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THE DSI SMALL SATELLITE LAUNCHER

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ABSTRACT

A new launcher has been developed by DSI, that is compatible with the GAS canisters. It has the proven capability to deploy a satellite from an orbiting Shuttle that is 18 inches in diameter, 31 inches long, and weighing 190 pounds.

These DSI Launchers were used aboard the Discovery (STS-39) in May 1991 as part of the Infrared Background Signature Survey (IBSS) to deploy three small satellites known as Chemical Release Observation (CRO) satellites A, B, and C. Because the satellites contained hazardous liquids (MMH, UDMH, and MON-10) and were launched from GAS Cylinders without motorized doors, the launchers were required to pass NASA Shuttle Payload safety and verification requirements.

Some of the more interesting components of the design were the V-band retention and separation mechanism, the separation springs, and the launcher electronics which provided a properly inhibited release sequence operated through the Small Payload Accommodations Switch Panel (SPASP) on board the Orbiter.

The original plan for this launcher was to use a motorized door. The launcher electronics, therefore has the capability to be modified to accommodate the door, if desired.

INTRODUCTION

DSI developed the launchers because the presence of the hazardous payload required more safety inhibits than existing designs could provide. As the satellite design evolved, more volume and weight capability was needed. The new launcher has the electronics and spring drive outside the GAS canister envelope.

This paper will describe the design features, the qualification program that was performed that led to the approval to fly on the Shuttle, and the operational results.

The DSI launchers were used aboard the Shuttle, Discovery (STS-39) in May 1991 as part of the Infrared Background Signature Survey (IBSS). Fig. 1 shows the canisters and launchers mounted in the Shuttle payload bay. The SPASP was mounted on the aft flight deck, as seen in Fig. 2.

These launchers can be reused. Information is provided as to the processes required for reuse and possible modifications to adapt to other requirements.

DESIGN

Mechanical Details

The DSI launcher was designed to interface directly to the standard GAS canister cylinder. The launcher base plate interfaces to the cylinder via the standard attachment of 32 3/8" screws. The mounting to the Shuttle is a typical GAS canister technique, using 4 attach brackets and a shear pin.

The primary components of the launcher mechanism, as shown in Fig. 3A, are the pusher plate, v-band, mounting ring, separation springs (2), base plate, guide rod, retention springs, electronics/battery boxes, and the electronics enclosure. Most of the parts are made of aluminum. The springs are made of music wire; the guide rod is stainless steel; the bushings supporting the guide rod are bronze.

The launcher pusher plate sits inside the launcher mounting ring and mates with the inside edge of the satellite mounting ring. The segmented v-band, as shown in Fig. 3B, holds the rings together with approximately 3000 lbs tension. The springs were sized to provide a nominal separation velocity of 3.7 ft/sec for a 190 lb satellite. With these springs, the separation velocity will vary with the satellite weight. All components and hardware are mounted directly to the launcher base plate.

The launcher electronics system is contained in a single box as shown in Fig. 4. The batteries and electronics are mounted beneath the base plate in the payload bay, and protected by the electronics module cover plate, which provides an additional 8 inches of usable height inside the GAS can, while utilizing only 2 inches of height inside the canister for launcher volume.

The DSI launcher utilizes a 4-segment, aluminum v-band, using A-286 CRES links at two joints and separation pyrotechnics at the other two joints. The launcher mechanism is released by firing clamp separators, actuated by NASA Standard Initiators (NSI). Each half of the separated v-band is retained in the cylinder after deployment by retention springs. Damage to the GAS cylinder is prohibited by band stops, which retain the v-band and prevent flipping and resulting contact with the cylinder walls. As the v-band separates from the satellite and launcher mounting rings, the spring/pusher plate then pushes the satellite out of the cylinder. Fig. 5 is a photo of a launcher in flight configuration, after deployment.

Electrical Description

The launcher electronics system accommodates three launchers with control by the SPASP. As shown in Fig. 2, this panel has 6 switches and 6 indicators. Two of the switches were used for the launcher address code, three were used for the remaining three inhibits, and the final switch, "execute", was used to initiate the command selected by the other switches. The indicators were used to verify the commands.

The launcher electronics block diagram is shown in Fig. 6. Two 12-volt batteries provide power, each with six lead-acid cells with 4 amp-hour rating. Power converters include a 12V-28V unit for the indicators on the SPASP and a 12V-5V unit for the logic and relay drivers. The pyro circuit uses the 12 volts from the battery. Both batteries and both converters are protected by fuses.

