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Technical Report

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DEVELOPMENT OF A MINIATURE SOLID PROPELLANT ROCKET MOTOR FOR USE IN PLUME SIMULATION STUDIES

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FOREWORD

This report was prepared by the Calspan Corporation, Buffalo, New York, 14221, to cover the work accomplished from June 1973 to November 1973 under Contract No. NAS8-26701 for the N.A.S.A., George C. Marshall Space Flight Center, Huntsville, Alabama. The program was administered for N.A.S.A. by Mr. John Reardon of Remtech Inc., Huntsville, Alabama.

At Calspan, this work was under the technical supervision of Mr. Kenneth C. Hendershot of the Aerodynamic Research Department.

Acknowledgement is made to F. A. Vassallo and Mr. Hendershot of the Calspan Corporation and Mr. Reardon of Remtech Inc. for their valuable technical comments. The contributions of T. J. Maj to the completion of the live rocket tests and S. T. Liszewski to the fabrication of the specialized electronics required during the program are also gratefully acknowledged.

ABSTRACT

A miniature solid propellant rocket motor has been developed to be used in a program to determine those parameters which must be duplicated in a cold gas flow to produce aerodynamic effects on an experimental model similar to those produced by hot, particle-laden exhaust plumes.

Phenomena encountered during the testing of the miniature solid propellant motors included erosive propellant burning caused by high flow velocities parallel to the propellant surface, regressive propellant burning as a result of exposed propellant edges, the deposition of aluminum oxide on the nozzle surfaces sufficient to cause aerodynamic nozzle throat geometry changes, and thermal erosion of the nozzle throat at high chamber pressures.

A series of tests was conducted to establish the stability of the rocket chamber pressure and the repeatibility of test conditions. Data are presented which define the tests selected to represent the final test matrix.

Qualitative observations are also presented concerning the phenomena experienced based on the results of a large number of rocket tests not directly applicable to the final test matrix.

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INTRODUCTION

During the ascent of a rocket-propelled vehicle, rapidly decreasing ambient pressure causes the rocket exhaust plume to expand to large diameters. At sufficiently high altitudes, the plume interacts with the vehicle flowfield and produces a boundary-layer separation that can occur well forward of the vehicle base. Because this phenomenon significantly influences vehicle stability, heating, and stage separation, the conditions at which plume-induced flow separation can occur are naturally of interest.

Wind tunnel tests performed to date using existing cold-flow jet simulation techniques have not adequately demonstrated the validity of the similarity parameters for assuring proper simulation of the hot-exhaust free stream.

Calspan Corporation is in the process of performing a program which deals with the aerodynamic exhaust plume interaction effects on the Space Shuttle. The subtask with which this report is concerned is a part of the larger program and is specifically concerned with the experimental investigation of hot, particle-laden, exhaust plumes from solid propellant rockets, to determine those parameters which must be duplicated in cold gas flows to produce comparable aerodynamic effects on an experimental model.

The following describes the development of a miniature solid propellant rocket motor for use in plume simulation studies to be performed in the NASA/George C. Marshall Space Flight Center's 14 x 14 inch Trisonic Wind Tunnel Facility.

SUMMARY

The test matrix proposed for the rocket exhaust plume study to be conducted at the NASA/Marshall Space Flight Center consisted of high and low aluminum content solid rocket propellants to be tested at rocket chamber pressure levels of from 400 to 2000 psia using uncooled copper exhaust nozzles of area ratios 4 and 8. Chamber pressures, stable to within 5 percent, were required for a minimum of 0.150 seconds.

During the rocket motor calibration test phase conducted at Calspan Corporation, it was found that the propellant loading configuration in the rocket chamber was critical, and certain configurations could cause high propellant burning rates in localized areas.

These locally high burning rates (erosive burning) produced rocket chamber pressure variations that were characterized by varying degrees of high initial pressure and large pressure decay rates.

Two propellant loading configurations were found that would minimize erosive burning of the propellant. These configurations would allow a maximum propellant surface area of 23 square inches to be placed in the rocket chamber. A maximum propellant surface area of 53 square inches was required to complete the initially proposed test matrix.

The reduction in the maximum propellant area and the thermal erosion of material in the throat of the rocket nozzle combined to produce a test matrix that was limited to maximum rocket chamber pressures of 1200 and 1600 psia for the area ratio 8 nozzle and to 400 and 750 psia for the area ratio 4 nozzle using the low and high aluminum content propellants, respectively.

It was felt by MSFC that, even though the test matrix had been reduced by problems associated with the small size of the rocket chamber and the uncooled copper nozzle, a significant contribution could be made to distinguish those parameters that are critical to the simulation of a hot, solid propellant exhaust plume with a cold air jet.

TEST EQUIPMENT

Lockheed Missiles and Space Company's Huntsville Research and Engineering Center was responsible for the fabrication of the rocket chamber and the two nozzles to be used during testing at MSFC. While this fabrication was in progress, Calspan fabricated similar pieces of hardware specifically for use during the rocket motor development and calibration tests to be performed at Calspan.

The selected rocket configuration consists of a cylindrical steel chamber with a removable copper nozzle block at the downstream end and a removable forward closure containing a replaceable electric pyrotechnic igniter. A Mylar diaphragm, installed at the chamber/nozzle interface, contains the gases produced by the igniter until propellant ignition and burning is well established. Propellant is loaded into the chamber on three longitudinal plates which are retained in parallel longitudinal slots cut into the chamber periphery. Operating chamber pressure levels are varied by inserting propellant loads with more or less propellant surface area. Details of the rocket motor combustion chamber fabricated for the calibration tests are presented in Figure 1.

The removable nozzle blocks fabricated by Calspan for the calibration tests were modified versions of the test nozzles. The modification consisted of the elimination of the downstream nozzle contour and all of the nozzle pressure orifices with the exception of an orifice located at the nozzle throat. It was felt that these modifications would not compromise test data and would result in a considerable reduction in fabrication costs. The modified calibration test nozzle blocks used to represent the aerodynamic nozzles of area ratios 4 and 8 are delineated in Figures 2 and 3, respectively. A typical calibration test configuration is shown in Figure 4.

