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INTERNAL NOTE

SATURN IB INFLIGHT PHOTOGRAPHIC INSTRUMENTATION SYSTEM

by

A. J. Davis
P. L. Hassler, Jr.

MEASURING BRANCH
ASTRIONICS LABORATORY

GEORGE C. MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

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ABSTRACT

This Internal Note presents the development of the Saturn inflight photographic instrumentation program from its original development requirement concept to the flight hardware application on Saturn vehicles. A comprehensive description of the inflight photographic instrumentation system is given along with data concerning testing, operation, application, and evaluation of the system after recovery.

This Internal Note shows that the system has been successfully developed, that valuable information has been obtained from film retrieved from recovered capsules, and that the system can be used with a high degree of reliability.

NASA-GEORGE C. MARSHALL SPACE FLIGHT CENTER

INTERNAL NOTE

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PROPULSION AND VEHICLE ENGINEERING LABORATORY

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SECTION I. INTRODUCTION

A. PROJECT HISTORY

In 1961 inflight photographic instrumentation was first used to record the separation of the warhead from a Redstone Rocket booster. Two camera systems were used--one was mounted in the booster and one on the warhead. The Saturn inflight photographic instrumentation system to some extent is a continuation and refinement of the Redstone Development program. The need for optical instrumentation to provide visual analysis of flight phenomena unobtainable by other methods became evident early in the Saturn development program. (figure 1)

Inclusion of motion picture and television cameras on SA-1 was first proposed by DSLA Memo No. 303, April 25, 1960, Proposal for Movie and TV Cameras on the S-I Stage of SA-1. This proposal outlined a plan for movie camera and TV camera coverage for the SA-1 vehicle. The desired objective was to develop a camera capsule using previously developed and proven components and to modify these components to meet the inflight camera instrumentation requirements. However, limited time and funds precluded camera or TV coverage for the SA-1 vehicle. Effort was therefore directed toward the development of a more complete program for incorporation on later vehicles.

In September 1961, a memorandum, Agreement of Inflight Film Camera Instrumentation, established the joint responsibilities of the Astrionics Laboratory and Propulsion and Vehicle Engineering Laboratory of MSFC for film camera instrumentation aboard NASA vehicles under MSFC cognizance. A joint feasibility study was completed and a new proposal for inflight camera instrumentation was presented. The inflight camera instrumentation project was approved for SA-5 on October 6, 1961. Cook Technological Center, a division of Cook Electric Company, Chicago, Illinois, was selected as the contractor for the design, development, and fabrication of ejectable recoverable camera capsules under Contract NAS8-2516.

This Internal Note covers the development of the Saturn inflight instrumentation program from its original measurement requirement concept to the flight hardware application on Saturn vehicles. A comprehensive description of the inflight photographic instrumentation system is presented, along with information covering testing, operation, application, and evaluation of the system after recovery.

B. INFLIGHT PHOTOGRAPHIC INSTRUMENTATION SYSTEM

The Saturn IB, inflight photographic instrumentation system was designed to provide a permanent visual record for future study of activities aboard SA-5, 6,

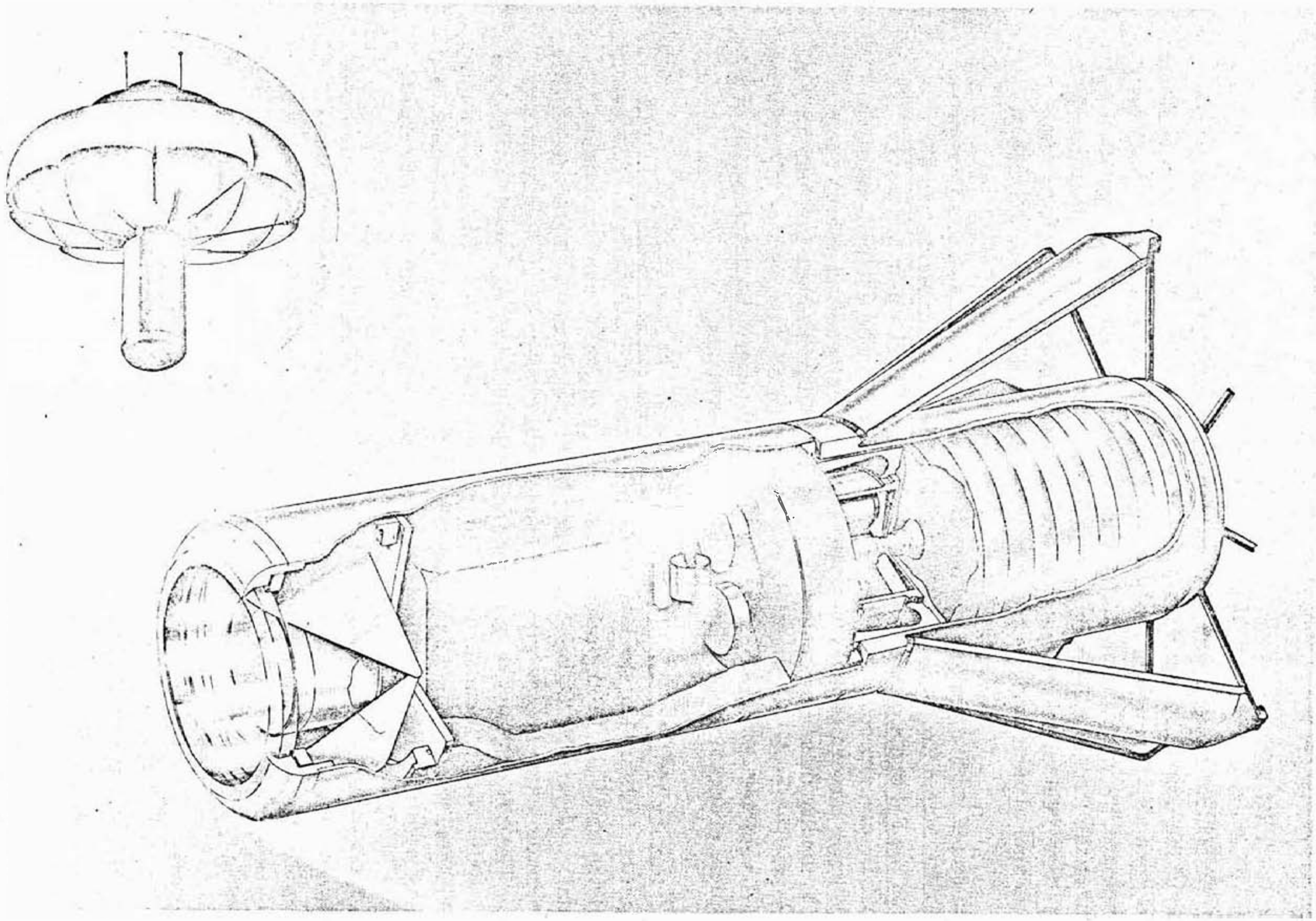

FIGURE 1. RECOVERABLE CAMERA CAPSULE

and 7. The cameras obtain a history of inflight vehicle behavior by monitoring preselected functions and events, thereby providing visual data for analysis of events that cannot be economically simulated by ground testing or recorded through telemetry techniques. Figure 2 shows the locations of the eight motion picture cameras and the other major components. Table 1 is a listing of the events filmed by these cameras.

An environmentally conditioned capsule houses each camera during the S-I stage powered flight, capsule ejection, reentry, impact, and saltwater immersion. The capsule is divided into three compartments: a lens compartment housing the camera lens and quartz viewing window; a camera compartment housing the camera and control unit assembly; and a recovery compartment housing a paraballon assembly, a radio assembly, a light beacon assembly, and other recovery aids. Each capsule has stabilization flaps that open at ejection to stabilize the capsule during its descent. Two configurations of the camera capsule are used: model A and model B. The two capsule models are very similar, differing mainly in size of viewing window and position of electrical disconnect assembly.

The four model A camera capsules, containing cameras 1, 2, 3, and 4, are installed in model A ejection tubes with the camera lenses upright for filming forward along the external areas of the Saturn vehicle. The four model B camera capsules, containing cameras 5, 6, 7, and 8, are installed in the inverted position in model B tubes to facilitate the filming of events inside LOX containers O-C and O-3 and in the S-IVB aft interstage areas. The camera capsule ejection tube mechanisms are mounted in pairs around the S-I stage spider beam at each of the four fin positions as shown in figure 1. Each pair consists of a model A and model B ejection mechanism.

Fiberoptic bundles with lenses on each end transmit the image from the vehicle interior to the model B capsules on the spider beam. An incandescent light illuminates the interior of each LOX container that is filmed, and two incandescent lights and a strobe system illuminate the interstage area prior to separation.

All eight camera capsules are programmed to be ejected from the vehicle by the capsule ejection system approximately 25 seconds after stage separation. Ejection velocity for the cameras is 8 m/s relative to the S-I stage. Predictions based on the 8-engine SA-5 flight showed that vehicle speed is 2,600 m/s, altitude is 100 km, and range is 150 km at the time of capsule ejection. The camera capsule reaches an apogee of approximately 175 km. Upon reentry into the atmosphere, at approximately Mach 10, the capsule assumes a stable attitude. The opening of the paraballon and ribbon parachute causes the capsule to decelerate to

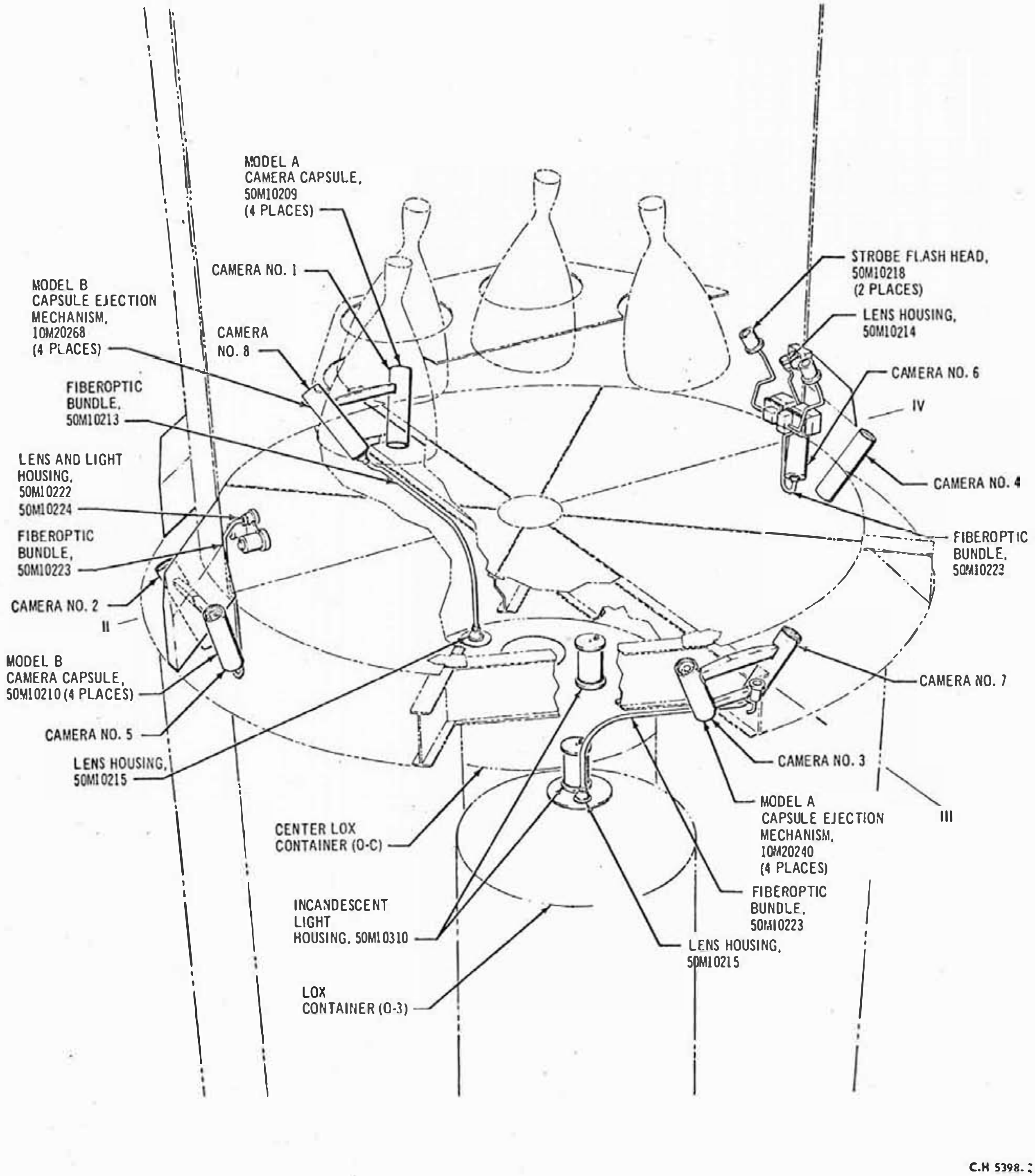

FIGURE 2. INFLIGHT PHOTOGRAPHIC INSTRUMENTATION SYSTEM

a terminal velocity of approximately 60 ft/sec. The paraballon keeps the capsule afloat after impact, while electronic and visual recovery aids guide recovery teams to the capsule.

TABLE 1. EVENTS COVERED BY FILM CAMERAS

Event	Camera Coverage							
	1	2	3	4	5	6	7	8
LOX motion in center LOX container (O-C)								x
LOX motion in outboard LOX container (O-3)							x	
Interstage ports opening	x	x	x	x	x*			
S-IVB engine chilldown					x			
LOX-SOX disposal					x			
Ullage rocket ignition	x	x	x	x				
Stage separation	x	x	x	x		x*		
Retrorocket ignition	x	x	x	x				
S-IVB engine ignition						x		
S-IVB engine initial burning	x	x	x	x		x		
Booster tumbling rate	x	x	x	x		x		

*Limited coverage

SECTION II. CAMERA AND OPTICAL AIDS

A. CAMERA

The cameras in the photographic instrumentation system are modified off-the-shelf, lightweight, 16-mm units manufactured by the D. B. Milliken Company. These cameras are designated DBM-3A RFI, MSFC drawing number 50M10221 (figure 3). The camera has an aluminum alloy casing, has a mass of 3.39 kg, and is 10 cm wide, 3 cm high, and 16.8 cm long (without lens).

The camera is mounted in the central compartment of the camera capsule. It is held rigidly in position inside the capsule by a beveled aluminum dovetail plate on the base of the camera housing, which mates with a plate on the interior capsule wall, and by the capsule lens compartment bulkhead. The two-speed camera motor, which operates on 28 Vdc from the S-I stage electrical system, is preset to the required film frame rate. The film magazine has a capacity of

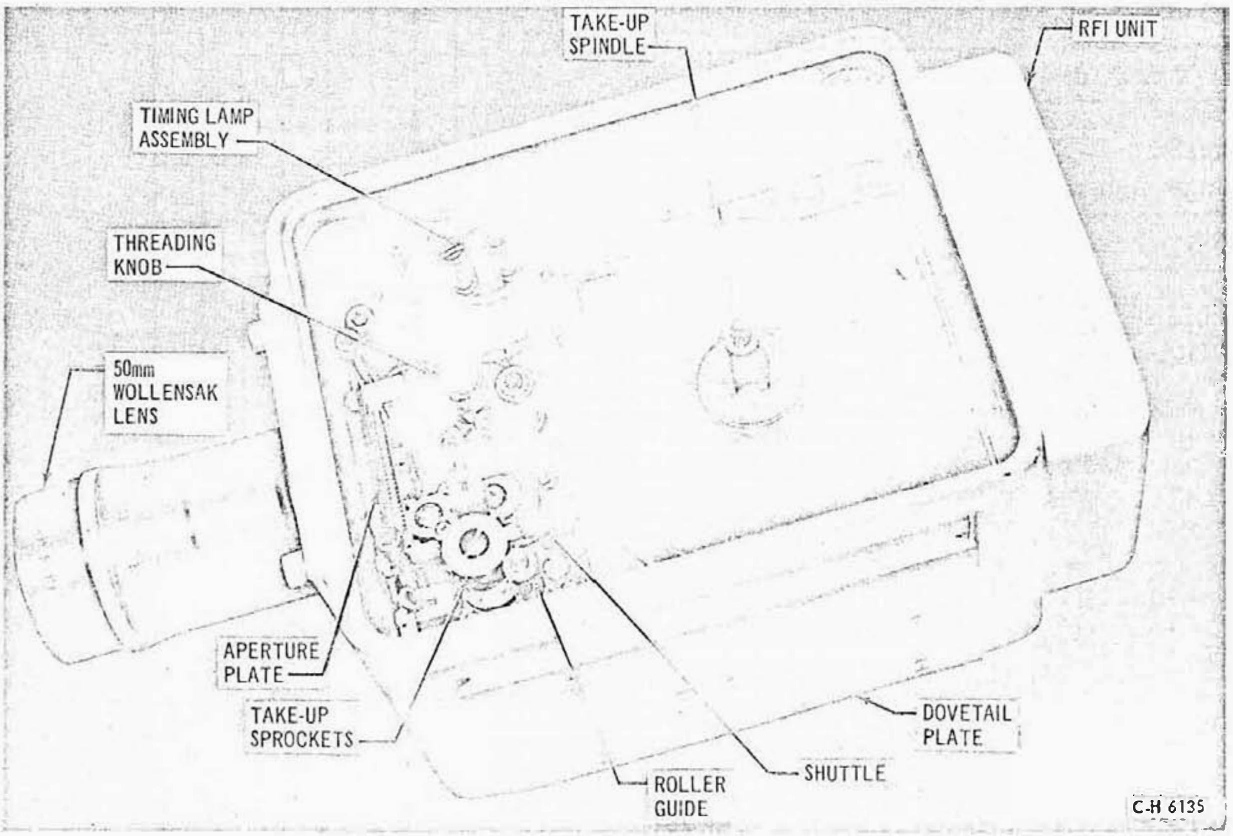

FIGURE 3. DBM-3A RFI CAMERA

30.5 m, providing 330, 164, or 64 s coverage at 12, 24, or 64 frames per second (fps), respectively. Two neon lights in each camera are controlled by a timer located on the spider beam in line with fin IV. This timer, described in Section II.B, supplies 115 Vac to the neon timing lights. The neon timing lights mark flight time and the time of major events on the camera film. The heater is thermostatically controlled and maintains a temperature of 27° to 40° C within the camera prior to launch to prevent film brittleness. The DBM-3A RFI camera schematic is shown in figure 4.

A radio frequency interference (RFI) filter unit attached to the back of the camera housing minimizes electrical noise on the camera circuits. Components within the unit are shielded with aluminum mesh, which, in conjunction with the filter, prevents radiated and conducted RFI.

A frame rate converter indicates through telemetry signals if the camera is running and at what speed. Frequency pulses from the sync generator in the camera are converted to a dc signal proportional to frequency for modulating telemetry subcarrier oscillators.

The standard single module frame rate converter is mounted in the measuring rack on the S-I stage. The two printed circuit boards with polyurethane-potted components are contained in an aluminum alloy casing. The required 28 Vdc power is supplied by the vehicle electrical system.

B. TIMER

Timing pulses on the film in all eight cameras are controlled by the timer located on the spider I-beam in line with fin IV. The timer operates on 28 Vdc, but is regulated by 115 Vac, 400 Hz power from the vehicle inverter. Accuracy of this system is within 0.012 Hz.

The timer, which is housed in an aluminum shell, contains eight transistorized printed circuit boards, each coated with polyurethane (figure 5). Seven boards make up the binary code system, and one board is used for the flight event marker.

The timer converts the 400 Hz signal from the vehicle 400 Hz inverter into 10 Hz coded signals. Each 10 Hz signal is amplified and activates one of the two neon lamps (LT2-24-IR) mounted on each camera wall interior. One lamp (A) marks binary coded time, and the other (B) marks vehicle flight events on the film. The lamp A signal registers flight time on one edge of the film, providing timed sequential documentation of general conditions during vehicle flight. Lamp B marks selected flight events on the opposite edge of the film. Sensing elements

S3	CAM-OPERATED SWITCH
S2	THERMO SWITCH
R7	12K OHM
R3,4,5,6	3.6K OHM .5 WATT
R2	200 OHM 25 WATT
R1	2 OHM 25 WATT
NE1,2	NE2JL
L3	ROTOR
L2	STATOR
L1	PICKUP COIL
H1	HEATER, 100 WATT
CR3	750 MA. 400 VOLT
C6	0.01 μF 200 VOLT
C5	0.02 μF 150 VOLT
C4	0.01 μF 150 VOLT
C3	.0 μF 100 VOLT

* ALL CAMERAS EXCEPT CAMERA 6 (STROBE CAMERA, FIN LINE IV).
** CAMERA 6 ONLY.

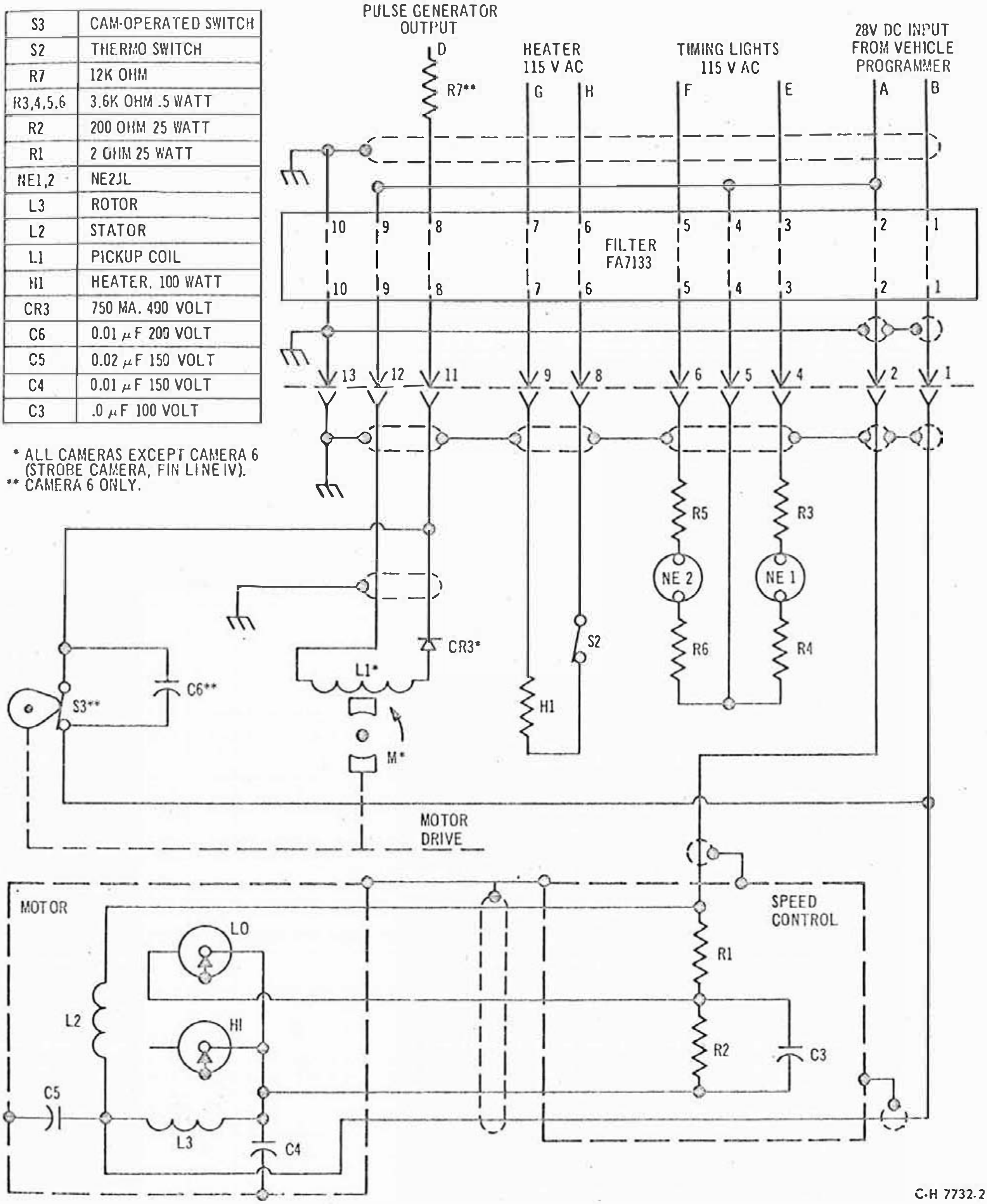

FIGURE 4. DBM-3A RFI CAMERA--SCHEMATIC

energize relays that switch a signal to the timer to release a pulse to lamp B at liftoff, liquid hydrogen cooldown, S-I stage inboard engine cutoff, and stage separation.

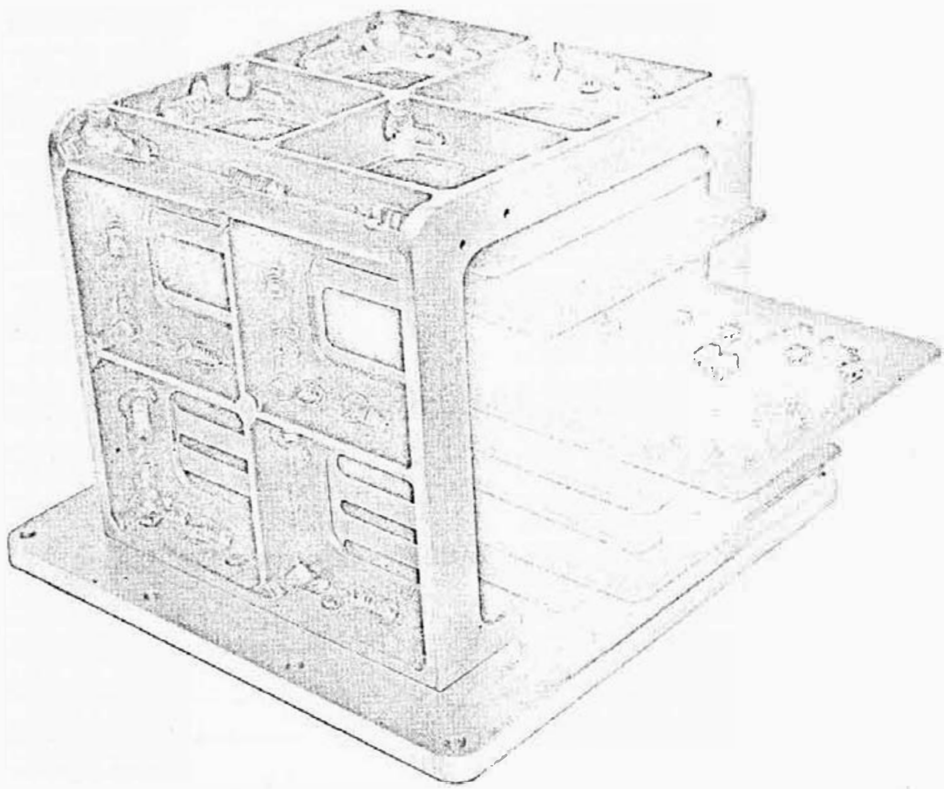

FIGURE 5. TIMER, INTERNAL VIEW

C. FIBEROPTIC BUNDLE

Fiberoptic bundles and lenses (Section II. D.) allow the cameras to film events inside the vehicle even though the capsules are located on the outer ends of the spider beam for easy ejection. The fiberoptic bundles, which can be bent during installation, provide one-to-one image transfer from the interstage area or the LOX containers to the cameras. The four bundles used are prefocused multi-fiber optical bundles encased in a hose of stainless steel braided over a Teflon core (figure 6). Resolution of each bundle is better than that of standard television. Tests conducted on a typical bundle having a resolution of 56 lines/mm vertically and 56 lines/mm horizontally.

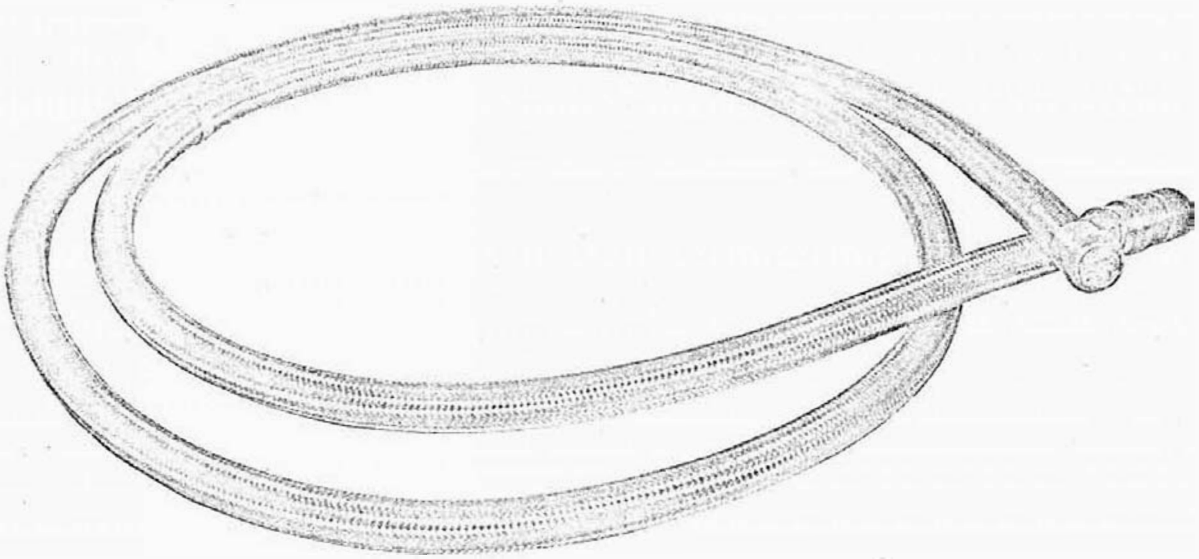

FIGURE 6. FIBEROPTIC BUNDLE C-H 6133-1

D. LENSES

The four types of lens used in the inflight photographic instrumentation system are pictured in figure 7.

1. Model A Capsules

A model 725 Pacific Optical Company Periphoto 4.2 mm lens is installed in each of the cameras for the four model A capsules. This lens has a 1.92 radian (rad) field of view. The Periphoto lens is in an aluminum alloy casing mounted to the camera face plate.

2. Model B Capsules

A 50 mm Wollensak lens is mounted on each of the cameras in the four model B capsules, and a 10 mm Angenieux or a 5.7 mm Kinoptic lens is mounted on the viewing end of the fiberoptic bundles.

The image formed at the camera end of the fiberoptic bundle must be transferred to the camera film. This is accomplished by a 50 mm Wollensak coupling lens system. One lens is attached to the end of the fiberoptic bundle and couples through the quartz capsule window to another 50 mm Wollensak lens attached to the camera face plate. This combination of lenses is more efficient than a single lens, and the distance between the two lenses is not critical.

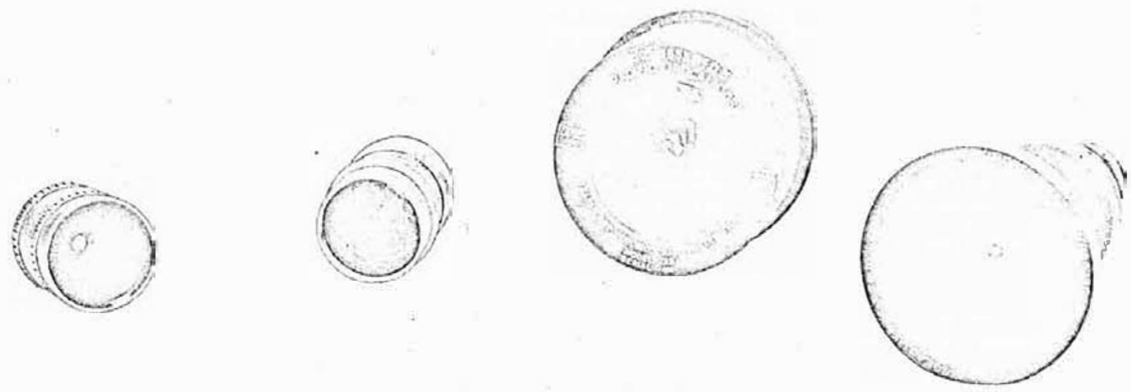

FIGURE 7. LENSES

The Wollensak lenses function as two thin convex lenses to provide a 1.1 inverted image on the camera film. As shown in figure 8, the object (image formed on the end of the fiberopic bundle) is at the focal plane of the first lens, lens A, so that all the rays from any particular point on the object are parallel after traveling through lens A. Parallel rays entering the second lens, lens B, converge at the focal plane of lens B to form the inverted image at the focal plane (camera film). Because of the symmetry of this system, it is free of distortion, and spherical aberration is at a minimum.

On the viewing end of the fiberoptic bundle, either of two different types of lens may be housed. With the fiberoptic bundle servicing camera number 6 (fin line IV), a 5.7 mm Kinoptic lens with 1.46 rad circular coverage is used. (However, the field of view of the camera is 1.5 rad vertical.) To each of the other three fiberoptic bundles is attached a 10 mm Angenieux lens that gives 0.887 rad horizontal and 0.713 rad vertical coverage. The housings for these lenses are discussed in Section II. G.

E. FILM

Cameras number 1 through 7 use color film, and camera number 8 used black and white. Color film is used where possible because it presents an element of depth to the image, providing a better three-dimensional effect than the gray tones of black and white photography. Black and white film must be used for camera

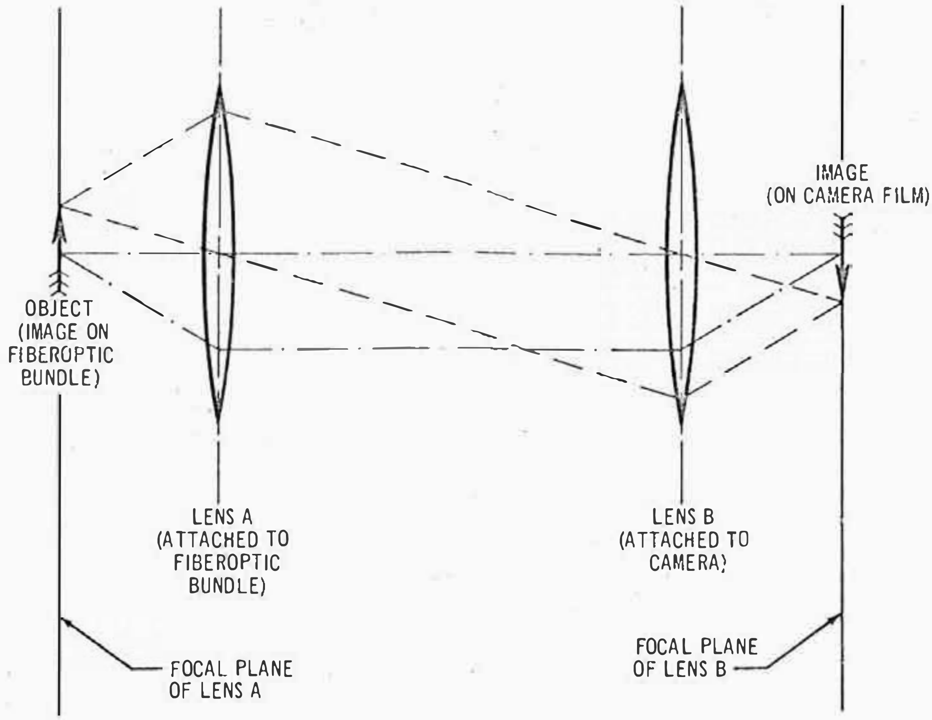

FIGURE 8. COUPLING LENS SYSTEM

number 8 because it films events in the center LOX container with the aid of a 3.8 m fiberoptic bundle and an incandescent light. The light attenuation of the larger fiber-optic bundle is too great (approximately 75 percent) for color film; however, color film can be used for the three cameras using the 1.8 m fiberoptic bundles (light attenuation approximately 50 percent) and for the four exterior viewing cameras.

Kodak Ektachrome 7255 Commercial film is loaded in the cameras for the four model A capsules. This film provides a low-contrast original from which color release prints of good projection quality can be made. Contrast is purposely low, so that the print contrast is comparable to that of a good original made on daylight film. The Ektachrome 7255 film in these four cameras is exposed at the rate of 64 fps.

Interior viewing cameras 5, 6, and 7 in the model B capsules use Kodak Ektachrome medium speed film. This product, created for aerospace use, has special development properties so that an overexposure of two stops or an under-exposure of four stops will still present an acceptable print. Cameras 6 and 7 operate at 24 fps; camera 5 is set at 64 fps.

For camera 8, black and white Kodak Royal-X Pan, an extremely fast panchromatic film, is used. Of medium contrast and coarse grain, Royal-X Pan is highly sensitive for extremely short exposure periods. Camera shutter rate for center LOX container viewing is 12 fps.

The exposure setting for each camera is determined by film type, camera location, artificial lighting conditions, predicted vehicle trajectory, and predicted weather conditions. The shutter speed and aperture are preset before launch. The exposure settings for the SA-5 cameras are shown in table 2.

F. LIGHTING

The areas to be photographed during a Saturn flight normally have different and variable lighting conditions:

(a) Constant direct sunlight - external areas.

(b) Constant light reflected from the earth - external area on one side of the vehicle.

(c) Constant total darkness - LOX container interiors.

(d) Intermittent flash or engine-flash illumination followed by direct sunlight after separation - interstage.

Categories (a) and (b) require no supplementary control of light and require only that the lens setting for cameras 1, 2, 3, and 4 be based on anticipated sunlight conditions and anticipated intensity of light reflected from the earth during flight above the atmosphere. Category (c) has a lighting problem solved by installation of one 250 W lamp on each LOX container that is photographed. Category (d), the interstage, presents the most interesting problem. The two cameras viewing the interstage, photograph events that are in direct sunlight after stage separation and they must be preset to these conditions. Before staging, however, the interstage is in intermittent darkness. A lighting system that will provide light of approximately the intensity of sunlight is required for illumination of the events filmed in the interstage prior to separation. This is accomplished by two 150 W incandescent lamps and a strobe system. The incandescent lamps are installed at fin line II, and the strobe system is installed at fin line IV.

The strobe system, which operates on 28 Vdc power from the vehicle, includes two flash tubes, to provide an intense light flash synchronized with every fourth frame of the shutter in camera 6. The film from this camera is step-printed after recovery. (Each exposed frame is printed four times.) The strobe system, consisting of flash heads, power supplies, interconnecting cables, and pulse divider, provides light at an intensity approximately equal to the intensity of sunlight. The

TABLE 2. CAMERA EXPOSURE SETTINGS FOR SA-5

					Exposure	
Camera No.	Frames per second	Objective Lens	Illumination	Film	Aperture Angle (rad)	Aperture
1	64	110°	Daylight	Kodak ECO 7255	0.86	2
2	64	110°	Daylight	Kodak ECO 7255	0.86	2
2	64	110°	Daylight	Kodak ECO 7255	0.86	4
4	64	110°	Daylight	Kodak ECO 7255	0.86	4
5	24	10 mm on Fiber-optic bundle	Incandes-cent	Kodak MS430	2.8	1.5
6	64	10 mm on Fiber-optic bundle	Strobe, Daylight	Kodak MS430	0.26	1.8
7	24	10 mm on Fiber-optic bundle	Incandes-cent	Kodak MS430	2.8	1.5
8	12	10 mm on Fiber-optic bundle	Incandes-cent	Kodak Royal-X	2.8	1.5

system requires one-third the power that would be necessary for a steadily-burning incandescent light providing the same intensity.

