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ALTITUDE TESTING OF HERCULES POWDER COMPANY BE-3 ROCKET MOTORS

(Phase II–Qualification and Acceptance Testing)

By

A. L. Cannell and C. F. Nokes, Jr.
Rocket Test Facility
ARO, Inc.

NOX

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a subsidiary of Sverdrup and Parcel, Inc.

April 1962
ARO Project No. 134114

(This abstract is classified CONFIDENTIAL.)

ABSTRACT

Eight Hercules Powder Company, BE-3, solid-propellant rocket motors designed for application as a retrorocket for the lunar impact capsule of the Ranger series of spacecraft were tested at pressure altitudes in excess of 100,000 ft as part of an acceptance and qualification program to determine ignition reliability and motor performance and to evaluate the erosion resistance of the inert components of the motors.

Seven motors were fired successfully. Vacuum total impulse varied from 52,329 to 52,736 lb_f-sec. Specific impulse based on the vacuum total impulse and the manufacturer's stated propellant weight varied from 275.1 to 276.0 lb_f-sec/lb_m. The average ignition lag time was 8 millisec. During one firing, the motor forward closure burned through as a result of failure of the igniter support tube.

Considerable nozzle deterioration (also encountered during previous tests) was prevalent in the region of the exit plane. A nozzle modification, consisting of an aluminum stiffening ring bonded to the nozzle exit cone, was used on two firings to strengthen the nozzle and was successful for the one firing during which the ring stayed in place.

(Catalog cards with an unclassified abstract may be found in the back of this document.)

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1.0 INTRODUCTION

The Hercules Powder Company (HPC), BE-3, solid-propellant rocket motor (Fig. 1) is to be used as the retro-rocket for the lunar impact capsule of the Ranger series of spacecraft (Fig. 2). The motor is designed to reduce the velocity of the capsule an incremental value of approximately 8660 fps prior to lunar impact.

Because of the necessity of knowing accurately the ballistic performance of these motors at near vacuum conditions, the final weight of the inert components, and also to establish motor reliability, a two-phase program was established at the Rocket Test Facility (RTF), Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC). Phase I of this program (Ref. 1) conducted during the period September 29 to October 6, 1961, consisted of development tests to determine performance of two motors and evaluate erosion resistance of the motor inert parts. The results of Phase II tests which consisted of qualification and acceptance tests of a series of similar motors are presented in this report. The Phase II tests were conducted during the period November 20-30, 1961.

The test was sponsored by AFSC and was conducted at the request of National Aeronautics and Space Administration. Personnel from Jet Propulsion Laboratory, Aeronutronics Division of Ford Motor Company, and Hercules Powder Company provided technical liaison for the test.

The primary objectives of the second phase of the test were to determine ignition capability; to accurately determine and evaluate repeatability of ballistic performance and to study erosion resistance of motor inert components (motor case, liner, nozzle, etc.).

The motors tested during Phase II differed from the Phase I motors (Ref. 1) in two respects. The nozzle wall thickness in the region of the exit plane (Fig. 1b) was doubled because of the deterioration experienced during Phase I testing. Also, the phenolic igniter support tube contained a steel sleeve (Fig. 1c) which was incorporated after difficulties were encountered during Phase I testing when one of the tubes was broken during connection of the CO_2 quench system (Ref. 1).

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2.0 APPARATUS

2.1 TEST ARTICLE

The HPC, BE-3, solid-propellant rocket motor (Fig. 1) is a full-scale flight weight motor. It has a spun fiberglass case with a Buna-N rubber insulator and contains a case-bonded propellant charge. The 16-deg conical nozzle (19:1 expansion ratio) has a graphite throat insert and a phenolic exit cone. The nozzle wall in the region of the exit plane was strengthened after Phase I testing by increasing the wall thickness as illustrated in Fig. 1b. For two firings (T3-41-09 and T3-41-10), an aluminum stiffening ring (Fig. 1f) was bonded on the nozzle exit cone to strengthen the nozzle in an attempt to prevent exit cone deterioration.

A modification of the igniter support tube was also incorporated which consisted of a steel sleeve in the phenolic tube (Fig. 1c).

Nominal design and performance characteristics are as follows:

Motor length, in.	32.6
Motor diameter, in.	18.18
Propellant weight, lb	190
Inert weight (including igniter and nozzle closure), lb	24
Total motor weight, lb	214
Mass fraction	.89
Throat area, in.^{2}	6.16
Nozzle expansion ratio	19.1
Average flow rate, lb/sec	18
Motor chamber pressure, psia	470
Average thrust, lb	5000

The motor propellant grain is cast in the configuration presented in Fig. 1a. The aluminized, double base propellant has the following composition:

Ingredient	Weight Percent
Nitrocellulose (12.6-percent) nitrogen content)	22.1
Nitroglycerin	28.3
Ammonium perchlorate	20.7
Aluminum	21.4
Triacetin	5.0
Resorcinol	1.5
2 Nitrodiphenylamine	1.0

The BE-3 motor is equipped with a basket type igniter (Fig. 1a) containing 100 gm of $BKNO_3$ pellets which are ignited with 2 U. S. Flare squibs. The nominal ignition current is 1.75 amp. The phenolic support tube for the basket igniter was utilized as a chamber pressure tap and also as a supply line for the CO_2 quench system provided for cooling the unit after burnout. For one firing (T3-41-09) a flight type igniter support tube which is filled with potting compound was used. Therefore no chamber pressure measurements were obtained for this firing.

The motors were equipped with a styrofoam throat closure. The closures were punctured prior to sealing the test cell; therefore the motor chamber pressure was equal to ambient pressure prior to ignition.

2.2 INSTALLATION

The aft end of the motor was installed in a mounting ring. This mounting ring rested on 4 roller bearings attached to a box frame structure. A steel plate attached to this box frame structure by tie rods was mounted on the front end of the motor. A thrust adapter was attached to this steel plate. The design of the harness allowed for longitudinal thermal expansion of the motor thus eliminating strain on the motor case. The firing harness was rigidly attached to a thrust cradle (Fig. 3) which was supported by three horizontal and two vertical flexure columns to permit a single degree of freedom. Load cells resisted and measured the axial thrust of the motor.

All motors were fired in Rocket Altitude Cell T-3 (Ref. 2). The pressure altitudes for these firings were established by using the RTF rotating exhaust machinery in conjunction with an auxiliary steam ejector. During the firings, the rocket motor exhaust was used as the driving gas for the ejector-diffuser system which maintained the simulated altitude in excess of 100,000 ft during the seven successful firings. During the first 5 firings, a 42-in.-diam, uncooled diffuser was used. A 48-in., water-cooled diffuser was used for the remaining three firings. A CO_2 quench system, exhausting through the igniter support tube, was installed to cool the motor after burnout in order to protect the motor case, case liner, and nozzle materials from the effects of residual heating, so that evaluation of the structural integrity of these components could be accomplished.

2.3 INSTRUMENTATION

Instrumentation (Table 1) was provided to measure four thrust signals, two chamber pressure signals, three cell pressure signals, one cell temperature signal, one motor case temperature signal, and one ignition pulse signal. The four thrust measurements were obtained from two double-bridge, strain-gage-type load cells having a range of 0 to 10, 000 lb. Chamber pressure was measured by two bonded strain-gage transducers, and the cell pressure was measured by unbonded strain-gage transducers. Iron-constantan (I-C) thermocouples were used to measure motor and cell temperatures.

The output signals from the load cells and transducers, were indicated in totalized digital form on a visual readout millivolt-to-frequency converter and also in analog form on a photographically recording, galvanometer-type oscillograph. A magnetic tape system recording in frequency form stored the signals from the converter. The signals stored on the magnetic tape system were reduced to engineering unit data by an ERA-1102 digital computer. The computer system provided data printouts of absolute values and accumulative integral values for all signals at 0. 1-sec intervals.

The printouts from the digital computer were considered to be primary data. Oscillograph records (trace speed of 25 in. /sec) provided an independent backup for all data. Continuous recording null-balance potentiometers recorded thrust, chamber pressure, cell pressure, and cell temperature to provide data for analysis immediately following the firings.

Visual observation of the motor firings was provided by a closed circuit television monitor. A visual record of the firings was provided by high speed (1000 frames per second) movie cameras.

3.0 PROCEDURE

After arrival at AEDC, the BE-3 rocket motors were radiographically inspected and then stored in a temperature-controlled storage area (70° ± 1°F) for a period of at least 72 hr prior to firing. Continuous temperature records were maintained for each motor on strip-type recorders. These recorders were regularly checked against a mercury bulb thermometer to document any drift in calibration. During storage, further visual and dimensional inspections of the motors were made including a fit check of the thrust adapter hardware, measurement of the igniter squib resistances, weighing and photographing the motor, and measurement of the nozzle exit plane diameters. The nozzle throat diameter measurements could not be obtained because of the nozzle throat closures. These measurements along with weights of all motor components were provided by the manufacturer.

After inspection and temperature conditioning were completed, the motor was transported to the cell in an insulated container. Temperatures were continuously monitored during this period with the same strip recorders used in the storage area. After the motor was installed in the test cell, all instrumentation connections were made, the circuits checked and the CO_2 quench system connected. The ignition circuit was continuity checked from the control room, and the sea-level calibrations were performed.

The pressure and temperature data systems were calibrated at sea level by a four-step electrical calibration system using known resistances to simulate known signal levels. The resistors were located in the test cell, and the systems were remotely energized from the control room.

The thrust measuring system was calibrated at sea level by a remotely operated dead weight calibrator at nominal thrust levels of 0, 2000, 4000, 4800, 5600, and 6400 lb_f. The weights used on the calibrator had previously been compared with secondary standard weights whose error is 20 parts in a million. The thrust calibrator also was calibrated with a high accuracy load cell, (± 10 lb) prior to motor installation and system calibration.

The test cell was sealed, the pressure in the test cell was reduced to simulated altitude conditions, and the required checks and altitude calibrations were accomplished. The ignition circuit resistance was then adjusted to provide the proper firing current and the motor was then fired. Approximately 10 sec after motor burnout, the CO_2 quench system was actuated allowing approximately 50 lb of CO2 to flow through the engine in a period of 15 min. After firing and while the test cell was still at simulated altitude conditions, the calibration procedures were repeated.

After removing the motor from the test cell, post-firing inspection was performed which consisted of weighing the motor assembly and measuring the nozzle throat diameter. The nozzle exit diameters were impossible to obtain to any degree of precision because of nozzle deterioration. Photographic documentation of the motor post-fire condition was also accomplished.

4.0 RESULTS AND DISCUSSION

4.1 GENERAL

Eight HPC, BE-3, solid-propellant rocket motors were fired at simulated altitudes in excess of 100,000 feet for qualification and acceptance tests. The typical variation of axial thrust, chamber pressure, and test cell pressure throughout one of the firings is shown in Fig. 4. Typical build-up of thrust and chamber pressure immediately following ignition are shown in Fig. 5. The primary objectives were to determine ignition capability, to establish accurately and to determine the repeatability of motor performance. In addition, an evaluation of the erosion resistance of inert components including a strengthened nozzle exit cone was accomplished. The nozzles used on these motors were strengthened in the exit plane region by doubling the thickness of the nozzle wall at the exit plane (Fig. 1b) from that used on the motors tested during Phase I.

Seven motors were fired successfully. During the firing of one motor, the igniter support tube failed (Fig. 6) allowing hot gases to be expelled through the front end of the motor and finally resulting in burn-through of the forward closure.

The igniter support tube as used during this firing did not simulate the flight application. For flight application, the tube is filled with potting compound which substantially increases its strength. The pressure adapter used for chamber pressure measurements and CO_2 quench system connections was not flight hardware.

The motors were essentially identical units except for variations in treatment during manufacture, such as prolonged vacuum conditioning, ethylene trioxide treatment, and sterilization. The difference in treatment is outlined in Table 2 along with variations in testing environment and nozzle modifications for each unit. The motor tested in run T3-41-09 had a flight type igniter support tube which is filled with potting compound. Therefore, no chamber pressure measurements were obtained for this firing.

4.2 IGNITION CAPABILITY

All motors ignited successfully at pressure altitudes above 100, 000 ft. The ignition lag time, defined as the time interval between the application of current to the igniter squibs and the first perceptible rise in thrust varied from 3 to 18 millisec, the average value being approximately 8 millisec (Table 3). A reproduction of a portion of the high-speed, oscillograph-type analog data showing thrust and chamber pressure build-up characteristics for a typical firing are presented in Fig. 5. The natural frequency of the thrust stand used for this test was approximately 40 cps.

4.3 VACUUM PERFORMANCE

4.3.1 Thrust and Impulse

Total impulse corrected to vacuum conditions for the seven successful firings varied from 52, 329 to 52, 736 lbf-sec. Total impulse as presented in this report is the average of 4 independent channels of thrust integral data. The vacuum correction based on prefire nozzle exit area is approximately 0. 3 percent of the total measured values. The average total impulse, corrected to vacuum conditions, for seven firings was 52, 595 lbf-sec. Propellant weights for the motors tested varied from 190. 03 lb to 191. 26 lb (Table 4); this variation accounts in part for the difference in delivered total impulse.

Total impulse at vacuum conditions as a function of total propellant weight is shown in Fig. 7 for the seven successful firings. Also shown is the predicted variation of total impulse as a function of propellant weight based on the average specific impulse for seven firings. It is noted that all data fall within ±0. 25 percent of the predicted values based on the average specific impulse for the seven firings.

Specific impulse for each motor is based on the average of the 4 channels of total impulse corrected to vacuum conditions. Specific impulse based on the manufacturer's stated propellant weight for the seven successful firings varied from 275. 1 to 276. 0 $\text{lb}_f\text{-sec/lb}_m$. The average value for the seven motors was 275. 4 $\text{lb}_f\text{-sec/lb}_m$.

Specific impulse based on the pre- and post-firing weight difference is presented in Table 3 for comparison with values obtained using the stated propellant weights. The pre-firing weights used were supplied by HPC and are believed to be accurate within ±0. 01 lb. The post-firing weights were measured at AEDC and are believed to be accurate to ±0. 012 lb. Specific impulse calculated on this basis varied from 269. 5 to 271. 1. The average specific impulse for the seven successful firings was 270. 4.

4.3.2 Thrust Coefficient

Average thrust coefficients for total propellant burning time were calculated for each motor by the following method

$$C_f = \frac{I_v}{A_{throat} \int_0^{t_B} P_{chamber}\, dt}$$

where

I_V = Vacuum corrected total impulse

A_{throat} = Average of pre-firing and post-firing throat areas

$\int_{0}^{t_B} P_{chamber}\, dt$ = Integral of chamber pressure with respect to time for total burning time.

The thrust coefficients calculated by this method varied from 1.82 to 1.86; an average value of 1.84 was obtained for the six firings. When an area ratio of 19:1 and a ratio of specific heats of 1.17 is assumed, the theoretical thrust coefficient for this nozzle would be 1.81.

4.3.3 Repeatability of Performance

Total and specific impulse were the performance parameters which were obtained with the greatest precision for these motors. Total impulse was obtained by averaging four channels of thrust integral data and correcting this average to vacuum conditions. Standard Deviation of the four channels of thrust integral data varied from 0.024 to 0.093 percent (Table 5). This is an indication of the precision of instrumentation used.

Specific Impulse was calculated for each motor based on the vacuum corrected total impulse and the manufacturer's stated propellant weights. These weights were stated to be accurate to ±0.01 lb or approximately ±0.005 percent of the total propellant weight.

It was decided to base repeatability of performance on specific impulse rather than total impulse because of the variation in propellant weights of the motors. While the variation in performance due to difference in propellant weights is of interest, it can be predicted from specific impulse. It is therefore of primary importance to know what variation in specific impulse can be expected for a group of motors because this value reflects performance variations due to slight differences in propellant composition, nozzle area ratio, nozzle misalignment, etc.

Maximum deviation of specific impulse from the average of the seven firings was 0.22 percent which is within the estimated maximum deviation of the instrumentation used to obtain the impulse data. Standard deviation of specific impulse for seven firings was 0.11 percent (Table 5).

