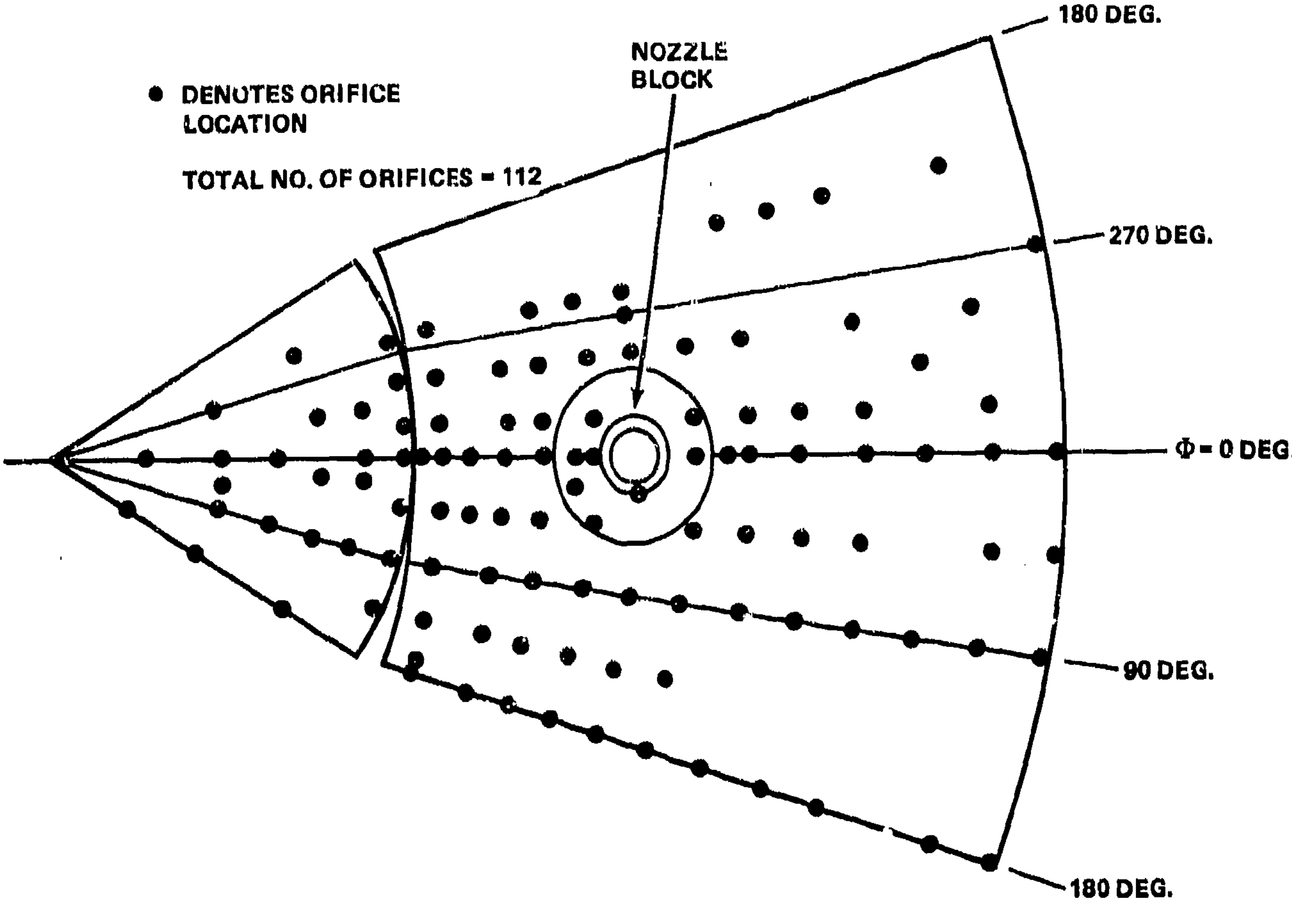

FIGURE 5. SURFACE PRESSURE ORIFICE LOCATIONS

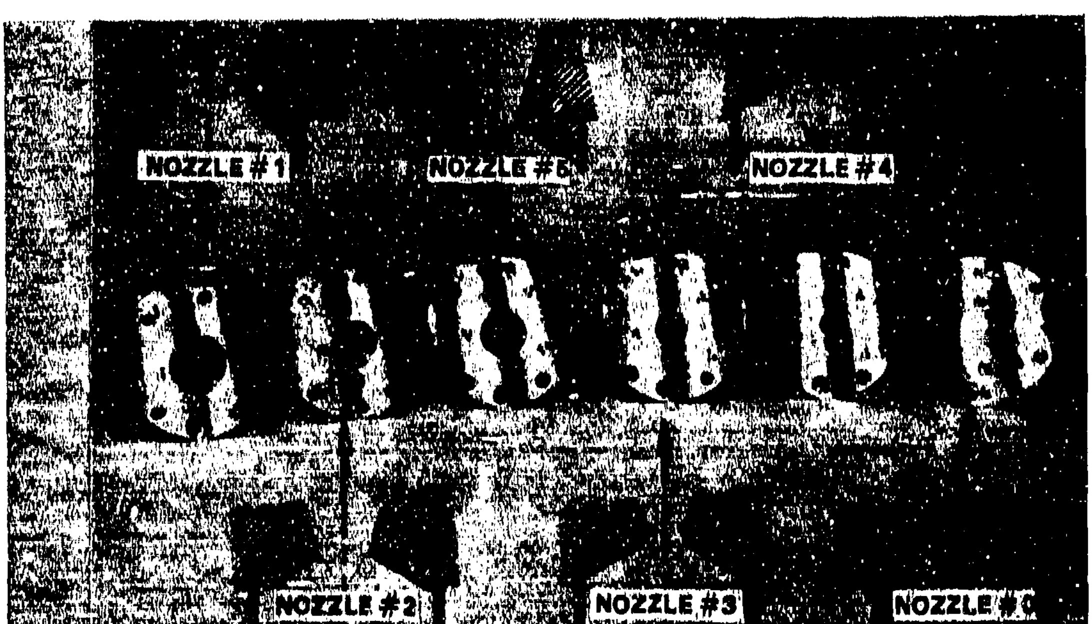

FIGURE 6. INTERCHANGEABLE THRUSTER NOZZLE BLOCKS

support system, the N_2 gas supply system, and to provide a plenum chamber for the thruster capable of withstanding high internal pressure.

The forebody, centerbody and afterbody sections are shown in the photograph in Figure 7. Due to response time errors typical of short duration facilities, transducers must be mounted inside the model. This requirement along with the large quantity of surface instrumentation resulted in the complex mounting of the transducers. The forebody section was an aluminum shell capable of housing the sixty-two pressure transducers that serviced forebody and centerbody surface pressure orifices. The remaining pressure transducers were mounted in the magnesium shell afterbody. Passageways for pressure taps in the solid centerbody section allowed surface pressure measurements to be made by tranducers in the forebody and afterbody sections.

INSTRUMENTATION

Surface pressure measurements were made with Microswitch 130PC four-active-element piezoresistive bridge tranducers mounted in close proximity to the pressure taps. Tygon jumper tubes were used between the model pressure taps and the transducers. In those parts of the model where transducers were heavily packaged, metal springs were wrapped around the bent sections of the tygon tube or miniature metal bellows were used to avoid collapsing the jumper tubes.

Other instrumentation included Microswitch transducers measuring two base pressures and Statham unbonded wire strain gage transducers measuring the thruster plenum pressure at two locations, gas supply pressures at four locations and tunnel wall pressures at five locations.

Photographic data consisted of 70mm and 16mm black and white Schilerens taken at rates of 10 and 600 frames per second, respectively. Color 70mm Schlierens were obtained for one run. Large, approximately 30 in. square, windows made wide field of view possible. Time signals from the flash unit used with the 70mm camera were sent to the data acquisition system and to the 16mm camera so that the photographic data could be correlated with model attitude or jet pressure data.

BLOCKAGE STUDIES

Investigations of blockage occurring during a tunnel run were performed prior to and during the major test entry. A blockage test was conducted in November 1981, using a mock-up of the actual test configuration. The thruster nozzle was machined into a cylindrical centerbody and an existing forebody cone and afterbody flare were attached. The blockage test determined blockage limits during a typical Hypervelocity Tunnel run and provided performance evaluation of the gas supply system, of the Microswitch pressure transducers (eight transducers were mounted in the forecone) and thruster jet uniformity (by Schlieren photographs of the jet).

