Martin

Martin-CR-65-93 (Vol I) Copy No. 2

SOLID PROPELLANT ROCKET EXHAUST EFFECTS (SPREE) AND METHODS OF ATTENUATION

VOLUME I PROJECT SUMMARY

JANUARY 1966

prepared by
MARTIN MARIETTA

GPO PRICE $ ________

CFSTI PRICE(S) $ ________

Hard copy (HC) 2.00

Microfiche (MF) .50

ff 653 July 65

FACILITY FORM 602

N66 30606	
(ACCESSION NUMBER)	(THRU)
40	1
(PAGES)	(CODE)
CR-76236	33
(NASA CR OR TMX OR AD NUMBER)	(CATEGORY)

prepared for
KENNEDY SPACE CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
JOHN F. KENNEDY SPACE CENTER, FLORIDA

Martin-CR-65-93 (Vol I) Copy No. 2

Contract NAS10-2300

SOLID-PROPELLANT ROCKET EXHAUST EFFECTS (SPREE) AND METHODS OF ATTENUATION

VOLUME I: PROJECT SUMMARY

January 1966

Authors

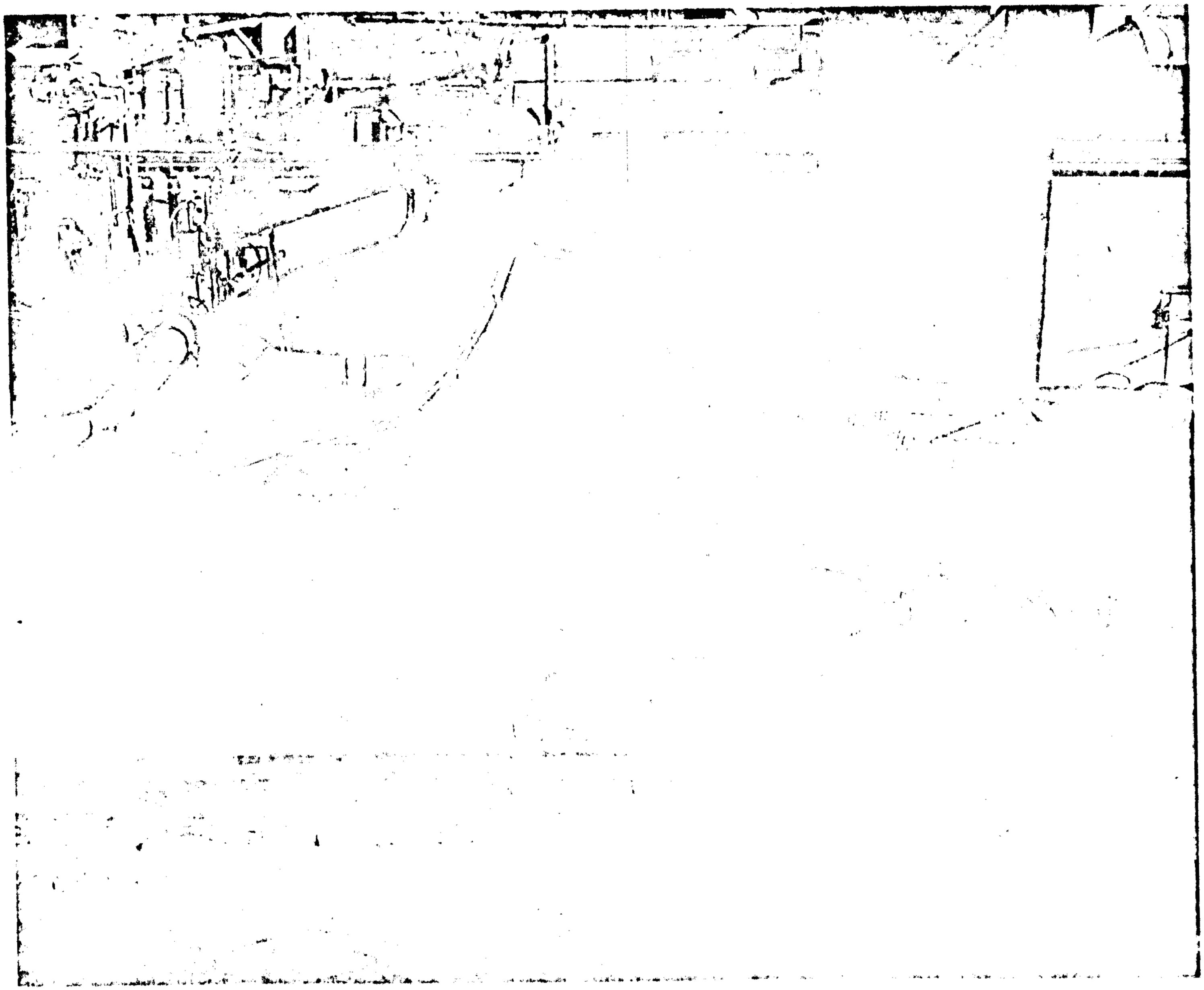
E. A. Darrow, BSMS, MSEE, P.E.
E. Lays, BSAE, P.E.

Approved

E A Darrow

E. A. Darrow
Program Manager, SPREE

MARTIN-MARIETTA CORPORATION
MARTIN COMPANY
Denver Division
Denver, Colorado



Dual-Motor Firing

FOREWORD

This report provides the data developed during Part II, Phase II of the Solid-Propellant Rocket Exhaust Effects (SPREE) Program under Contract NAS10-2300 and also includes data previously provided from the Phase I and Part I, Phase II SPREE investigations performed under Contracts NAS10-389 and NAS10-1107. The three contracts have been administered by NASA at the Kennedy Space Center, John F. Kennedy Space Center, Florida. Technical direction of the program has been provided by the Future Studies Branch of the Launch Support Equipment Engineering Division, Kennedy Space Center. The report is submitted in two volumes that are described below.

Martin-CR-65-93 (Vol I), SPREE Project Summary - This volume is submitted as a project summary of all work performed, results obtained, and recommendations evolved during the entire SPREE program.

Martin-CR-65-93 (Vol II), Part II, Phase II Final Report - This volume documents the results of the test program performed during the Part II, Phase II portion of the program. Where applicable, this volume also updates portions of the data previously provided by earlier reports developed during Part I, Phase II of the SPREE program. Also reported are data obtained from full-scale tests, which were made available for this effort and which have been used to correlate subscale model test results developed during this program.

A design handbook will be issued at a later date, which can be used as a guide in the design of portions of launch facilities from which vehicles using solid rocket motors are to be launched. The handbook will be compiled from data developed and evaluated during the SPREE program.

These investigations were prompted by a need to develop supporting technology for launch facilities applicable to potential improvements of the Saturn IB and Saturn V space vehicle systems.

CONTENTS

Figure

Table

I. PROGRAM GOALS

The Solid-Propellant Rocket Exhaust Effects (SPREE) program was established to evaluate and recommend materials and techniques for protection of launch facilities from potentially damaging environments created by the exhaust gases of solid rocket motors (SRMs). As a result of the program, a design handbook will be prepared for use in designing new, or modifying existing, facilities from which vehicles using SRMs will be launched. Empirical and analytical data and criteria that evolved as part of the program will form the basis for the design handbook. The handbook will provide criteria defining when and where protective materials should be used and what the materials should be. The handbook will further specify how the materials should be applied, how much they will deteriorate during a launch usage, and how they should be refurbished. These recommendations will be made so they apply to the types of solid-propellant rockets being considered for the Improved Saturn launch vehicle systems and will be based on environmental exposure criteria as well as economic considerations.

The overall program was planned to be implemented in three discrete steps; two of these are completed and the third is nearing completion. SPREE Phase I (NAS10-389) consisted of data gathering and evaluation. Data from industrial and governmental sources pertaining to SRM performance parameters, exhaust characteristics and environments, and commercially available protective (insulators and ablators) materials, were accumulated, reviewed, and analyzed. From these data, mathematical prediction models were developed to define both the environments produced on the launch pad and affiliated equipment by the rocket exhaust gases and the effects these gases would have on materials used to protect the facilities.

The purpose of SPREE Phase II, Part I (NAS10-1107) was to validate the Phase I mathematical prediction models and the associated theories. This was accomplished by performing scale-model tests using both cold and hot jet streams of single nozzle configurations. The jet streams were impinged on various deflector configurations that, in turn, were coated with the most promising protective materials selected from Phase I studies.

A Part II (NAS10-2300) was added to SPREE Phase II with two major purposes. First, additional hot jet tests were needed to obtain data required to provide a more complete and competent basis for the design handbook. Second, time and effort was allocated to allow preparation of the handbook. The tests scheduled for Part II provided data that allowed the selection and quantitative evaluation of protective materials, determined the effects of liftoff rates on protective materials and the environments imposed on the launch facility, and defined the environments and effects to launch facilities associated with multiple engine exhaust plumes. Simultaneously, the effects prediction models were refined, and the revised models were correlated with facility effects and environmental data obtained from a Titan IIIC launch using full-scale 120-in. SRMs.

The effects to the launch facilities resulting from the first launch of a Titan IIIC vehicle confirmed the necessity for extending facilities design technology to account for the severe exhaust environments created by the exhaust of solid-propellant rocket motors. The Titan III launch facilities were designed predicated on effects data extrapolated from liquid-propulsion exhaust environments and very limited scale-model tests using SRMs. Figure I-1 is a postlaunch photograph of the aerospace ground equipment (AGE) building, located about 40 ft from the Titan III launch pad, which illustrates the damaging effects that can accrue from exposure to the exhaust of SRMs. The SPREE program results indicate that economical and maintainable protection can be provided to launch facilities to minimize costly and time-consuming damage.