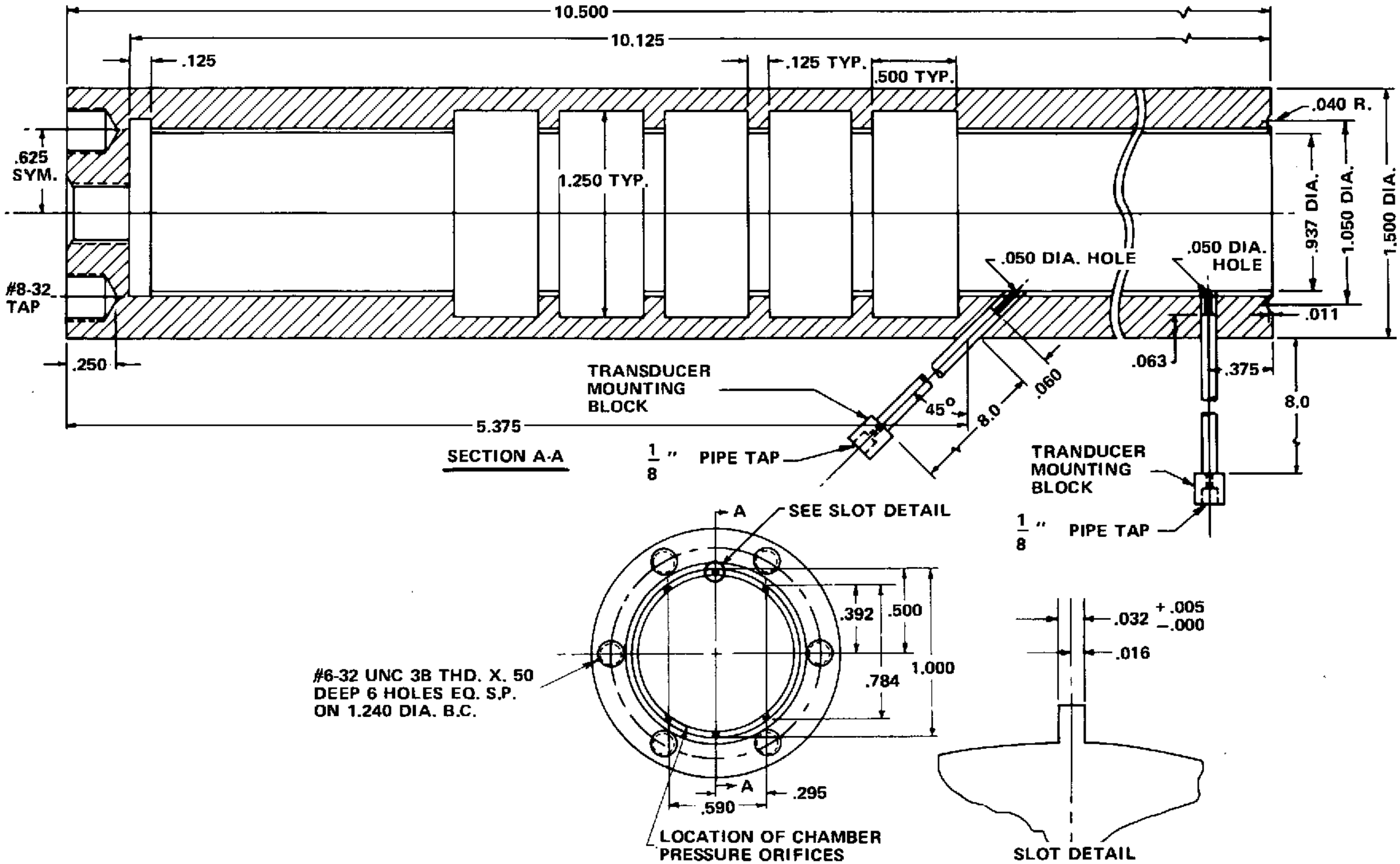


Figure 1 SOLID ROCKET MOTOR COMBUSTION CHAMBER DETAILS

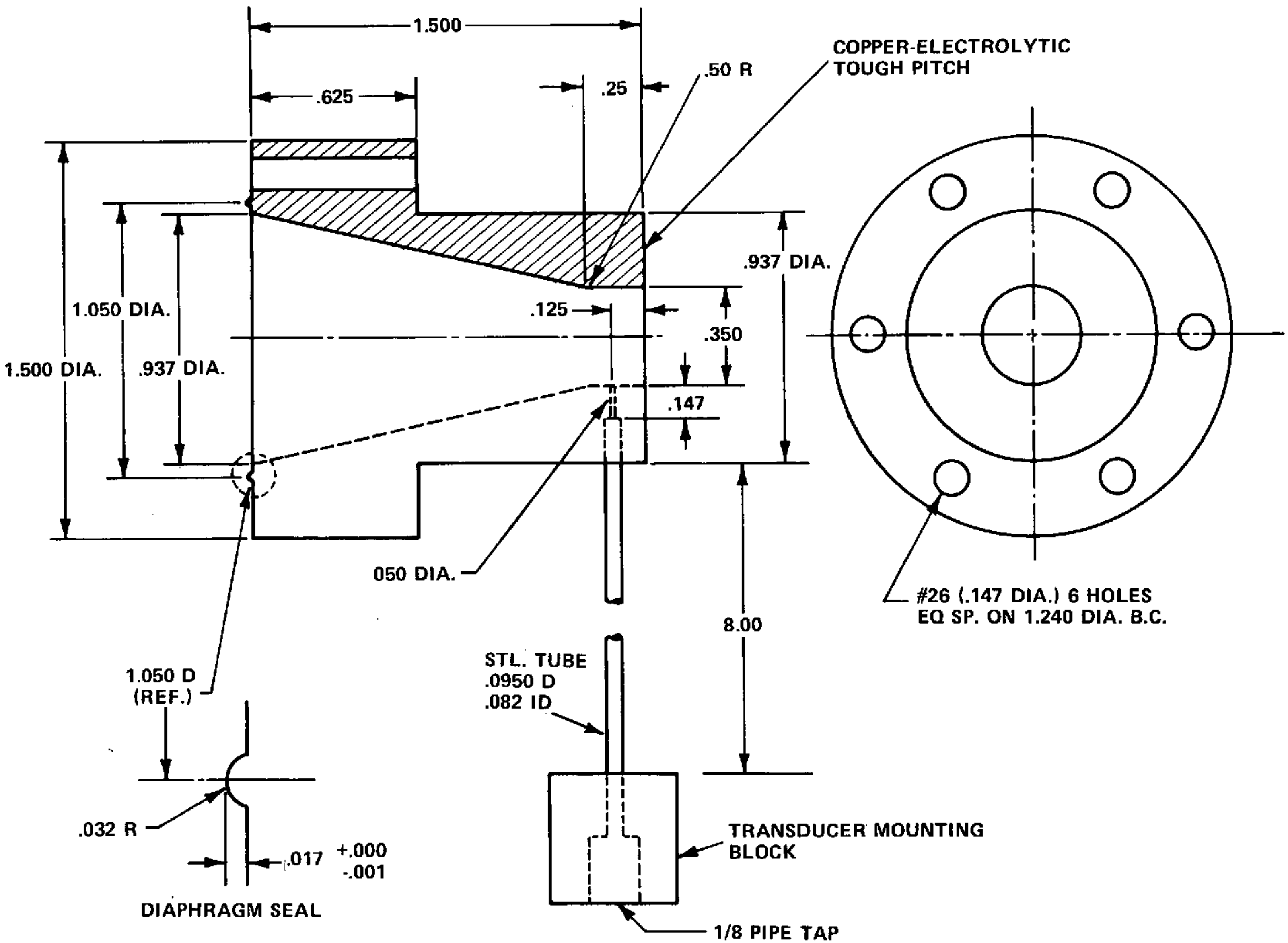


Figure 2 SOLID ROCKET MOTOR CALIBRATION NOZZLE - AREA RATIO =4

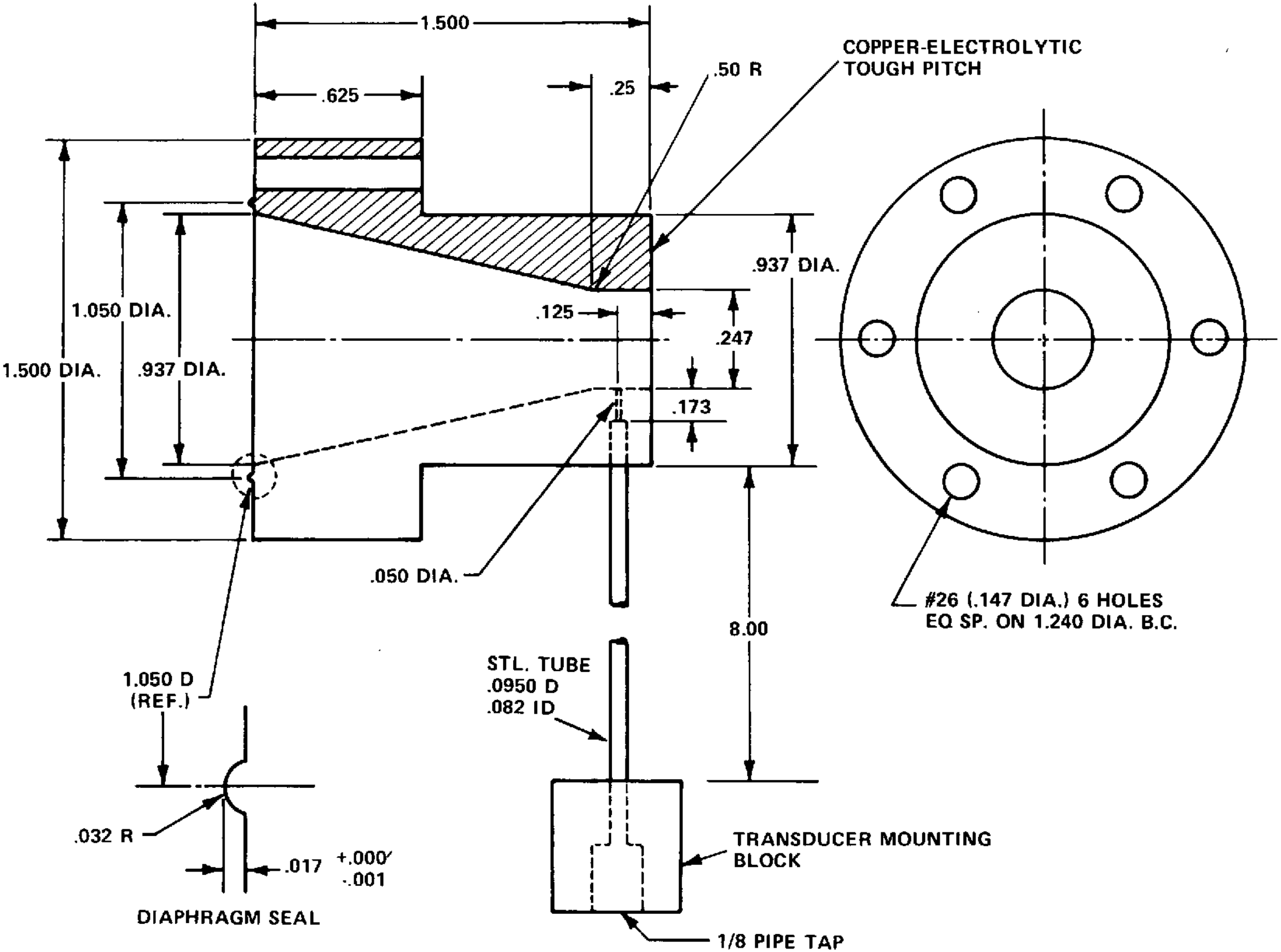


Figure 3 SOLID ROCKET MOTOR CALIBRATION NOZZLE - AREA RATIO = 8

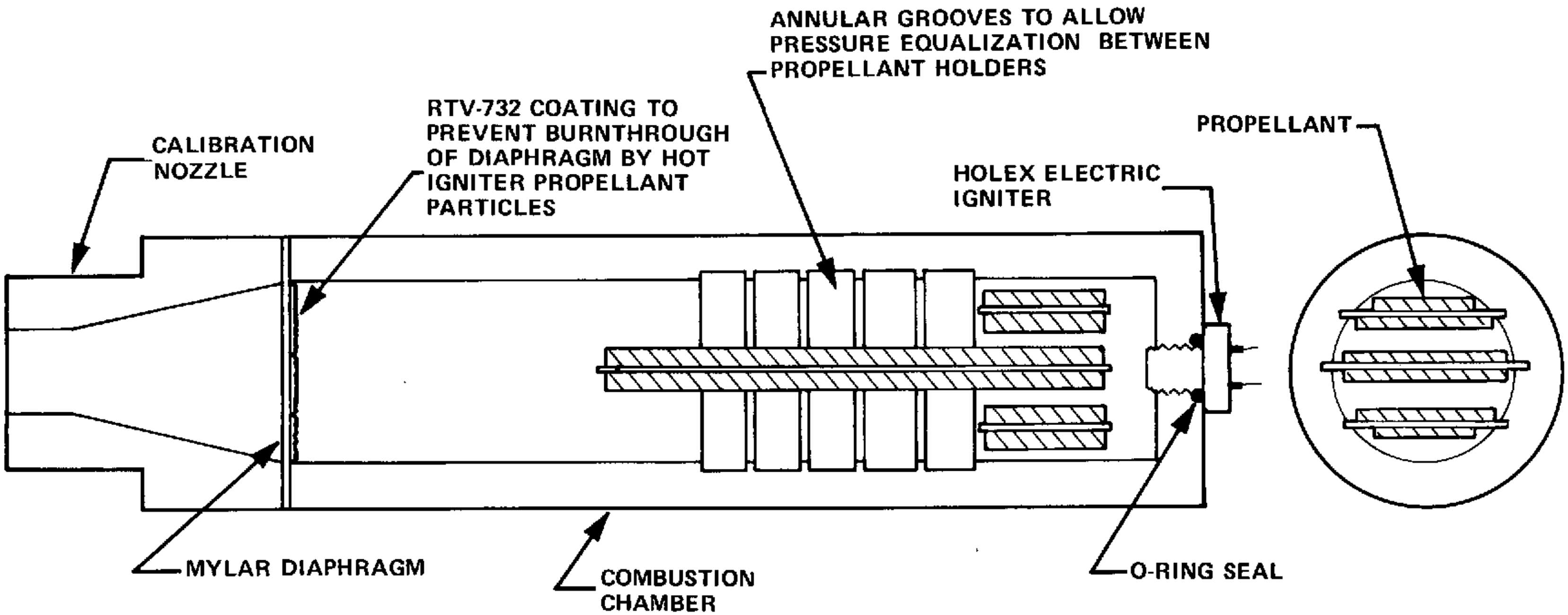


Figure 4 TYPICAL CALIBRATION TEST CONFIGURATION

The test rocket was mounted on a braced angle iron stand located in an underground ballistics range. Personnel testing the rocket motor were separated from the test chamber by a one-foot thick concrete wall, and observation of the rocket motor during a test was permitted by a 1.75 inch thick Lexan plastic window. A ventilation system located in the test cell allowed expulsion of the rocket exhaust gases from the test cell prior to personnel entry.

The solid rocket propellants selected to be used in the rocket motor were UTP-3001, a propellant containing 16% aluminum, manufactured by the United Technology Center of San Jose, California, and ANB-3335-1, a propellant containing 2% aluminum, manufactured by the Aerojet Solid Propulsion Company of Sacramento, California.

These propellants were selected by Calspan and approved by MSFC, both because of prior Calspan experience with their use and because of their substantially different aluminum content, thus allowing a good basis for comparison of the effect of metal content upon the aerodynamic characteristics of a hot rocket exhaust plume.

Thermodynamic data and chemical properties for both propellants are presented in Table I. Data obtained from the propellant manufacturers showing the variation of the ratio of propellant surface area to nozzle throat area, K_N, with rocket chamber pressure is presented in Figures 5 and 6.

Propellant thicknesses were selected to produce approximately 200 milliseconds of total burning time at a chamber pressure of 2000 psia for each propellant. Burning rate data supplied by each propellant manufacturer indicated that a thickness of 0.090 in. for the high aluminum content propellant (UTP-3001) and a thickness of 0.076 in. for the low aluminum content propellant (ANB-3335-1) would produce the required results.

TABLE I

PROPELLANT THERMODYNAMIC AND CHEMICAL PROPERTIES

Propellant Designation:	ANB-3335-1	UTP-3001
Molecular Weight:	25.20	23.97
Gas Constant (ft-lbf/lbm-^{o}R):	53.90	56.57
Flame Temperature in Combustion Chamber (^{o}R):	5340 (P_{CH} = 510 psia)	6100 (P_{CH} = 300 psia)
Specific Heat Ratio:	1.14	1.18
Aluminum Content (%):	2	16
Combustion Chamber Gas Constituents (%)		
$AlCl_2$	0.0*	--
Al_2O_3	0.90	7.00
Cl	0.00	1.00
CO	9.00	25.00
CO_2	18.00	1.00
H	0.00	3.00
H_2	20.50	28.00
HCl	--	13.00
H_2O	25.00	12.00
H_2S	0.03	--
N_2	9.00	8.00
OH	0.0	--

*0.00 shows an indication of less than 0.001%.