1. Incandescent Lights

Sylvania DXM 30 V, 250 W bulbs are used in the LOX containers. These lights operate on 28 Vdc through a power connector on each light housing and are turned on just prior to liftoff. The two incandescent lights mounted in the interstage at fin line II are General Electric 1959 quartz bulb, 150 W, 24 Vdc tracking lamps. These also operate on 28 Vdc from the vehicle electrical system through power connectors in their housings.

The envelope of the G.E. lamp is filled with halogen gas, which reacts with the tungsten burned off the filament. The resulting compound is then deposited on the hot filament where the halogen gas is released, leaving the metal on the filament. This cyclic action prevents the buildup of metal deposits on the glass, giving longer lasting high efficiency and a slight increase in life to the lamp.

The housings and mounting fixtures for these lights are described in Section II.G.

2. Strobe System

The strobe system has two flash heads installed on either side of fin line IV on the wall of the interstage, a power supply installed below each flash head, an interconnecting cable between each power supply and flash head, and a pulse divider located on the spider beam at fin line IV. (Refer to figure 9.) The system must have two inputs to produce a flash: 28 Vdc directly from the vehicle electrical system (turned on just prior to liftoff), and the output from the pulse generator in camera 6. The system provides an intense light flash synchronized with every fourth frame of the camera.

a. Component Description. The flash head contains a cold cathode flash tube, a discharge capacitor, and a voltage amplifier (figure 10). The flash tube is filled with silicone oil and GN_2 through a hole drilled in the unit. The hole is resealed at atmospheric pressure. A pressure switch, installed on the flash tube to protect the tube from prolonged operation, is connected to the power supply circuit so that the power to the flash tube is interrupted when the heat expands the oil and gas enough to actuate the switch. An aluminum honeycomb screen, placed at the front of the flash head, is an effective RFI shield while reducing light intensity only 5 percent. The use of this type of RFI shield was developed by the Dynamics Unit of the MSFC Astrionics Laboratory (R-ASTR-IMS).

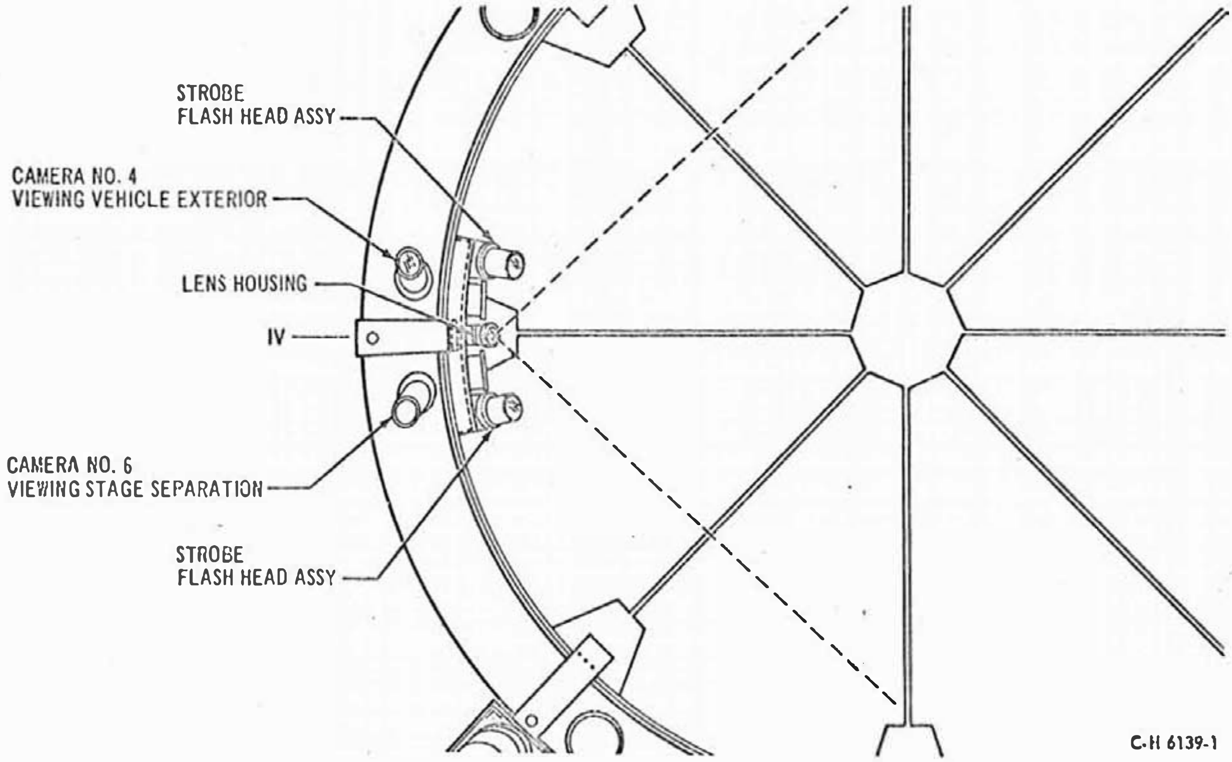

FIGURE 9. LOCATION OF STROBE FLASH HEADS

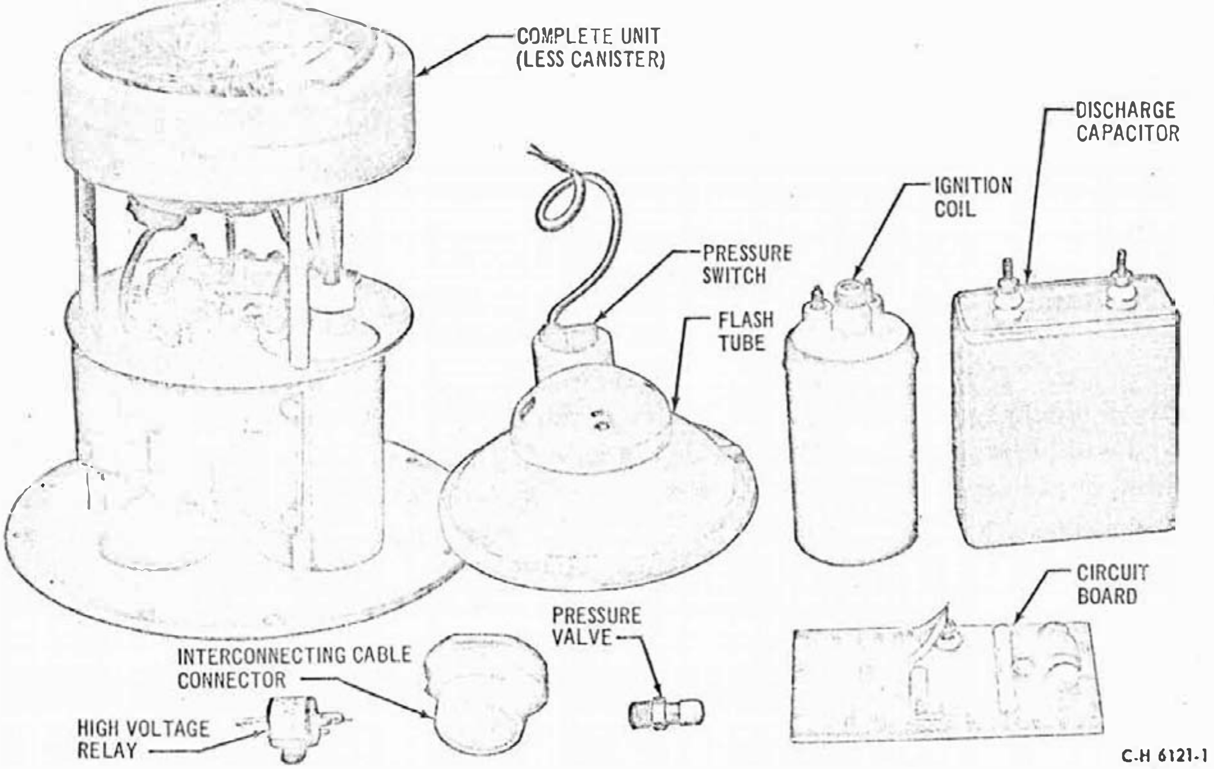

FIGURE 10. FLASH HEAD, DISASSEMBLED

The strobe system power supply is made up of a trigger amplifier, oscillator, power amplifier, saturable reactor, and high voltage transformer rectifier. These components are housed in a container of aluminum alloy (figure 11).

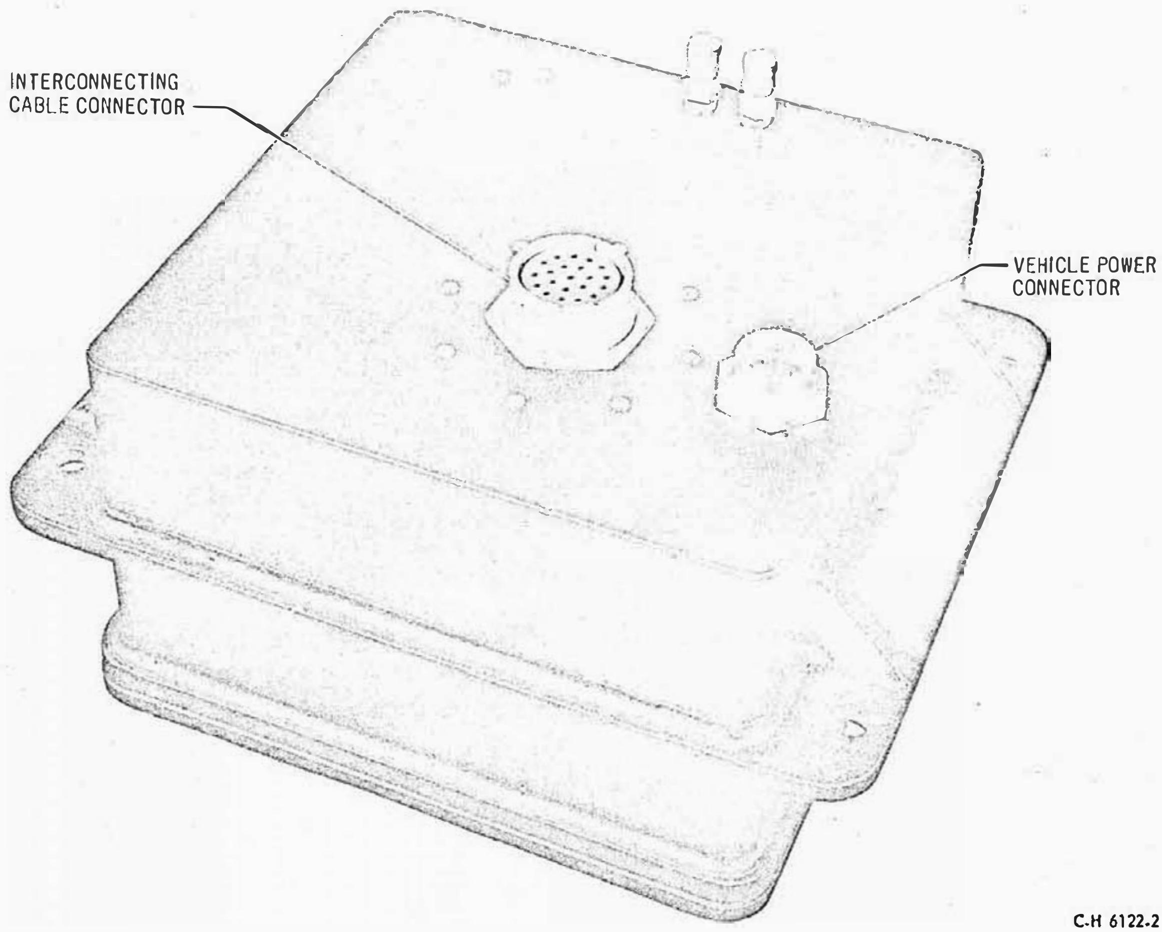

FIGURE 11. POWER SUPPLY

A flexible hydraulic hose with electrical connectors and housings at its ends (figure 12), connects each power supply and flash head. The hose is used as a pneumatic pressure container and RFI shield for the electrical cables that join these components. The cable provides metal-to-metal contact between the power supply and flash head to prevent RFI through the cable connectors. Each power supply, flash head, and cable is sealed at atmospheric pressure. O-rings at each cable connection maintain pressurization during flight.

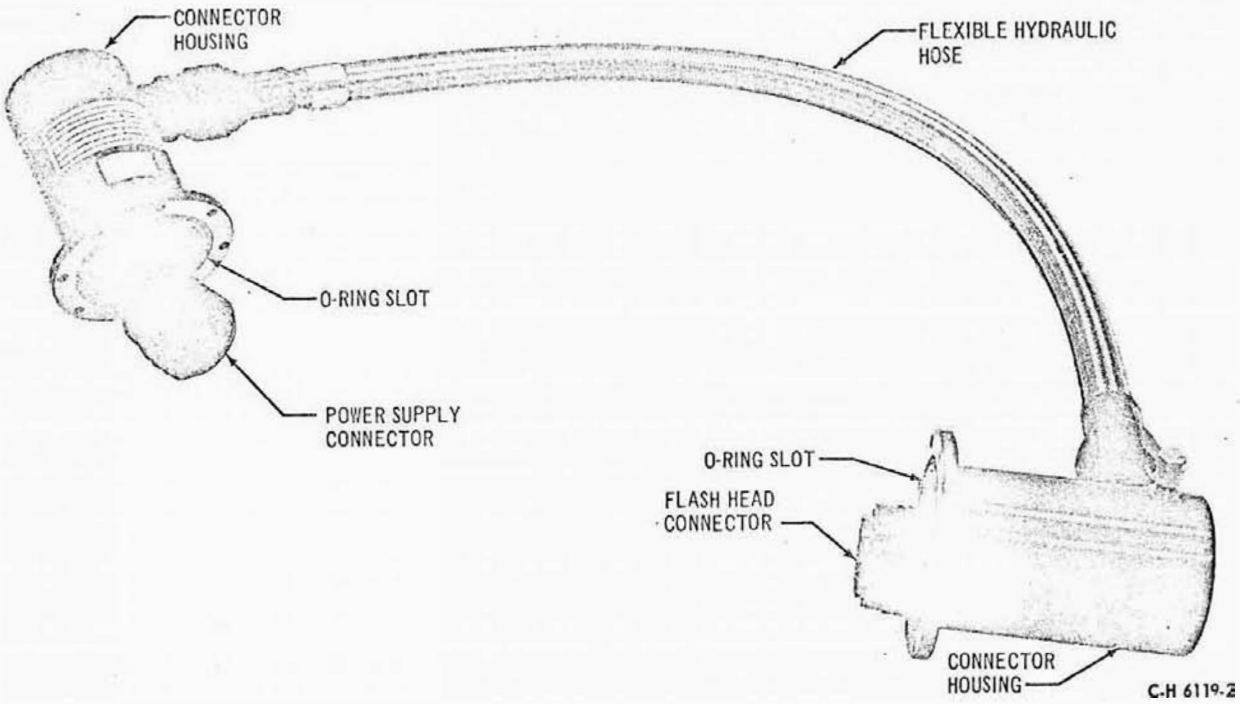

FIGURE 12. INTERCONNECTING CABLE

The pulse divider (figure 13) amplifies the signals from the camera pulse generator, and divides the frequency by four, providing an output pulse that is synchronized with every fourth opening of the shutter in camera 6.

The pulse divider consists of two printed circuit boards with transistorized circuitry potted with a clear, silicone rubber material. The use of two facing rectangular boards eliminates cross wiring. Housed in a 0.79 cm aluminum alloy shell, the pulse divider is mounted on the spider I-beam at fin line IV.

b. Principle of Operation. The operating principle of the strobe system is discussed in relation to the block diagram shown in figure 14. The main circuits are made up of the following components:

POWER CIRCUIT

1. Input filter
2. Oscillator
3. Power amplifier
4. Saturable reactor
5. High-voltage transformer
6. Rectifier

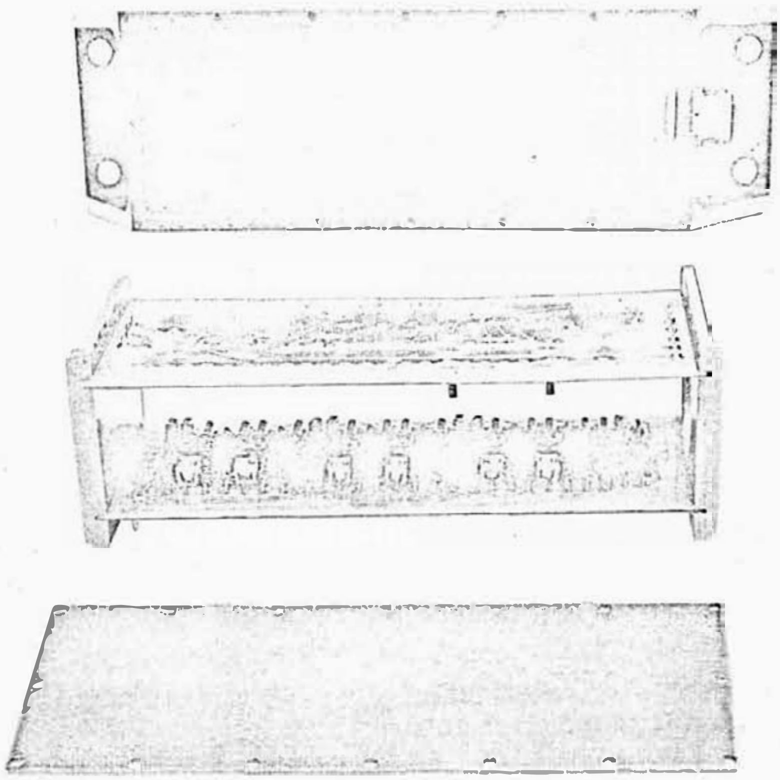

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FIGURE 13. PULSE DIVIDER

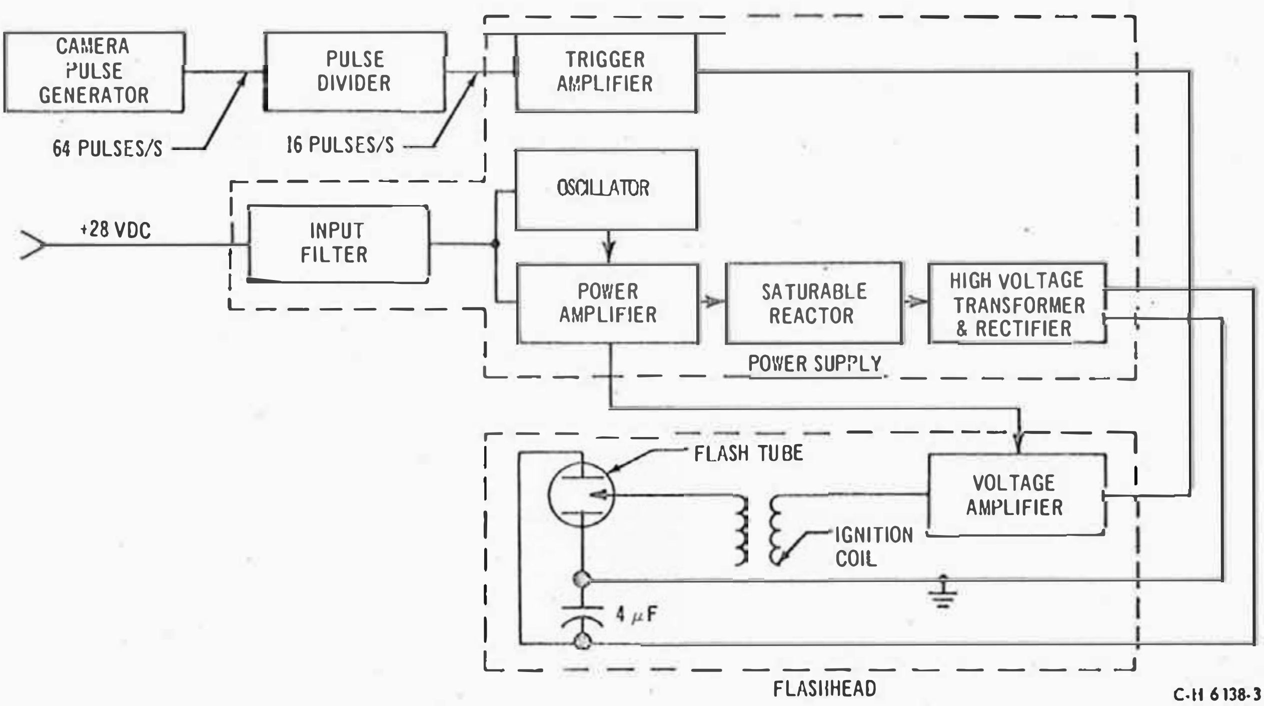

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FIGURE 14. STROBE SYSTEM--BLOCK DIAGRAM

TRIGGER CIRCUIT

1. Synchronization pulse generator (on camera 6)
2. Trigger amplifier (in power supply)
3. Voltage amplifier (in flash head)
4. Ignition coil (in flash head)

(1) Power Circuit Functions. A potential of 28 Vdc is applied through the input filter to the oscillator and power amplifier. The oscillator power drives the bases of the power amplifier transistors. The power amplifier output goes to the primary winding of the trigger coil and to a 10 kΩ resistor to charge a 1 μF capacitor in the voltage amplifier to 200 V.

Another voltage from the power amplifier is applied to the saturable reactor, which in turn applies voltage to the high voltage transformer and rectifier. The power supply delivers 3000 V output under static conditions and approximately 2400 V during operation. The saturable reactor maintains a constant current flow with very little power loss, while the transformer output voltage varies from to 2400 V. This current is rectified and passed directly to the 4 μF discharge capacitor in the flash head.

(2) Camera Shutter, Pulse Divider, and Trigger Circuit Function. Operation of the camera shutter causes the cam-actuated switch in the synchronization pulse generator in camera 6 to make and break intermittently, applying 28 V from the camera circuit to the trigger circuit at the rate of 64 pulses per second (pps). The pulse generator is illustrated schematically in figure 3. A voltage divider reduces the 28 V to between 2.5 and 4.0 V, and the pulse divider (figure 13) counts down 4 to 1, providing a 16 pps signal. The pulse divider also increases the voltage to between 4 and 8 V.

The trigger circuit shapes and amplifies the synchronization pulse. To do this, the trigger amplifier increases the power without increasing the voltage. This increased power is applied to the voltage amplifier in the form of negative pulses. The voltage amplifier inverts the signal polarity; and boosts the pulse to approximately 20,000 V by means of 1 μF capacitor discharging through the primary winding of the high voltage ignition coil in the flashhead. (The 1 μF capacitor is previously charged to 200 V by the power amplifier. The trigger amplifier causes a silicon rectifier to conduct, discharging the 200 V into the transformer ignition coil.)

The 20,000 V pulse applied to the flash tube (previously charged to 2400 V) causes ionization in the tube. Ionization allows the 4 μF capacitor to discharge through the flash tube to produce a high-intensity light flash.

G. SUPPORT STRUCTURE

1. Fiberoptic Bundle and Coupling Lenses

An optical housing and end plate assembly is attached to the forward end of each model B camera capsule ejection tube to house the 50 mm Wollensak coupling lens and spacer coupling (figures 15 and 16). The spacer coupling fits over the fiberoptic bundle end between the bundle and the Wollensak lens.

The unit is made of aluminum alloy with a black anodize finish. Eight screws lock the 20 cm diameter end plate on the ejection tube. Another eight screws hold the 10.8 cm diameter optical housing on the end plate. Two valve assemblies on the housing permit hand purging of the coupling lens with dry gaseous oxygen before installation, when the assembly is sealed at atmospheric pressure.

The optical housing and end plate assembly provide a rigid support for the coupling end of the fiberoptic bundle; the remainder of the bundle is held rigidly in place by clamps attached to the vehicle.

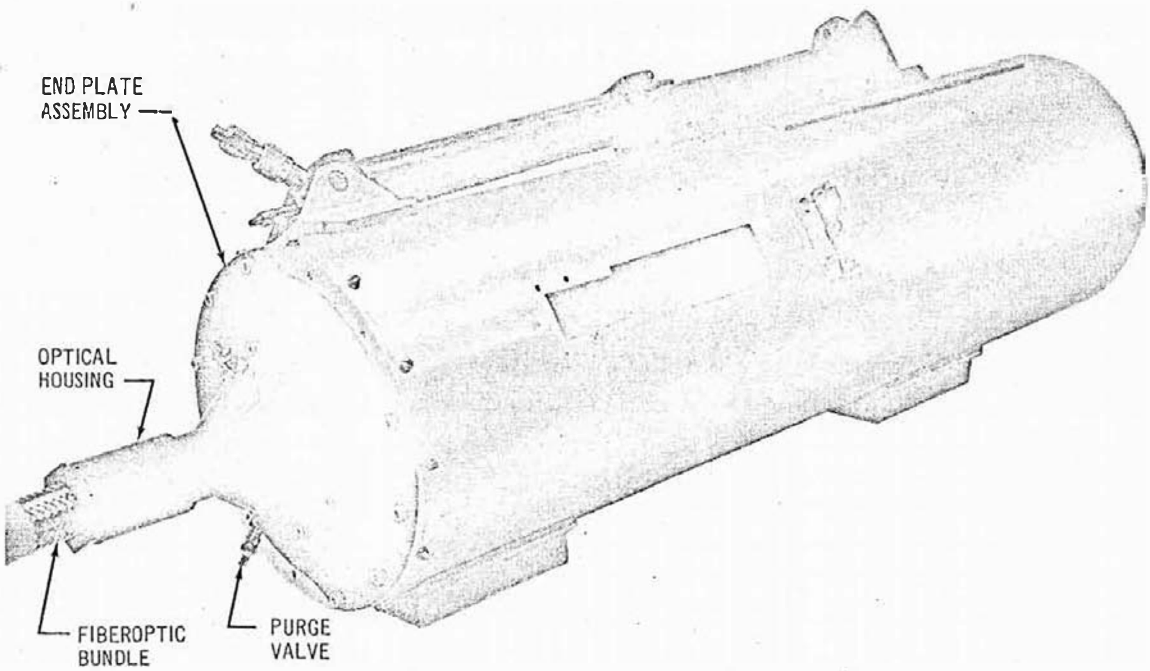

FIGURE 15. OPTICAL HOUSING AND END PLATE ATTACHED TO EJECTION TUBE C-H 6132.1

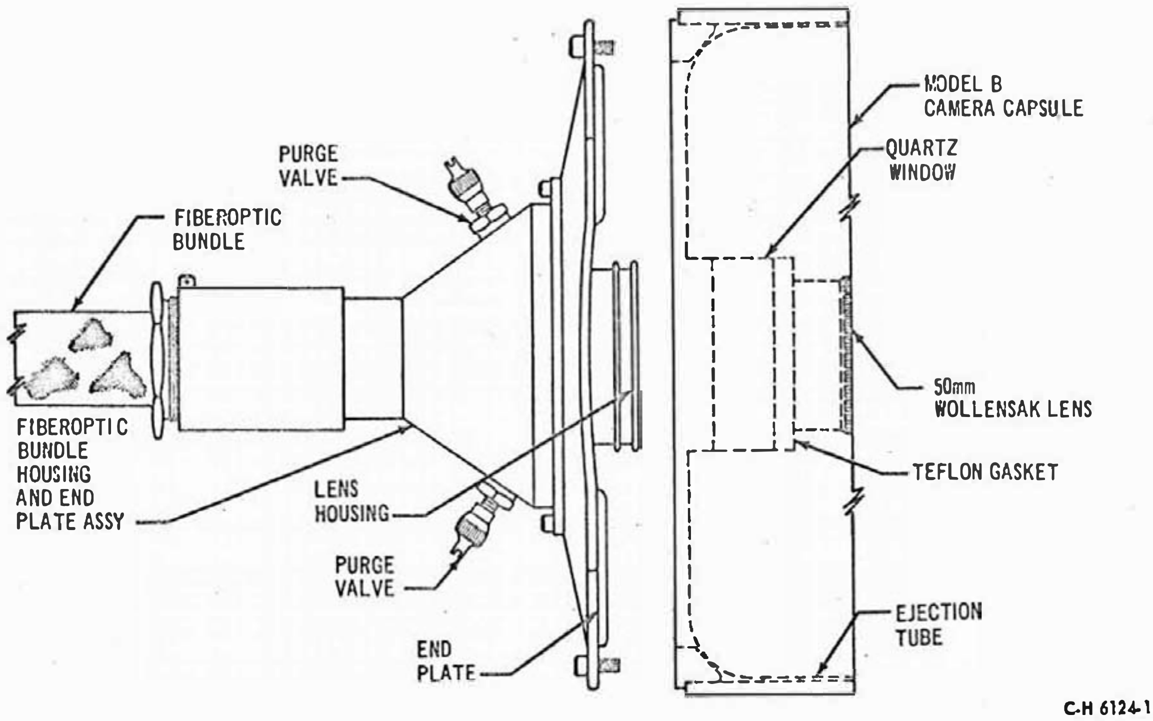

FIGURE 16. OPTICAL HOUSING AND END PLATE ASSEMBLY

2. LOX Containers

a. Lens and Light Housings. The Lox container lens and light housings provide:

(1) A rigid supporting structure for the lights and lenses.

(2) Protection against LOX container depressurization if the quartz windows on the LOX containers should break.

(3) A closed path for the GN_2 purge flow.

The housings for the two LOX containers are identical. The aluminum alloy light housing (figures 17 and 18) is a cylindrical canister 38 cm long and 16 cm diameter with a pyrex window. The lens housing (figure 19) is also of aluminum alloy, 8.6 cm at the largest diameter and 22.1 cm long when attached to the lens.

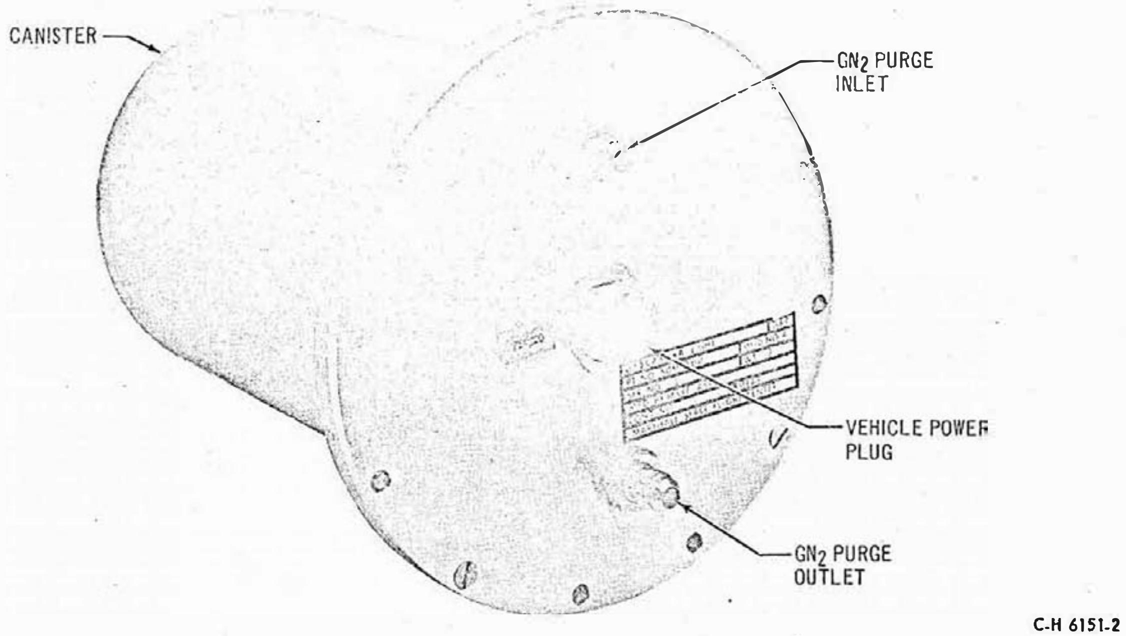

FIGURE 17. LOX CONTAINER LIGHT HOUSING

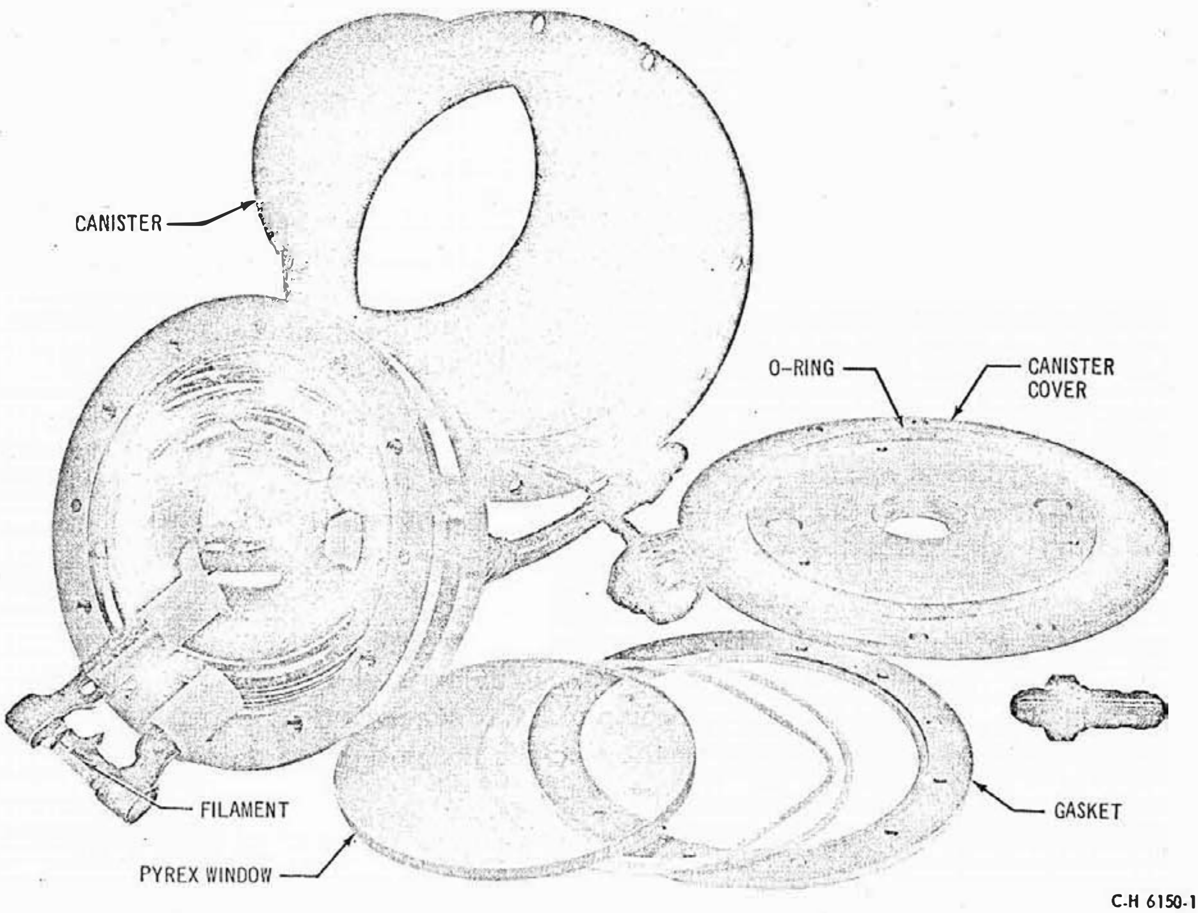

FIGURE 18. LOX CONTAINER LIGHT HOUSING, DISASSEMBLED

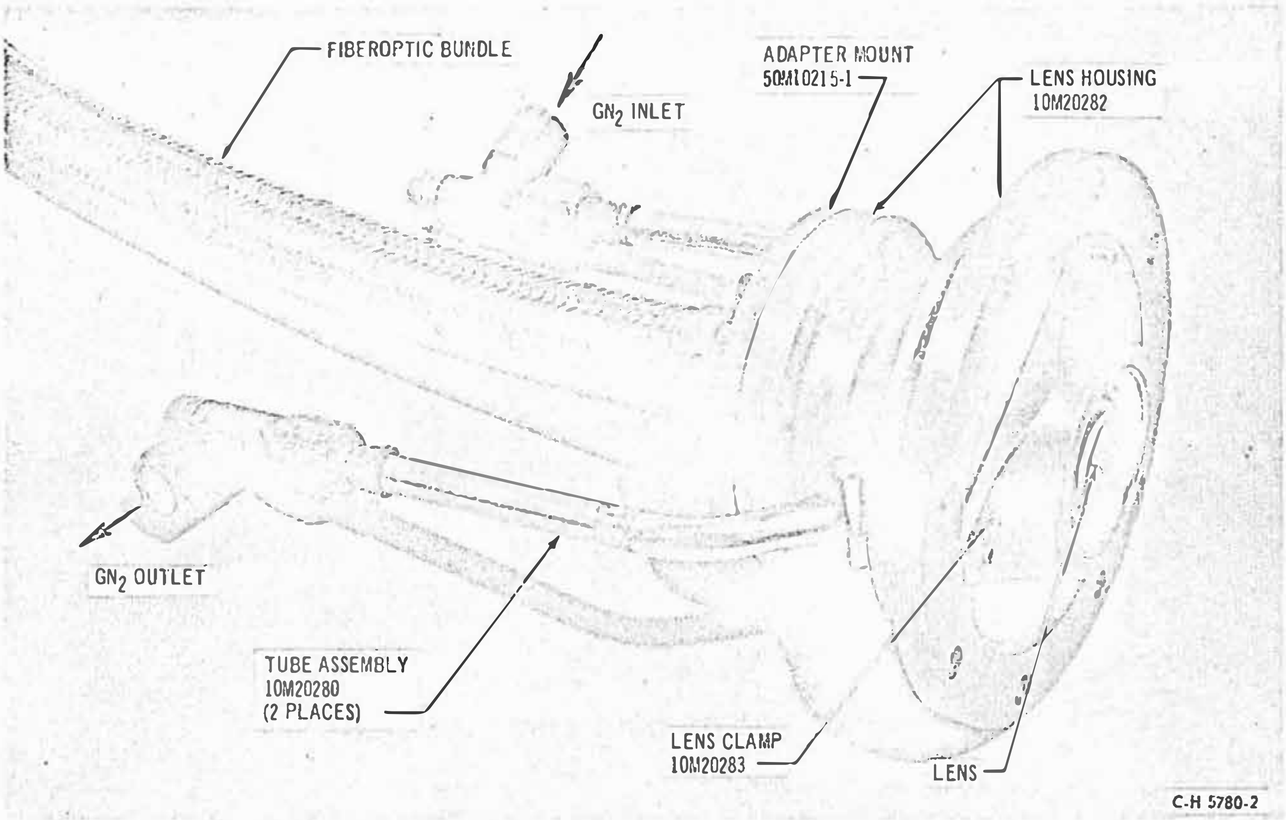

FIGURE 19. LOX CONTAINER LENS HOUSING

b. LOX Container Structures. The tops of the two LOX containers were redesigned to accommodate the lens and light housings. In container 0-3, the manhole cover was redesigned to contain two quartz windows (one window for camera viewing and one window to admit light), and the tapped holes for the bolts that attach the lens and lamp container assemblies to the manhole cover (figure 20).