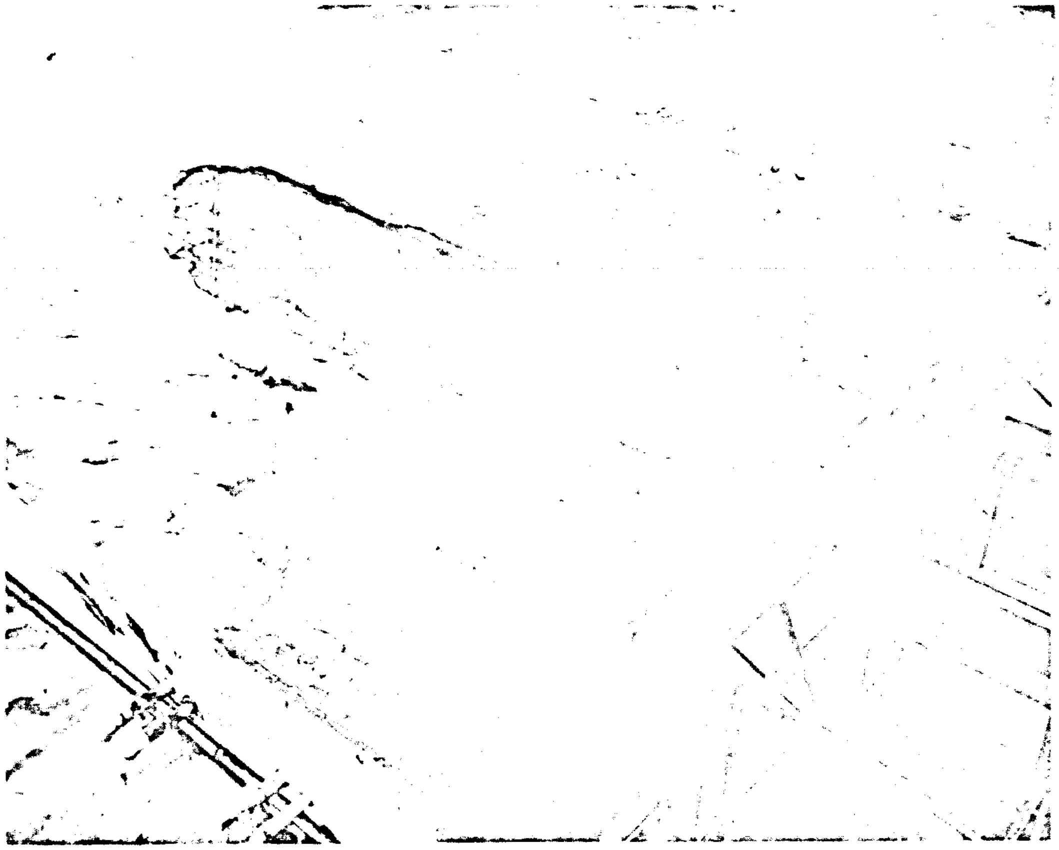


Fig. I-1 Postlaunch Condition of Titan IIIC AGE Building

II. CONCLUSIONS

The SPREE program was initiated to provide that specific technology necessary to properly configure launch facilities for vehicles using SRMs as the first stage, or as thrust augmentors to the first stage, equivalent to vehicle configurations being considered for Improved Saturn. The SPREE investigations revealed a number of technical findings that bear significantly on configurations and designs of launch facilities. These are summarized below.

A. PREDICTION MODELS

The mathematical models developed early in the SPREE program for predicting environments and effects on deflectors and other associated launch pad structures correlate, within reasonable limits, with scale-model tests of SRMs. These models were also correlated with the full-scale launch effects and environmental data obtained from captive firings and launches of 120-in. SRMs. Sufficient confidence is presently held in these modles to apply them practically in design considerations. Further correlation with full-scale launch data, however, is necessary to refine the models for general application. The publication of the first edition of a design handbook is certainly warranted at this time.

B. SCALE-MODEL TESTS

With care being exercised to strictly control configuration and performance parameters, scale-model testing to develop environmental data and duplicate effects created by exhaust gases of solid motors is very practical, economical, and productive. The results derived with scale-model tests should, however, be validated on at least a limited basis with full-scale results to measure true effectiveness of the prediction models constructed from the subscale model tests.

C. EFFECTS OF MOTOR CHARACTERISTICS

The basic characteristics of a solid-propellant rocket motor that influence the environments created by the exhaust plume are chamber temperature, chamber pressure, burning rate (mass flow), expansion ratio, and chemical composition of the propellant. Propellant composition, however, primarily influences temperature, pressure, and burning rates, which, in turn, are the major contributors to environmental and effects configurations. The chemical products in the exhaust plume would, of course, be greatly influenced by propellant composition, and, from purely chemical reactions, damaging effects to launch facilities could be realized. This result did not occur with PBAA propellants used for the SPREE tests.

The alumina particles in SRM exhaust plumes, although contributing to increased erosion, are not considered the primary cause for the increased erosive effects on facilities. The analysis, which indicates that only 10% of the total erosion effect may be attributable to existence of particles in the exhaust stress, has been substantiated although not completely explained. Closer examination of the phenomenon indicates that the addition of aluminum powder to solid propellants increases the chamber temperature, and it is the additional heat generated that creates the greatest erosive effects.

The effect of varying expansion ratios off optimum was not evaluated by scale-model tests. However, effects resulting from either underexpansion or overexpansion can be conservatively extrapolated from ideal expansion data.

D. PRIMARY EFFECTS PARAMETERS

Heat resulting from high temperatures of the exhaust plume is the main environment that can damage launch facilities. However, impact pressures cannot be disregarded, because they are also relatively high. But design can be more easily performed for high pressure conditions than for high heat conditions. It is due to the high heat fluxes at high temperatures that as liftoff accelerations are decreased from the nominally used 1.3 g, the magnitude of erosive effects fects inflicted on launch facilities is greatly increased.

E. PROTECTIVE MATERIALS

A number of commercially available compounds can be applied to steel and concrete launch pad structures that will provide good protection against SRM exhaust environments. Fondu Fyre WA-1 exhibited the best mechanical, physical, and chemical properties. This material can be applied where direct impingement of the exhaust plume is expected. It also provides excellent protection against heat to structures exposed to the high temperatures created by the proximity of the exhaust plume. The structures requiring this kind of protection are those whose structural integrity would be affected by exposure to excessively high temperatures. Erosion of the Fondu Fyre was observed to decrease when it was moist. A need for protective paints was not determined, and none is recommended.

F. DEFLECTORS

It was demonstrated that a radius of deflector curvature equivalent to 1.7 nozzle exit diameters gives satisfactory exhaust plume deflection with predictable and nonexcessive deflector erosion effects. This curvature criterion, as well as the distance between the exit plane of the motor nozzle and deflector, is governed more by vehicle constraints than by facility considerations if the distance is greater than the length of the first supersonic shock diamond. Deflector curvature and impingement distance determine vehicle base pressures and temperatures. These parameters, plus location of the nozzle exit plane above the launch deck exhaust opening, thrust buildup of the motor, and exhaust duct configuration contribute to the ignition overpressure pulse, which must be considered in design of the launch pad structure as well as the vehicle engine compartment.

The concept investigated for eliminating or minimizing mechanical deflectors by diversion of exhaust streams with secondary jets of gases did not prove effective even when the auxiliary jet flow rates approached 20% of the flow rates of the primary exhaust stream. The concept might prove effective if the diverting jet mass flow rate approached 50% of the primary exhaust stream, but, at this point, economics would indicate that mechanical deflectors are more practical.

The potential application of the Coanda effect for deflector usage showed promise. By use of a suitable configuration involving a Coanda-type deflector, the pressure of a jet stream impinging on a flat plate can be substantially reduced, and the exhaust plume can be effectively turned. More detailed investigations of this concept are warranted.

G. UMBILICAL TOWERS AND OTHER SUPPORT EQUIPMENT

In general, when umbilical towers and other launch pad equipment are farther than 1½ nozzle exit diameters from the centerline of the SRMs and nozzles are canted toward the structures no heat protection is necessary except in very localized areas. This can be readily accomplished with available materials, such as Martyte or Dynatherm E-300. Pressures, on the other hand, require greater attention. With the data available from SPREE to predict the pressure environments, this effect can be easily handled with proper structural design.

III. PROGRAM DESCRIPTION

A. ORGANIZATION OF PROGRAM

The SPREE program was technically performed in five consecutive steps. The first of these was an extensive literature search for SRM characteristics and performance data, rocket engine exhaust characteristics data, deflector design and performance data, and data pertaining to available protective insulator and ablator materials. These data were carefully reviewed, evaluated, and analyzed to develop theories that would allow prediction of environments associated with and effects resulting from the exhaust plumes of solid propellant rocket motors.

The second step consisted of a series of cold jet tests aimed at validating some of the theories pertaining to pressure and flow relationships existing for various impingement angles of a high velocity gas stream on a deflector surface. Also tested were several unique concepts of deflector configurations with the purpose of evolving simpler and more effective gas deflection techniques.

The third step consisted of 30 hot-jet firing tests using a small-scale SRM. The characteristics of these tests and their associated objectives are shown in Table III-1. The overall objective of the tests was to validate data obtained from the cold-jet tests and to resolve some of the assumptions made in the environment and effects prediction models.

The fourth step also consisted of hot-jet tests (27 firings), but the purpose of these was to correlate the effects of launch facility configurations to the results predicted by mathematical models. The detailed objectives achieved and the characteristics of the test firings are specified in Table III-2.

The fifth step will consist of preparing a handbook that can be used for development of criteria and specifications necessary for designing a launch pad to withstand the exhaust environments of solid propellant rocket motors.

B. TEST CONFIGURATIONS

Scale-model tests, to a large degree, were the source of particularly significant and important data during the SPREE program. Since quantitative as well as qualitative data were sought, special precautions had to be exercised in test configurations to obtain the data desired in a precise manner. Figure III-1 illustrates the test fixture and associated instrumentation used for the cold-jet test program. Pressure and jet-gas temperature data were recorded on oscillographs, whereas flows were recorded by Schlieren photographs. The figure shows the mechanisms used to vary the impingement angle and the distance between the exit plane of the jet nozzle and deflector surface.

Table III-1 SPREE Phase II, Part I Hot-Firing Data

Firing No.	Run Date	Test Objectives	Grain Data			Nozzle Exit Dia (in.)	Deflector Data			Remarks
			Al (%)	$\dot{w}$ (lb/sec)	P_c (psi)		Type Coating	(deg)	Motion	
1	6/5/64	Deflector Heating and Pressure Data	14	10	700	4.89	1/2-in. Mild Steel Flat Plate	15	Static	
2	6/10/64		14	10	700	4.89		30		
3	6/12/64		14	10	700	4.89		45		
4	6/17/64		14	10	700	4.89		60		
5	6/23/64		14	10	700	4.89		90		
6	7/1/64	Plume Measurements and Materials Evaluation	14	10	700	4.89		30	Static	NRL 1010 specimen burned out. Failed to retract from plume. Specimen in plume for 10+ sec
7	7/6/64		18	10	700	4.89		30		RL 1010 specimen in exhaust 1.86 sec, erosion rate = 0.141 in./sec P_c = 689 psia
8	7/8/64		18	10	700	4.89		30		NARMCO Conolon 525-T2 specimen in exhaust 2.96 sec, erosion rate 0.081 in./sec, P_c = 711 psia
9	7/16/64		14	10	700	4.89		30		Asbestos fiber w/91LD phenolic binder, specimen in exhaust 4.02 sec, erosion rate 0.093 in./sec, P_c = 614 psia
10	8/15/64	Flat Slab Erosion Measurement	18	10	700	4.89	Flat Plate, Silica Coated	15	0.3 g	
11	8/17/64		14	10	700	4.89	Plate 6, Silica Coated	30	↓	
12	8/18/64		14	10	700	4.89	Plate 3, Silica Coated	45	↓	
13	8/19/64		14	10	700	4.89	Plate 1, Silica Coated	60	↓	
14	8/21/64		14	10	700	4.89	Flat Plate, Silica Coated	30	0.3 g	Deflectors used during runs 11 and 12 were reused in this firing. Impingement centerline was at the center of the plate used in Firing 11.
15	9/8/64		18	10	700	4.89	↓	30	25 dia in 10 sec	
16	9/9/64		18	10	700	4.89	↓	45	↓	
17	9/10/64		14	10	700	4.89	Flat Plate, Silica Coated	30	25 dia in 10 sec	
18	9/22/64	J Deflector Erosion Measurement	14	10	700	4.89	J, Silica Coated, Medium Size	30	25 dia in 10 sec	
19	9/23/64		18	10	700	4.89	↓	30	↓	No P_c readout
20	9/28/64		22	10	700	4.89	J, Silica Coated, Medium Size	30	25 dia in 10 sec	Deflector damaged, burned through, no erosion data
21	10/13/64		22	5	700	3.46	J, Silica Coated, Small Size	30	50 dia in 10 sec	
22	10/23/64		14	5	700	3.46	↓	90	↓	
23	10/27/64		18	5	700	3.46	J, Silica Coated, Small Size	90	↓	
24	11/2/64		22	10	700	[illegible]	J, Fondu Fyre Coated, Medium Size	90	↓	
25	11/6/64		22	10	[illegible]	[illegible]	J, [illegible] (1/2 and Silica Tile, Medium Size	90	↓	Motor blew out at P_c 1380 psia at 2 sec after PS_1.
26	11/29/64		14	15	[illegible]	[illegible]	J, Silica Coated, Large Size	90	↓	
27	11/20/64		18	15	[illegible]	[illegible]	↓	90	↓	
28	11/23/64		22	15	[illegible]	[illegible]	J, Silica Coated, Large Size	90	↓	Deflector material completely eroded and spalled off.
29	11/24/64		14	10	[illegible]	[illegible]	J, Silica Coated, Medium Size	90	↓	
30	11/24/64		18	10	[illegible]	[illegible]	J, Silica Coated, Medium Size	90	50 dia in 10 sec	