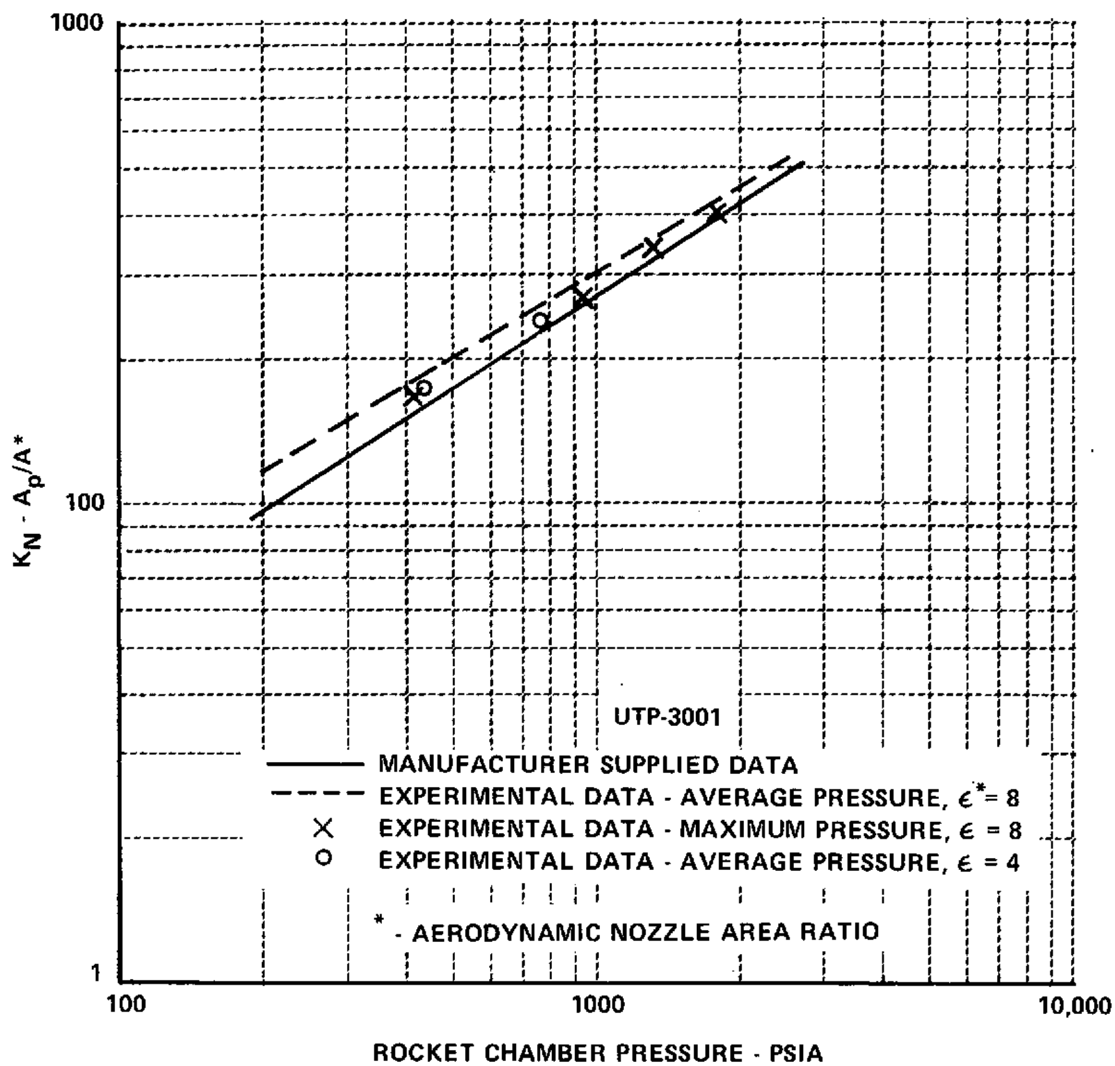


Figure 5 PROPELLANT SURFACE AREA TO NOZZLE THROAT AREA RATIO AS A FUNCTION OF PRESSURE - HIGH ALUMINUM CONTENT PROPELLANT

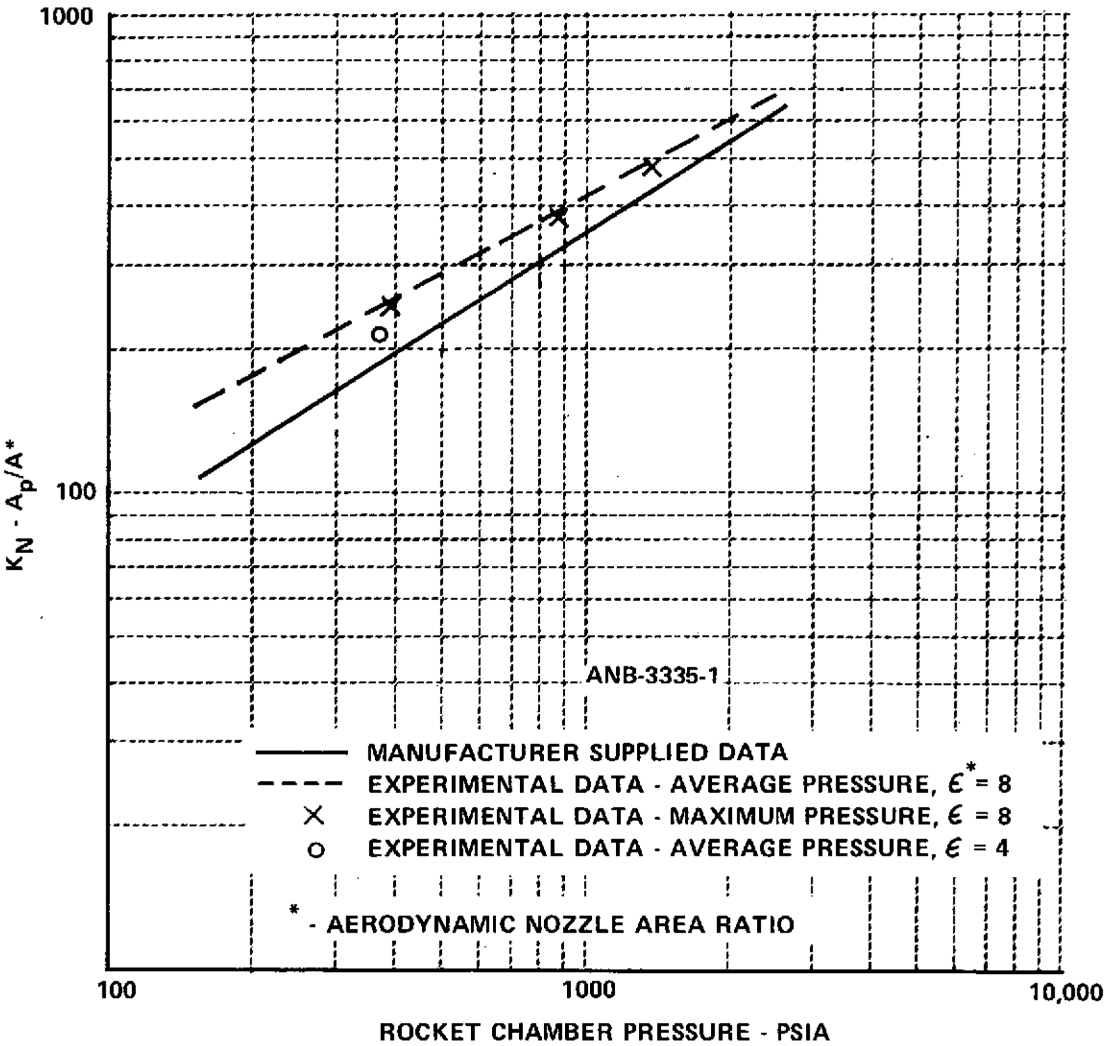


Figure 6 PROPELLANT SURFACE AREA TO NOZZLE THROAT AREA RATIO AS A FUNCTION OF PRESSURE - LOW ALUMINUM CONTENT PROPELLANT

The propellants were received from the manufacturers in 7 to 10 pound blocks, and a subcontractor was assigned the responsibility of slicing these blocks to a specified thickness of 0.090 to 0.095 in. for the UTP-3001 propellant and 0.075 to 0.080 in. for the ANB-3335-1 propellant. Measurement of 10 random samples of each propellant after receipt from the subcontractor indicated that the UTP-3001 propellant was 0.090 $\pm$0.008 in. thick and the ANB-3335-1 propellant was 0.079 $\pm$0.009 in. thick.

The solid propellant was bonded to the surfaces of the 0.025 in. thick 3003-H14 aluminum propellant holders using Saureisen contact cement, and in the final configuration a 0.030 to 0.050 in. thick coating of Dow-Corning 732 adhesive/sealant was placed on all exposed propellant edges to prevent their premature ignition. Propellant holder widths were established at 0.995 $^{+0.000}_{-0.005}$ in. for the center holder and 0.780 $^{+0.000}_{-0.005}$ in. for the side holders.

Figure 7 shows three representative propellant distributions tested during the calibration test phase of this program.

The ignition of the solid rocket propellant was accomplished through the use of a Holex Inc. electric pyrotechnic igniter. The igniter was initiated by applying a 45-volt D.C. pulse to the leads of the igniter from a Calspan-designed electronic firing circuit. The igniter configuration is shown in Figure 8. A schematic of the firing circuit control box is presented in Figure 9, and a diagram of the electronics of the firing circuit is shown in Figure 10.

Rocket chamber pressures were recorded at the two locations shown in Figure 1 during each test. Kistler Model No. 603A piezoelectric pressure transducers were used to record these pressures on Tektronix Inc. Model No. 502 and 5103N oscilloscopes.