4.4 INERT COMPONENTS EVALUATION

As in Phase I testing, the motor case and case liner material appeared to be satisfactory. However, additional strengthening of the nozzle exit cone to prevent deterioration may be necessary.

Film sequences of three firings considered typical are presented in Fig. 8. Post firing photographs of all motors are presented in Fig. 9. As can be seen, extensive nozzle deterioration was experienced during the first four firings. During the fifth firing a 2-in. strip of reflective aluminum tape was placed on the nozzle exit cone. No improvement in nozzle condition was noted. After the fifth firing the 42-in. uncooled diffuser was removed and replaced with a 48-in. water-cooled diffuser. This was done to determine if nozzle deterioration was being caused by radiation from the diffuser walls. During the first firing using this diffuser, the chamber pressure adapter failed; however, the nozzle deterioration still occurred during this firing which verified the existence of a nozzle problem rather than a testing environment problem.

The last two units were fired with an aluminum stiffening ring incorporated on the nozzle exit cone (Fig. 1f) as explained in Apparatus Section. This ring worked well on the first attempt (Fig. 9g) even though it became partially unbonded. On the second trial it became completely unbonded from the nozzle during firing and extensive deterioration was again experienced (Fig. 9h). It is believed that this ring would be sufficient as a fix if a sufficiently strong bonding agent is used and provided the weight penalty involved is not prohibitive.

Nozzle deterioration of this type could result in difficulties in using these motors because of the possibility of intolerable thrust vectors resulting from unsymmetrical deterioration.

The loss of motor inert component weight was obtained by comparing the manufacturer's stated inert components weight with the measured post-firing weight. This was done for runs T3-41-03 through T3-41-07. Run T3-41-08 was not used because of failure which occurred. Runs T3-41-09 and T3-41-10 were not used because the post firing weights included the aluminum ring and insulating material used in the nozzle modification. For the five firings considered, the loss in inert components averaged 3.67 lb. This value includes the nozzle closure (0.22 lb) and the igniter (0.82 lb). The remainder of the loss was case liner, nozzle material, potting compound, etc.

4.5 TEST FAILURE (FIRING T3-41-08)

During motor firing T3-41-08, a failure of the phenolic igniter support tube occurred and resulted in loss of the motor forward closure (Fig. 6d).

The motor ignited normally, but approximately 0.8 sec after ignition both chamber pressure signals were lost (Fig. 6a). The motor continued to burn at a slightly reduced thrust level until approximately 6.5 sec, and at this time thrust began to decay gradually; burning ceased at 11.25 sec.

From inspection of the motor condition, chamber pressure and thrust data, and movie film, the following sequence of events was surmised.

Since both chamber pressure signals were in phase and were lost simultaneously, it is believed that the igniter support tube failed at the threaded portion (Fig. 1c) during or shortly after ignition causing loss of chamber pressure signals and leaving the pressure adapter supported only by the CO_2 quench line and the pressure transducer sensing lines. The hot gases expelled through the 3/8-in.-diam hole in the phenolic tube caused the erosion of the pressure adapter (Fig. 6b) and also burned the CO_2 and pressure transducer lines.

The hot gases were deflected by the thrust pylon and adapter and recirculated around the motor until the forward closure separated. At this time (6.5 sec) axial thrust began to decay until burning ceased at 11.25 sec.

5.0 SUMMARY OF RESULTS

Eight BE-3, solid-propellant rocket motors were fired at simulated altitude conditions in excess of 100,000 ft. The results of the test may be summarized as follows:

1. All motors were ignited successfully, the average ignition lag time being 8 millisec.
2. The average total impulse corrected to vacuum for seven successful firings was 52,595 lb_f-sec.
3. The average specific impulse for seven firings was 275.4 lb_f-sec/lb_m based on the manufacturer's stated propellant weight.

4. Repeatability of motor performance based on specific impulse using one sigma standard deviation was ±0.11 percent.
5. Extensive nozzle exit cone deterioration was experienced during the first six firings. For the seventh and eighth firings, an aluminum stiffening ring was bonded on the nozzle exit cone to prevent deterioration. This ring prevented the nozzle exit cone deterioration on the seventh firing and presumably would have worked on the eighth firing if it had remained bonded to the nozzle.
6. During one firing the motor forward closure burned through because of the failure of the igniter support tube at the forward end. This tube is used as a chamber pressure sensing line and a support for the chamber pressure adapter during static firings. For flight application the tube is used only as a conduit for ignition leads and is filled with potting compound.

REFERENCES

1. Cannell, A. L. and Nokes, C. F., Jr. "Altitude Testing of Hercules Powder Company BE-3 Rocket Motors (Phase I)." AEDC-TDR-62-8, January 1962. (Confidential)
2. Test Facilities Handbook (3rd Edition). "Rocket Test Facility, Vol. 2." Arnold Engineering Development Center, January 1961.

TABLE 1
INSTRUMENTATION

Parameter	Units	Range of Sensing Instrument	Recording Method	Calibration Method	Estimated System Deviation at Operating Level
Axial Thrust Total Impulse	lb_f lb_f-sec	10, 000 lb ———	Magnetic Tape "	Dead Weight ———	±1. 00% ±0. 25%
Chamber Pressure Integral	psia psia-sec	750 psia ———	" "	Electrical ———	±3. 00% ±1. 00%
Cell Pressure Integral	psia psia-sec	1 psid ———	" "	Electrical ———	±5. 00% ±2. 00%
Cell Temperature	°F	I-C (0-150°F)	Null Balance Potentiometer	Electrical	±3. 00%
Manufacturer's Stated Pre-Firing Motor Weight	lbm	Not Known	Visual	Dead Weight	±0. 005%
Post-Firing Motor Weight	lb_m	0 - 20 lb	"	"	±0. 08%

TABLE 2

MOTOR AND TEST ENVIRONMENT VARIATIONS

Parameter	T3-41-03 S/N 37	T3-41-04 S/N 38	T3-41-05 S/N 39	T3-41-06 S/N 42	T3-41-07 S/N 40	T3-41-08 S/N 41	T3-41-09 S/N 45	T3-41-10 S/N 43
HPC Treatment	(2)	(2)	(2)	(3) and (4)	(1)	(1)	(3) and (4)	(3) and (4)
Temperature Conditioning at AEDC	*	*	*	*	*	*	*	*
Average Cell Pressure during Firing, psia	.1313	.1246	.1225	.1205	.1310	.3009	.1098	.1088
Type of Diffuser Installation	42 in. Uncooled	42 in. Uncooled	42 in. Uncooled	42 in. Uncooled	42 in. Uncooled	48 in. H_2O Cooled	48 in. H_2O Cooled	48 in. H_2O Cooled
Nozzle Shielding or Modification	None	None	None	None	2 in. Width Aluminum Reflective Tape	None	Aluminum Ring Stiffener	Aluminum Ring Stiffener

(1) No special treatment
(2) 72 hr at vacuum
(3) Ethylene trioxide treatment
(4) Sterilize entire assembly
* 72 hr at 70° ± 1°F

TABLE 3

MOTOR PERFORMANCE TABLE

Parameter	T3-41-03 S/N 37	T3-41-04 S/N 38	T3-41-05 S/N 39	T3-41-06 S/N 42	T3-41-07 S/N 40	T3-41-08 S/N 41	T3-41-09 S/N 45	T3-41-10 S/N 43
Total Motor Burning Time, sec	9.897	9.869	10.075	10.120	9.882	11.250	9.794	10.005
Ignition Lag Time, sec	0.007	0.003	0.018	0.005	0.005	0.010	0.012	0.005
Measured Impulse, lb-sec								
Sys. 1	52,502	52,537	52,565	52,640	52,459	45,121	52,220	52,426
Sys. 2	52,482	52,491	52,540	52,617	52,464	45,190	52,173	52,485
Sys. 3	52,451	52,451	52,472	52,528	52,435	45,077	52,169	52,434
Sys. 4	52,462	52,482	52,489	52,577	52,454	45,121	52,255	52,438
Average	52,474	52,490	52,517	52,591	52,453	45,127	52,204	52,446
Chamber Pressure Integral, psia-sec							Not Measured	
Sys. 1	4371	4304	4387	4348	4324	—		4303
Sys. 2	4442	4368	4461	4426	4382	—		4310
Average	4407	4336	4424	4387	4353	—		4307
$\int_0^{t_B} P_{chamber} \times dt \times A_{throat}$, lb_f-sec	28,835	28,331	28,888	28,779	28,599	—	Not Measured	28,238
Cell Pressure Integral, psia-sec								
Sys. 1	1.30	1.21	1.23	1.21	1.29	—	1.071	1.104
Sys. 2	1.30	1.25	1.24	1.23	1.30	—	1.080	1.075
Average	1.30	1.23	1.235	1.22	1.295	—	1.076	1.0895
$\int_0^{t_B} P_{cell} \times dt \times A_{exit}$, lb_f-sec	152.5	143.9	144.8	144.6	152.0	—	125.0	127.1
I_v (Vacuum Corrected Total Impulse), lb_f-sec	52,626	52,634	52,662	52,736	52,605	—	52,329	52,573
Specific Impulse, lb_f-sec/lb_m								
(1) Manufacturer's Propellant Weight	275.5	275.2	275.3	276.0	275.1	—	275.4	275.6
(2) Measured Weight Difference	271.1	270.3	269.9	269.5	270.6	—	271.4	270.3
$C_f\ I_f\ \frac{I_v}{\int P_{chamber} \times dt \times A_{throat}}$	1.825	1.858	1.823	1.832	1.839	—	—	1.862

TABLE 4

PHYSICAL DATA TABLE

Parameter	T3-41-03 S/N 37	T3-41-04 S/N 38	T3-41-05 S/N 39	T3-41-06 S/N 42	T3-41-07 S/N 40	T3-41-08 S/N 41	T3-41-09 S/N 45	T3-41-10 S/N 43
Pre-Firing Weight (HPC)	213.90	214.74	214.19	214.70	213.90	213.75	212.97	214.37
Post-Firing Weight (AEDC)	19.77	20.02	19.08	19.08	19.34	———	20.16	19.87
Manufacturer's Stated Inert Components Weight	22.862	23.504	22.934	23.640	22.701	22.903	22.937	23.598
Propellant Weight								
(1) HPC (Manufacturer)	191.04	191.24	191.26	191.06	191.20	190.84	190.03	190.78
(2) Measured Weight Difference Based on Pre- and Post-Fire Weights	194.13	194.72	195.11	195.62	194.56	———	192.81	194.50
Mass Fraction								
(1) Based on Manufacturer's Stated Weights	0.893	0.891	0.893	0.890	0.895	0.892	0.892	0.890
(2) Based on Weight Difference	0.908	0.907	0.911	0.911	0.910	———	0.905	0.907
Throat Area								
(1) Pre-Fire	6.290	6.290	6.290	6.290	6.290	6.290	6.290	6.290
(2) Post-Fire	6.796	6.777	6.770	6.830	6.850	6.834	6.933	6.824
Average	6.543	6.534	6.530	6.560	6.570	6.562	6.612	6.557
Percent Change	8.04	7.74	7.63	8.23	8.52	8.29	10.22	8.49
Exit Plane Area								
(1) Pre-Fire	117.302	117.014	117.263	118.553	117.398	117.321	117.129	116.688
(2) Post-Fire	*	*	*	*	*	*	115.352	*
Average	*	*	*	*	*	*	116.241	*
Pre-Firing Area Ratio	18.649	18.603	18.643	18.848	18.664	18.652	18.621	18.551

*Not Available

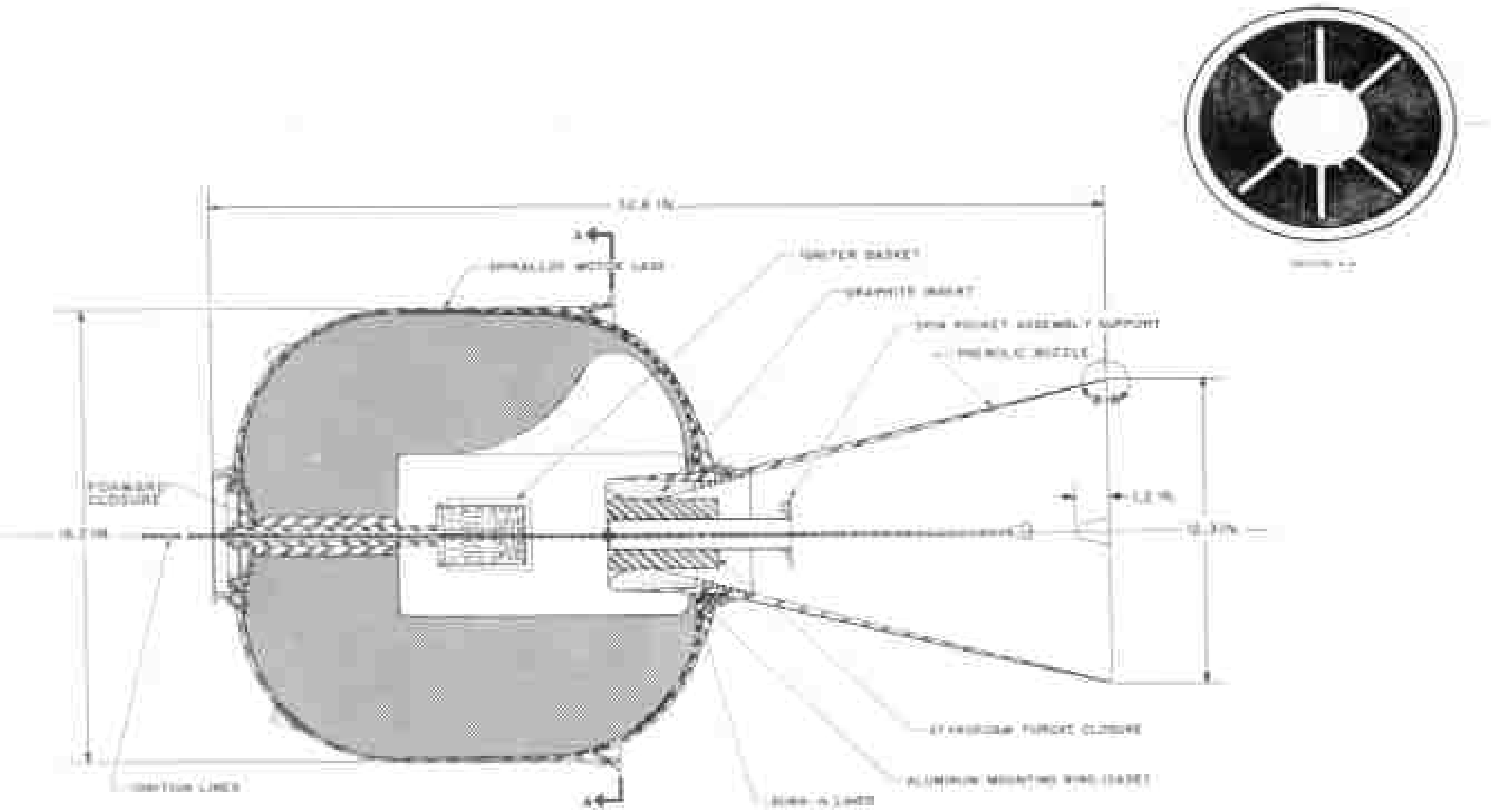
TABLE 5
PERFORMANCE REPEATABILITY

	T3-41-03 S/N 37	T3-41-04 S/N 38	T3-41-05 S/N 39	T3-41-06 S/N 42	T3-41-07 S/N 40	T3-41-08 S/N 41	T3-41-09 S/N 45	T3-41-10 S/N 43
Measured Impulse, lb_f-sec								
(1)	52, 502	52, 537	52, 565	52, 640	52, 459	45, 121	52, 220	52, 426
(2)	52, 482	52, 491	52, 540	52, 617	52, 464	45, 190	52, 173	52, 485
(3)	52, 451	52, 451	52, 472	52, 528	52, 435	45, 077	52, 169	52, 434
(4)	52, 462	52, 482	52, 489	52, 577	52, 454	45, 121	52, 255	52, 438
Average	52, 474	52, 490	52, 517	52, 591	52, 453	45, 127	52, 204	52, 446
Standard Deviation, from Mean, of 4 Channels								
lb-sec	±21.71	±35.57	±43.39	±49.10	±12.69	±46.69	±41.00	±26.64
percent	± .0414	± .0678	± .0826	± .0934	± .0242	± .1035	± .0785	± .0508
Specific Impulse Based on Average Total Impulse Corrected to Vacuum lb_f-sec/lb_m	275.5	275.2	275.3	276.0	275.1	Not Applicable	275.4	275.6