Redesign of the center LOX container was more extensive because of the GOX distribution manifold on the top. Two wells, one on each side of the manifold, were necessary. Circular holes were cut on the projected container top, then fitted with flanged tubes that were welded in place.

c. Lens and Light Housing Purge System. The GN_2 purge system prevents moisture condensation on the LOX container window, lenses, lights, and reflecting surfaces. Such moisture would prohibit successful photographic viewing. Figure 21 shows a light and lens housing and GN_2 purge lines on the vehicle. Figure 22 is a schematic of the purge system.

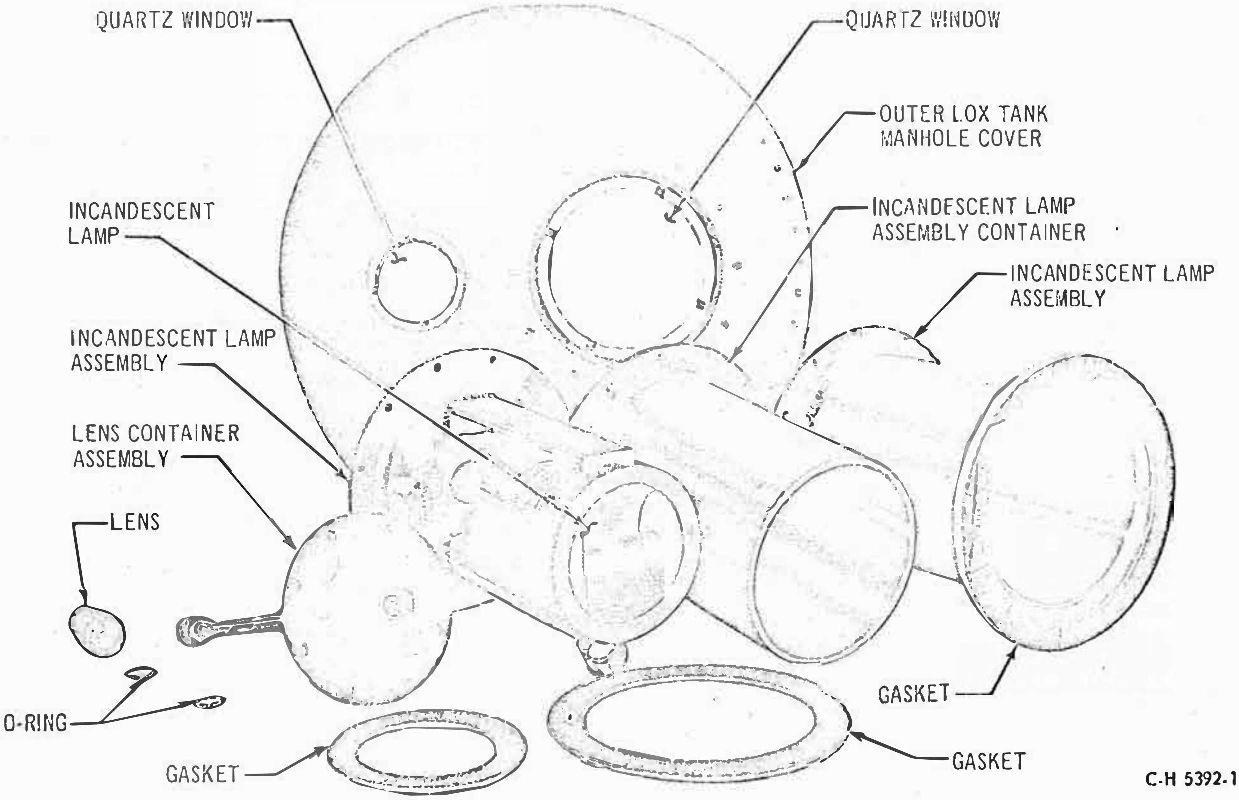

FIGURE 20. LOX CONTAINER LIGHT AND LENS HOUSING INSTALLATION

GN_2 at 4° to 22° C and 17.1 ± 3.4 N/cm^2 is supplied from the ground to the quick-disconnect coupling located on the fin III upper umbilical plate, starting 30 minutes prior to LOX tanking. The normally closed solenoid vent valve closes at liftoff to seal the system. During ground purge operations, GN_2 flows through the quick-disconnect coupling and enters branched feeder lines. Each line services a LOX container (O-C or O-3) optical assembly, passing the GN_2 first through the lens housing. After purging the light housing, the GN_2 enters the solenoid vent valve.

At liftoff, the check valve of the quick-disconnect coupling closes when the umbilical arm disconnects. Simultaneously, the solenoid vent valve is deenergized. This seals both ends of the circuit to a minimum pressure of 48 N/cm^2 and prevents loss of LOX container pressure if a quartz window should break.

3. Interstage

In the interstage, a lens and two incandescent lights are mounted at fin line II, and the strobe system and lens are at fin line IV.

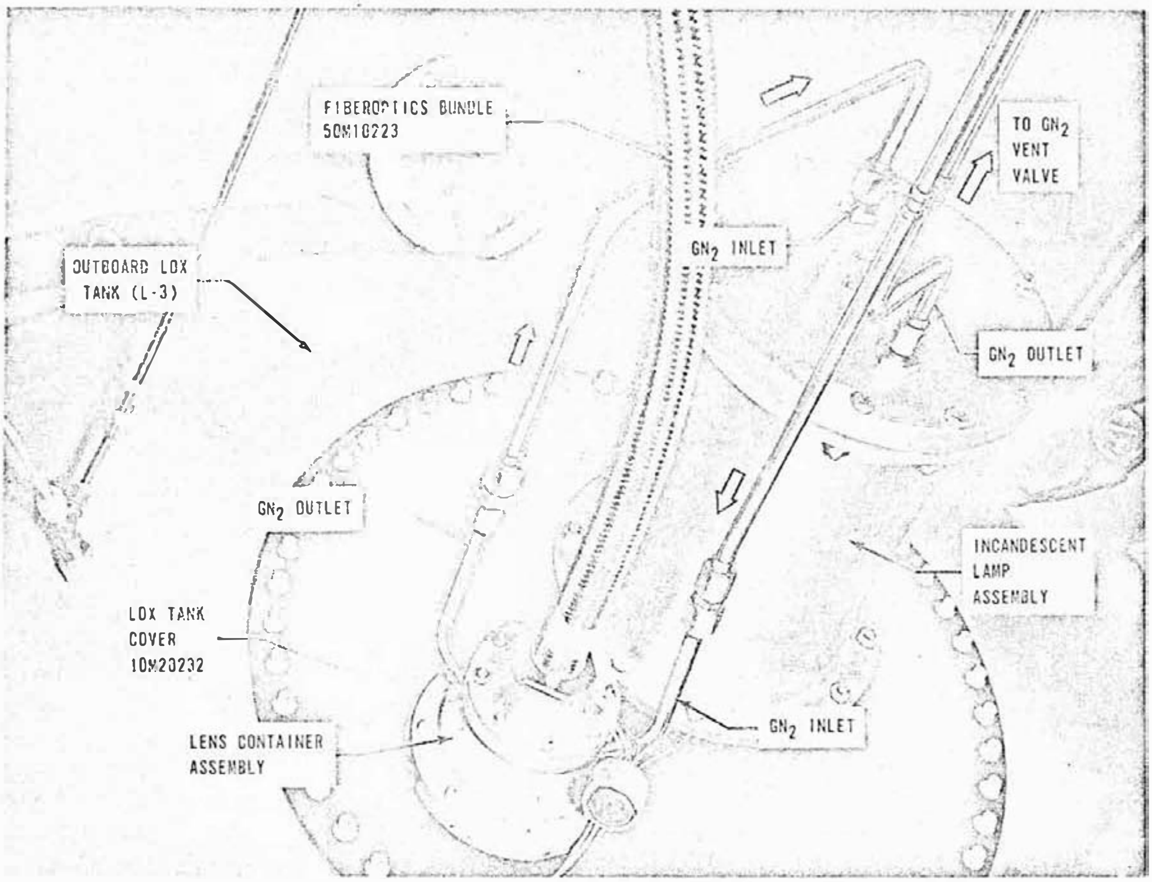

FIGURE 21. LOX CONTAINER LIGHT AND LENS HOUSING INSTALLED ON VEHICLE

The housings for the incandescent lights are aluminum alloy step canisters 12.1 cm at the larger diameter and 12.7 cm long. The window in the housing is translucent Teflon, sealed and unbreakable. The light attenuation of the Teflon is approximately the same as that of clear glass. (This application for Teflon was originated at R-ASTR-IMS.) The lens housing is of the same aluminum alloy as the light housing, and is 7.0 cm at the larger diameter and 10.2 cm long.

The light and lens housings are bolted to an adapter (figure 23) that attaches to a mounting bracket on the interstage wall.

The strobe system is installed about fin line IV. The two flash heads, each containing a discharge capacitor, ignition coil, circuit board, and flash tube (figure 10) are mounted on brackets on the interstage wall and are turned upward at a 0.698 rad angle. The aluminum alloy canisters, are 15.5 cm in diameter and 26 cm long (figure 24).

The power supplies are connected to the flash heads with flexible cables, and each is mounted in a bracket on the interstage wall below its respective flash head.

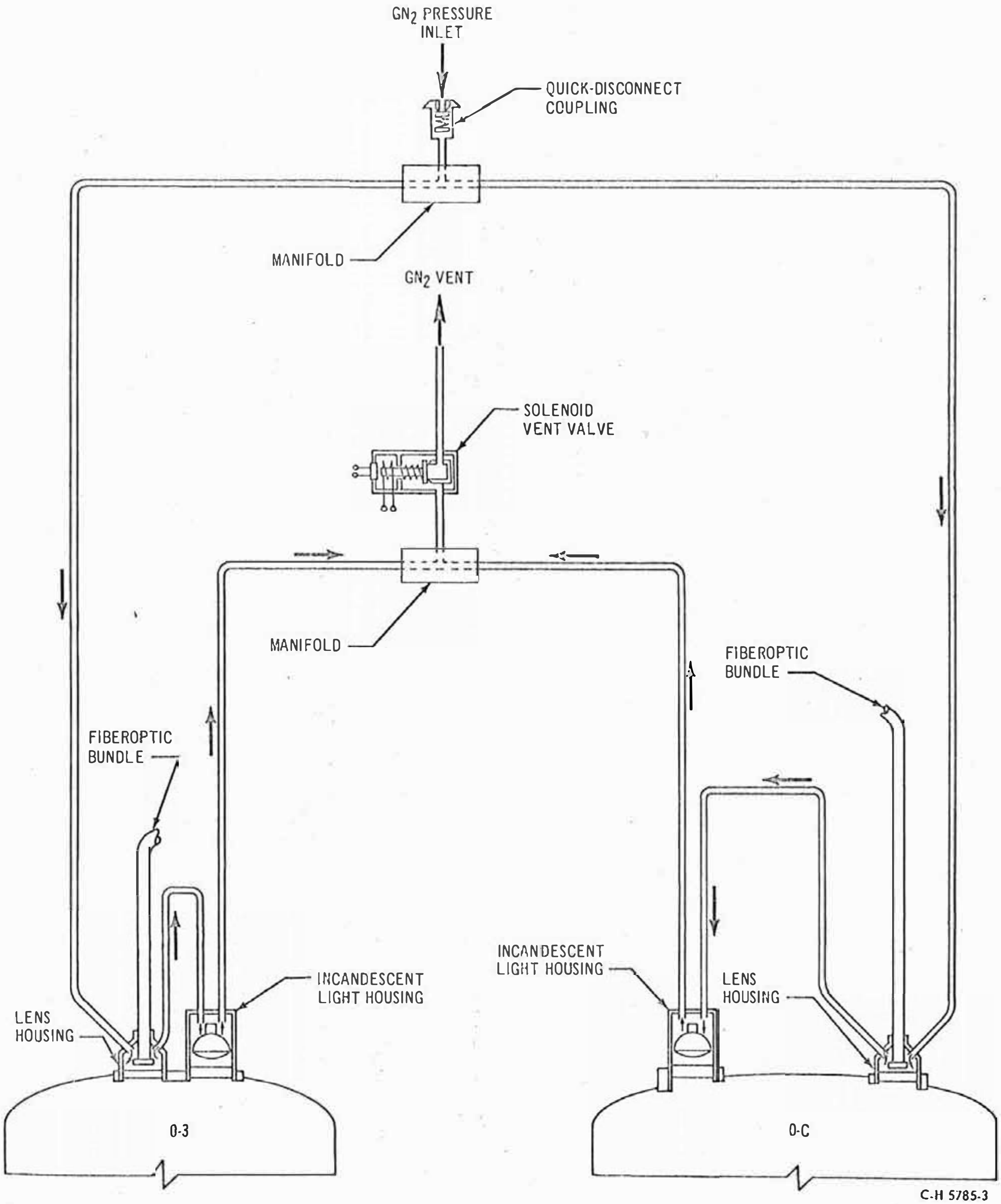

FIGURE 22. GN_2 PURGE SYSTEM--SCHEMATIC

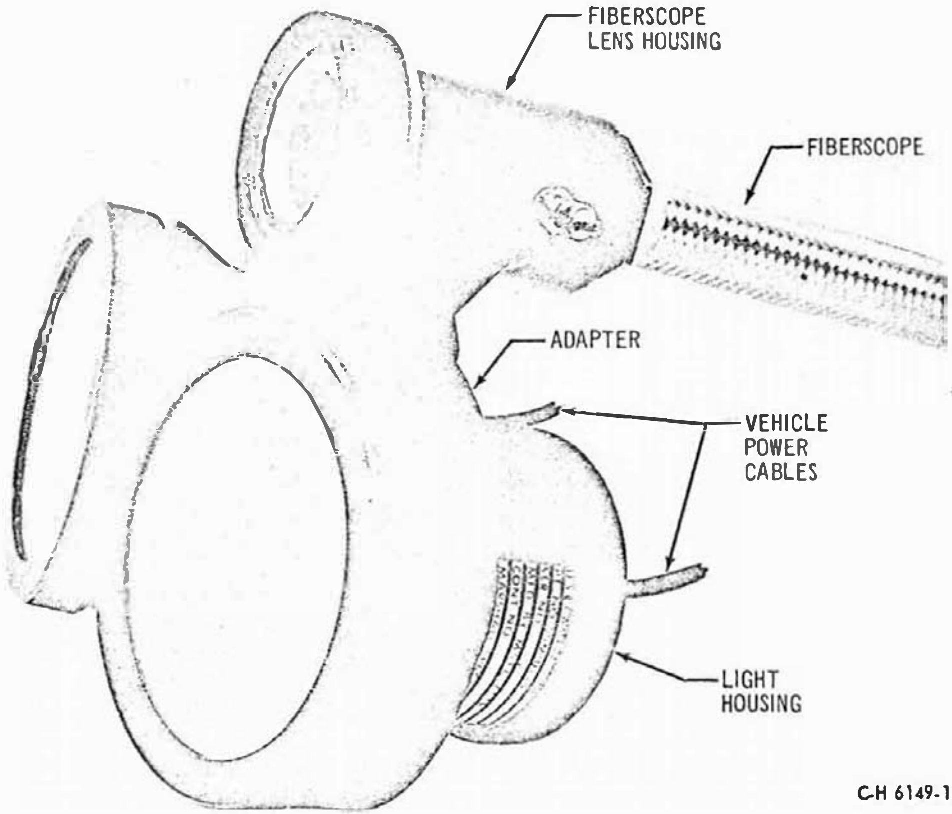

FIGURE 23. INTERSTAGE INCANDESCENT LIGHT AND LENS HOUSINGS IN ADAPTER

The flash heads, cables, and power supplies all have purge fittings. They are purged with dry GN_2 before installation, and individually sealed at atmospheric pressure.

The lens housing at fin line IV, servicing camera number 6, is mounted in a bracket on the interstage wall, and tilted upward 0.698 rad. The aluminum alloy housing is 22 cm long and 13 cm diameter.

II. TESTING

1. Camera

Because the camera is basically an off-the shelf unit, the only tests necessary were those peculiar to the Saturn environment.

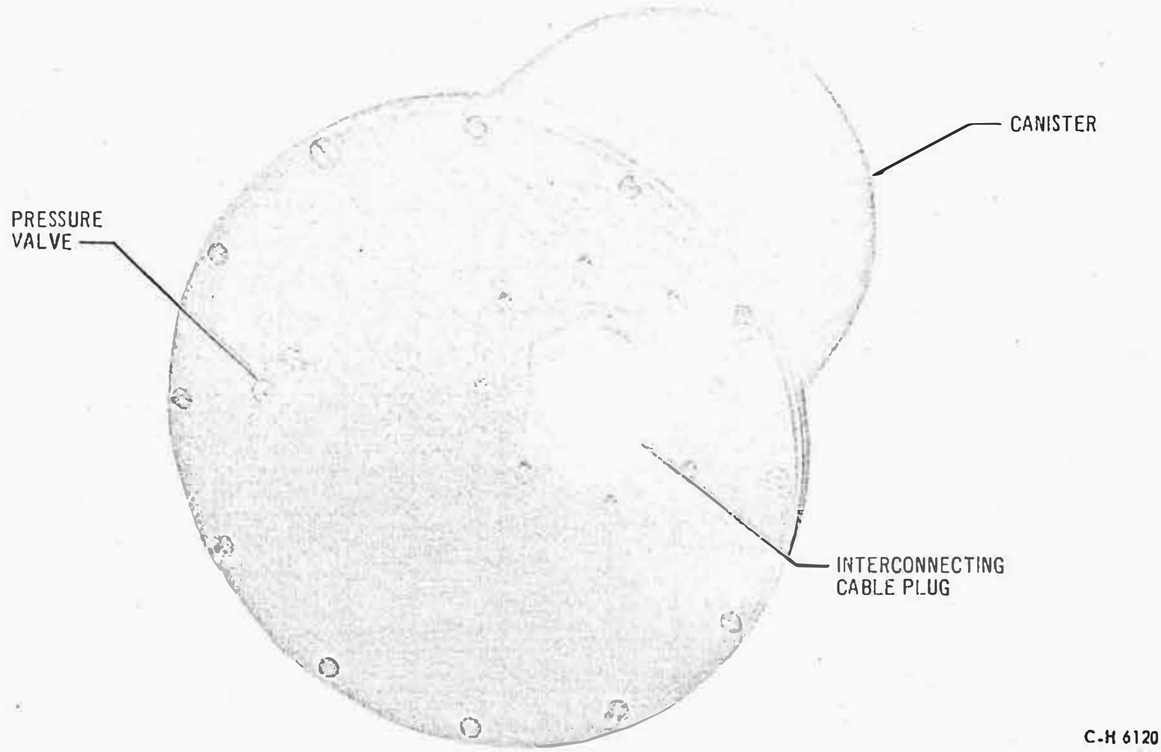

FIGURE 24. FLASH HEAD

The conducted interference readings for the camera inside the capsule were found to be above the limit set by MIL-I-6181D before the camera components were shielded with aluminum mesh, but far below this limit after shielding. Figures 25 and 26 illustrate radiated broadband and conducted interference readings before and after the components were shielded.

The drop tests (Section IV.E. 3) were also tests of camera operation in the camera capsule during capsule flight. These tests indicated that the camera would operate satisfactorily under flight conditions.

2. LOX Container Light and Lens Housings

The LOX container light and lens housing design was tested for necessary torque, leakage and effectiveness of GN_2 purge.

a. Torque Tests. To determine the torque that should be applied to the quartz window bolts in order to prevent leakage, torque tests were conducted with the test-setup tanks filled with LN_2 (figure 27.)

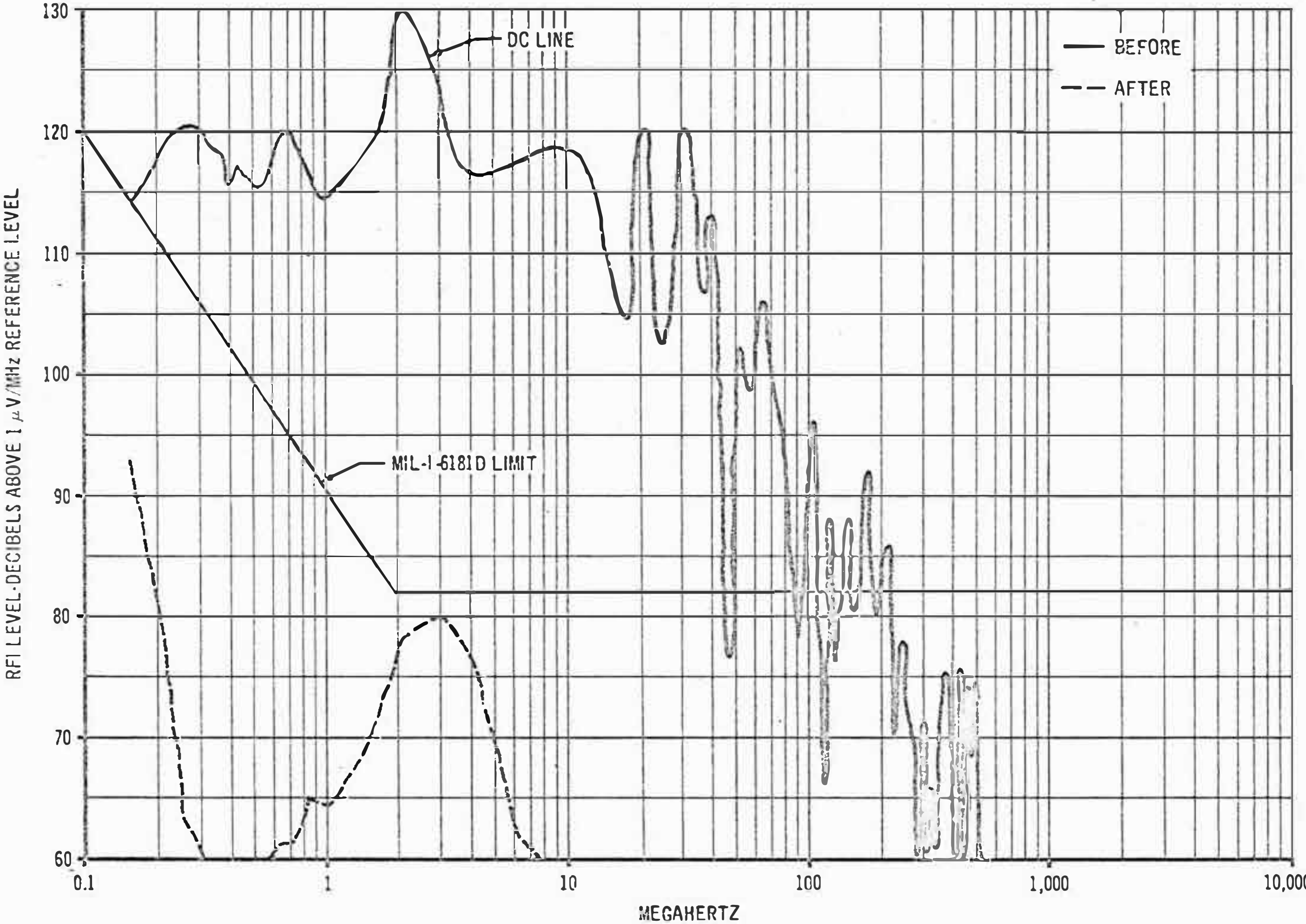

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FIGURE 25. CAMERA CONDUCTED RFI TESTS

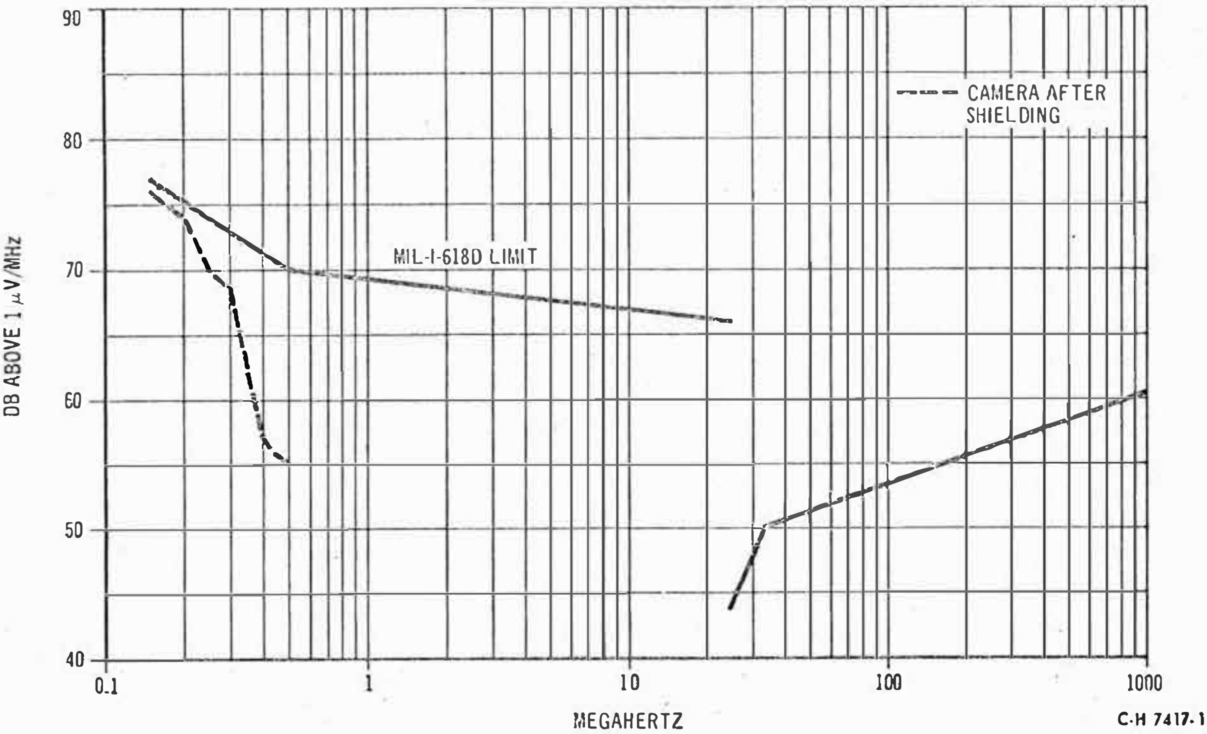

FIGURE 26. CAMERA RADIATED BROADBAND RFI TESTS

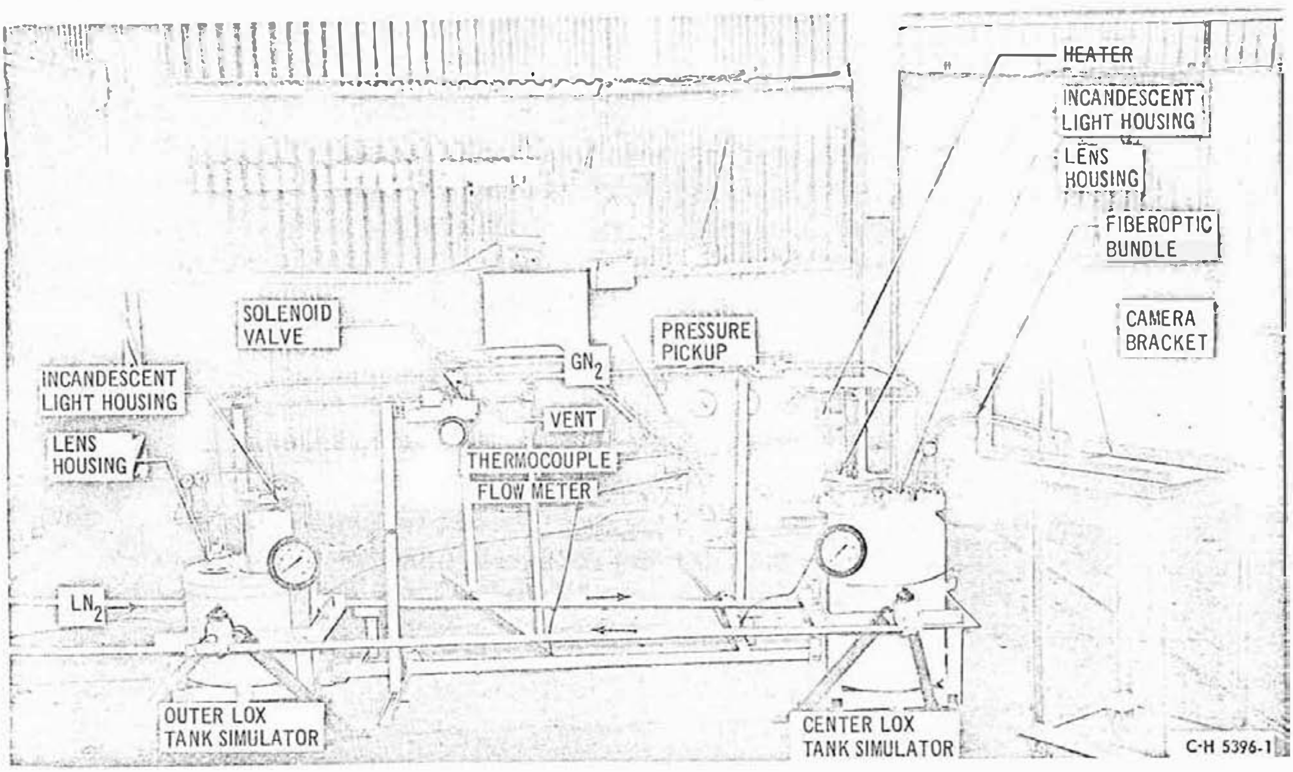

FIGURE 27. TORQUE TEST SETUP

The bolts for the light housing window were torqued to 67.8 Nm and the bolts holding the smaller lens housing window were torqued to 56.4 Nm. The window retainers were checked at room temperature each day for 15 days and torqued when necessary. Results indicate that torque on all window retainers had stabilized by the eighth day to within 5 percent of the initial torque value.

b. Leak Tests. Hydrostatic proof tests were performed to determine whether window seals would leak or window housings would distort under operating conditions. The test tanks were hydrostatically tested at 68.9 N/cm^2 for 15 minutes, and no distortion or leakage was observed. The tanks were then pressurized in increments of 10.3 N/cm^2 to 46.1 N/cm^2, using GN_2. Pressure was maintained for eight minutes, and no leakage was observed. The tank was then half-filled with LN_2, and pressurized in increments of 1.05 kg/cm to 46.1 N/cm^2. Pressure was maintained for ten minutes; no leakage was detected.

c. Purge Tests. During the torque and leak tests, dry GN_2 at 30.9 N/cm^2 and 21° ± 3° C was passed through the lens and lamp container assemblies. Purging began 30 minutes prior to LN_2 tanking, and all optical surfaces remained free from frost and fog. When purging was not begun until two hours after tanking had begun during a later test on the static test stand, the quartz window became fogged. After these two tests, the procedure was established that the GN_2 purge will be started 30 minutes prior to LOX tanking and will be maintained until liftoff.

3. Strobe System

The first RFI shields for the strobe system consisted of only the flexible hydraulic hose over the interconnecting cable and machined connecting surfaces, with O-ring seals at the cable connecting housings. Later, aluminum wire screen was tied over the flash lamp. RFI tests, conducted according to Specification MIL-I-6181D, revealed that the strobe system with such shields did not adequately meet the requirements of the specification. (See figure 28.)

In addition to being a poor RFI shield, the aluminum screen reduced the light intensity by 30 percent. The screen has been replaced by an aluminum honeycomb shield that restricts only five percent of the light and that more than adequately meets the RFI requirements in the flash lamp area. RFI through the cable connections is prevented by metal-to-metal connection between the flash head housing and the power supply, with O-rings to maintain pressurization. To make the system a completely sealed-in unit without conducted or radiated RFI. a plate containing feed-through capacitors is installed just inside the power supply at the battery cable plug.

Results of the most recent tests for conducted RFI are shown in Figures 29 and 30. Radiated interference levels were also checked, but were too low to be measured.

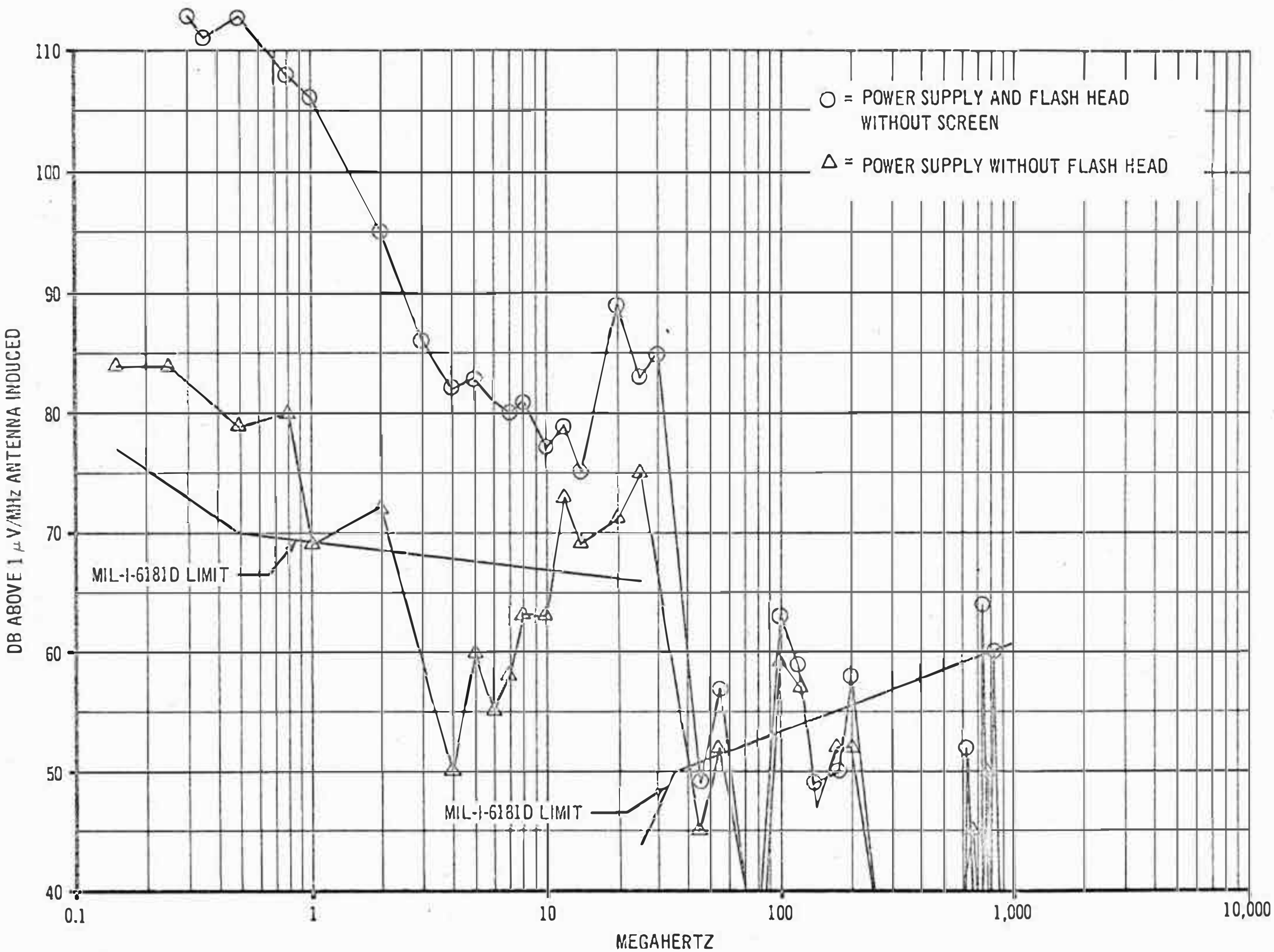

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FIGURE 28. STROBE SYSTEM RADIATED BROADBAND RFI TESTS

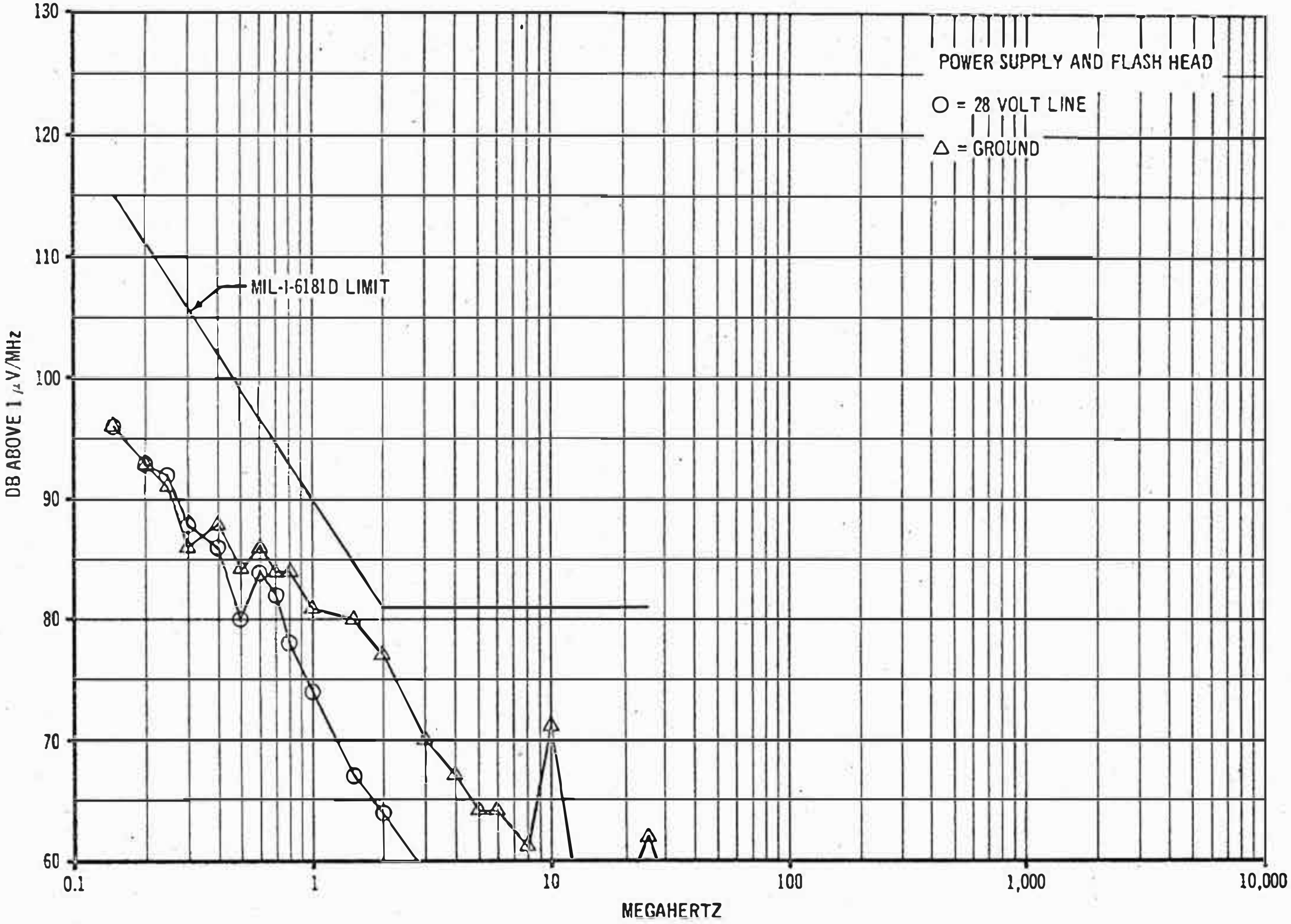

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FIGURE 29. STROBE SYSTEM CONDUCTED BROADBAND RFI TESTS

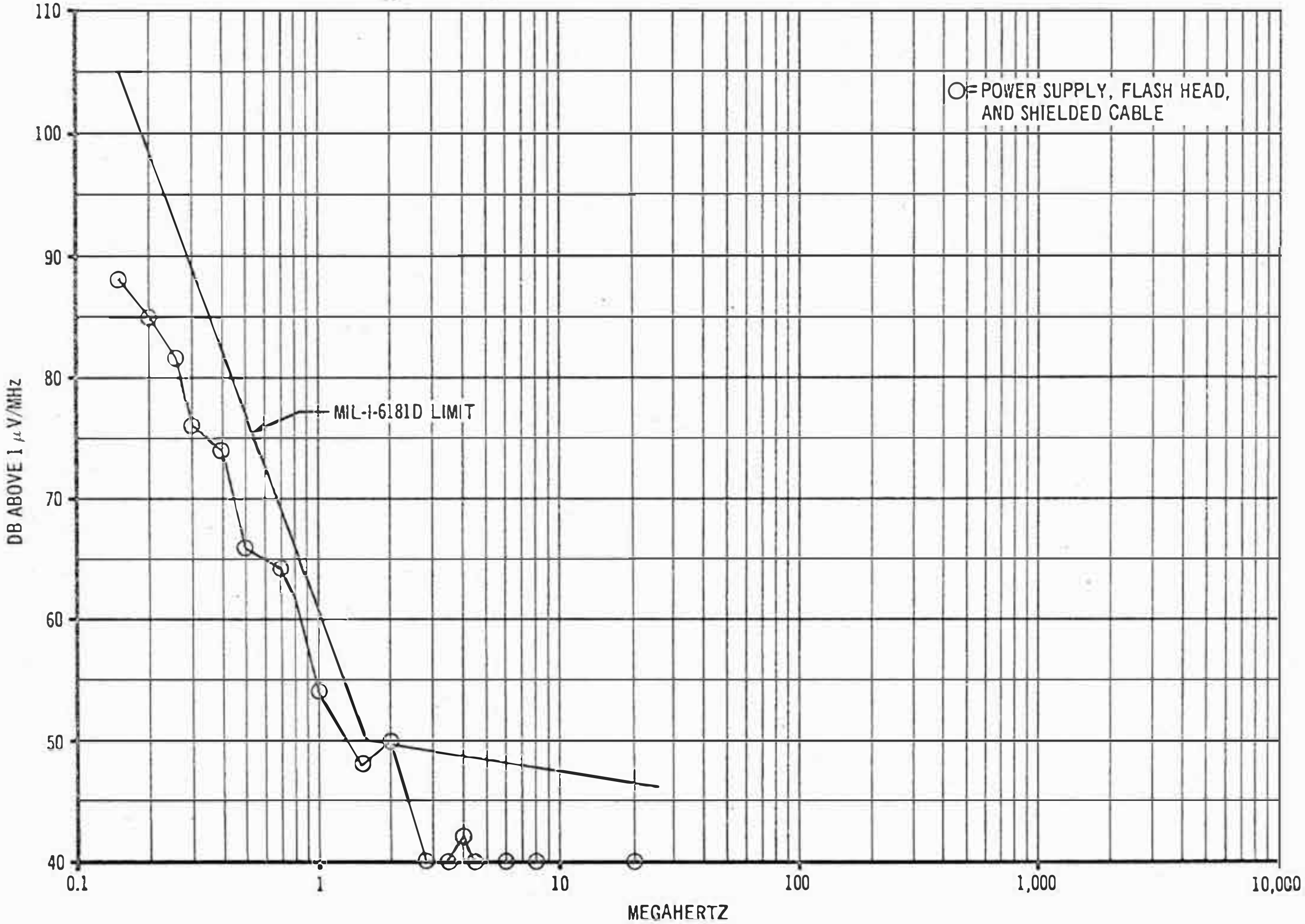

FIGURE 30. STROBE SYSTEM CONDUCTED BROADBAND RFI TESTS USING CURRENT PROBE

SECTION III. CAPSULE EJECTION SYSTEM

A. COMPONENT DESCRIPTION

The capsule ejection system consists of eight pneumatically operated ejection mechanisms, a high-pressure GN_2 storage sphere, and the associated valves and hardware necessary for control of pressurized GN_2. The ejection system provides protection and rigid mounting for the capsules during vehicle ascent. The capsules are ejected from the vehicle approximately 25 s after stage separation.