9-1

FIRING NO.		GROUP	OBJECTIVE		
Planned	Actual			No.	P_c (nom) (lb/in^2)
1	1	I	Performance of deflector Coating materials	1	650
2	2				
3	3				
4	4				
5	5				
6	6				
7	7	II	Effect of propellant aluminum content on launch complex heating and pressure, evaluation of protective coatings, and collection of data for comparison with full scale Titan III erosion data (scale effects investigation)	1	600
8	8				
9	9				
10	10	II,III			
11	11	II			
12	12				
13	13				
14	14				
15	16	III	Effect of lift-off rate on erosion and launch complex Heating	1	600
16	18				
17	19				
18	20				
19	15	IV	Special evaluations	1	600
20	17				
21	21				
22	22				
23	26	V	Launch complex heating and Pressure with clustered engines	2	600
24	27				
25	23	VI	Single and clustered engine Plume characteristics	1	600
26	24			1	
27	25			2	

(1) Initial nozzle-to-deflector separation distance = 3 D_e. Hold-down

(2) Four parts fine quartzite aggregate to one part Type III Portland C

(3) Atlantic Research Corporation grain.

(4) Initial nozzle-to-deflector separation distance = 2.5 D_e. Hold-dow

(5) Initial nozzle-to-rake separation distance = 8 D_e. Hold-down time

(6) Initial nozzle-to-rake separation distance = 12 D_e. Hold-down time

MOTOR DATA				NOZZLE-TO-DEFLECTOR SEPARATION DISTANCE	DE
$\dot{W}$ (nom) (lb/sec)	D_e (in)	%Al	Burn Time (nom) (sec)	(nozzle exit diameters)	Type
4 ↓	2.84 ↓	15 ↓	4 ↓	3 ↓	30° flat plate ↓
8.2 ↓	4.77 ↓	18 22 14 18 22 7 22(3) 14	13 ↓	130 D_e in 10 sec(1) ↓ 50 D_e in 10 sec(1) ↓ 130 D_e in 10 sec(1)	J(similar to TIII) ↓ J J(similar to TIII)
8.2 ↓	4.77 ↓	18 ↓	13 24 ↓	100 D_e in 10 sec(1) 35 D_e in 10 sec(1) 25 D_e in 10 sec(1) 15 D_e in 10 sec(1)	J ↓
8.2 ↓	4.77 ↓	18 ↓	13 ↓	50 D_e in 10 sec(1) ↓	J ↓
8.2 ↓	4.77 ↓	18 ↓	13 ↓	50 D_e in 10 sec(1) 50 D_e in 10 sec(4)	Wedge (similar to J's back to back)
8.2 ↓	4.77 ↓	7 ↓	13 ↓	50 D_e in 10 sec(5) ↓ 50 D_e in 10 sec(6)	N/A ↓

ime = 0.2 sec. These values must be subtracted from values shown for evaluation of
ment apportioned by weight.

time = 0.2 sec. Cart moved 47 D_e in 9.8 sec.
0.2 sec. Cart moved 47 D_e in 9.8 sec.
= 0.2 sec. Cart moved 47 D_e in 9.8 sec.

Table III-2 SPREE Phase II, Part II Run Schedule

FLECTOR DATA	DISTANCE COATED PLATE FACE TO NOZZLE ℄	REMARKS
Coating		
Fondu-Fyre WA-1	N/A	Base run
Fondu-Fyre XB-1	↓	Evaluation of higher cost Fondu-Fyre mix
Fondu-Fyre WA-1 with crack		Evaluation of self healing quality
Ice (H_2O)		Materials evaluation
Portland cement mix$^{(2)}$		Evaluation of Titan III deflector material
Refurbished Fondu-Fyre WA-1		Refurbishability evaluation
Portland cement mix$^{(2)}$	1.5 D_e	(Cart travel malfunction)
↓	↓	
Fondu-Fyre WA-1		
↓		Complements runs 15-18
H-W Harcast ES	1.25 D_e	(Rocket motor malfunction)
Portland cement mix$^{(2)}$	1.25 D_e	Rocket motor canted 6°
Fondu-Fyre WA-1	1.5 D_e	Run 10 complements this group of runs
↓	↓	
Fondu-Fyre WA-1	1.25 D_e	Rocket motor canted 6°
↓	1.5 D_e	Repeat firing on deflector used for run 10
H-W Special Mix 13-65	↓	Evaluation of newly developed deflector coating (a fused silica castable)
Fondu-Fyre WA-1		Deflector used for firing 17 (actual) refurbished
Fondu-Fyre WA-1	N/A	(Cart jumped track). Ridge radius = 0.25 D_e
↓	↓	Ridge radius = 0.50 D_e
N/A	N/A	Single motor plume probe
↓	↓	Duplicate run
		Two motor plume probe

lift-off acceleration (e.g. 50 D_e in 10 sec. is really 47 D_e in 9.8 sec.)

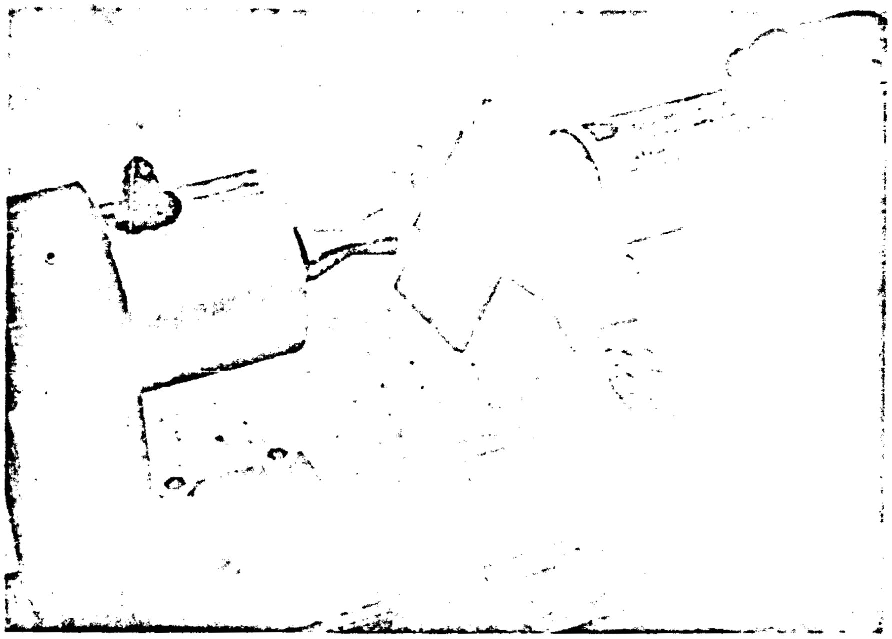


Fig. III-1 Cold-Jet Test Configuration

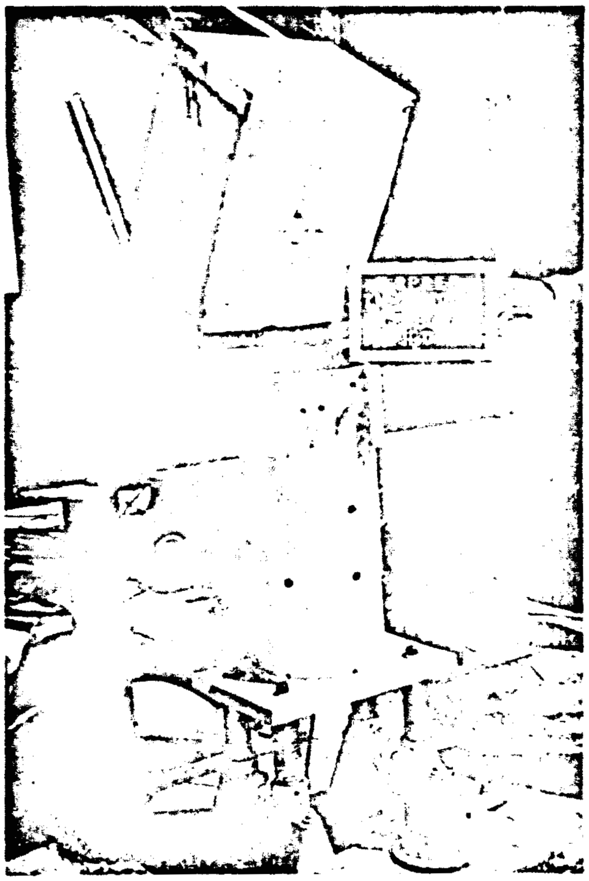


Fig. III-2 Hot-Jet Materials Evaluation Test Fixture

To evaluate the performance of materials when exposed to a hot-jet gas stream, the test configuration shown in Fig. III-2 was used. For these tests, small scale-model SRMs of 4-second burning time were fired vertically up onto deflectors coated with candidate protective materials. Data are obtained primarily by performing weight and dimensional measurements of the deflector. The SRM was monitored for chamber pressure and burning time only, since other performance data peculiar to motors used were known. Shown in the figure is a block of ice used as a deflector during one test firing. The ice failed mechanically after approximately 2 seconds of exposure to the hot-gas jet stream.

The majority of the test firings were performed under dynamic conditions where launch vehicle liftoff accelerations were simulated. Shown in Fig. III-3 is the general equipment arrangement used for these tests. The test run shown consisted of a simultaneous firing of two SRMs to determine environments in complex exhaust plumes. It may be noted that, at the point of intersection of the supersonic portions of the individual motors, turbulent, subsonic flow is experienced. The cart, shown in the foreground, is equipped with a pressure/temperature rake and a trailing instrumentation cable. The 500-lb weight in the tall structure in back of the motors provided the necessary acceleration forces for movement of the cart. Figure III-4 shows the mechanism used to provide control for the desired acceleration. The arm was rotated by means of a constant speed 40-hp electric motor that, in turn, controlled the movement of the cart attached to the arm by 3/8-in. wire rope. By changing ratios in the gear train and varying the equivalent length of arm, various acceleration rates were obtained. The cart configuration used to measure single-motor plume pressures and temperatures is shown in Fig. III-5. Although the cart was essential to simulate liftoff accelerations when deflector effects data were sought, it was used for the plume data runs to obtain continuous measurements of the solid-motor exhaust characteristics for a length of plume equivalent to 50 nozzle exit diameters.