Average Specific Impulse = 275.4 lb_f-sec/lb_m

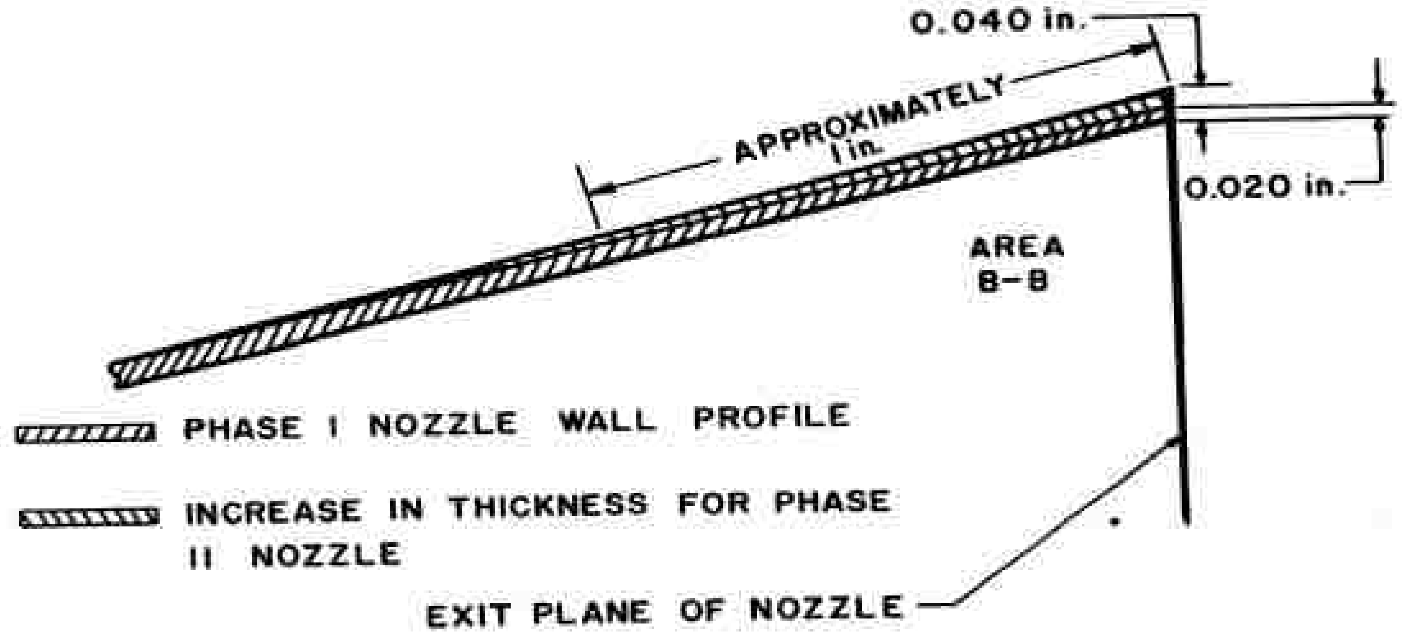
Standard Deviation between Firings Based on Secific Impulse Values $= \sqrt{\dfrac{(.1)^2+(.2)^2+(.1)^2+(.6)^2+(.3)^2+(0)^2+(.2)^2}{6}} = \pm .30 \ \dfrac{lb_f - sec}{lb_m}$