There are two configurations for the ejection mechanisms: model A, which houses a model A capsule in the ejection tube, and model B, which houses a model B capsule. The model A ejection mechanism is shown in figure 31. The model B ejection mechanism is similar, differing only in the position of the electrical disconnect assemblies. On the outside of the ejection tube, parallel to the centerline of the tube, is a pneumatic ejector cylinder and piston with an ejection finger at its forward end. The ejection finger fits into a slot in the camera capsule shell to hold the capsule in place. The finger is secured with two nuts and a lock bolt, which are in turn held in place by a shear pin.

One of each type of ejection mechanism is mounted on the S-I stage spider beam fairing at each fin line (figure 32). The ejection mechanisms are canted 0.18 rad outboard from the vehicle centerline so that the ejected capsules will avoid collision with the S-I Stage. In addition, the model A ejection mechanisms are canted 0.31 rad outward from the fin plane, while the model B ejection mechanisms are canted 0.27 rad outward from the fin plane; this variation in attitude greatly reduces the probability of capsule collision after ejection.

Because the ejection mechanisms extend into the vehicle flight airstream, they are insulated with Thermo-Lag T-230, and the exposed portions of the model B capsules are covered with the same material.

The GN^2 storage sphere and manifold are located on the spider beam member between fin lines I and II. The volume of the fiberglass sphere is 0.014 m^3. Figure 33 is an installation view of the sphere and manifold.

B. SYSTEM OPERATION

The ejection system is represented schematically in figure 34. The GN_2 storage sphere is pressurized to 19.3 MN/m^2 approximately 15 minutes before liftoff through the same lines and at the same time as the fuel container pressurization spheres. The GN_2 flows through the fuel pressurization spheres, the pressurization quick-disconnect coupling, the GN_2 distribution manifold, the

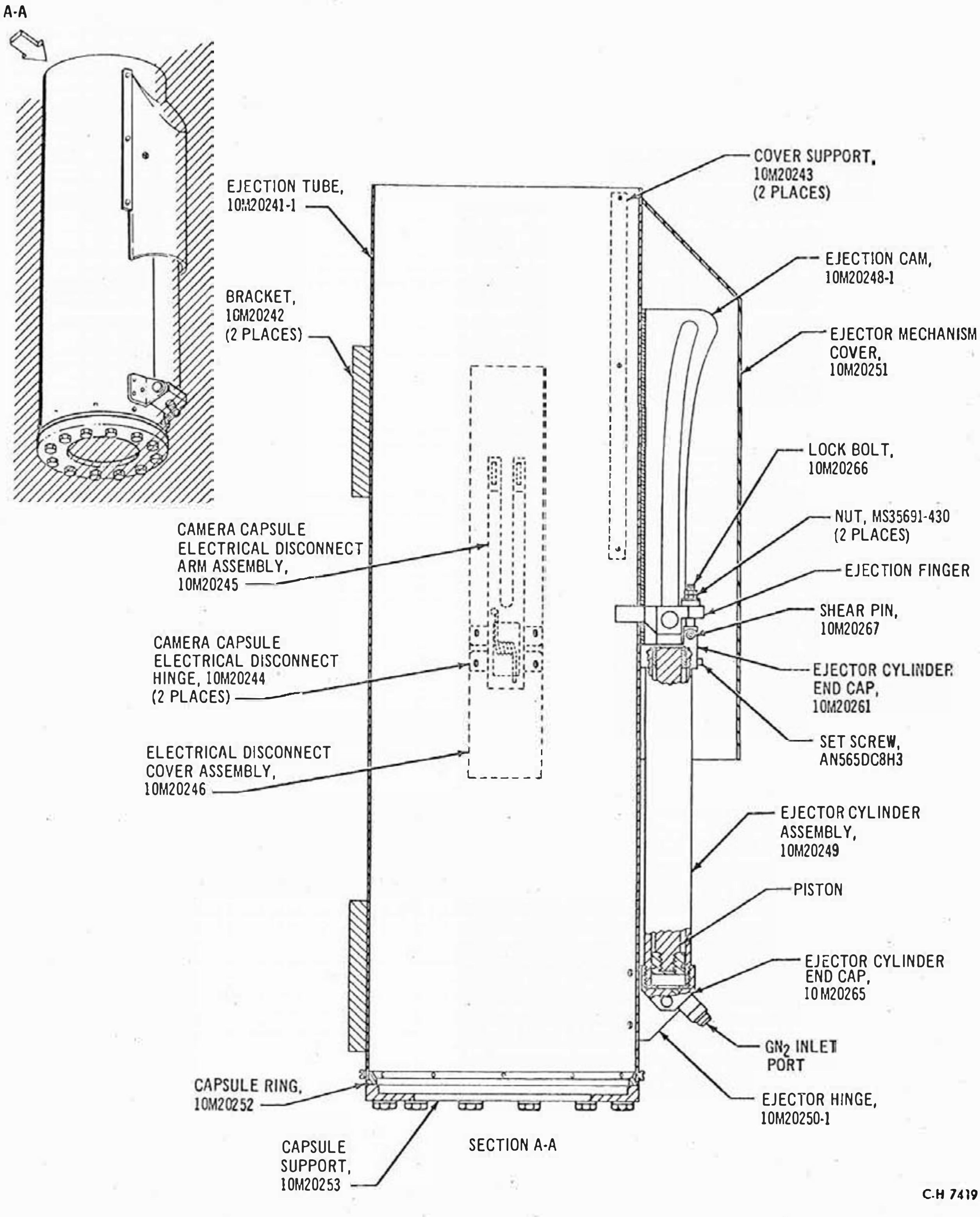

FIGURE 31. MODEL A CAPSULE EJECTION MECHANISM, SECTIONAL VIEW

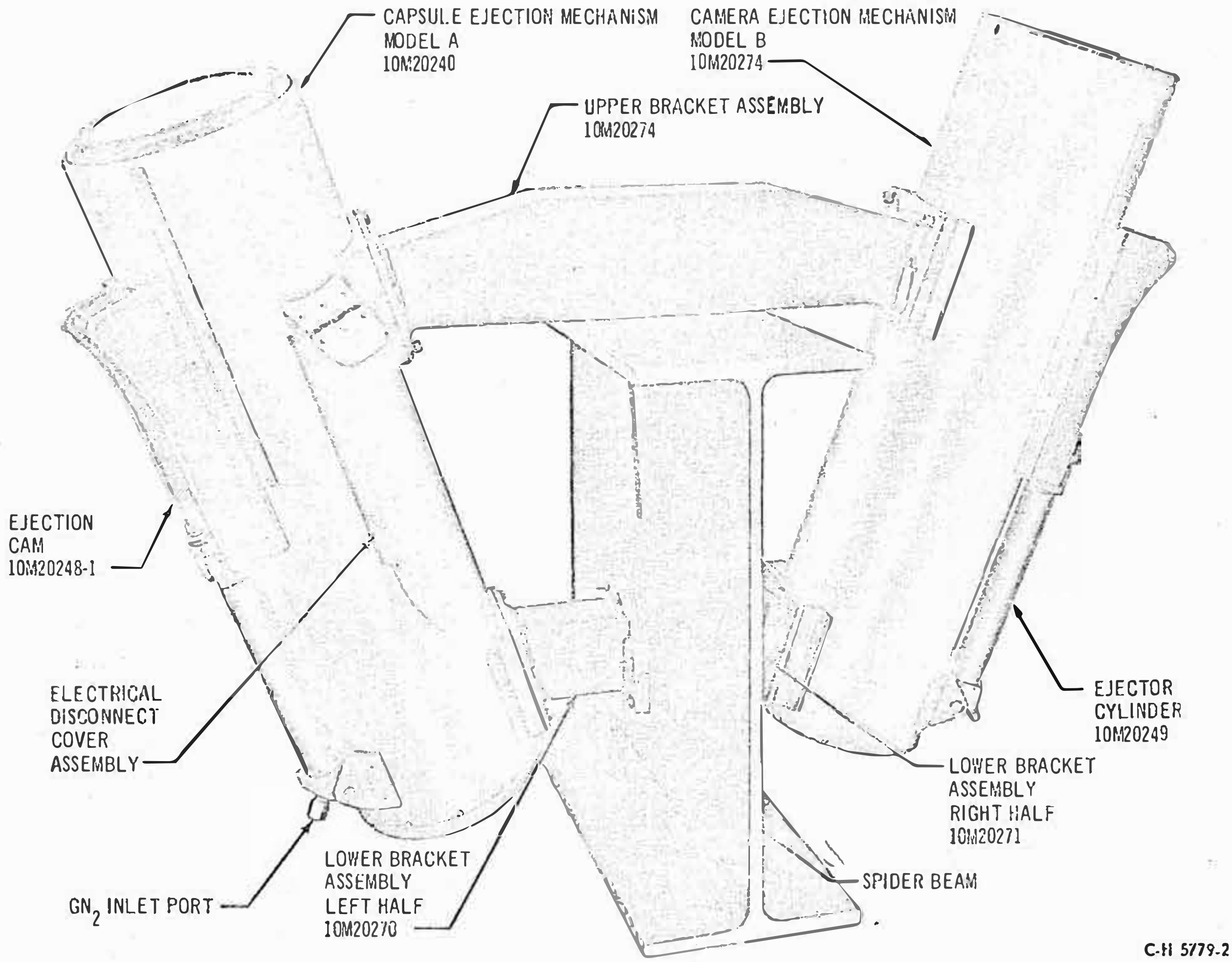

FIGURE 32. EJECTION MECHANISM INSTALLATION

solenoid control valve, and the fill and vent control valve to the GN_2 storage sphere. Charging and maintaining of the GN_2 supply in the sphere are controlled by a normally closed solenoid control valve.

Energizing of the normally closed fill-and-vent control valve opens the high-pressure sphere vent port. A special fitting on the valve acts as a junction for the fill and ejection lines, and admits pressure to the pressure gage providing measurement information to the Launch Control Center.

A signal from a timer energizes the normally closed solenoid control valve approximately 25 seconds after stage separation. This valve opens, allowing high pressure GN_2 to flow through the camera ejection manifold to the camera ejection mechanisms. A constant flow regulator, installed in the manifold, prevents premature actuation of the ejection cylinders by leakage in the solenoid control valve.

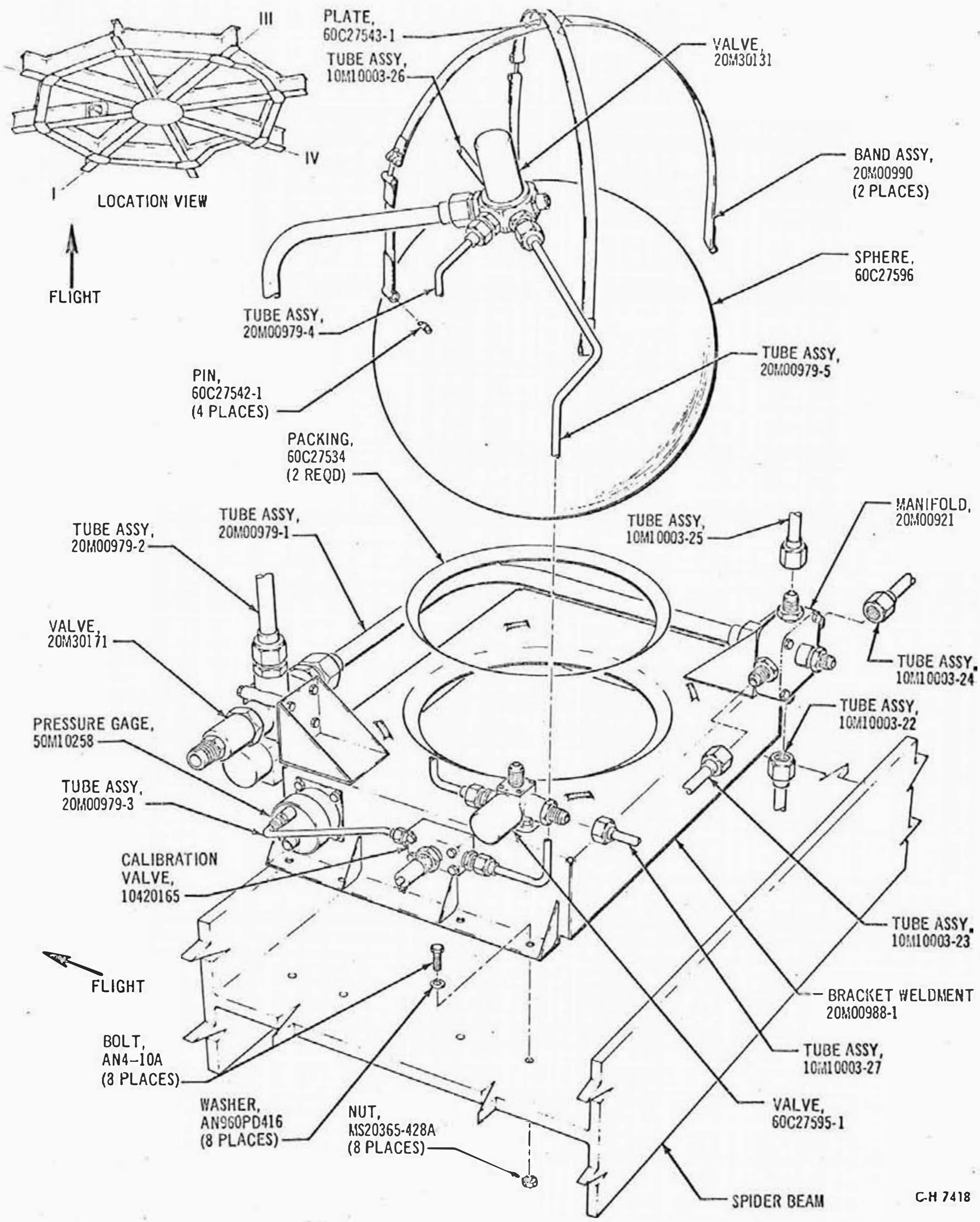

FIGURE 33. GN_2 STORAGE SPHERE AND MANIFOLD, INSTALLATION VIEW

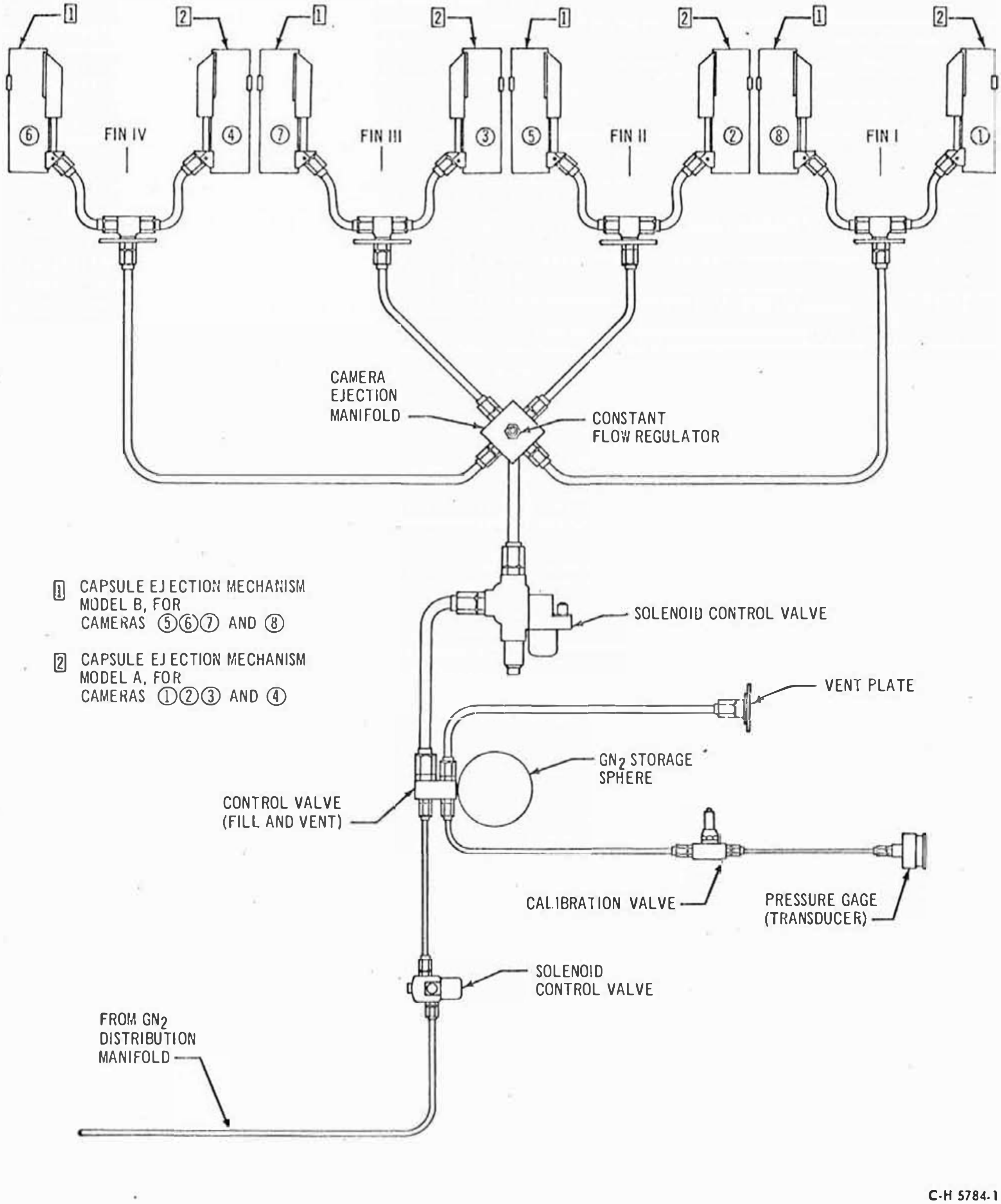

FIGURE 34. CAPSULE EJECTION SYSTEM--SCHEMATIC

Upon receipt of the ejection signal, GN_2 pressure builds up. When GN_2 pressure reaches approximately 5.5 MN/m^2 the piston in the ejection mechanism causes the shear pin to shear, allowing the ejection finger to force the camera capsule through the ejection tube. The upper part of the finger follows the path of the ejection cam (figure 35), causing the finger to be withdrawn from the capsule at the end of the piston stroke. The ejector cylinder assembly, secured by the ejector hinge, pivots as the ejection finger travels along the ejection cam.

A spring-loaded electrical disconnect arm (figure 36) provides an electrical connection from the vehicle programmer to the capsule. As the camera is forced out of the ejection tube, electrical connectors are disengaged and the disconnect arm moves out of the path of the capsule.

Figure 37 shows the position of the ejection finger and lock bolt as a model A capsule is ejected, and also shows how the capsule stabilization flaps are deployed. The capsule exit velocity is approximately 8 m/s relative to the S-I stage.

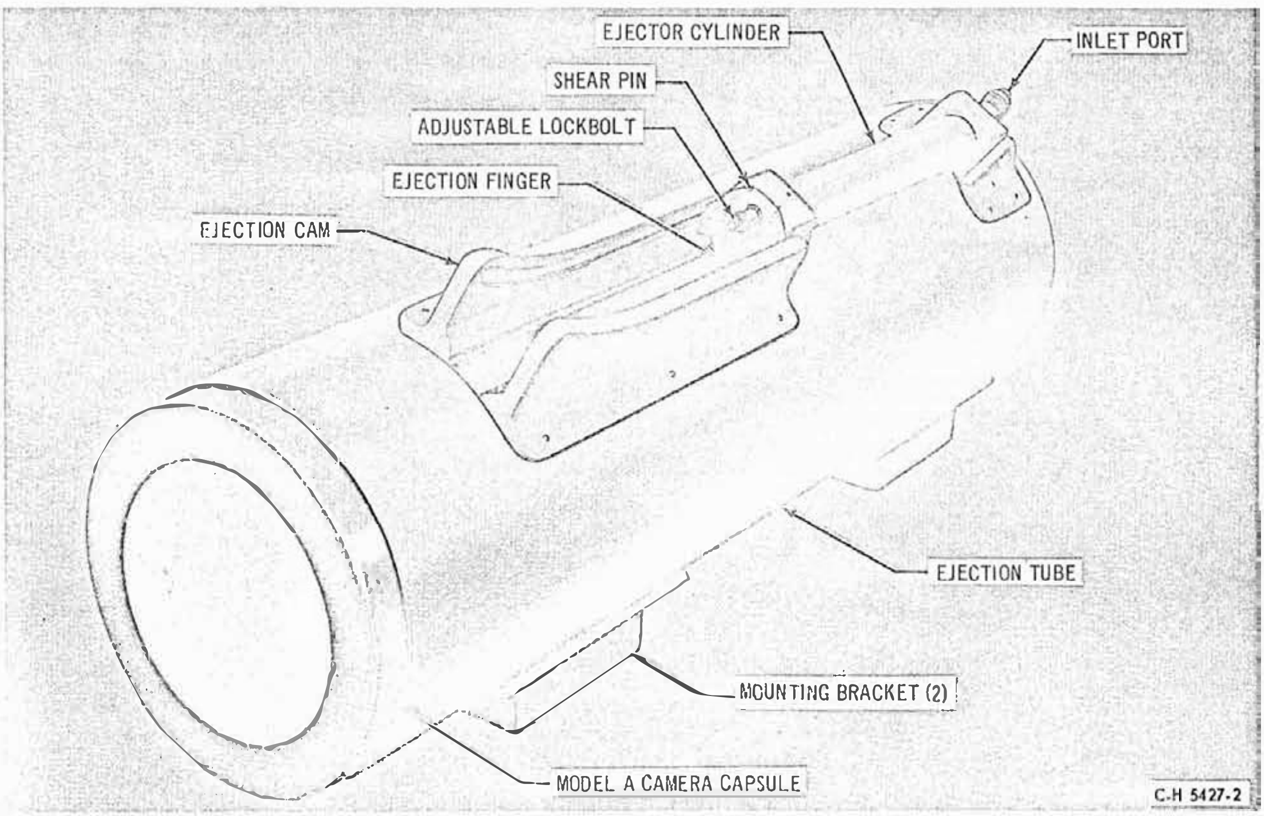

FIGURE 35. CAPSULE EJECTION MECHANISM

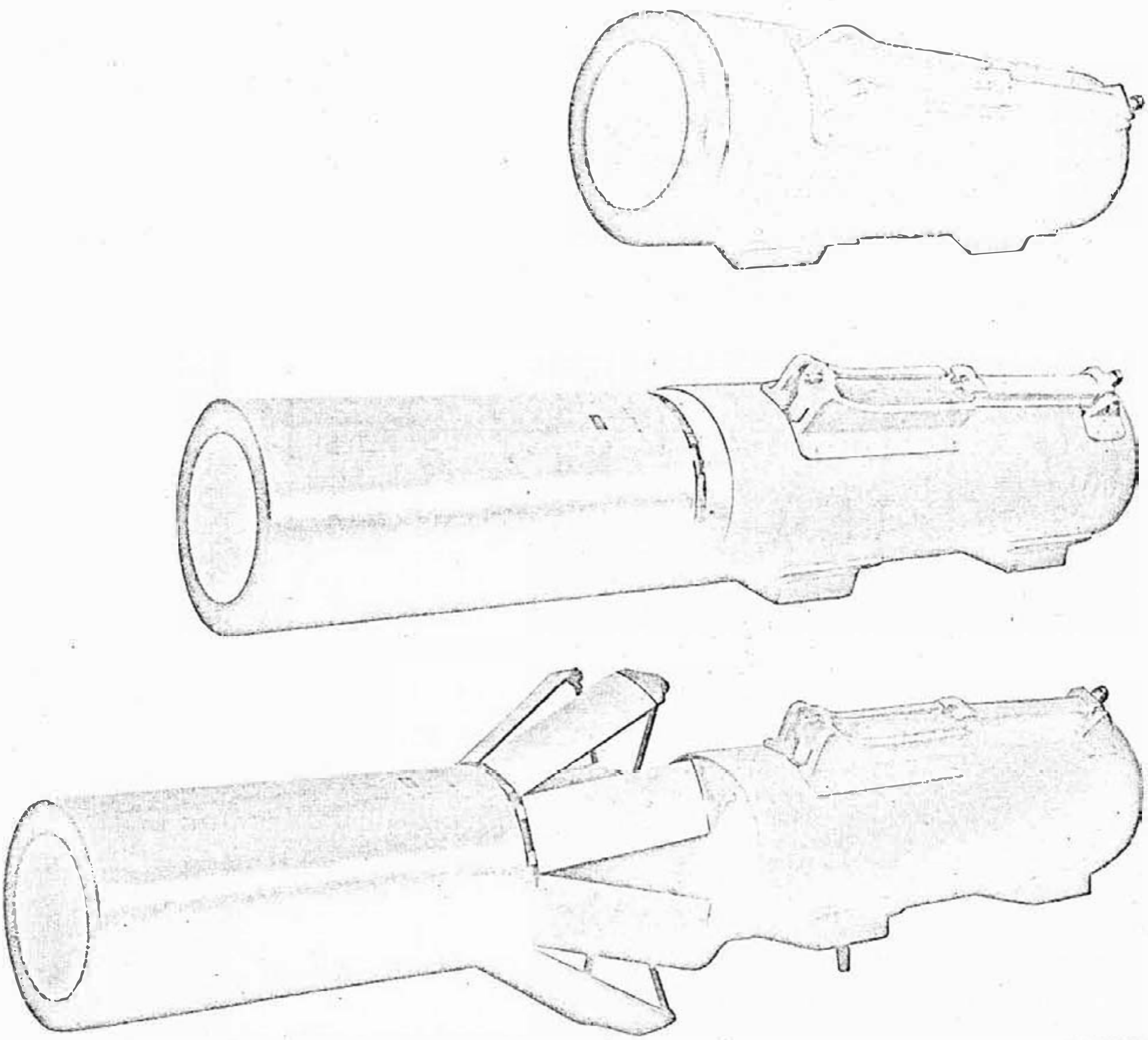

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FIGURE 37. SEQUENCE OF STABILIZATION FLAPS DEPLOYMENT

model B capsules. The insulating material chosen for this purpose was Thermo-Lag T-230. Temperatures of the quartz windows and retaining rings without added insulation were investigated and found to be within the specifications of Saturn design criteria.

Figure 38 shows the points on the ejection mechanism where temperature investigations were made. Figure 39 shows the results of an investigation at a particular point with the Thermo-Lag insulation. Temperature at the insulated point remained below 77°C.

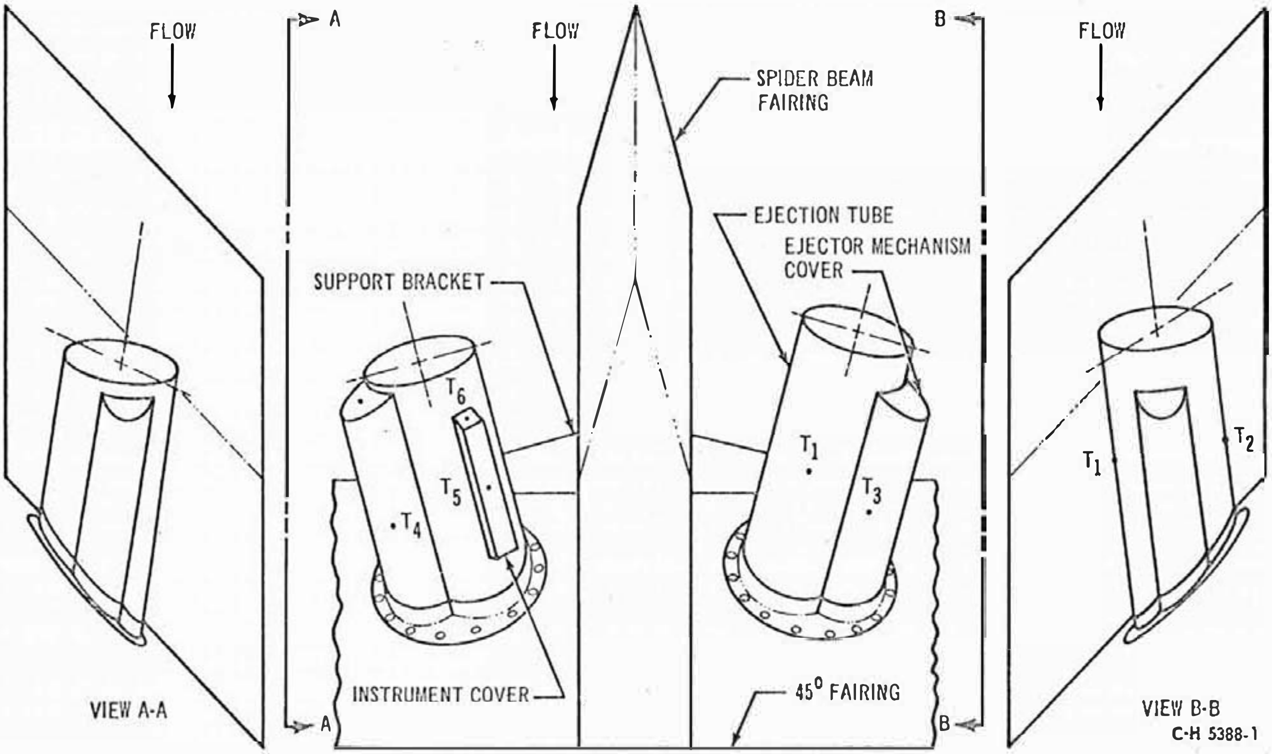


FIGURE 38. EJECTION MECHANISM TEMPERATURE ANALYSIS POINTS

2. Ejection Mechanism Tests

The capsule ejection mechanism was tested to verify its design values under actual operating conditions. The test setup (figure 40) had the same line lengths and system components as the SA-5 vehicle. A dummy camera capsule with a mass of 20.4 kg and a flight camera capsule of 25.4 kg were used during these tests. The dummy capsule was ejected horizontally and the flight capsule was ejected 0.175 rad above the horizontal.

Various aluminum alloy shear pins were used in the tests. Table 3 gives the results of the tests that were conducted using the 6061-T6 aluminum alloy shear pins. The test series indicated that a 0.335 cm-diameter shear pin of 6061-T6 aluminum alloy should be used on the flight hardware. This is the pin that is now used in the Saturn IB configuration.

The ejection system performed according to design expectations, and no fractures or deformations caused by ejection acceleration forces occurred.

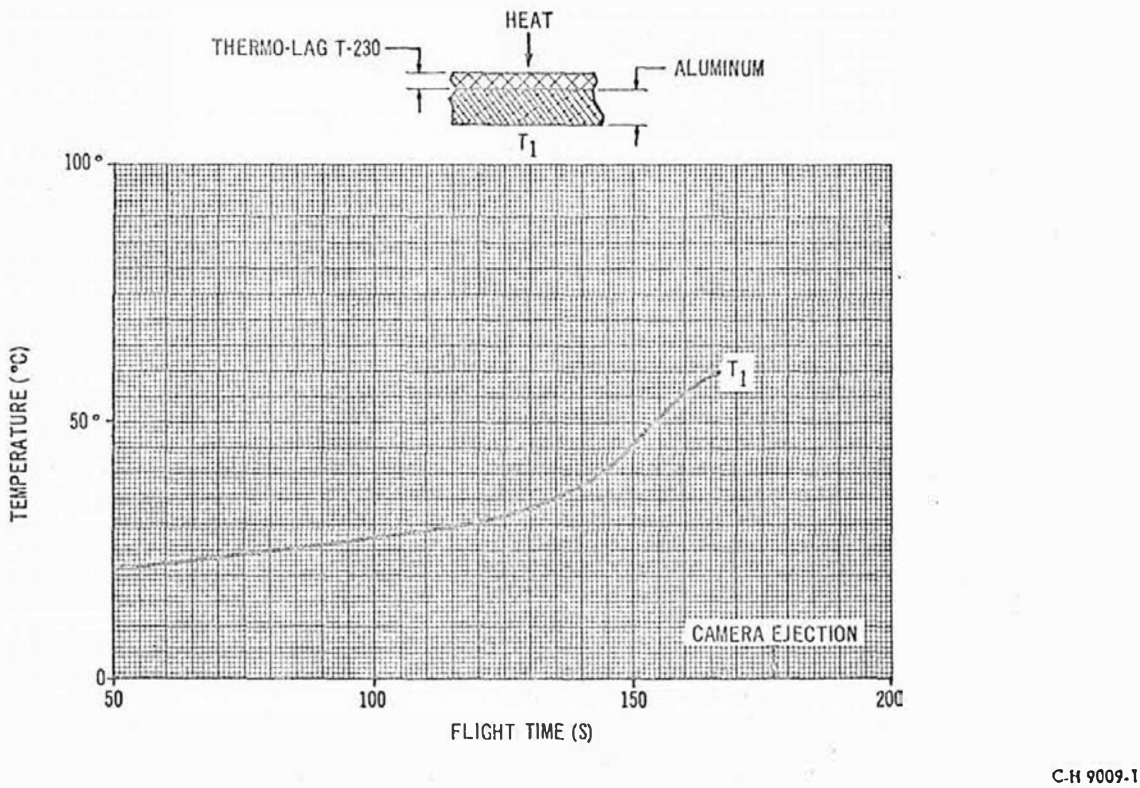


FIGURE 39. RESULTS OF TEMPERATURE ANALYSIS AT A PARTICULAR POINT

TABLE 3. EJECTION TEST RESULTS

Test Capsule	Supply Sphere (MN/m^2)		Ejection Cylinder (MN/m^2)		Acceleration (G)		Exit Velocity (cm/s)	Time From Ej. Signal (μs)	
	Init.	Final	Pin Shear	Final	Peak	Avg.		Pin Shear	Eject
Dummy	19.9	16.7	6.02	16.7	35.5	19.6	848	86	182
Dummy	19.8	14.4	5.64	16.5	36.8	20.5	924	81	165
Flight	19.8	18.3	6.48	17.5	25.0	12.5	634	71	208
Flight	19.9	17.9	6.73	17.9	28.0	8.4	533	87	252
Flight	19.8	17.8	6.89	17.7	31.0	7.3	485	93	268

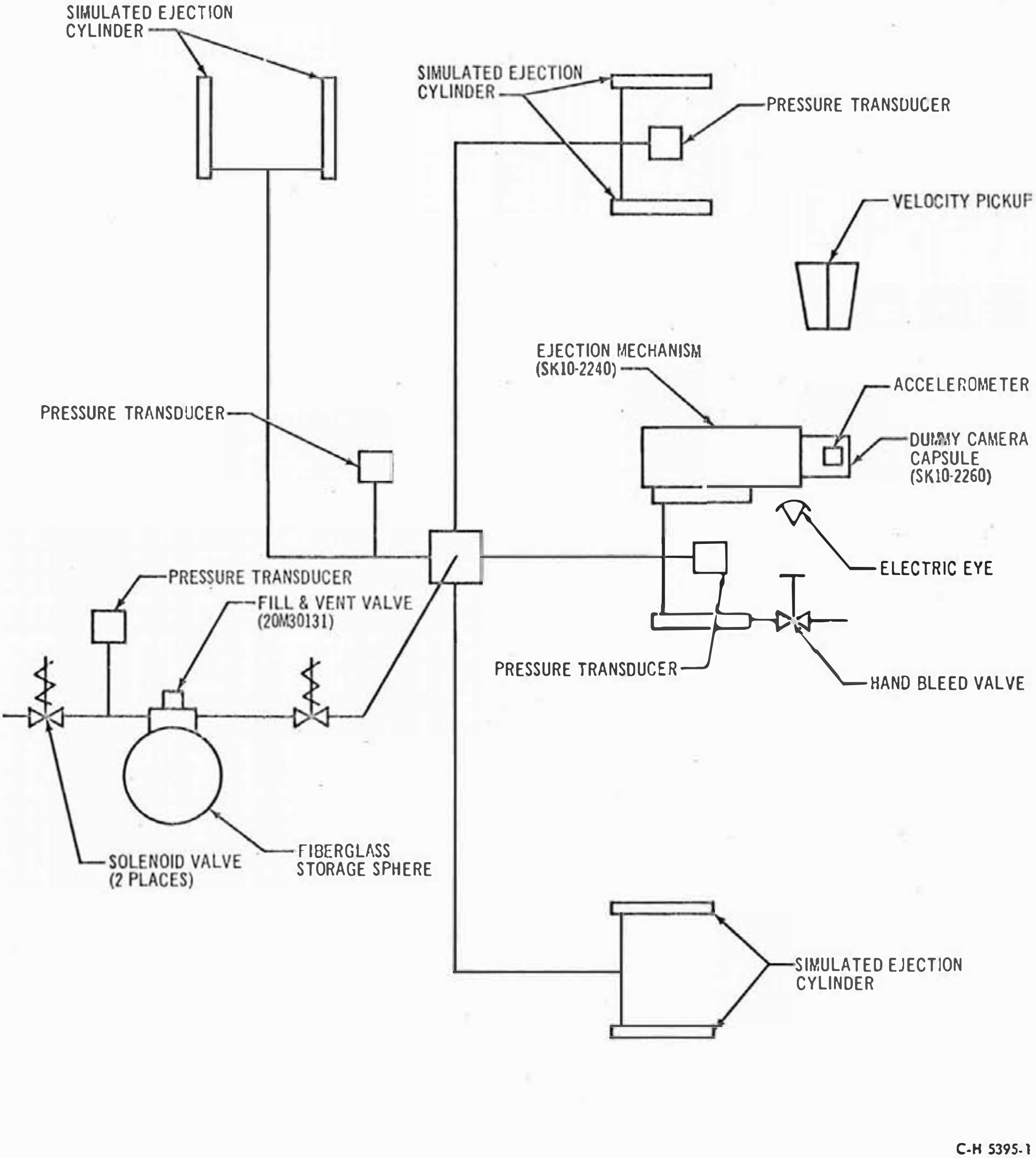


FIGURE 40. EJECTION SYSTEM TEST--SCHEMATIC

3. Vibration Tests

Vibration tests were conducted to investigate the shear pin reliability under flight vibration forces.