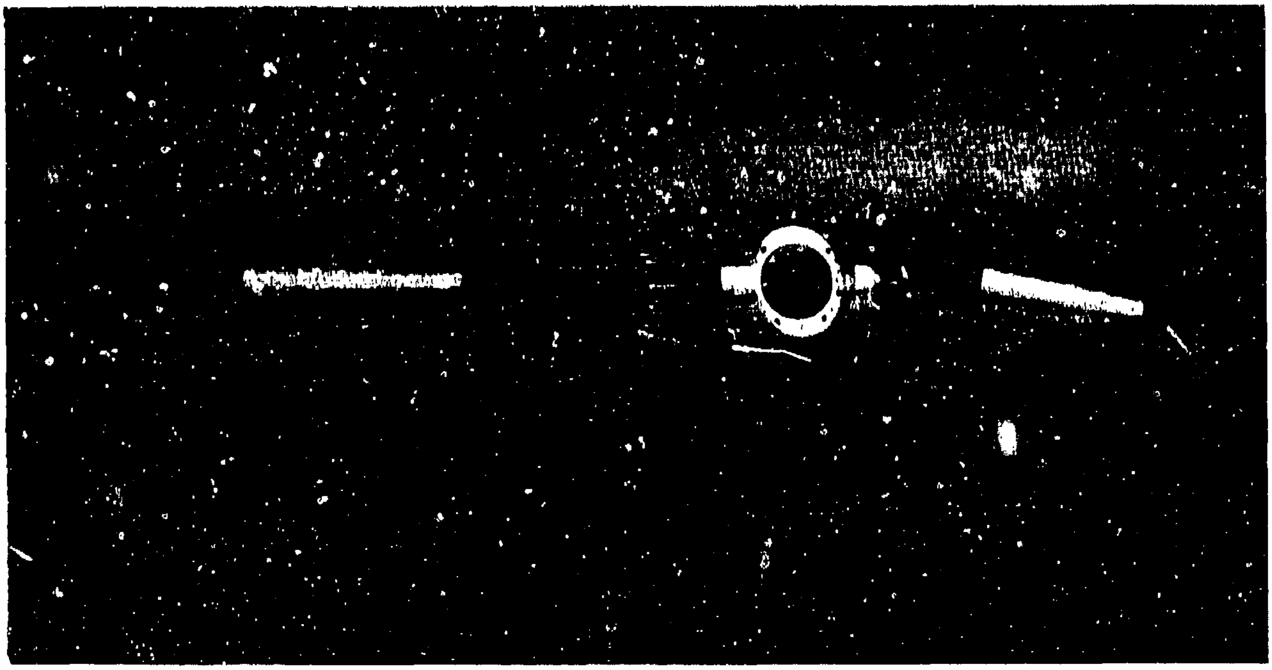

FIGURE 7. AFTERBODY, CENTERBODY AND FOREBODY SECTIONS

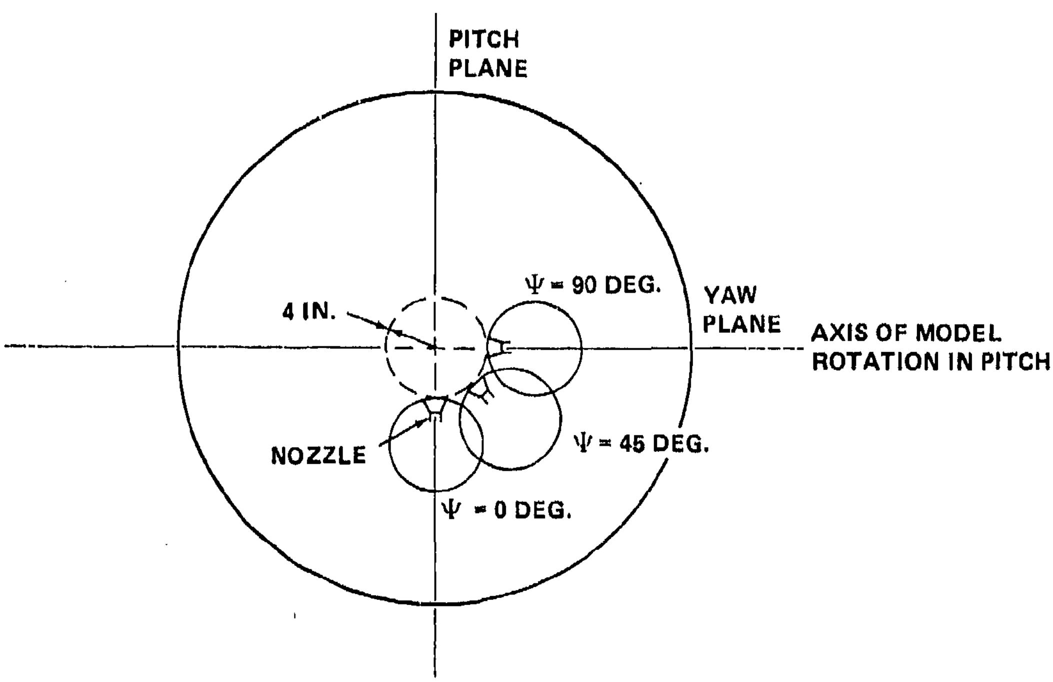

FIGURE 8. MODEL POSITION IN TUNNEL

The scale of the final model was selected based on the model size that would accommodate the 112 pressure transducers and still provide the largest jet parameter. The jet parameter is proportional to the jet size divided by the base area. Therefore, the smallest base area and the largest jet size that will not cause blockage provide the largest jet parameter. Offsetting the model so that the nozzle exit plane was four inches from the center of rotation, allowed the jet to be more nearly centered in the tunnel and reduced the chance of blockage. This change along with a slightly smaller model size allowed much higher jet parameters to be reached in the primary test entry than were reached in the blockage test.

The occurrence of blockage or flow breakdown was measured using four independent sources. These sources consisted of impact or pitot pressures, model surface pressures, tunnel wall pressures and Schlieren photographic data. The data obtained during the blockage test indicated that the most sensitive indication of flow breakdown was the Schlieren photographic data. The tunnel wall pressure was less sensitive followed by pitot pressures and surface pressures which was the least sensitive of the four.

Schlieren pictures from the blockage test indicated flow breakdown at smaller jet parameter values than the pressure data for a Mach 10 run at -10 deg. angle of attack. The wall pressures indicated flow breakdown at a jet parameter of 5.0, the pitot pressure indicated flow breakdown at a value of 5.3 and the Schlieren photographs indicated breakdown at values between 4.9 and 3.0. The surface pressures taps were located on the forecone of the model, too far from the base of the model to aid in this discussion.

In the primary test entry four additional wall pressure taps and an additional pitot pressure tube were installed to better identify flow breakdown. The surface pressure data at the rear of the model also provided an indication of flow breakdown. During a Mach 10 run at zero angle of attack in the primary entry flow breakdown occurred in the vicinity of the most sensitive wall pressure tap (directly over the jet) at a jet parameter of 4.2, in the vicinity of a wall pressure tap further upstream at a jet parameter of 5.7 and in the vicinity of the surface pressure taps at a jet parameter of 6.5. During another run at -13 deg. angle of attack flow breakdown occurred in the vicinity of the surface pressure taps at a jet parameter of 5.6, while during another run with jet parameter values of 7.9 flow breakdown occurred at +2 deg. in the vicinity of the surface pressure taps. The indication of flow breakdown by surface pressure tap data was used during the test to determine the amount of disturbance-free surface pressure data obtained during each run.

MODEL SUPPORT SYSTEM

The model used in the test program was mounted in the test cell using a model support system specifically designed to provide; 1) an offset so that the model center of rotation was about a point 4 in. outboard of the center of the thruster nozzle exit plane for every bank angle (see Figure 8), 2) thruster bank angles of 0, 45 and 90 deg., 3) angle of attack sweeps from -15 to +15 deg., 4) support for the model and the 1550 lb. lateral thrust, 5) ducting for high pressure thruster gas, and 6) aerothermal shielding for the electrical wire bundle. Fairing of blunt model support parts to reduce aerodynamic interference as well as to reduce the possibility of blockage was included in the design.

The final design of the model support system resulted in the stings illustrated in Figures 9, 10 and 11. The three bank angles were provided by hardware changes and rotations of the stings. A twenty degree bend in the 0 deg. roll sting allowed angle of attack sweeps from -15 to +15 deg. The 45 deg. and 90 deg. roll sting allowed model angle of attack from -5 to +15 deg. The tunnel sector was capable of supporting the 1550 lb. thrust in the pitch plane, but a restraint cable (shown in Figure 11) was required to support the thrust in the other two orientations. Figure 12 shows the model and sting assemblies in the Hypervelocity Tunnel test cell.

GAS SUPPLY SYSTEM

The Hypervelocity Tunnel gas supply system was designed and constructed specifically to meet the needs of this test program. Figure 13 shows the features of the system. The working fluid was nitrogen at a design pressure of 5000 psia. up to the control valve and 3000 psia. downstream of the valve. The storage reservoir was an 86 cubic ft. horizontal vessel (shown in Figure 14) supplied directly from the nitrogen storage field. The large volume of the reservoir assured practically constant thruster supply pressures during operation. The gas supply was controlled by an Autoclave control valve operated as an on/off valve or programmed for variable control. The valve is shown in Figure 15.

