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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

NASA PROGRAM APOLLO WORKING PAPER

LUNAR MODULE REACTION CONTROL SYSTEM ENGINE EXHAUST-PLUME EVALUATION TEST IN THE SPACE-ENVIRONMENT SIMULATION LABORATORY

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LUNAR MODULE REACTION CONTROL SYSTEM ENGINE EXHAUST-PLUME EVALUATION TEST IN THE SPACE ENVIRONMENT SIMULATION LABORATORY

PREPARED BY

Space Environment Test Division

AUTHORIZED FOR DISTRIBUTION

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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By the Space Environment Test Division

INTRODUCTION

The configuration of the lunar module ascent and descent stages (fig. 1) is such that the exhaust plumes from the ascent-stage downfiring reaction control system thrusters impinge on various parts of the spacecraft. Original thermal evaluation and design of the lunar module skins and insulation blankets were based upon a maximum reaction control system engine ontime equivalent to a steady-state burn of 15 seconds during descent from an altitude of 500 feet to the lunar surface. This evaluation was predicated on results of shock tube tests on one-tenth-scale analytical models, and on results of tests on insulated panels which were heated with quartz lamps and subjected to hot carbon dioxide. However, astronaut experience in the lunar module simulator has resulted in a need for design criteria for a reaction control system engine ontime of 23 seconds during the 2-minute period from the 500-foot descent altitude to touchdown on the lunar surface. Therefore, the lunar module reaction control system plume-impingement test program was initiated to study the effects of the increased firing time on the structure of the lunar module. Included in the test evaluation were samples of the lunar module skins and insulation blankets, a lunar module landing window, a lunar module docking window, samples of the command module structure, and an ablative foam panel. The test was also planned to qualify an exhaust-plume deflector design.

The test operations were conducted from April 27 to May 2, 1969, by the Space Environment Test Division. The test was performed in chamber A of the Space Environment Simulation Laboratory (SESL) at the NASA Manned Spacecraft Center.

The objectives of the test program were the following:

1. To determine if the reaction control system plume deflector is required for the LM-5 mission

2. To certify the reaction control system plume deflector according to the lunar module certification test requirements (ref. 1)

3. To develop design data in support of analysis of lunar module thermal insulation

4. To develop test data on plume-impingement effects on the windows of the lunar module and NASA surface finish materials

A more detailed breakdown of the test program objectives is given in the test plan (ref. 2).

This report covers test operations, data analysis, and test results based on real-time data only. A final test report, including a detailed data analysis, will be released by the spacecraft contractor at a later date.

TEST CONFIGURATION

The lunar module reaction control system plume-impingement test was conducted with the following facilities and equipment:

1. Chamber A, Space Environment Test Division

2. General Electric Acceptance Checkout Equipment-Spacecraft (ACE-SC) Station Number 1

3. Grumman Aircraft Engineering Corporation (GAEC) test-article support structure and associated instrumentation

4. Structures and Mechanics Division test articles and associated instrumentation

5. Space Environment Test Division high-speed, frequency-modulated (FM) data acquisition system

6. Propulsion and Power Division reaction control system engine pallet and firing console

Chamber A

Chamber A is a stainless-steel vessel 65 feet in diameter and 120 feet high. The floor of the chamber includes a 45-foot-diameter platform (the lunar plane) which can be rotated 180° in either direction from its center position. The lunar plane can rotate at speeds up to 1-2/3 rpm, and rotation can be controlled automatically or manually. The surface of the lunar plane can be maintained at temperatures between 80° and 400° K.

The chamber incorporates a 40-foot-diameter, hydraulically operated, remote-controlled side-hinged door. The test volume of the chamber is surrounded by black, liquid-nitrogen-cooled heat sink panels.

The chamber vacuum system consists of mechanical and diffusion pumps and a 20° K helium-cooled cryopump. The pumping system can pump the chamber to 3×10^{-6} torr in 18 hours with a gas-leak load of 27.6 torr-liters/sec. Instrumentation penetrations are provided in the chamber walls and the lunar plane.

The chamber systems also include airlock entrances and solar simulators which were not used in this test. Liquid nitrogen panels were placed over the side solar simulator (fig. 2), and a gas-flow diverting baffle was suspended beneath the top solar simulator (fig. 3) in order to divert gases behind the liquid nitrogen panels. A cross section of the chamber, illustrating the test setup, is shown in figure 4.

Test-Article Support Structure

Figure 5 is an illustration of the support structure, reaction control system engine pallet (propulsion assembly), and test articles as installed in chamber A. The frame structure was fabricated from aluminum I-beams which were mounted on the stationary grating around the perimeter of the lunar plane. Strip heaters were mounted on the structure to maintain an even temperature and, thereby, limit thermal distortion which would result in engine-pallet/test-article misalinement. The frame was wrapped with multilayer H-film and aluminized Mylar insulation, which kept the frame thermally isolated from the chamber grating.

Test Articles

Thirteen test articles were exposed to the lunar module reaction control system plume. All but one of the test articles, the lunar module landing window, were mounted on the test-article support structure. The landing window was mounted on a separate support structure. In three cases, two test articles were mounted at the same station in such a manner that they would be exposed simultaneously to the plume.

Plume rakes.- The four plume rakes were arrays of plume-surveying instrumentation (pressure transducers and heat-flux sensors). The rakes were placed at heights of 22, 45, 84, and 165 inches above the reaction-control-system engine exit plane. Figures 6 and 7 show examples of these test articles.

Lunar module docking window.- The lunar module docking window was a flight configuration of only the exterior pane of the multipaned docking window (fig. 8). The docking window was positioned at the same test station as the plume rake that was 45 inches above the engine exit plane.

Quad II simulator module.- An aluminum boilerplate assembly (figs. 9 and 10), representative of the lunar module descent-stage quad II envelope and the lower portion of the ascent stage, was instrumented with thermocouples, pressure transducers, and heat-flux sensors. The assembly was suspended from the support structure to simulate the geometric relationship of the corresponding parts of the lunar module to the lunar module reaction control system engine. Guard heaters were used to thermally isolate the assembly from its support points.

Reaction control system plume deflector.- A flight configuration of the reaction control system plume deflector was tested. The front face is constructed of five 12-inch-wide, seam-welded, 1.25-mil Inconel sheets. The front face is backed by Inconel mesh and nickel foil, with a back face of 0.5-mil Inconel. The deflector was instrumented with thermocouples and suspended from the support structure to simulate the geometric relationship of the corresponding part of the lunar module to the lunar module reaction control system engine. Thermal shielding and heaters were placed near the deflector to simulate the thermal properties of the lunar module and the thermal environment during flight. The deflector is shown in figures 11 and 12.

Uninsulated (heating rate) panel.- The uninsulated panel consisted of a 2-foot-square metal plate instrumented with calorimeters and pressure transducers. The panel was centrally located in the reaction control system plume (fig. 13). The ablative foam was attached to the perimeter of this panel.

Ablative foam.- A 2-inch layer of ablative foam was attached to the perimeter of the uninsulated panel (fig. 13). This specimen was evaluated for possible coating application as a plume-impingement protective coating.

<u>Command module coating test panel</u>.- The command module coating test panel consisted of a metal plate (fig. 14) containing a sample of an ablator and a sample of thermal coating from the command module.

<u>Insulated panel</u>.- The insulated panel consisted of a 2-foot-square panel representative of the lunar-module thermal shielding and lunar module insulation blankets (fig. 15). The panel was instrumented with thermocouples, and thermal heaters were provided for emittance testing.

<u>Lunar module landing window</u>.- The lunar module landing window consisted of a flight-type exterior pane of the lunar module commander's window mounted in a boilerplate in the exterior part of the lunar module crew-compartment forward bulkhead. The pane was instrumented with thermocouples. The landing window was not suspended from the same support structure as the other test articles, but was mounted on a separate structure which rested on the grating surrounding the lunar plane. The landing window is shown in figure 16.

<u>Plume visualization device</u>.- The plume visualization device (fig. 16) consisted of a rod from which several nickel foil tabs were suspended. The device was positioned at the same test station as the landing window so that the foil tabs would be deflected by the plume and give some indication of the size and force of the exhaust plume.

Reaction Control System Engine Pallet

The reaction control system engine pallet (figs. 17 and 18) was a self-contained propulsion system intended to duplicate the exhaust-plume characteristics of a single lunar module reaction control system engine. Included in the pallet configuration was a 100-pound thruster and sufficient quantities of fuel (50 percent hydrazine and 50 percent unsymmetrical dimethylhydrazine), oxidizer (nitrogen tetroxide), and pressurant (helium) to sustain a minimum of 120 seconds of engine-firing without reservicing. A schematic diagram of the pallet propellant and pressurant systems is presented in figure 19. The pallet was designed to be fueled and checked out in an engine test cell prior to installation in chamber A.

The pallet was provided with an internal environmental heating system and an external insulation scheme to allow reliable pallet operation in cryogenic temperature ranges. The internal heating system and the external insulation were designed to maintain the enclosed engine pallet plumbing at 75° ± 15° F at a chamber A environmental temperature of -320° F.

The engine pallet was mounted on the rotating lunar plane and was rotated, as required, to a position beneath each test article. The engine was fired upward and inward toward the center at an angle of 15° from the vertical. This 15° cant angle was provided to limit the effect of the chamber walls on the plume. The mounting configuration is shown in figure 4.

Engine-firing control was accomplished by use of a single firing control console (fig. 20), which was designed for use with the lunar module reaction control system engine pallet. The console was portable and self-contained, and it provided remote control of pallet operations, which included the following:

1. Engine valve safety interlocks
2. Engine arm switch
3. Control valve switches and valve position indicators
4. Gaseous helium regulator control and position indicator

Also contained in the console were three independent 28-V dc power supplies: a console and control-valve power supply; an engine-propellant-valve power supply; and a spare power supply for emergency power requirements. Remote panel meters provided real-time propulsion status information.

Instrumentation System

The instrumentation system consisted of all test-article and engine-pallet transducers, a solenoid control box, an instrumentation-controller assembly, the ACE-SC data system, an FM data acquisition system, a thermal control system, and a television and motion-picture camera system.

Test article sensors.- The GAEC and NASA test-article instrumentation included heat-flux sensors, copper/constantan thermocouples, pressure transducers, strain gages, and accelerometers. The distribution of the instrumentation for the test articles is shown in table I. The pressure transducers used the chamber A pressure as a reference, thus necessitating a means of closing off the reference ports prior to the engine-firing. This was accomplished by means of a solenoid valve on each reference port and by means of a solenoid control box to open and close the solenoid valves as required. A patch panel was fabricated by GAEC to provide a quick patch capability between tests. Firing signals to the console were activated through the ACE-SC up-link system.

Chamber A pressure sensors.- The chamber instrumentation included two types of high-response pressure sensors for measurement of chamber dynamic pressure during engine-firings, three Baratron mechanical diaphragm-type sensors with an operating range of approximately 1 × 10 [illegible] to 1 × 10 [illegible] torr, and four [illegible] ionization gages with an operating range of approximately [illegible] torr. The sensors were located around the engine pallet to measure pressures occurring in the plume boundary. The sensor heads were placed below the exit plane of the plume boundary and were shielded to limit the effects of high velocity gas flow from the engine pallet and the plume boundary. Pressure measurements were recorded during the test by both the high-response FM data acquisition system and the ACE-SC station.

Frequency-modulated data acquisition system.- The elements of the FM data acquisition system (fig. 21) were provided by NASA. The system consisted of differential amplifiers, voltage-controlled oscillators, tape record/reproduce machines, and oscillographs. The data flow for the data acquisition system is shown in figure 22.

Acceptance checkout equipment-spacecraft.- The Space Environment Test Division ACE-SC Station Number 1 (as defined by the block diagram in fig. 23) was used to support the reaction control system thermal-vacuum test. The ACE-SC provided real-time computerized processing and display of data measurements, and precise control signals to the firing control console for use in commanding firings of the engine pallet.

Thermal control system.- The thermal control system (fig. 24) maintained the test-article support structure at a constant temperature of 70° F through the use of strip heaters and insulation wrapping of the support structure. The thermal control system also used quartz lamps (fig. 25) to provide the desired thermal conditions on the plume deflector and insulated panel during firing of the engine pallet on these articles.

Television and motion-picture camera system.- Television and motion-picture cameras were placed inside the chamber to facilitate engine-pallet/test-article alinement and to provide a visual record of the effect of the plume upon certain test articles. Arrangement of these television and motion-picture cameras is shown in figure 26.

TEST OPERATIONS

On April 27, 1969, the reaction control system engine pallet was transported from the Thermochemical Test Area to the SESL with authorized safety procedures in force. The pallet was installed and alined in chamber A in accordance with approved Interface Control Document provisions. Pumpdown was initiated at 0020 hours on April 28, 1969.

The test was conducted in accordance with Operation Checkout Procedures (ref. 3). Pumpdown and cryostabilization was slowed by minor difficulties with the helium cryopumping system and an apparent, unexplained chamber gas load which was determined to be partially helium. The chamber pressure reached 1.0×10^{-6} torr at 29:10 hours elapsed test time (e.t.t.), and the first firing took place at 29:37 hours e.t.t. at station 2 where one of the four rakes tested was positioned. A summary of engine-firings is presented in table II. For plume definition, each of the four rake stations required a 0.5- and 1.0-second burn during the first phase of testing, which lasted until 39:41 hours e.t.t. The passive docking-window specimen was included in the firing on the rake that was positioned 45 inches above the engine exit plane at station 4. There was a hold in chamber A for chamber B manning operation (not related to this test) from 33:50 to 35:20 hours e.t.t.

The second test phase required 0.5- and 1.0-second burns on the quad II simulator module, and was completed at 48:25 hours e.t.t. A 4-hour delay occurred during instrumentation patching for the high-response data system; the normal patching procedure also involved troubleshooting of calibration and dropout instrumentation.

The firing on the quad II simulator module was closely followed by the heating-rate burns; that is, 0.5- and 1.0-second burns on the uninsulated panel. The heating-rate burns were completed by 56:00 hours e.t.t. The helium cryopumping panels were then relieved of accumulated gas loads prior to the first normal-duty-cycle profile burn (36 seconds of engine ontime) on the insulated panel. A tabulation of the reaction-control-system normal duty cycle is presented in reference 2. This duty-cycle firing was accomplished at 61:28 hours e.t.t. with a maximum chamber pressure of 100 microns (1.0×10^{-1} torr), well within the engine ZOT limit of 500 microns (5.0×10^{-1} torr). (The ZOT limit indicates probability of engine damage caused by propellant buildup in the injector if the engine is reignited at a pressure above 500 microns.)

During the 6.5-hour recovery period before the duty-cycle firing (certification test), on the plume deflector, difficulties with engine parameter dropouts were resolved, and engine-heater power loss was corrected. The firing profile was initiated on the plume deflector at 67:07 hours e.t.t. and was completed without apparent damage to the test article. Maximum chamber pressure was indicated on the Baratron sensors to be 100 microns — again, well within the ZOT limit.

The remaining test firings, for secondary objectives, were concluded without difficulty. The firings were on the following test articles:

1. Command module coating test panel: A 6.0-second burn occurred at 75:13 hours e.t.t.

2. Plume visualization device: A 4.0-second burn occurred at 77:03 hours e.t.t.

3. Lunar module landing window: A 10.0-second ontime burn (consisting of four hundred 25-msec pulses, 2 seconds apart) occurred, starting at 78:20 hours e.t.t.

4. Ablative foam: The polyurethane ablative foam specimen attached to the uninsulated panel was subjected to the last 10 minutes 20 seconds of the normal duty cycle (including 13 seconds of engine ontime) at 80:14 hours e.t.t.

Chamber sublimation techniques were used to remove water vapor and engine exhaust products from the chamber. Repressurization was completed at 96:20 hours e.t.t. The reaction control system engine pallet, after post-test calibration, was removed from chamber A and transported to the Thermochemical Test Area, and the test operations ended at 0545 hours, May 2.

TEST RESULTS

Chamber A Performance Evaluation

All chamber A test objectives were achieved, and operations were nominal except for problems with the cooldown and stabilization of the gaseous helium system.

Chamber A pressure performance.- There were three primary test requirements concerning the vacuum pumping system in chamber A:

1. To provide a prefiring chamber pressure of 1×10^{-5} torr or less, with liquid nitrogen panels and cooled, stable gaseous helium panels

2. To maintain the chamber pressure at less than 1×10^{-3} torr during the first 0.1 second of each firing of the reaction-control-system engine pallet

3. To maintain the chamber pressure below the 5×10^{-1} torr abort limit during the 36-second engine ontime of the normal duty cycle

The first requirement was accomplished by maintaining the chamber pressure prior to each firing in a range of 7×10^{-7} to 1.5×10^{-6} torr. The liquid nitrogen panels were stable, and there was no evidence of temperature rise during the engine-firings. However, there was difficulty in establishing helium cryopumping temperatures. During the cooldown phase, it became necessary to reduce the chamber heat load and the amount of cryopumping surface in order to obtain stable conditions. During the 0.5- and 1.0-second firings, helium panel temperatures increased rapidly. However, these surfaces continued to cryopump throughout the recovery period. During the 6.0-second and normal-duty-cycle burns, helium panel temperatures increased rapidly until these surfaces were no longer effectively cryopumping.

The second requirement was accomplished as indicated by the plots in figures 27 to 40, which show that the chamber pressure was below 1×10^{-3} torr for the first 300 to 400 msec of each firing. These measurements indicate that the engine exhaust plume was sufficiently expanded to obtain valid pressure, temperature, and heating-rate values on the test articles.

During the 0.5-second firings, the increase in chamber pressure varied from 1×10^{-2} to 8×10^{-1} torr. During the 1.0-second firings, the increase in chamber pressure varied from 8×10^{-4} to 8×10^{-2} torr. In the initial stages of the normal duty cycle, the chamber pressure rapidly increased to approximately 2×10^{-2} torr, reaching a maximum value of approximately 7×10^{-2} torr in the final stages of the simulated lunar landing.

Chamber pressure rapidly recovered after each of the short-duration firings; much more rapidly than had been theoretically predicted. In addition, the rise in chamber pressure was not as great as had been previously calculated. The rapid recovery rate and the limited pressure rise could be a result of a lower amount of noncondensables (primarily hydrogen and helium) present in the exhaust products from the fuel used during this test than in the reported values for previous burns of Aerozine-50 and nitrogen tetroxide.

Maintenance of the chamber pressure below the 5×10^{-1}-torr abort limit during the lunar-landing duty cycles fulfilled the third chamber pressure requirement. The maximum pressure was 1×10^{-1} torr as indicated by the Baratron sensors.

Lunar plane operation.- Analysis of test data indicates that the lunar plane was accurately positioned in the required location for each phase of the test. There is some evidence, however, that the plume was off center of the uninsulated panel at station 6. Attempts to measure movement of the test specimens in relation to the rocket nozzle were limited by the restricted view from rigid external chamber points. However, the measurements which were obtained and the use of visual devices indicated an allowable amount of movement resulting from the transition from ambient to thermal-vacuum conditions.

Test-Article Evaluation

The results presented in this report are based on real-time test monitoring and available, uncorrected oscillograph data. Only real-time oscillograph data for three of the four plume rakes, and only approximately 20 percent of the quad II simulator module data are available for this report.

Heat-flux data for rake 4 positioned at station 2 fall within a ±15-percent range around the GAEC shock tunnel data. Heat-flux and pressure data for the remaining three rakes contain more scatter and must be analyzed. One interesting fact should be noted: heat-flux data for the core of the plume indicate a dip at the center, rather than peak values which were obtained from shock-tunnel data. However, since the center of the plume does not impinge the lunar module, the data concerning the portion of the plume outside the plume core are of most interest from a thermal design viewpoint.

Three problem areas encountered during the test prevented presentation of quick-look data on the uninsulated panel, the insulated panel, and the plume deflector:

1. A calibration error was found which caused the real-time oscillograph data to be invalid as far as absolute values are concerned. However, the data were sufficient to determine that valid data were recorded by the FM data acquisition system.

2. The malfunction of a thermopile in the insulated panel affected the accuracy of the effective emittance test. Post-test data analysis will determine the magnitude of the effect of this malfunction.

3. The improper switching of instrumentation channels for real-time oscillographs prior to the plume deflector test resulted in the loss of real-time data on the plume deflector.

The contamination test articles (the ablator and thermal-coating samples on the command module coating test panel, the docking window, and the landing window) require post-test analysis and will be reported in separate documentation.

The following observations were recorded from the post-test inspection of the test articles:

1. Plume rakes: All plume rakes appeared to be in good condition with no evidence of damage.

2. Docking window: Contamination and optical degradation of the docking window were evident, as shown in figure 41. Figure 8 is a pretest photograph of the window.