The vibration test setup is shown in figure 41. A 10-minute logarithmic sweep was performed from 15 to 2000 Hz at 7g sinusoidal vibration in each of three planes.

During the first test, the shear pin failed when the torque on the inner lock bolt nut MS35691-430 (figure 31) exceeded 0.226 Nm. Subsequent tests showed that the addition of three 0.159 cm rubber washers behind the nut alleviated this problem. After the rubber washers were installed, the nut was proof-torqued to 0.565 Nm (adequate for flight purposes) with no failure resulting.

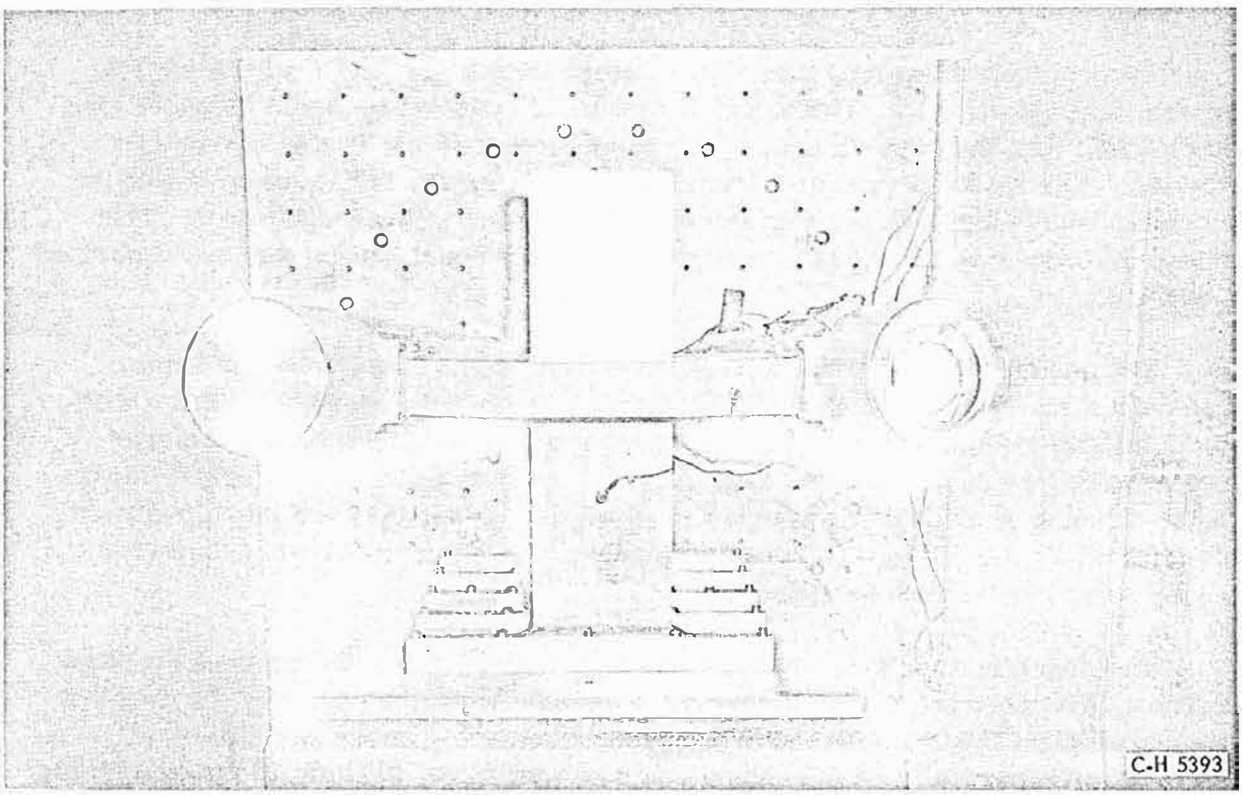


FIGURE 41. EJECTION MECHANISM VIBRATION TEST SETUP

SECTION IV. RECOVERABLE CAMERA CAPSULE

A. GENERAL DESCRIPTION

The recoverable camera capsule is basically an insulated aluminum cylinder. Through the incorporation of Teflon sealed bulkheads, it is divided into three compartments: a watertight lens compartment, a watertight camera compartment, and a recovery systems compartment. The capsule is designed to provide:

a. A rigid mounting with the camera in proper attitude.

b. A controlled atmosphere in which the camera can operate properly.

c. Protection for the camera during vehicle flight, capsule ejection and reentry, water impact, and salt water immersion.

d. Systems for recovery of the cameras.

Two configurations are used in the Saturn IB vehicle, the model A capsule and the model B capsule. The model A capsule (figure 42) requires no accessory optics other than the camera lens. This model is used for filming the vehicle exterior. The model B capsule (figure 43) is designed for indirect viewing of the interior of the Saturn vehicle with the aid of accessory optical equipment. The lenses, fiberoptic bundles, and lights used with the model B capsules are described in Section II.

The configurations for the two capsule models are essentially the same. Both have a maximum diameter of 19.7 cm and an overall length of 72.4 cm with the stabilization flaps closed. The recovery systems assemblies and cameras used in each type of capsule are identical, but the lens and camera compartments are modified to provide different electrical connector and ejection block positions and to allow the mounting of dissimilar lenses. The major differences between the two capsules can be seen in figure 44.

The differences in configuration are also reflected in total mass of the flight capsule. Prepared for flight, the model A capsule weighs 26 kg, and the model B capsule weighs 27.5 kg. The recovery system assembly cover and stabilization flaps, which separate from the capsule at ejection at an altitude of approximately 4,300 m, weigh 4.48 kg. Therefore, mass of the model A capsule at impact is 21.5 kg, and mass of the model B capsule is 23 kg.

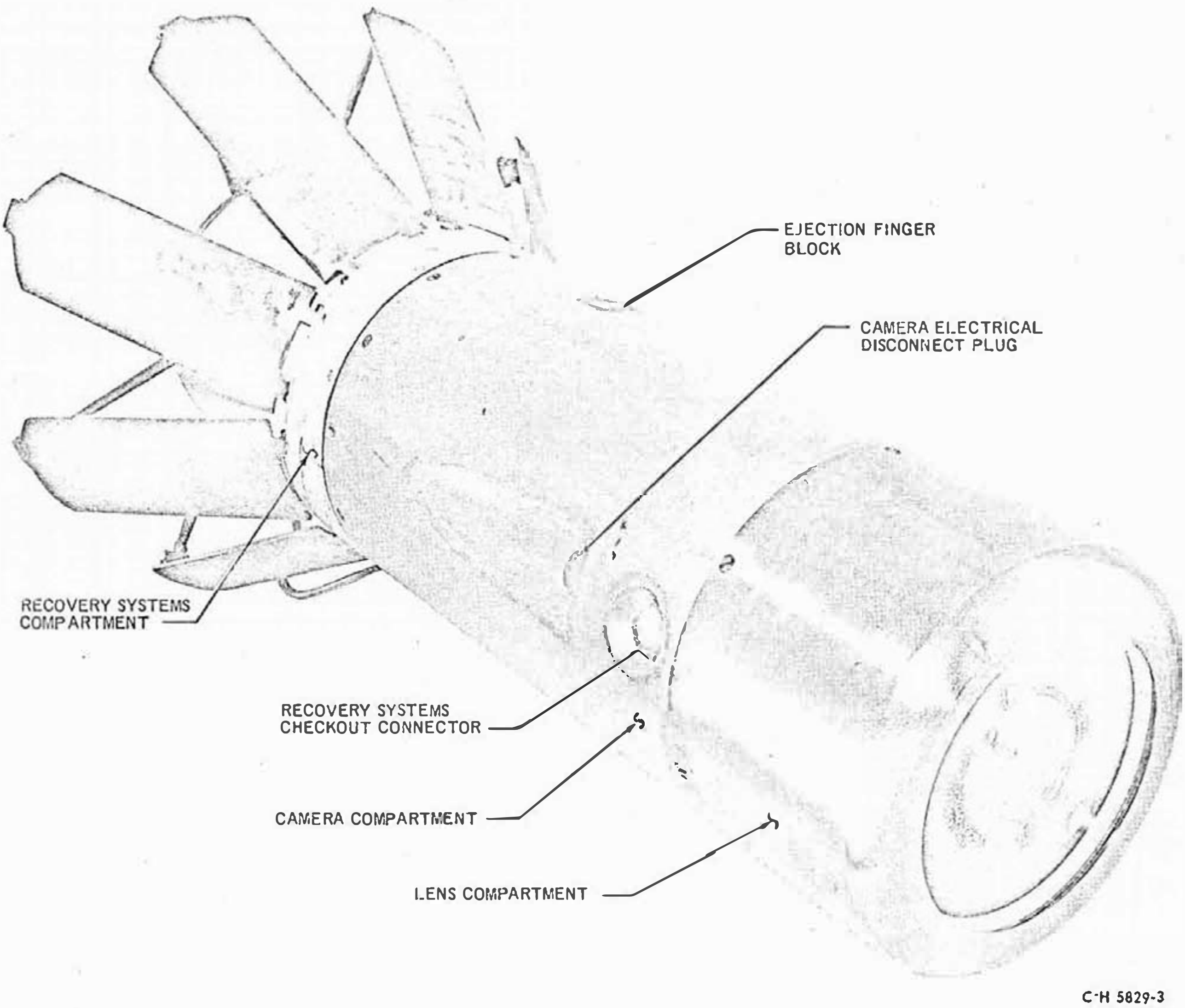


FIGURE 42. MODEL A CAMERA CAPSULE

B. LENS COMPARTMENT

The lens compartment of the model A capsule houses a 4.2 mm Periphoto wide-angle lens with a 1.92 rad field of view. The viewing window of the capsule is quartz, 1.91 cm thick and 15.3 cm diameter.

The model B capsule facilitates indirect viewing of the interior of the capsule with a coupling lens system that attaches to a fiberoptic bundle. (The 50-mm Wollensak lens in the lens compartment couples, through the 1.91-cm thick, 7.65-cm diameter quartz viewing window in the capsule with a similar 50-mm lens that attaches to a fiberoptic bundle.)

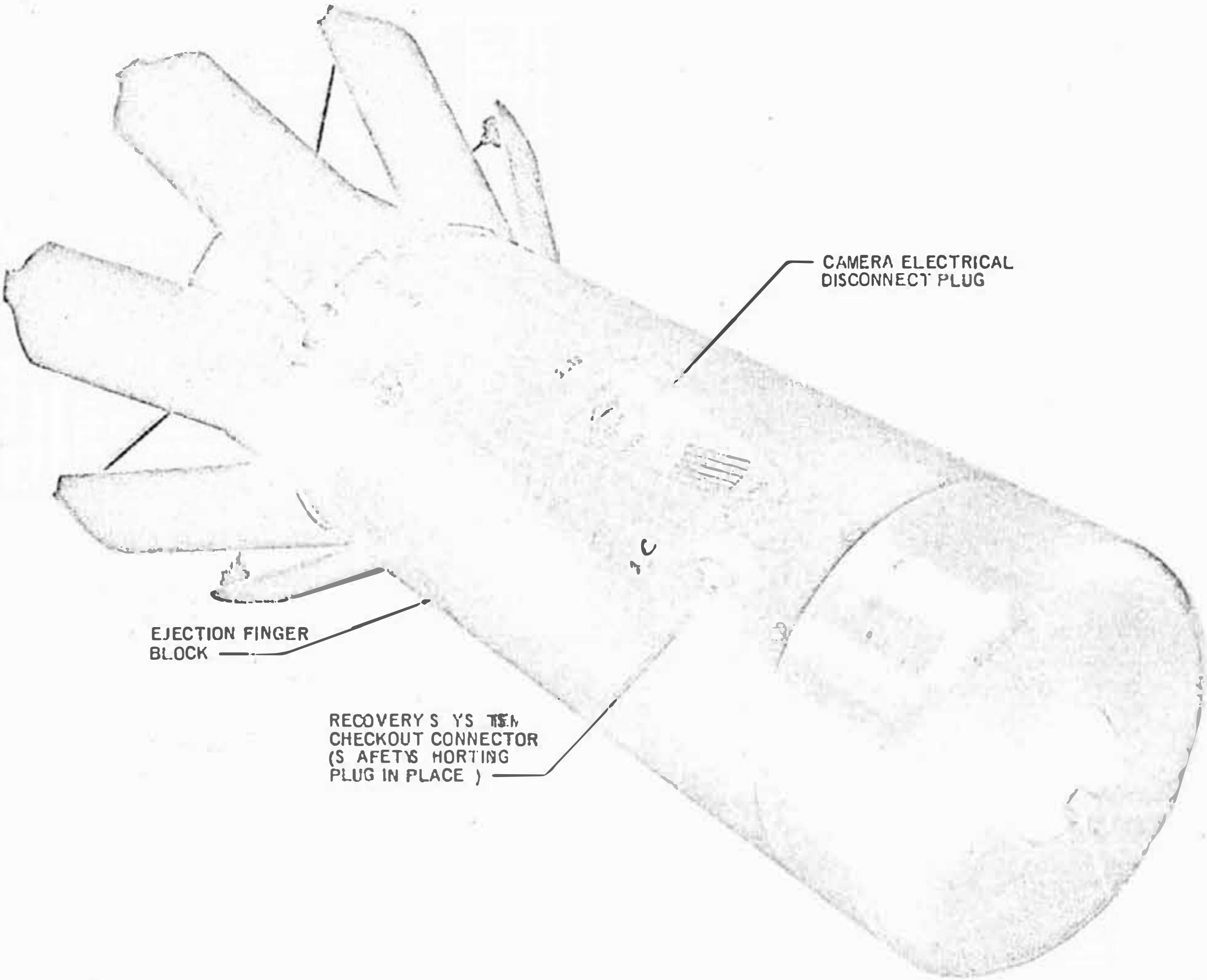


FIGURE 43. MODEL B CAMERA CAPSULE

The quartz window in each type of capsule is sealed with two Teflon gaskets; a stainless steel retainer ring secures both the Teflon seal and the window.

C. CAMERA COMPARTMENT

The camera is mounted in the central compartment of the capsule, between two bulkheads. The camera mounting supports and bulkheads hold the camera rigidly in place. The camera is installed and removed through the forward end of the capsule by removing the lens compartment and the bulkhead to which the camera is attached. The camera compartment is sealed and insulated to maintain a minimum pressure of 6.8 N/cm^2, and a temperature range of -29° to +66°C.

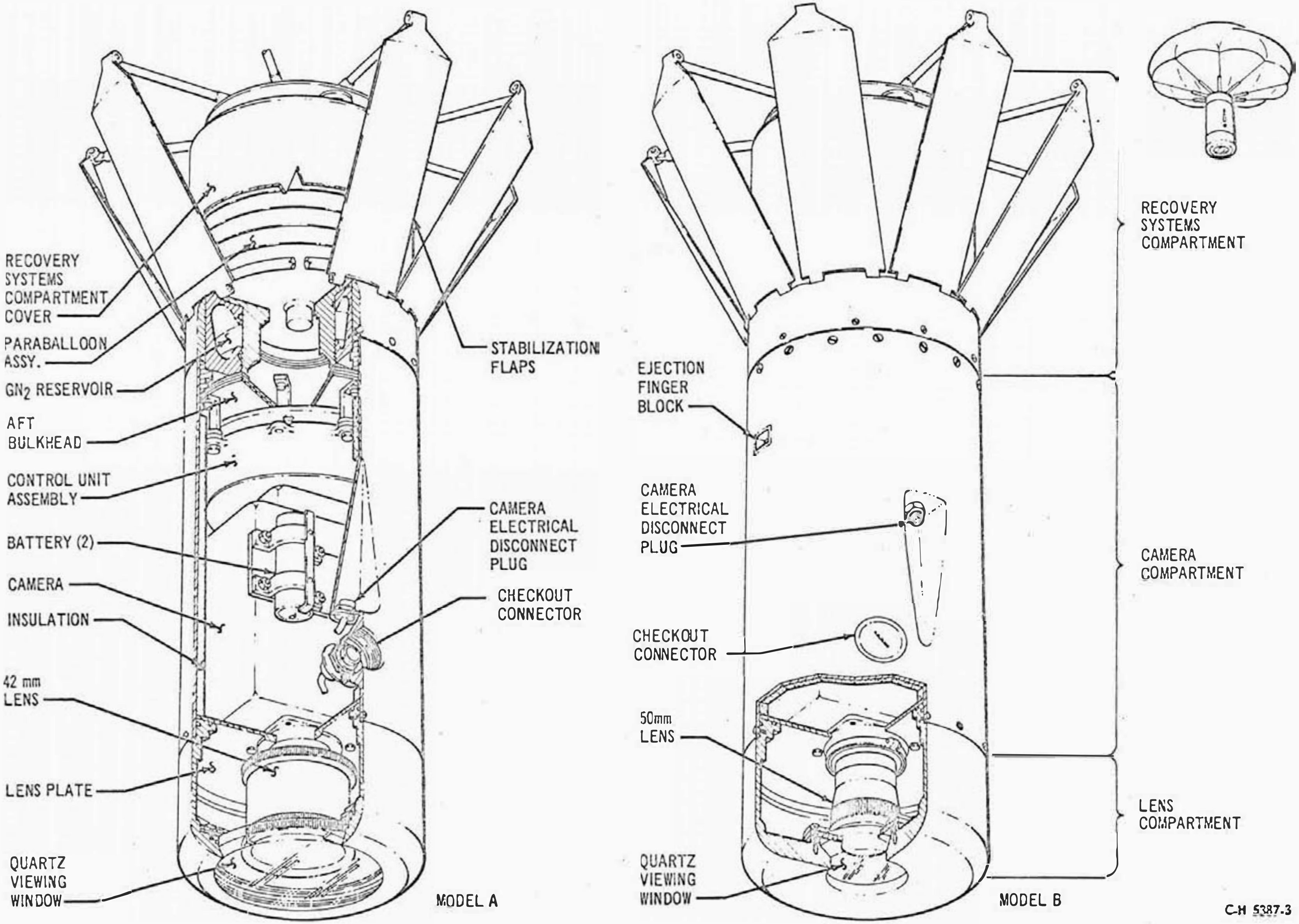


FIGURE 44. COMPARISON OF CAPSULE CONFIGURATIONS

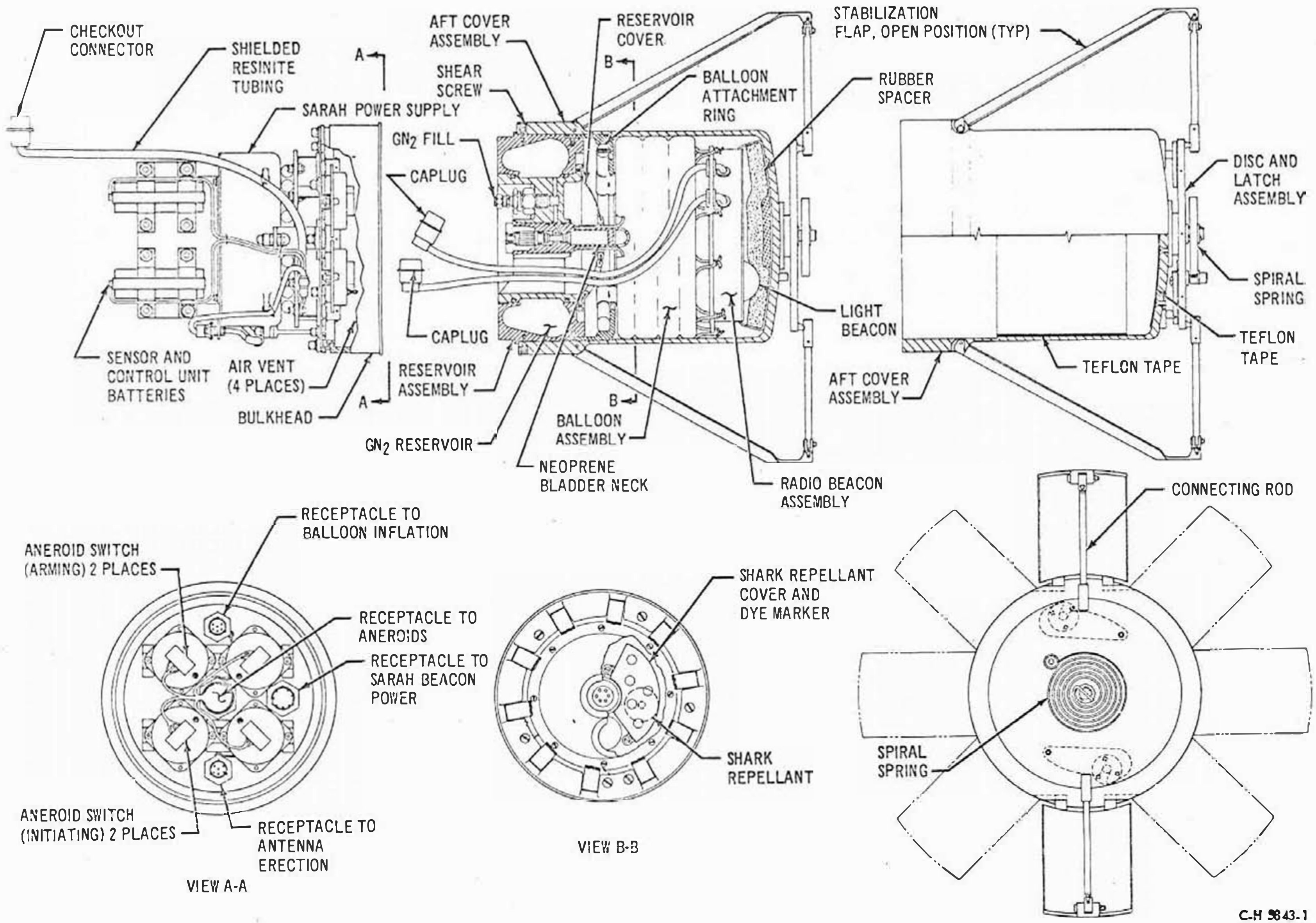


FIGURE 45. RECOVERY SYSTEM ASSEMBLIES

The camera operates on vehicle power through an electrical quick-disconnect plug (figure 36). The vehicle power receptacle is mounted on a hinged arm (in the ejection tube) that connects with the camera electrical disconnect plug through an opening in the capsule, and disconnects at ejection.

The ejection finger block is a rectangular slot in the capsule. The finger of the ejection mechanism is inserted into this slot when the capsule in installed in the ejection tube.

The GN_2 reservoir and sensor and control unit for the recovery systems are mounted on the aft bulkhead of the camera compartment. Checkout of the sensor and control unit is accomplished through a checkout connector on the capsule. At all times before flight, except during checkout, a safety shorting plug is inserted in this connector.

D. RECOVERY SYSTEMS COMPARTMENT

The recovery systems compartment, the aft portion of the capsule, occupies approximately 1/3 of total capsule length. This compartment houses all the capsule recovery aids (figure 45), which provide:

a. Descent stabilization (stabilization flaps).

b. Descent deceleration (paraballoon).

c. Flotation (paraballoon).

d. Location (radio transmitter and antenna, light beacon, sea-marker dye).

e. Safety (shark repellent).

The GN_2 reservoir for the paraballoon and for the sensor and control unit, which controls the operation of the recovery systems, is located on the aft bulkhead of the camera compartment.

1. Stabilization Flaps

Until ejection, confinement in the ejection tube holds the eight stainless steel stabilization flaps closed. At ejection, a spring loaded disc and latch assembly releases, extending and locking the flaps at a 0.523 rad angle to the capsule axis (figures 37 and 46). This forms a slotted frustum afterbody on the capsule, thereby stabilizing the capsule at reentry and maintaining the capsule at a zero angle of attack up to the point of paraballoon deployment. The force of paraballoon inflation shears the eight cover assembly screws, causing the entire aft cover assembly, including the stabilization flaps, to fall free of the capsule.

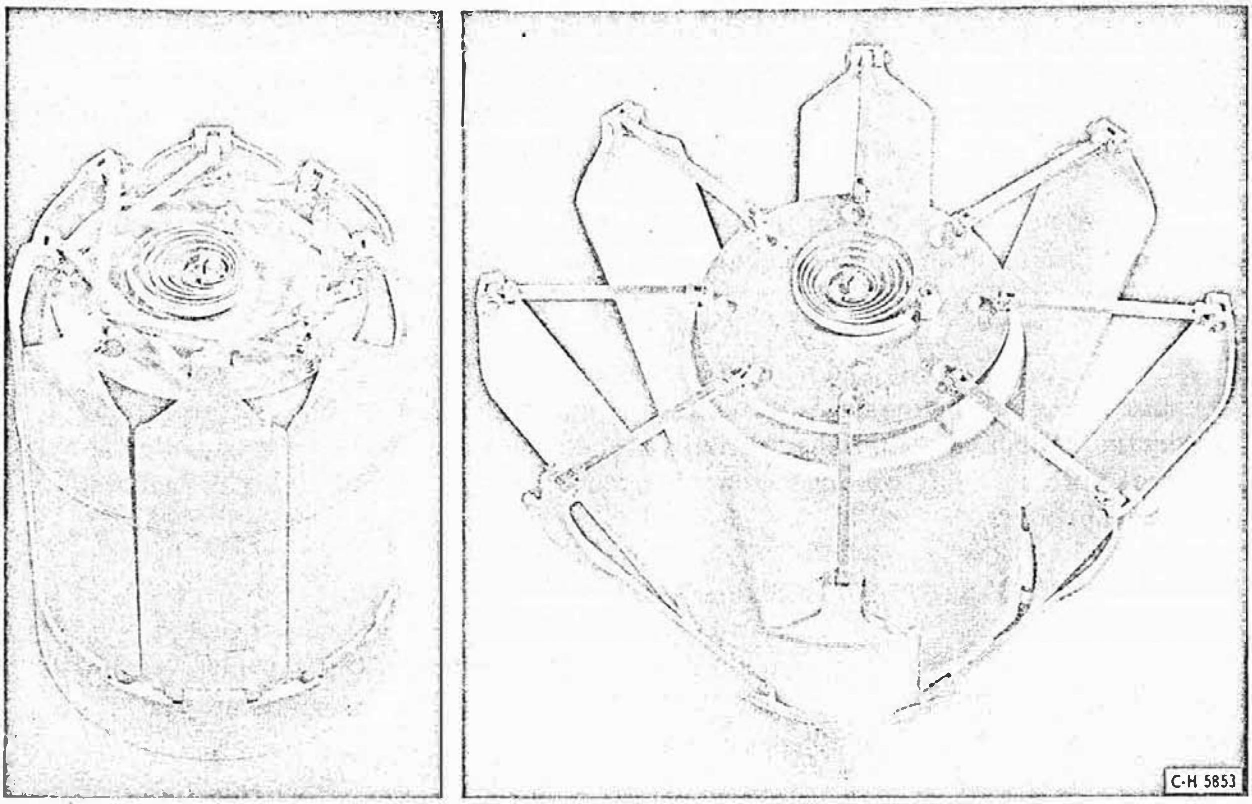


FIGURE 46. STABILIZATION FLAPS, CLOSED AND OPEN POSITIONS

2. Paraballoon

The paraballoon provides deceleration drag for the capsule during descent and holds the capsule afloat during recovery operations. The para-baloon is folded and packaged in the aft portion of the recovery compartment until deployed at 4,300 m during descent. The paraballoon assembly consists of a reinforced nylon cloth outer cover, a nylon bladder impregnated with neoprene rubber, and a thin nylon para-drag skirt that fits around the center of the balloon (figure 47).

The paraballoon is the sole means for capsule flotation after impact. The top of the paraballoon provides an above-the-waterline platform for the flashing light beacon and the radio transmitter and antenna (figure 48). The upper hemisphere of the balloon cover has alternating panels of white and dayglow orange to increase visibility during recovery; the lower hemisphere is a dark purple that is unattractive to sea life.

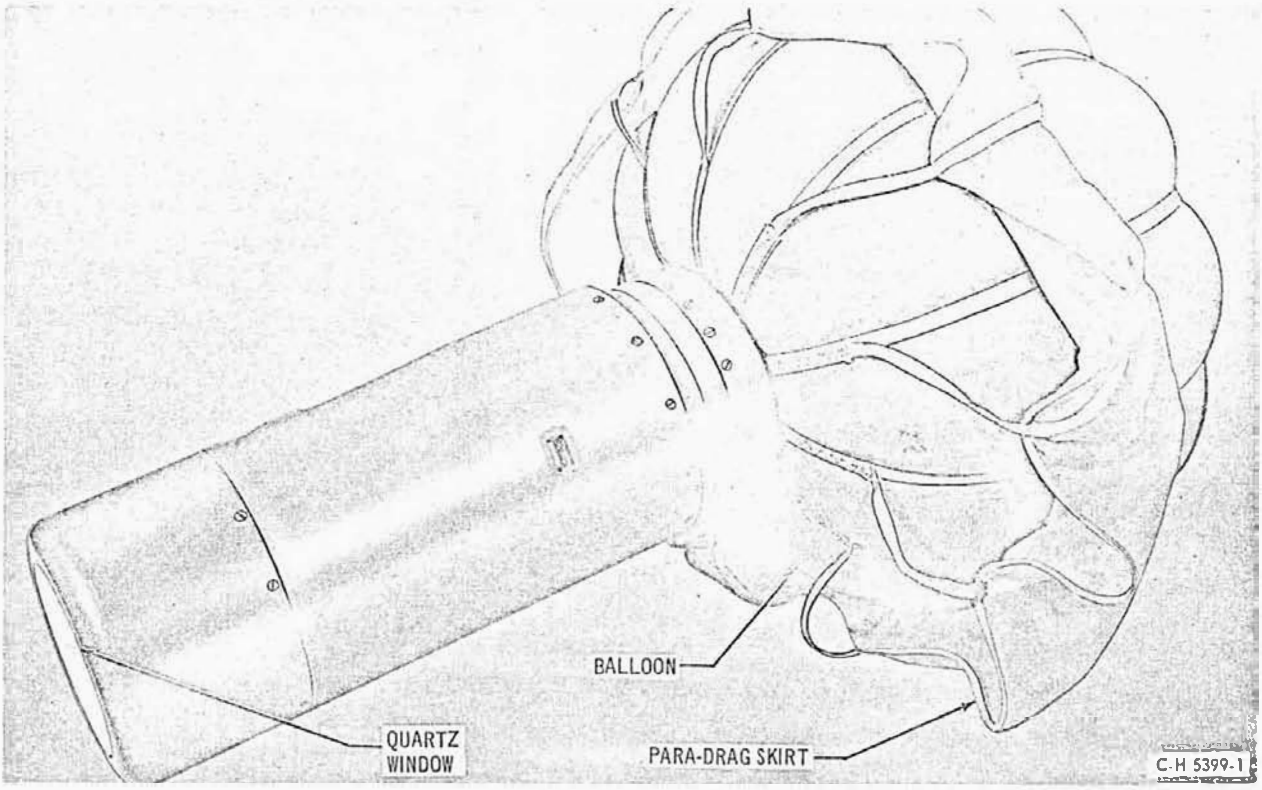


FIGURE 47. CAPSULE WITH PARABALLOON DEPLOYED

3. Shark Repellent

An 85-gram, water soluble plug of cupric acetate is packaged in a metering container in the recovery compartment, and is metered out to the surrounding water after capsule impact for a period of 2 to 4 hours, depending on water temperature and sea state. Sharks and other sea life that might damage the paraballoon, capsule, or recovery team are repelled by this cupric acetate.

4. Radio Transmitter and Antenna

The primary capsule location aid is a SARAH 311026-001 radio transmitter located on the paraballoon (figure 48). The radio transmitter case includes a self-erecting spring steel antenna that is deployed 6 seconds after the capsule reaches 4,300 meters during descent, when the capsule sensor and control unit fires a squib release on the antenna cover.

A SARAH power supply, packaged separately inside the capsule shell, contains a single dc-to-dc converter for operation of both the transmitter and the flashing light. Five stacks of three Mallory RM-1450 RS cells are connected

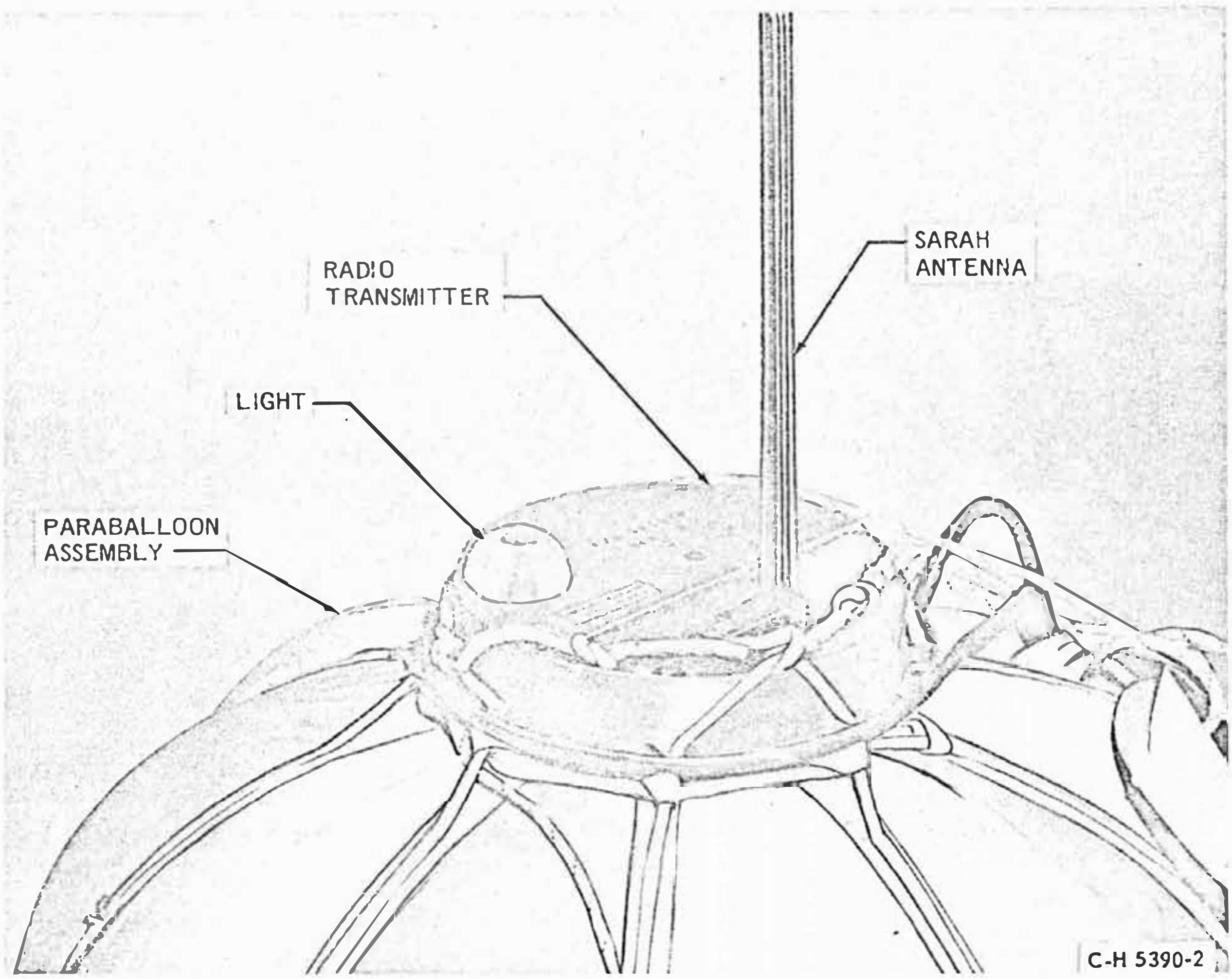


FIGURE 48. PARABALLOON PLATFORM

to provide 7.1 V to the converter for a period of 20 hours. The converter output is 500 V.