Transducers to monitor the pressure of the gas supply system were located at the storage reservoir, at the inlet and outlet of the Autoclave valve, at the sting near the model and at the thruster plenum. Figure 16 shows the calibration curve used to select the proper pre-run reservoir pressure to obtain the desired model plenum pressure.

DATA REDUCTION

Pressure data obtained during the test were reduced in the following manner; 1) raw data were filtered at a cut-off frequency that would remove all but the desired aerodynamic data, 2) the filtered data were converted to pressure by using pre-test, pre-run or post-test calibration factors in the correct pressure range, 3) the pressures were corrected for pressure shifts attributable to transducer or data system zero shifts after calibration by utilizing a pre-run static taro, 4) the pressures were corrected for pressure lag (a function of the rate of change of pressure), 5) the data were reduced to coefficient form by using the freestream static and dynamic pressures derived from the tunnel condition data, and 6) the pressure coefficients versus angle of attack were tabulated and plotted. Other data that were recorded included tunnel supply pressure and temperature, freestream pitot pressures, angle of attack, model base pressures, tunnel wall pressures, gas supply pressures and

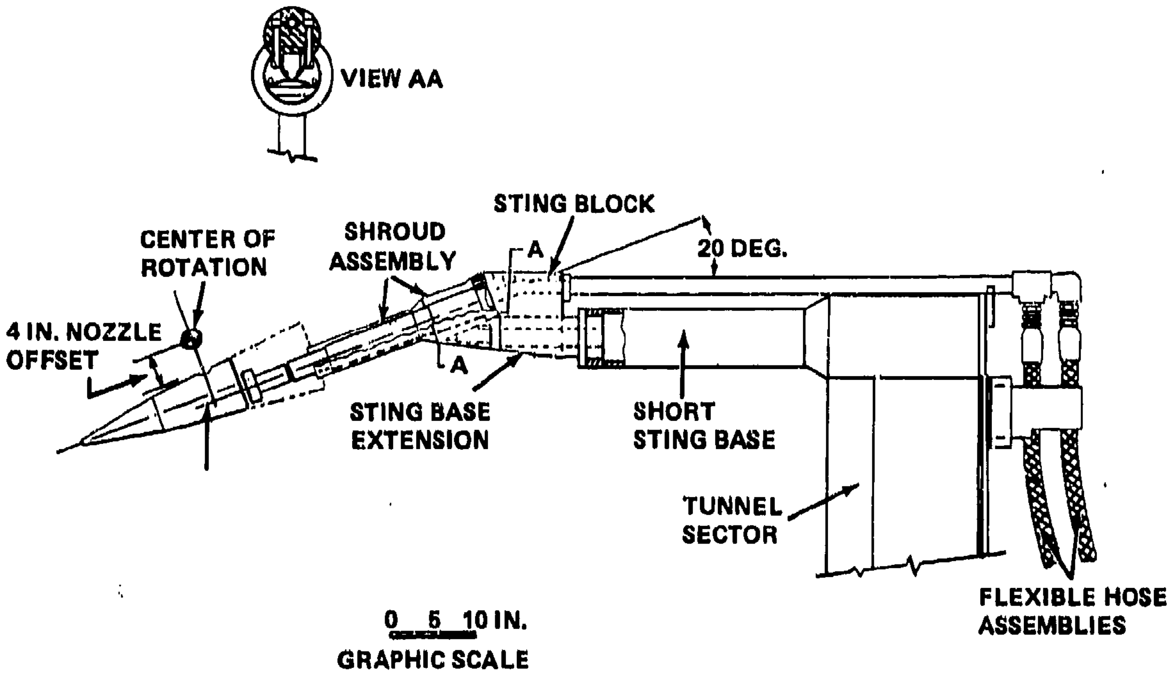

FIGURE 9. H.A.M.C. MODEL/STING ASSEMBLY 0 DEG. ROLL CONFIGURATION

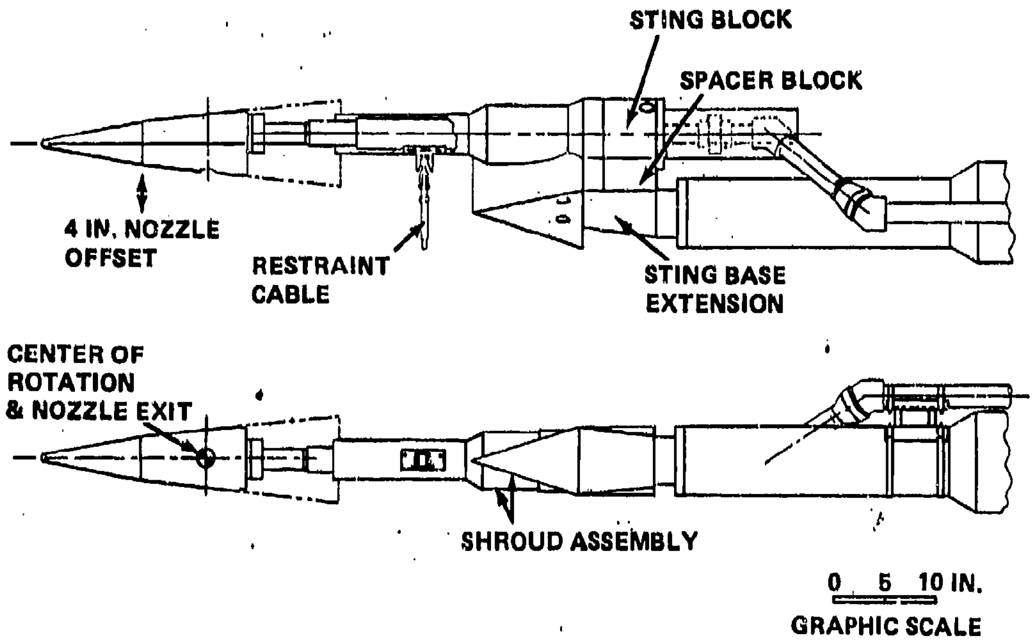

FIGURE 10. H.A.M.C. MODEL/STING ASSEMBLY 90 DEG. ROLL CONFIGURATION

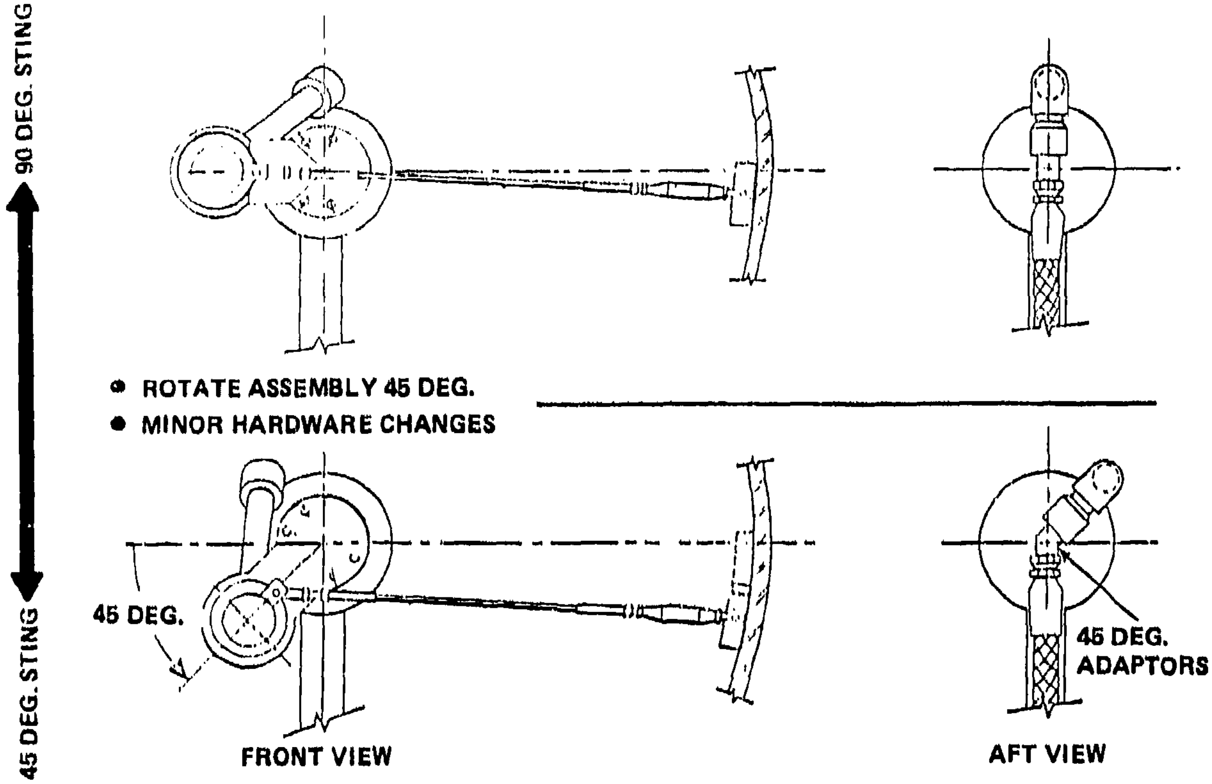