The logic circuit is a set of static gates that convert switch actions into responses to satisfy operational and safety requirements. The circuit design does not allow the failure of a single device to cause an improper action, such as a premature or simultaneous release.

Toggling the execute switch turns the electronics on for 15 minutes. The logic circuit decodes the address switches and closes the select relay if the address is correct. The prearm switch closes the prearm relay which is on the return side of the pyro circuit, and a one minute inhibit timer is started. This timer prevents an premature release by inadvertently throwing the switches. The arm switch closes the arm relay. The deploy switch completes the pyro circuit and releases the satellite from the launcher.

Ground Support Equipment

Pallets were developed to support the CRO satellites and the DSI launchers. They were used during the fuel and oxidizer loading operations on CRO, for transport of the satellites, during the mating of the satellites to the launchers and the installation of the V-bands, and through final flight preparation until integration into the GAS cylinders.

The battery charger provided monitoring and charging of the batteries during testing, integration and prelaunch activities. Final battery charge is performed up to 2 months before launch.

The ordnance circuit monitor unit (OMU) is connected to the launcher in place of the Safe or Arm plug. This connection safes the ordnance by shorting the wires while monitoring the activity on the launcher ordnance circuit. When the fire voltage is applied, a light comes on in the OMU. This unit also verifies no operation during the environmental tests.

The function and power monitoring unit (FPMU) is used in conjunction with a SPASP to verify operation of the four inhibit relays and to monitor battery and converter voltages. Proper timer operation and addressing functions may also be verified using the FPMU.

An interface verification test (IVT) set was developed to verify the copper path of the interconnect wiring system from the SPASP to all three launchers. Each launcher has two connectors for system wiring which are wired together, with branch wiring to the launcher electronics. The first two launchers each have two cables connected, while the third launcher only has one cable connected.

By attaching the IVT set to the unused connector on the third launcher, all wiring could be verified in one test without disturbing flight connections. A test (GSE) SPASP was built to allow operational tests before integration with the Shuttle.

SHUTTLE QUALIFICATION PROGRAM

Design Verification Tests

These tests verified that the operations were in compliance with predicted and required performance characteristics. The GSE SPASP was used with two flight cables and one non-flight cable. All credible scenarios were verified.

A low voltage test was performed to verify the ability of the logic to perform as the supply voltage decayed. The test result showed that the relay drivers ceased operations at a higher voltage than the logic.

Environmental Tests

EMI and EMC testing was performed to verify the ability to operate in the Shuttle RF environment and that no inadvertent emissions from the launcher could effect the Shuttle.

The Thermal Cycling Test was performed to verify operation of the launchers over the temperature range of -30°C to +40°C. The Thermal Vacuum Test, performed at GSFC, was a 90 hour duration test with holds of up to 8 hours at both -20°C and +40°C, the low and high temperatures. The Vibration Test, was conducted with the launchers in the GAS cylinder with the CRO satellite attached in flight configuration. The vibration levels were 5.5 Grms, 7.7 Grms, 7.0 Grms in the X, Y, & Z axes respectively, as specified in the Orbiter Cargo Bay Random Vibration Payload Sidewall Adapters/Orbiter Interface Per ICD-A-17559. The Static Load Test, performed at GSFC, tested the launcher structure and separation mechanism, and showed minimum safety factors of 2.78 on yield and 3.34 on ultimate.

Operational Verification

As part of the design process, an extensive deployment velocity analysis was performed. The goal of the analysis was to provide a nominal separation velocity of 3.7 ft/sec for a 190 lb satellite. Deployment testing was performed to verify the analysis, and some surprising results were found. Deployment testing showed a separation velocity of approximately 60% the predicted value. The losses were discovered to be due to dynamic response of the spring mass system. All previous analysis considered only static properties. Design modifications were incorporated to achieve the required deployment velocity.

An analysis was required to ensure the launcher would not "jam", creating an unsecured, undeployed satellite situation. The inherent design prevents the launcher from jamming itself. Analysis also showed that all possible angles of incidence of the satellites and the cylinder walls, given the coefficients of friction (minimized by placing delrin rails on the outer edges of the satellite) were too small to cause the satellites to jam within the cylinder after separation from the launcher. Flight experience shows that all three CRO satellites contacted the cylinder wall with no jamming.