The radio transmits a characteristic double-pulsed SARAH signal at 242.0 ± 2 MHz at a pulse group repetition rate of 200 groups. The RF power output is approximately 15 W peak envelope power. The transmitted signal aids sea and air recovery craft to obtain a directional bearing on the capsule during descent and after impact. The radio signal is the only recovery aid for both long and short range, day and night searching.

A transistorized, crystal-controlled, tone-modulated, continuous-wave (CW) transmitter was flown on SA-5 in place of one SARAH beacon. The Saturn IB vehicles will continue to use the SARAH beacons, but Saturn V requirements will be met with these CW beacons because the SARAH beacons are being phased out by Patrick Air Force Base.

5. Flashing Light

A high-intensity light beacon located on the radio transmitter receives its power from the dc-to-dc converter in the SARAH power supply. Flashing at a rate of 40 flashes per minute, the light has a life of approximately 20 hours.

6. Sea-Marker Dye

Fifty seven grams of yellow-green fluorescein dye packaged in carbowax are located in a metering compartment at the base of the paraballoon, below the water line. When the container is exposed to salt water after impact, the dye is metered out over a period of four to six hours, depending on water temperature and sea state. Four parcels of fluorescein dye (a total of 84 g) are placed in the aneroid compartment. This dye is packaged in starched paper; when the sea water flows through the compartment, the starch dissolves and the dye flows through the porous paper to mark the surrounding waters.

E. CAPSULE FLIGHT AND RECOVERY

This section describes the events that take place from capsule ejection until capsule recovery.

1. Ejection

Approximately 25 s after stage separation, camera capsule ejection takes place. Each capsule travels in an upward and outward direction at approximately 0.35 rad from the vehicle centerline with a velocity of 8 m/s relative to the ejection tube; S-I stage velocity at this time is approximately 2,600 m/s. To prevent collision after ejection, there is a difference of 0.044 rad (with respect to the spider beam members) in the ejection angles of the model A and model B capsules. The ejection system is described in detail in Section III.

The stabilization flaps, held flush with the sides of the capsule while in the ejection tube, deploy and lock at an angle of 0.523 rad to the longitudinal axis of the capsule to provide stabilization at ejection. Until the paraballoon is deployed, the capsule follows a ballistic trajectory.

2. Flight

During capsule flight a sensor and control unit located in the camera compartment initiates events in the recovery sequence. The basic components are aneroid switches, squibs, and pyrotechnic time delay devices. The sensor and control unit is schematically shown in figure 49, and the sequence of operation of the unit is shown in figure 50.

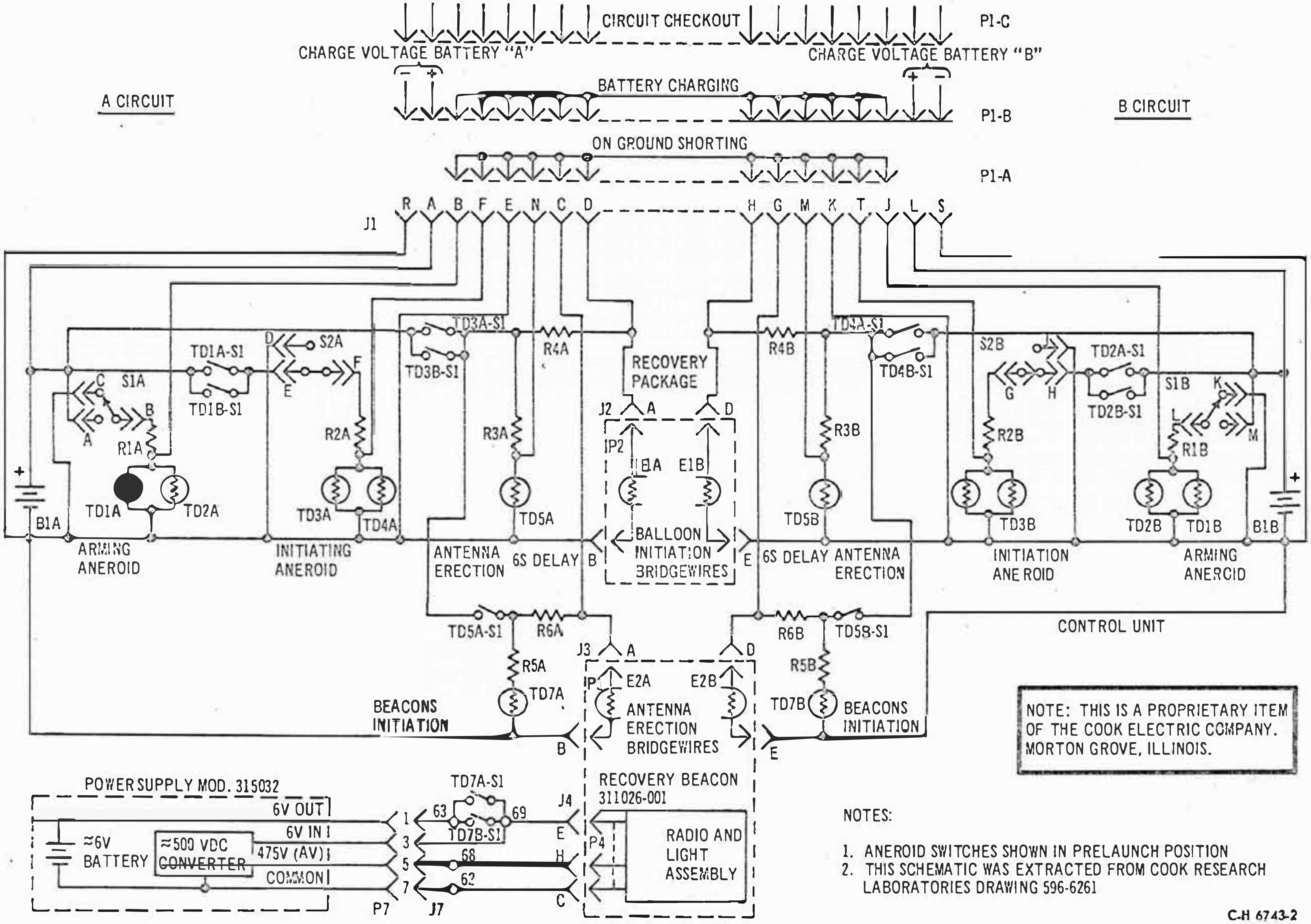


FIGURE 49. SENSOR AND CONTROL UNIT--SCHEMATIC

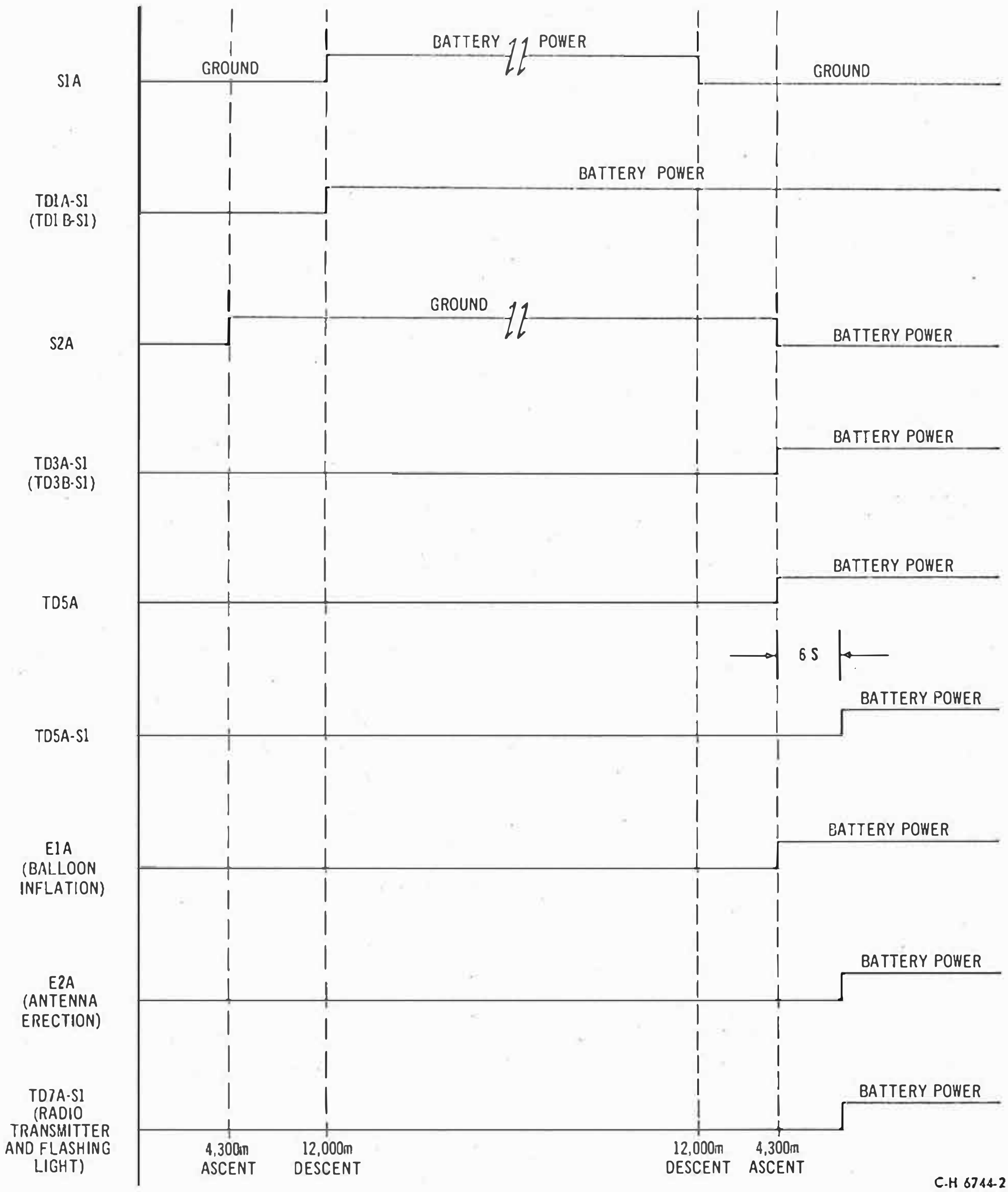


FIGURE 50. SENSOR AND CONTROL UNIT OPERATION SEQUENCE

The control unit uses two duplicate circuits, either of which is capable of controlling and initiating the various recovery functions. Where practical, components are duplicated within each circuit. The following description pertains to circuit A; circuit B operates in the same manner.

A single on-ground shorting plug (P1A) provides ground safety. A single inflight arming plug (P1B) brings battery power to S1A, TD1A-S1, and TD3-S1.

The first circuit action occurs at approximately 4,300 m altitude during vehicle ascent. At this time, initiating aneroid S2A actuates and grounds the parallel combination of TD3A and TD4A in series with R2A. At an altitude of 12,000 m the arming aneroid actuates, allowing battery power from B1A to fire both TD1A and TD2A. The resulting action of TD1A-S1 brings battery power to the ungrounded side of S2A. An inconsequential action of S1A occurs at 12,000 m during descent; TD1A-S1 remains closed because of the single-action nature of TD1A. All other bridgewire switches and pyrotechnic devices employed in these circuits are single-action type. At an altitude of 4,300 m after reentry, the initiating aneroid S2A returns to its original prelaunch position, allowing battery power to fire both TD3A and TD4A. The closing action of TD3A-S1 allows the firing of TD5A and E1A. The firing of E1A initiates balloon inflation.

The paraballoon is located behind the aft bulkhead of the camera compartment. During inflation the pressure of the balloons shears the eight cover assembly screws, allowing the entire aft cover assembly of the capsule to fall free. The paraballoon then inflates to a diameter of 46 cm to stabilize and decelerate the capsule so that terminal velocity is about 50 m/s.

After a 6 s delay from the time of firing of TD5A, TD5A-S1 closes, firing both TD7A-S1 and E2A. The latter two actions permit the powering of the radio transmitter and flashing light circuits (TD7A-S1) and the initiating of the antenna erection cycle (E2A).

Table 4 presents a detailed description of a typical capsule flight. Various parameters are given so that an analysis of the flight environment of the capsule can be made. The actual values of a capsule flight depend upon the vehicle flight, wind conditions, etc.; the data in Table 4 show only one typical flight.

3. Impact Area

Between the time of capsule ejection and the opening of the paraballoon, the capsule follows a ballistic trajectory that primarily determines the impact

TABLE 4. TYPICAL CAMERA CAPSULE FLIGHT ENVIRONMENT

Time (s)	Altitude (km)	Range (km)	Mach No.	Dynamic Pressure (N/m^2)	Velocity (m/s)	Acceleration (m/s^2)	Angle of Attack (rad)
*147	73.43	92.33	9.54	205	2670	-4.65	1.12
157	84.73	115.98	10.18	26.4	2627	-4.01	1.15
167	95.16	139.54	10.03	2.94	2589	-3.68	1.18
**172							
397	99.14	674.07	9.98	0.98	2574	+3.56	1.95
407	89.06	697.60	10.12	10.8	2611	+3.82	1.97
417	78.12	721.21	10.02	97.0	2650	+3.91	2.02
427	66.32	744.87	8.91	554	2687	+3.21	2.04
437	53.71	768.50	8.04	2,390	2705	-0.42	2.06
447	40.53	791.68	8.08	1,290	2622	-22.31	2.09
457	28.15	812.11	6.74	50,600	2034	-103.27	2.13
467	20.12	824.32	3.16	38,500	920	-92.68	2.18
477	16.62	828.78	1.12	8,550	318	-30.26	2.33
487	14.73	830.34	0.72	4,930	208	-4.59	2.58
497	13.02	831.16	0.61	4,580	178	-2.10	2.79
507	11.39	831.60	0.53	4,580	160	-1.54	2.95
517	9.88	831.82	0.48	4,570	146	-1.25	3.04
527	8.48	831.92	0.43	4,540	135	-1.04	3.09
537	7.18	831.96	0.39	4,490	125	-0.88	3.11
547	5.97	831.98	0.36	4,450	117	-0.76	3.12
557	4.84	831.99	0.33	4,420	110	-0.67	3.13

* Separation
** Capsule Ejection

TABLE 4. TYPICAL CAMERA CAPSULE FLIGHT ENVIRONMENT (CONT.)

Time (s)	Altitude (km)	Range (km)	Mach No.	Dynamic Pressure (N/m^2)	Velocity (m/s)	Acceleration (m/s^2)	Angle of Attack (rad)
***561	4.40	831.99	0.32	4,410	107	-0.63	3.13
567	3.77	832.00	0.31	4,390	104	-0.58	3.14
577	2.76	832.00	0.29	4,370	98	-0.51	3.14
587	1.80	832.00	0.27	4,340	93	-0.45	3.14
597	0.89	832.00	0.26	4,330	89	-0.40	3.14
607	0.02	832.00	0.25	4,310	85	-0.34	3.14
**** 607.218	0.00	832.00	0.25	4,310	85	-0.34	3.14

*** Paraballoon deployment
**** Impact

area. Paraballoon inflation at 4,300 m increases the drag area and reduces capsule velocity. The graph of capsule trajectory for the SA-5 flight is presented in figure 51.

The impact area for an eight-engine SA-5 flight occurs at 26 degrees north geodetic latitude and 72 degrees west longitude (figure 52). The impact point can drift in the direction of the wind nearly 17 km prior to paraballoon deployment, and an additional 2.5 km after deployment, under influence of a wind with a velocity within the standard deviation for that area. An engine failure can shorten the range at impact by as much as 200 km. The impact areas resulting from engine failure at various times during S-I stage powered flight for SA-5 are presented in figure 53.

4. Recovery Operations

a. Weather and Sea State Limitations. Weather and ocean-surface conditions during search and recovery operations greatly affect the probability of recovery. High seas, stormy weather, and low visibility could make recovery impossible. AFMTC has established a Code 4 (International Scale) sea state as the worst weather conditions under which recovery operation will be attempted. Some conditions of a Code 4 sea state are a wind of 31 to 38 km/hr, and moderate waves with an average of 1.5 m and many white caps. Sea conditions at the expected impact area are Code 4 or better approximately 90 percent of the time.

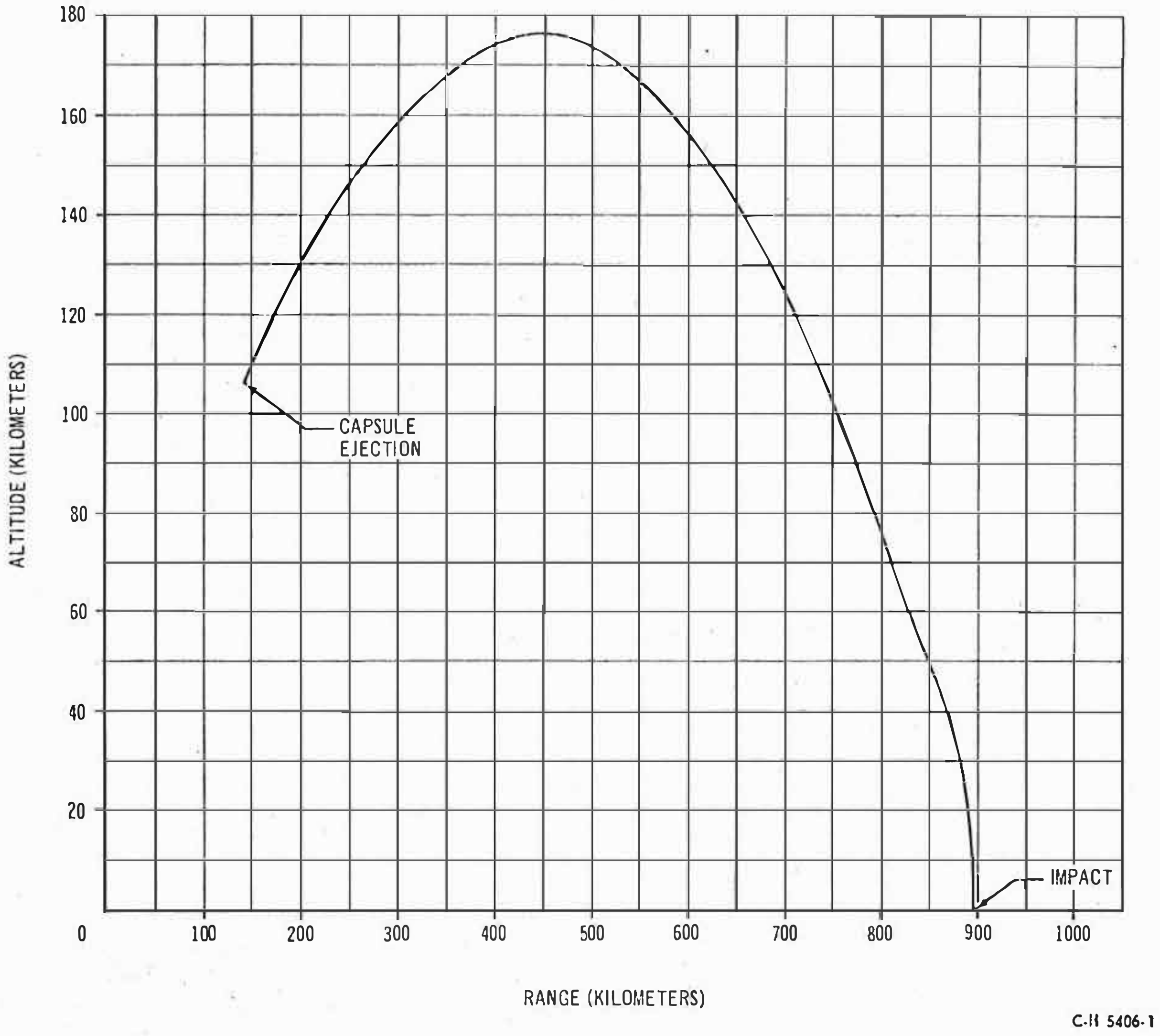


FIGURE 51. SA-5 CAPSULE TRAJECTORY

b. Standard Recovery Operations. Both air and sea recovery teams participate in capsule recovery operations. Before flight, a JC-54G aircraft is assigned to a station approximately 24 km from the predicted impact area, while a ship is stationed 16 km downrange from the impact area.

When the impact area has been determined, both the aircraft and ship begin recovery operations. In the event that a bearing cannot be established within a reasonable time, the aircraft begins a systematic search from an altitude of 1,500 to 2,500 m. If the capsules are located before the ship reaches the area, the aircraft summons the ship and descends to a lower altitude to maintain SARAH-transmitter contact and visual surveillance.

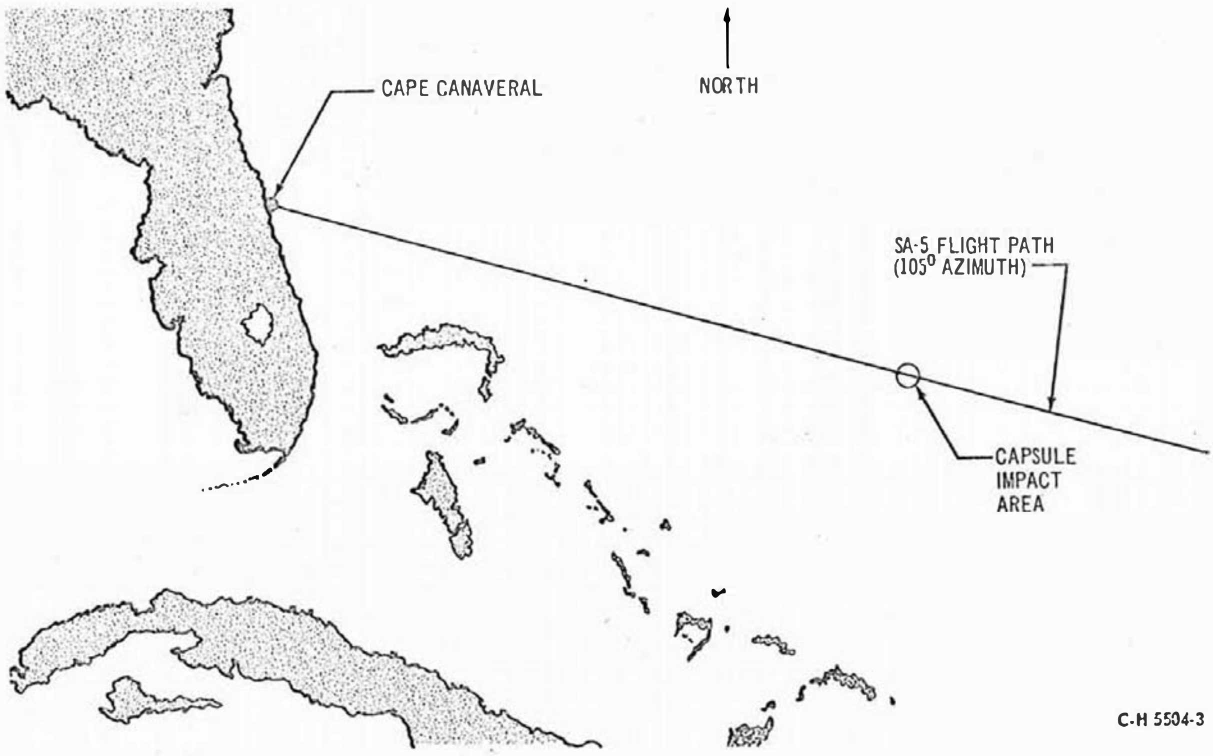


FIGURE 52. CAPSULE IMPACT AREA, SA-5 FLIGHT

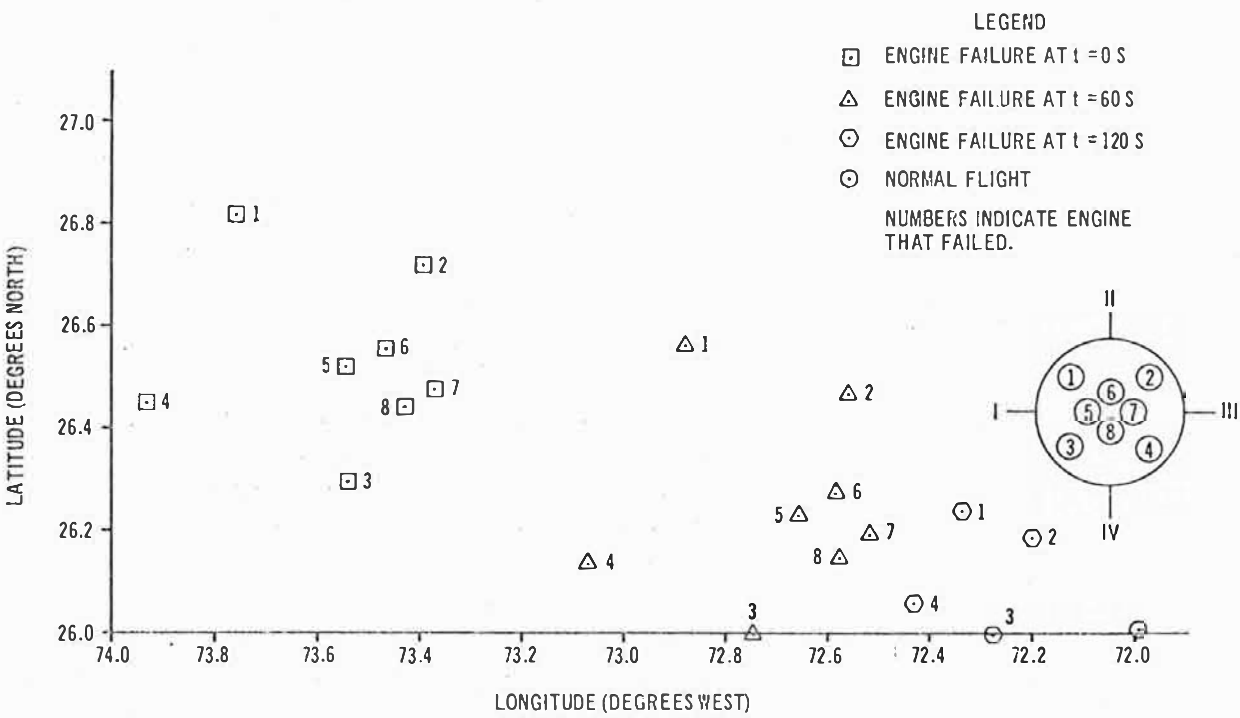


FIGURE 53. IMPACT AREAS CAUSED BY ENGINE FAILURE

c. Special Recovery Operations. If a deviation from the scheduled vehicle flight path is anticipated, the new capsule impact area is computed and the search aircraft is redirected.

The rescue ship, previously positioned for standard recovery operations, proceeds to the newly computed impact area (maximum travel time nine hours). Although the capsule is designed for a 20-hour flotation period, the recognized vulnerability of the capsule would probably dictate aircraft pararescue diver recovery, thereby effecting recovery within 1 to 2 hours after impact.

5. Probability of Recovery

The following factors affect the probability of a successful recovery:

a. Atmospheric conditions.

b. Saturn IB reliability.

c. Ejection system reliability.

d. Capsule flotation system reliability.

e. Radio beacon reliability.

f. Sea conditions.

g. Search and recovery efficiency.

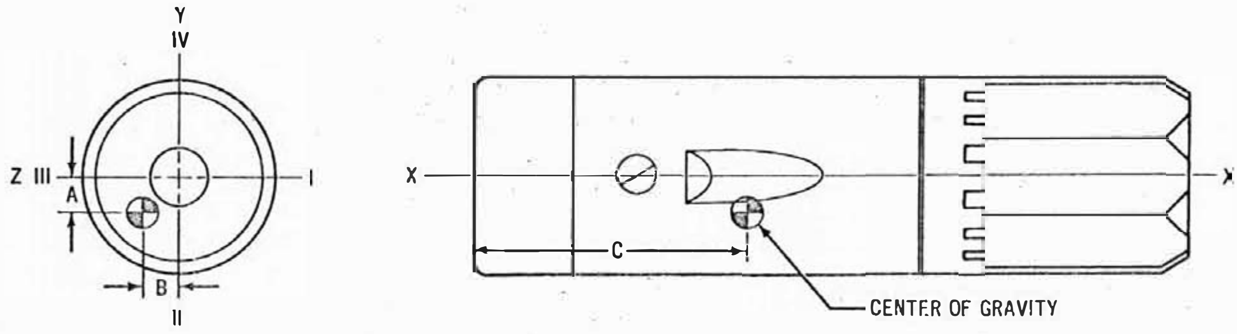
Improvement in any of these factors will improve the overall probability of camera capsule recovery.

F. DESIGN AND TESTING

A series of tests and analyses to determine the final capsule characteristics became necessary once a preliminary functional capsule design was decided upon. Preliminary analyses were run, and the mass characteristics, aerodynamic properties, stability, temperature effects, and trajectory were investigated. The capsules also underwent three series of flight tests before the SA-5 flight.

1. Capsule Mass Characteristics

The mass of a flight ready model A capsule was calculated on a shadowgraph balance scale to an accuracy of 6.81 gm. A two-point suspension method was used to determine the longitudinal center of gravity, and a balance machine was used to determine the radial center of gravity. The mass moment-of-inertia data were obtained by use of a torsional pendulum system. The mass characteristics of the flight-ready model A capsule are shown in figure 54.



WEIGHT (Kgs)	A (cm)	B (cm)	C (cm)	Ixx*	Iyy*	Izz*
26.1	0.13	0.43	33.3	1.41	13.5	13.6

*INERTIA UNITS: Kgs - cm - seconds2

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FIGURE 54. CAPSULE MASS CHARACTERISTICS

2. Wind Tunnel Tests

The Aero-Astrodynamics Laboratory (R-AERO) wind-tunnel tested a stainless steel, 0.1613-to-1 scale model capsule (figure 55) for stability and coefficient of drag through the Mach number range of 0.40 to 4.96. Typical shadow-graphs of the scale model under test in the R-AERO transonic facility are shown in figure 56.

The curves of parameters that influence configuration stability are presented in figure 57. The capsule is statically stable about a center of gravity 1.7 calibers from the base. The curves shown indicate that the maximum dynamic stability margin exists at approximately Mach 1, and the minimum stability margin exists in the Mach number range from 1.1 to 1.9.

Figure 58 presents curves of the relationship between Mach number and coefficient of drag, and agrees closely with similar computations performed by Cook Technological Center.

3. Capsule Heating Analyses and Tests

To determine reentry heating on critical portions of the camera capsules, heating analyses and tests were performed.

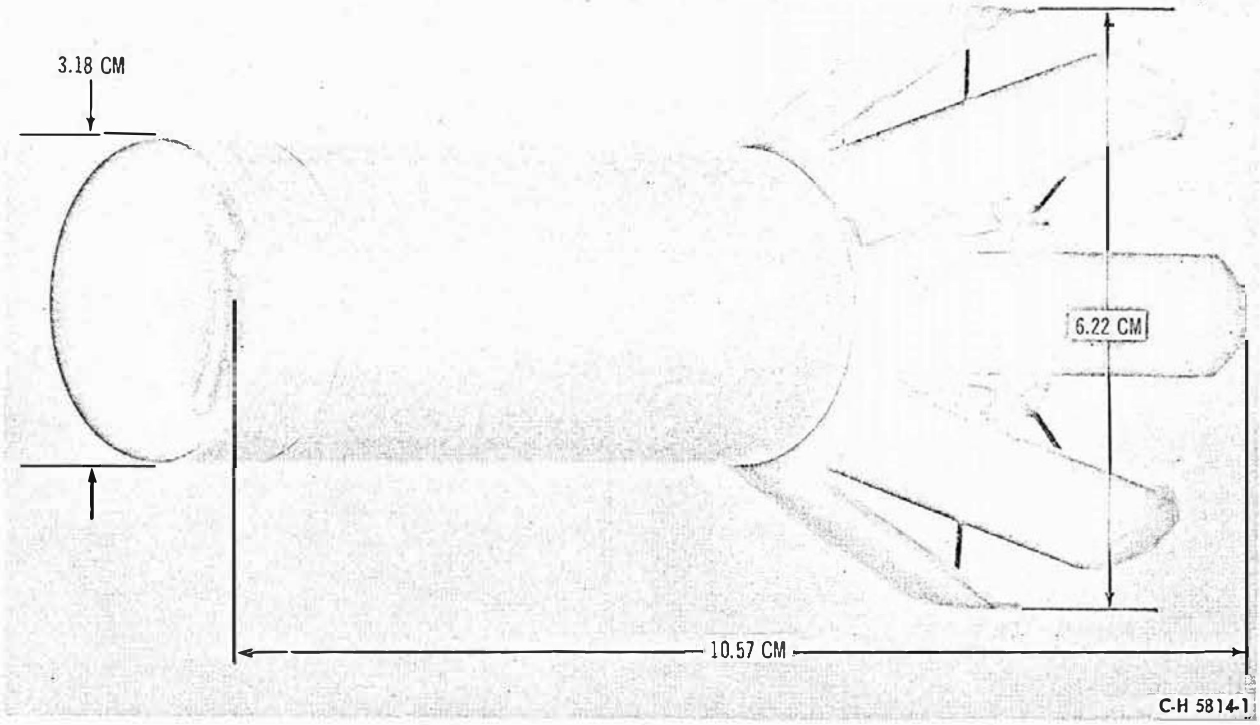

FIGURE 55. WIND TUNNEL TEST CAPSULE

a. Predicted Capsule Temperatures. Predicted temperature histories were prepared for critical areas of the camera capsules. The heat transfer study predicted maximum component temperatures during ascent, coast, and reentry.

The temperature history of the quartz window (figure 59) shows that maximum anticipated exterior surface temperatures are 600°C during ascent and 580°C during reentry. Inside surface temperatures reach a maximum of 204°C during reentry, but only 66°C during the period of camera operation.

The stainless steel retainer ring history (figure 60) shows a maximum exterior surface temperature of 121°C during ascent and 275°C during reentry. In close association with the retainer ring and quartz window, the Teflon seals (figure 61) are predicted to reach a maximum of 272°C during reentry.

The 0.318 cm aluminum camera container skin and the 0.318 cm stainless steel stabilization flaps are protected during ascent. Their predicted temperatures (figure 62) become significant during late coast and reentry. Skin temperature is expected to reach a maximum of 100°C without insulation, and stabilization flap temperature is expected to reach 345°C.

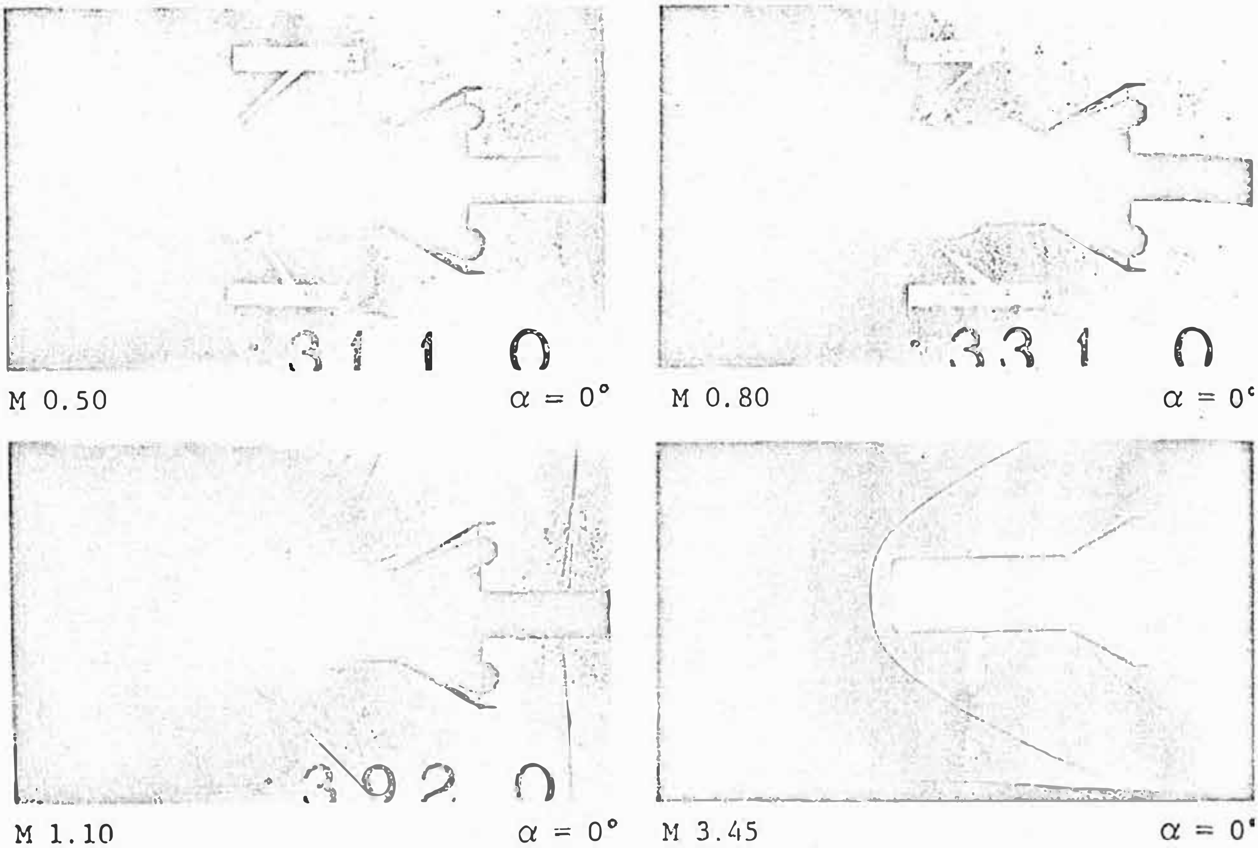

FIGURE 56. SHADOWGRAPHS OF TEST CAPSULE IN MSFC TRANSONIC FACILITY

Cook Technological Center performed temperature calculations to determine the maximum temperature on the aluminum skin. As a result of this analysis, 0.635 cm fiberglass with Mylar liner is placed on the inside of the aluminum skin for insulation. With this liner, the maximum inside skin temperature will be 33°C.

b. Simulated Capsule Temperatures. Simulated aerodynamic heating tests were conducted to duplicate closely the ascent, coast, and reentry environment, and to measure component temperatures. The temperature-time histories were programmed into three control sections of the capsule structure through the use of infrared radiant heat furnaces (figure 63). Results of this test, which agree with the predicted temperature histories, are given in figure 64.

As a result of the large thermal gradients shown by the previous computations and tests, a test to check quartz characteristics under high thermal gradients was recommended. The Pilot Manufacturing Development Branch (R-ASTR-P) performed this test, which showed that no problems existed.