3. Quad II simulator module: No anomalous conditions were apparent.

4. Plume deflector: Separations of 2 inches and 0.25 inch were noted along the first and third seams, respectively, from the narrow end of the plume deflector (fig. 42). Plume deposits, or heating effects, or both, were evident on the front face. The H-film insulation on the support structure approximately 10 feet above the plume deflector was torn away from the frame by the force of the exhaust plume (fig. 43).

5. Uninsulated panel: Discoloration existed approximately 6 inches to the right of the panel center line. It is suspected that the discoloration was a result of plume deposits, while the offcenter location was a result of panel/engine pallet misalinement. Misalinement of this magnitude should be reflected in the heat-flux and pressure data profiles on the panel.

6. Ablative foam: The ablative foam was charred slightly on the front face. A long strip of ablative foam was missing from the center of one side of the uninsulated panel (fig. 44).

7. Command module coating test panel: No anomalous conditions were noted.

8. Insulated panel: The toy tab seam on the front-face Inconel sheet was gapped open approximately 0.25 inch, and discoloration of the front face was observed (fig. 45). One area of discoloration around the seam appears to be in the same location as the discolored area noted on the uninsulated panel.

9. Landing window: Contamination and optical degradation were evident on both glass surfaces of the lunar module landing window (fig. 46). Figure 17 is a pretest photograph of the window.

10. Plume visualization device: Many of the nickel foil tabs were bent and discolored by heat and pressure from the exhaust plume.

Reaction Control System Engine-Pallet Performance

The performance of the lunar module reaction control system engine during all firings was nominal and compared favorably with the pretest firings conducted in the Thermochemical Test Area. Satisfactory control of the internal thermal environment system was maintained throughout the test.

No flow-measurement capability was provided for the pallet; therefore, it was not possible to calculate oxidizer/fuel (O/F) ratios for each test. However, the engine performance indicated that the engine was operating at a nominal O/F ratio. The engine-chamber pressures were corrected by post-test, in-place calibration and are accurate to within ±1.0 percent of tabulated values. The engine-pallet performance and firing profile are summarized in table II.

Instrumentation System Performance

Chamber A pressure sensors.- The millitorr gages functioned properly with minor exceptions. Data evaluation revealed that these gages indicated an initial drop in chamber pressure during the 0.5- and 1.0-second firings and an intermediate pressure drop during the 1.0-second firing pulses (figs. 30 and 32).

Two of the Baratron sensors did not function properly because of a zero shift. The remaining gages, because of their limited operating range, were useful only during the final stages of the lunar-landing (normal) duty-cycle burns.

Frequency-modulated high-response data acquisition system.- Functionally, the FM data acquisition system performed well with few component failures during the test series.

Acceptance checkout equipment-spacecraft.- The ACE-SC system supported the reaction control system thermal-vacuum test with no significant discrepancies. The only notable discrepancy was a possible failure of a portion of the analog-to-digital conversion equipment during the plume deflector test. This possible failure may have resulted in the acquisition of a few questionable data measurements during the test.

Motion-picture camera system.- Post-test examination of the motion-picture camera system revealed that camera number 1 worked properly and gave excellent films. The other three high-speed (400 frame/sec) cameras failed. Two environmental housings overheated and destroyed both the film and the cameras. The third high-speed camera jammed.

Of the three low-speed cameras (24 frames/sec), two worked properly. Camera number 5 on the plume deflector apparently had faulty film. Camera number 6 on the landing window performed satisfactorily; however, the information may be of little value because of contamination on the back side of the landing window.

An unsuccessful effort was made to photograph the plume by using a Scotchlite panel (fig. 47) to reflect any amount of light emitted during firing. High-speed camera number 7, located in manlock MA-1, was focused on the Scotchlite panel during the firings on the insulated and uninsulated panels.

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1. Grumman Aircraft Engineering Corporation: LM Certification Test Requirements, Plume Deflector. LCQ-280-36, Revision A, Mar. 28, 1969.

2. Grumman Aircraft Engineering Corporation: Test Plan, LM RCS Plume Impingement Test. LTP-257-2, Dec. 2, 1968.

3. Grumman Aircraft Engineering Corporation: Operational Checkout Procedures, LM RCS Plume Impingement Test. OCP-B-90009-TM-9, Apr. 22, 1969.

TABLE I.- QUANTITY OF INSTRUMENTATION ON TEST ARTICLES

Test article	High-response data acquisition system					Slow-response ACE-SC	Total
	Heat-flux sensors	Copper/constantan thermocouple	Pressure transducers	Strain gages	Accelerometer	Copper/constantan thermocouple	
Plume rakes and docking window	70	70	35	0	0	13	188
Quad II simulator module	96	96	33	0	0	26	251
Uninsulated panel	10	10	5	0	0	4	29
Insulated panel	1	18	6	0	0	8	33
Plume deflector	0	24	0	24	1	39	88
Command-module-coating panel	2	0	0	0	0	8	10
Landing window	0	0	0	0	0	4	4
Support structure[a]	0	0	0	0	0	4	4
All articles	—	—	—	—	—	—	607

[a]Thermocouples were placed on the support structure near the plume rakes positioned 22, 45, and 84 inches above the RCS engine exit plane and near the plume deflector.

TABLE II.- LUNAR MODULE REACTION CONTROL SYSTEM

ENGINE-PALLET PERFORMANCE SUMMARY

Date	Elapsed test time, hr:min	Station	Test article	Engine ontime, sec (a)	Steady-state engine-chamber pressure, psia
4-29-69	29:37	2	Plume rake[b]	0.5	97
	31:08	2	Plume rake[b]	1.0	97
	32:32	3	Plume rake[c]	.5	97
	33:19	3	Plume rake[c]	1.0	97
	35:43	4	Plume rake[d]	.5	97
	37:00	4	Plume rake[d]	1.0	97
	38:15	8	Plume rake[e]	.5	96
	39:47	8	Plume rake[e]	1.0	97
	47:33	5	Quad II simulator module	.5	99
4-30-69	48:25	5	Quad II simulator module	1.0	97
	50:31	6	Uninsulated panel	.5	97
	56:00	6	Uninsulated panel	1.0	97
	61:28	7	Insulated panel	[f]36.0	101
	67:07	1	Plume deflector	[f]36.0	102
5-1-69	75:13	9	Command-module-coating panel	6.0	103
	77:03	10	Plume visualization device	4.0	101
	78:20	10	Landing window	[g]10.0	--
	80:14	6	Ablative foam	[h]13.0	102

[a]Total engine ontime = 114.0 seconds.

[b]Positioned 165 inches above the engine exit plane.

[c]Positioned 84 inches above the engine exit plane.

[d]Positioned 45 inches above the engine exit plane.

[e]Positioned 22 inches above the engine exit plane.

[f]Duty cycle consisting of multiple pulses (ref. 2).

[g]400 pulses; 0.025 second/2.0 seconds off.

[h]The last 10 minutes 20 seconds of the normal duty cycle.

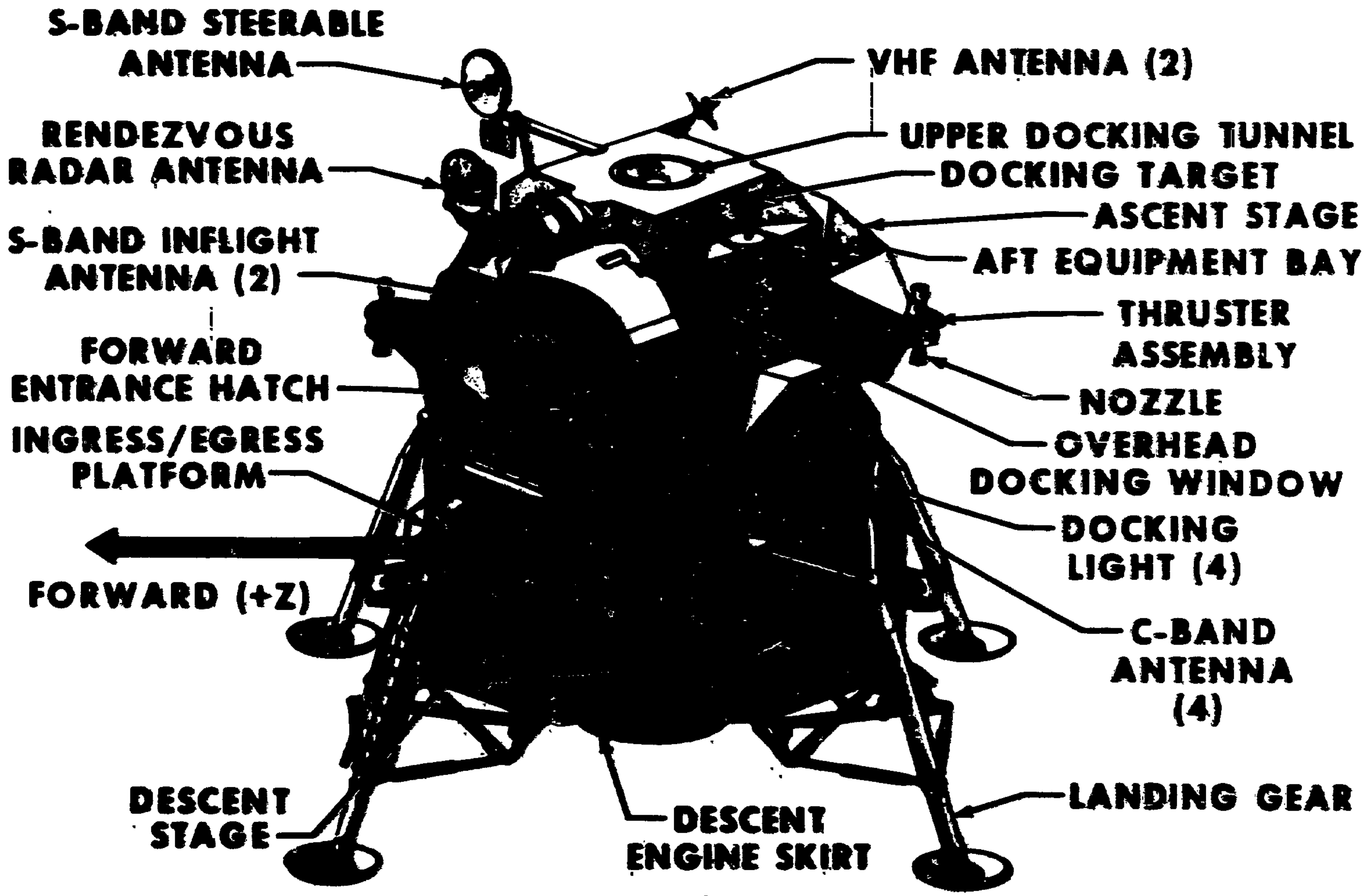

Figure 1.- Lunar module.

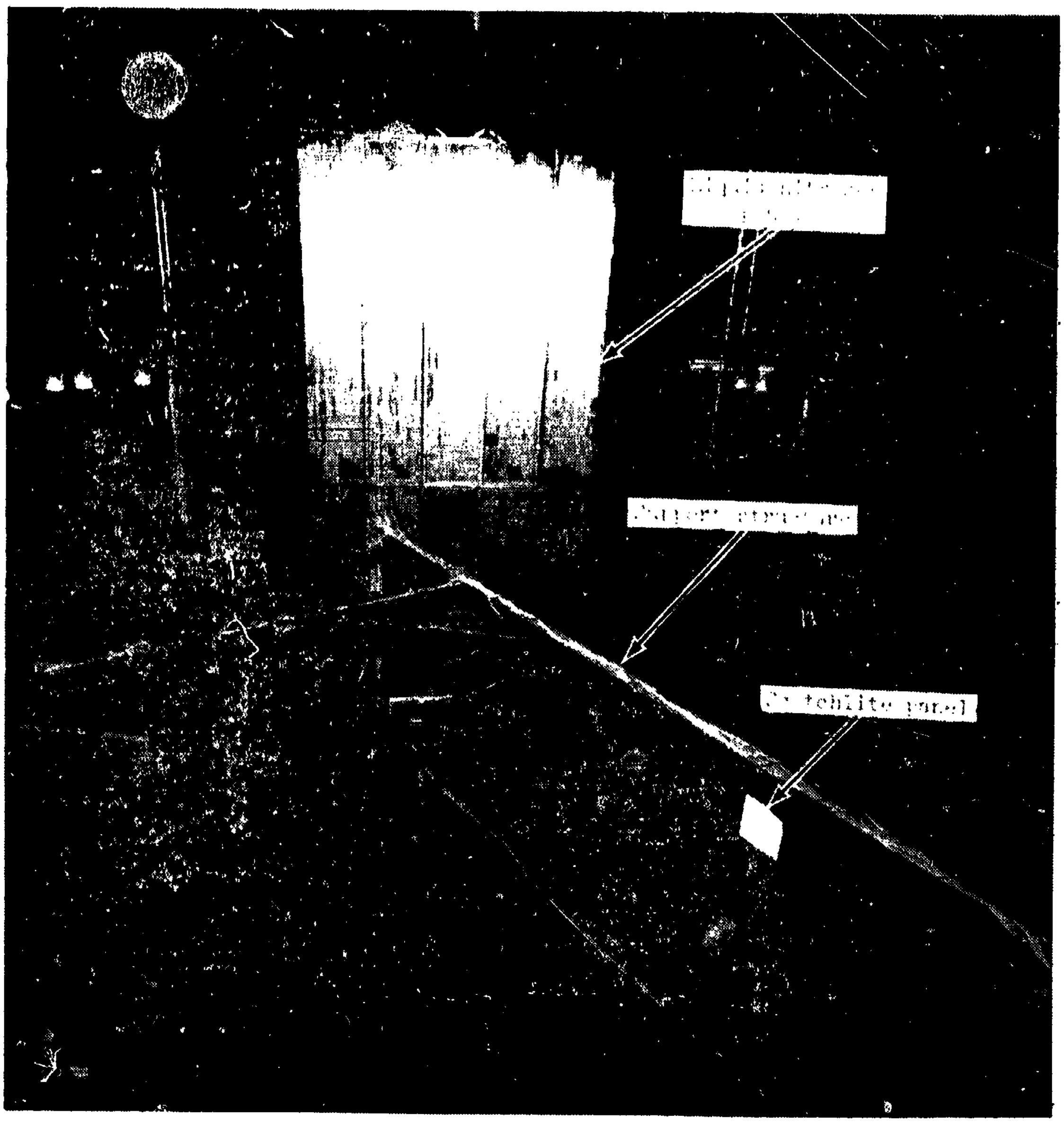

Figure 2.- Liquid nitrogen panels placed over the side solar simulator.

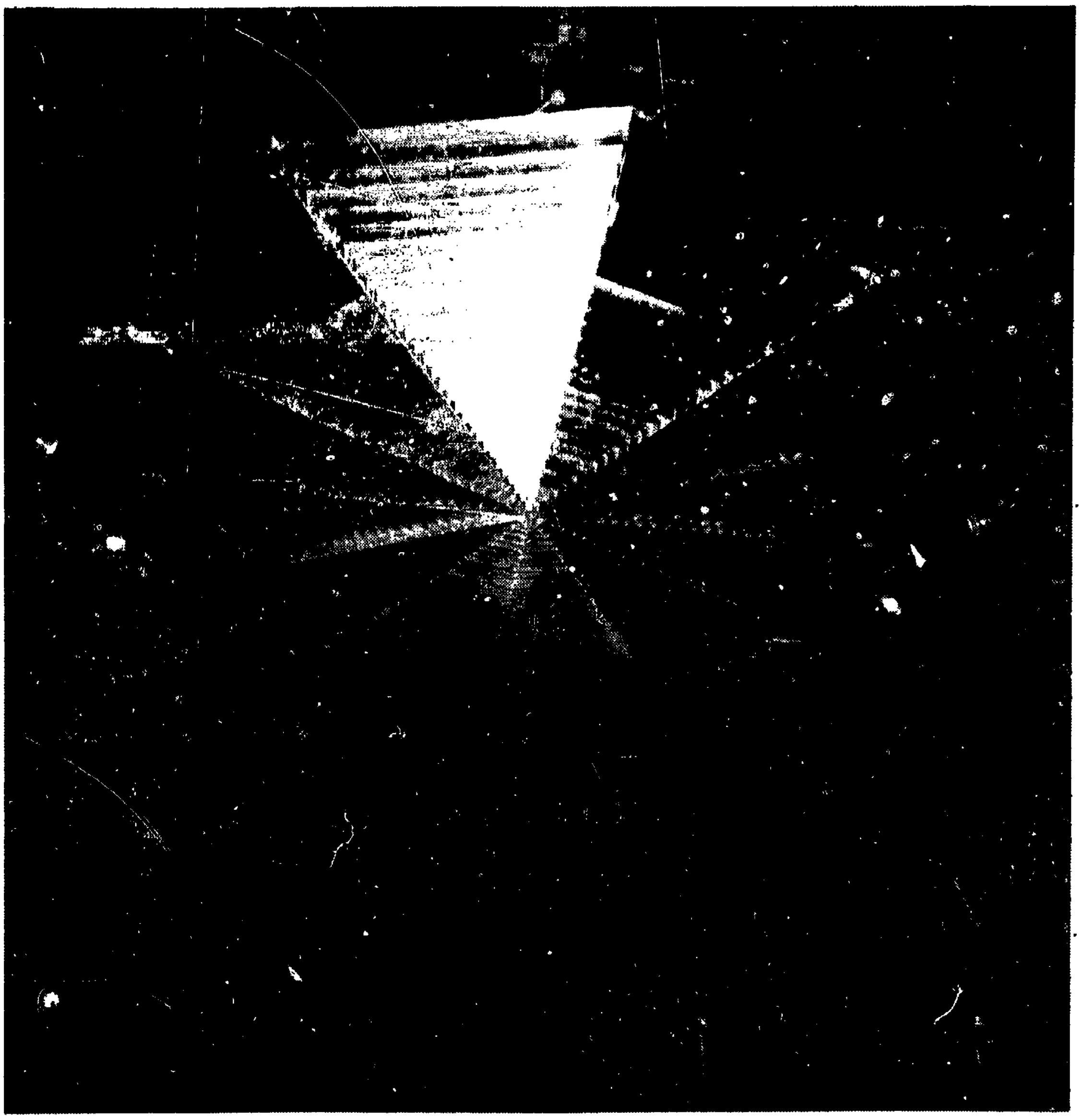

Figure 3.- Gas-flow diverting baffle suspended under the top solar simulator.

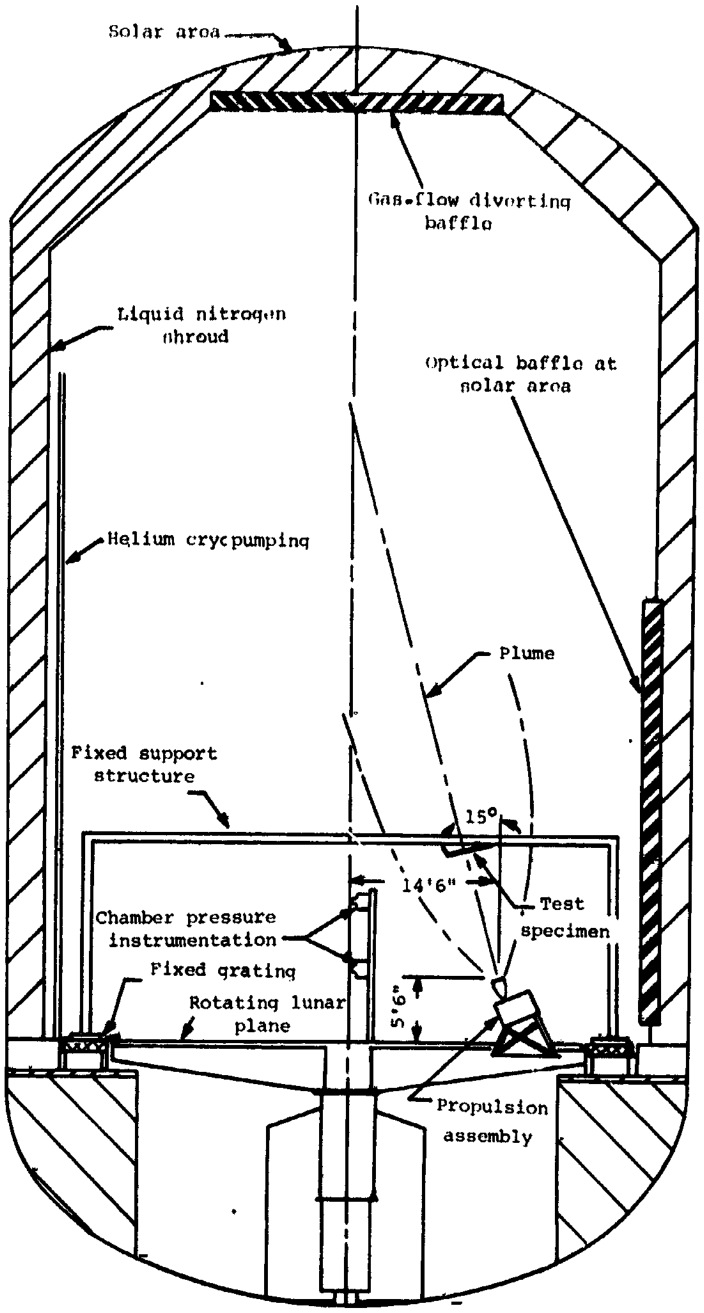

Figure 4.- Cross section, chamber A.