Tipoff analysis was required to determine expected tumbling, interference, etc. The GAS cylinder walls limit tumbling due to tip-off, but is still possible, as post flight data indicates. Tip-off is a function of satellite mass properties, so a tip-off analysis must be performed for each mission. Post flight data does not correlate to previous analysis well, simply because tumbling was not of concern to the first mission.

V-band release testing was performed in numerous configurations to verify the structural design, the release capability, and the retention test. Ground tests verified that the v-band would release by firing only one of the two clamp separation bolts. Ground testing also verified the v-band attachment procedures as well as leading to the final v-band retention design, which was difficult to design based on analysis alone. The v-band was also test fired using bolts and a bolt cutter, instead of the recommended clamp separator. Tests proved the band released nominally in this configuration.

Flight Safety Qualification

The DSI launcher has passed extensive flight safety qualification and certification in accordance with NSTS 1700.7B, the STS Flight Safety Handbook. Extensive electrical, structural and kinematic analyses and tests were performed to certify that the launchers would pose no operational threat to the Orbiter. In addition, the launcher handling is controlled by a certified and approved Fracture Control Plan, generated in accordance with NHB 8070.1, the STS Fracture Control Requirements. In the event of a re-flight, the Fracture Control Plan is still applicable for all handling and screening of flight hardware.

OPERATIONAL RESULTS

On-Orbit results

CRO satellite C was successfully deployed with a confirmed ejection velocity of 4.0 ft/sec which was within 5% of the predicted release velocity of 3.9 ft/sec. No operational problems were encountered during the deployment sequence.

CRO satellite B was successfully deployed as shown in Fig. 7. The measured release of 3.7 ft./sec. was within 5% of the predicted release velocity of 3.6 ft/sec. Upon initial attempt to deploy B, the astronauts were unable to activate the launcher B electronics. After following the malfunction procedures, there was still no response from the launcher. Ground control commanded the crew to try launcher B again, before going to launcher/satellite AM (which was the planned contingent mode). Launcher B performed nominally and satellite B was successfully deployed. This anomaly is discussed in the next section.

CRO satellite A was successfully deployed, with a measured velocity of 3.5 ft/sec, which is within 5% of the predicted release velocity of 3.4 ft/sec. Problems similar to those seen in the B deployment were observed here. Again, this anomaly is discussed in the next section.

Post Flight Activities

When the Shuttle returned to the Kennedy Space Center (KSC), a series of tests were performed during the deintegration as part of the anomaly resolution. In the Orbiter Processing Facility, operational and cable resistance tests were performed before any changes were made to the flight configuration. (Repeating the verification tests performed pre-flight.) Both tests were successful and showed no anomalies.

Stand-alone tests were performed by DSI both at KSC and in McLean, VA using the GSE. All tests were successful and no anomalies were found. The Shuttle payload bay operated at a cold temperature during the flight of STS-39. Therefore a cold temperature test was performed to simulate flight conditions. This test recreated the flight anomaly.

The flight anomaly was caused by the design of the electronic circuit. The execute switch is used for two functions: To turn on the electronics for 15 minutes; and to initiate switch selections. Due to a time delay caused by the toggling of a relay, the input voltage to the logic is applied before the logic is powered on. When the temperature is cold, this caused the logic to "lock-up" and not respond to further commands. Resetting the logic clears this condition and allows the logic to perform normally. The proper operation, in accordance with the electronics design, is to toggle the execute switch by itself to turn the launcher electronics on, and then to set the address switches and toggle the execute switch again.

Future Mission Capability

Refurbishment Requirements

Because the DSI launcher is flight proven, a minimal amount of refurbishment is required. In order for a payload to use the DSI launcher, there is a certain minimum hardware refurbishment and engineering support required.

Hardware Activities:
- Battery replacement.
- Mounting ring & v-band inspection and surface re-finish.
- Procurement of new clamp separators (although only 2 are required for flight, at least 20 must be procured due to STS ordnance qualification requirements).
- Standard testing (functional and temperature).
- Separation testing.

Analysis, Documentation, and Program Support:

- Separation analysis, which is spacecraft unique.
- Flight Safety Documentation. According to NSTS 1700.7B, must be re-certified for each flight. All analyses for flight 1 are in place, so it is a matter of verifying analyses are still adequate, re-submitting documentation as part of payload unique safety process.
- Support of system vibration testing and pyro shock testing (if customer required).
- Support Orbiter integration at KSC.
- Provide crew training (for SPASP operation).
- Support mission operations at JSC if required.