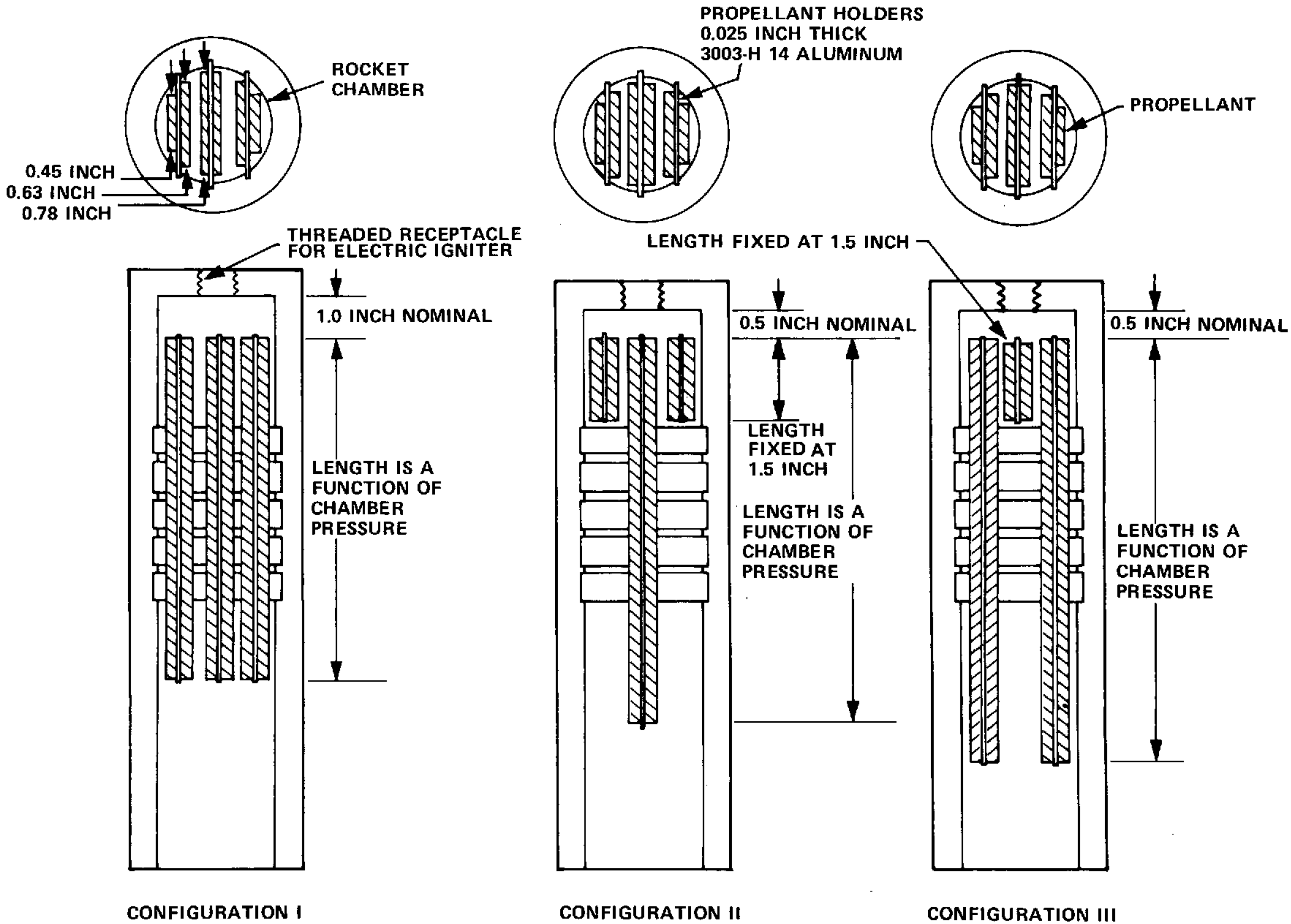
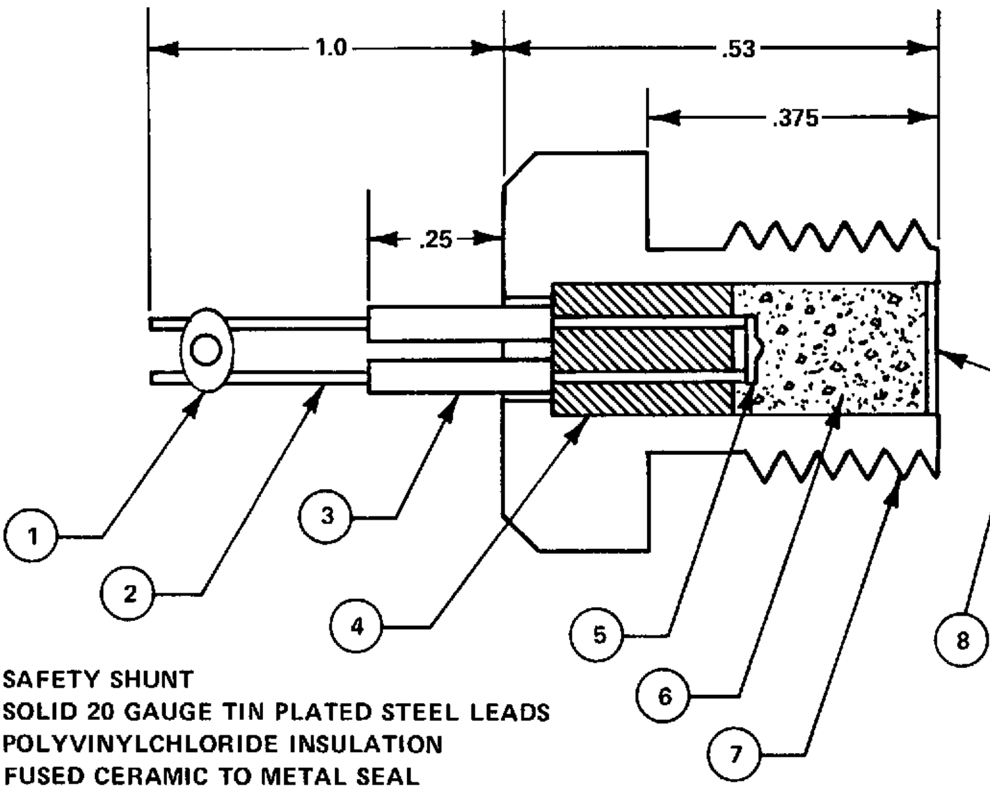


Figure 7 SOLID PROPELLANT DISTRIBUTION IN ROCKET CHAMBER



1. SAFETY SHUNT
2. SOLID 20 GAUGE TIN PLATED STEEL LEADS
3. POLYVINYLCHLORIDE INSULATION
4. FUSED CERAMIC TO METAL SEAL
5. PRIMED BRIDGE WIRE
6. EXPLOSIVE LOAD - HOLEX 1196A, 110 mg ZIRCONIUM BASED EXPLOSIVE, HOLEX 1196B, 160 mg FFFG BLACK POWDER
7. FUSION TIN PLATED STEEL BODY
8. NRC LAQUER SEALED ALUMINUM CLOSURE