Relative Standard Deviation $= \dfrac{.30 \times 100}{275.4} = 0.11$ percent



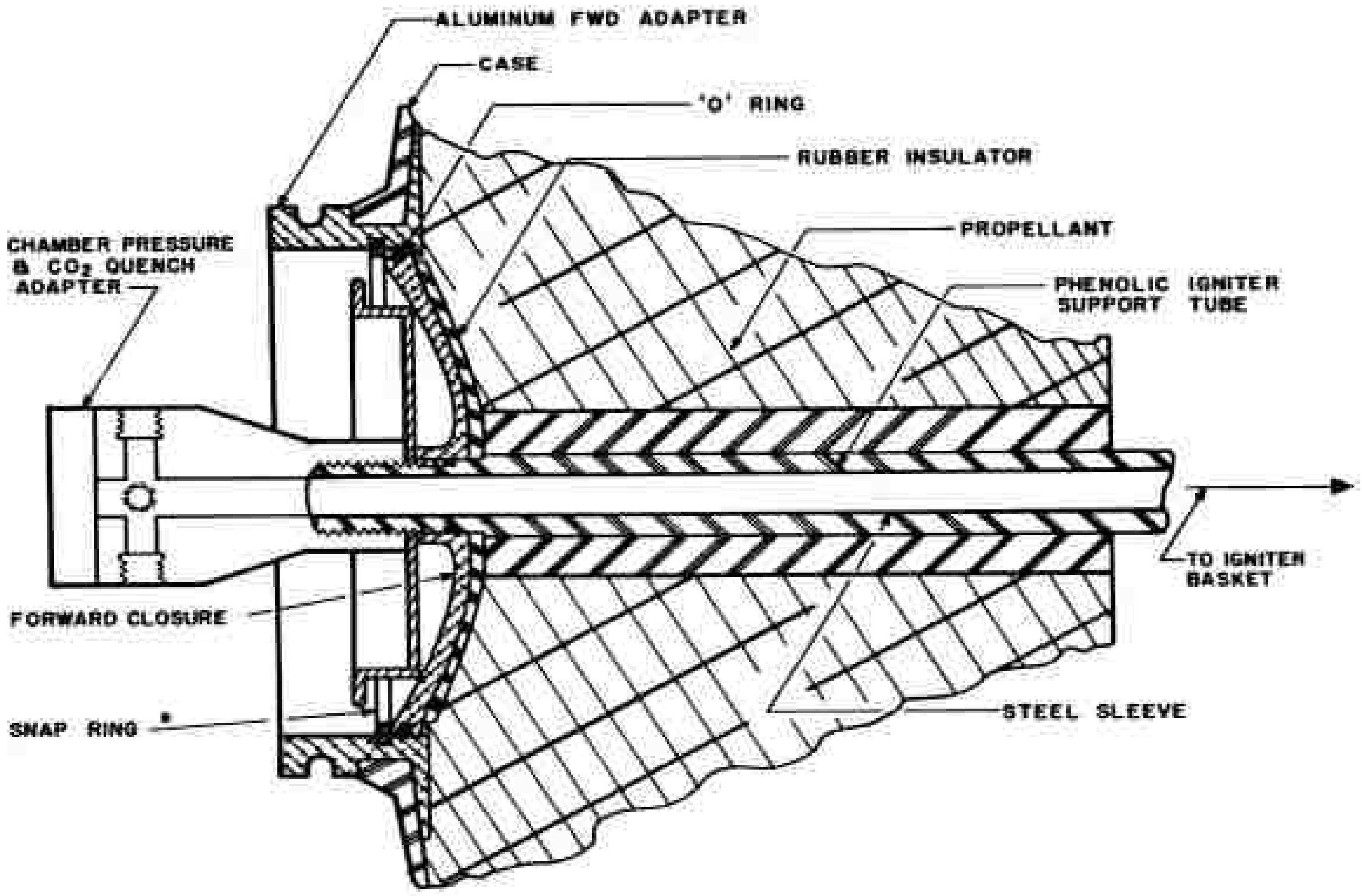
a. Schematic of Motor

Fig. 1 Typical BE-3 Rocket Motor



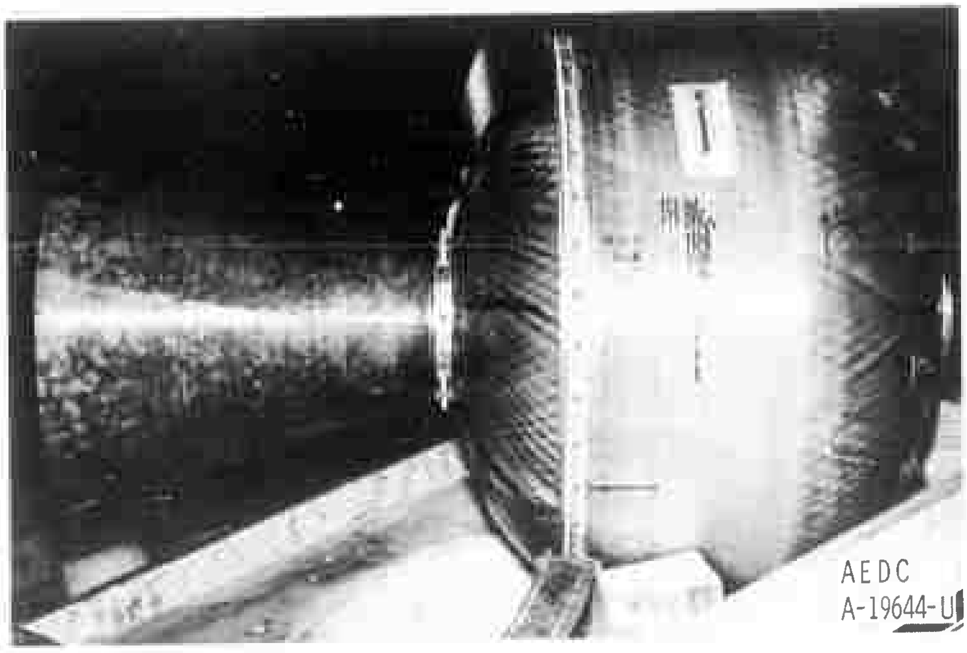
b. Nozzle Wall Profile

Fig. 1 Continued

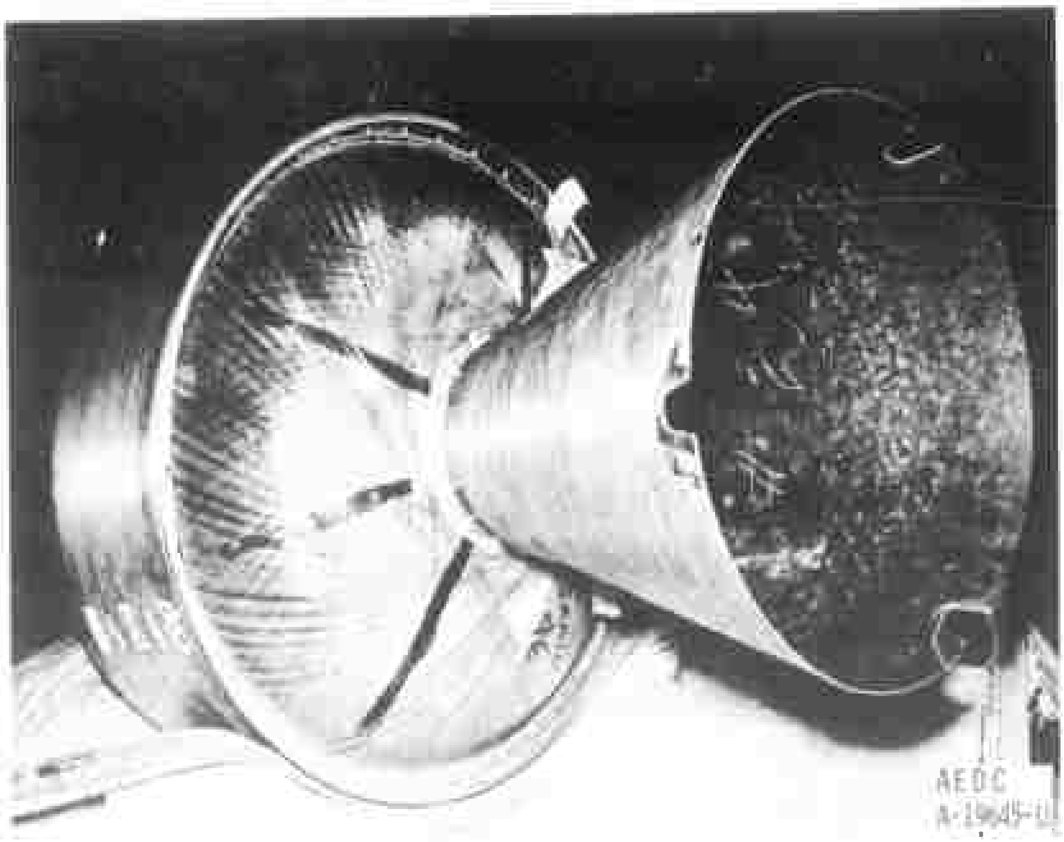


c. Schematic of Chamber Pressure Adapter and Support Tube Assembly

Fig. 1 Continued

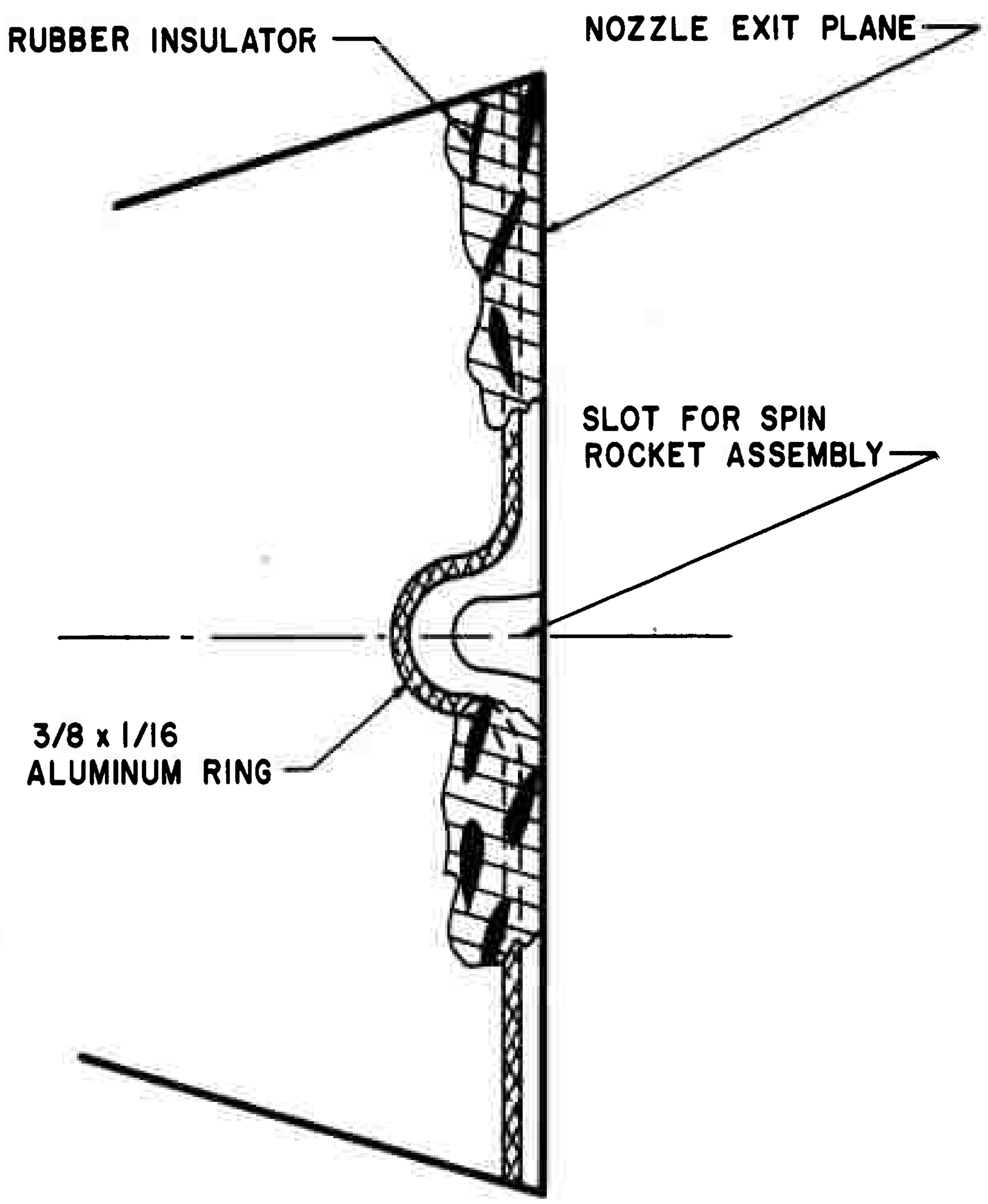


d. Photograph of Motor Assembly



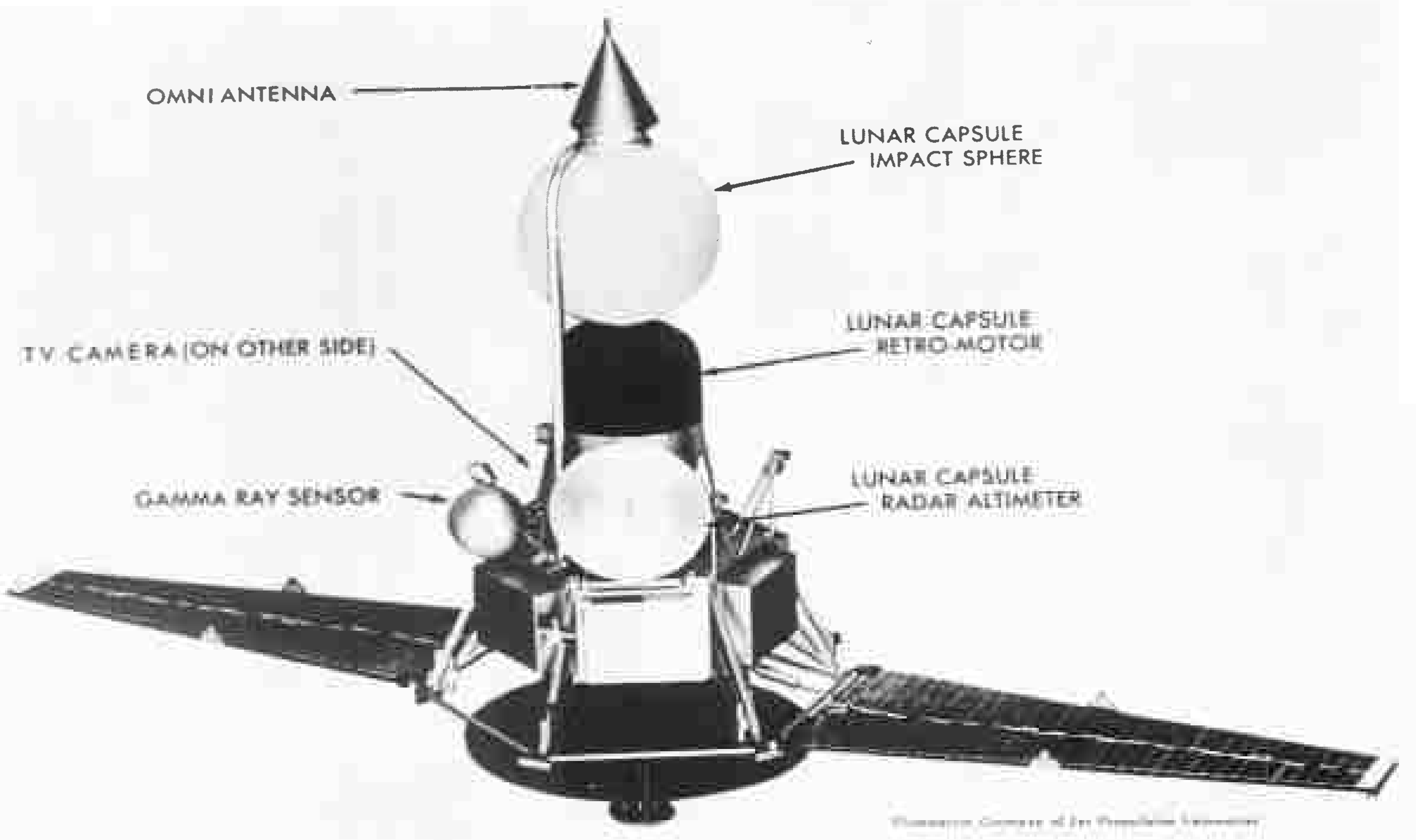
e. Photograph of Nozzle Exit Region

Fig. 1 Continued



f. Nozzle Modification (Runs T3-41-09 and T3-41-10)