Baseline Capability

The baseline capability of the DSI launcher is a 3.7 ft/sec deployment velocity for a 190 lb payload. The separation velocity vs payload weight, using the standard separation springs, is shown in Figure 8. If the deployment velocity is too great, smaller, standard springs may be substituted for the existing ones. If additional deployment velocity is required, custom springs would have to be developed.

In addition to deployment velocity limitations, the launcher does have payload weight vs cg limitations, as shown in Figure 8. These are based more on Flight 1 qualification levels than structural capability. If payloads meet this requirement, then little or no additional structural analysis would be required. If the payload does not fit the weight/cg curve, additional analysis would be required.

Motorized Door Option

DSI originally planned to have the capability to operate the GAS fully deployable motorized door. For this reason, two sets of high energy batteries were installed and a switch function was intended to be used to control the door. The DSI launcher could be used with the door by modifying the electronics. The changes made would be to add a printed circuit card which would utilize the Pre-Arm switch as the motor control and the Pre-Arm indicator as the motor control indicator. The one minute timer would now be replaced by the time to operate the door.

Standard GAS Can Launcher

DSI originally intended to utilize the standard GAS Can launcher (known also as the TSI launcher) marman (V) band design. If a payload would like to be (or is) compatible with the TSI V-band, and is in the acceptable weight vs cg range, TSI compatible separation system could be incorporated at a minimal cost.