Figure 8 ELECTRIC PYROTECHNIC IGNITER

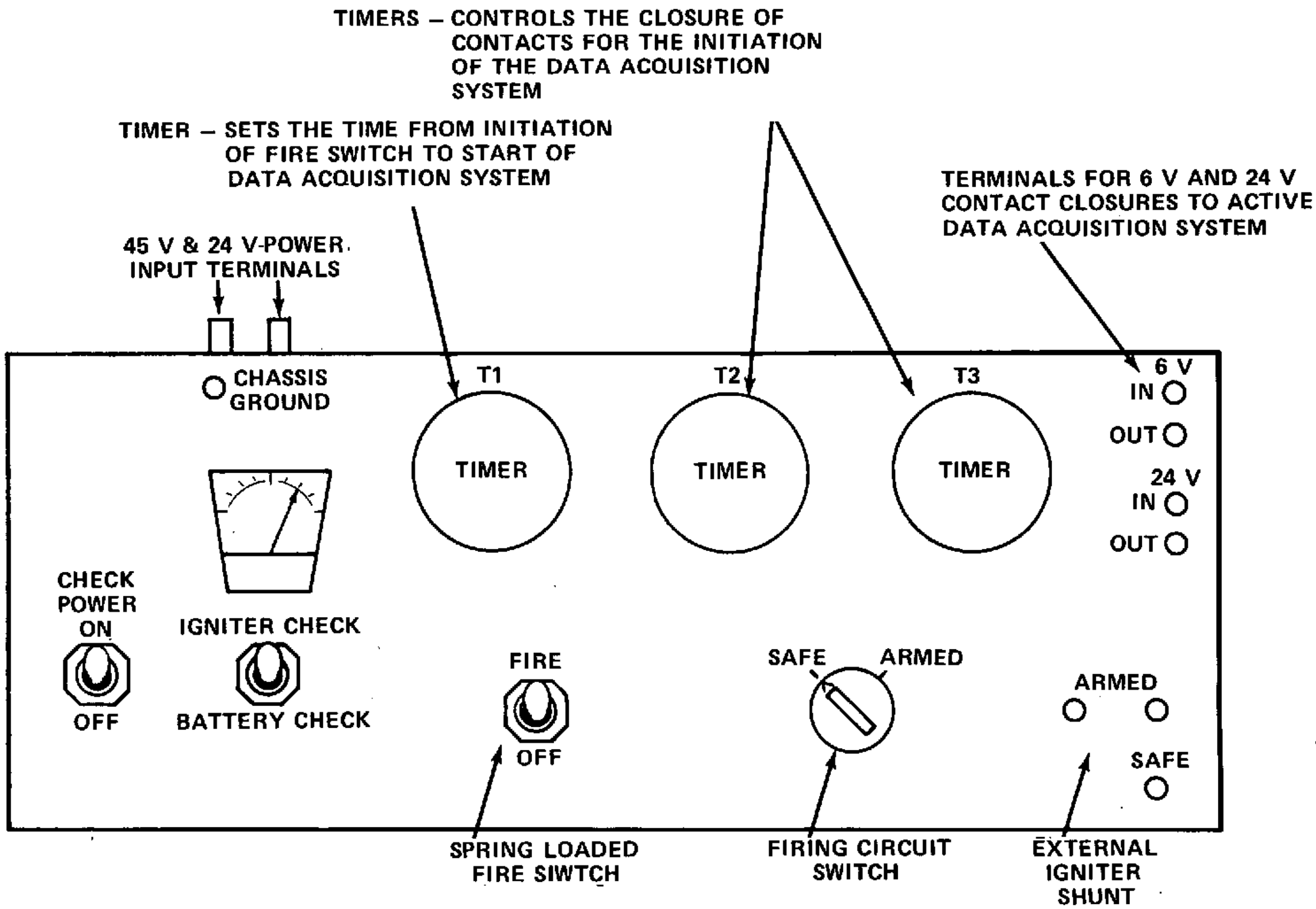


Figure 9 SCHEMATIC OF FIRING CIRCUIT CONTROL BOX

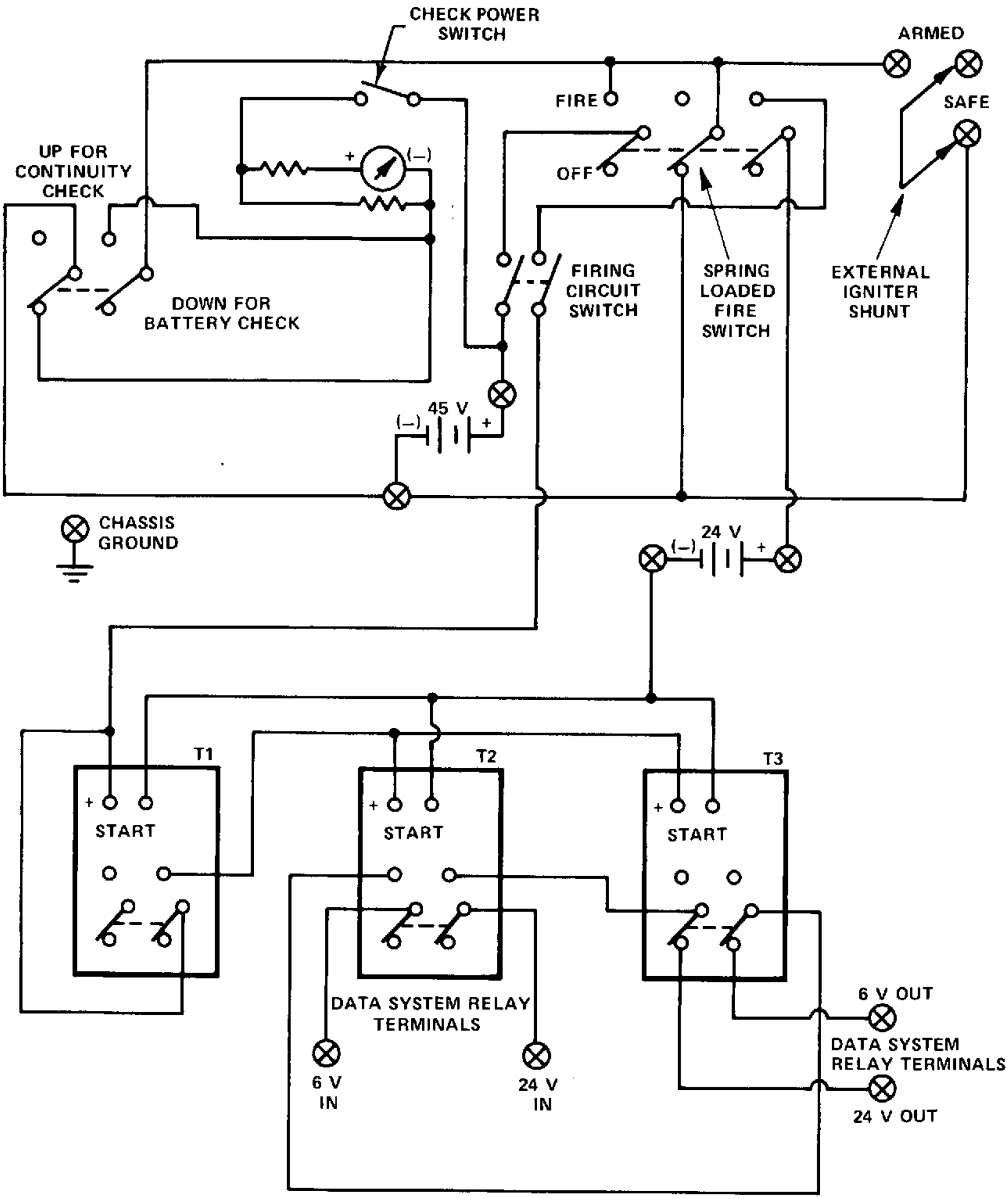


Figure 10 ELECTRICAL DIAGRAM OF FIRING CIRCUIT CONTROL BOX

Safety procedures were established for the conduct of all operations concerned with the testing of the live solid propellant rocket motors and were forwarded to NASA/MSFC for approval prior to completion of the calibration test phase at Calspan.

TEST PROGRAM

The following contains a description of the rocket motor calibration test program conducted at Calspan. The purpose of the test program was to establish the exact propellant configurations that were required to produce rocket chamber pressures and test durations that were consistent with the proposed test matrix.

Initially, experimental tests were concerned with the reliable ignition of the rocket propellant. The original propellant configuration consisted of three 10 in. long propellant holders on which propellant sections of equal length were bonded. The propellant was to be located at the nozzle end of the chamber to minimize the heat loss from the propellant gases to the chamber walls.

Data obtained during several tests indicated that the loss in energy to the chamber walls and exposed surfaces of the aluminum propellant holders by the igniter gas before it reached the propellant was sufficient to reduce its energy to a level that was insufficient to ignite the propellant. The propellant configuration was changed to that indicated by configuration I of Figure 7. A workable propellant holder length was established as the propellant length plus 0.2 in. Premature burn-through of the Mylar diaphragm by the hot igniter propellant particles caused failure of the propellant to ignite in several cases. A light coating of Dow Corning 732 adhesive/sealant applied to the inner surface of the diaphragm eliminated this problem.

The new propellant configuration was tested to insure the reliable ignition of the propellant. Ignition proved to be positive though the time delay between the firing of the igniter and the ignition of the propellant was erratic and rather long at times. Propellant ignition delay times were found to vary from 20 to 1600 milliseconds and were not repeatable for any particular propellant configuration. The resulting chamber pressure-time history following the ignition lag, however, proved to be independent of ignition delay time.

It was concluded that an igniter that produced a substantially higher energy output than the initially available Holex Inc. Model No. 1196A igniter might shorten the long ignition delay times and lead to better repeatability.

Approximate calculations indicated that an increase of 300% in the energy output of the igniter could produce ignition chamber pressures of 150 to 200 psia and substantially increase the heat transfer to the propellant surfaces during the ignition phase.

Unfortunately, long delivery times and the space limitations of the wind tunnel test model precluded the acquisition of other than the Holex Inc. Model No. 1196B electric igniter, which produced an energy only 35% greater than the existing igniter. Subsequent tests of this slightly more energetic igniter produced no significant improvement in the ignition delay time of the propellant.

Chamber pressure data obtained during the initial rocket motor firings indicated that ignition of the exposed propellant edges produced nonlinear (regressive) burning of the propellant. These pressure-time histories were not sufficiently flat to satisfy test requirements. Chamber pressure decay rates approaching 1.5 psi per millisecond were observed.

At this point, an investigation was initiated to develop an effective method to prevent ignition of the propellant edges. Various coating/thickness combinations were applied to the edges of the propellant, including Glyptol (an electrical insulating paint), Pyromark type 17A high temperature paint, Saureisen contact cement, Dow Corning 732 RTV adhesive/sealant, and mixtures of Tolual and 732 RTV (Tolual was added to thin the viscous RTV, making application easier). A zinc-based nonorganic paint was also ordered but was received too late to be included in the evaluation.

Tests to evaluate the various edge coatings were conducted at a chamber pressure of 400 psia with the area ratio 8 nozzle. It was determined that a 0.030 to 0.050 in. thick layer of Dow Corning 732 RTV effectively inhibited edge burning as evidenced by more than an order of magnitude decrease in the pressure decay rate during the nominal "steady" burn time from 1.5 to 0.1 psi/msec.

The effect of propellant edge coating on the chamber pressure-time history is presented in Figure 11, and a representative example of the post-test appearance of a propellant holder that had experienced propellant edge burning is presented in Figure 12.

With the edge burning problems resolved, effort was directed toward tests to obtain the higher chamber pressures with both the high and low aluminum content propellants. Initially, the propellant configuration consisted of three propellant holders of equal length positioned such that the propellant was located 1.0 in. from the electric igniter. (Figure 7, configuration I.) The results of several tests with both propellants indicated that if the propellant length was greater than 2.0 to 2.5 in., certain sections of the propellant would experience high burning rates. At times, this phenomenon pro-produced chamber pressures as high as 3500 psia. Damage to the rocket chamber and nozzle occurred on two separate occasions as a result of these high overpressures.

It was concluded that the accelerated burning rates (erosive burning) were associated with the increase in propellant gas mass flow through the small cross-sectional areas between the adjacent surfaces of the propellant. The areas which were subjected to the high burning rates were clearly evident from the appearance of the propellant holder surfaces after a test. Tests for which