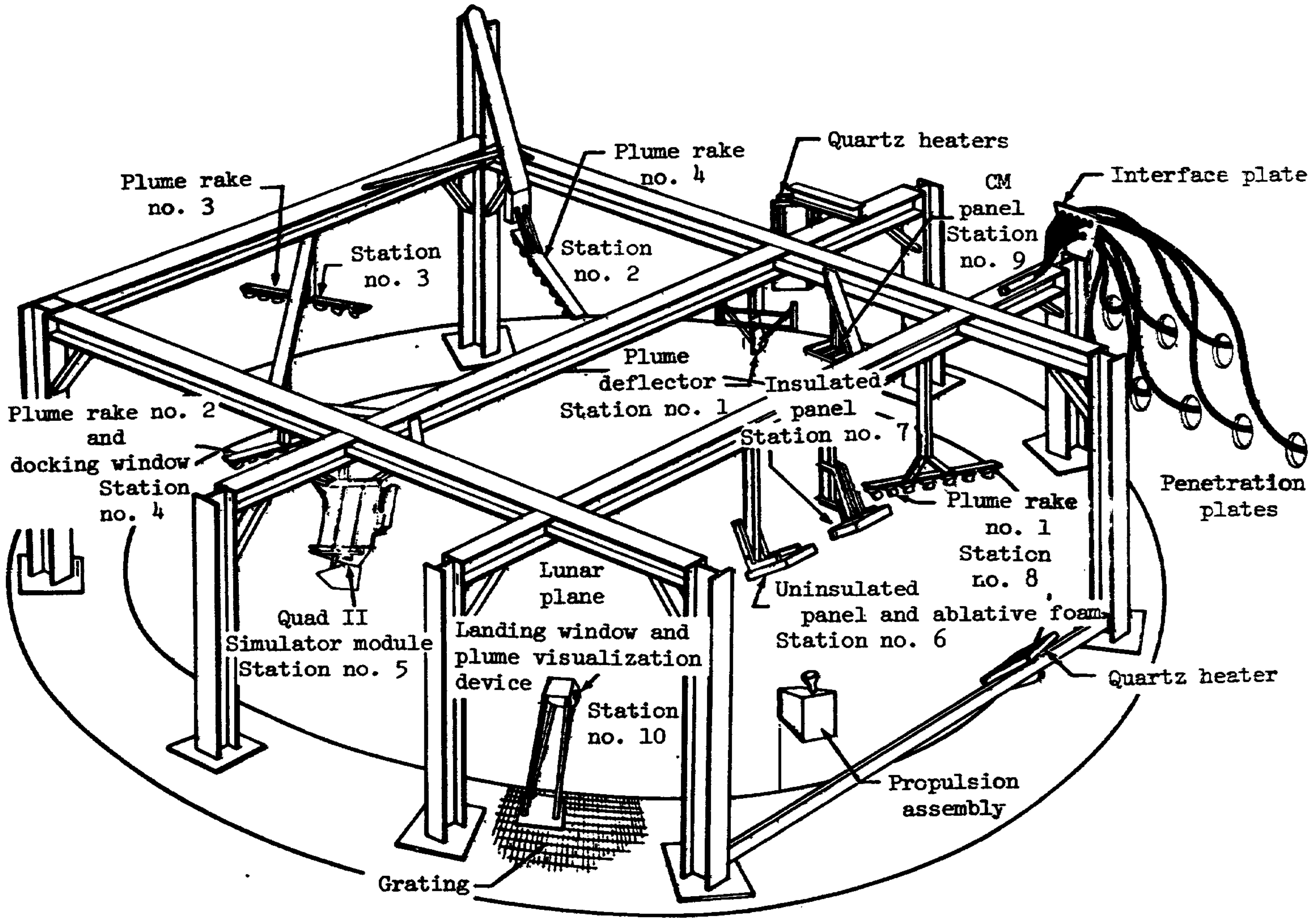

Figure 5.- Test-article support structure.

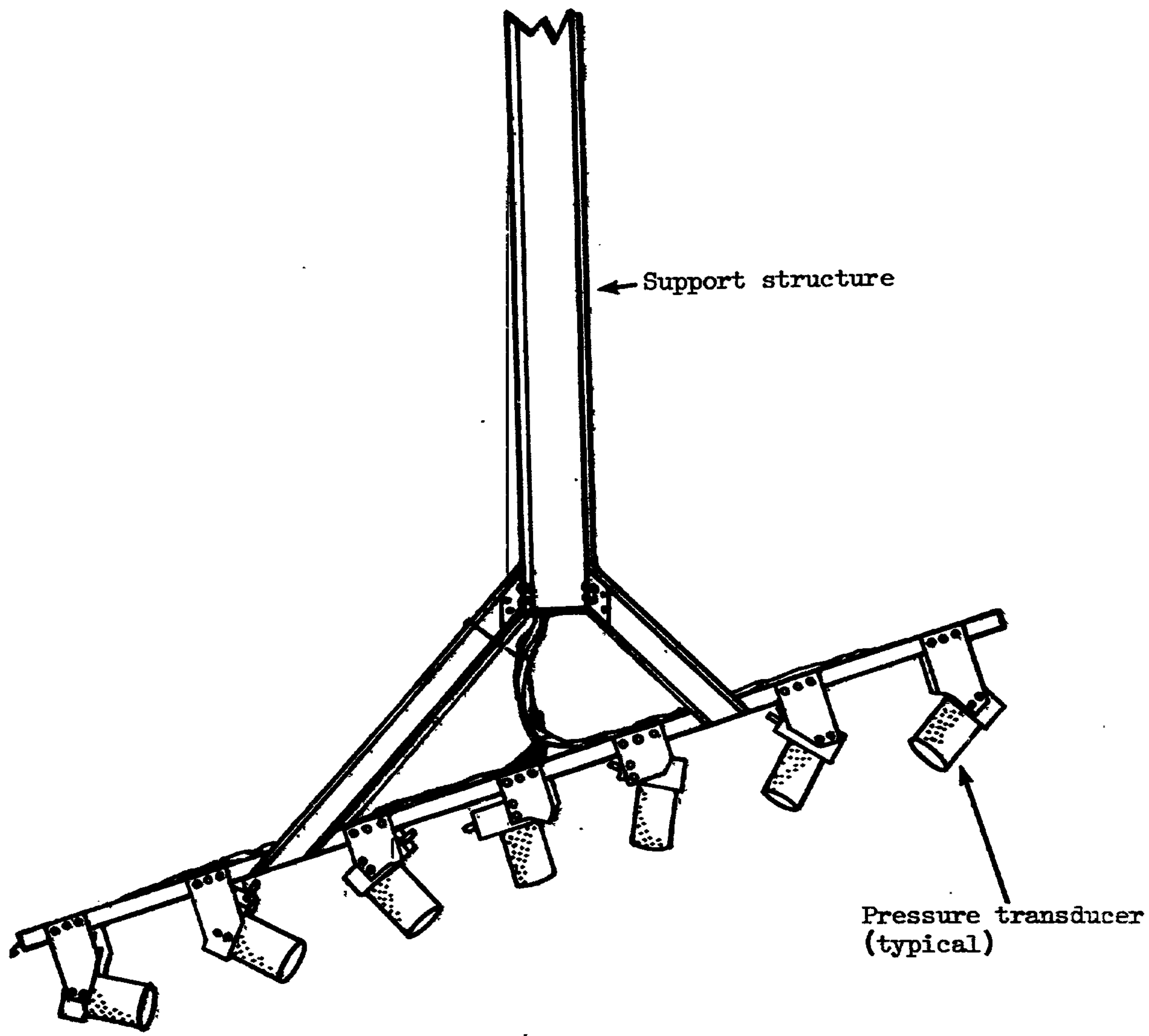

Figure 6.- Plume rake (typical).

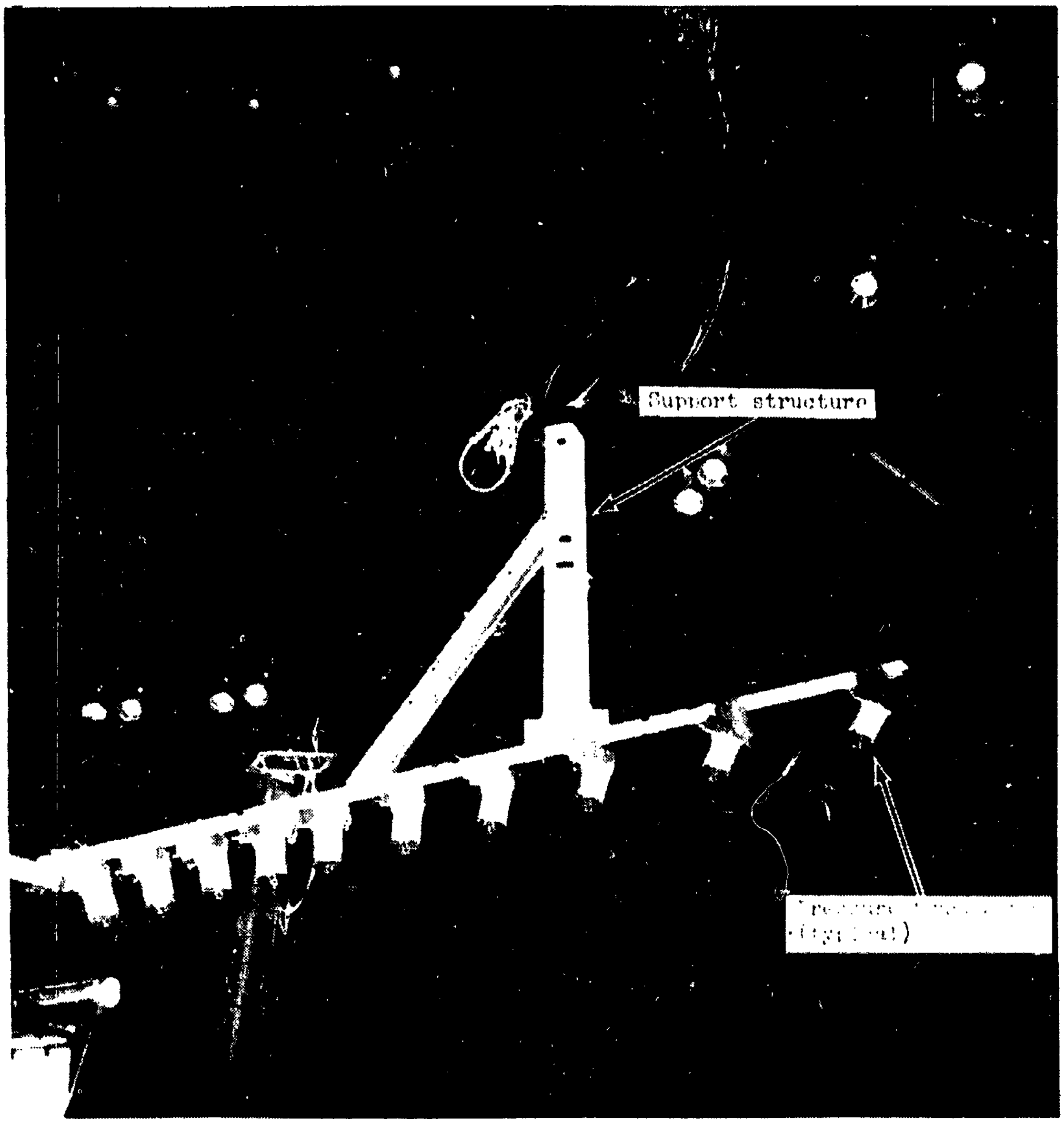

Figure 7.- Plume rake mounted on support structure.

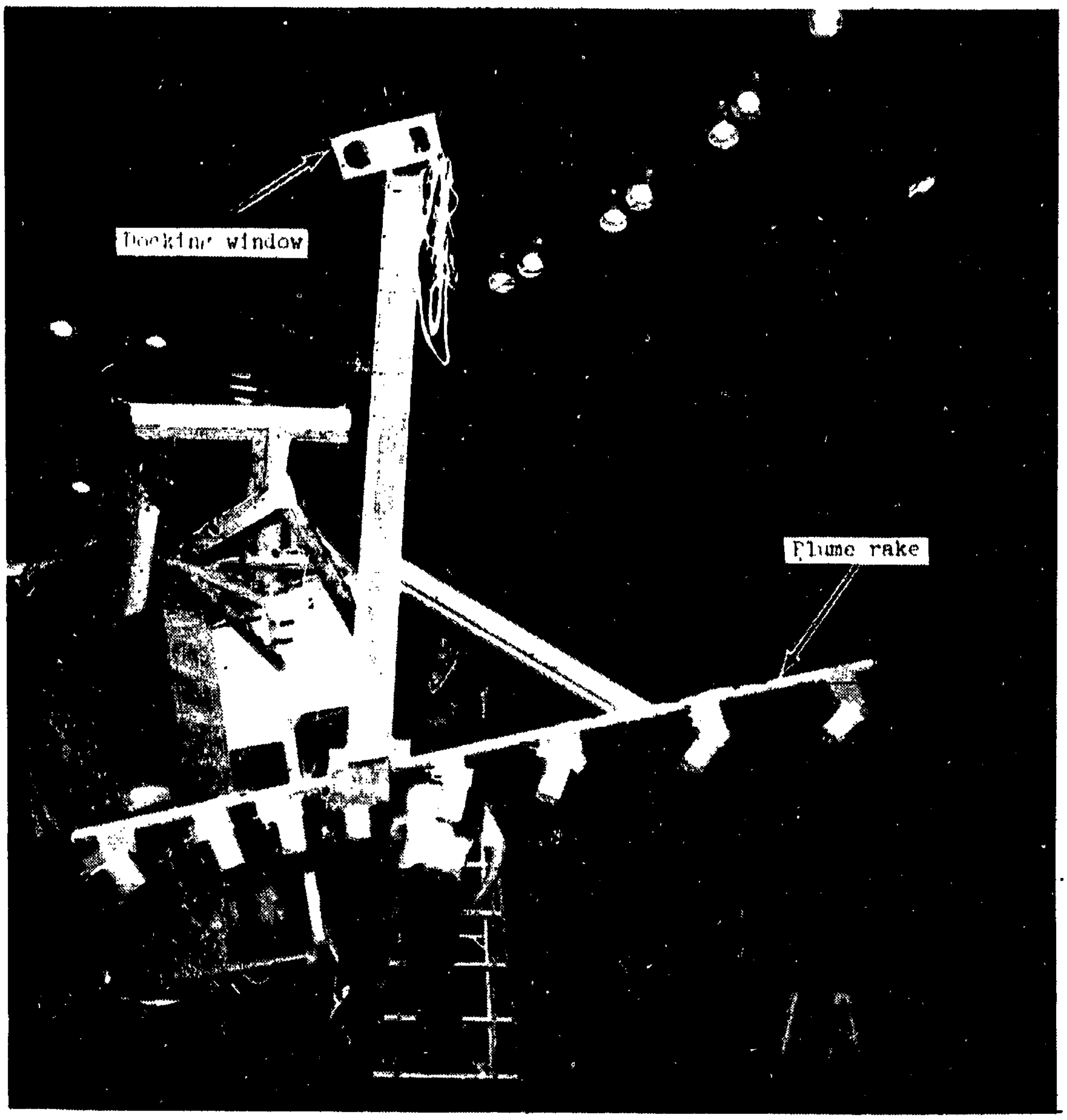

Figure 8.- Docking window mounted on support structure.

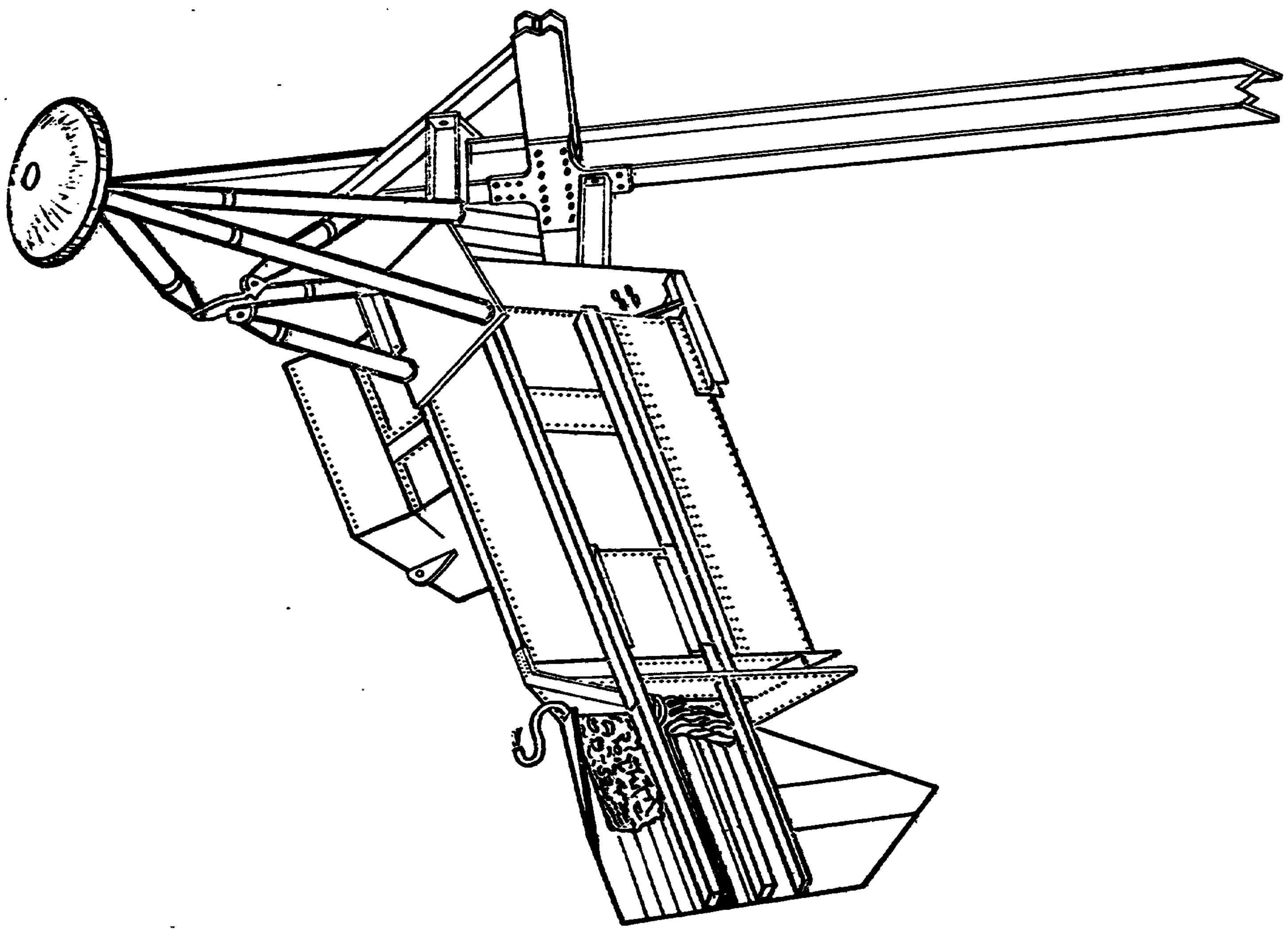

Figure 9.- Quad II simulator module.