The SRMs used were of a special design to allow maximum reusability. It was desired that mass flow of the motor remain constant from firing to firing and also over the duration of each firing. The motors shown in Fig. III-6 provided a nominal thrust of 2500 lb each at a chamber pressure of approximately 600 psia and a mass flow of about 8.2 lb/sec. Replaceable solid propellant grains of PBAA composition, furnished by Atlantic Research Corporation and Rocket Power Incorporated, were used. The nozzle of the motor was made of graphatite G carbon and was replaced for each run to minimize potential throat erosion as a variable from motor performance. Ideal expansion was realized from the nozzles shown. The motors performed extremely well throughout the hot-firing test program giving reasonable repeatability of motor performance.

Examples of the deflector configurations used during the effects determination portion of the test program are shown in Fig. III-7 and III-8. A launch deck, with a simulated umbilical tower, to which samples of protective materials were attached, was employed with the J type of deflector. The deflector and the umbilical tower test panels were replaced for each firing. Some environmental measurements in terms of temperatures, pressures and total heat fluxes were made under simulated liftoff conditions. Deflectors and test panels were [illegible] for weight and dimensions before and after exposure to the motor exhaust. It is particularly significant that the simulated launch deck was coated with Fondu-Fyre WA-1 and do not require refurbishment even after repeated firings. Refurbishment procedures were, however, developed and evaluated for the deflectors since they did erode with each firing. Tests of the wedge deflector indicated that minor effects occur to the ridge line when the motors are located on either side of the ridge.

In addition to the directly instrumented data obtained during the tests, motion pictures were taken. However, the film data is of only minor value because gases of the plume prevented good viewing of the behavior of the test article. A summary film has been compiled to provide a visual historical record of the test program.