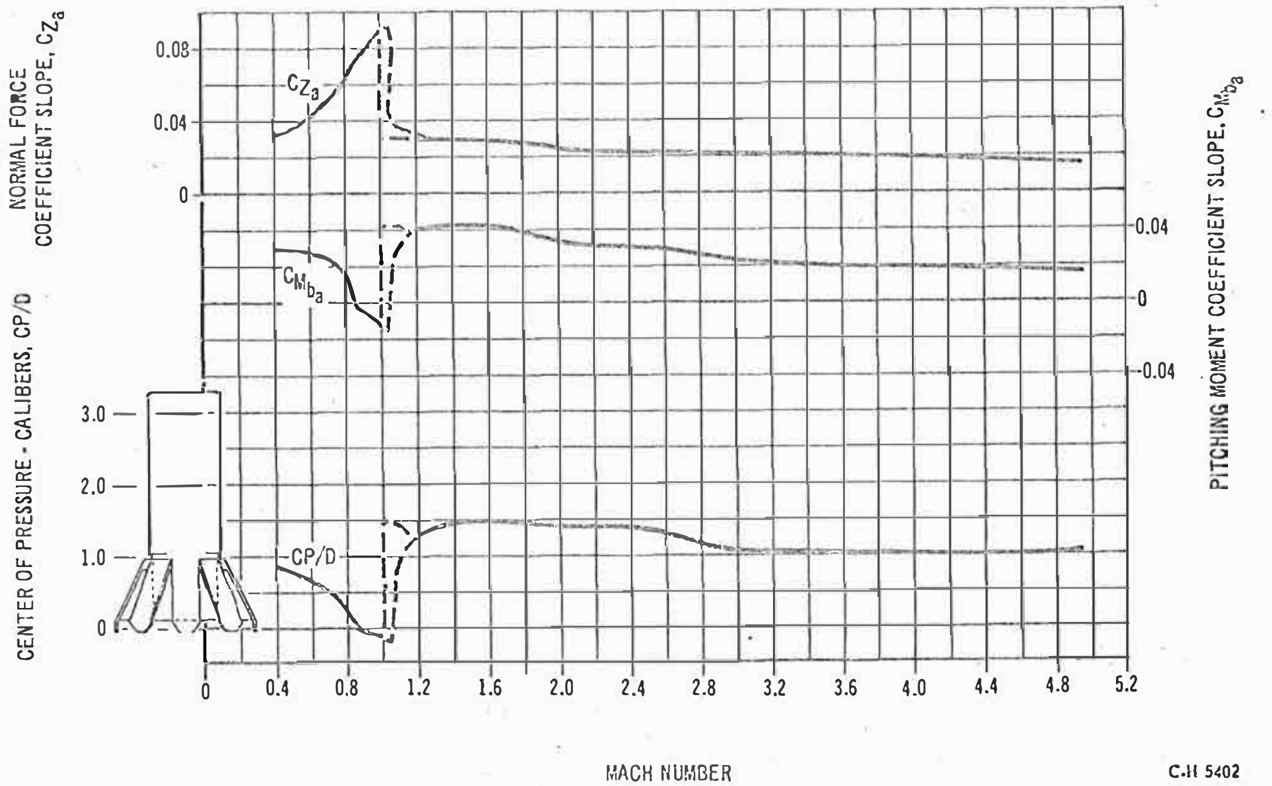

FIGURE 57. RELATIONSHIP OF CAPSULE AERODYNAMIC CHARACTERISTICS TO MACH NUMBER

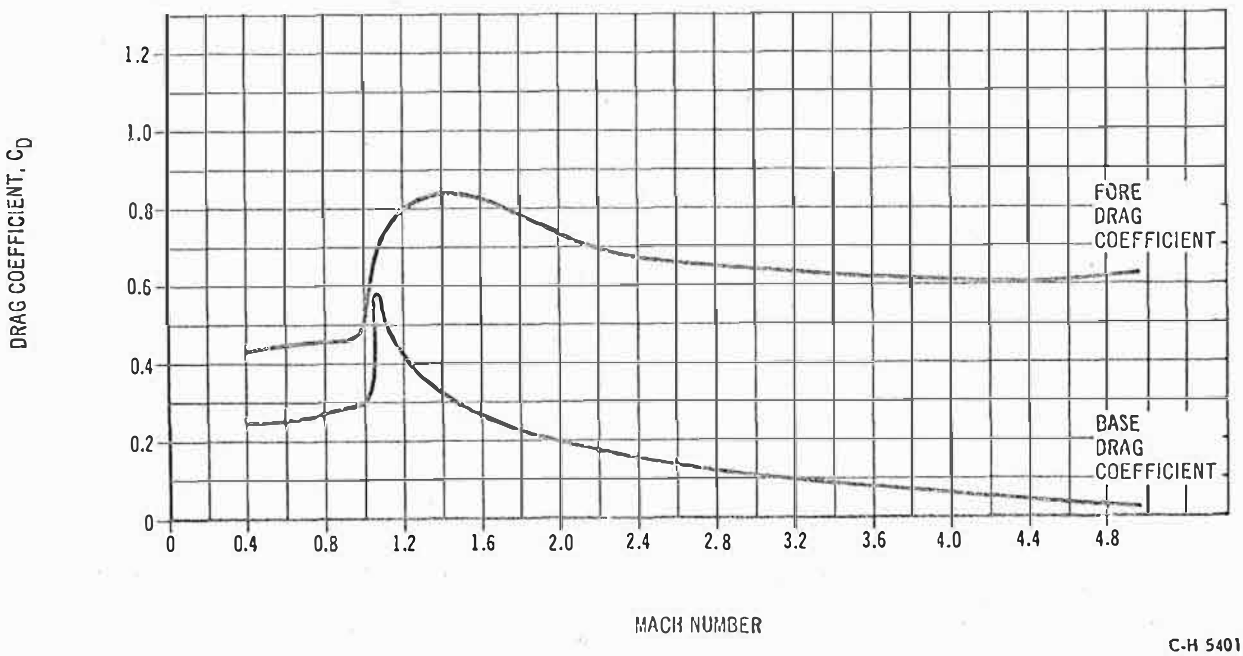

FIGURE 58. RELATIONSHIP OF CAPSULE DRAG CHARACTERISTICS TO MACH NUMBER

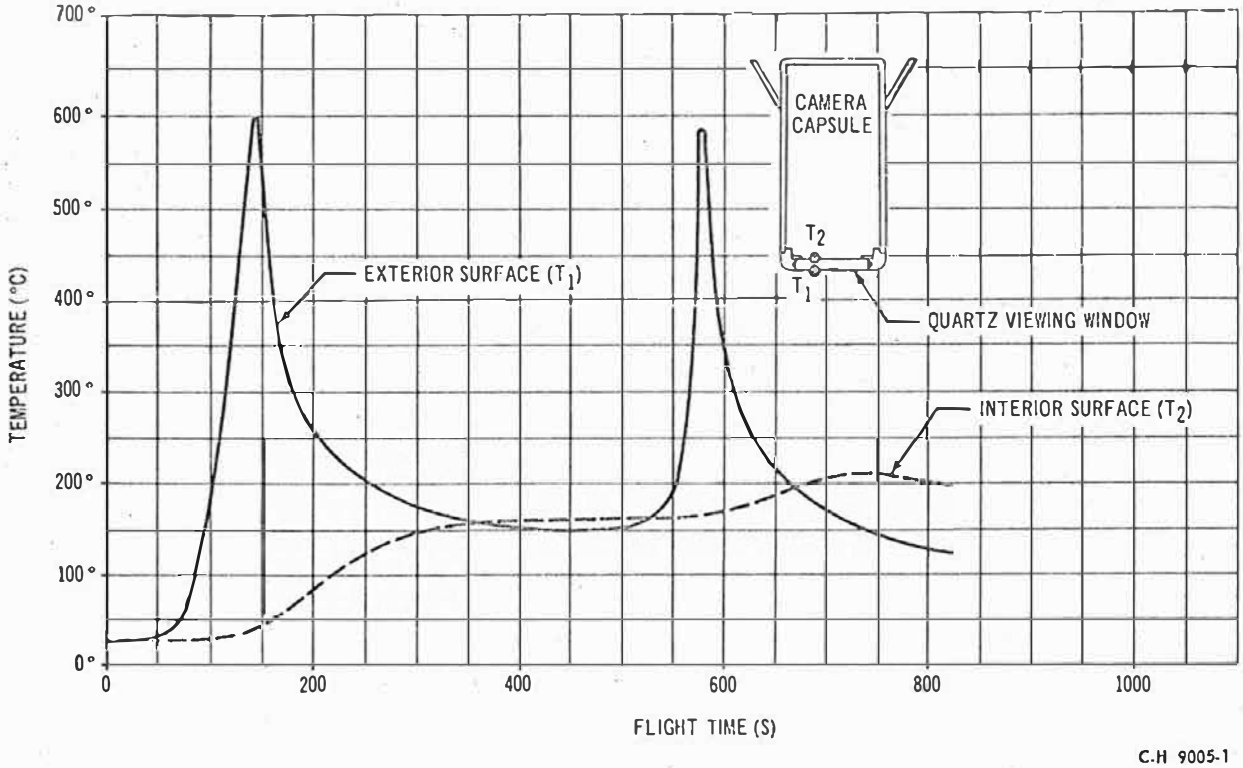

FIGURE 59. PREDICTED QUARTZ WINDOW FLIGHT TEMPERATURES

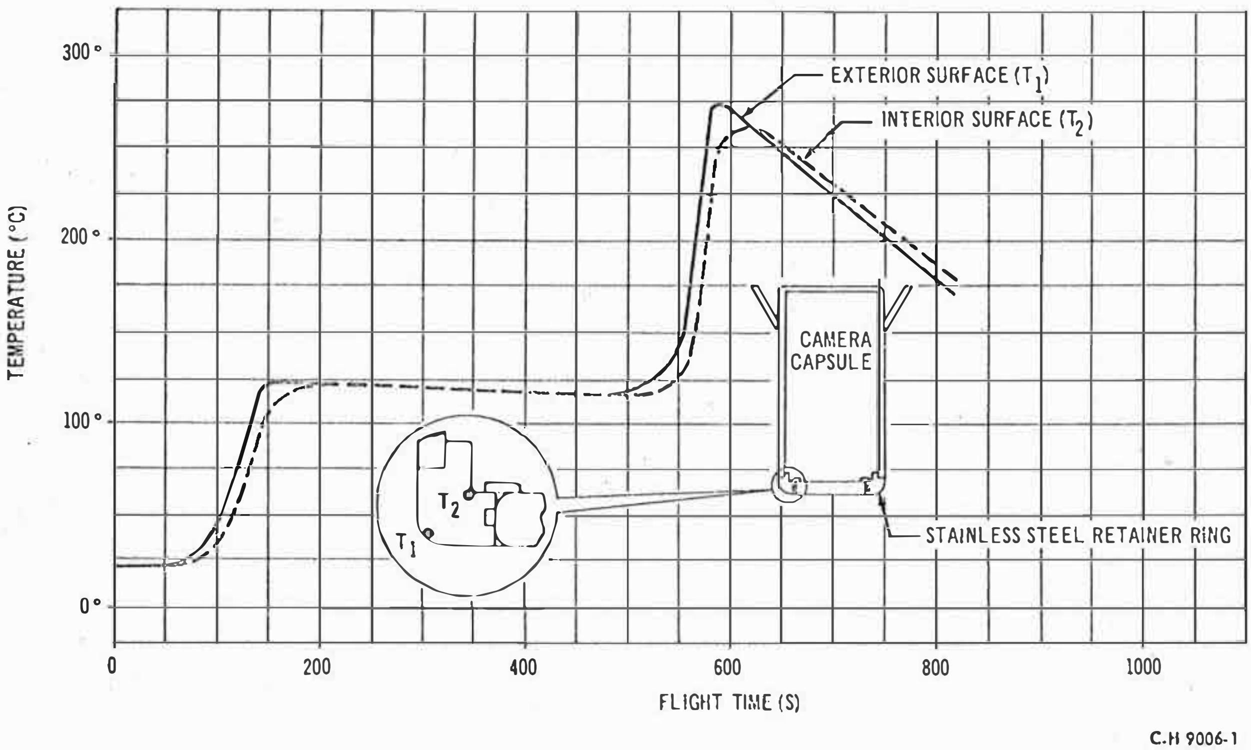

FIGURE 60. PREDICTED RETAINER RING FLIGHT TEMPERATURES

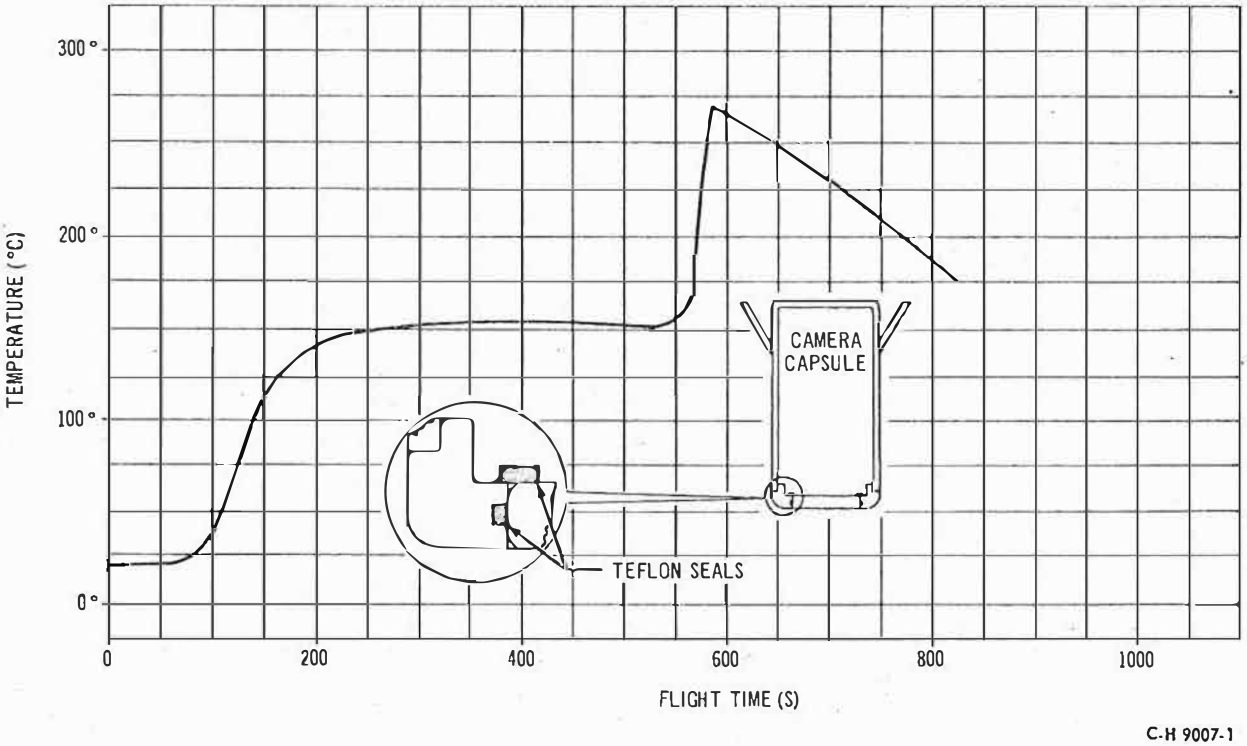

FIGURE 61. PREDICTED TEFLON SEALS FLIGHT TEMPERATURES

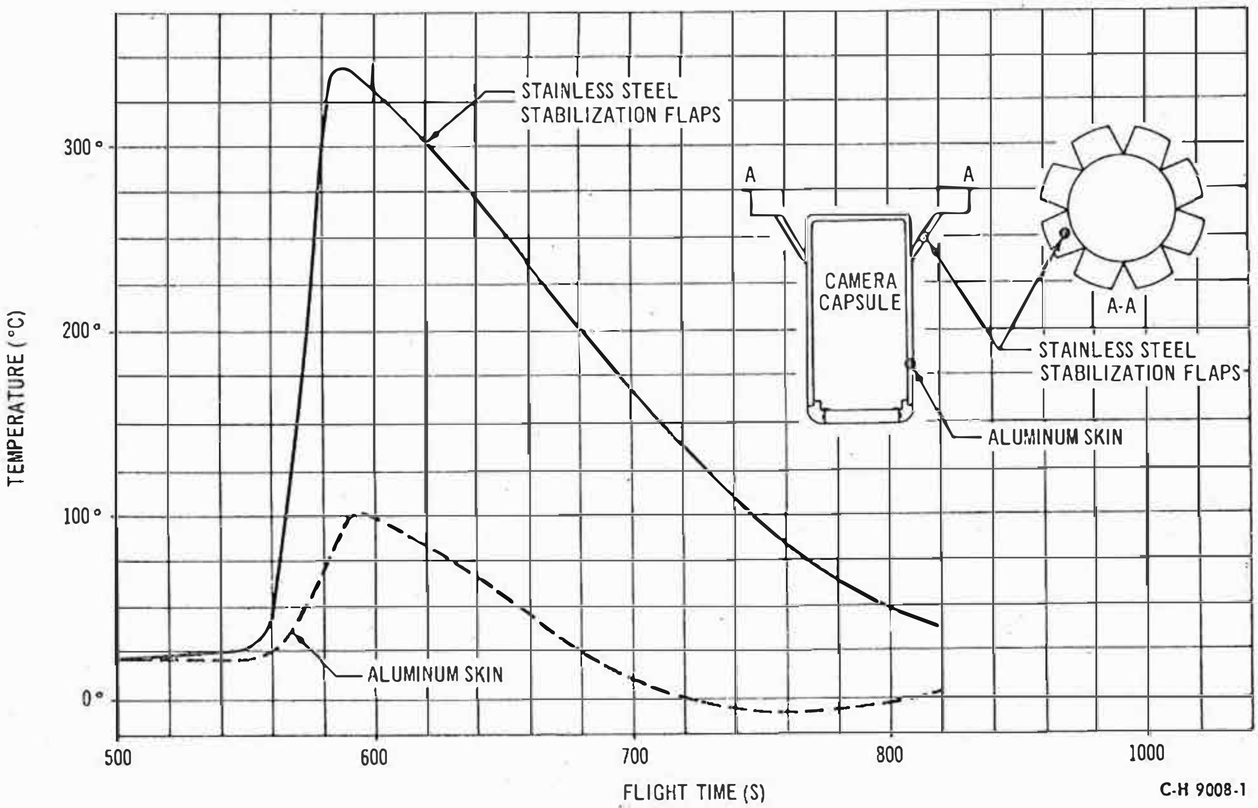

FIGURE 62. PREDICTED CAPSULE SHELL AND STABILIZATION FLAPS FLIGHT TEMPERATURES

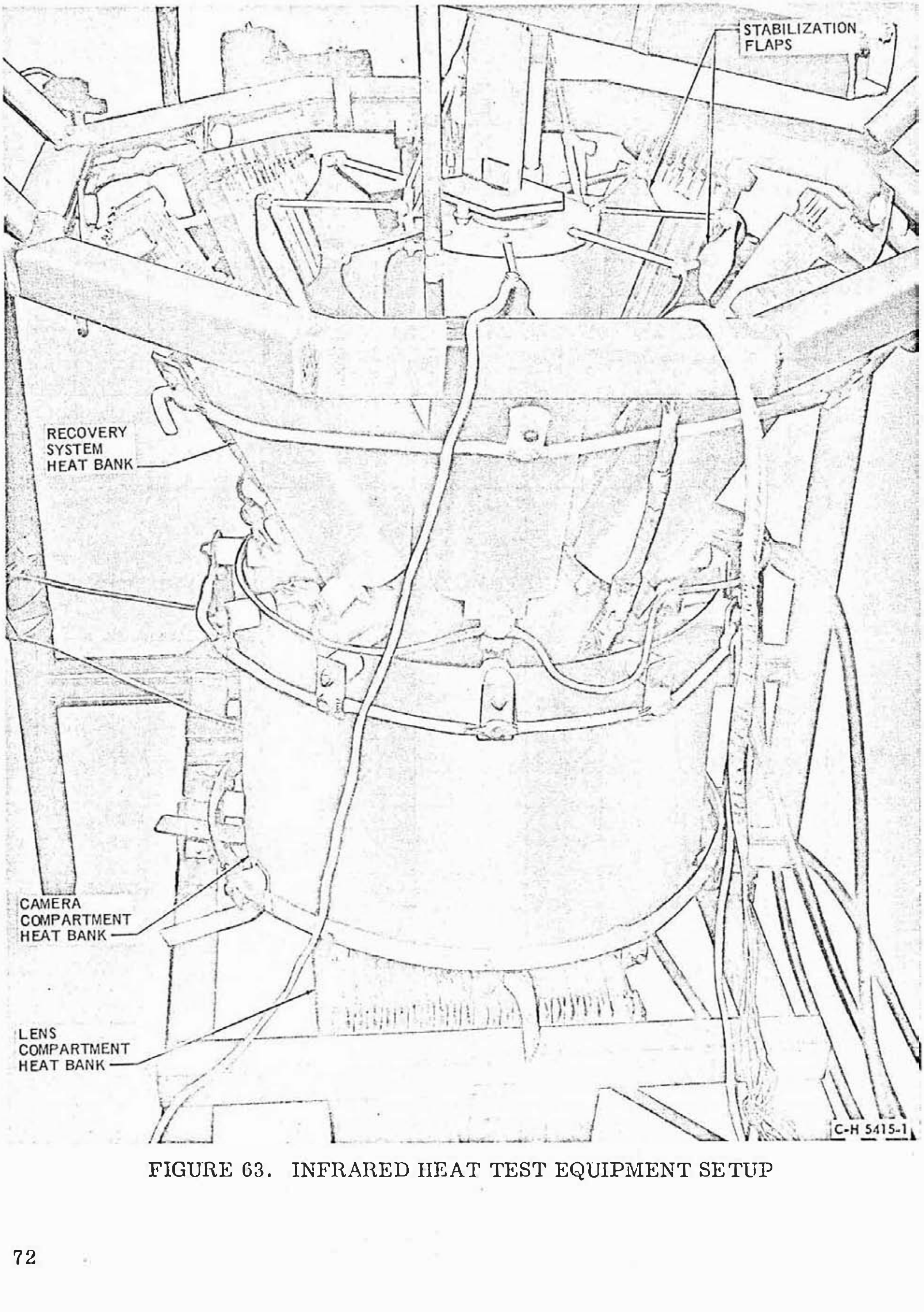

FIGURE 63. INFRARED HEAT TEST EQUIPMENT SETUP

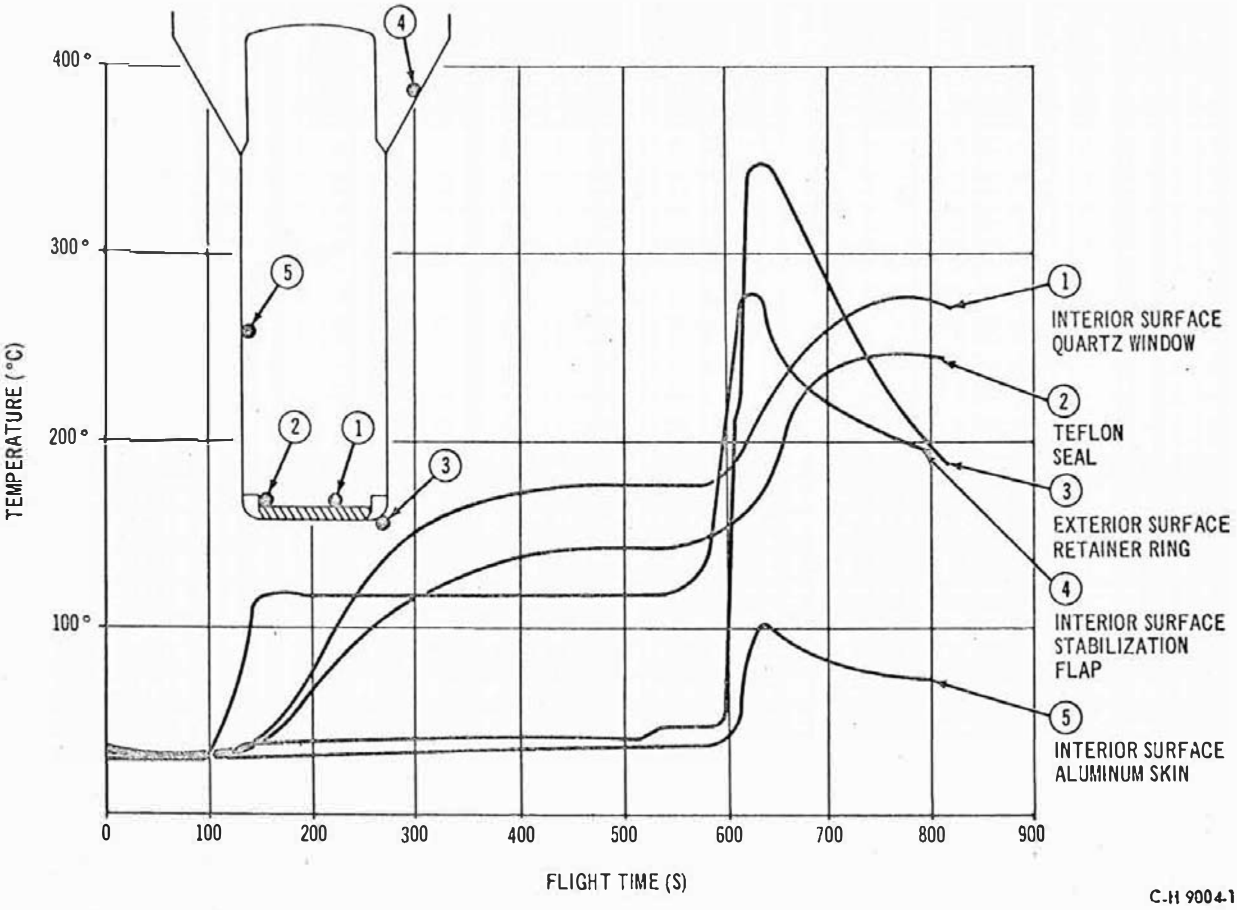

FIGURE 64. INFRARED HEAT TEST DATA

During reentry, the Teflon seals approach sublimation temperatures for short periods. During the heat tests, no Teflon seal sublimation was observed.

c. Lens Heat Shield Evaluation. Cook Technological Center performed an analysis to determine whether a two-piece clam shell protective shield that would close over the camera lens after ejection to prevent excessive radiant heat from reaching the camera would be necessary. Two sources of radiation were considered: (1) stagnation temperature effects and (2) heating of the quartz window. In the analyses of both sources, extremely conservative assumptions were made; even so, the total heat input to the camera was negligible. As a result of this test, it has been determined that no protective shield for the camera lens is required.

4. Capsule Flight Tests

In March 1962, negotiations among Propulsion and Vehicle Engineering Laboratory (R-P&VE), Astrionics Laboratory (R-ASTR), Launch Operations Center (LOC), Atlantic Missile Range (AMR), and Cook Technological Center were completed for a series of drop tests. The first series of tests consisted of four drops from an altitude of 3,000 m to evaluate the camera, recovery package, and SARAH transmitters, and to familiarize recovery personnel with recovery procedures. These tests, conducted in June 1962, are referred to as the June drop tests. The capsules for the June drop tests were modified from the actual flight configuration by adding batteries to operate the cameras and a 5 s delay timer to actuate the paraballoon inflation mechanisms. These tests led to a second series of drop tests conducted in November and December 1962, under the supervision of Atlas project personnel. In these tests, referred to as the high altitude drop tests, capsules were dropped from an altitude of 6,000 m after being flown to 12,000 m to actuate the aneroid switches. Between June and December 1962, actual flight missions using capsules similar to the Saturn capsule configuration were carried out on Atlas missiles.

During drop testing, the original paraballoon rubber bladders, which were so greatly weakened by folding that they snagged and tore easily, failed. Under R-ASTR contract NAS8-2692, the bladders have been redesigned and requalified. They are now made of nylon impregnated with neoprene rubber, and are extremely rugged and reliable. Other improvements in the bladder reliability have been effected through reduction in GN_2 reservoir pressure, smaller reservoir exit orifice, and increased emphasis on quality control during manufacture. Reduction in GN_2 reservoir pressure and size of the orifice has increased bladder inflation time from 0.55 s to approximately 3 s. Previously, rapid inflation caused rupture of the bladder neck.

a. June Drop Tests. The first June test, a successful dry test, consisted of placing four capsules on land at locations unknown to the search aircraft. Each capsule transmitted a SARAH signal to aid the aircraft in locating the capsule.

The first drop test consisted of releasing the capsules, singly, as close to the recovery vessel as possible to minimize the radio search aspects of the recovery. Both the drop aircraft and the ocean range vessel were instructed to obtain a directional bearing on the SARAH beacons and to recover the capsules.

The first camera test capsule (dropped on June 12, 1962) sank because the timer failed to initiate the paraballoon inflation mechanism. When the second capsule was dropped on the second day with similar results, the two remaining capsule drops were postponed. That evening, additional batteries were added to the timer circuit in an attempt to render the paraballoons operative.

Dry tests were repeated on June 13, and were successful. On June 14, the third capsule was air deployed and the fourth was dropped preinflated. Radio signals were received from the SARAH beacons, and both capsules were recovered. The third capsule had a slight paraballoon leak, and had 30 s of exposed film; the fourth had a broken quartz window and 10 s of exposed film. The drag skirts on both paraballoons were torn extensively.

An MSFC rubber specialist examined the air bladder that had leaked on one test and reported that the tear had originated from a small cut or snagged area. Cook Technological Center was instructed to examine four other capsules that were ready for shipment. The bladder specification was then modified to require the contractor to inspect the surface of the paraballoons, and to reject those having any minute cuts or snags.

b. High Altitude Drop Tests. Because capsules very similar to the Saturn prototypes had been obtained for inflight photography on Atlas missiles, a drop test series designed to simulate the dynamic conditions of recovery from Atlas missiles was executed at AMR. These tests were conducted jointly by NASA and Atlas personnel in November and December of 1962.

As a result of the June drop tests and previous Atlas flights on which the prototype capsules had been flown, certain modifications of the capsule had been made: reinforcement had been added to the neck of the bladder, inflation air pressure had been reduced, and the size of the inflation valve opening had been decreased.

To facilitate the location and recovery of these capsules if they sank on impact, a pinger device was attached to the capsule. The pinger emits an audible sound that is easily picked up by the SONAR gear on the recovery vessel. The pinger is used for shallow water recovery only, and is not flight hardware.

The first test of the series was conducted on November 15, 1962. A JB-57B aircraft, equipped with a specially built capsule-release mechanism, flew a camera capsule to 12,000 m to actuate the arming aneroid switches. The aircraft descended to 6,000 m and attempted capsule release which was unsuccessful. However, since successful paraballoon inflation was reported by the photochase plane, aircraft descent continued. At 1,800 m, release was again attempted and was successful. Aircraft visually located the capsule and maintained visual contact for 30 minutes before the capsule sank. A weak SARAH signal was reported at Cape Kennedy for a few minutes after impact. SONAR contact with the capsule was made three hours after impact, the area was marked with buoys, and the capsule was recovered from the ocean bottom approximately 15 hours later.

Examination of the camera capsule showed the following:

(1) SARAH beacon, flashing light, and pinger were inoperative.

(2) Antenna was broken off near base.

(3) Bladder and cover had been punctured near top by thrashing antenna.

(4) Camera had not started.

(5) Ninety percent of dye marker was still in container.

(6) Quartz window and camera lens were intact.

As a result of the test, recommendations were made for changing the dye marker formula and rate of water flow through the dye chamber.

A dummy capsule release test was performed on November 20. Since this release was also unsuccessful, the JB-57B release mechanism was modified. On November 21, the dummy capsule was again flown and was successfully released.

The second live capsule drop of the series was conducted on December 3. The aircraft flew to 12,200 m to actuate the aneroid switches and then descended to 6,400 m and released the capsule.

The chase-plane photographs show the release to have been clean, with $\pi/4$ rad capsule oscillation prior to stabilization. Impact occurred approximately 1.4 km from the recovery vessel. Personnel on the recovery ship visually located the capsule and recovered it approximately 15 minutes after impact.

Examination of the recovered capsule revealed the following:

(1) Quartz window and wide-angle lens: Shattered. The lens were loose because the O-ring between the wide-angle lens and the lens mounting plate had been omitted and the lens housing mounting screws had not been tightened.

(2) Camera: A few inches of film exposed. No usable data could be obtained. The extremely low temperature (-40 °C) at 12,200 m is believed to have congealed the grease lubricant on the camera mechanism, because the capsule was airborne for two hours.

(3) Beacon and flashing light: Inoperative. Water leakage had caused arcing in the circuit board.

(4) Drag skirt: Severely torn.

(5) Antenna: Tangled in drag skirt and bent toward water line.

c. Atlas Missile Flight Tests. Between August and December 1962, ten camera capsules similar to the model A capsules designed for the Saturn project were flown on Atlas missiles launched at AMR. Two capsules were installed on each missile. One capsule was ejected with the flaps first; the other with the lens first. At 69,000 m altitude a Saturn-type pneumatic system ejected both capsules in an aft direction through the engine exhaust at about -60m/s relative to the missile velocity of 2,700 m/s.

The Atlas tests have qualified the capsule under conditions that exceed the Saturn requirements. At ejection, the Atlas vehicle velocity was approximately 2,700 m/s, as compared to 2,600 m/s for SA-5. This created a reentry environment different by 90 percent more peak reentry heat flux and 55 percent more heat load than that expected for SA-5. In addition, the capsules were ejected through the plume of the upper stage sustainer engines.

Recovery operations were conducted for all ten capsules, and seven were successful. The AMR search and recovery support consisted of three to five aircraft, two of which were Air Rescue Service SA-16 aircraft equipped with air-sea snatch gear and para-rescue divers. Each recovery effort was also supported by an AMR ocean vessel equipped with a small recovery work boat. A summary of these results is given here:

Capsules recovered	7
Capsules not recovered	3 (2 unsuccessful recoveries were attempted during very rough seas.)
Quartz window	4 shattered 1 missing
Capsule damage	3 dented camera compartments 1 dented lens compartment
Leakage	5 in lens compartment 1 in camera compartment
Evidence of severe heating	3 on paraballoon cover and drag skirt 4 on camera container exterior 1 on camera container interior

Camera condition	5 good 1 wet 1 jammed by film
Film condition	5 very good 1 bunched and broken 1 with water marks
Paraballoon bladder	2 were of old type with small reinforced neck area (1,030 N/cm^2 reservoir pressure). 1 leaked slowly. 1 leaked badly and sank (recovered by para-rescue divers). 4 were of the new type with large neck area reinforcement of nylon impregnated with neoprene (860 N/cm^2 reservoir pressure). 1 was undamaged. 3 were lost (2 in heavy seas). 2 were of the new type with heavier rubber (961 N/cm^2 reservoir pressure). 1 leaked slowly. 1 was undamaged.
SARAH beacon	8 signals were received; 2 were not. 2 were normal signals. 2 weak signals later became normal. 1 signal failed after 2 hours. 1 signal failed after 15 minutes. 2 signals terminated at impact.
Light beacon	3 did not operate.
Dye marker	5 were inadequate 2 were good (powdered dye added to recovery package).

4. Practice Recovery Capsule

After the June drop tests, ten practice recovery capsules were designed at the M-P&VE Laboratory (eight for testing and two spares). The practice models have provided training and experience for the recovery team, thus increasing successful recovery probability.

The practice models (figure 65) have the general external configuration of an ejected camera capsule. A foam ball rather than a paraballoon acts as the flotation device. Each capsule is equipped with a pulse code O SARAH beacon operating on a frequency of 240 MHz, and a high-intensity xenon light which flashes at approximately 30 times per minute. An auxiliary battery pack increases the operating time from 20 to 70 hours.

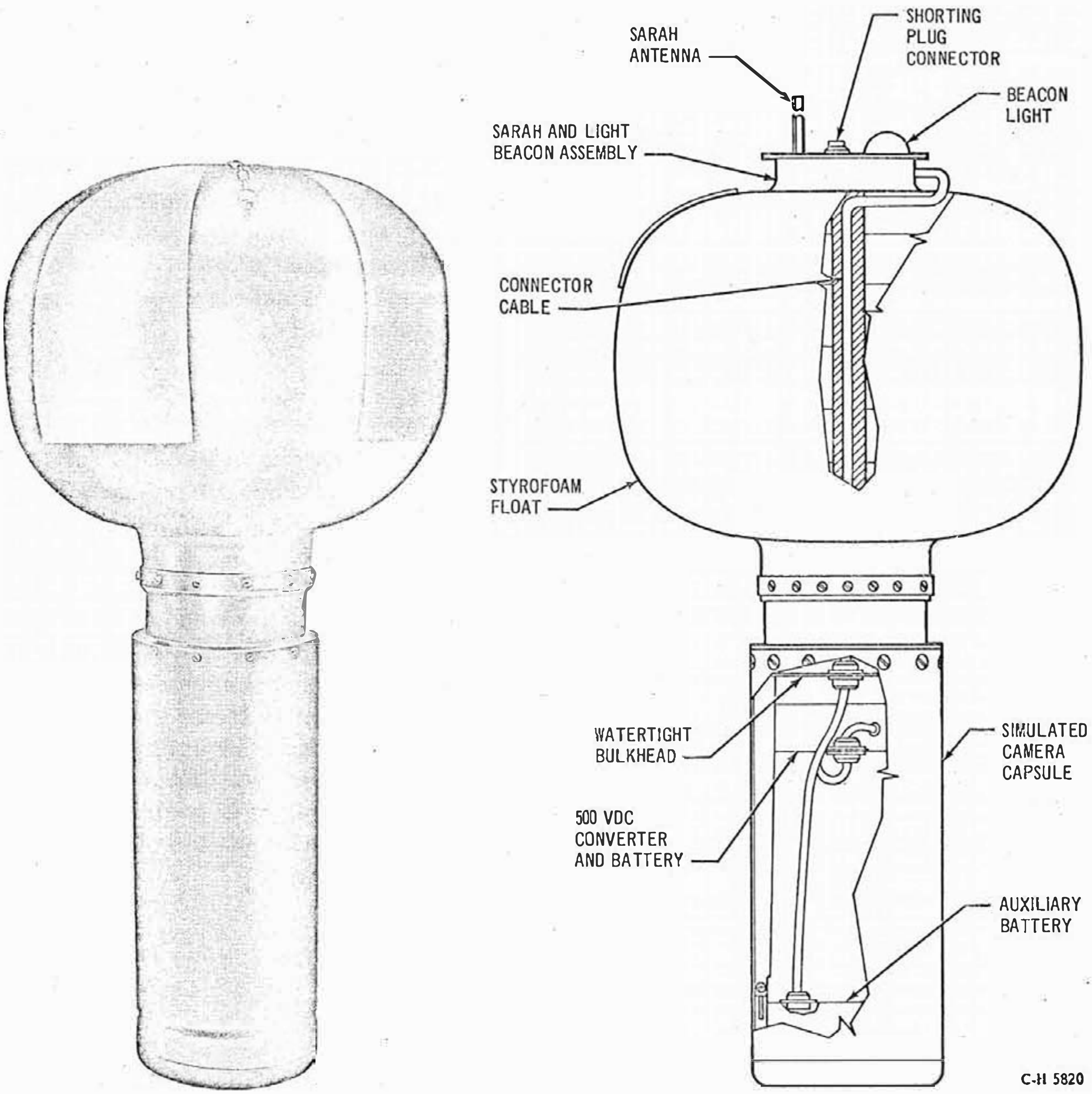

FIGURE 65. PRACTICE RECOVERY CAPSULE

SECTION V. SATURN I (SA-5) FLIGHT AND RECOVERY

On January 29, 1964, the Saturn SA-5 was launched and the booster flight and second stage separation was filmed, using the inflight photographic instrumentation system described in Sections I through IV of this document. All but one of the eight capsules used in the project were retrieved from the recovery area (figure 66). This section presents a discussion of the areas filmed, flight sequencing, and film recovery. Also, the photographic data is evaluated with comparable telemetered data.