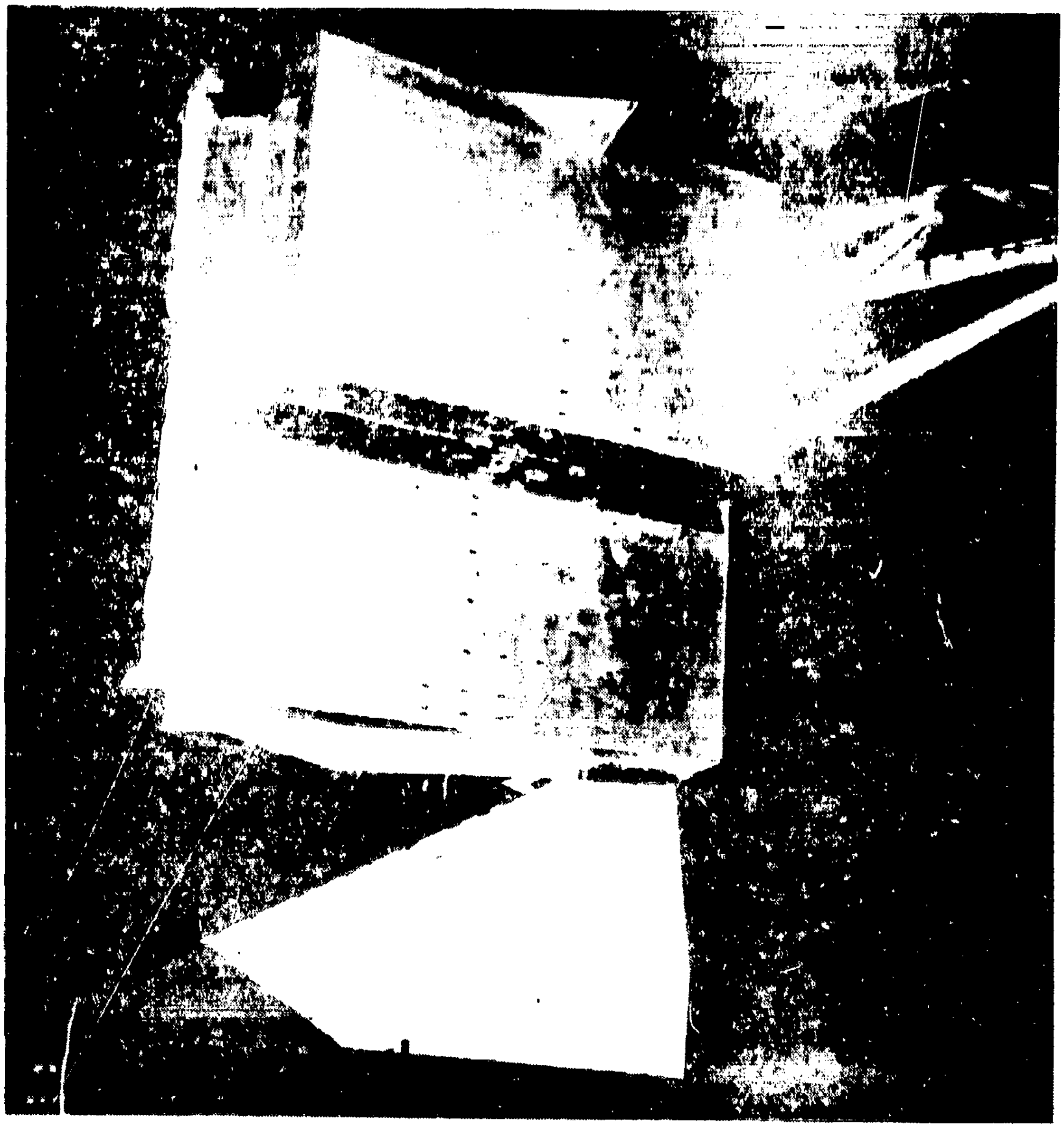

Figure 10.- Quad II simulator module mounted on support structure.

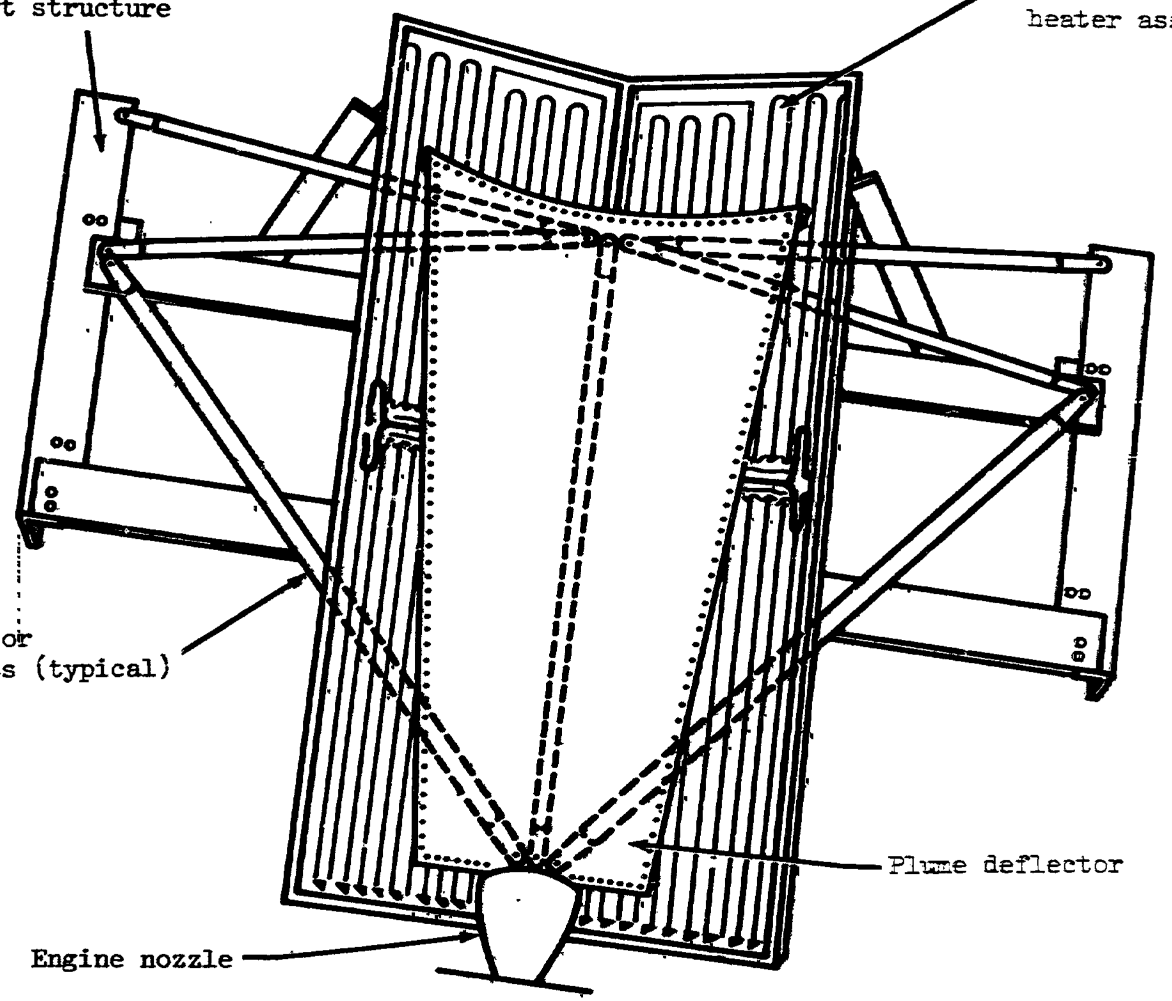

Figure 11.- Plume deflector.

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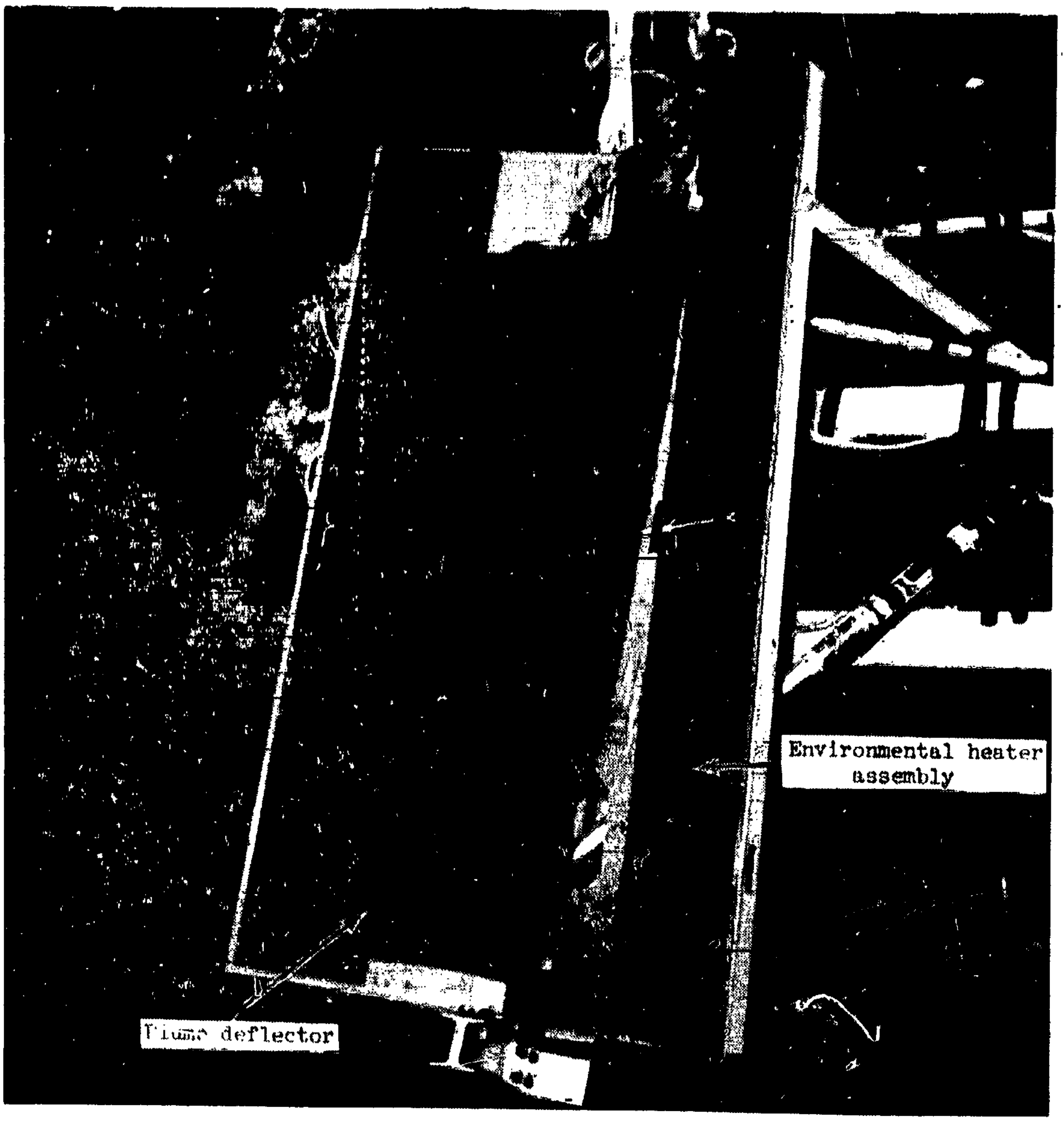

Figure 12.- Plume deflector mounted on support structure.

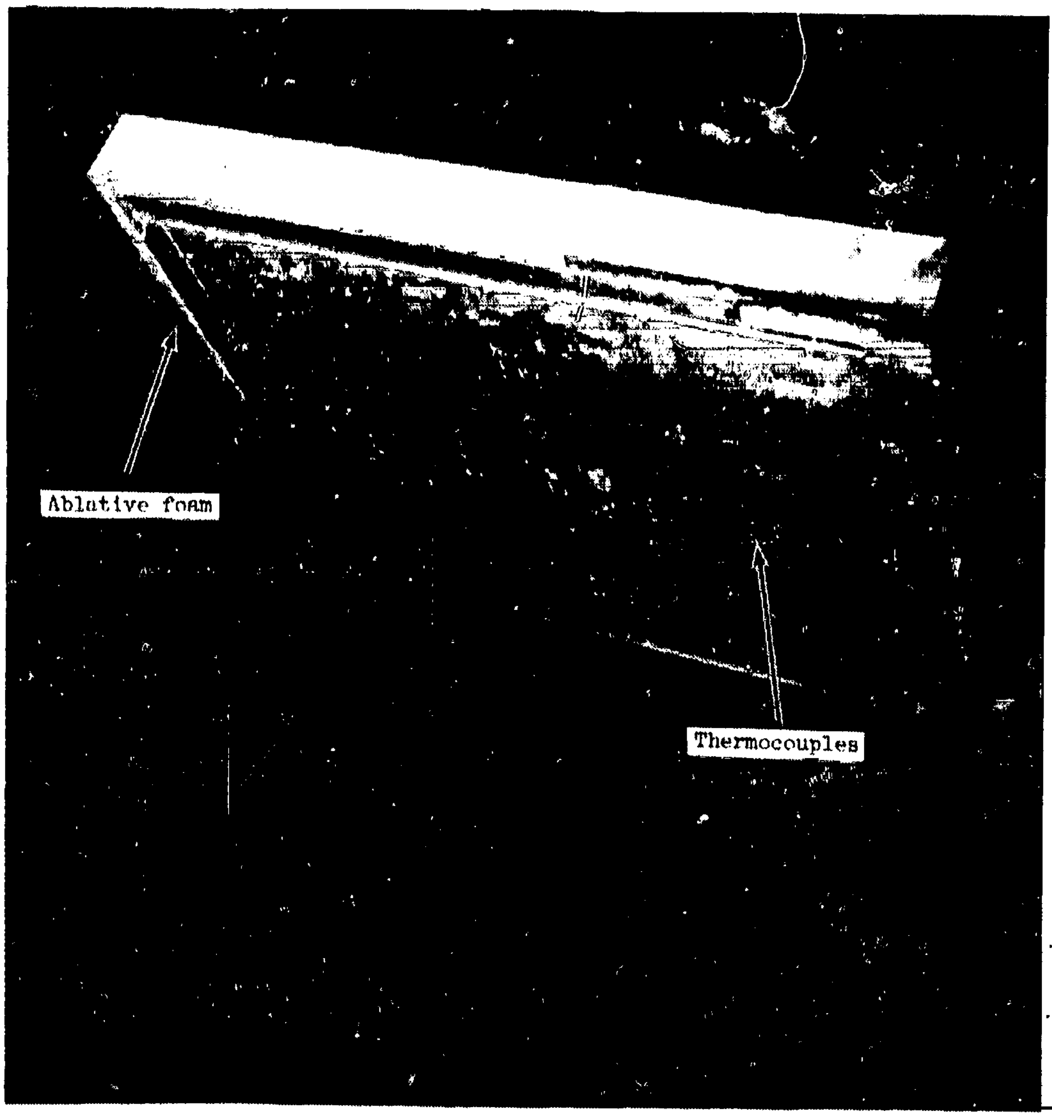

Figure 13.- Uninsulated panel and ablative foam.

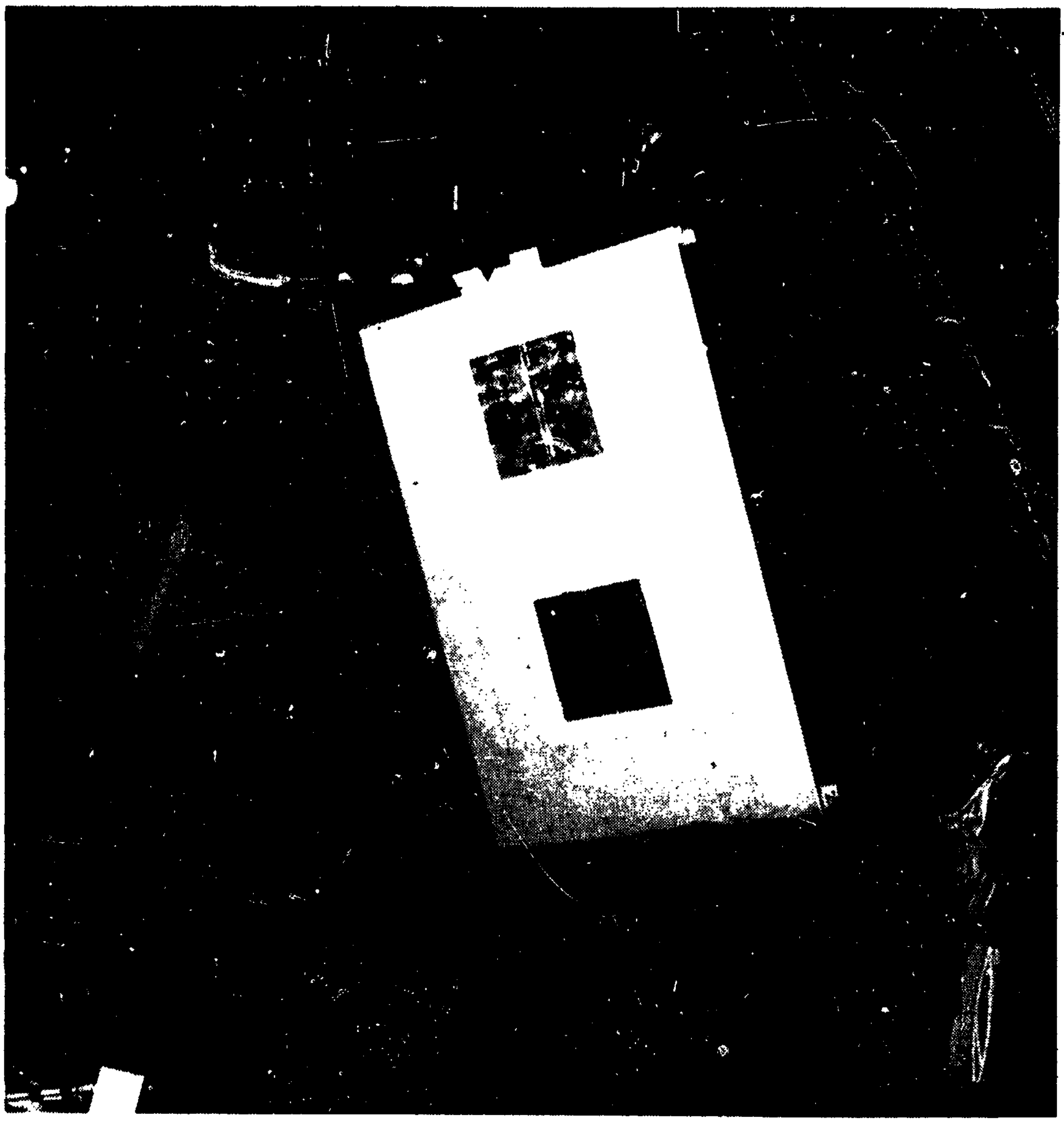

Figure 14.- Command module coating test panel mounted on support structure.

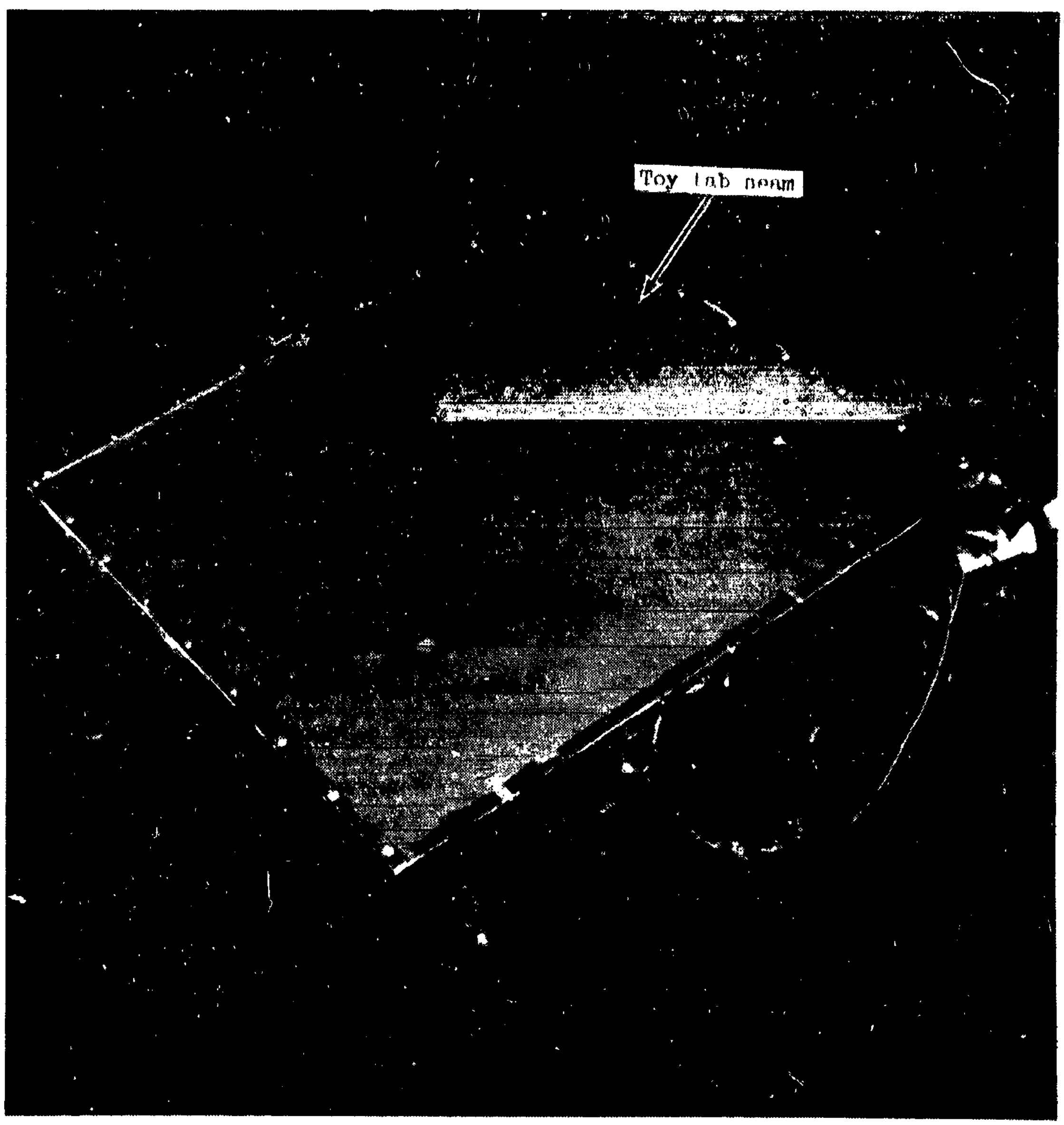

Figure 15.- Insulated panel mounted on support structure.