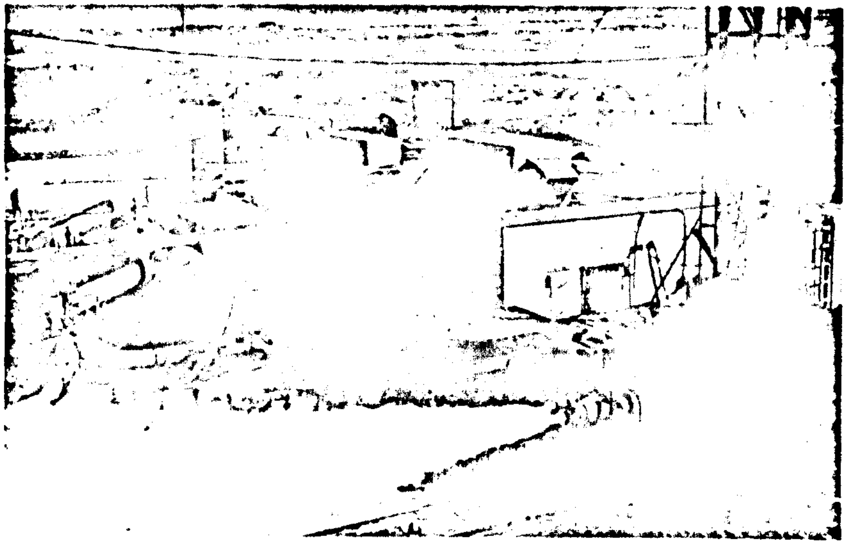


Fig. III-3 Acceleration Simulation Fixture

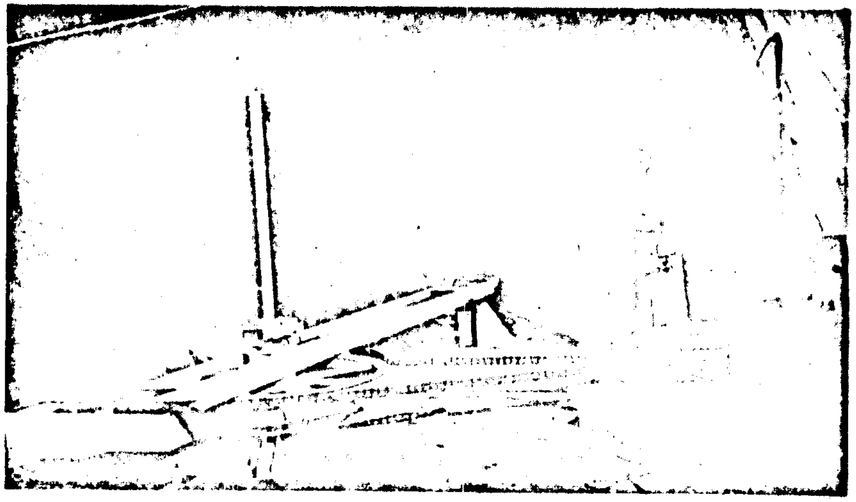


Fig. III-4 Acceleration Control Mechanism

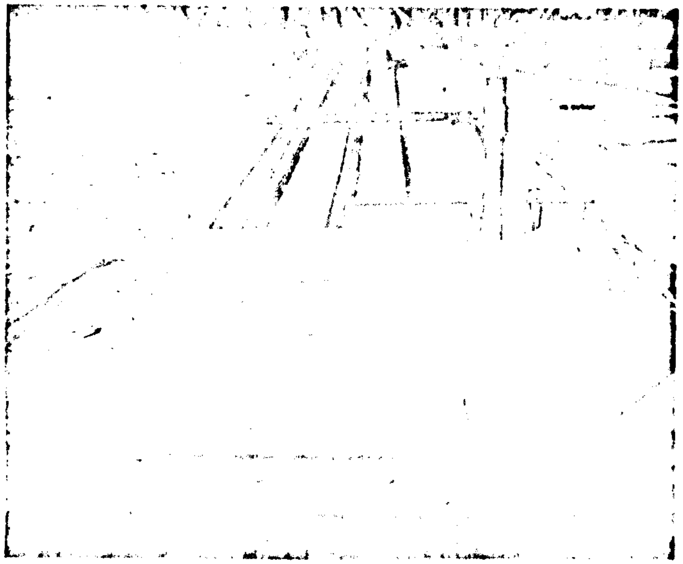


Fig. III-5 Test Cart

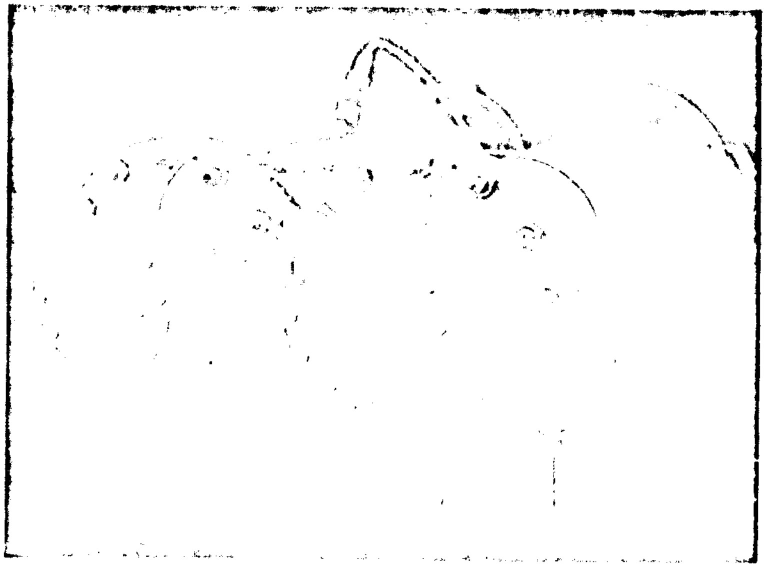


Fig. III-6 Scale Model Solid Propellant Rocket Motors

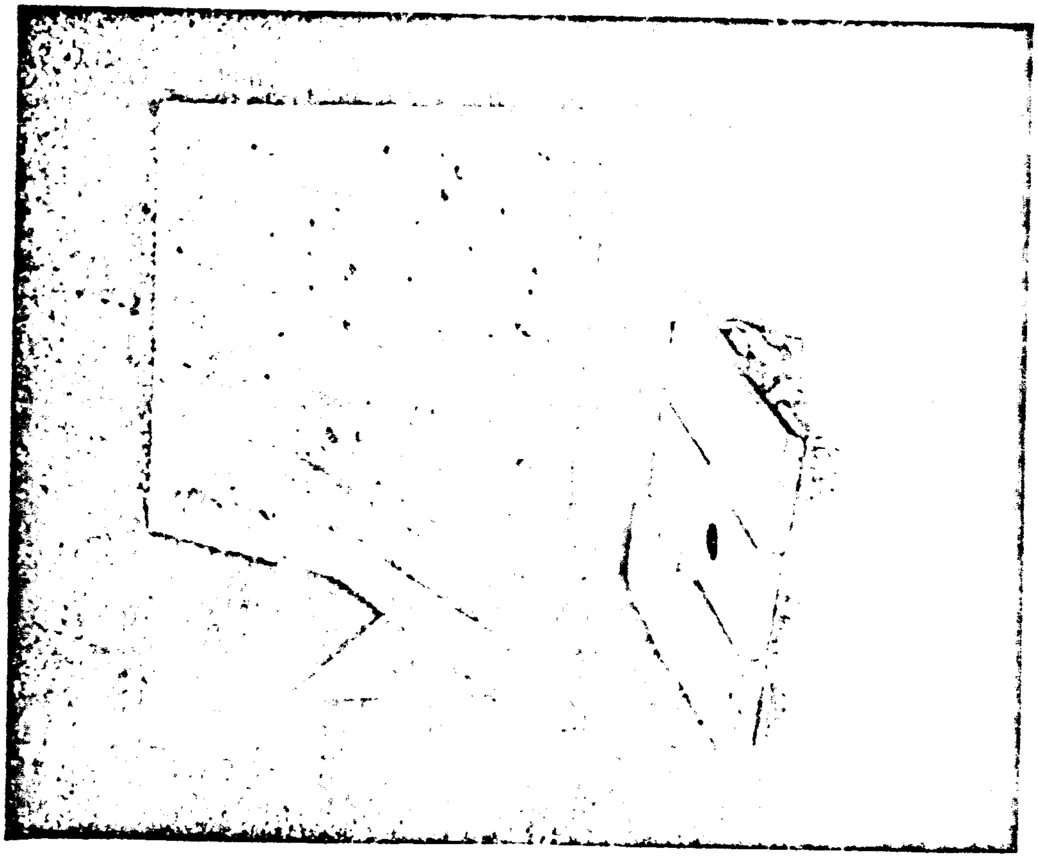


Fig. III-7 Simulated Launch Facility Effects Model

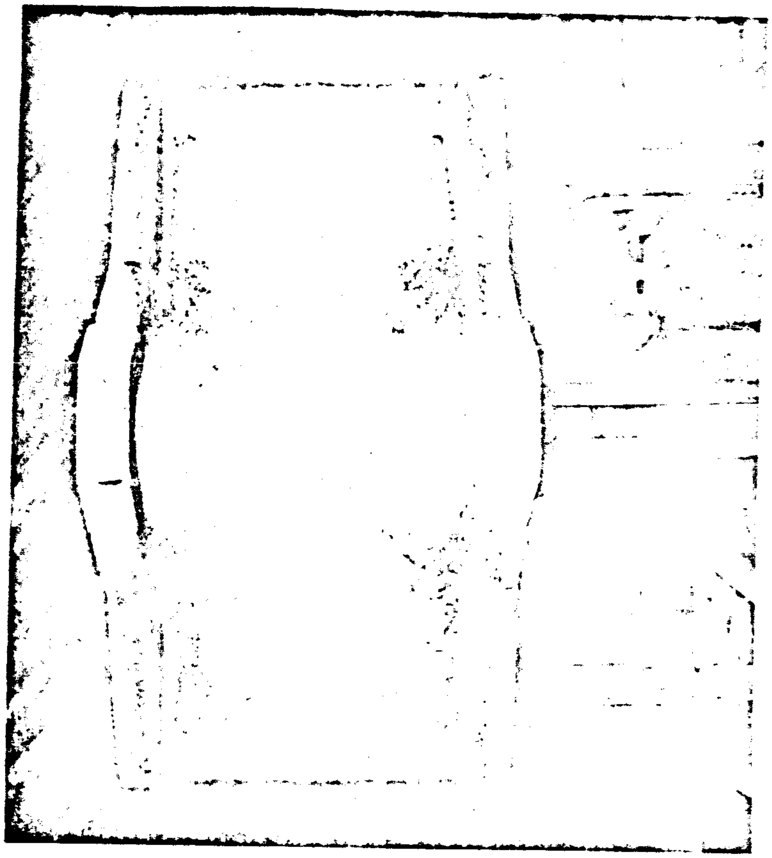


Fig. III-8 Wedge Deflector Used with Multiple Motor Firings

IV. SIGNIFICANT RESULTS

Considerable data were accumulated during the SPREE program. These data and the analyses thereof are documented in the Solid Propellant Rocket Exhaust Effects (SPREE) and Methods of Attenuation reports, NASA-CR-64-84, NASA-CR-64-87, and NASA CR-65-93. A handbook applying these data to practical design considerations of launch facilities will subsequently be published. Illustrated herein are some of the typical and significant data realized from the program.

A. MATERIALS

More than 100 commercially available silicone and ceramic based materials were evaluated for potential application as protective coatings for launch facilities. Fondu Fyre WA-1, which is a formulation of refractory aggregates and a hydraulic setting base, was the material best suited for the applications investigated. This material is about 1.5 stronger in compression than Portland cement concretes and exhibits other excellent mechanical and chemical properties. When exposed to the heat of the SRM exhaust, it will heal itself in areas where cracks have occurred. It is relatively easy to install and is reasonably economical. Also, its refurbishment is simple and relatively economical. Although excellent, its erosion qualities are not as good as other materials. Economy, simple application methods, easy maintenance, and good mechanical properties qualify Fondu-Fyre WA-1 for application to deflector surfaces as well as other elements of the launch pad that require protective coatings.

B. EXHAUST PLUME CHARACTERISTICS

To measure exhaust plume pressures and temperatures, a graphite-coated rake was installed on the test cart. The cart was pulled away from the exit plane of the motor at a constant acceleration rate to obtain measurements of the plume for a distance of 50 nozzle exit diameters. To prevent clogging of the pressure ports, only 7% aluminum content grains were used. The rake shown in Fig. IV-1 was actually used to sample the composite plume of two motors being fired simultaneously. The temperature and pressure profiles for the dual plume are shown in Fig. IV-2 and IV-3 in terms of nondimensionalized parameters to allow application for variably sized SRMs. Single-motor data did not differ from the profiles obtained for each motor of the dual-motor firings. The temperature and pressure profiles existing between the two exhaust streams were of primary interest for the dual motor firing test.

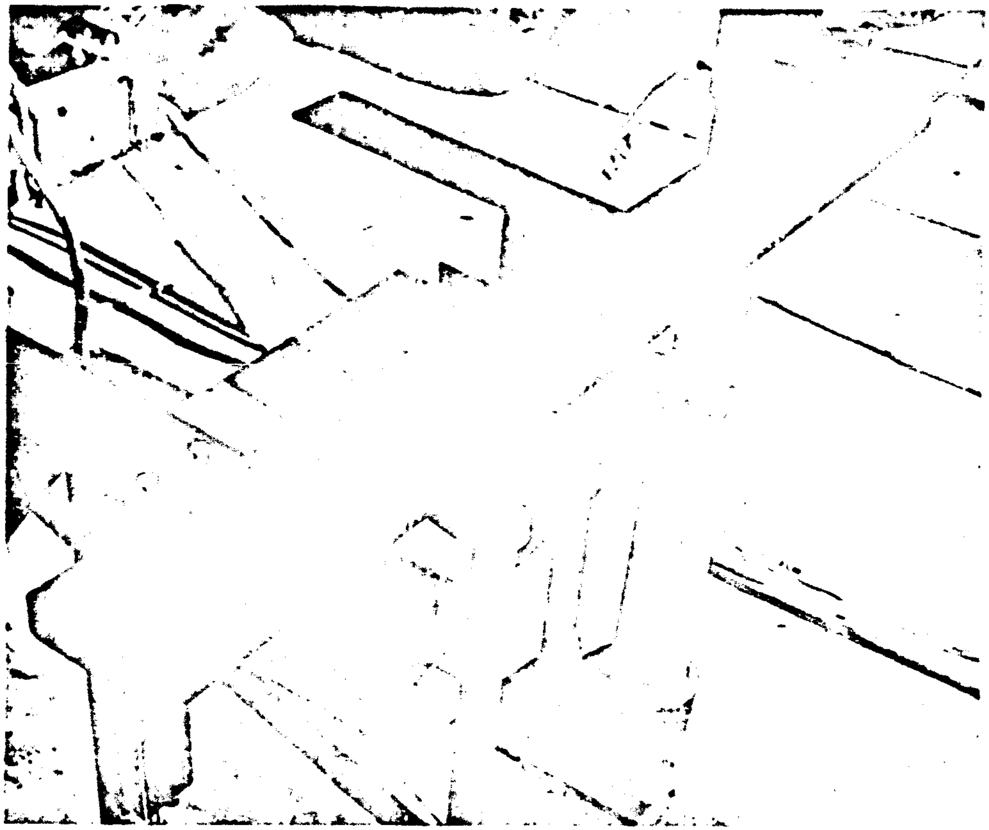
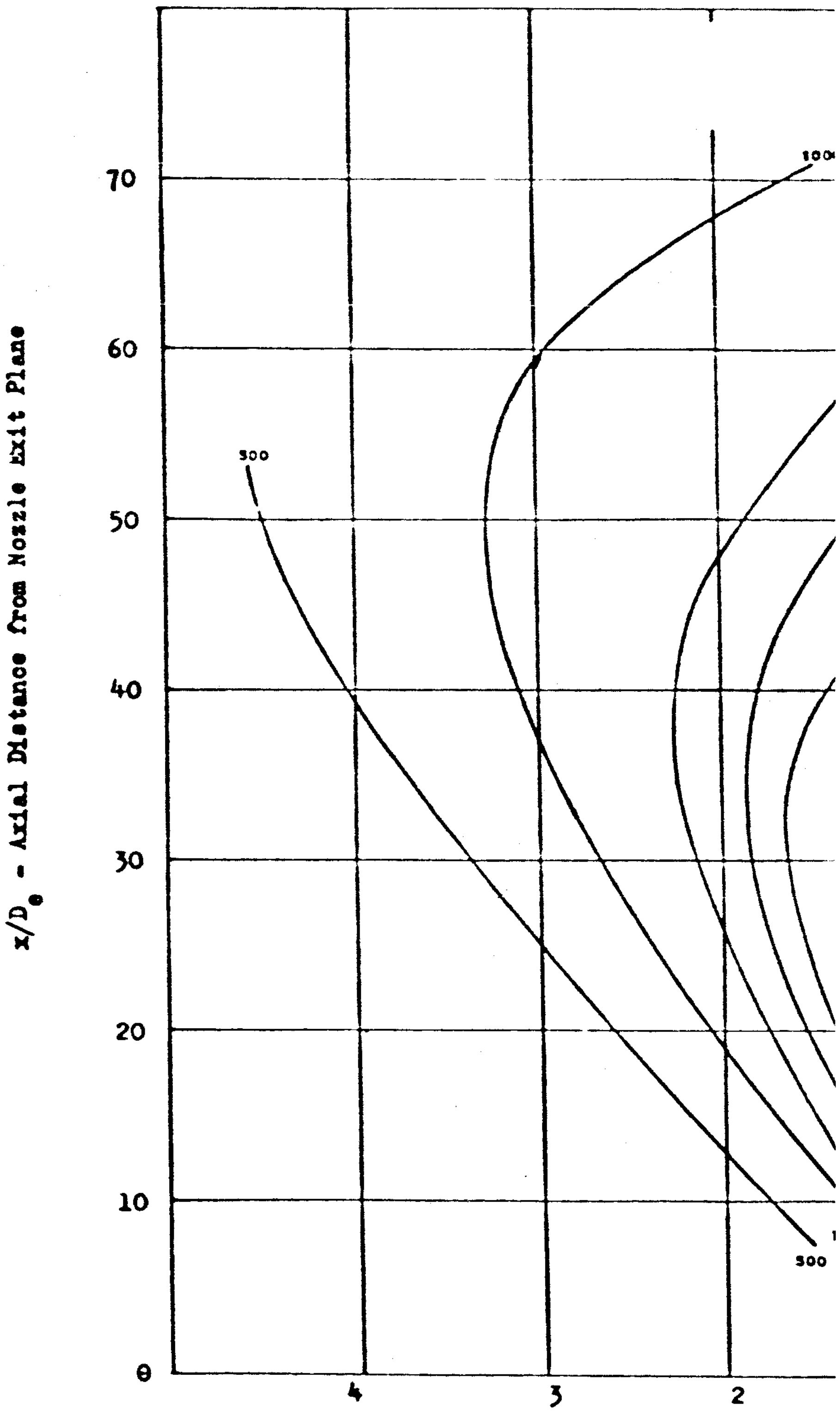