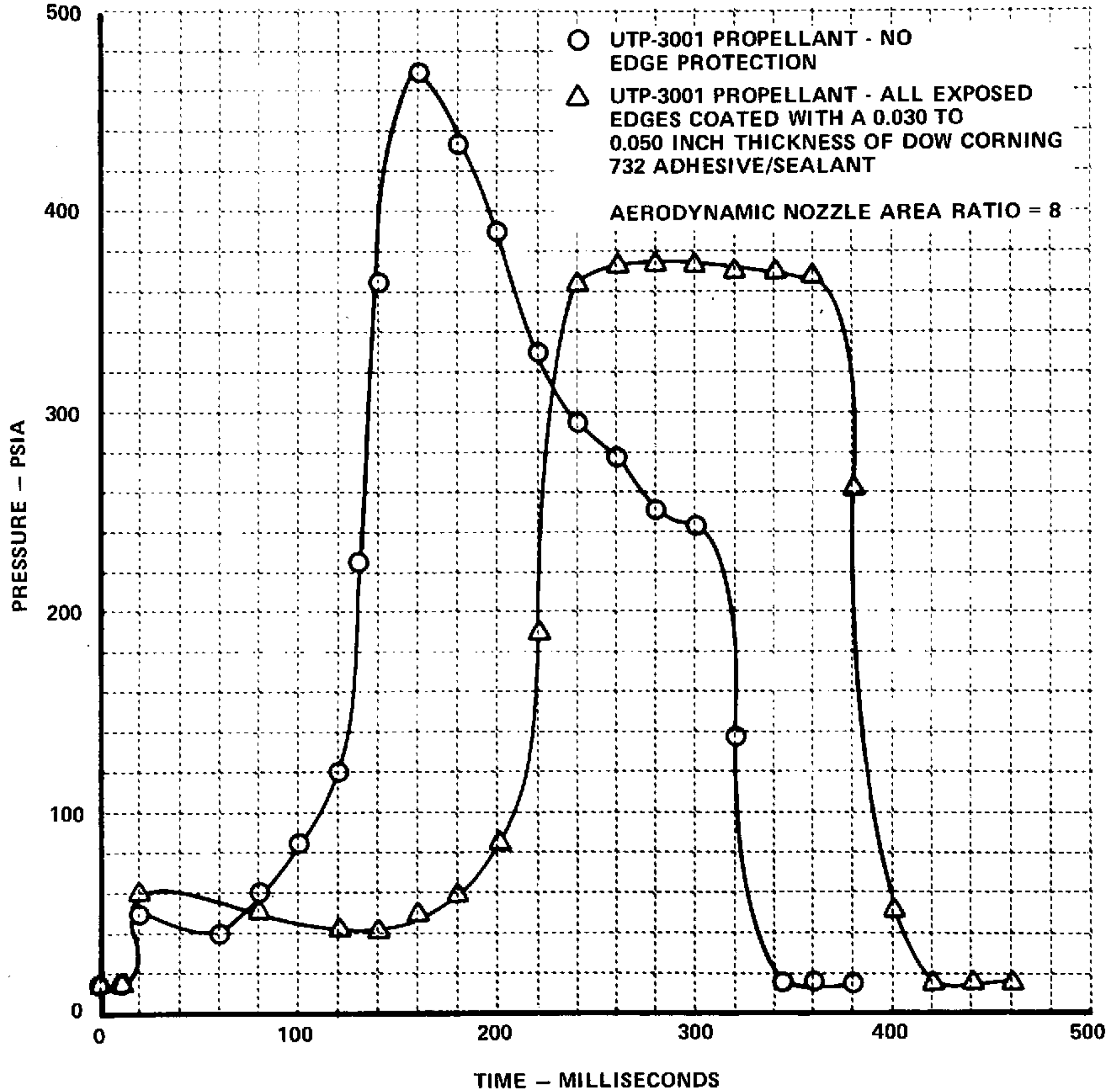


Figure 11 COMPARISON OF ROCKET CHAMBER PRESSURE – TIME HISTORY WITH AND WITHOUT PROPELLANT EDGE PROTECTION

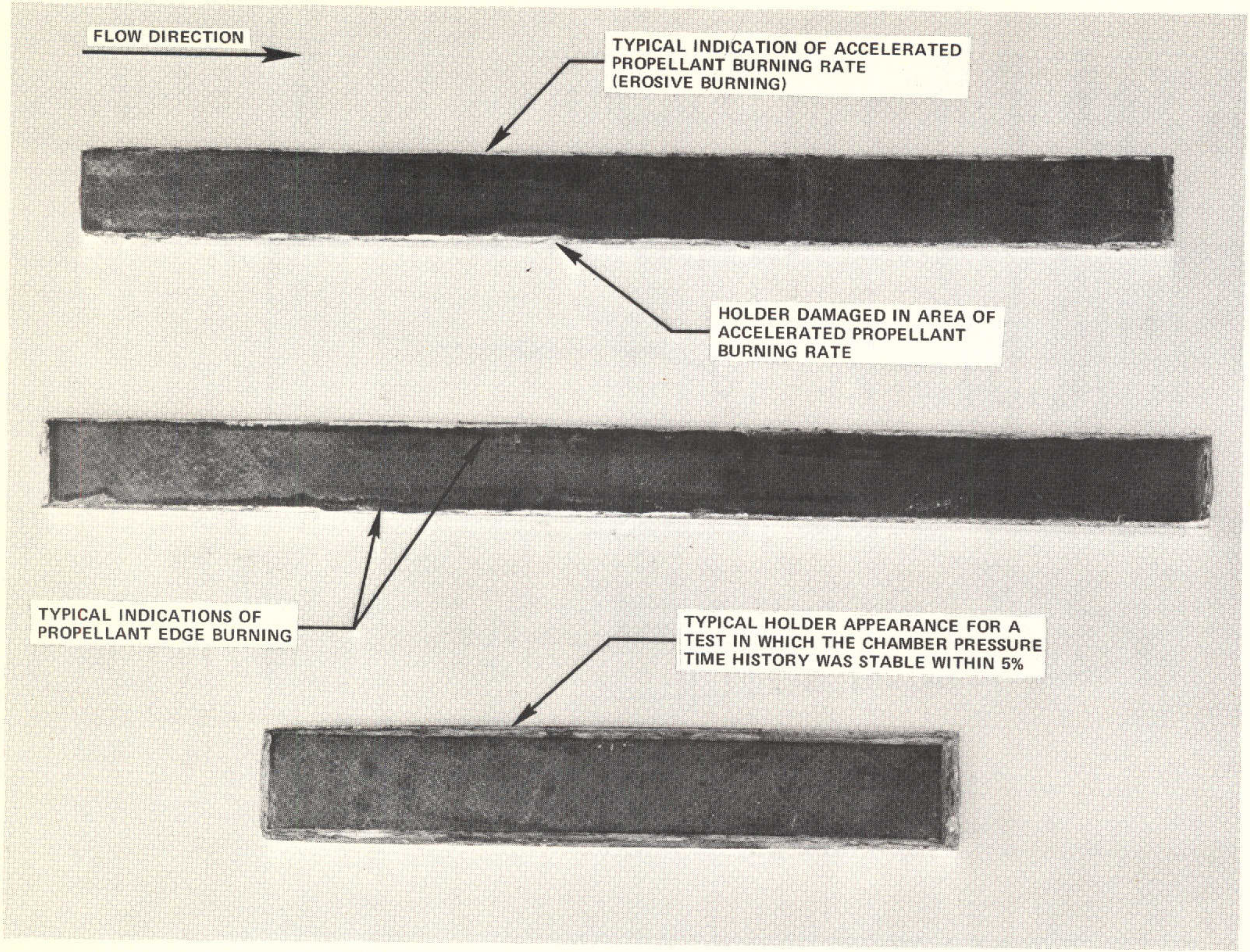


Figure 12 POST-TEST APPEARANCE OF PROPELLANT HOLDER SURFACE

evidence of erosive burning was detectable were also characterized by burning of portions of the aluminum propellant holders and, in some cases, by substantial changes in position of the propellant holders. Figure 12 shows the appearance of a propellant holder surface as a result of a mildly erosive propellant burn.

Various propellant configurations were tested to determine whether a configuration could be found which would minimize the possibility of erosive burning. The most promising configurations were found to be a long center propellant section with two short side sections (Figure 7, configuration II) and a combination of a short center propellant section and longer side sections. (Figure 7, configuration III). These configurations restricted the maximum propellant surface area to less than 23 square inches compared with a required maximum of 53 square inches. Hence, the propellant area limitation substantially reduced the proposed test matrix for the area ratio 4 nozzle.

The remainder of the calibration test phase was concerned with obtaining the exact propellant surface areas that would produce the chamber pressure levels specified by the reduced test matrix.

Propellant configurations II and III of Figure 7 were tested exclusively with preference given to configuration II whenever possible. Configuration II allowed a maximum propellant surface area of only 18.1 square inches, but the propellant distribution in the rocket chamber allowed larger cross-sectional flow areas for a greater portion of the propellant length than did the propellant distribution of configuration III.

A total of 39 tests were required to establish the repeatability and absolute chamber pressure levels of the various test configurations.

The maximum chamber pressures attainable using the low aluminum content propellant were 1200 psia for the area ratio 8 nozzle and 400 psia for the area ratio 4 nozzle. Both of these limits were established by the maximum propellant surface area allowed by propellant holder configuration III.

The maximum chamber pressures attained for the high aluminum content propellant were limited to 1600 psia with the area ratio 8 nozzle and 400 psia with the area ratio 4 nozzle. An additional point in the test matrix for the area ratio 4 nozzle was obtained when it was decided to add a test which used a maximum propellant load. A slightly elevated propellant burning rate at the nozzle end of the propellant load resulted in a change of 15% from the maximum chamber pressure recorded over the 150 millisecond test time. The average chamber pressure over the same time interval was 754 psia.

The factors limiting the test matrix for the high aluminum content propellant were erosion of the area ratio 8 nozzle throat at the 2000 psia test point and the limited propellant surface area of propellant holder configuration III for the area ratio 4 nozzle.

Chamber pressure variations of from 3.9 to 12.8% were recorded over the required 150-millisecond test duration. A reduction in the width of the propellant sections on the center propellant holder and the inboard surfaces of the side propellant holders allowed application of RTV edge coatings that were of more uniform thickness. This change was attempted prior to conduct of the repeat tests for each test point and is believed to account for the slight improvement in chamber pressure stability.

Inspection of the rocket chamber and nozzle after each test in which the high aluminum content propellant was used indicated that a coating, suspected to be aluminum oxide, was forming on the interior surfaces of the rocket chamber and nozzle. The configuration of the nozzle throat after a test varied from a complete coating of the throat circumference approximately 0.010 to

0.012 in. thick to the absence of a coating. In a slight majority of cases, the throat was fully coated. It is believed that the coating configuration in the forward nozzle cone and throat was continuously changing during a test.

A total of 110 live rocket firings were completed during the calibration test program. Fifty propellant holder configurations were assembled at Calspan and transported with 75 electric igniters and a quantity of unassembled propellant and propellant holders to the Marshall Space Flight Center.

A representative of Calspan was present at Marshall Space Flight Center on the 26th and 27th of November 1973 to present a review of the rocket motor calibration tests and to supervise the conduct of two successful rocket motor firings in preparation for the aerodynamic test phase of the program.

TEST RESULTS AND OBSERVATIONS

The results of the rocket motor calibration tests are presented in this section. Also included are descriptions of the phenomena experienced during these tests and qualitative observations based on the results of the many tests which were not directly applicable to the final test matrix.

The calibration tests selected to represent the test matrix for the aerodynamic test program to be conducted in MSFC's 14 x 14 inch trisonic wind tunnel exhibit a chamber pressure stability, over 150 milliseconds of test time, of from 3.9 to 12.8% of the maximum pressure recorded during a test. A reduction of propellant width prior to conduct of the repeat tests for each pressure level succeeded in improving the pressure stability slightly. The repeatability between tests at a particular chamber pressure varied from 0.6 to 4.6%. Tabulated values of the results of the calibration tests are presented in Table II. This table includes the propellant surface area; the resulting maximum, minimum, and average chamber pressures; and the percent change in chamber pressure over 150 milliseconds of test time. Table III, "Rocket Motor Test Information," includes a correlation between propellant type, chamber pressure, and propellant configuration. Table III was forwarded to M.S.F.C. prior to conduct of the wind tunnel test program to be used as an integral part of the test safety procedures as specified by Calspan. A representative chamber pressure-time history for each of the propellant types is shown in Figures 13 and 14.

A time exposure photograph of a typical high aluminum content propellant rocket plume is presented as Figure 15.

The processes which govern the ignition characteristics of a solid propellant are both physical and chemical in origin, delays being associated

TABLE II

ROCKET MOTOR CALIBRATION TEST RESULTS

Propellant	Nominal Chamber Pressure (psia)	Nozzle Area Ratio	Propellant Surface Area (in^2)	Maximum Chamber Pressure (psia)*	Minimum Chamber Pressure (psia)*	Average Chamber Pressure (psia)*	Percent Change In Chamber Pressure*
ANB-3335-1	400	8.0	12.0	407	385	396	5.4
	400	8.0	12.0	385	370	378	3.9
ANB-3335-1	800	8.0	18.2	880	803	841	8.3
	800	8.0	18.2	847	811	829	4.3
ANB-3335-1	1200	8.0	22.9	1340	1205	1273	10.0
	1200	8.0	22.9	1355	1215	1285	10.2
ANB-3335-1	400	4.0	21.2	385	370	378	3.9
	400	4.0	21.2	378	355	367	6.1
UTP-3001	400	8.0	8.1	420	375	398	10.7
	400	8.0	8.1	400	370	385	7.5
UTP-3001	800	8.0	13.0	835	735	785	12.8
	800	8.0	13.0	815	770	793	5.5
UTP-3001	1200	8.0	16.7	1325	1155	1240	12.8
	1200	8.0	16.7	1245	1185	1215	4.8
UTP-3001	1600	8.0	19.2	1740	1555	1648	10.6
	1600	8.0	19.2	1790	1615	1703	9.8
UTP-3001	400	4.0	16.8	445	415	430	6.7
UTP-3001	750	4.0	22.9	815	693	754	15.0**

*Test data are based on a 150-millisecond test duration.

**Evidence of erosive propellant burning was present for this test.

TABLE III

ROCKET MOTOR TEST INFORMATION

Propellant Description	Propellant Color	Average Chamber Pressure (psia)	Propellant Configuration*	Active Propellant Charge Length (in.) Sides	Active Propellant Charge Length (in.) Center	Nozzle Area Ratio	Mylar Diaphragm Material Thickness (in.)
ANB-3335-1	Black	390	II	1.50	5.60	8	0.005
		835	II	1.60	9.50		0.007
		1280	III	9.50	1.50		0.009
		375	III	8.75	1.50	4	0.005
UTP-3001	Red	395	II	1.50	3.15	8	0.005
		790	II	1.50	6.20		0.007
		1230	II	1.50	8.60		0.009
		1680	III	7.80	1.50		0.011
		430	II	1.50	8.55	4	0.005
		750	III	9.50	1.50	4	0.007

NOTE: A Holex electric igniter No. 1169B is to be used for all propellant configurations.

* See Figure 7 for configuration details.