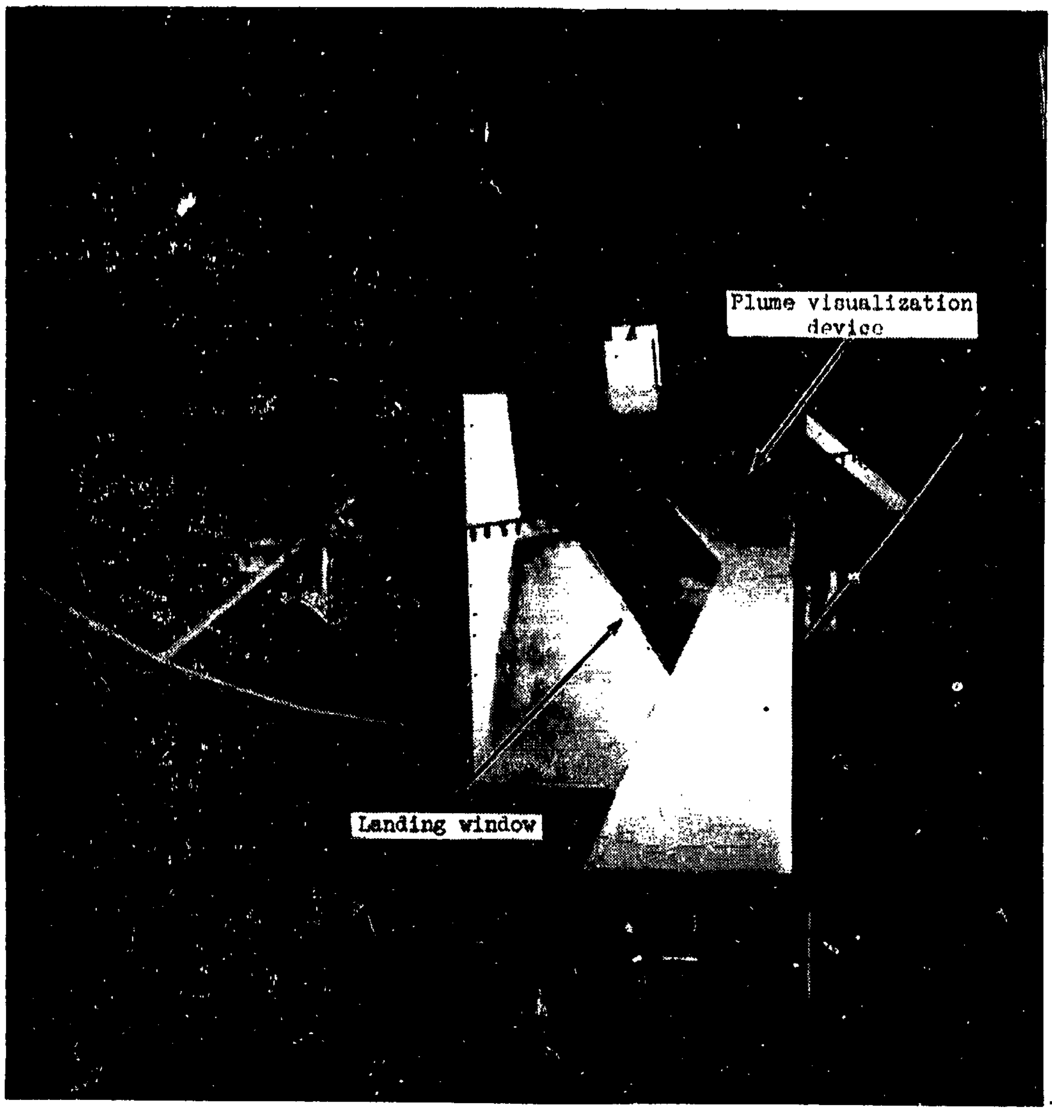

Figure 16.- Landing window and plume visualization device.

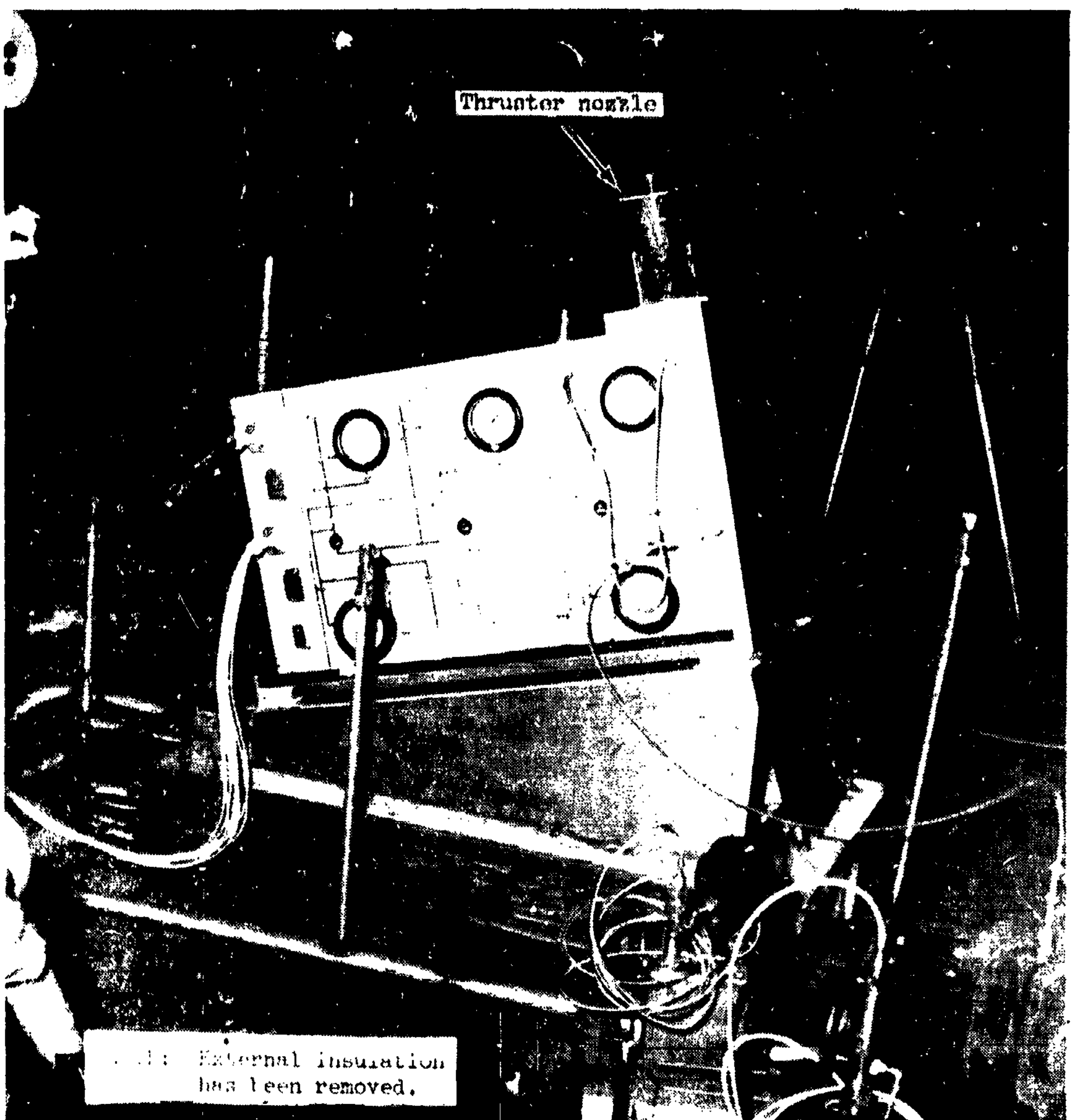

Figure 17.- Engine pallet, exterior.

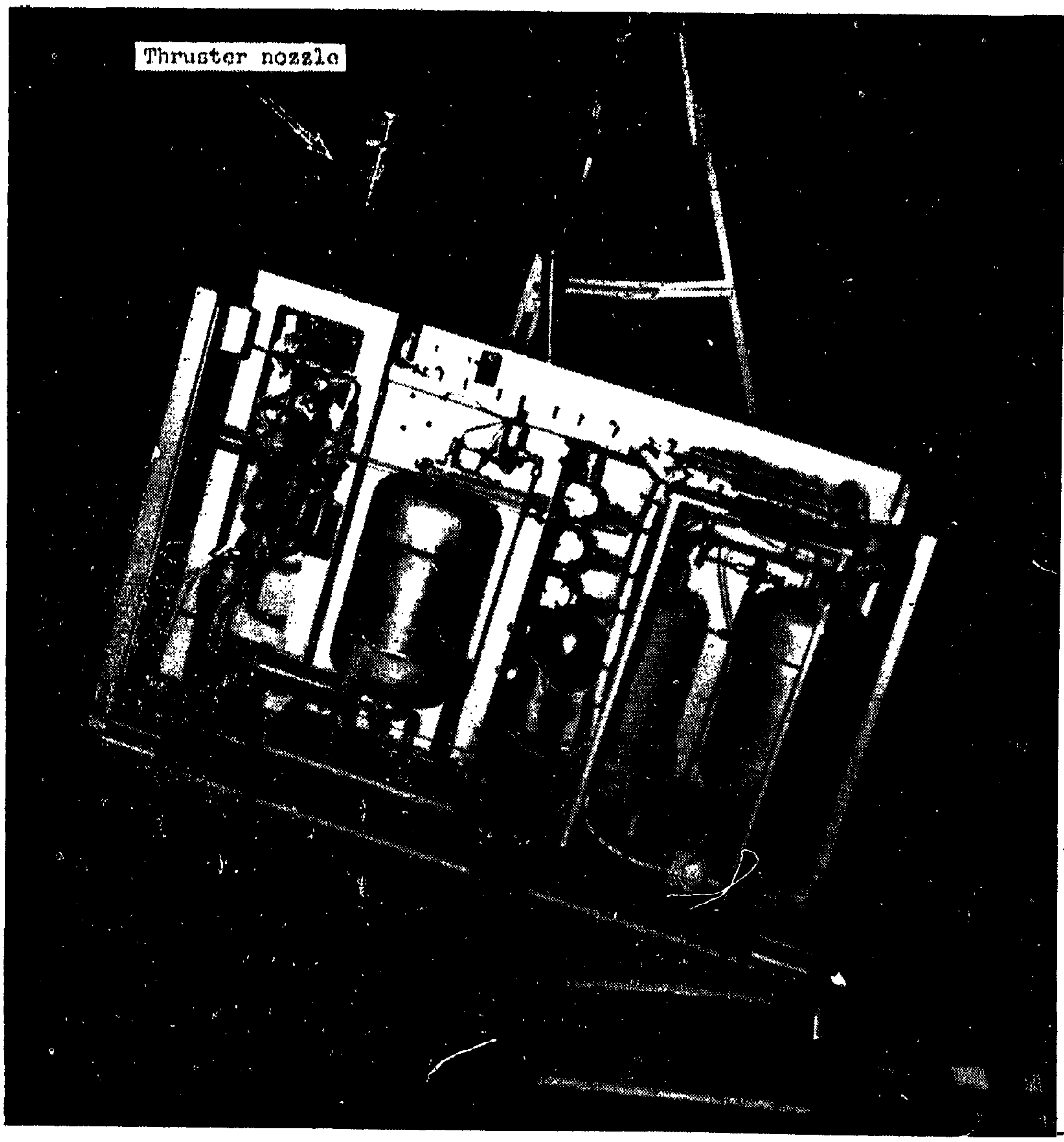

Figure 18.- Engine pallet, interior.

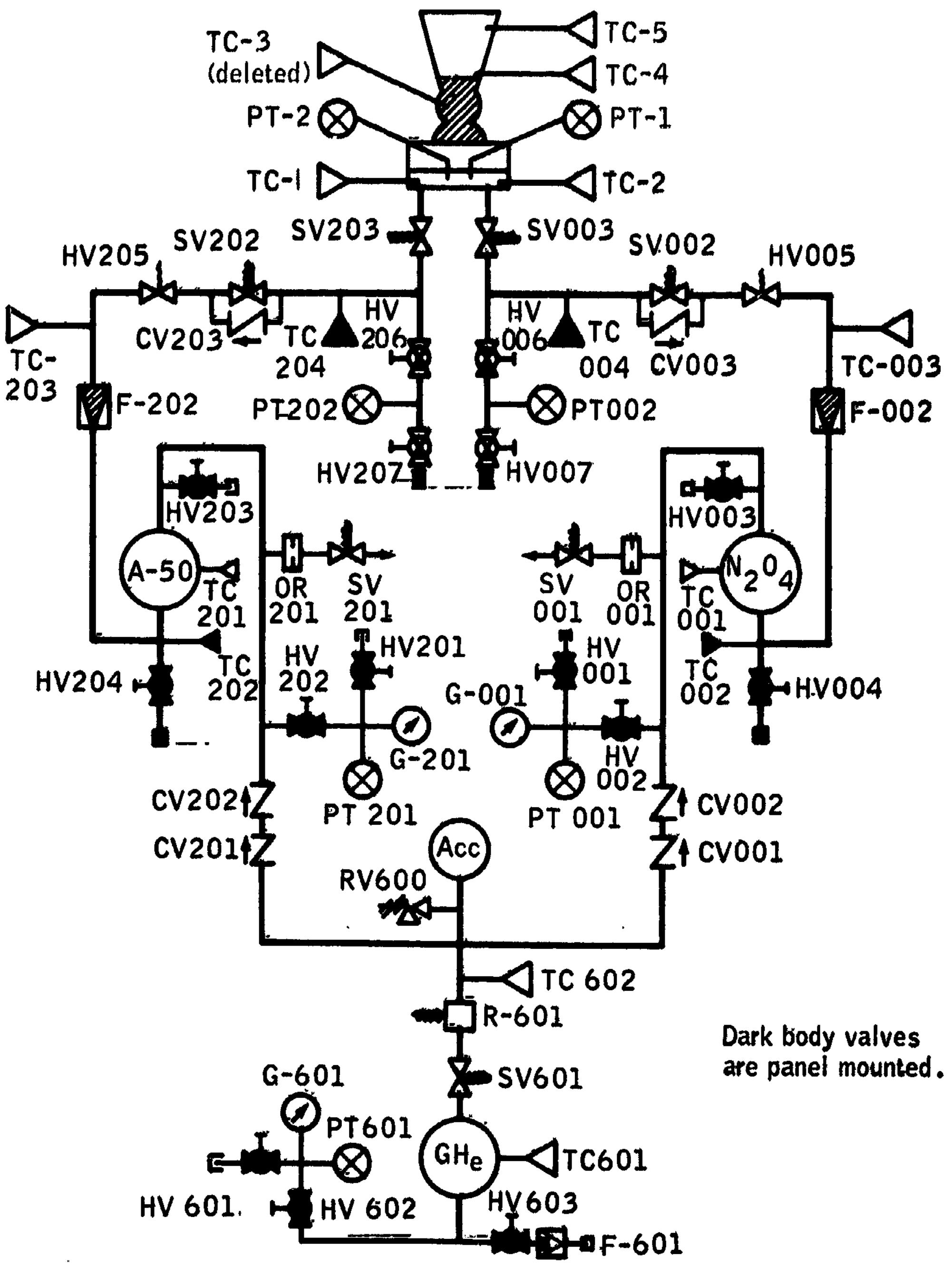

Figure 19.- Schematic diagram of the reaction control-system engine-pallet propellant and pressurant systems.

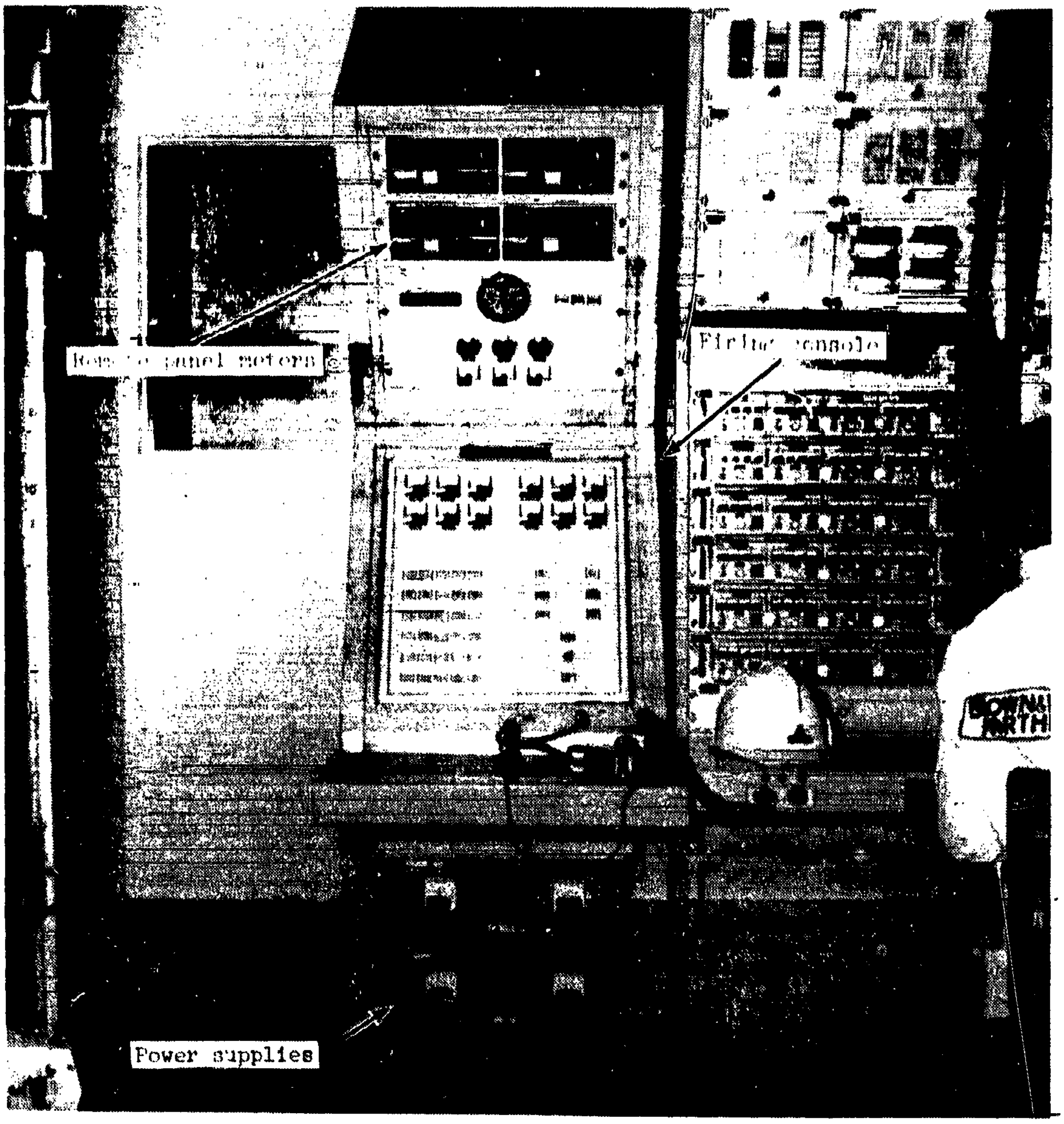

Figure 20.- Engine-firing console.