FIGURE 11. H.A.M.C. ROLL STING ASSEMBLIES 90 DEG. TO 45 DEG. (OR VICE VERSA) STING CHANGE DETAILS

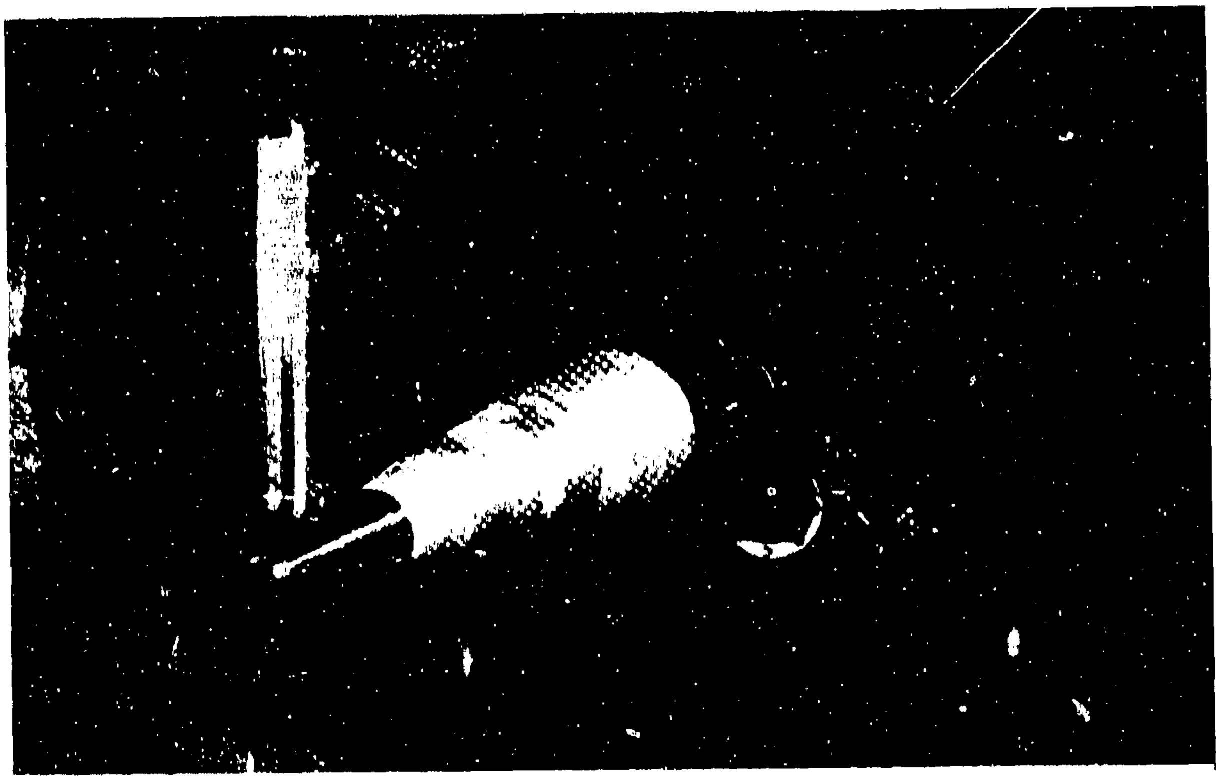

FIGURE 12. H.A.M.C. MODEL AND 0 DEG. ROLL STING IN THE HYPERVELOCITY TUNNEL

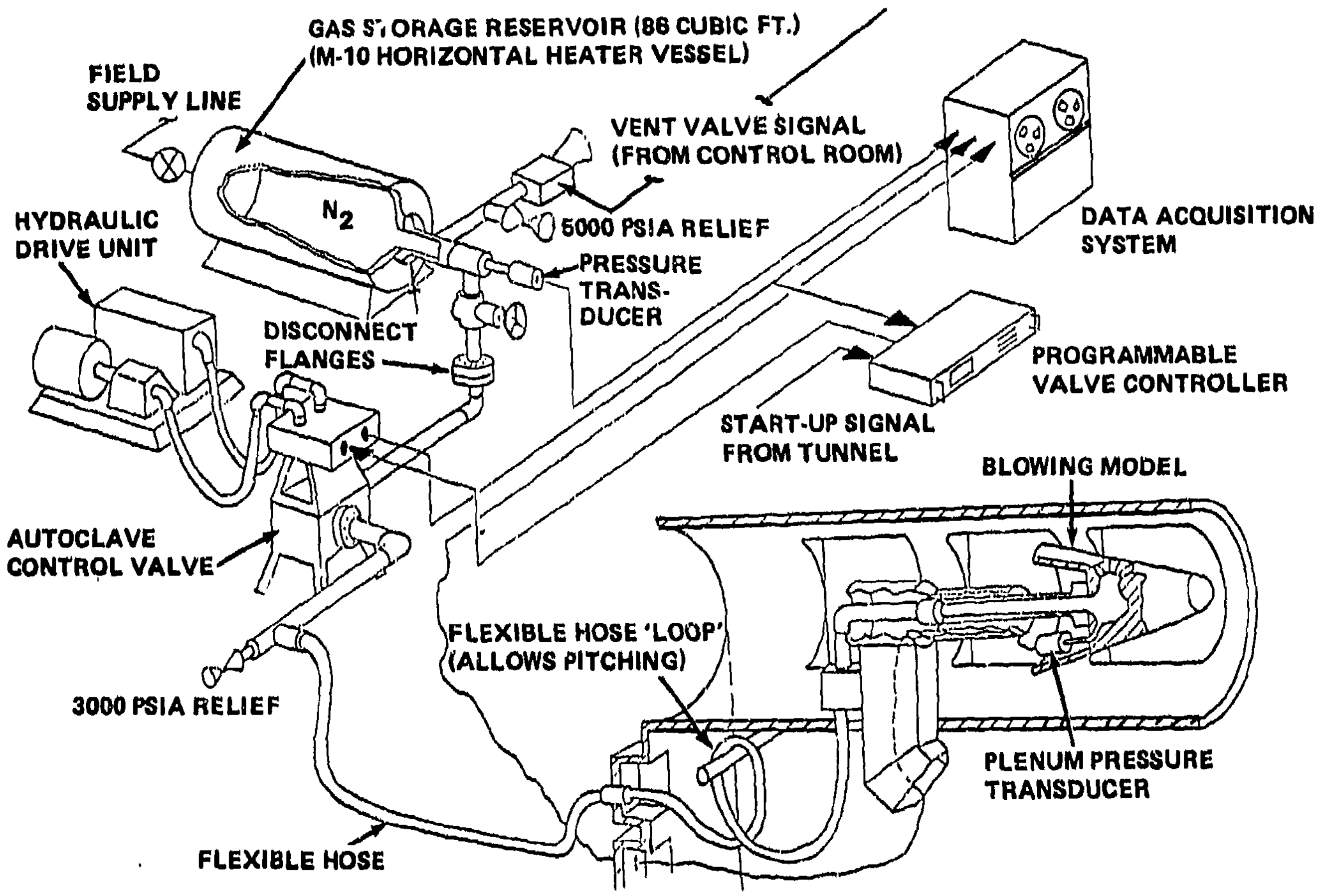

FIGURE 13. TUNNEL BLOWING GAS SUPPLY SYSTEM

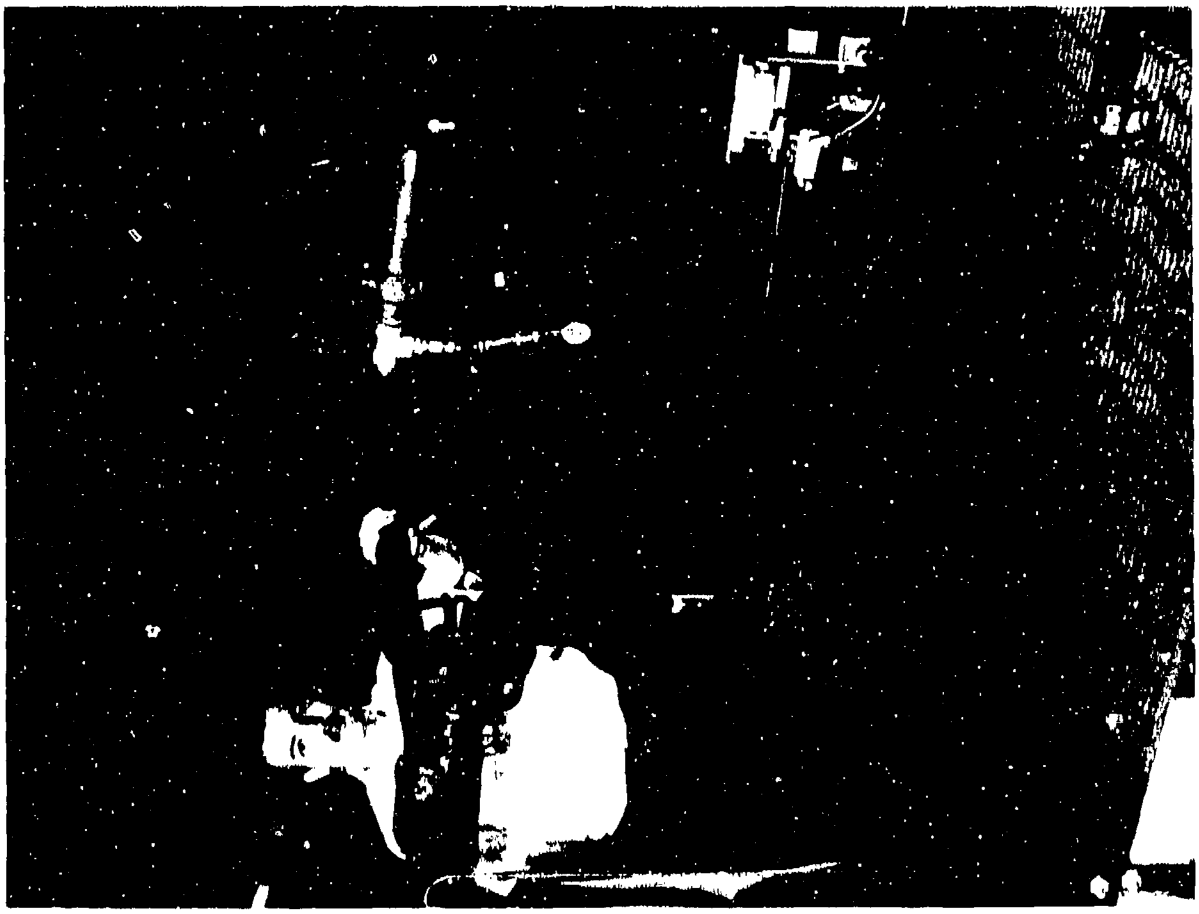

FIGURE 14. STORAGE RESERVOIR FOR THE TUNNEL GAS SUPPLY SYSTEM

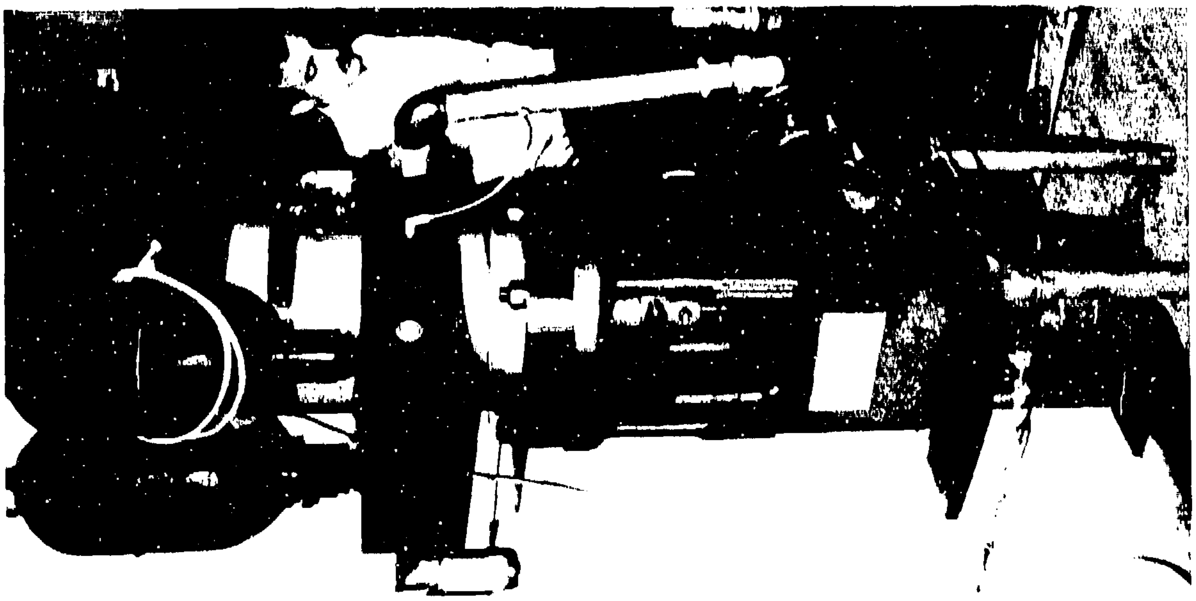

FIGURE 15. AUTOCLAVE VALVE FOR THE TUNNEL GAS SUPPLY SYSTEM

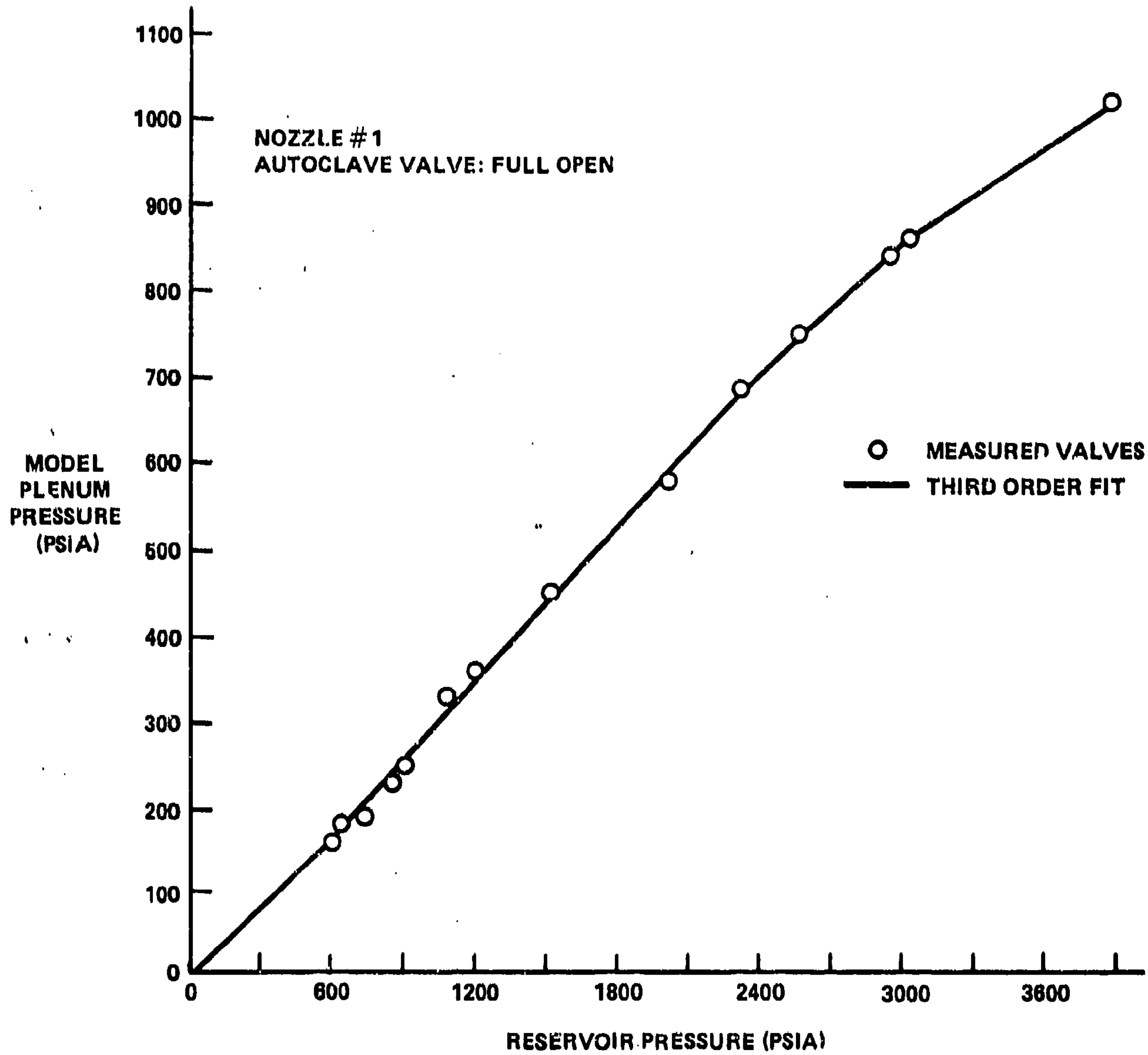

FIGURE 16. GAS SUPPLY CALIBRATION CURVE

model plenum pressures. These data were filtered and converted to engineering units using the proper calibration. Correction of model angle of attack data for flow misalignment was performed using measured values.