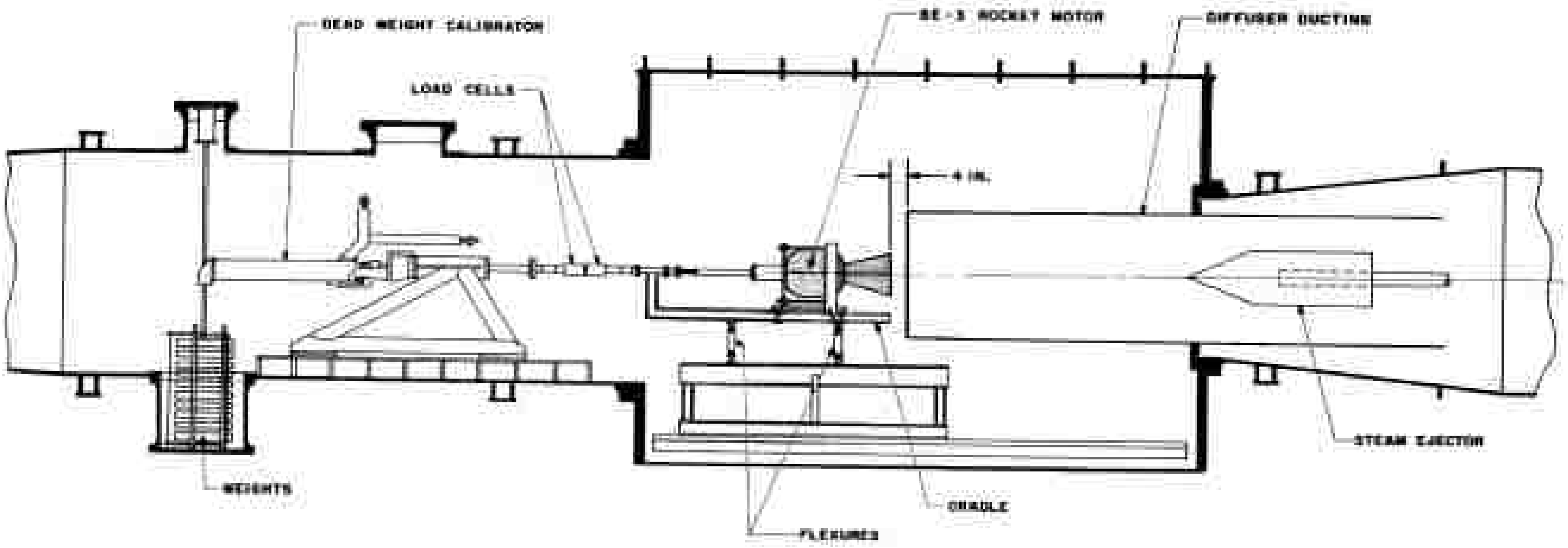
Fig. 1 Concluded



Photograph Courtesy of Jet Propulsion Laboratory

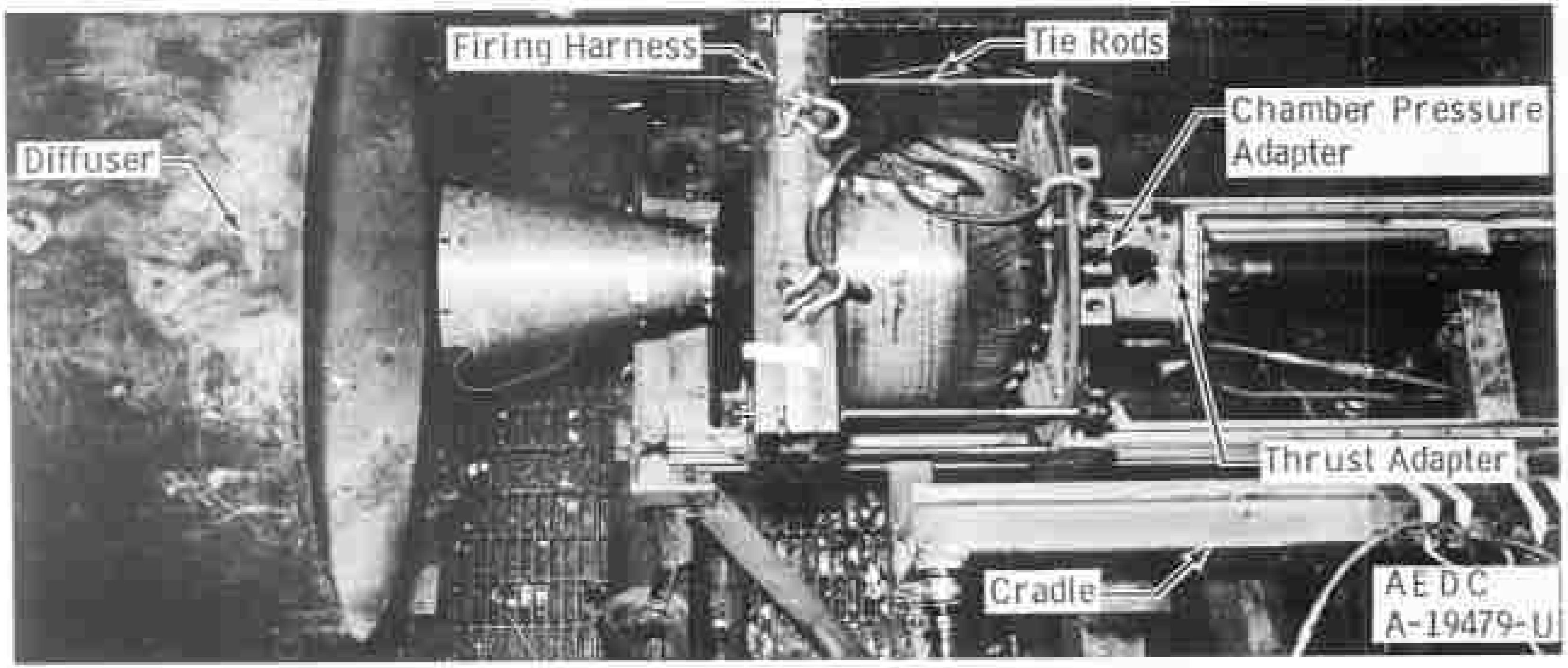
RA-3

Fig. 2 Schematic of Ranger Vehicle



a. Schematic

Fig. 3 Installation of Motor in T-3 Test Cell



b. Photograph

Fig. 3 Concluded

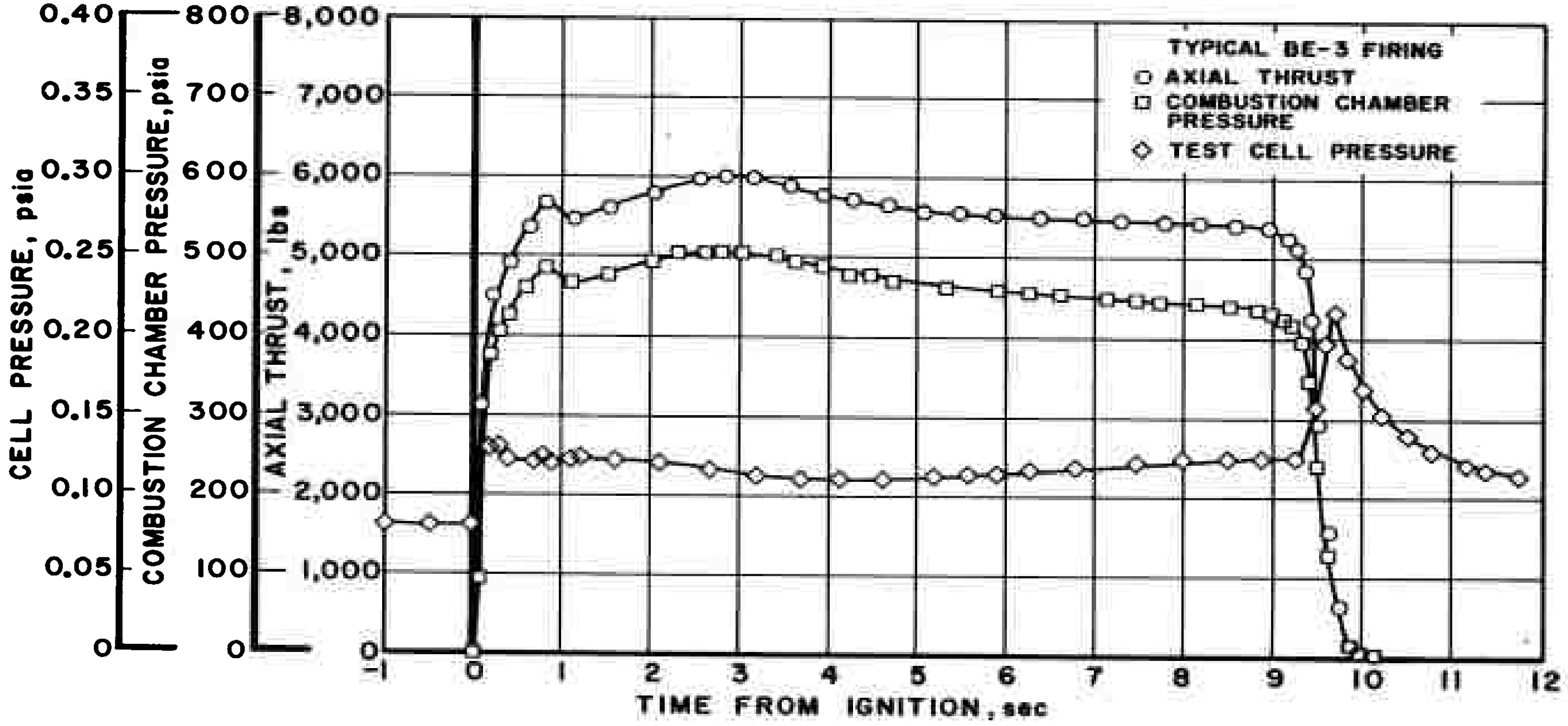


Fig. 4 Typical Motor Firing

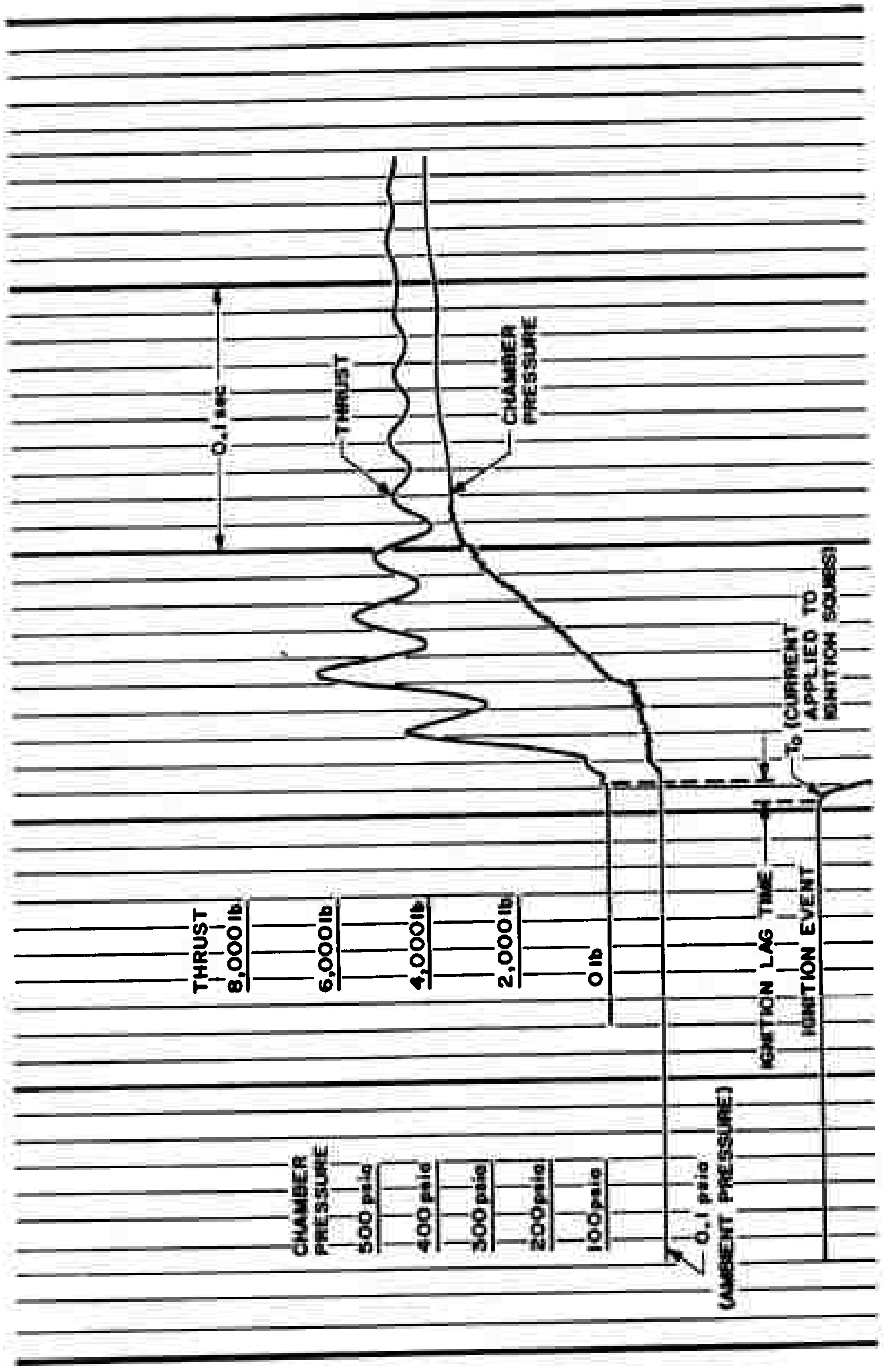
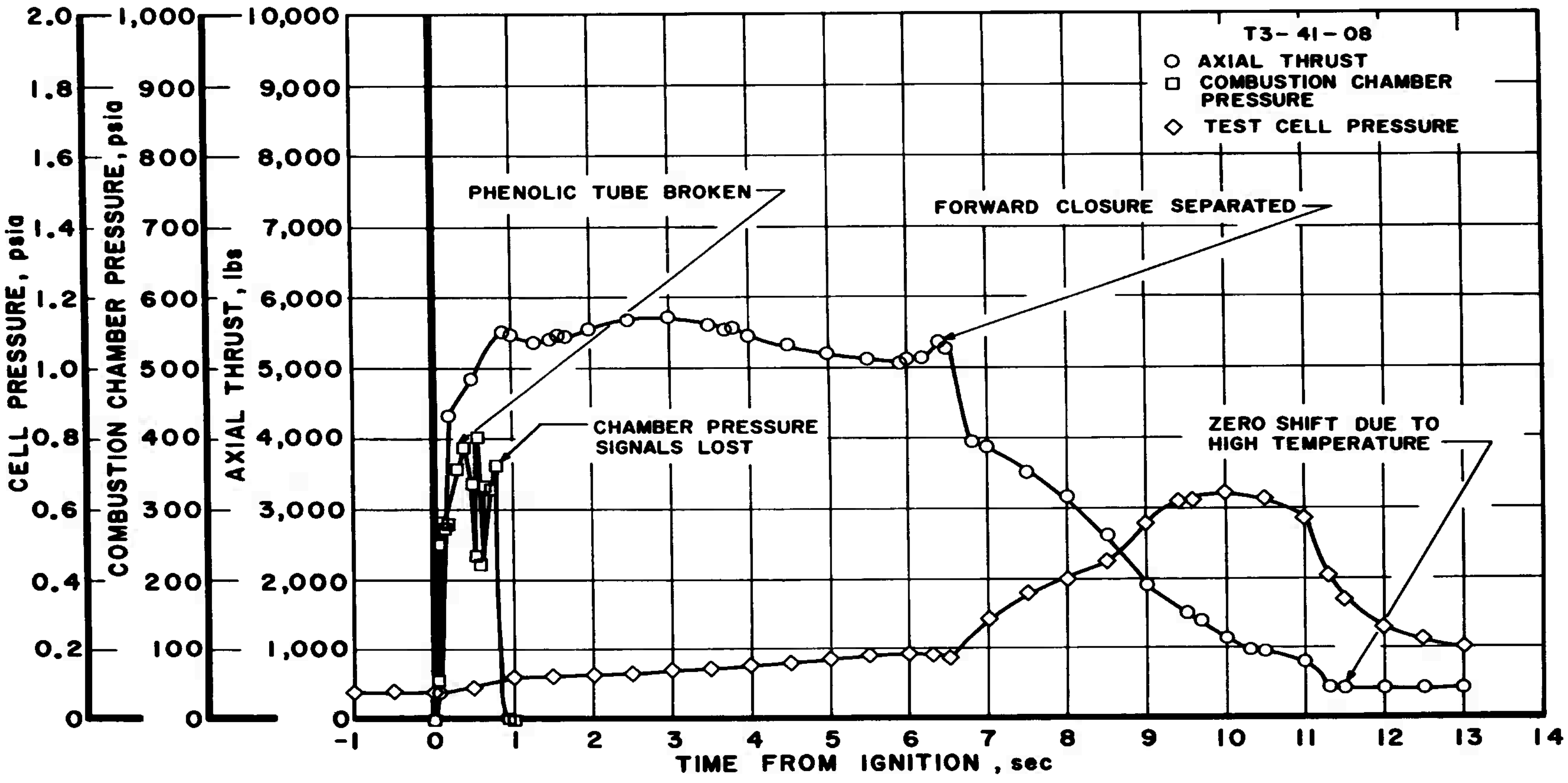
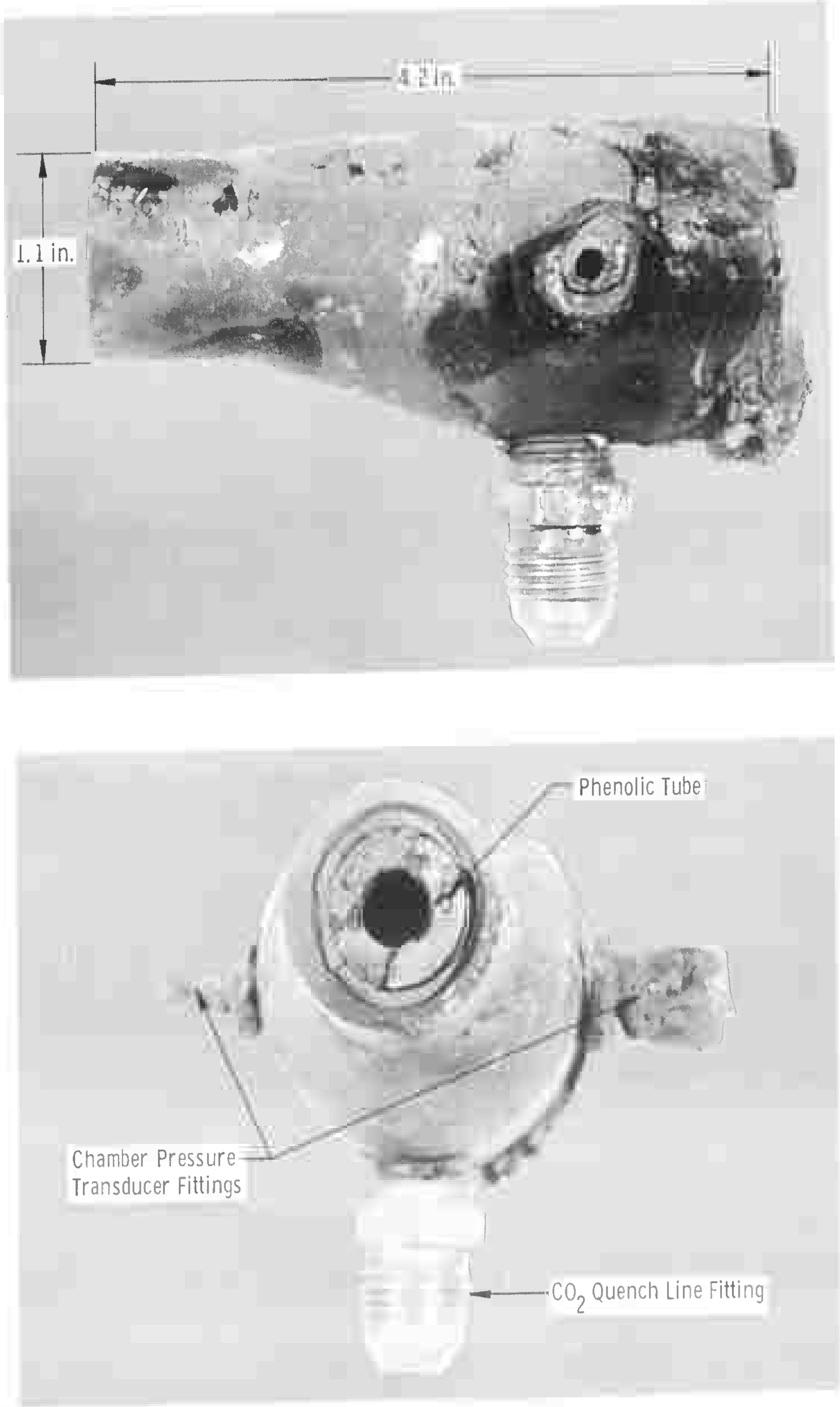


Fig. 5 Typical Thrust and Chamber Pressure Buildup Characteristics



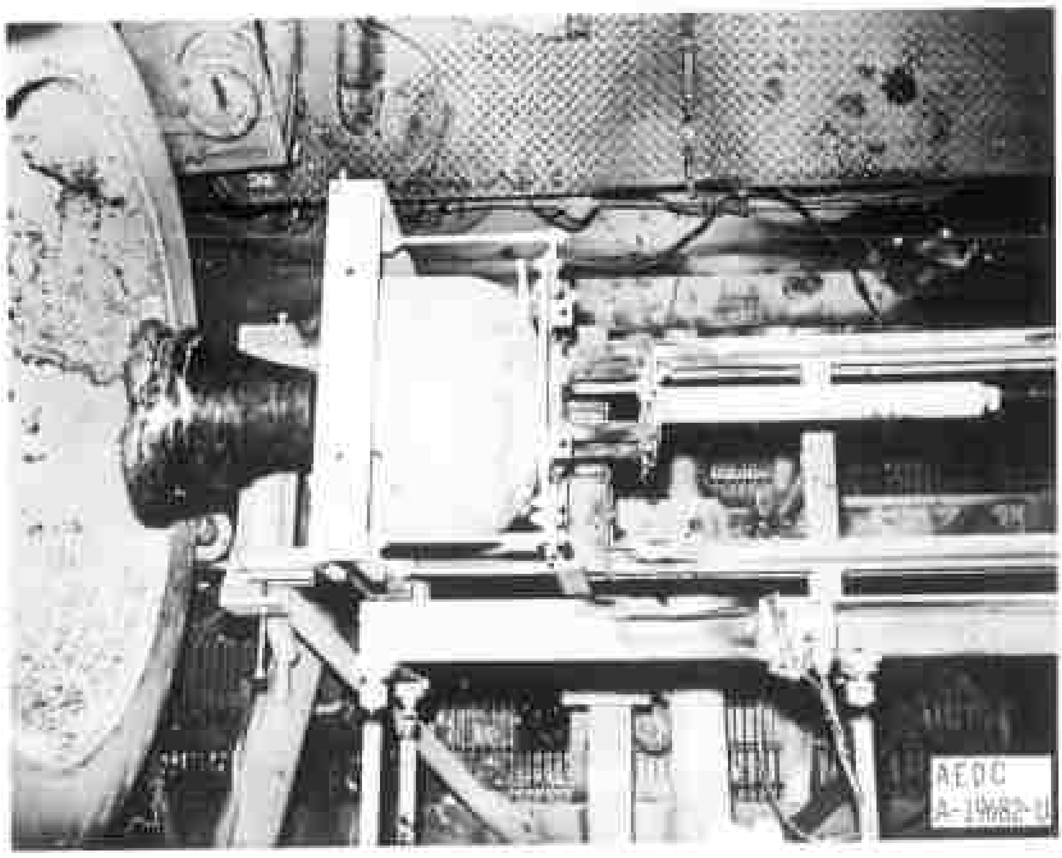
a. Thrust, Chamber Pressure, and Cell Pressure vs Time

Fig. 6 Firing T3-41-08 (Motor Failure)