Figure 21.- Data acquisition system.

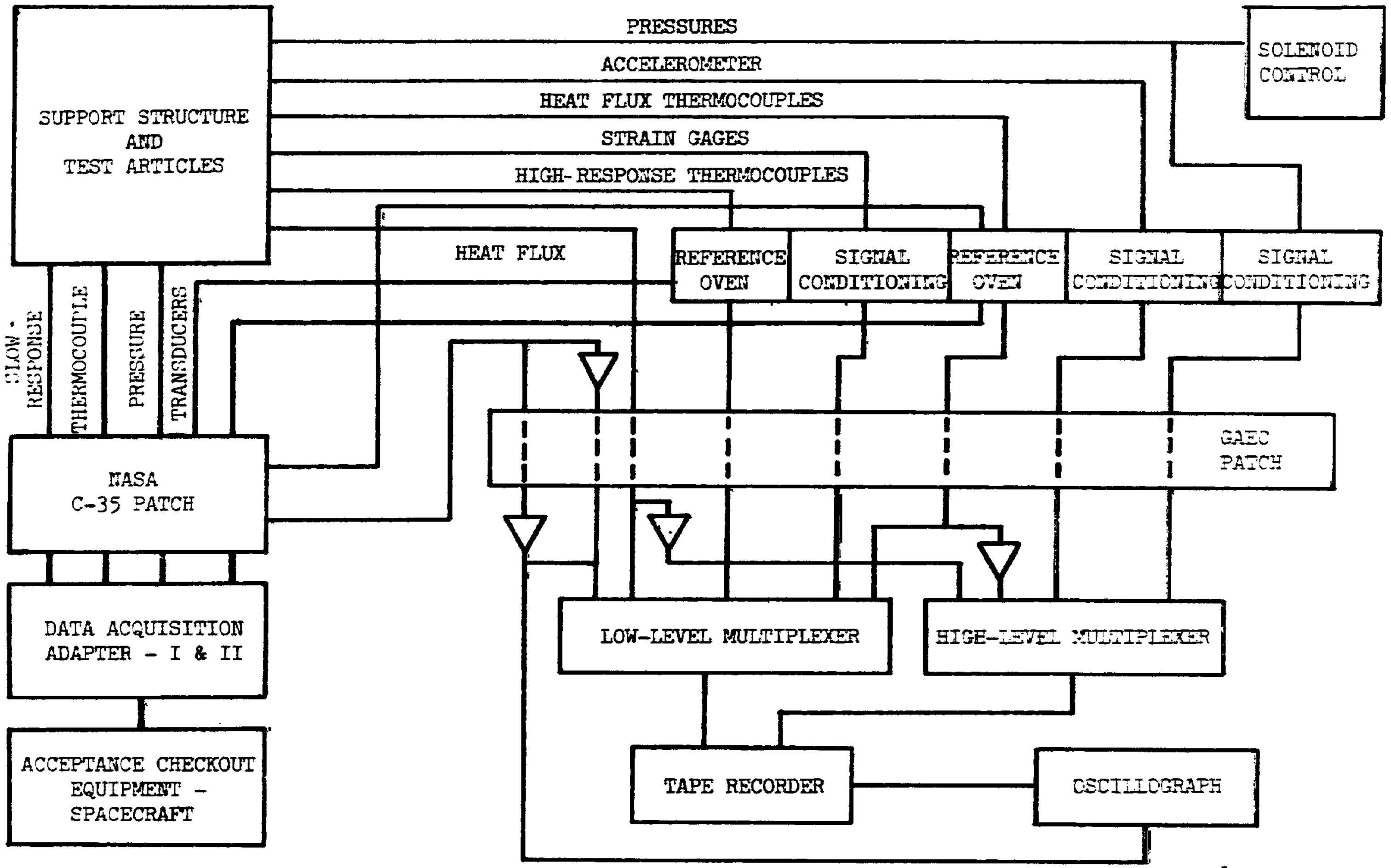

Figure 22.- Data acquisition system data flow.

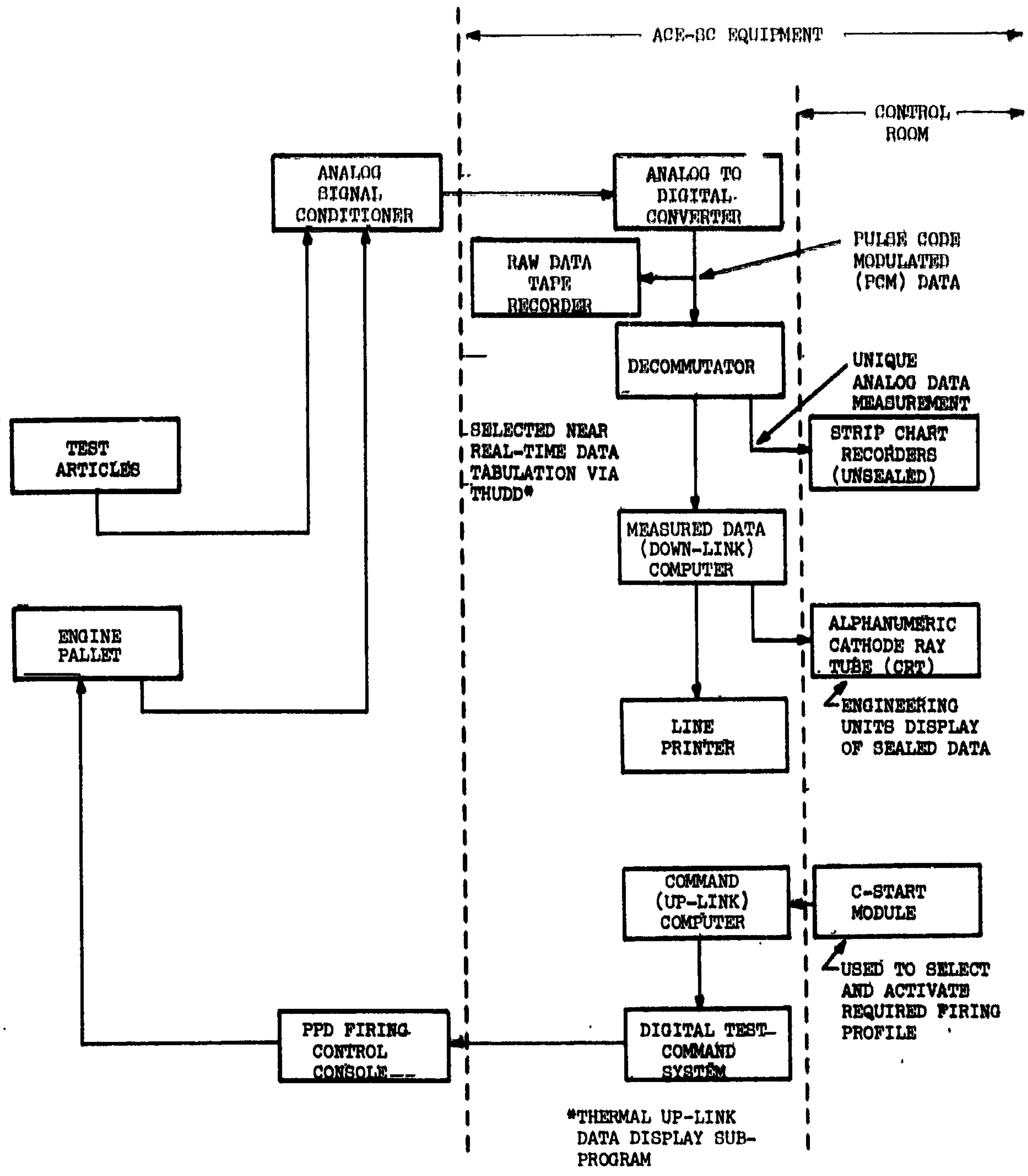

Figure 23.- Acceptance Checkout Equipment-Spacecraft, block diagram.

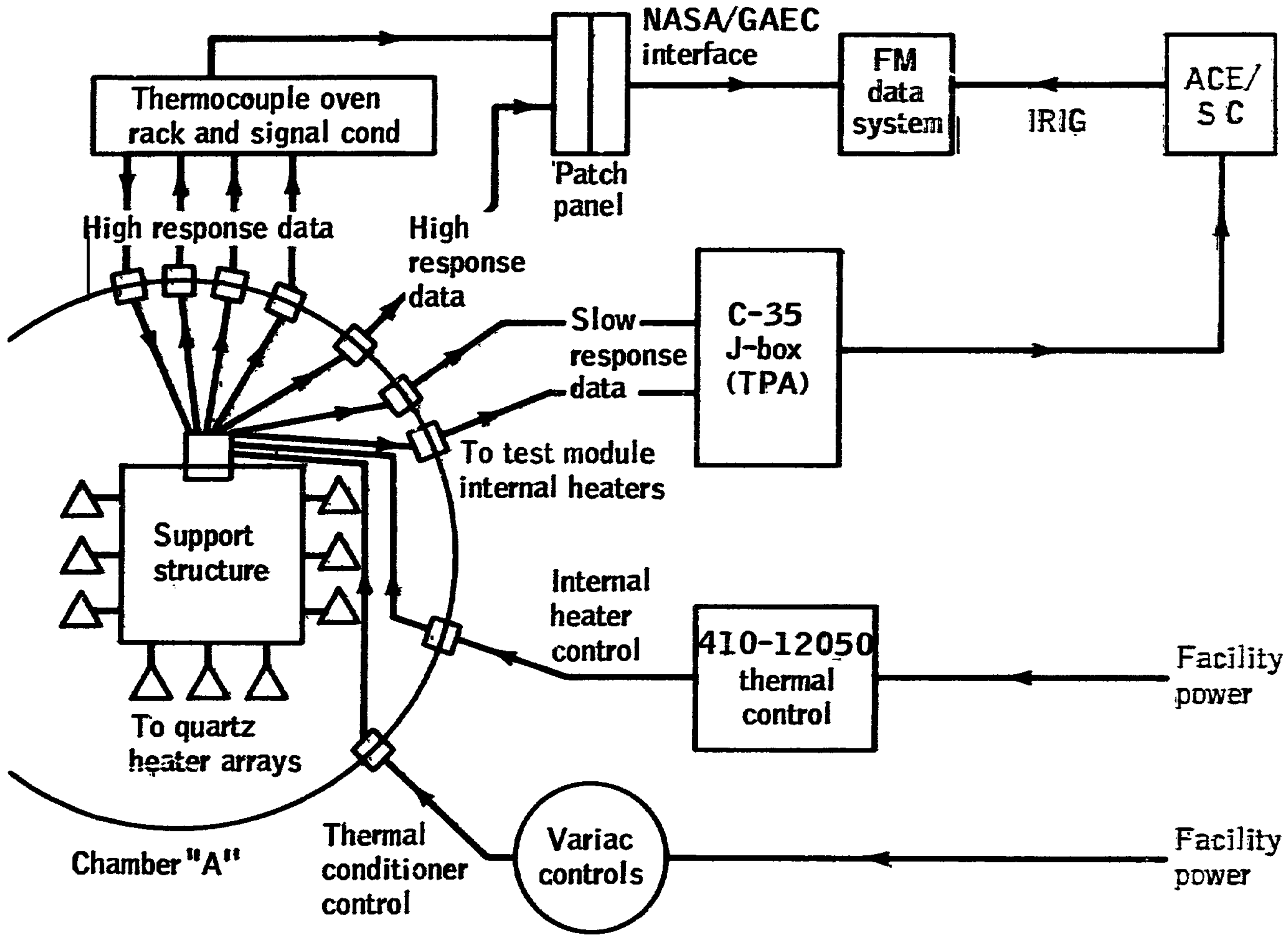

Figure 24.- Schematic diagram of the thermal control system.

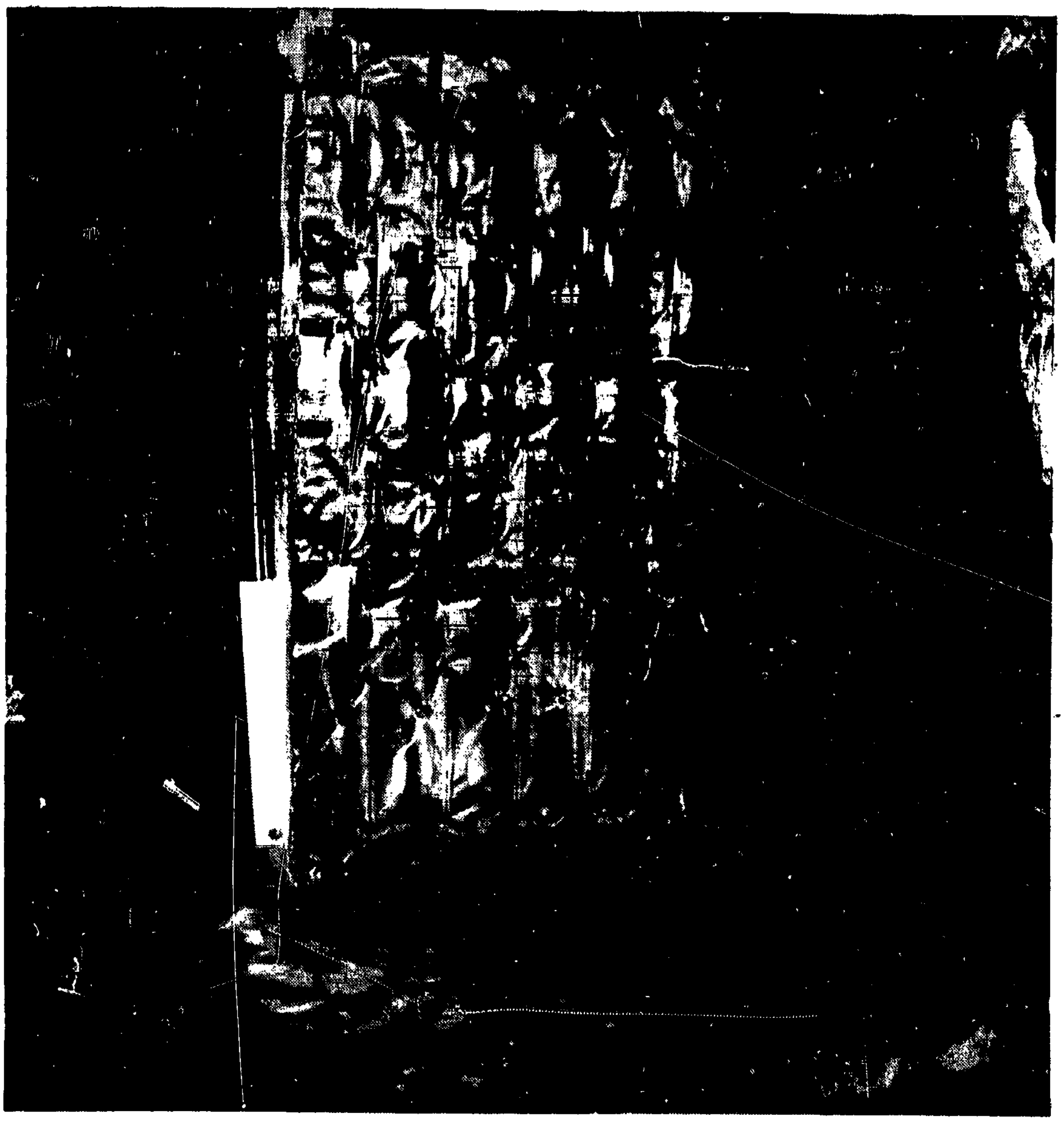

Figure 25.- Quartz lamp heater.

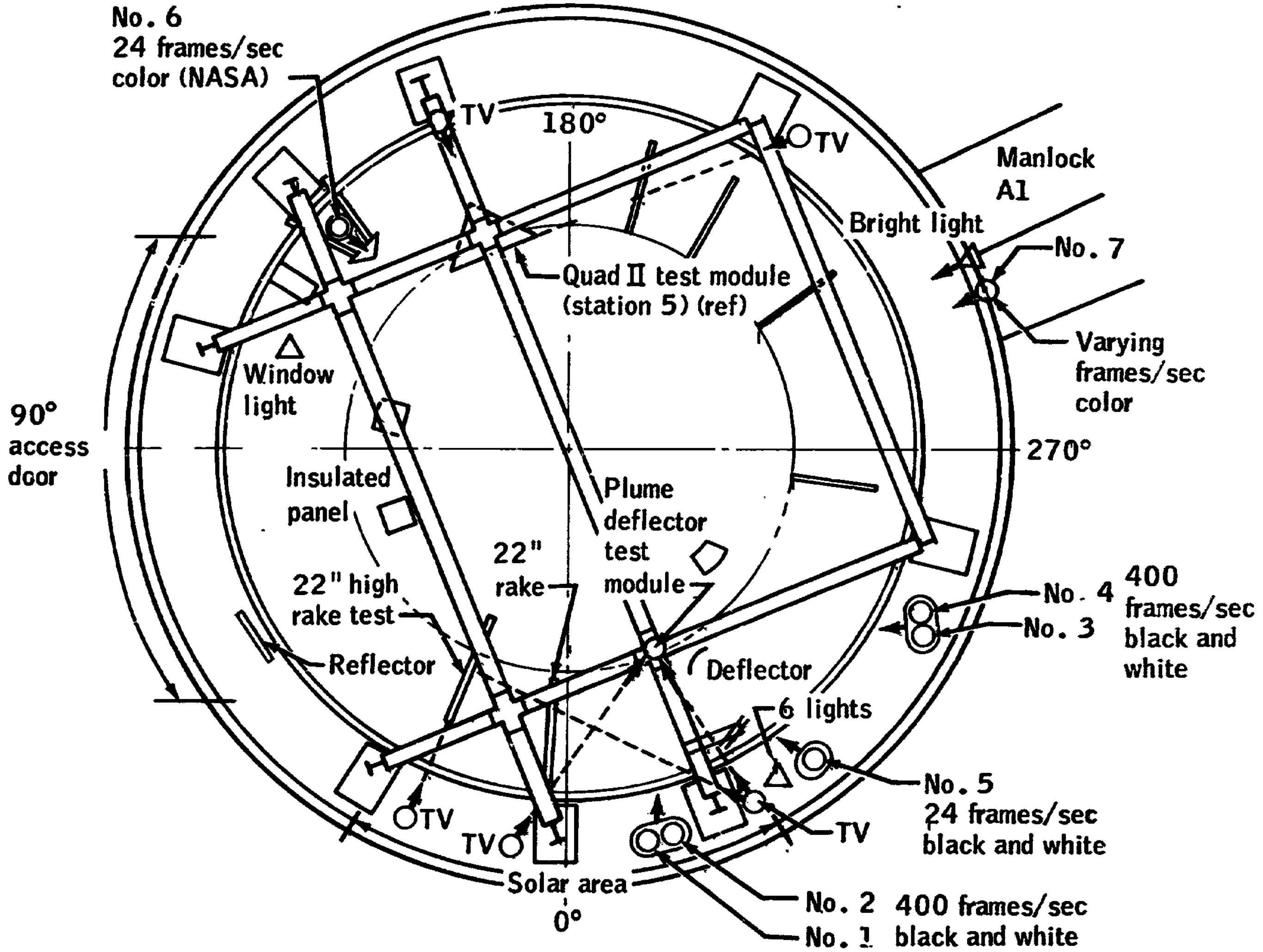

Figure 26.- Camera locations in chamber A.