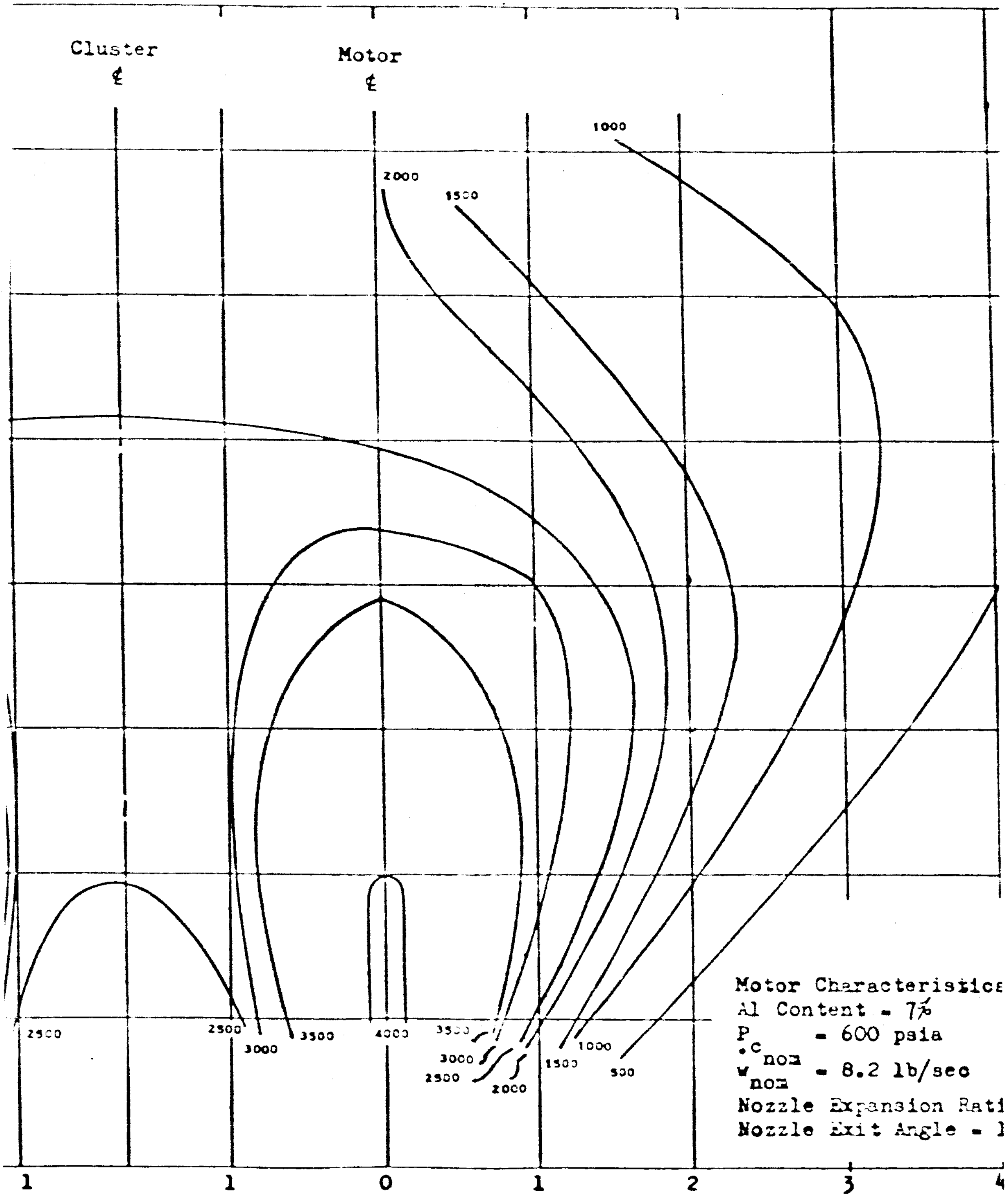
Fig. IV-1 Dual-Motor Pressure Temperature Rake

C. UMBILICAL TOWER PRESSURES AND TEMPERATURES

Temperature, total heat flux, and pressure measurements were made at various locations along the height of a simulated umbilical mast. Figure IV-4 shows the calorimeter location and method of installation onto the test panels used for evaluating exhaust effects on samples of protective materials. Also shown are two pressure ports drilled in the test panels. Thermocouples were attached to the back side of the test panels to measure temperatures. These panels were installed at 1½ nozzle diameters away from the centerline of the rocket motor. Liftoff was simulated so the test panels remained parallel to the centerline of the exhaust plume. The pressures measured under these conditions for several pertinent data runs are shown in Fig. IV-5 where the pressure ports were located nearest to the top of the umbilical tower. These data correlate well with full-scale data obtained from the first launch of the Titan IIIC space vehicle.

The maximum total heat flux measured was 85 Btu/ft^2-sec, and the highest panel temperature achieved for a steel plate located at the bottom umbilical position was 1200°F. Evaluation of the steel plate did not indicate major damage despite the relatively high temperature experienced.


x/D_e - Axial Distance from Nozzle Exit Plane
70
60
50
40
30
20
10
0
4
3
2
500
500
100
1



Motor Characteristics
Al Content - 7%
P_c - 600 psia
$\dot{w}_{nom}$ - 8.2 lb/sec
Nozzle Expansion Rati
Nozzle Exit Angle - 1