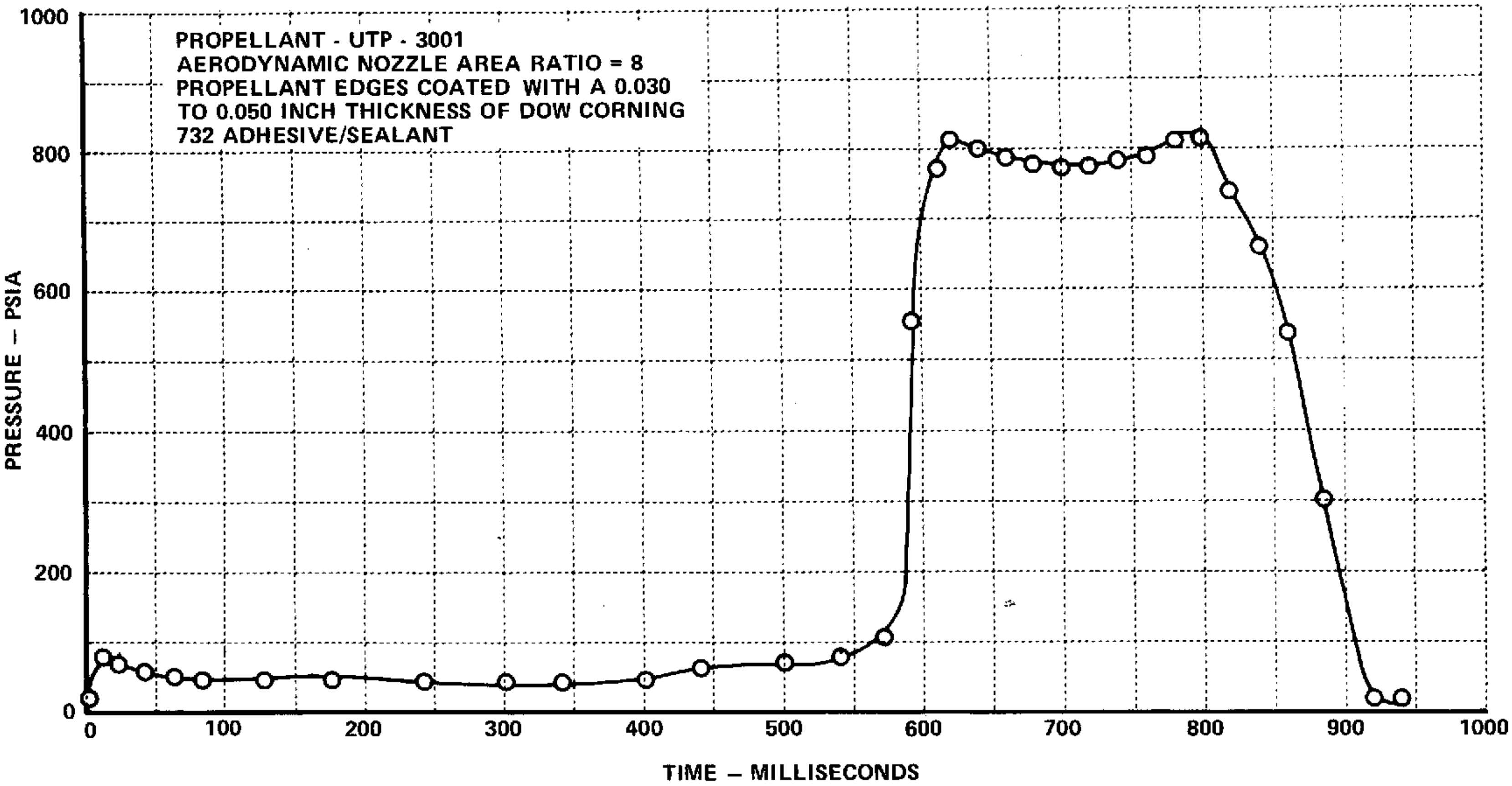


Figure 13 TYPICAL ROCKET CHAMBER PRESSURE – TIME HISTORY – HIGH ALUMINUM CONTENT PROPELLANT

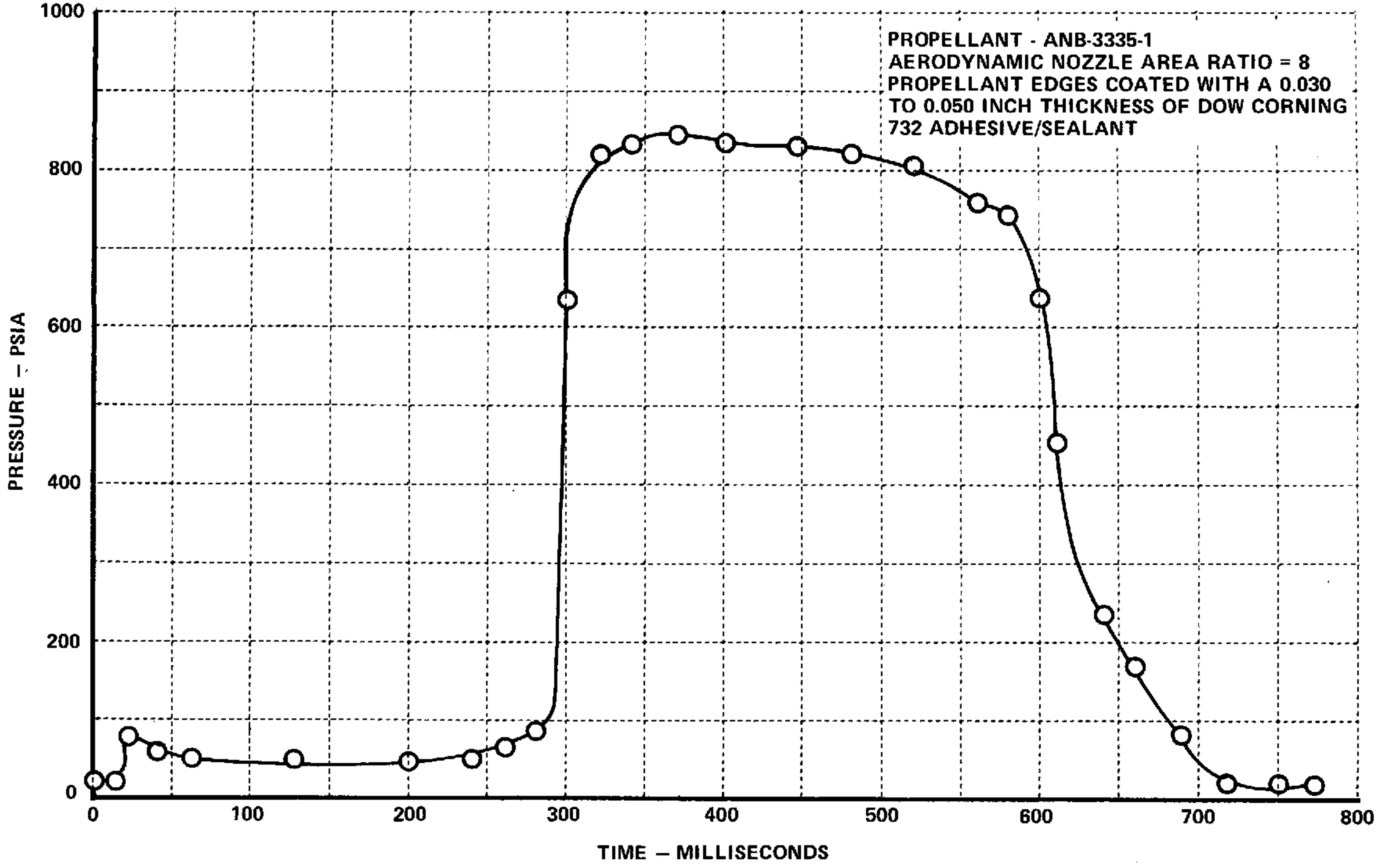


Figure 14 TYPICAL ROCKET CHAMBER PRESSURE – TIME HISTORY – LOW ALUMINUM CONTENT PROPELLANT

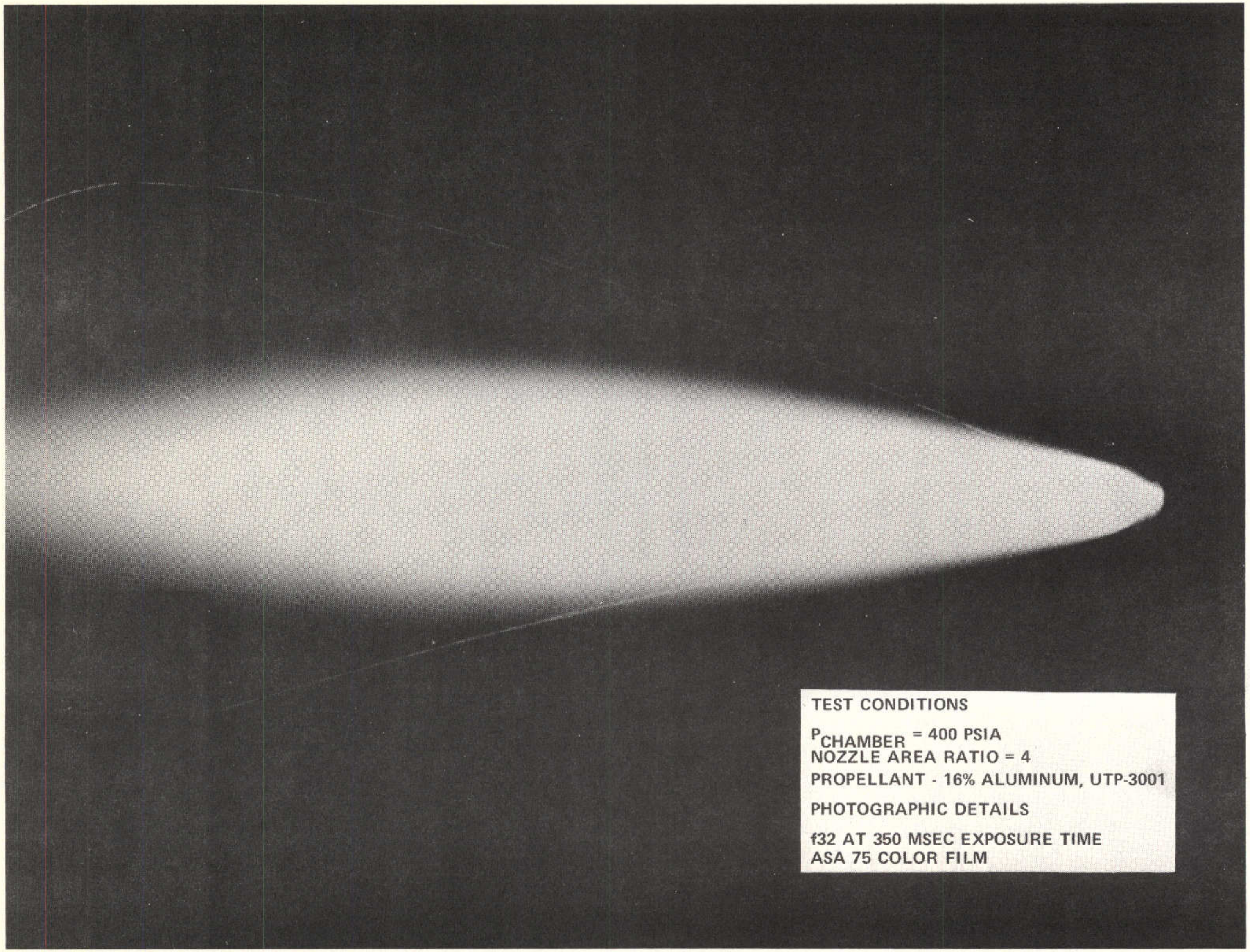


Figure 15 TIME EXPOSURE PHOTOGRAPH OF A SOLID PROPELLANT ROCKET PLUME

with the vaporization and mixing of the product vapors and solid particles, particle size, pressure, and the rate at which heat is transferred to the surface of the propellant.

Propellant ignition during the calibration program was characterized by large variations in ignition time. As was noted earlier, the time for the propellant to ignite varied from 20 to 1600 milliseconds.

The maximum pressure developed by the electric igniter in the rocket chamber during the calibration tests exhibited a random variation of from 45 to 85 psia. The maximum pressure was below 75 psia for a large majority of tests.

The black powder in the igniter is ignited by a resistance element, around which is placed a small quantity of zirconium-based propellant. As the primer powder ignites the black powder charge, the pressure developed in the igniter cavity ruptures the thin aluminum closure of the cavity, blowing both burning and unburned black powder grains into the rocket chamber.

Observations made during the calibration program indicate that the primary propellant charge of the igniter is distributed along the length of the rocket chamber at the time of ignition.

The apparent variation in output of the igniter during a particular test may be attributed to that portion of the black powder grains for which combustion is partially or completely inhibited as a result of contact with the cold chamber walls or the lack of a sufficiently high gas temperature in the downstream portions of the chamber.

Closed bomb tests of black powder predict burn times of the order of 20 to 30 milliseconds. In most cases, the black powder charge of the electric igniter was pyrolized long before propellant ignition occurred. For this reason, it is believed that the primary mechanism causing propellant ignition is the heat transferred from the hot chamber gases to the propellant surface.