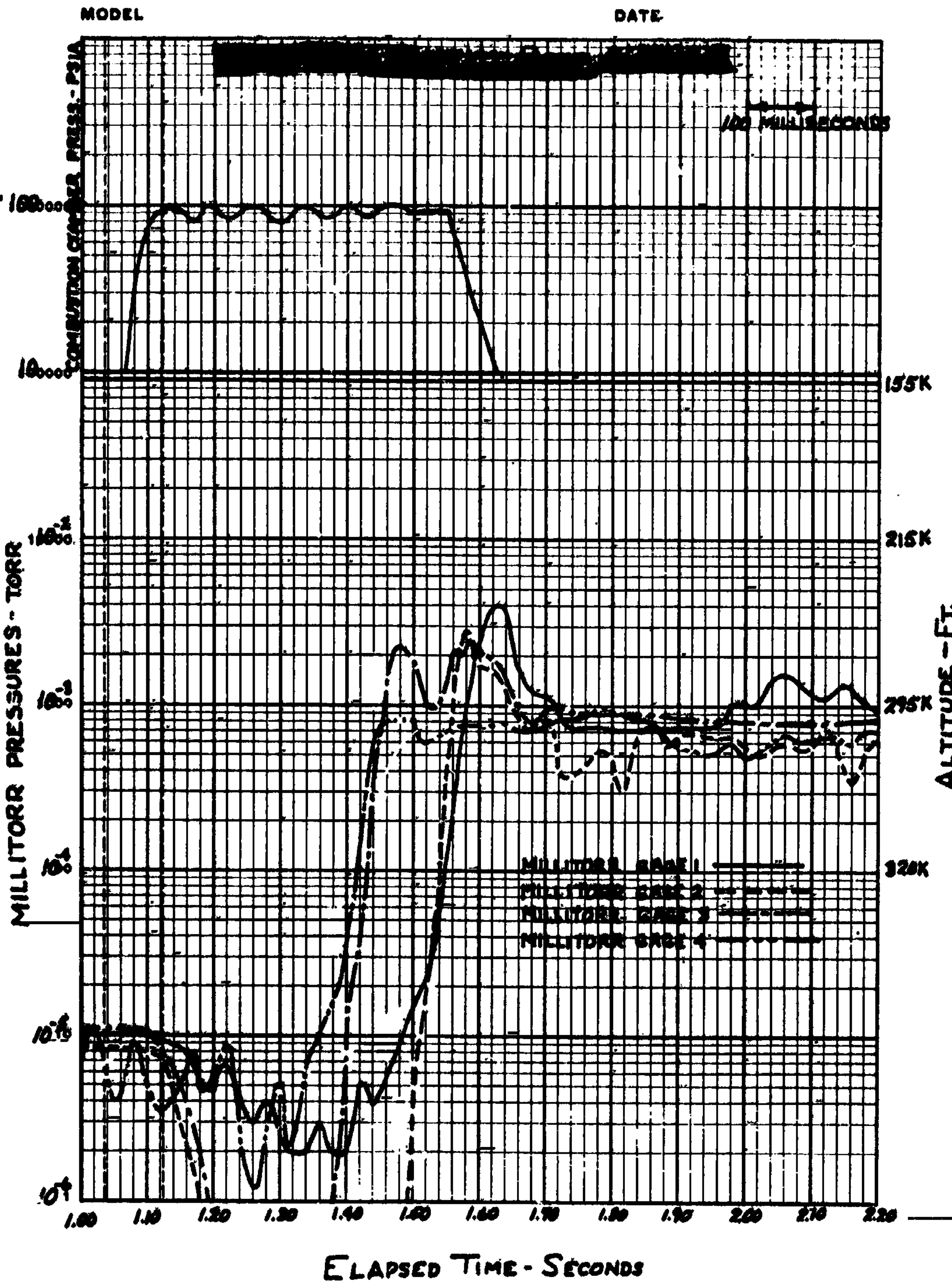

Figure 27.- Dynamic chamber pressure, rake 4, burn 1.

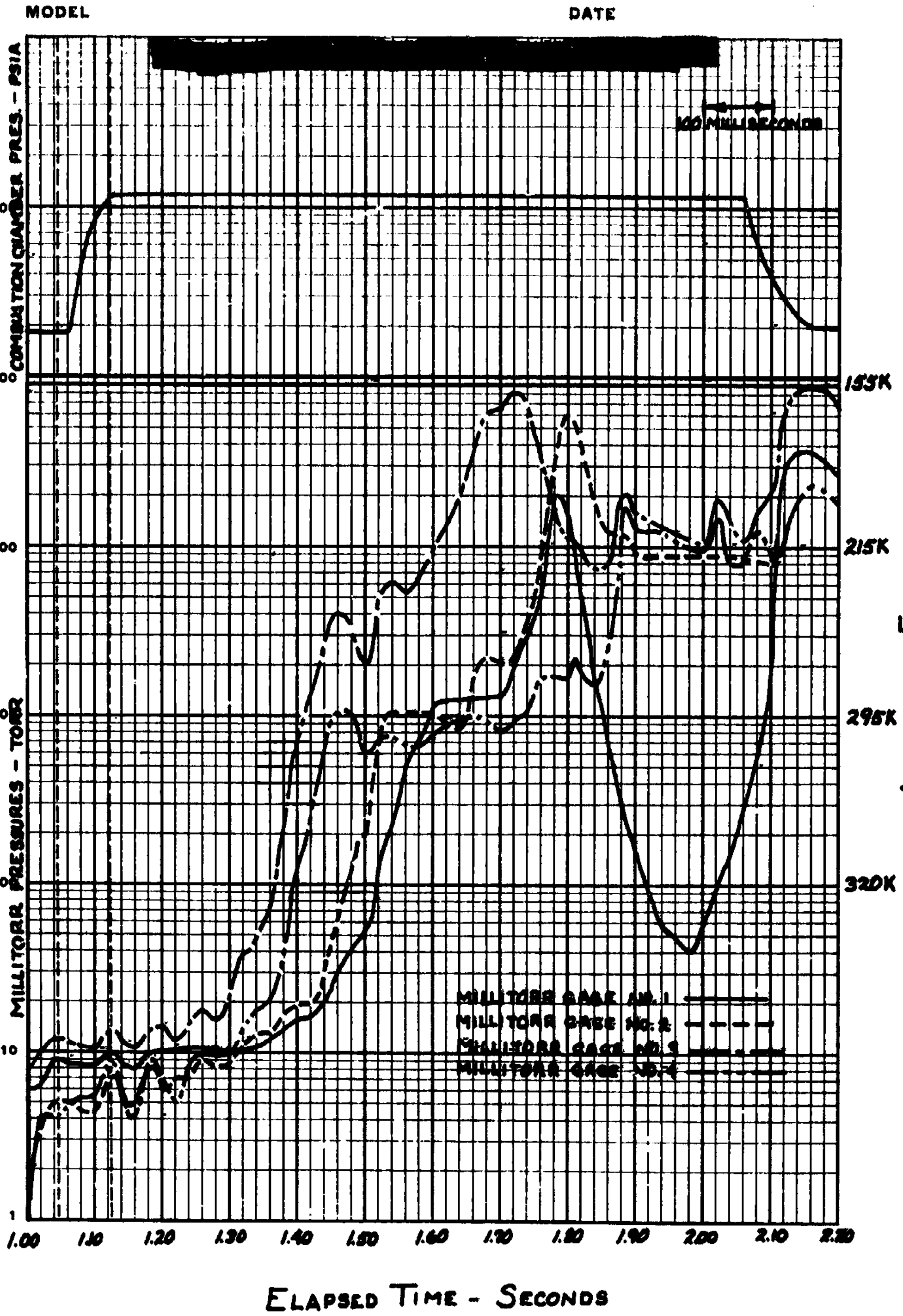

Figure 28.- Dynamic chamber pressure, rake 4, burn 2.

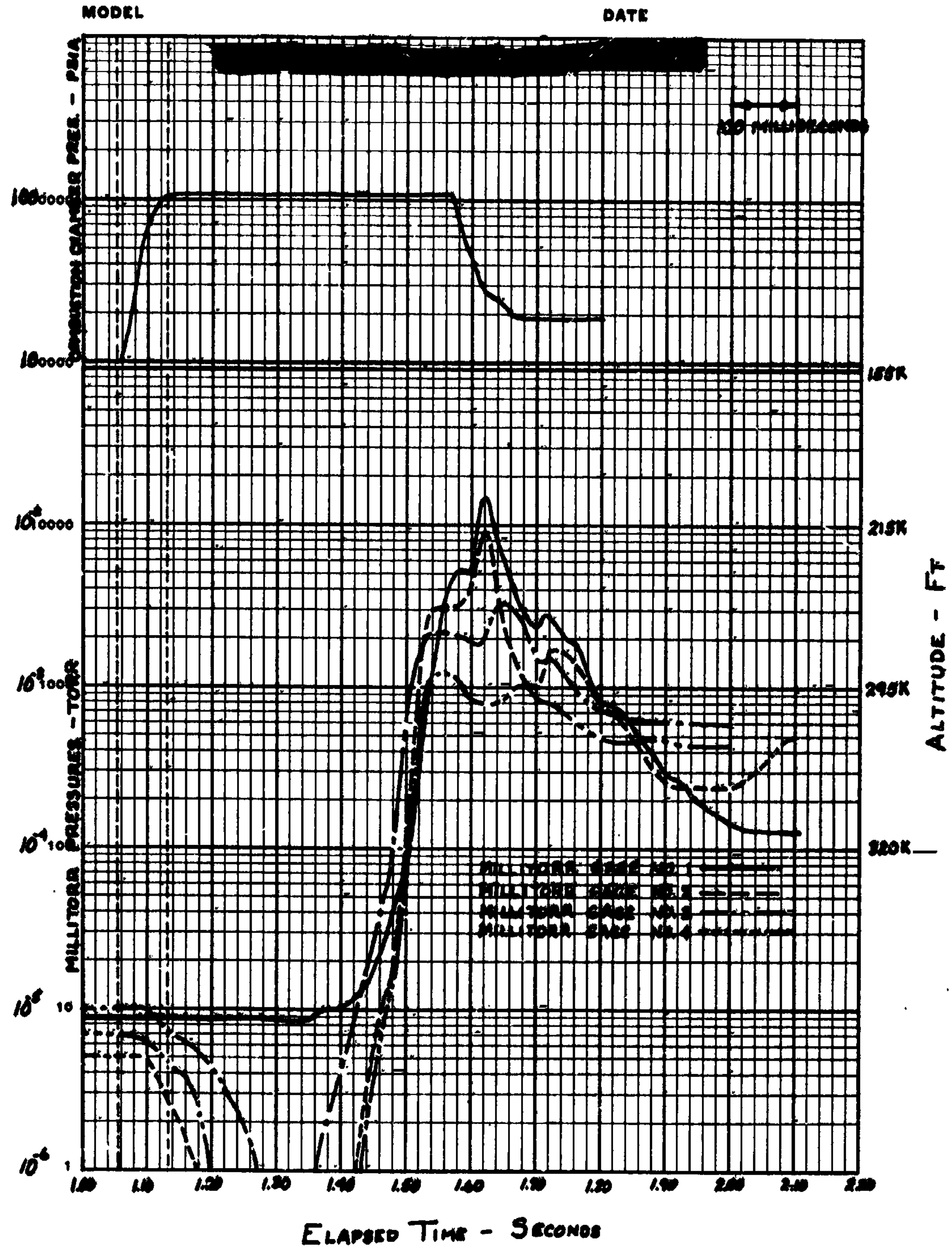

Figure 29.- Dynamic chamber pressure, rake 3, burn 1.

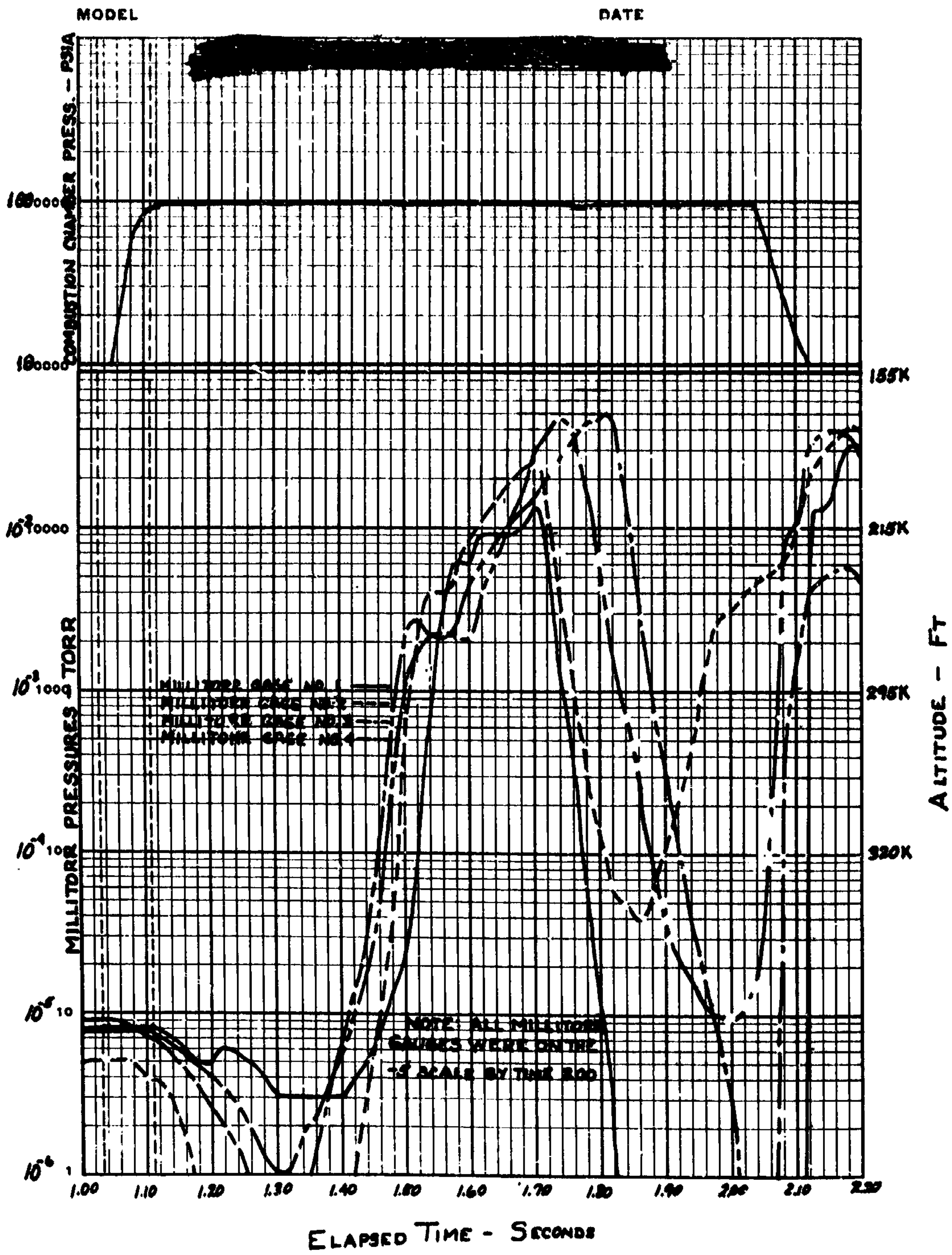

Figure 30.- Dynamic chamber pressure, rake 3, burn 2.

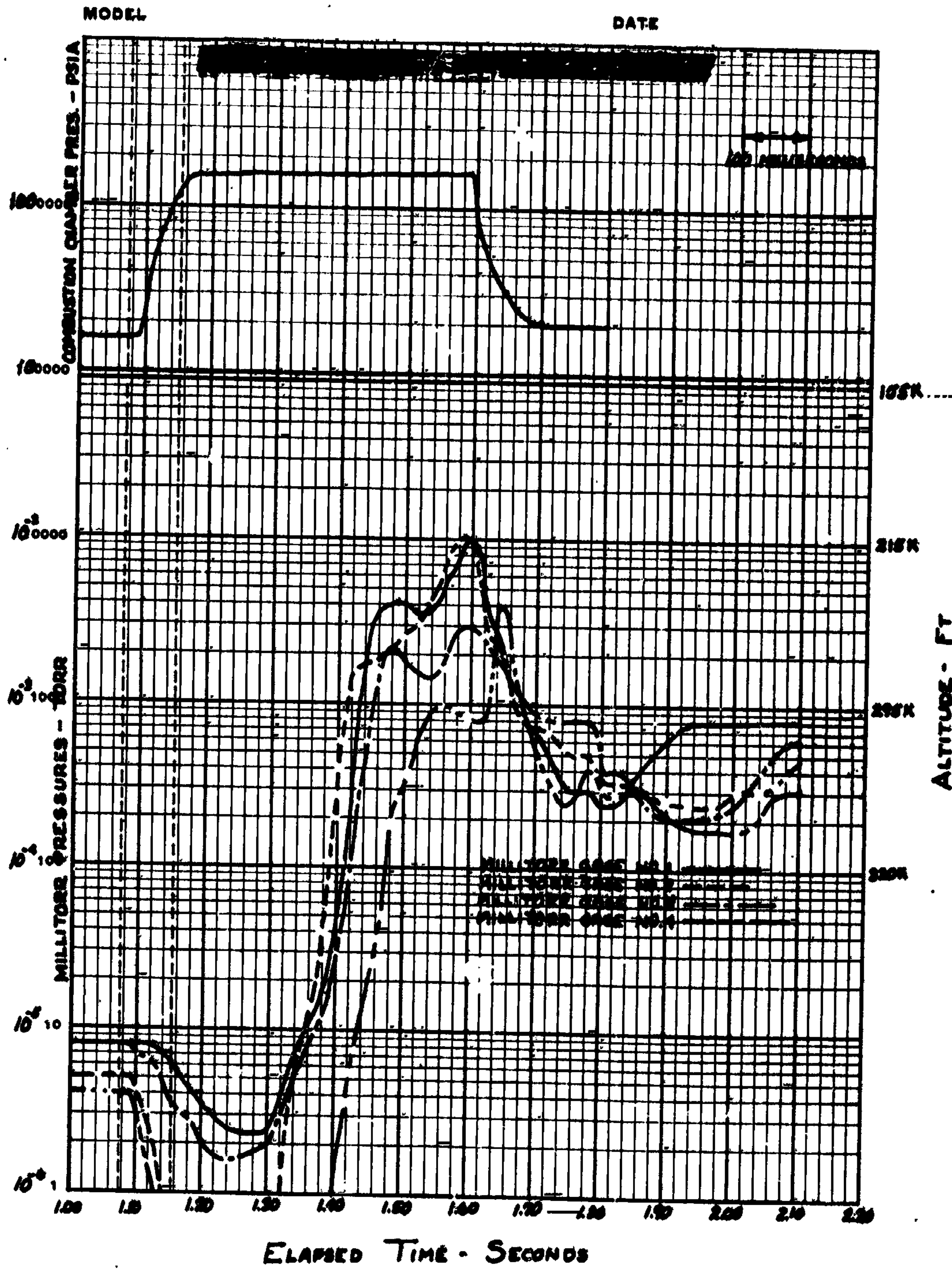

Figure 31.- Dynamic chamber pressure, rake 2, burn 1.

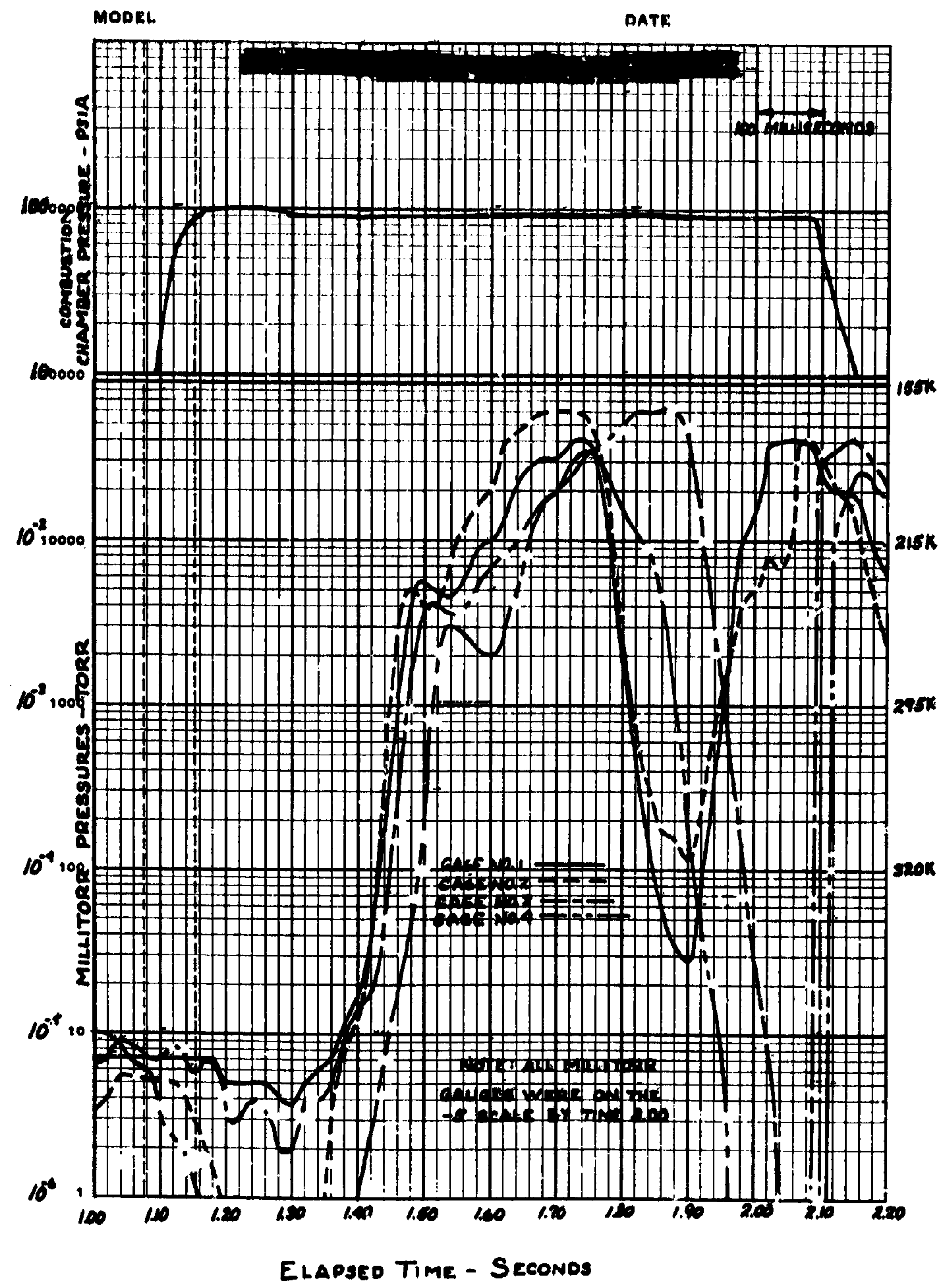

Figure 32.- Dynamic chamber pressure, rake 2, burn 2.