The Microswitch pressure transducer was evaluated prior to the test to determine the inherent noise level and the effects of digital filtering. Two transducers were connected to the data acquisition system at various amplifier gain settings and data were recorded at a sampling rate of 100 samples/sec. The result is shown in Figure 17 for peak signal of noise expressed in psia. versus the cut-off frequency of the digital filter. The unfiltered data values are plotted at 50Hz corresponding to the folding frequency of the 100/sec. sample rate. Filtering at any frequency below 15Hz will reduce the noise below the acceptable level that was set based on past experience.

Filter cut-off frequency selection was unique for this test program due to the pressure steps introduced by the jet. For simplicity, the selection of cut-off frequencies was made for three categories of pressure data; 1) pitot pressure data for all runs, 2) surface pressure data for the runs with no jet blowing, and 3) surface pressure data for jet interaction runs.

The pitot pressure data (category #1) was filtered using a cut-off frequency that would reduce the noise level but not introduce errors due to filter ringing. Filter ringing is produced by filtering of data that changes abruptly. Figure 18 shows pitot pressure data unfiltered and filtered at 5 and 3Hz cut-off frequencies. The ringing or settling time for each of the filtered data is shown and illustrates the reason for selecting the 5Hz cut-off frequency to reduce all pitot pressure data.

The surface pressure data for category #2 runs typically included a sharp rise in pressure at the start of the run, similar to the rise in pitot pressure, then changes in pressure due to pitching of the model. Again, the filter cut-off frequency was selected so that errors due to ringing would be minimized. Synthetic pressure data was created as a function of the angle of attack data and was filtered to observe the effect of using a 5Hz cut-off frequency. Figure 19 shows synthetic data and real data filtered and unfiltered. There is no appreciable error introduced by the filtering except at the ends (this data is not used).

The surface pressure data for category #3 runs were reduced using a cut-off frequency based on the smearing effect of filtering on pressure steps. The smearing effect is illustrated in Figure 20 which shows surface pressure data filtered at 26Hz, 20Hz and 15Hz cut-off frequencies. Note that the pressure rise from an undisturbed condition at -2.7 deg. to 66% of the maximum pressure occurs over a 1.5 deg. angle of attack range for the 26Hz cut-off frequency. The 15Hz cut-off frequency, however, which would reduce the noise level to the acceptable limit, smears the pressure step more than could be considered acceptable. The 20Hz cut-off frequency was selected to reduce the smearing effect of filtering.

The correction of pressure data for pressure lag effects was determined to be necessary based on pressure data obtained from the Microswitch evaluation test conducted as part of the blockage test mentioned previously. In that test a mock-up of the actual test configuration was instrumented with eight pressure