Tests in which the electric igniter produced a maximum chamber pressure greater than 75 psia were characterized by ignition delay times of less than 300 milliseconds. This pressure level corresponds to a chamber gas temperature after mixing of the hot igniter produced gases with the initial volume of ambient air in the chamber of approximately 1200°F.

It appears that the ignition delay time of the propellant can be shortened substantially by increasing the thermal energy output of the igniter such that igniter pressures developed in the rocket chamber are of the order of 100 psia. Of course, experimental tests would be required to verify this conclusion.

Figures 11, 13, 14, and 16 show typical pressure-time histories produced by the electric igniter prior to propellant ignition.

The burning of composite rocket propellants can be described as the gasification, mixing, and ignition of separate streams of fuel, oxygen, and solid particles from the surface of the propellant. The propellant burning rate depends, among other things, on the rate of surface decomposition and, at higher pressures, the energy responsible for this decomposition comes from a flame zone that is displaced from the propellant surface by a zone in which the fuel and oxygen are mixed. The rate of energy transfer to the propellant surface depends on the separation of the flame from the burning propellant surface and on the thermal conductivity of the mixing zone. The increase in the linear burning rate of the propellant can be attributed to the displacement of the flame zone toward the propellant surface and also to an increase in the effective thermal conductivity and diffusion coefficients in the developed small-scale, turbulent, product flows close to the propellant surface as the velocity of the combustion products over the propellant surface increases beyond some limiting value. The sensitivity of the linear burning rate of the propellant to the gas flow velocity parallel to the burning surface is usually called erosive burning.

Local flow velocities prior to the onset of erosive burning were calculated for several tests based on the initial geometry of the propellant gas flow area and the position at which erosive burning appeared to start, as taken from the post-test appearance of the propellant holder. The point at which erosive burning became evident usually was located 2.0 to 2.5 in. from the forward end of the propellant holders. Calculated local flow velocities prior to the start of erosive burning ranged from 325 ft/sec for the high aluminum content propellant to 385 ft/sec for the low aluminum content propellant. Velocities of greater than 1000 ft/sec have been computed at locations downstream of the initial indication of erosive propellant burning by assuming linear burning rates only 20% greater than those presented by the propellant manufacturer. Separate experiments conducted by Zucrow (6), Larue (7) and Herron (8) indicate that linear burning rates will be increased 1.6 to 3.0 times when the gas velocity reaches 1500 ft/sec, depending on the chemical composition of the solid propellant. Other experiments by Vilyunov and Dvoryashin (9) suggest that velocities as low as 500 ft/sec can produce increases of 10 to 15% in the linear burning rate of certain types of solid propellants.

The results of increased propellant linear burning rate have a direct effect on rocket chamber pressure stability, and the magnitude of this effect can be great enough to cause structural damage to the rocket chamber, aerodynamic nozzle, or instrumentation of a rocket motor. For these reasons, the flow velocity in the rocket chamber must be maintained at a value which will preclude the possibility of erosive propellant burning.

The post-test appearance of the holder surface thought to represent the presence of erosive burning is shown in Figure 12. A pressure-time history of a propellant configuration that experienced severe erosive burning is presented in Figure 16.

As was mentioned earlier, tests in which the chamber pressure-time histories indicated erosive propellant burning were also characterized, upon occasion, by propellant holder movement and the burning of holder material.

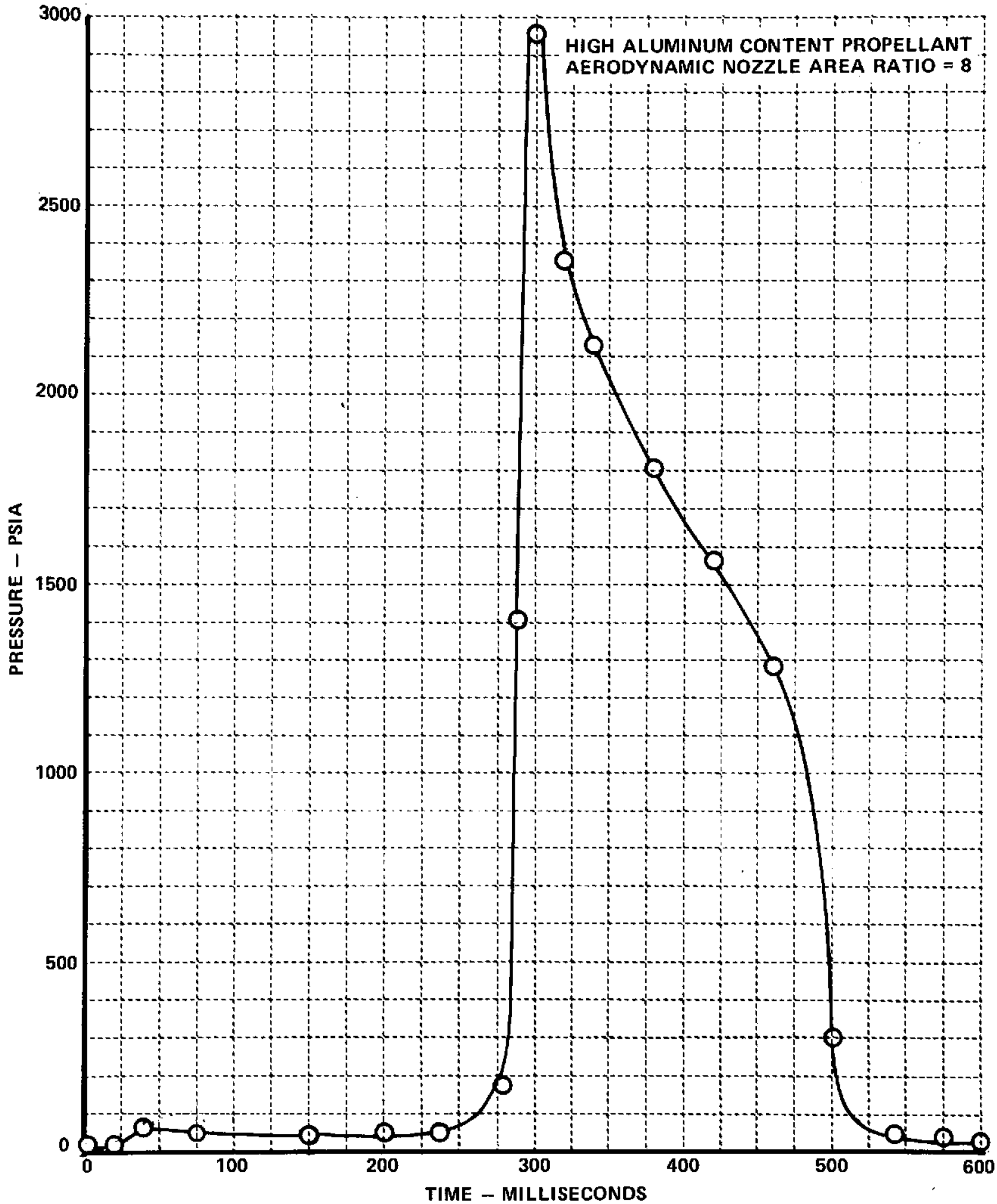


Figure 16 ROCKET CHAMBER PRESSURE – TIME HISTORY AS A RESULT OF EROSIVE PROPELLANT BURNING

The propellant holders generally exhibited material removal in the area of the annular grooves of the rocket chamber (see Figure 1) and in that area exposed to erosive propellant burning.

The material removal in the area of the annular grooves of the rocket chamber appeared to be caused by flow across the exposed edges of the side propellant holder into the outside flow channel formed by the side holder and the chamber wall. The burning of holder material in that area exposed to erosive propellant burning can be attributed to the loss of thermal insulation of the holder surface by the propellant prior to the complete pyrolysis of the propellant sections upstream of the erosive burning. The resulting flow over the exposed holder surfaces, in some cases, was sufficiently long to thermally erode portions of the propellant holder.

The rearward movement of the propellant holders during tests experiencing erosive burning is thought to be caused by a pressure differential between the upstream and downstream ends of the propellant holders that produced a net thrust in the direction of the aerodynamic nozzle with slight contributions from the skin friction drag at the surface of the propellant as the result of the high velocities in the area between the propellant surfaces.

A sample of the coating present on the upstream nozzle contour after a test of the high aluminum content propellant was examined using an Etec Autoscan scanning electron microscope with a Kevex Ray x-ray energy spectrometer attachment. Interpretation of the data obtained from the spectrometer revealed that the coating was primarily aluminum oxide; small quantities, less than 1%, of copper, iron, and hydrogen chloride were also present.

During the experimental tests, it was found that the propellant performance deviated somewhat from that suggested by data obtained from the propellant manufacturer. Figures 5 and 6 show the comparison of the experimentally determined K_N (ratio of propellant surface area to nozzle throat area) and data supplied by the manufacturer.

The experimental K_N was 21 and 30% higher than the manufacturers' data at 400 psia chamber pressure for the high and low aluminum content propellant, respectively. As the chamber pressure increased, the experimental data approached that of the manufacturer for both propellants, although the high aluminum content propellant performed more closely to manufacturers' specifications for all pressures tested.

Characteristic velocities for each propellant were computed both from experimental data* and from the propellant flame temperature supplied by the manufacturer. A comparison of these data at 400 psia chamber pressure indicates (C^*) efficiencies of approximately 97% for the high aluminum content propellant and approximately 93% for the low aluminum content propellant; these values correspond to approximately 7% and 14% reductions in flame temperature from the theoretical value. The coating of aluminum oxide that formed on all interior chamber and nozzle surfaces during tests of the high aluminum content propellant may have been responsible for the apparent decreased flame temperature of the combustion products.

Thermal erosion of the nozzle throat was experienced during one of two tests conducted at a chamber pressure of 2000 psia. On the basis of these tests, it was concluded that operation at this pressure level was marginal and would not be included in the test matrix. The presence of an 0.050 in. diameter pressure orifice in the nozzle throat did not cause or contribute to the thermal erosion noted.

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* An average value based on an integration of the Pc(t) history and total propellant weight; i.e., $\overline{C^*} = \frac{\int P_c dt A^* g}{W}$

CONCLUSIONS

The results of the rocket motor calibration tests indicate that there are several phenomena concerned with the development of miniature, solid propellant rocket motors that can substantially affect their design, performance and safety.

The linear burning rate of a solid propellant can be affected by the magnitude of the propellant gas flow velocity parallel to the propellant surface. This effect may range from a decrease of the burning rate, from that measured in a solid strand burner at quiescent flow conditions, to an increase of several hundred percent depending on the magnitude of the flow velocity and the chemical composition of the solid propellant.

If the solid propellant is of sufficient thickness, burning of the propellant edges may contribute to pressure transients which preclude the measurement of parameters that require steady flow be developed, for example, the rocket exhaust plume shape and associated nozzle static pressures.

Solid propellants having a high aluminum content produce relatively large quantities of aluminum oxide in their exhaust gases. The solidification of the aluminum oxide on the surfaces of the exit nozzle contour and throat can produce changes in the geometry of the nozzle. This effect is particularly important in small throat diameter nozzles where thin coatings can cause a large percentage change in the throat area thus increasing the effective area ratio of the nozzle.

If the energy of an igniter is not sufficient to produce adequate heat transfer rates to the propellant surface, the result may be erratic ignition times. The chamber pressure-time history appears to be independent of the ignition delay time.

Thermal erosion of material in the copper, exit nozzle throat occurred during one of two tests at 2000 psia chamber pressure. The solution to this problem may be found in water cooling of the nozzle throat or using a material, such as tungsten, that has a sufficiently high melting temperature.

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