A. OPTICAL COVERAGE

Eight camera capsules were spaced around the top of the S-I stage so that each capsule would be ejected clear of the vehicle and so all of the capsules would fall into the same general recovery area. Each camera was assigned a subject and from this assignment the required orientation was determined (see Table 5). In most cases, however, the orientation specified was designed to make maximum use of the entire field of view by filming other areas of interest in addition to the primary subject. Exposure time and focus settings were determined and set prior to the flight to provide maximum reliability.

Cameras 1, 2, 3, and 4 were oriented to view in the direction of flight. The optical center lines were tilted out at 10 degrees to the vehicle's axis and rotated clockwise through an angle of 18 degrees. Through this orientation scheme, a total of 130 degrees of angular coverage was obtained. After the S-IV stage moved a few feet it left the field of vision of cameras 5 and 6; at this point, cameras 1, 2, 3, and 4 continued to monitor the flight.

Cameras 5, 6, 7, and 8 provided fiberoptic coverage of the remote areas of the interstage. Camera 5 covered the chilldown area prior to separation and then recorded the separation sequences. Camera 6 gave general coverage of the interstage area looking up at engine 6. Camera 6 records at 16 frames per second until separation begins, then records at 64 frames per second. Light prior to separation is furnished by a strobe light flashing at 16 flashes per second and exposing every fourth frame. After separation, outside sunlight is available and coverage is obtained at the camera rate. Camera 7 provided coverage of LOX tank 3, and camera 8 provided coverage of the center LOX tank.

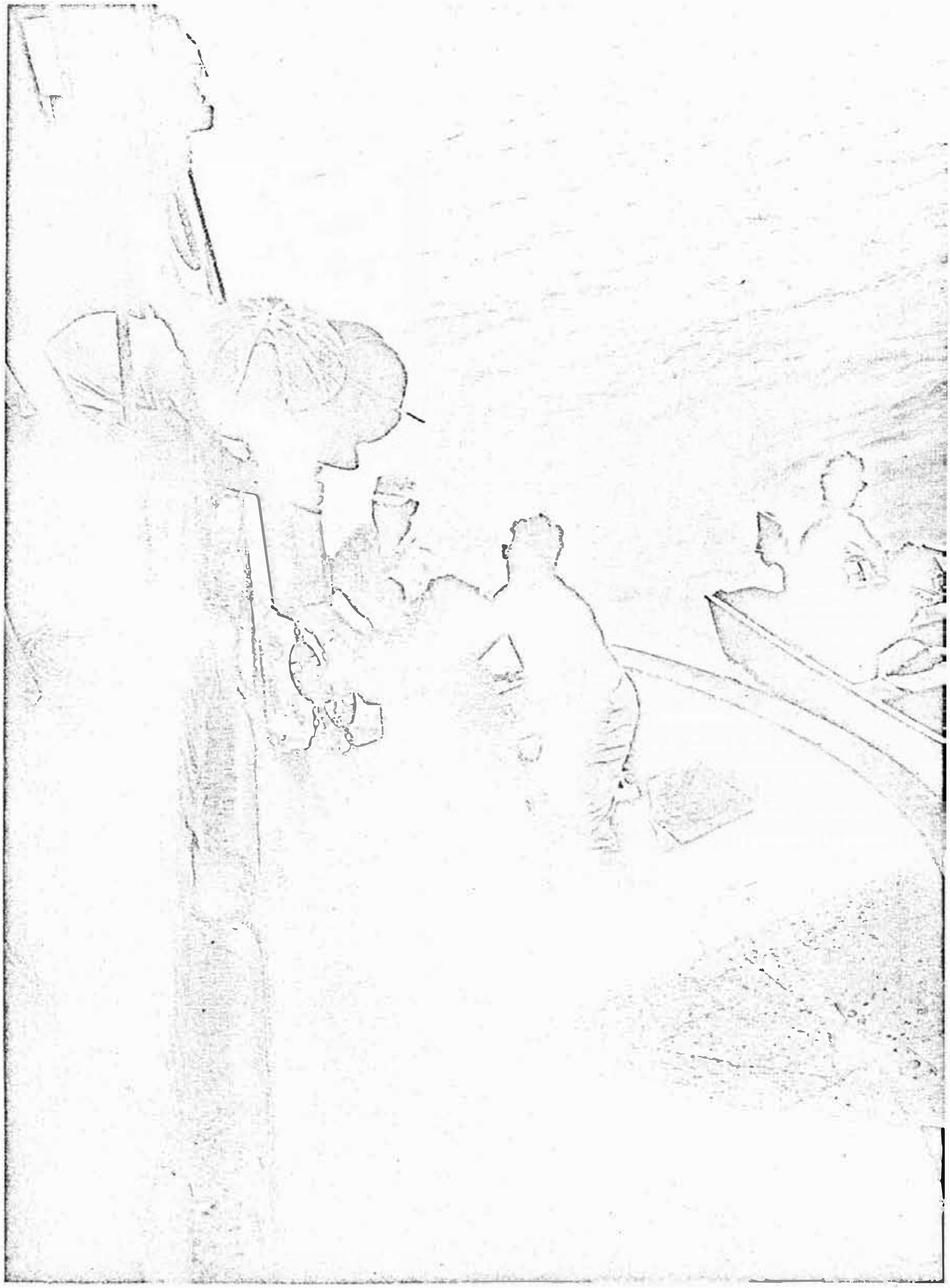

FIGURE 66. RECOVERY OF CAMERA CAPSULE

TABLE 5. PHOTOGRAPHIC COVERAGE

Camera Model No.	Location Sta. Fin	View	Subject	Frame Rate (frame/sec)	Length Lens Focal	Light and Exposure (shutter aperture)	Film	Film Processing
A 1	969 I	Outside Upward Direct	Ullage rocket Fin I side of S-IV stage	64	110° wide-angle; 4.2037 mm (0.1655 in)	Natural 72° shutter @ f2	Ekta-chrome (commer-cial) 16 mm	Normal
A 2	969 II	Outside Upward Direct	Ullage rocket Fin II side of S-IV stage	64	110° wide-angle; 4.2037 mm (0.1655 in)	Natural 72° shutter @ f2	Ekta-chrome (commer-cial) 16 mm	Normal
A 3	969 III	Outside Upward Direct	Ullage rocket Fin III side of S-IV stage	64	110° wide-angle; 4.2037 mm (0.1655 in)	Natural 72° shutter @ f4	Ekta-chrome (commer-cial) 16 mm	Normal
A 4	969 IV	Outside Upward Direct	Ullage rocket Fin IV side of S-IV stage	64	110° wide-angle; 4.2037 mm (0.1655 in)	Natural 72° shutter @ f4	Ekta-chrome (commer-cial) 16 mm	Normal
B 5	969 II	Remote Internal	Fiberoptic coverage of interstage chill-down area and S-IV engine nozzles.	24	*54.9° by 40.9°; 10mm (0.394 in)	Incandescent lamps; 106° @ f1.8	Kodak MS 16 mm	Forced processed 3 "F" stops
B 6	969 IV	Remote Internal	Fiberoptic cover-age of S-IV engines looking up at engine 6.	64	*84° 5.7mm	Stroboscopic and Natural 15° shutter @ f1.8	Kodak MS 16 mm	**Forced processed 3 "F" stops
B 7	969 III	Remote Internal	Fiberoptic coverage of LOX tank 3	24	*54.9° by 40.9°; 10mm (0.394 in)	Incandescent lamps; 160° shutter @ f1.8	Kodak MS 16 mm	Forced processed 4 "F" stops
B 8	969 I	Remote Internal	Fiberoptic coverage of center LOX tank	12	*54.9° by 40.9°; 10mm (0.394 in)	Incandescent Lamps; 160° shutter @ f1.8	Royal-X Pan 16 mm	Forced processed approx. 4 "F" stops

* Only Objective Lens on fiberoptic bundle listed. Camera and optic coupling lenses were 50 mm Wollensak.
** Processing included step-printing of first part of film.

B. SEQUENCING AND RECOVERY

Speeds and operational sequencing were selected for each camera to achieve maximum use of the film (figure 67).

The capsule ejection occurred at 607.2 seconds after range zero. The telemetry indicated that ejection was normal for all capsules. The capsules impacted on the ocean surface 892 km downrange. Seven of the eight capsules were recovered after impact. The eighth capsule, capsule number one, was not located, probably the victim of a recovery malfunction.

Six of the seven capsules recovered were in excellent condition. The other, capsule number six, was found connected only to the paraballoon by the neck of the bladder; the retaining webs had been sheared from the capsule.

A large indentation similar to that created by a hydraulic force was found on capsule number six. Apparently the capsule experienced a freakish impact because of high winds and rough seas. Not to be discarded is the possibility that the expelled stabilizing fin and fiberglass housing unit did not clear the paraballoon until after the capsule struck the water. Such a possibility would cause the capsule to strike the surface at a high velocity and in an unusual and incorrect attitude.

The radio beacons on the seven capsules that were recovered functioned as expected. The dye solution was clearly seen from the air but dispersed within 45 minutes, necessitating that flares be dropped.

No signals from the missing capsule were received.

C. PERFORMANCE

In general, the instrumentation provided the required coverage and was fair in quality. Examination of the cameras revealed that they had operated as planned; however, a substance coated the capsule viewpoints on the externally viewing camera obscuring most of the coverage during separation. The four cameras using fiber-optics provided some qualitative data.

Through the use of reference marks on one edge of the film and event marks on the other edge, coordination of the data was possible. Fiducial marks on the film aided in locating the center of the frame. This presented an unusual opportunity because seldom has an instrument with its own record been recovered thereby enabling the verification of telemetered data.

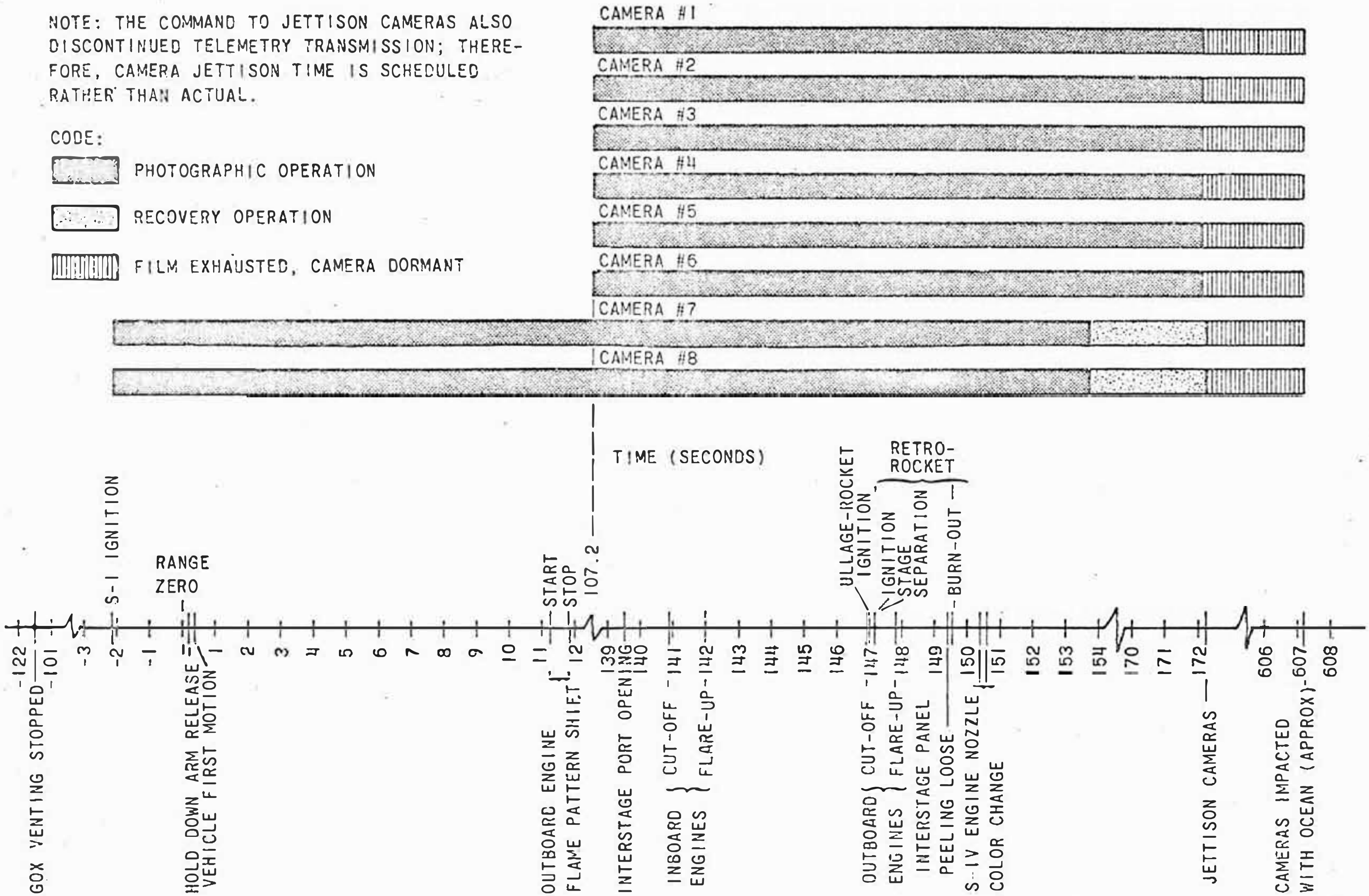

FIGURE 67. CAMERA SEQUENCING FOR SA-5 FLIGHT

Although camera 1 was not recovered, telemetered data indicated that operations and ejection were normal.

On cameras 2, 3, and 4 the first one-second film segment was a test at 24 minutes before liftoff. When the cameras turned on, the quartz windows were coated with a soft opaque substance that continued to accumulate, obscuring any coverage. At times, small, unidentified particles fell onto the viewports. Then, with the firing of the ullage and retro rockets, an increase in sunlight was noted, indicating a partial clearing of this material. The ullage rocket ignition and separation were visible. At this point, a radical change in the obliterating layer was observed, reducing to some extent its effects, but not removing it. Failure of the chilldown ports to open could be observed as the booster's tumble brought the earth into view. Cameras 2 and 3 also observed the S-IV stage engine ignition and bell heating (color change).

Camera 5's coverage shows the LOX-SOX chilldown dispersal system on the S-IV stage. LH_2 chilldown was in progress when the camera turned on. There was some vapor in view, but it gradually dispersed. The interstage fogged up when the LOX prestart occurred. One second later the effect of the LOX-SOX dispersal system was seen. Simultaneously, the partial operation of the blowout panels was seen. At the time of separation, sunlight flooded the interstage, over-exposing the film.

Sea water entered camera 6 and damaged the latent image on a portion of the film. Smoke or vapor was observed in the interstage at camera turn-on, but it cleared prior to separation. Vapor streaming from the engine and overboard pipes was visible when the S-IV stage lifted out at separation. The ullage rockets were seen firing as they came into view. Operation of the S-IV stage engine igniters was observed as was a change in engine color. At approximately this time,. violent buffeting of the interstage wall was seen. A large flap, believed to be a section of the honeycomb interstage panel, was seen blowing about in the interstage. As the booster tumbled, the earth came into view and patches of sunlight, reflecting from the tops of the blowout panels, were seen moving along the interstage wall.

The center portion of camera 7's film was lost when it broke in processing. No large amount of sloshing was observed in LOX tank 3 during the flight. When the retro rockets fired, the residual LOX was seen accelerating to the forward end of the tank. In free fall immediately afterwards, small globules of LOX were seen gyrating under weightlessness.

Camera 8 observed sloshing in the center LOX tank of a very low amplitude with a high frequency ripple on the surface of the LOX. The LOX was observed to pass the interior rings of the tank as depletion occurred (table 6). Residual LOX was seen accelerating to the forward end of the tank during retro firing.

TABLE 6. DATA REDUCTION REPORT ON LOX LEVEL IN CENTER TANK

RING NO.	TIME AFTER RANGE ZERO
19	11.831
18	19.866
17	26.961
16	33.880
15	42.176
14	47.904
13	54.998
12	61.922
11	68.590
10	75.514
9	81.839
8	87.908
7	94.319
6	101.585
5	107.398
4	113.980
3	121.074
2	127.143
1	133.725
Quartz Window	148.360

NOTE: Each frame that the LOX was observed to completely pass an interior ring was noted. This frame was then timed and converted to t = seconds after range zero.

NOTE: The accuracy of determining the exact frame where the LOX had completely passed a ring decreased with tank depth. The magnitude of this error varied from ± 50 milliseconds at the top to ± 200 milliseconds at the bottom of the tank.

D. EVALUATION

The Saturn SA-5 project is considered a success although certain events, monitored by the externally viewing cameras, were almost completely obliterated during the high heat region by a foreign material accumulating on the viewport. The discovery of this substance is considered significant.

A study of the films and a chemical analysis of the recovered capsules indicate the material to be carbon particles (a residue of Thermo-Lag) that ablated from the forward section of the vehicle. These conditions were brought about by the vehicles aerodynamic environment,i.e., the leading shock wave created a separation area in which the cameras were located and caused low velocity circulatory turbulence next to the vehicle. These low velocity currents transported the carbon particles to the viewports.

The film records have permitted verification of telemetered data, i.e., the filmed record gives knowledge of how the selected events occurred; not only just that they did occur (table 7).

A fringe benefit of this type of instrumentation is the bonus of recording an unexpected occurrence such as the foreign material on the viewports; the discovery of smoke or vapor in the interstage area; the partial instead of full operation of the blowout panels; and the violent buffeting of the interstage wall and the subsequent destruction of a honeycomb interstage panel.

Recovery of seven of the eight camera capsules indicated a continued high recovery capability. The newest configuration, using a nylon-reinforced bladder, is responsible for this favorable development. Statistical analysis shows, on the basis of 16 capsules recovered out of a total of 17, that the reliability factor is greater than 79 percent at the 90 percent confidence level. Consequently, zero capsule failures were predicted for the SA-6 flight with 90 percent confidence.

E. FUTURE MODIFICATIONS AND PLANS

From experience gained on the SA-5 flight, a number of changes was made to the photographic instrumentation systems for the SA-6 and SA-7 vehicles.

A nitrogen purge system was added to minimize the Thermo-Lag ablation deposits on the viewport of the direct-viewing capsules. Three of the four direct-viewing capsules are purged--these are those containing cameras 1, 3, and 4. The system consists of three spheres containing gaseous nitrogen; these spheres manifold into a common line. A nozzle at each capsule viewport directs the gaseous nitrogen over each window. Purging is initiated at T + 97 seconds. Activation of the cameras continues to be T + 107 seconds--the same as on SA-5.

Camera number 2 was used to photograph the blowoff panel flutter during mach 1.0 and the "q maximum" region. Operation of camera 2 was reprogrammed from T + 107 seconds to T + 30 seconds. No purge was required for this camera.

TABLE 7. PHOTOGRAPHIC INSTRUMENTATION SYSTEM AND MONITORED EVENTS ON SA-5 FLIGHT

Event	Camera No.	Time*	Telemetry Time *	Comments
Ignition Command	No		Yes -2.99	
Commit	No		Yes 0.00	Range Zero
First Motion	No		Yes 0.31	
Liftoff Signal	(See Note)		Yes 0.40	Start Program Development
Begin Roll	No		Yes 8.78	
End Roll	No		Yes 13.05	
Begin Tilt	No		Yes 15.71	
S-IV Prestart (chilldown)			Yes 107.66	
Freeze Tilt	No		Yes 133.31	
Panel Blowout	Yes 5	138.44	No	
S-IV LOX Prestart (Ports open)	Yes 5 (See Note)		Yes 139.80	
Inboard Engine cutoff	No (See Note)		Yes 140.75	
Ullage	Yes 2	146.68		
Outboard Engines cutoff	No		Yes 146.73	
Separation Event Mark	Yes 6	146.75		
Retro Rockets Fire	Yes 2	146.79		Large Orange Flash
	Yes 4	146.80		Small Red Flash
Ullage Rocket Ignition			Yes 147.02	
S1S-IV Separation	(See Note)		Yes 147.14	
Retro Rocket Ignition			Yes 147.16	(147.17 on table)
Retro Pressure Buildup			Yes 147.19	
S-IV Ignition				
Engines 1 & 2	Yes 6	150.63		Engines 5 and 6 were hidden by fiducial marks
Engines 3 & 4	Yes 6	150.76		
Film Run-out				
Camera No. 7				
Camera No. 8				
Jettison Ullage Rockets			Yes 167.14	
Last Sec. Recorded	Yes 4	169.0		Ullage eject not visible
Camera ejection	Yes 5	172.95		
Resume Tilt			Yes 174.11	
S-IV Cutoff			Yes 629.97	

* Time (in seconds) from Range Zero

NOTE: Four event marks are printed on film for co-ordination. The marks are at liftoff, S-IV LOX prestart, inboard engine cutoff, and separation.

The 110 degree wide-angle lenses in the four direct viewing cameras was replaced with 160 degree lenses. The 160 degree lenses gave greater picture coverage, less distortion, and a sharper image. The use of the 160 degree lenses also allowed reduction of the overall length of the applicable capsules and capsule ejection tubes since the 160 degree lenses are smaller than the 110 degree lenses.

The angle of view of the fiberoptic lens, on camera 6, was changed from 28 degrees (from vertical) to 18 degrees. The new angle gave better coverage of stage separation.

SECTION VI. SATURN AS-201 FLIGHT, RECOVERY, AND PARACHUTE TEST PROGRAM

The Saturn AS-201 vehicle was launched from Cape Kennedy on February 26, 1966. The inflight photographic instrumentation system consisted of two camera capsules (Model A) as previously described in sections I through IV of this Internal Note with the following three modifications--a ribbon parachute system to decrease velocity; enlargement of AFT section capsule; and a change in the aneroid arming sequence. These changes are covered in more detail in this section.

The capsules used in this experiment were located on the floor area of the interstage area on top of the S-IB first stage. The purpose of the test was to check ignition of the engine and separation of S-IB from S-IVB.

A. RIBBON PARACHUTE

The camera capsules used on the Saturn I, Block II vehicles were highly successful, but the impact loads on the Saturn SA-6 were of such magnitude that the windows were broken on all of the capsules, and one capsule was severely dented. It was felt to prevent this damage on subsequent vehicles, the impact velocity would have to be decreased from 88 ft./sec. to approximately 44 ft./sec.

The camera capsules used on the previous vehicles had a paraballoon with an attached skirt to decelerate the capsule. There were two choices--increase the skirt or add a parachute. Since the required area of the parachute was approximately four times the total cross-sectional area of the paraballoon, it was almost impossible to extend the skirt. In addition, the wake created by the skirt would decrease the effectiveness of the parachute, so the skirt was removed entirely.

After some consideration, it was decided that the parachute design should be a ribbon parachute of a circular shape (See figure 68). The ribbon design was chosen since this type of parachute gives relatively low angles of oscillation (± 5°). It was suspected that the damage to the capsules, other than window damage, was due to the capsule impacting at an angle. Therefore, by selecting the ribbon design, the amplitude of the oscillation could be minimized. Also, the average drag coefficient of this type of parachute was determined to be approximately 0.5, but the drag decreases approximately 20 per cent by the wake.

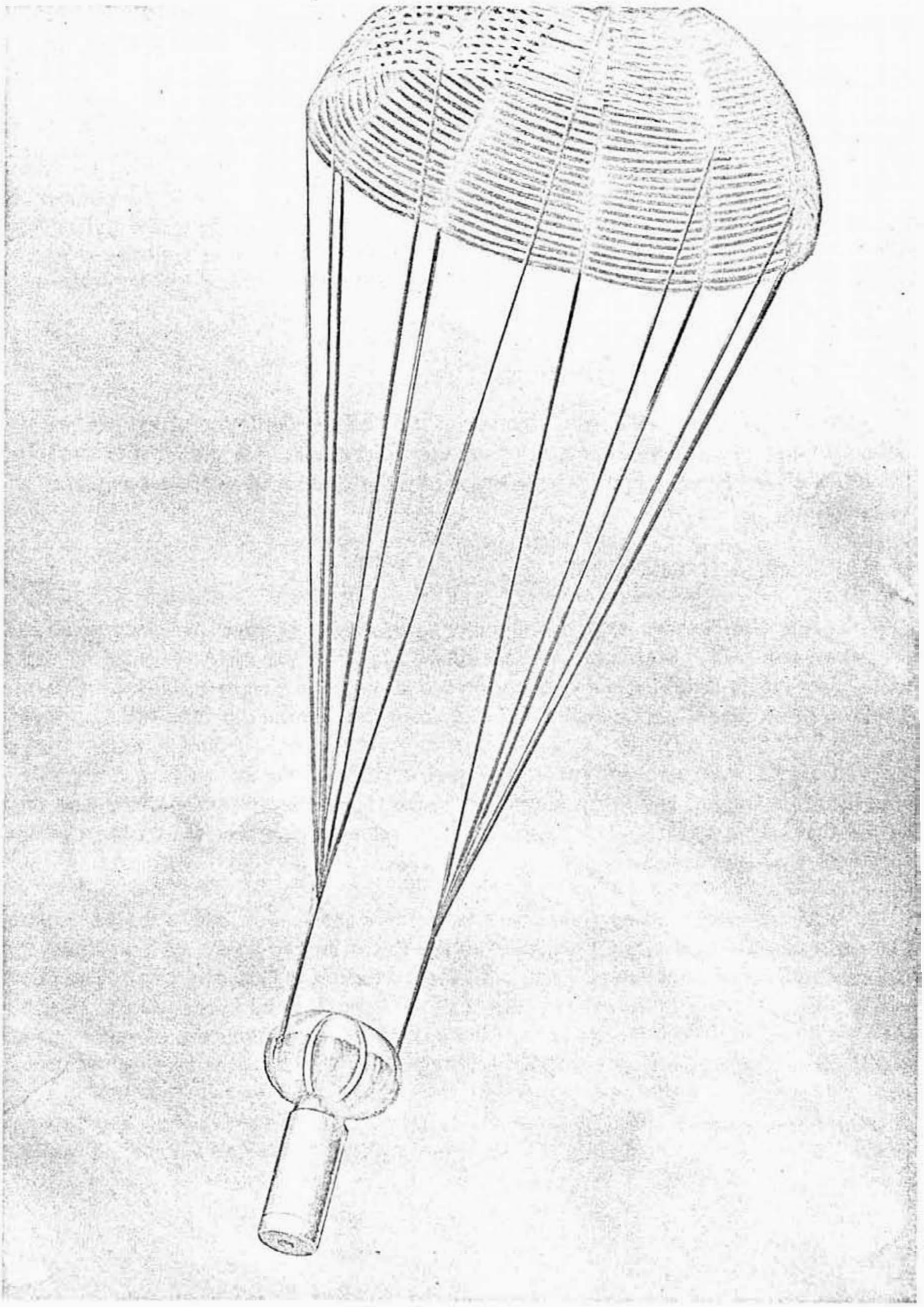

FIGURE 68. CAMERA CAPSULE RIBBON PARACHUTE

Using the weight of the capsule with the flap assembly ejected, approximately 45 pounds, the size of the parachute was determined to be a constructed diameter of approximately 8 feet or a deployed diameter of 5.5 feet.

B. CAPSULE AFT SECTION

The aft section of the capsule was modified to accommodate the packing of the parachute (See figure 69). As shown in figure 70, the walls of the cylinders were thickened, a metal separation plate was installed between the beacon and parachute bag, the stabilization flaps were moved forward, and the aft section was extended 6.125 cm.

C. ANEROID ARMING SEQUENCE

On the previous vehicles, arming of the balloon deployment system was accomplished by action of a 40,000 foot aneroid switch. On the AS-201 vehicle, the balloon deployment system was armed by the command vehicle program sequencer.

D. FLIGHT AND RECOVERY

Of the two camera capsules launched, one was successfully recovered and the other was lost. Both camera capsules were properly ejected from the vehicle and according to tracking data, they impacted near the preplanned impact point. There was no visual contact with the camera capsules during free fall.

A signal was received from the Sarah beacon of the first camera capsule shortly after impact and later the yellow-green fluorescein dye marker was spotted by the tracking aircraft. The location of the capsule was marked by the rescue ship with a smoke bomb.

After recovery of the first camera capsule it was found that the parachute had not opened, but was still attached to the flotation bag by its shroud lines. There was considerable damage to both the capsule and flotation bag. The quartz window was missing, the camera lens was broken, and the black plastic face plate was broken. The flotation bag was half torn loose from the camera capsule and its inner rubber bladder was protruding out of the flotation bag on the torn side. The capsule had three long scratch marks about 1 to 1-1/2 inches apart, running lengthwise on the outside of the capsule. The inside of the capsule was checked for water by removing the receptacle plug. No water was found inside the capsule.

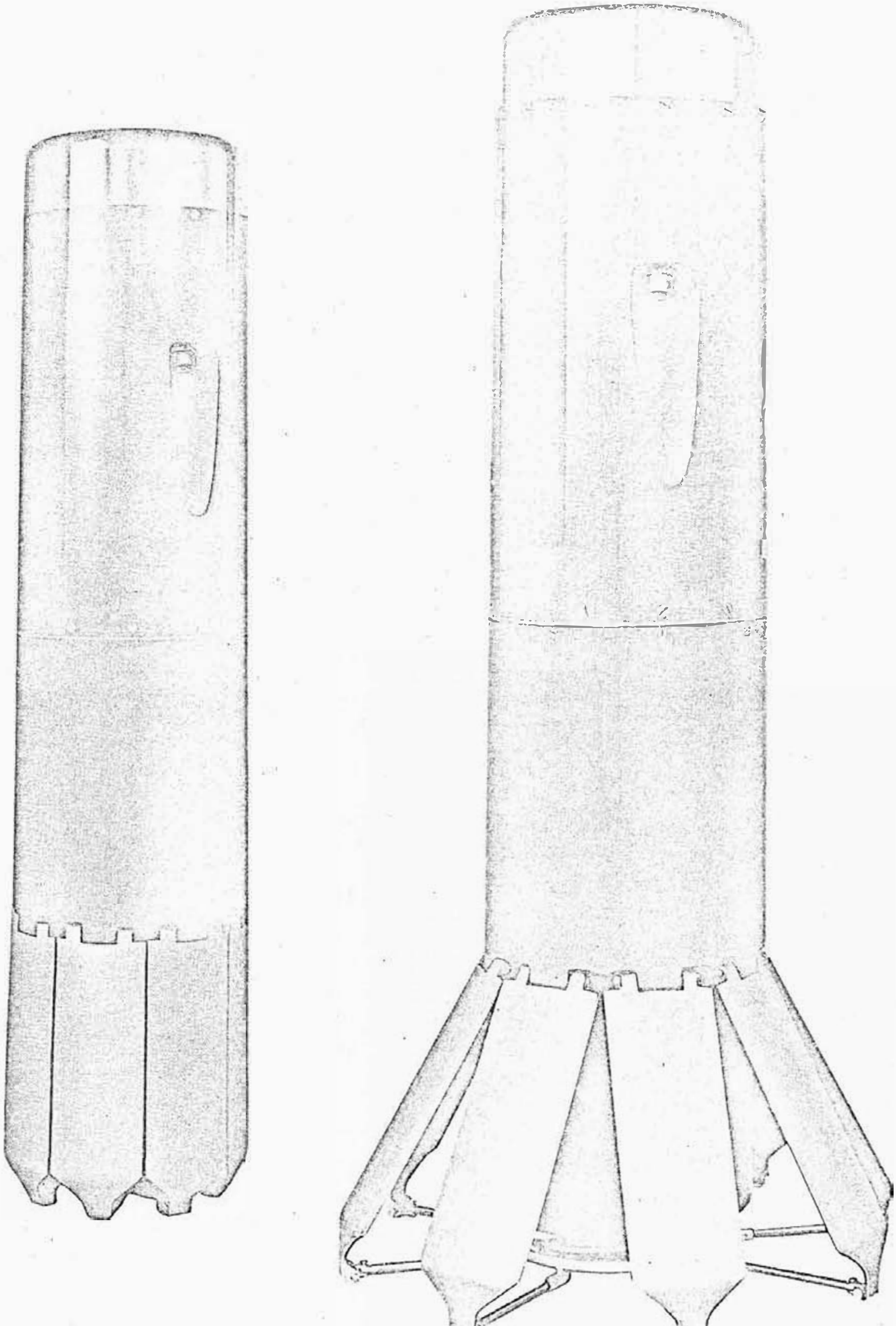

FIGURE 69. MODIFIED AFT RECOVERY SECTION WITH FLAPS IN OPEN AND CLOSED POSITION

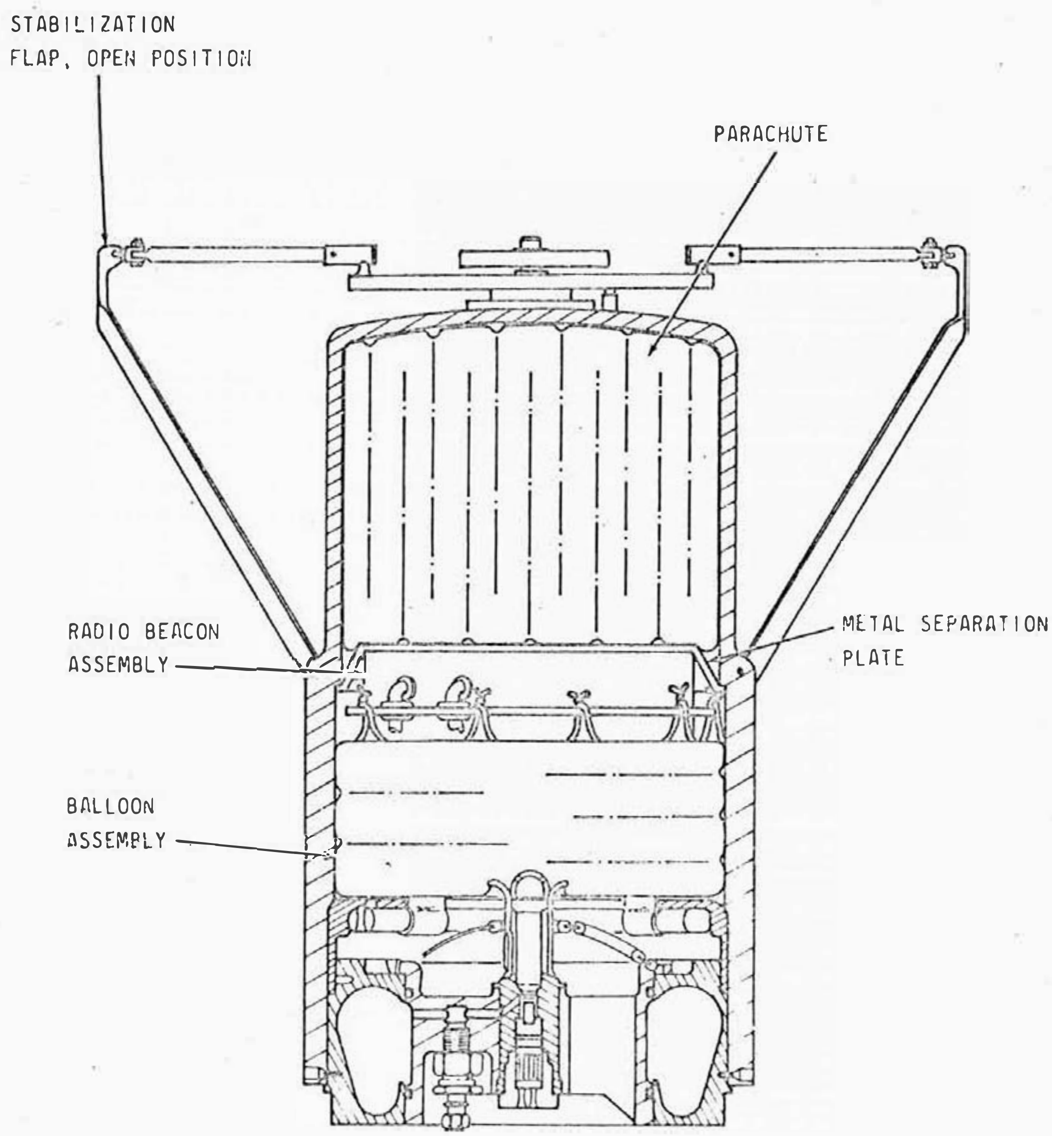

FIGURE 70. MODIFIED AFT RECOVERY SECTION

A Sarah signal was also received from the second camera capsule, but only the flotation bag, which was partially deflated, and the parachute, which was floating unopened on top of the water, was recovered by the recovery ship The camera capsule had torn loose from the flotation bag on impact with the water and was subsequently lost.

E. FILM COVERAGE

The film in the recovered capsule was processed and it was found that engine ignition and stage separation had occurred as planned.

F. CAMERA CAPSULE PARACHUTE TEST PROGRAM

Because of the parachute malfunction in the two AS-201 capsules, a new test program was begun by R-ASTR-IMS. Three drop tests were made during this program, two from a helicopter at an altitude of 1000 feet and one from a tower 100 feet high.

The first drop test was made from a helicopter at 1000 feet using a parachute repacked by Laboratory personnel. This packing used a rubber band as a release device rather than the string used on the AS-201 parachute. Due to the helicopter's slip stream, the parachute opened within the first 20 feet of the drop test. The helicopter had a forward velocity of approximately 50 miles per hour ground speed.

The second drop test from a 100 foot tower was performed at MSFC using the same configuration as that in the first test. Of the 12 rubber band release devices on the parachute, only 3 rubber bands had released by the time the capsule hit the ground. This indicated that the deployment chute did exert enough drag to cause the release of the parachute risers.

The third drop test was from a helicopter at 1000 feet. The helicopter maintained a stationary position at a velocity of 0 miles per hour ground speed. In this packing both the string and the rubber band device, which had replaced the string, were eliminated. The parachute began to deploy at approximately 1.25 seconds after release and was completely deployed from the packing bag by 4.5 seconds after release. The parachute was half opened when the capsule hit the ground.

After completion of these tests the objective, i. e., check deployment of the parachute from the packing bag, was successfully accomplished, and no further tests are considered necessary. Also, the force required for parachute deployment had been reduced from 10 pounds to 3 pounds by the elimination of the string from the release device.

SECTION VII. SATURN AS-202 FLIGHT AND RECOVERY PROGRAM

The Saturn AS-202 vehicle was launched from Cape Kennedy on August 25, 1966. The inflight photographic instrumentation system consisted of two Model A camera capsule systems. These two systems were located at Fins Two and Six in the floor area of the interstage area on top of the S-IB first stage. The purpose of the test was to check ignition of the engine and separation of S-IB from S-IVB.

A. FLIGHT AND RECOVERY

The two camera capsules were properly ejected from the vehicle, after launch, and a Sarah beacon was picked up after impact and the recovery helicopter vectored on the impact point. The recovery force retrieved the camera capsule system which had been located at Fin Six. The camera capsule system which was located at Fin Two was not recovered; the only trace of this system was a yellow dye slick on the water a short distance from where the recovered capsule was found.

The ribbon parachute obviously functioned normally and the camera capsule that was recovered was undamaged.

B. FILM COVERAGE

The film, from the recovered capsule, after being processed, indicated that engine ignition and stage separation occured as planned.

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