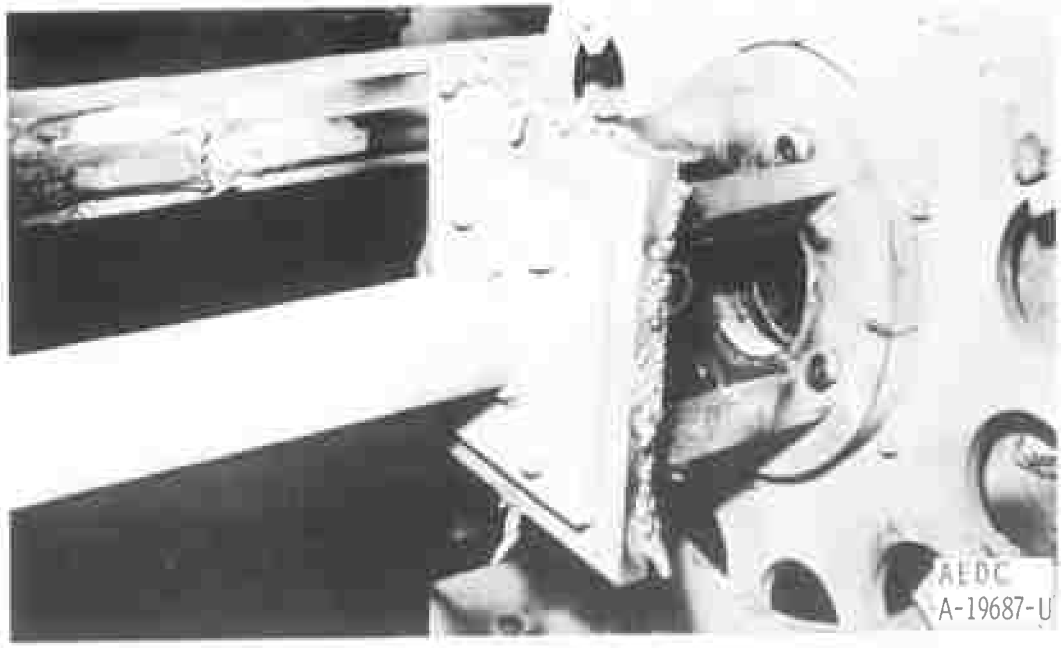


b. Damaged Chamber Pressure Adapter

Fig. 6 Continued



c. Overall Photo of Motor and Cell



d. Front End Detail

Fig. 6 Concluded

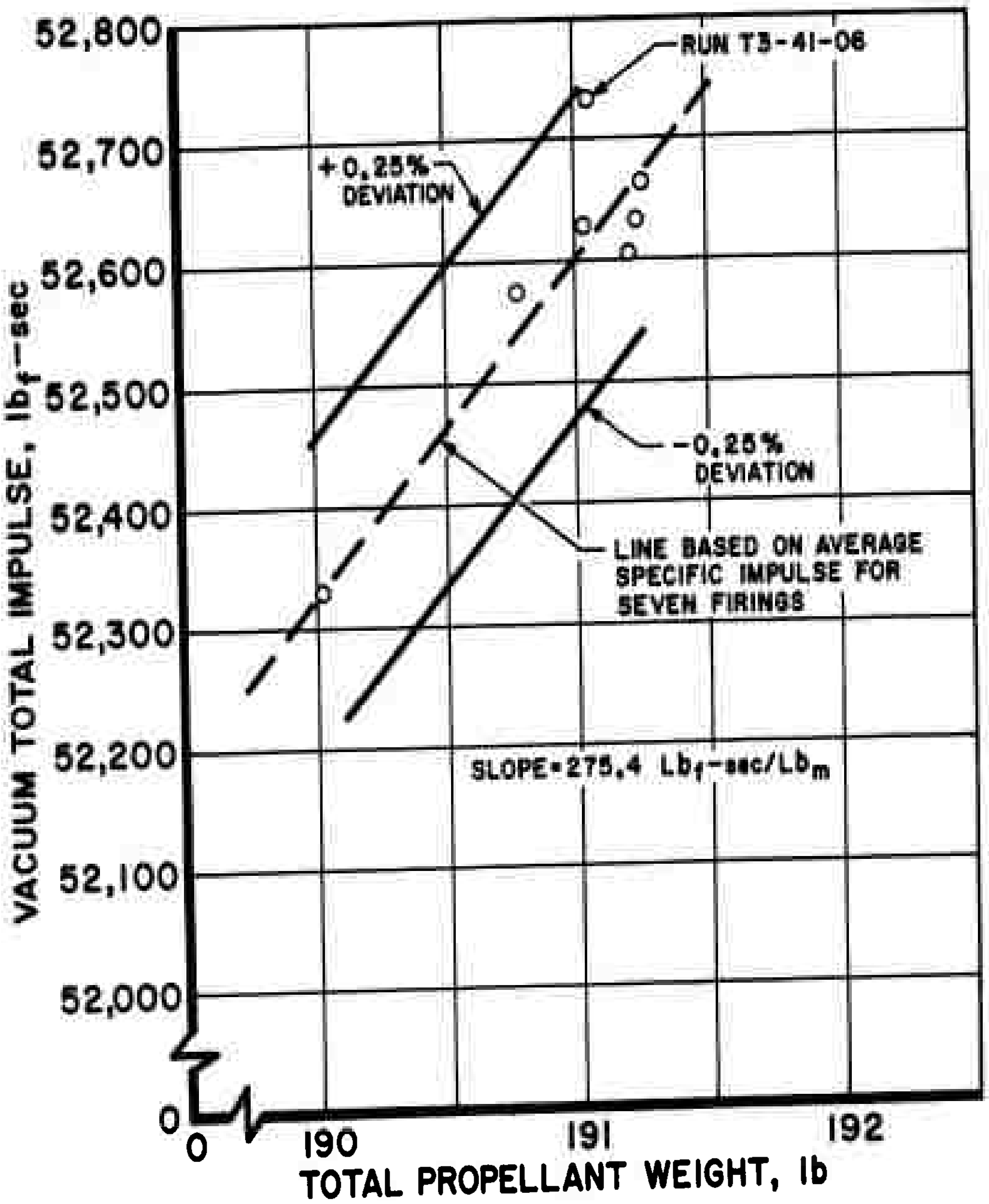
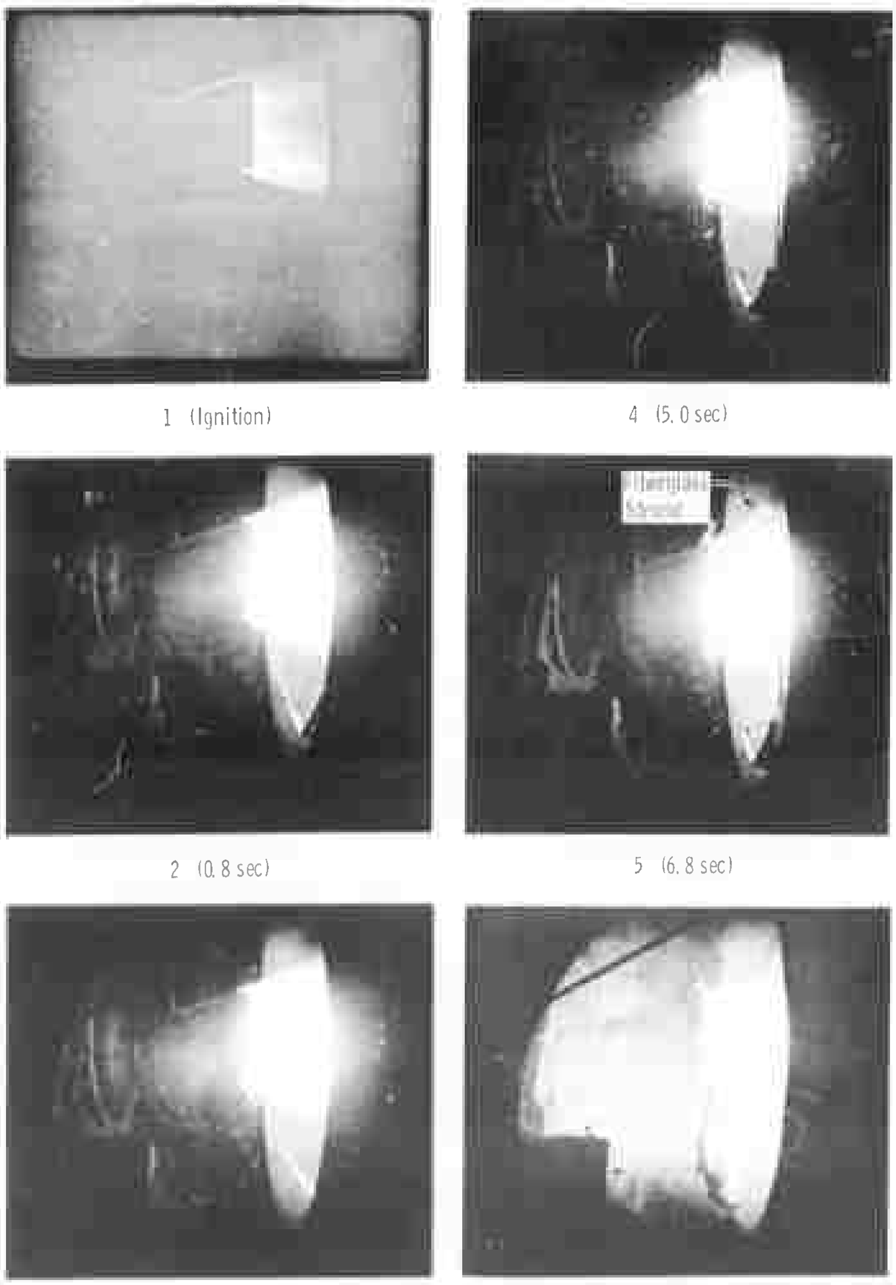


Fig. 7 Vacuum Total Impulse vs Propellant Weight



1 (Ignition)

4 (5.0 sec)

2 (0.8 sec)

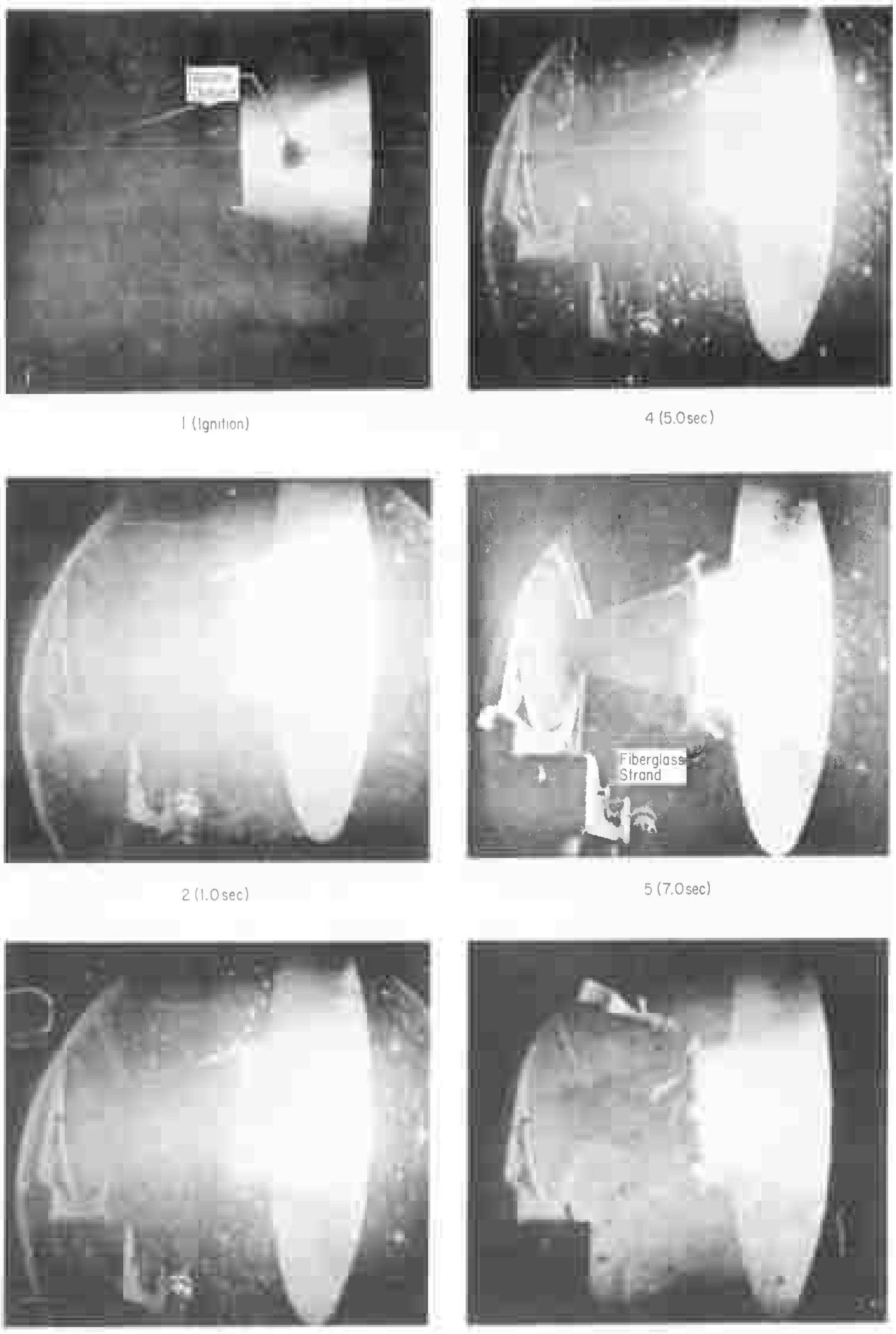
5 (6.8 sec)

3 (2.1 sec)

6 (9.0 sec)

a. Run Number T3-41-04

Fig. 8 Film Sequences of Firings



1 (Ignition)

2 (1.0sec)

3 (2.0sec)

4 (5.0sec)

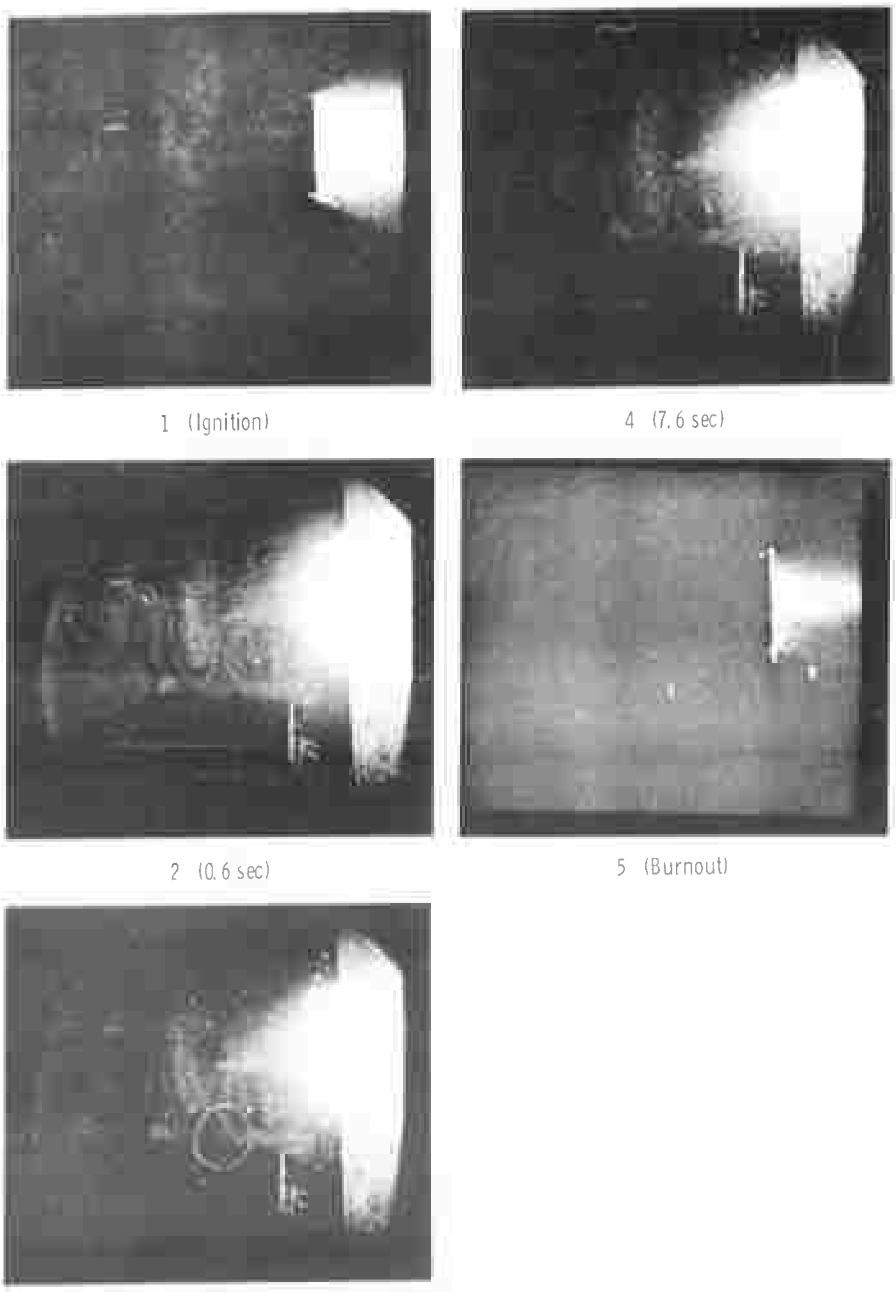
5 (7.0sec)

6 (9.0sec)

These times are approximate

b. Run Number T3-41-07

Fig. 8 Continued



1 (Ignition)

4 (7.6 sec)