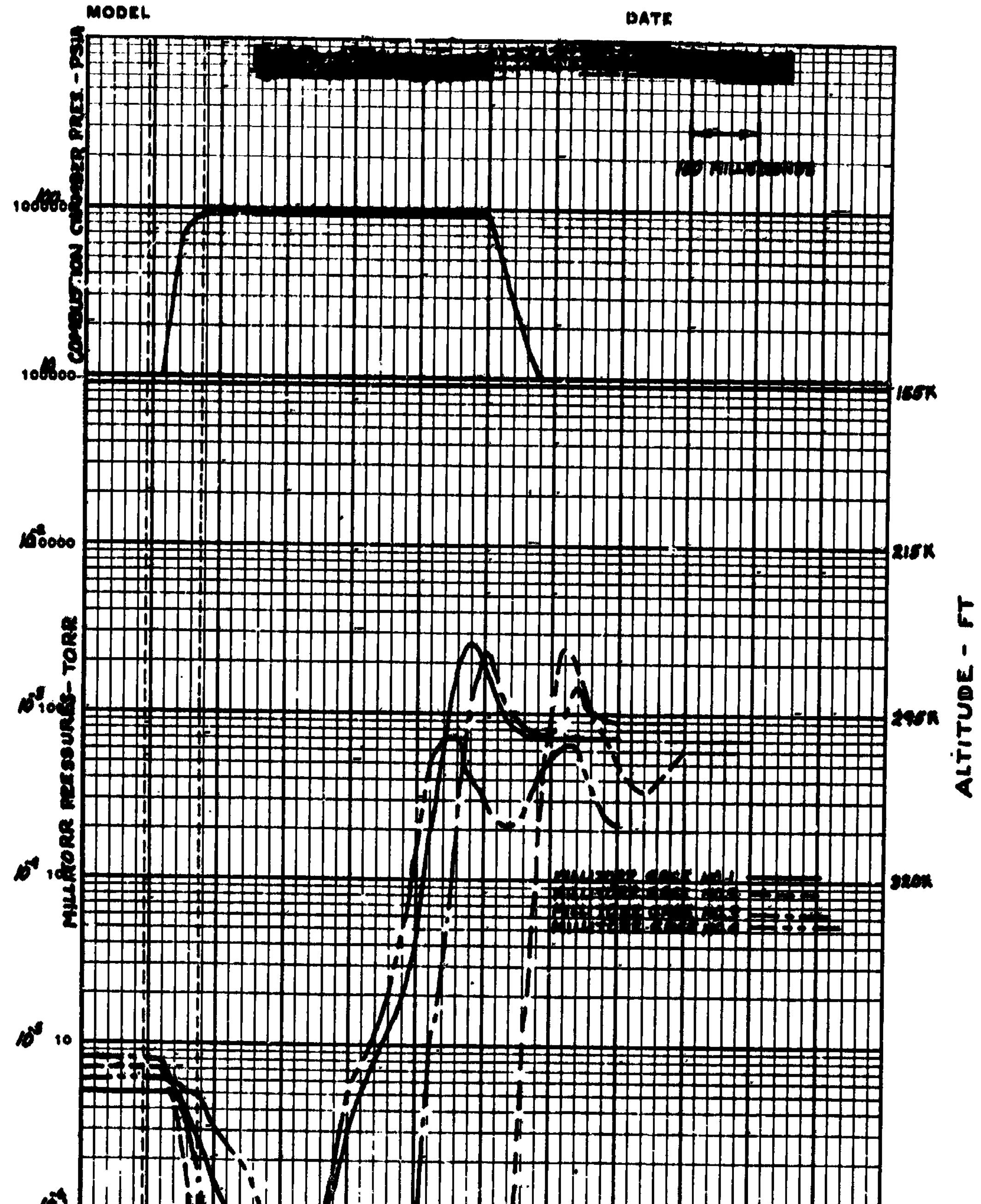

Figure 33.- Dynamic chamber pressure, rake 1, burn 1.

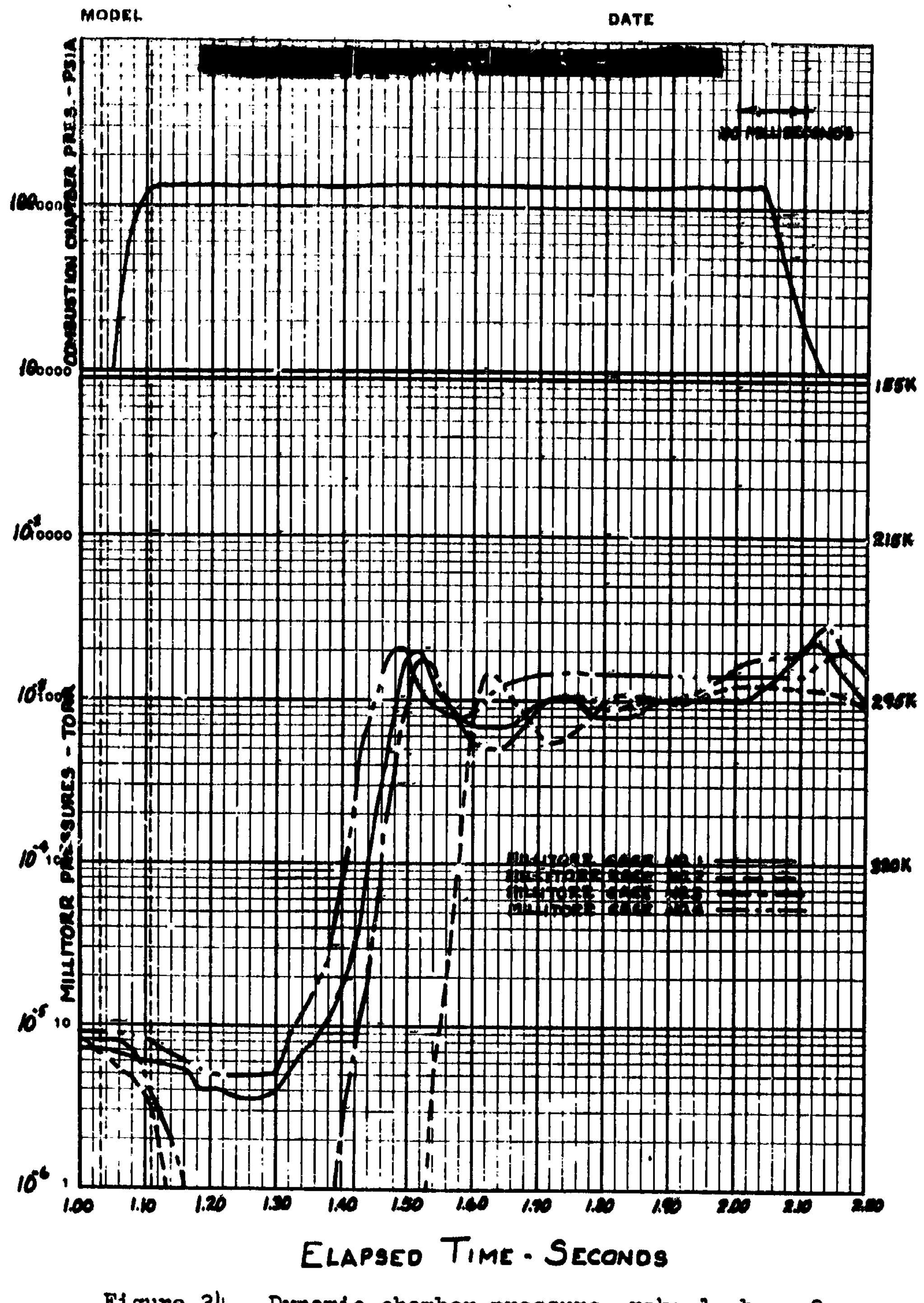

Figure 34.- Dynamic chamber pressure, rake 1, burn 2.

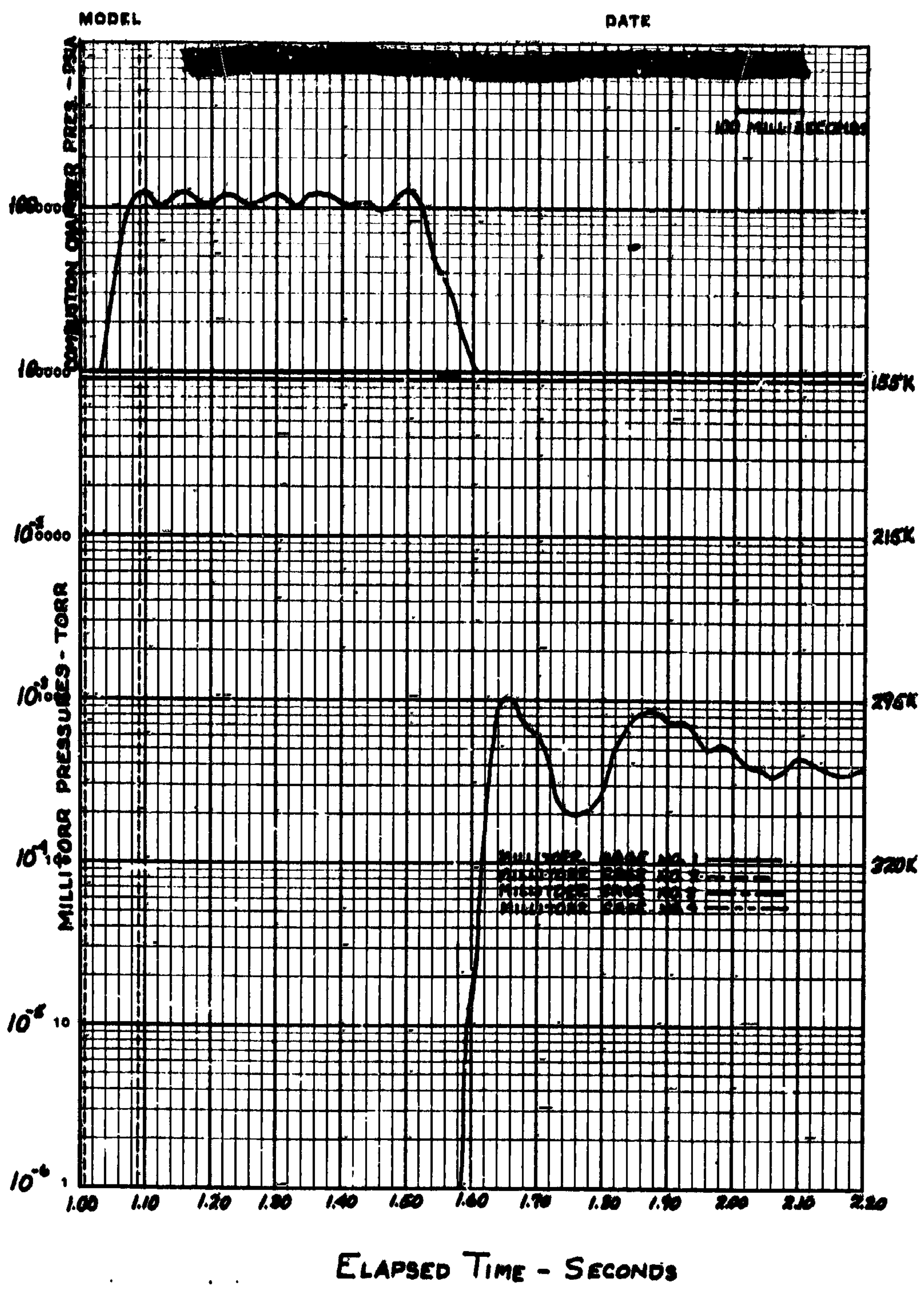

Figure 35.- Dynamic chamber pressure, uninsulated panel, burn 1.

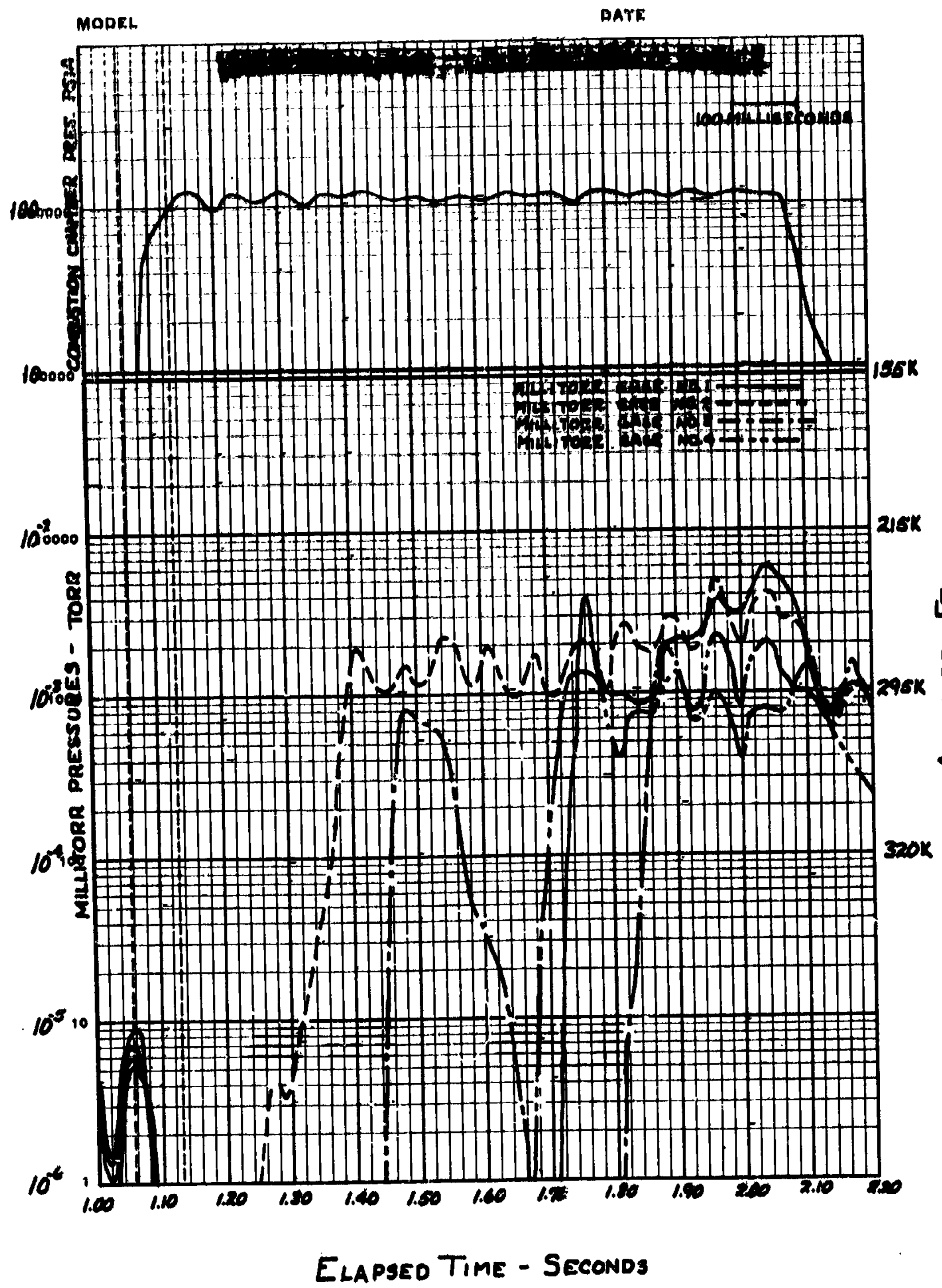

Figure 36.- Dynamic chamber pressure, uninsulated panel, burn 2.

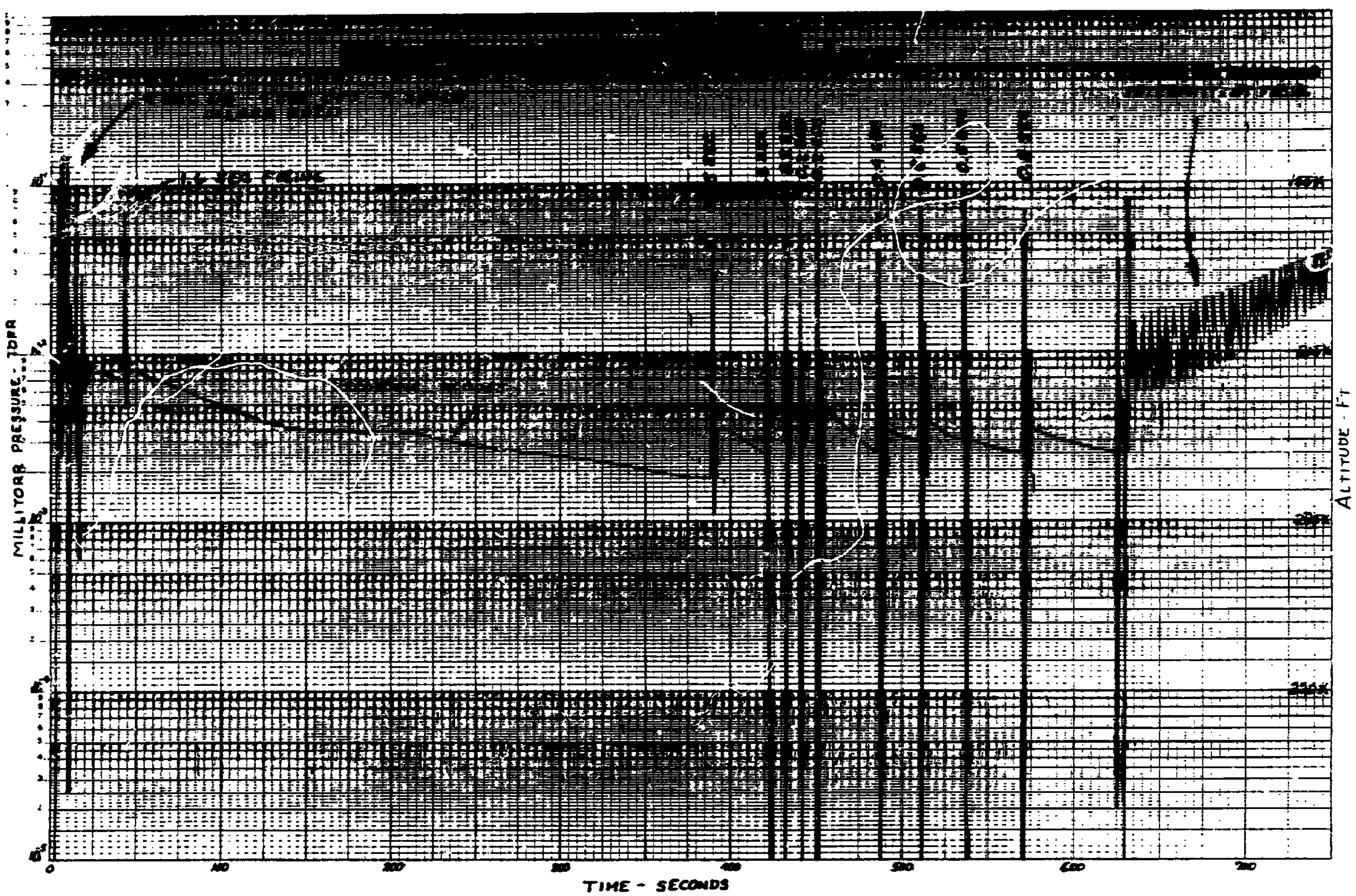

Figure 37.- Dynamic chamber pressure, insulated panel burn.