stance from Nozzle Centerline

Fig. IV-2 Dual-Motor Exhaust Plume Temperature I

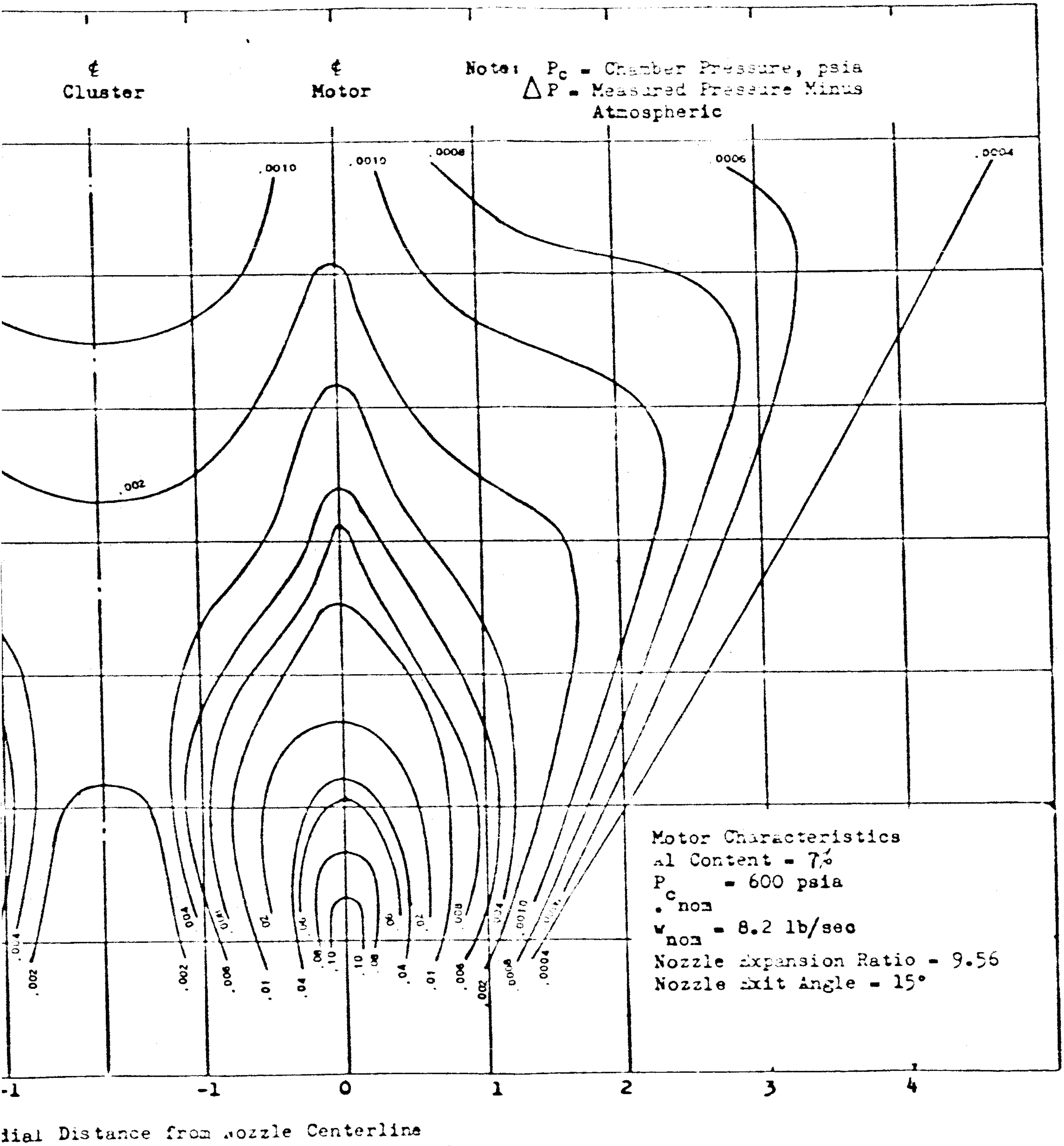


Fig. IV-3 Dual-Motor Exhaust Plume Pressure Profile

16-2

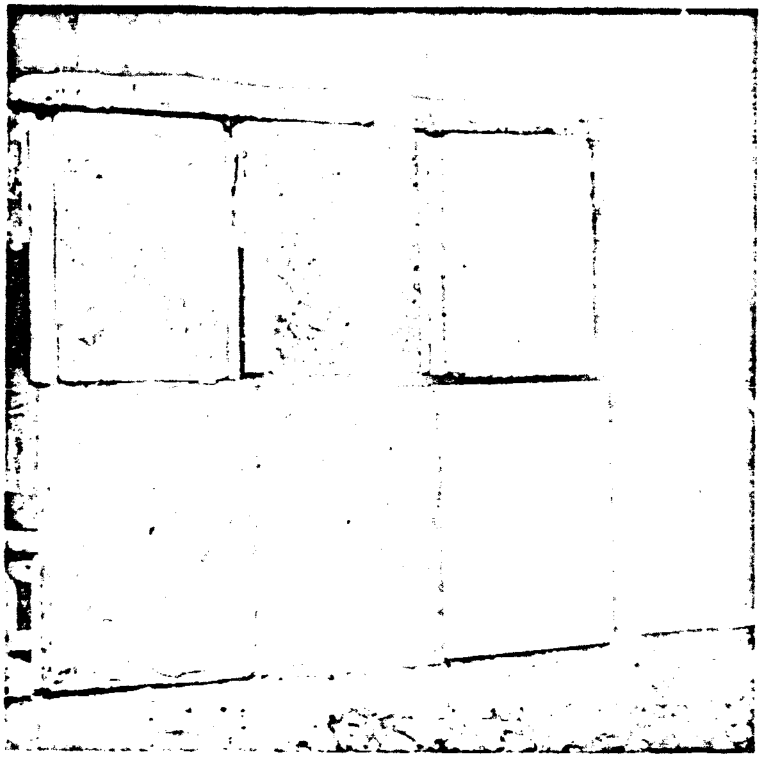


Fig. IV-4 Simulated Umbilical Tower Test Panels

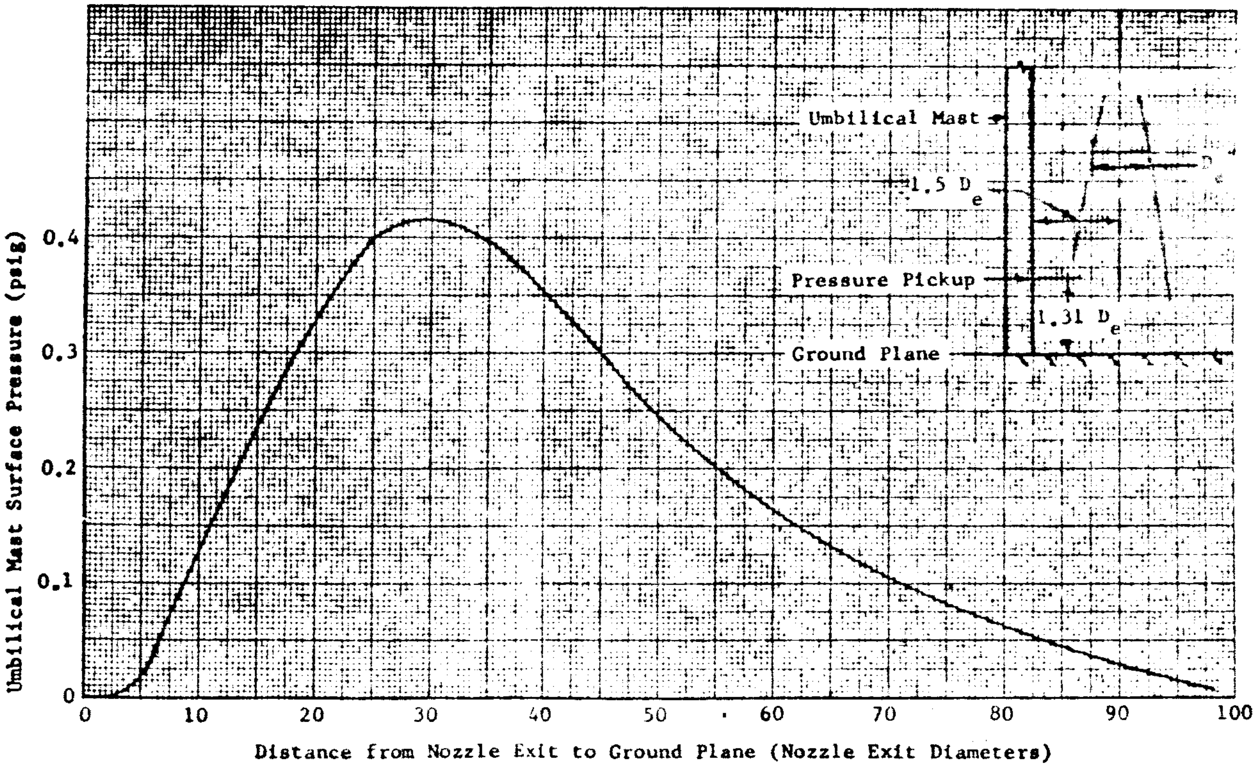


Fig. IV-5 Variation of Umbilical Mast Surface Pressure during Liftoff

D. PROPELLANT ALUMINUM CONTENT EFFECTS

The percentage of aluminum content in solid propellant rocket grains does have a predictable effect on the amount of erosion experienced by deflectors as is shown in Fig. IV-6. Significantly, the erosion effects increase over the spectrum from 7% to 18% aluminum and decrease at higher percentages of aluminum. These data partially correlate with results expected as a function of analytically predicted chamber temperatures for various contents of aluminum. However, they do not linearly correlate with the plume heat flux measurements made. This signifies that the popular analytical methods for predicting chamber temperature may be inaccurate and that the presence of alumina particles, and their behavior, in the exhaust stream have some contribution to the erosion realized. Little conclusive data are available on the composition or behavior of the alumina particles in the exhaust stream, and obtaining such data promises to be a difficult task.

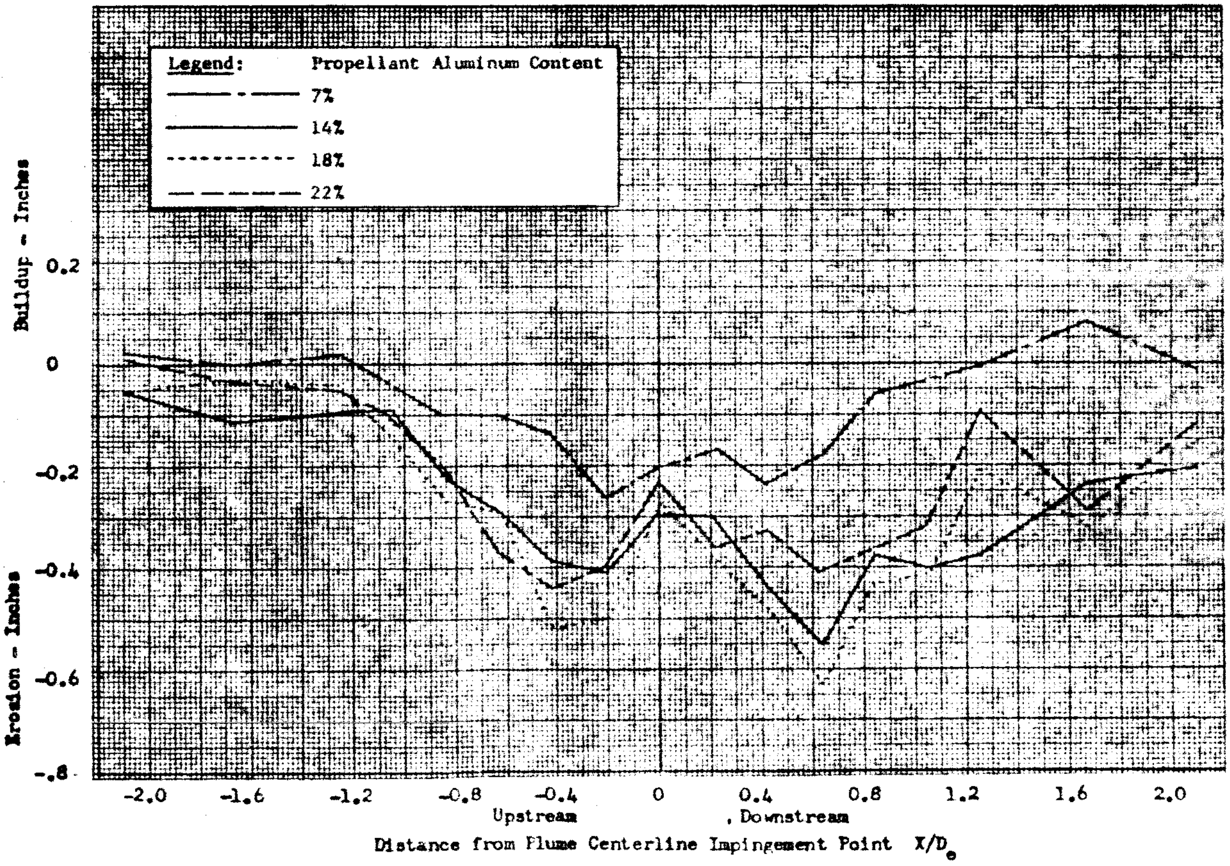
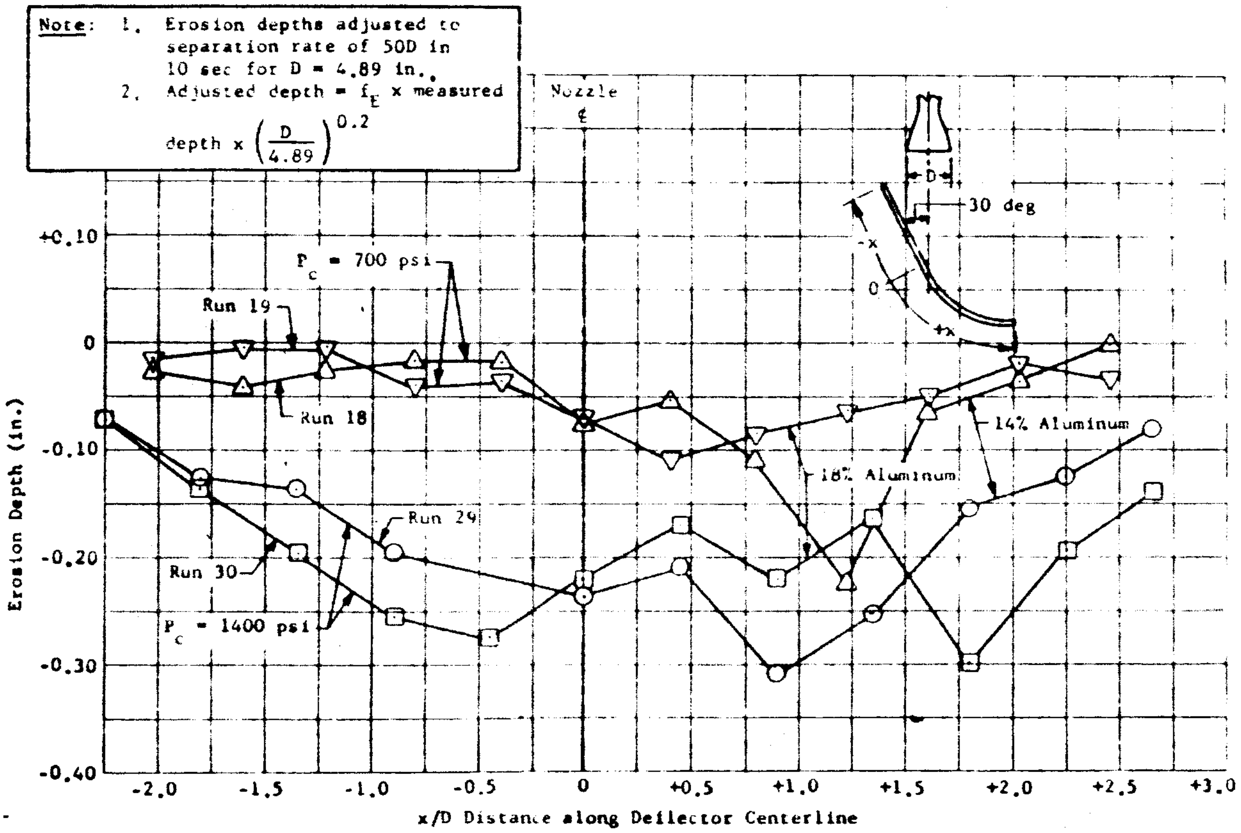


Fig. IV-6 Effect of Propellant Aluminum Content on Deflector Centerline Erosion

E. MOTOR CHAMBER PRESSURE EFFECTS

It would be expected that SRM chamber pressures are directly related to erosion effects. The magnitude of interrelationship is shown in Fig. IV-7 and can be seen as nearly linear. Doubling the chamber pressure doubles the effect. The primary contribution to this erosion appeared to be due to kinetic mechanical energy increase rather than the thermal energy increase in the exhaust plume.