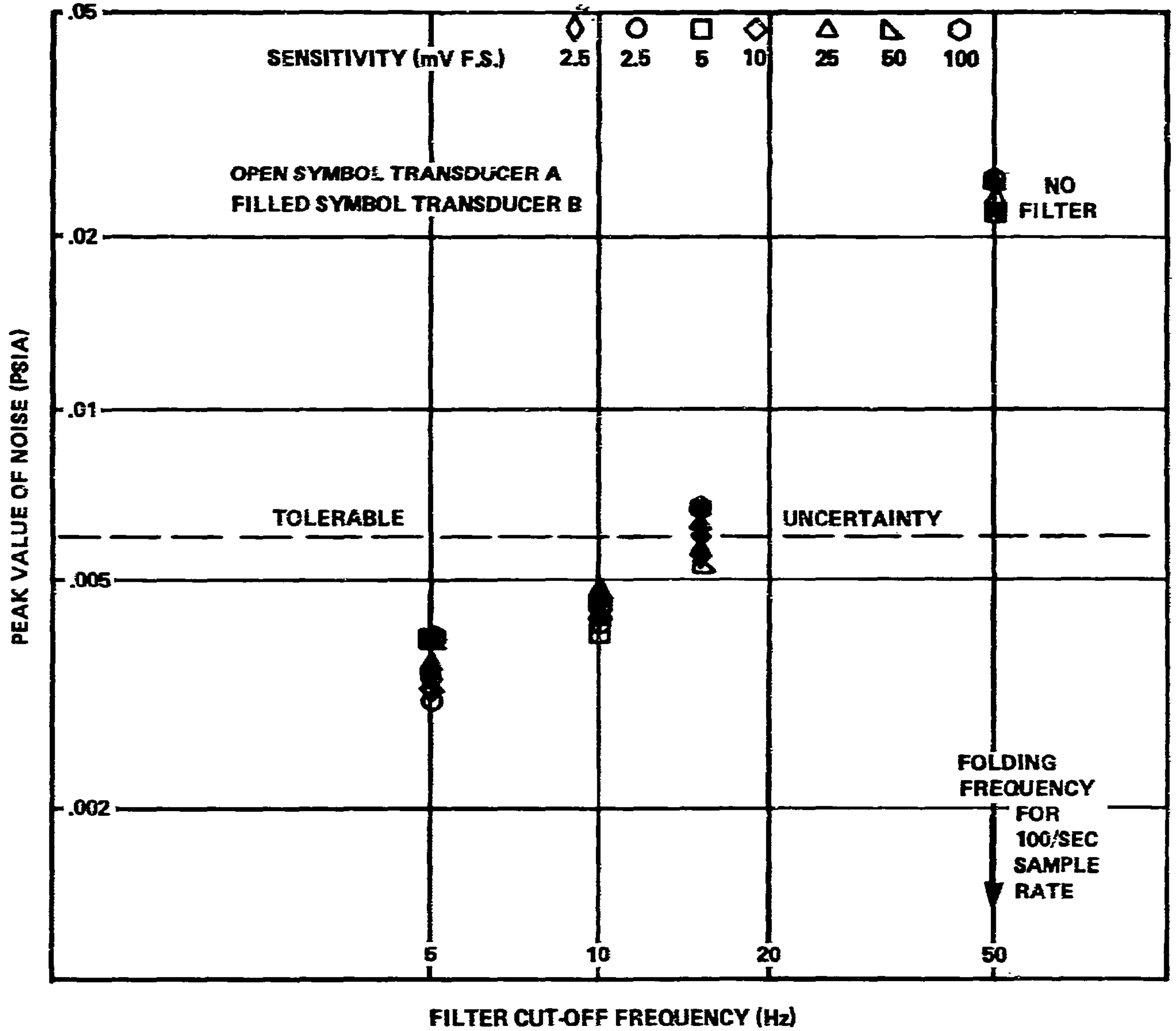

FIGURE 17. NOISE MEASUREMENTS ON MICROSWITCH PRESSURE TRANSDUCERS USING TUNNEL DATA ACQUISITION SYSTEM

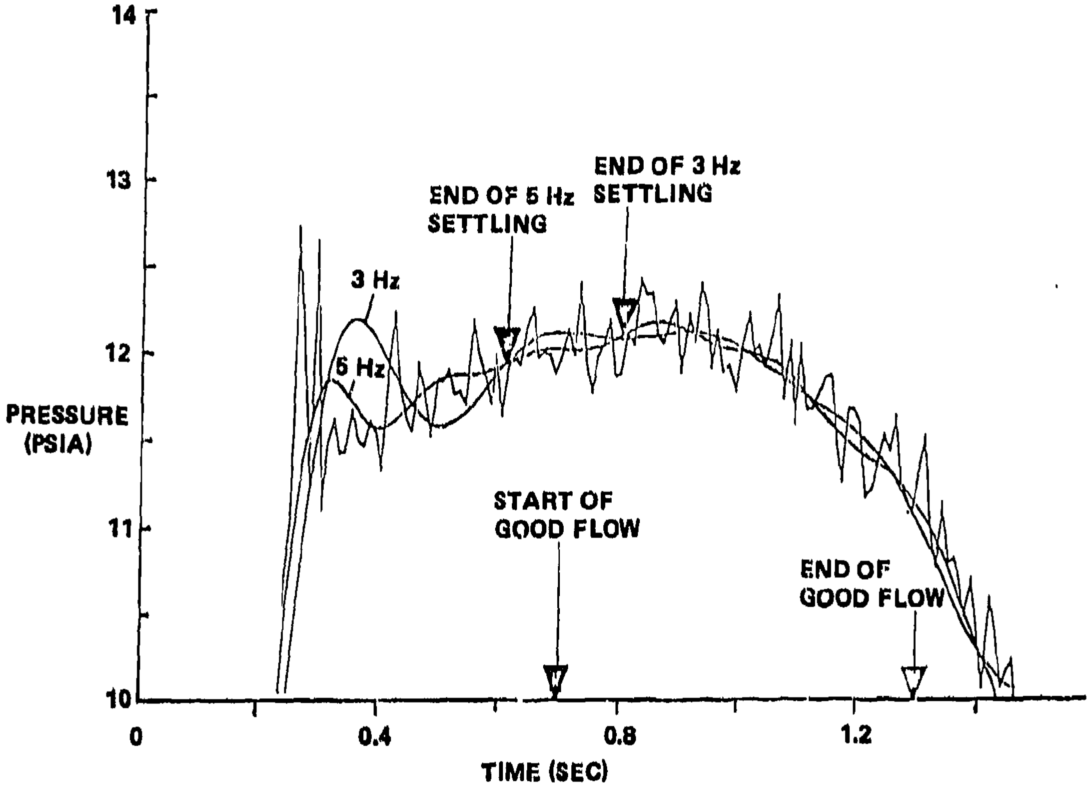

FIGURE 18. PITOT PRESSURE DATA UNFILTERED AND FILTERED AT 5 Hz AND 3 Hz CUT-OFF FREQUENCIES

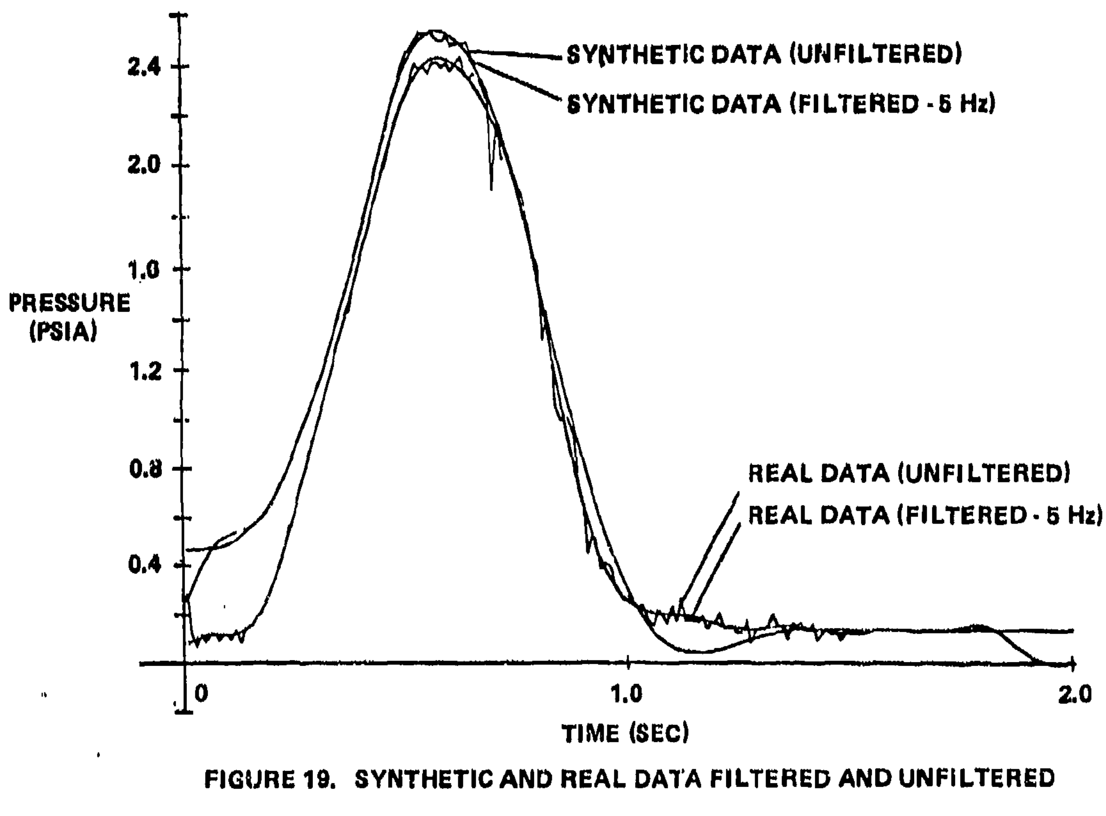

FIGURE 19. SYNTHETIC AND REAL DATA FILTERED AND UNFILTERED

transducers and measured the surface pressures at Mach 10 conditions on a 7 deg. half-angle forecone. A sweep of angle of attack from -10 deg. to +10 deg. was followed by a sweep in the opposite direction. The ΔP or the difference between the surface pressures recorded at the same angle but in opposite directions was ± 0.009 psia.

To correct for pressure lag the following equation was used:

$$P_{true} = P_{meas} + \tau (d\ P_{meas}/dt)$$

$$= P_{meas} + \tau P\ {}^{d}/dt\ (\ln P_{meas})$$

The value of τP, the product of the pneumatic time constant and pressure, for the Microswitch pressure transducer was measured prior to the test for various pressures using a quick opening valve. The response curve from these measurements shows that:

$$\tau P = 0.0022 \text{ psia.-sec.}$$

However, solving for τP using the ΔP and ${}^{d}/dt$ ($\ln P_{meas}$) from the blockage test data discussed above, a value of 0.0013 psia.-sec. was determined. The discrepancy in the correction factors is due to the difference in the gas used in the bench tests (ambient air) and the hot nitrogen gas present in an actual Hypervelocity Tunnel run. The value of 0.0013 psia.-sec. was used to correct all of the run data for pressure lag effects.

The final aspect of data reduction concerns the use of a flow misalignment angle to correct the angle of attack data. This angle has been measured for the Mach 10 and Mach 14 nozzles in the past and was measured for the Mach 14 nozzle in the present program. The angle was determined to be +0.29 deg. in the pitch plane by comparing the surface pressures on the top and bottom ray during a run at a stationary angle. The correction angle measured during past test programs is +0.28 deg. The measured value of +0.29 deg. was used to correct all Mach 14 angle of attack data and a correction angle of +0.25 deg., measured during past test programs, was used to correct all of the Mach 10 angle of attack data.

UNCERTAINTY OF MEASUREMENTS

The uncertainties of basic flow parameters for the Hypervelocity Tunnel have been determined by previous test programs[3] and are listed in Table 1. Nozzle calibration tests have also determined a $\pm 1.5\%$ variation in Po_2 across the test cell which, along with the basic flow parameter accuracies, were used to determine the free-stream flow property uncertainties.

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[3]Pronchick, S. W., Static Stability Tests of the Space Shuttle Orbiter at Mach 14 in the NSWC Hypervelocity Tunnel, NSWC/WOL MP 78-20, Dec 1978.

TABLE 1. UNCERTAINTIES IN FLOW PARAMETERS

Basic Flow Parameters			Free-Stream Flow Parameters			
Po	To	Po_2	Mach	P_{inf}	Q	Re/L
±0.4%	-1.7 to 0.5%	±0.3%	±0.4%	±0.9%	±1.7%	-1.0 to 2.8%

The uncertainty of pressure transducer measurement depends on the range of pressures over which they are calibrated. At a calibration range of higher pressures, the non-linearity is greater. During this test program careful selection of the calibration range based on the measured pressures resulted in reduced uncertaintites.

Two calibrations, one from 0.1 psia. to 2 psia. and another from 2 psia. to 15 psia. were performed prior to each run of the test program. The uncertainty was ±0.0008 psia. in the lower range and ±0.001 psia. in the higher range. In addition, nine tranducers measured pressures during the test program above 15 psia. range. Post-test calibrations of each overpressured transducer indicated that an uncertainty of ±0.002 psia. could be expected for all but three transducers. A third-order, post-test calibration was used for these three transducers which provided an uncertainty of ±0.002 psia.

The scatter in the pressure data can be attributed to "tunnel noise" and "electrical noise." Detailed measurements of the tunnel pitot pressure were performed in February 1982. "Tunnel noise," expressed as the peak value of the ratio of fluctuating to mean pitot pressures, was found to be in the range from ±4% to ±7%. "Electrical noise" of the Microswitch pressure transducers were measured (see DATA REDUCTION and Figure 17) and indicate that the peak value of noise is ±0.025 psia., independent of amplifier gain.

Filtering reduces the scatter of the pressure data in the following manner. The "tunnel noise" corresponding to a Mach 14 run, which is ±5.6% (peak value of the ratio of fluctuating to mean pressure), can be filtered to ±2.4% using a 20Hz cut-off frequency and to ±1.4% using a 5Hz cut-off frequency. The "electrical noise" can be reduced to ±0.007 psia. using a 20Hz filter cut-off frequency and ±0.0035 psia. using a 5Hz filter cut-off frequency. This results in an uncertainty of ±0.004 psia. or ±12% on leeward side data (mostly due to "electrical noise") and ±0.01 psia. or ±1.4% on windward side data (mostly due to "tunnel noise") for data filtered at a 5Hz cut-off frequency (category #1 and #2 pressure data). The runs with jet blowing can have surface pressures as low as 0.02 psia. behind the jet or as high as 42 psia. forward of the jet. Filtered at 20Hz, the former pressure data would have uncertainties of ±0.008 psia. or ±37% (again due to electrical noise) and the latter could have uncertainties of ±1.0 psia. or ±2.4%, provided that the fluctuating tunnel flow affects the regions that produce the high pressure.