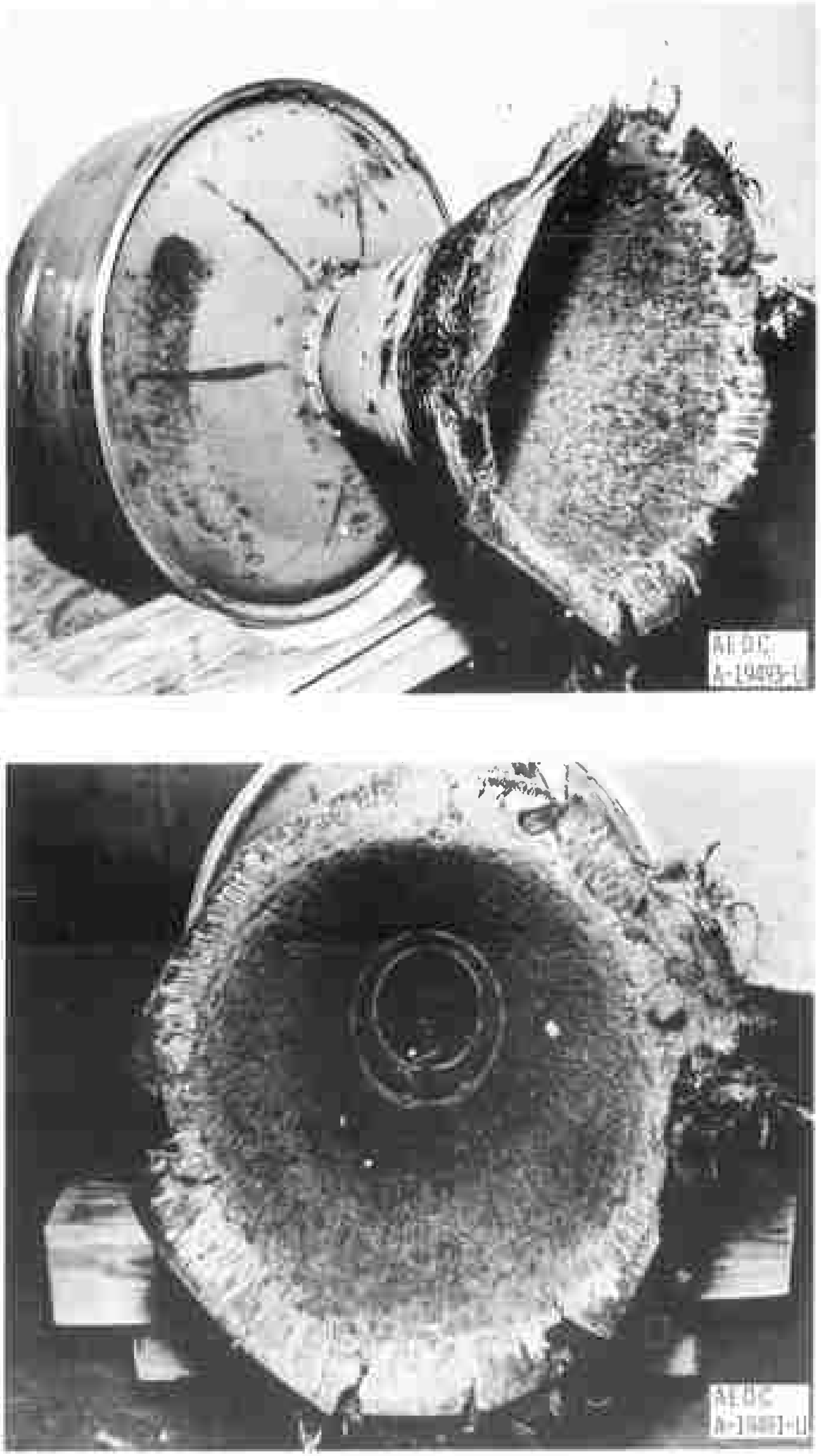
2 (0.6 sec)

5 (Burnout)

3 (3.8 sec)