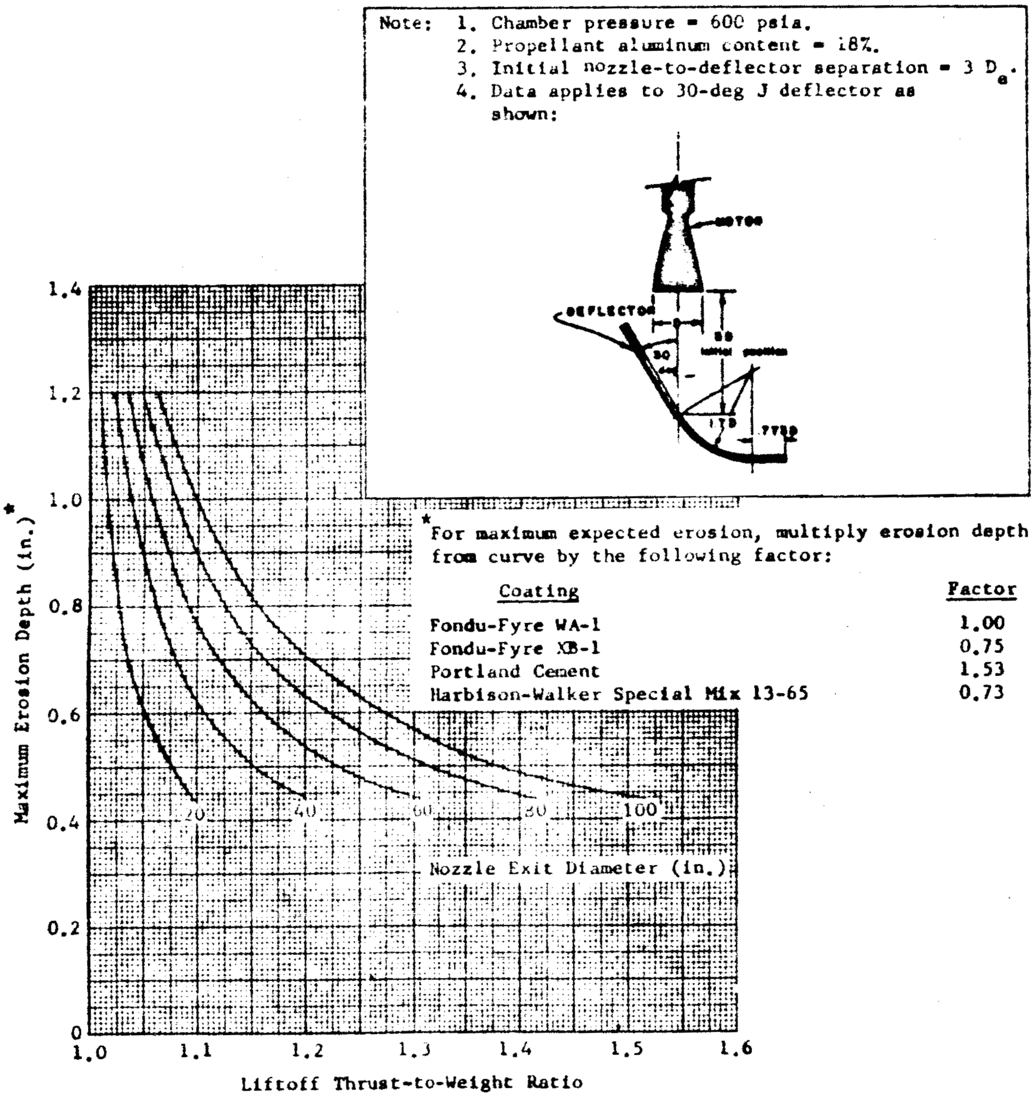
Note: 1. Erosion depths adjusted to separation rate of 50D in 10 sec for D = 4.89 in.

2. Adjusted depth = f_E x measured depth x $\left(\frac{D}{4.89}\right)^{0.2}$

Fig. IV-7 J Deflector Centerline Erosion Profiles Showing Effect of Varying Chamber Pressures

F. LIFTOFF ACCELERATION EFFECTS

Variations in thrust-to-weight ratios of launch vehicles directly affect the liftoff accelerations which, in turn, have very pronounced effects on launch facilities. **Figure IV-8** illustrates these effects in terms of resultant erosion depths on deflectors as a function of accelerations. Correlation of the subscale model results with Titan IIIC full-scale erosion data was very good with the subscale results slightly more severe (0.1 in. more erosion). The increasing erosion resulting from slower liftoff accelerations quite clearly correlates to the increase of heating.



Note:
1. Chamber pressure = 600 psia.
2. Propellant aluminum content = 18%.
3. Initial nozzle-to-deflector separation = 3 D_e.
4. Data applies to 30-deg J deflector as shown:

*For maximum expected erosion, multiply erosion depth from curve by the following factor:

Coating	Factor
Fondu-Fyre WA-1	1.00
Fondu-Fyre XB-1	0.75
Portland Cement	1.53
Harbison-Walker Special Mix 13-65	0.73

Fig. IV-8 Effect of Lift Off Thrust-to-Weight Ratio on Erosion of Several Deflector Coating Materials

V. RECOMMENDATIONS FOR FUTURE INVESTIGATIONS

A review of data obtained during the SPREE Program has resulted in identifying certain key investigations that should subsequently be performed. Although the SPREE Program produced much engineering data essential for design of deflectors, launch decks, umbilical towers, and other support equipment that may be exposed to the launch environments of SRMs, it was not planned to develop optimizing design criteria. The SPREE data was accrued from scale model tests that generally present more severe results than those encountered under actual full-scale conditions. Only limited reconciliation with full-scale rocket motor firing data was possible due to the limited amount of usable data available from such test programs. Major future emphasis should, therefore, be placed on providing proper and adequate instrumentation of full-scale launch operations of SRMs of all sizes so as to supplement the effects predictions models developed during the SPREE program. The recommendations presented herein are grouped into categories of scientific investigations and engineering investigations.

A. SCIENTIFIC INVESTIGATIONS

Data derived from these programs will ultimately allow development of environmental and effects prediction models as a function of more basic SRM characteristics and their exhaust parameters than those used in current technology. The results of these more generalized solutions will allow application of design data without the constraints of singular or a narrow selection of specific launch facility configurations.

1. Near Exit Plane External Ballistics Data

Little experimental data is available in terms of temperature and pressure distributions in the SRM exhaust jet stream in the vicinity of the nozzle exit. Most of the data available pertain to jet stream characteristics downstream of the sonic point. Available data have been analytically extrapolated from internal ballistics analyses. The data are required to determine particle compositions and velocities as a function of various chamber pressures, chamber temperatures, and exit Mach numbers.

2. Particle Sizes, Distributions, and Solidity

Data pertaining to alumina particle sizes, distributions, and solidity (whether some or all of the particles are hollow) in solid motor exhaust gases are required to theoretically evaluate the results of scale model tests. These data are imperative for specific impulse and thrust coefficient scaling. Data about where, in the jet stream, particles come into existence as solids would allow better description of plume characteristics. These data should be obtained for various chamber pressures and temperatures, for different propellant formulations, and for several thrust ranges of motors under varying expansion ratios.

3. Particle Velocities, Temperatures, and Paths in Exhaust Jets

Certain assumptions are necessary with regard to particle drag and heat transfer coefficients in a stream whose velocity and temperature vary along its length. Accurate definition of particle characteristics would permit more accurate assumptions and the development of a mathematical model that would allow a greater understanding of particle impact and erosion characteristics on hot surfaces. This latter condition is the fundamental difference between the effects of liquid engine and SRM exhaust gases.

4. Multiparticle Impact on Hot Protective Materials

The physical and chemical interaction of high-velocity alumina particles of a hot exhaust stream impinging on a nonmetallic surface needs to be investigated in some depth to better identify the characteristics of desirable protective materials. In addition to such physical properties as resiliency, hardness, and smoothness, the important chemical properties contributing to slow ablation should be identified, which will then allow materials to be selected on the basis of anticipated characteristics of exhaust plumes rather than on empirical methods.

5. Grain Particle Size Effects

The size of aluminum particles introduced into the solid propellant formulation under current technology is not critically controlled. The particles vary in size from 5 to 70 microns in somewhat random distribution. The question arises as to whether the size of the alumina particles in the exhaust stream is a function of the size of the aluminum particles in the grain itself. If there is such an interdependence that smaller aluminum particles result in smaller alumina particles, and these, in turn, result in reduced effects on launch facilities, it may be technically feasible to specify more stringent control of aluminum particle sizes in SRM propellants. This control may also produce improved performance of the SRM itself. This investigation can be performed by a relatively simple program in which erosion rates on a selected material are evaluated against the exhaust products of a standard SRM where size of aluminum particles is the only variable.

B. ENGINEERING INVESTIGATIONS

These investigations are aimed at expanding the use of SPREE data for applications more adaptable to variances in configuration details of launch facilities.

1. Flow Field of Deflected Jets

It is occasionally necessary to install some launch equipment in the direct path of a deflected exhaust stream. Current design philosophy assumes that the effects of the deflected exhaust are identical to a direct impingement condition, and, as such, temperature and pressure criteria are extrapolated for use. Frequently, unexpected damage to facilities is incurred that cannot be readily explained by this extrapolation process. Temperature and pressure measurements should be made to better identify the characteristics of the deflected exhaust stream.

2. Water Cooling of Ablative Deflectors

During the SPREE tests, it was observed that when a deflector was moist prior to exposure to hot exhaust products, the amount of erosion was reduced. It appears that refurbishment of ablative deflectors can be reduced if water is introduced on the deflector before or during exposure to SRM exhaust products. A complete investigation is merited to optimize the methods for wetting deflectors and to determine potential refurbishment cost savings.

3. Dual-Jet Impingement Investigations

With the advent of SRM strap-on configurations of liquid-propellant launch vehicles, complex exhaust plumes are possible. This is particularly true when the liquid motors and SRMs are ignited before vehicle liftoff. With these configurations, the liquid engines are ignited first and the SRMs are ignited just before liftoff. With the comparatively large thrust of the liquid engines, the exhaust stream could deflect the SRM exhaust so minimal effect is felt by the deflector. The cold-jet test program of SPREE demonstrated the effectiveness of a diversion jet that is approximately the magnitude of the jet to be diverted. Analyses and tests should be performed to evaluate the interaction of jets as a function of geometry.

4. Development of New Protective Materials

Adequate testing has now been accomplished to identify desirable characteristics of protective deflector coatings. Until such time as detailed quantitative chemical and mechanical properties can be specified, it will be necessary to evaluate new coatings by means of tests. This should only be undertaken, however, if (1) costs of these new materials, including installation and refurbishment, are competitive with the Fondu Fyre WA-1 now shown to be adequate for the launch facility applications, or (2) new generations of launch vehicles produce a considerably more severe exhaust gas environment than contemporary vehicles necessitating an improved deflector coating.

5. Ignition Overpressure Pulse Investigations

The magnitude of the pressure pulse generated at SRM ignition can be large (greater than 7 psig). This condition is centralized in the immediate vicinity of the engine compartment and presents a primary concern for vehicle engine compartment design on liquid vehicle tank bottoms. The magnitude of the pulse, however, is a function of the geometry at the launch duct, the deflector configuration, and separation distance between the exit plane of the SRM and first point of impingement of the exhaust gases. Previous analyses indicate that establishment of mathematical models to predict the phenomena are overly sensitive to detailed configuration effects. It will be necessary to perform scale model tests to measure the overpressure pulse for new and untried vehicle/launch facility configurations.

Martin-CR-65-93
(Vol I)

DISTRIBUTION

Copies	To
1 thru 20	Contracting Officer Kennedy Space Center National Aeronautics and Space Administration John F. Kennedy Space Center, Florida 32899
21	New Technology Representative Counsel for Patent Material Kennedy Space Center National Aeronautics and Space Administration John F. Kennedy Space Center, Florida 32899
22 thru 71	Martin Company Denver Division Denver, Colorado 80201 Attn: Libraries Section