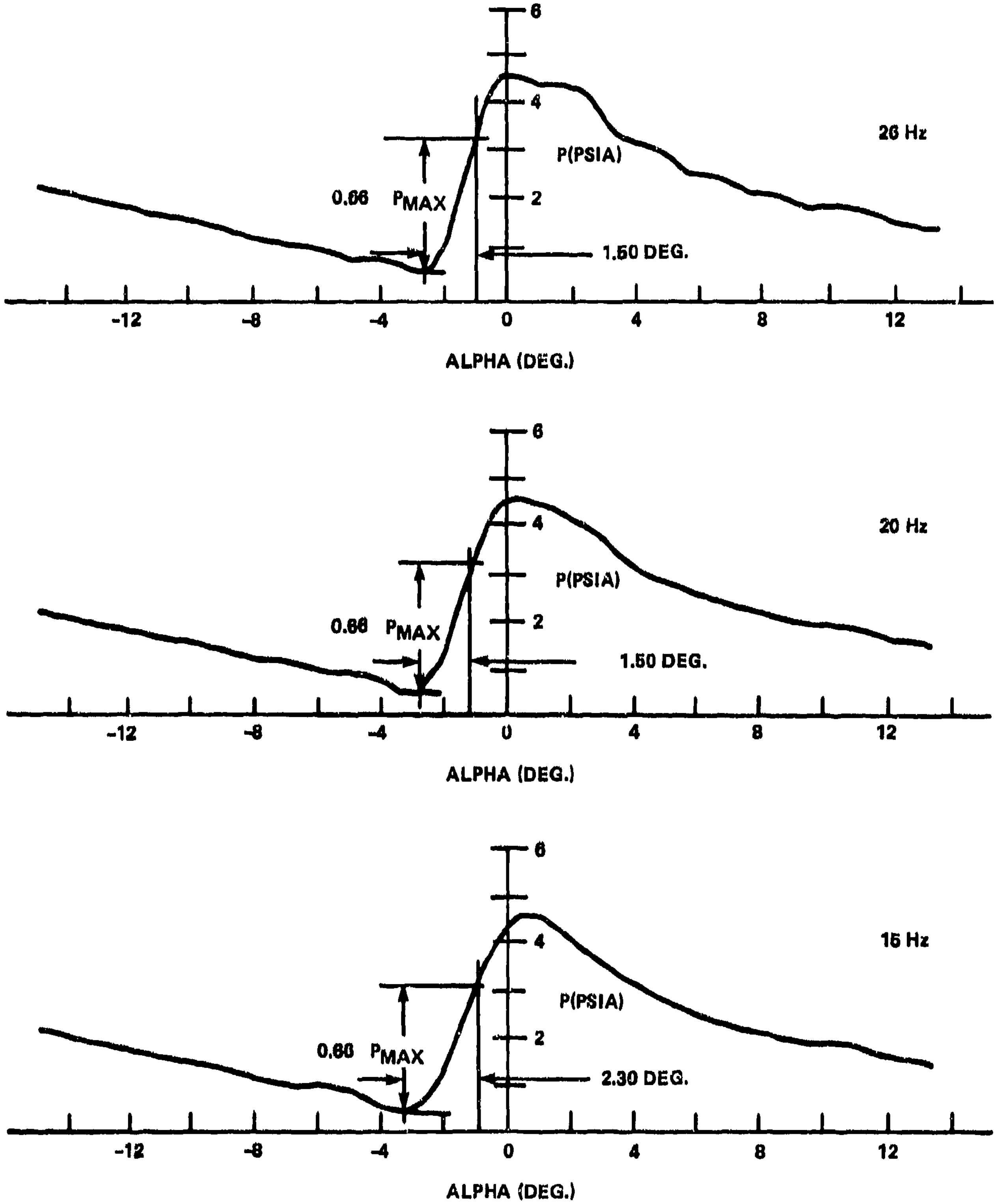

FIGURE 20. EFFECT OF FILTERING ON PRESSURE STEPS

The uncertainty for measurement of angle of attack was determined by repeat calibrations using an inclinometer positioned on the model support head. An electro-optical sensor, positioned on the carriage to which the support arm is attached, aids in providing an uncertainty in angle of attack measurements of only ± 0.034 deg.

RESULTS AND DISCUSSION

The conditions of the test program are summarized by the test matrix shown in Table 2. The three Mach 14 runs, at a nominal Reynolds number per ft. of 3.8×10^6, provided tare values, blockage limits with no pitch sweep and data for a jet parameter of 3.5 for the full pitch sweep.

Three Mach 10 runs provided similar conditions as the preceeding Mach 14 runs at a nominal Reynolds number per ft. of 3.3×10^6 and for a jet parameter of 5.5. In addition, eleven runs with the largest nozzle block provided three jet parameter values at the high Reynolds number (4.7×10^6/ft.), four jet parameter values at the medium Reynolds number (3.3×10^6/ft.) and four jet parameter values at the low Reynolds number (2.0×10^6/ft.) (one of these runs was a repeat run with opposite pitch sweep direction). Another three nozzle blocks were tested at the medium Reynolds number condition. The smallest nozzle block was used in an additional run to duplicate a run made in a previous test program.

Finally, four runs were conducted with the model at a bank angle of 90 deg. and at a bank angle of 45 deg. to study the 3-D effects of the jet. The two runs at each bank angle were conducted at a nominal Reynolds number per ft. of 3.3×10^6 and at jet parameter values of 3.9 and 2.5

TABLE 2. TEST MATRIX--HIGH ALTITUDE MANEUVER CONTROL PROGRAM

Run Date	Run No. Tunnel	Run No. Prog.	M	Re/L (10^6/ft.)	Alpha (deg.)	Ψ (deg.)	Nozzle	Jet Parameter
5-4-82	680	1	14.5	3.7	-15/+15	0	0	0
5-5-82	682	2	14.2	3.8	0	0	1	0/4.5
5-5-82	683	3	14.1	3.8	+15/-15	0	1	3.5
5-6-82	684	4	9.7	3.4	+10/-15	0	0	0
5-6-82	685	5	9.7	3.2	-15/+15	0	1	0
5-7-82	686	6	9.7	3.3	0	0	1	0/5.7
5-7-82	687	7	9.7	3.1	+15/-14	0	1	5.5
5-7-82	688	8	9.7	4.6	-15/+15	0	1	3.8
5-10-82	689	9	9.8	2.0	+15/+2	0	1	7.9
5-10-82	690	10	9.7	3.3	+15/-15	0	1	3.8
5-11-82	692	11	9.8	4.6	-15/+10	0	1	2.7
5-11-82	693	12	9.6	2.0	+15/-15	0	1	5.0
5-12-82	694	13	9.7	3.5	+15/-15	0	1	2.3
5-12-82	695	14	9.7	4.9	-15/+15	0	1	1.5
5-13-82	696	15	9.7	3.2	+15/-15	0	2	4.0
5-14-82	697	16	9.7	3.3	+15/-15	0	3	1.6
5-14-82	698	17	9.7	3.3	+15/-15	0	4	0.6
5-14-82	699	18	9.7	3.2	+15/-15	0	4	0.1
5-17-82	700	19	9.7	3.1	-15/+15	0	1	3.8
5-17-82	701	20	9.6	2.0	+15/-15	0	1	3.9
5-17-82	702	21	9.6	2.0	+15/-14	0	1	2.7
5-19-82	704	22	9.7	3.1	-5/+14	90	1	3.9
5-20-82	705	23	9.7	3.3	-5/+15	90	1	2.5
5-21-82	707	24	9.7	3.2	-5/+15	45	1	3.9
5-21-82	708	25	9.7	3.2	-5/+15	45	1	2.5

NOMENCLATURE

Alpha	Model angle-of-attack, deg.
deg.	Degrees, (angle)
ft.	feet, (length)
H.A.M.C.	High Altitude Maneuver Control
Hz	Hertz, (frequency)
in.	Inches, (length)
lb.	Pounds, (force)
M, Mach	Free-stream Mach number
mm.	Millimeters, (length)
m.v.F.S.	Millivolts Full Scale, (sensitivity)
N_2	Nitrogen
P	Surface pressure, psia.
P_{inf}	Static pressure, psia.
P_{max}	Maximum pressure, psia.
P_{meas}	Measured pressure, psia.
Po	Tunnel supply pressure, psia.
Po_2	Pitot tube pressure, psia.
psia.	Pounds per squared inch absolute, (pressure)
P_{true}	True pressure, psia.
Q	Free stream dynamic pressure, psia.
Re/L	Free stream unit Reynolds number, $ft.^{-1}$
°R	Degrees Rankine, (temperature)
sec.	Seconds, (time)
3-D	Three dimensional
To	Tunnel supply temperature, °R
ΔP	Delta P, pressure difference, psia.
τP	Tau P, product of time constant and pressure, psia.-sec.
ϕ	Phi, circumferential position, measured from center of jet, deg.
Ψ	Psi, bank angle, deg.

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