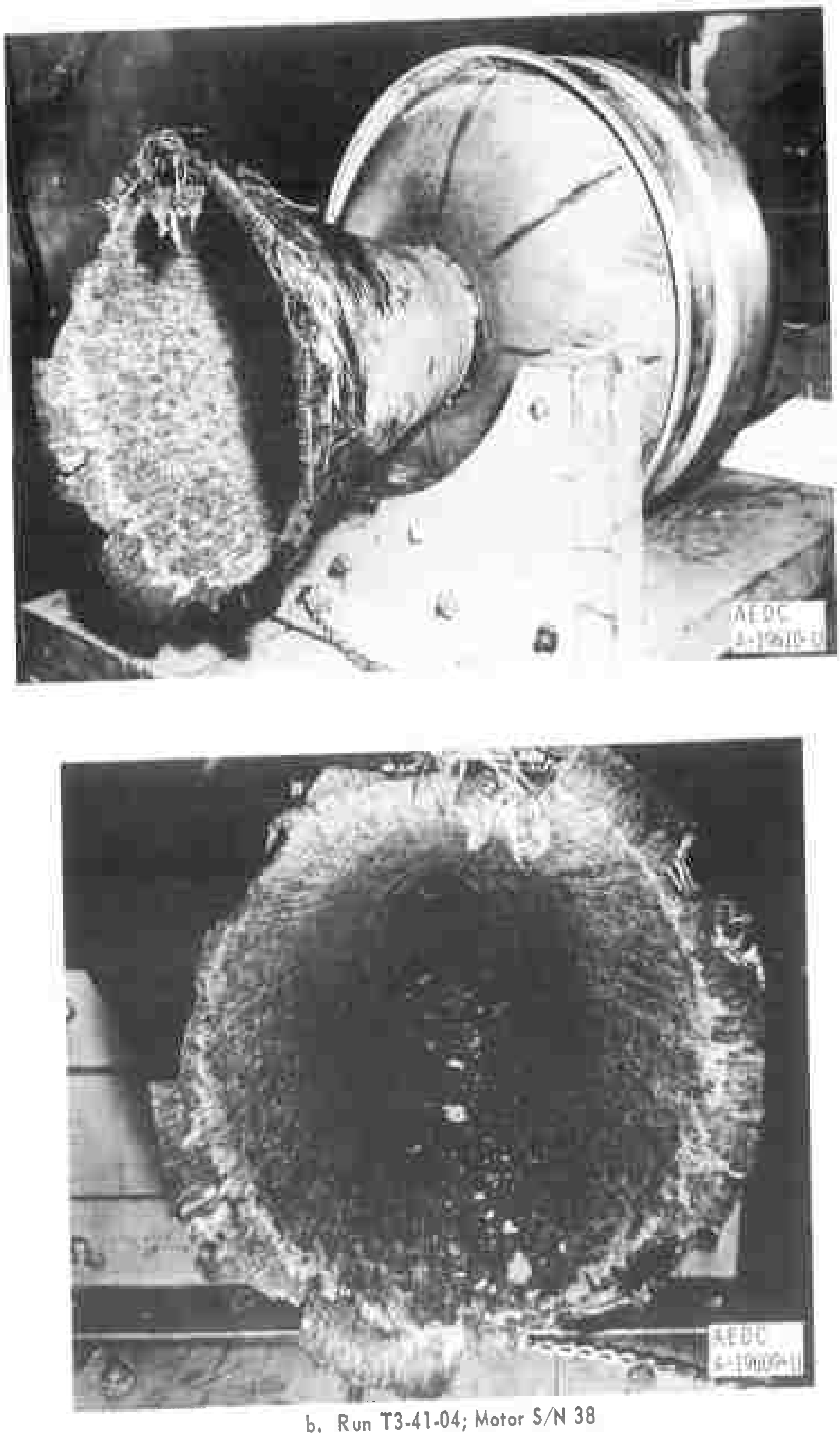
c. Run Number T3-41-09

Fig. 8 Concluded



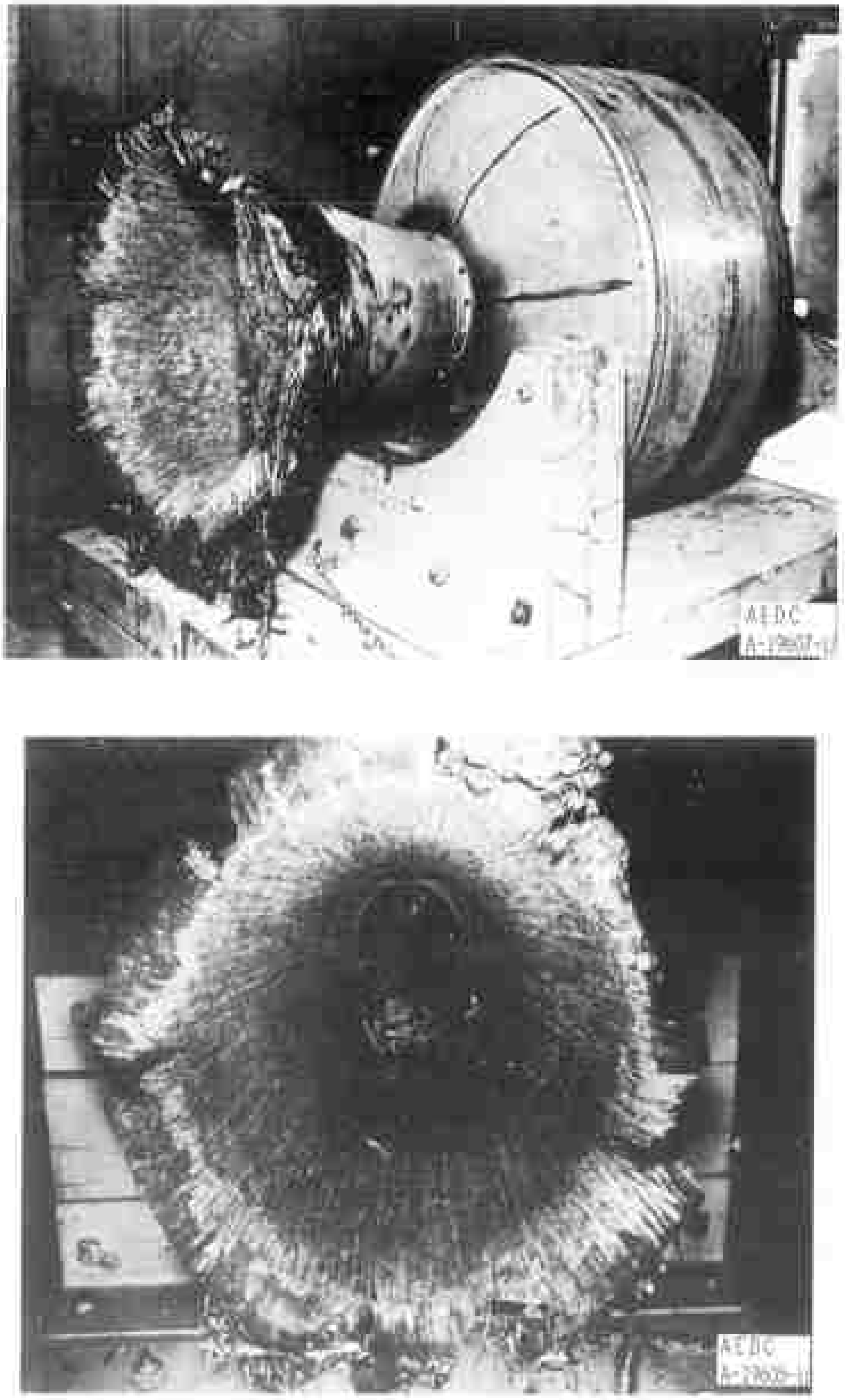
a. Run T3-41-03; Motor S/N 37

Fig. 9 Post-Firing Inspection Photos of Motors



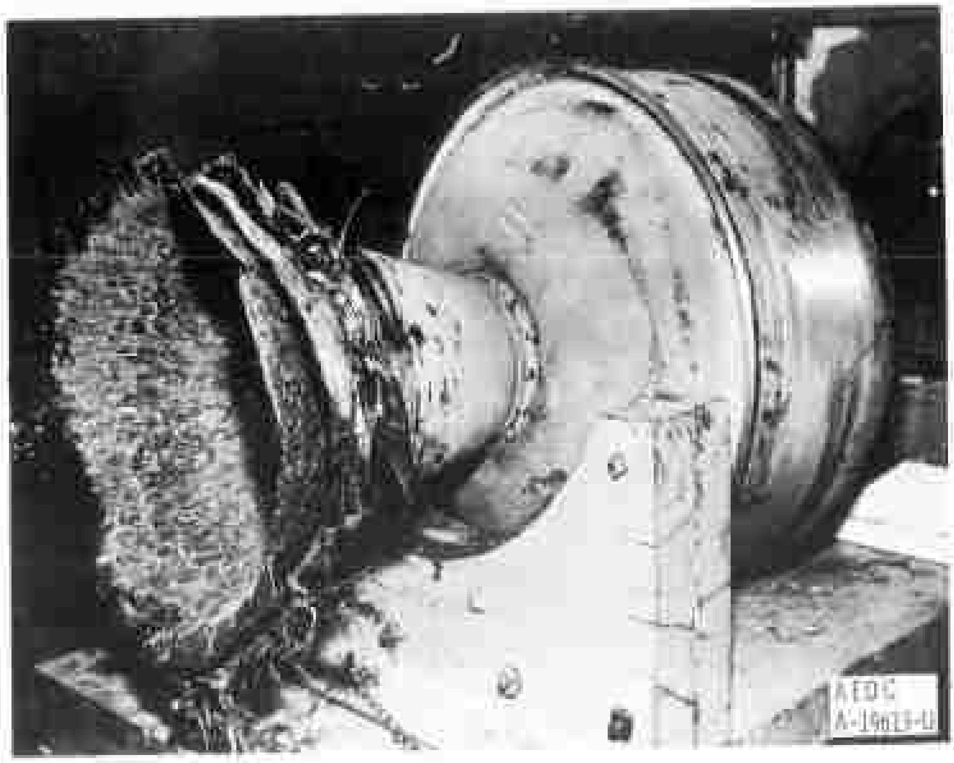
b. Run T3-41-04; Motor S/N 38

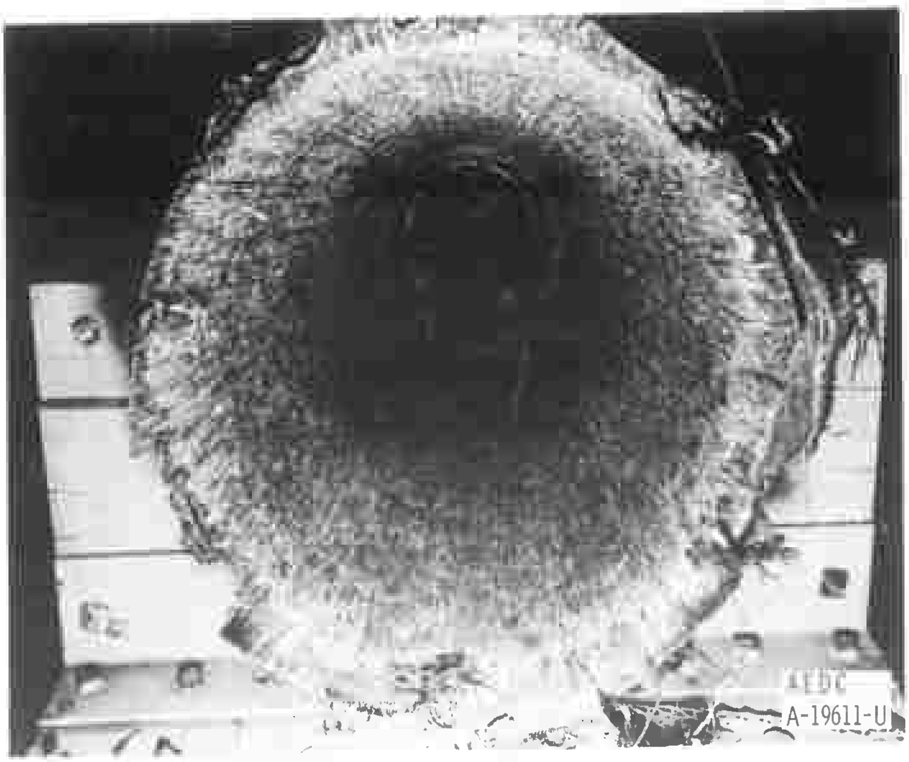
Fig. 9 Continued



c. Run T3-41-05; Motor S/N 39

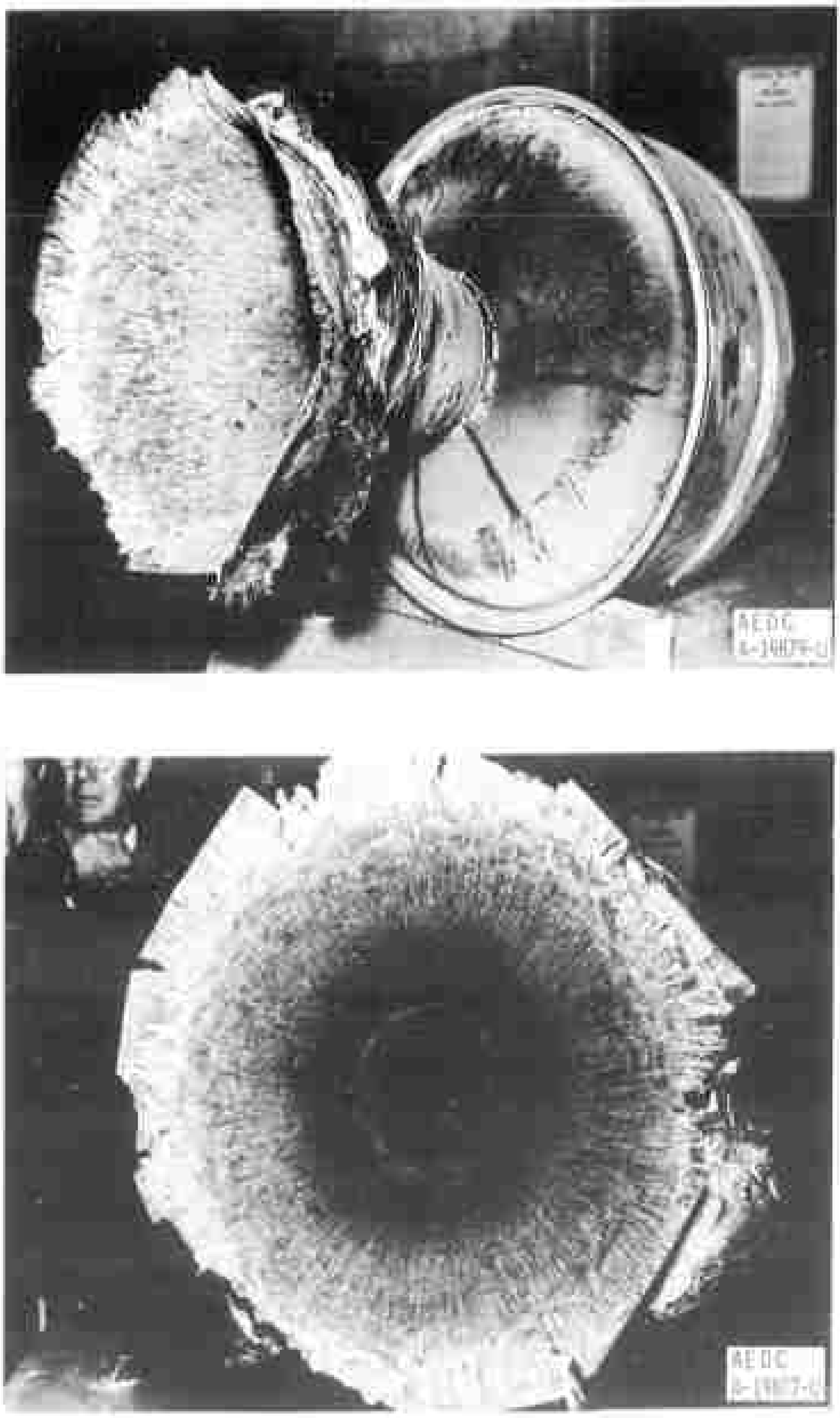
Fig. 9 Continued





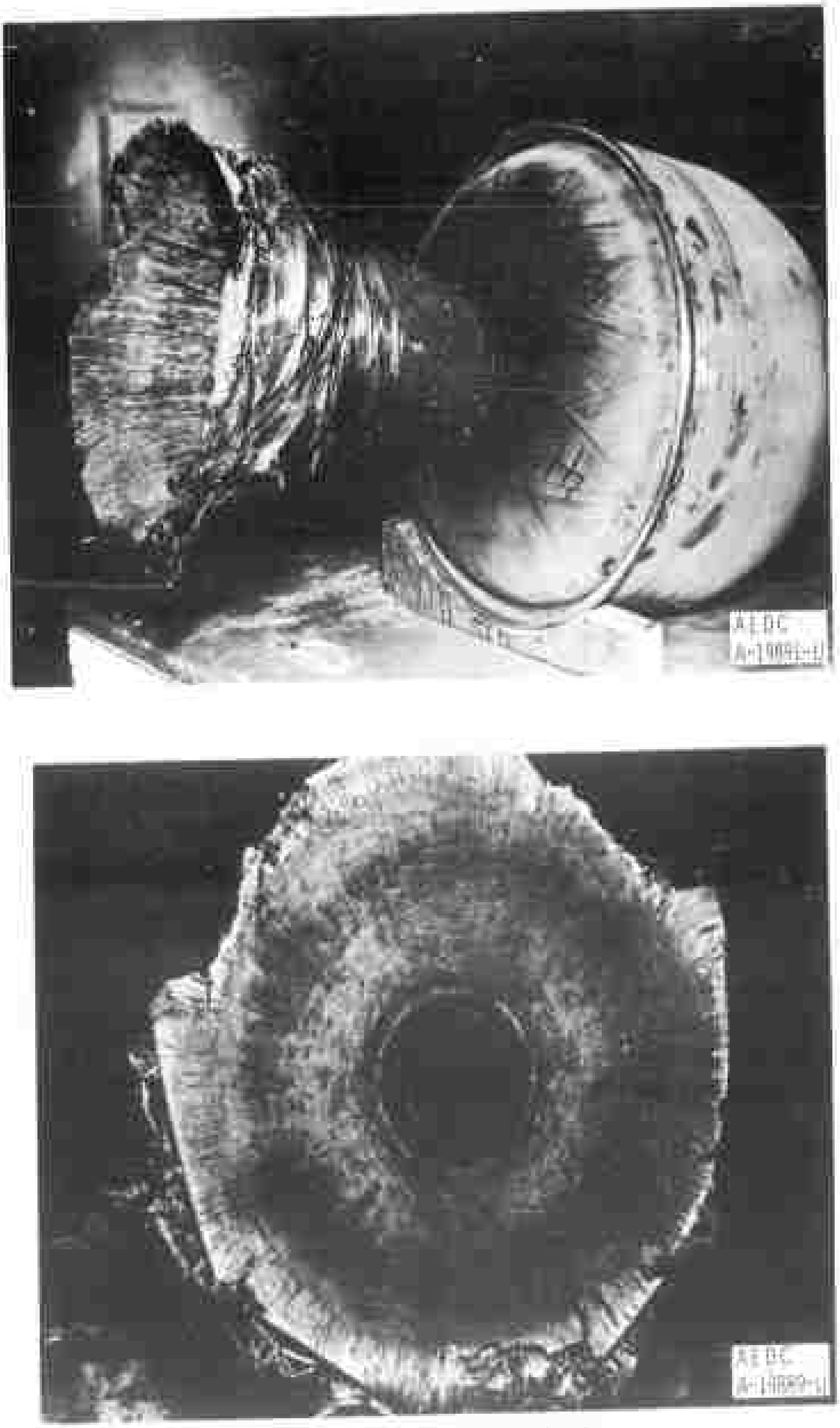
d. Run T3-41-06; Motor S/N 42

Fig. 9 Continued



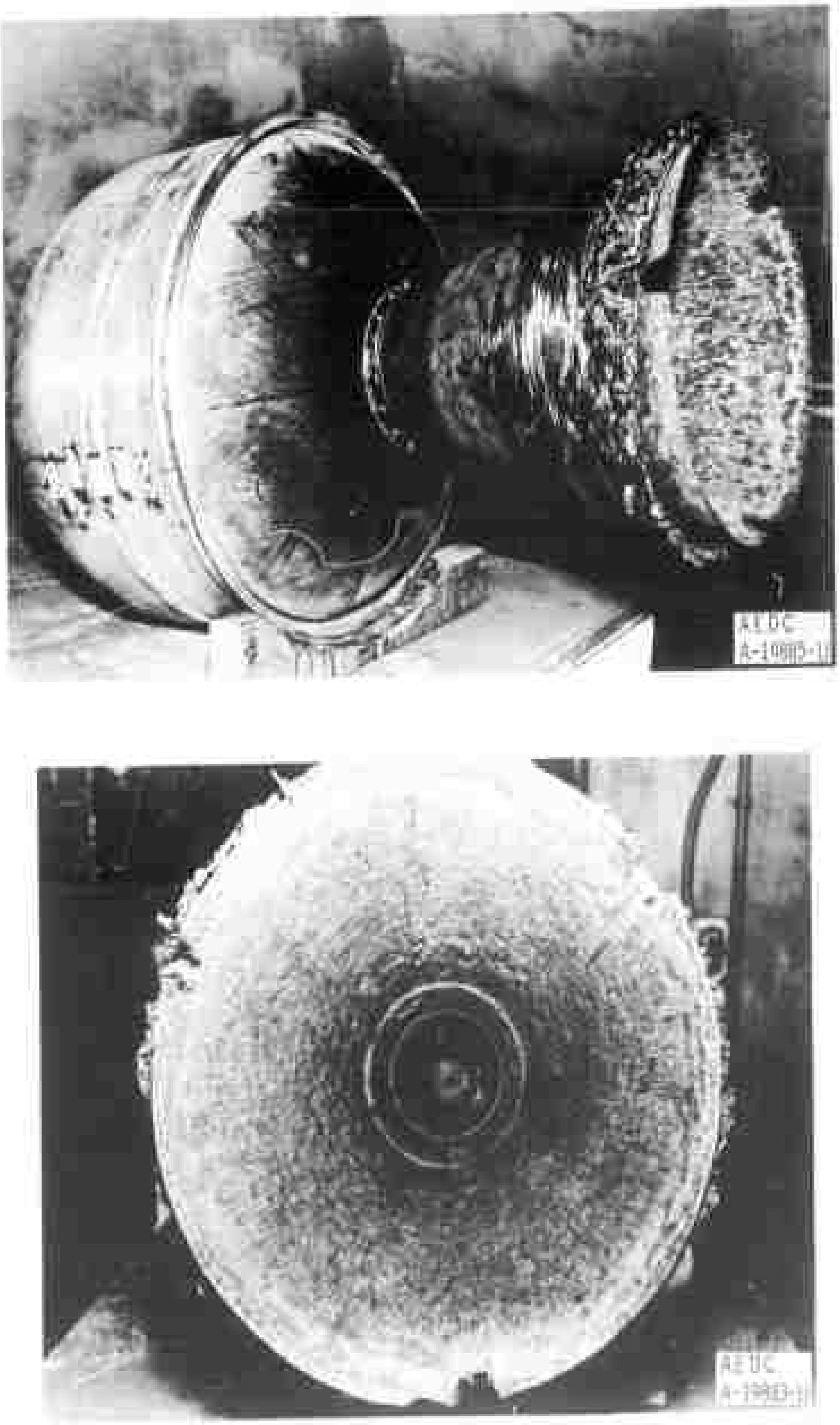
e. Run T3-41-07; Motor S/N 40

Fig. 9 Continued



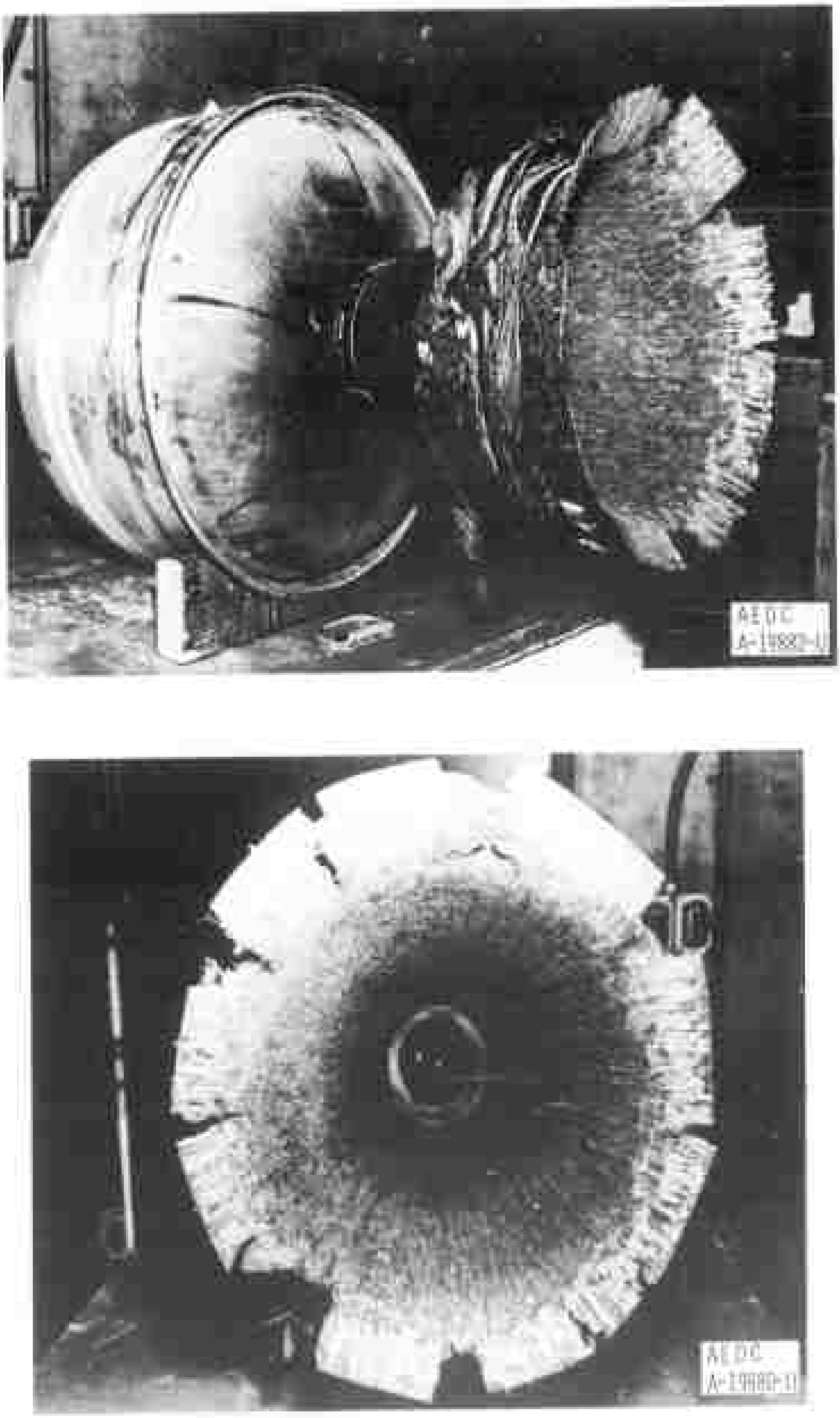
f. Run T3-41-08; Motor S/N 41

Fig. 9 Continued



g. Run T3-41-09; Motor S/N 45

Fig. 9 Continued



h. Run T3-41-10; Motor S/N 43

Fig. 9 Concluded

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3. Rocket motor nozzles
4. Deterioration
5. Rocket propulsion
6. Ignition
7. Specific impulse

I. AFSC Program Area 921E, Project 9042
II. Contract AF 40(600)-800 S/A 24(61-73)
III. ARO, Inc., Arnold AF Sta, Tenn.
IV. Cassell, A. L. and Nokes, C. F., Jr.
V. In ASTIA collection

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Rpt. No. AEDC-TDR-62-63. ALTITUDE TESTING OF HERCULES POWDER COMPANY BE-3 ROCKET MOTORS (Phase II - Qualification and Acceptance Testing) (U).
April 1962, 50 p. incl 2 refs., illus., tables.
Proprietary-Confidential Report

Eight Hercules Powder Company, BE-3, solid-propellant rocket motors designed for application as a retro-rocket for the lunar impact capsule of the Ranger series of spacecraft were tested at pressure altitudes in excess of 100, 000 ft as part of an acceptance and qualification program to determine ignition reliability and motor performance and to evaluate the erosion resistance of the inert components of the motors. Seven motors were fired successfully. Vacuum total impulse was obtained for each motor, and repeatability of performance based on specific impulse was determined. During one firing, the motor forward closure burned through as a result of failure of the igniter support tube. Considerable nozzle deterioration (also encountered during previous tests) was prevalent in the region of the exit plane. A nozzle modification, consisting of an aluminum stiffening ring bonded to the nozzle exit cone, was used on two firings to strengthen the nozzle and was successful for the one firing during which the ring stayed in place.

1. Rocket motors
2. Tests
3. Rocket motor nozzles
4. Deterioration
5. Rocket propulsion
6. Ignition
7. Specific impulse
I. AFSC Program Area 921E, Project 9042
II. Contract AF 40(600)-800 S/A 24(61-73)
III. ARO, Inc., Arnold AF Sta, Tenn.
IV. Cannell, A. L. and Nokes, C. F., Jr.
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