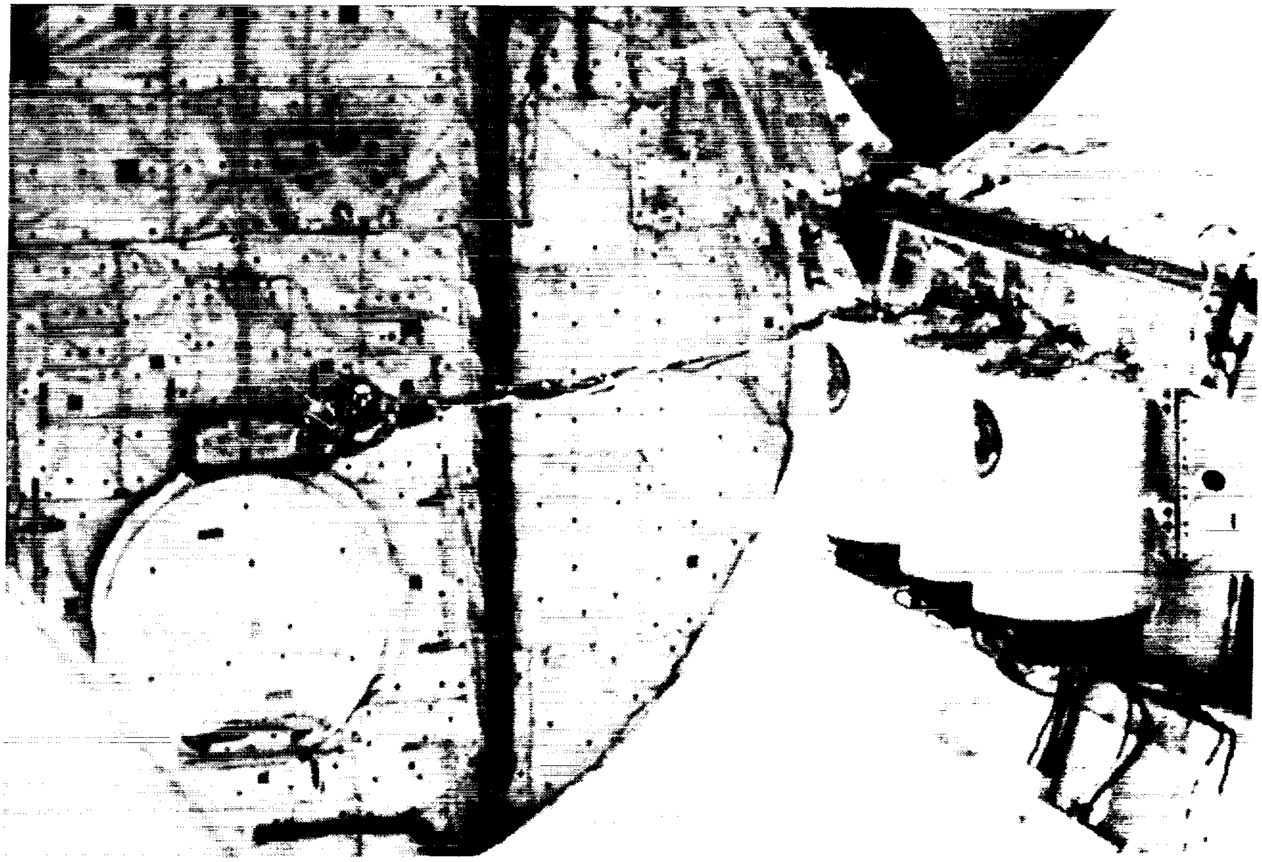

FIG 1. DSI Launchers Installed in Orbiter DISCOVERY

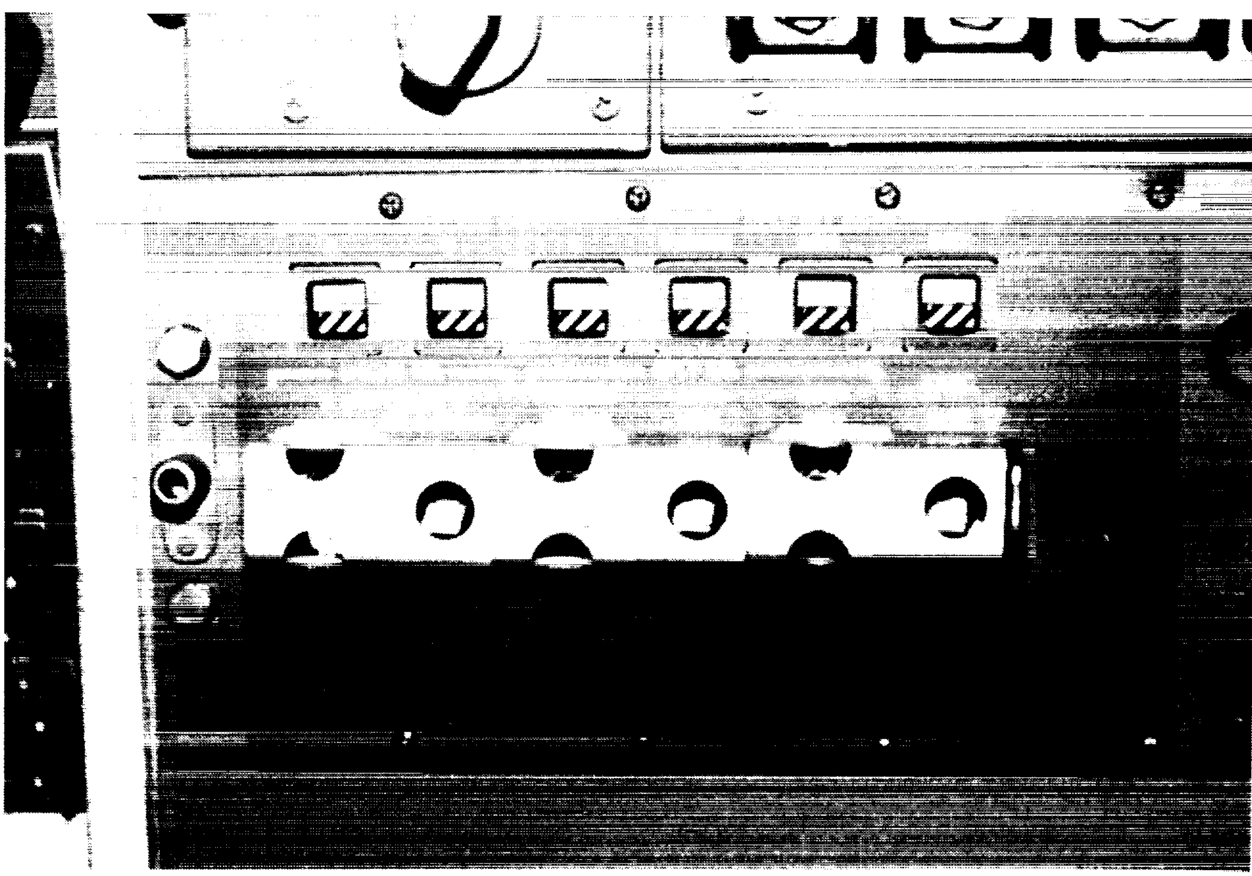

FIG 2. SPA Switch Panel in Aft Flight Deck of Orbiter

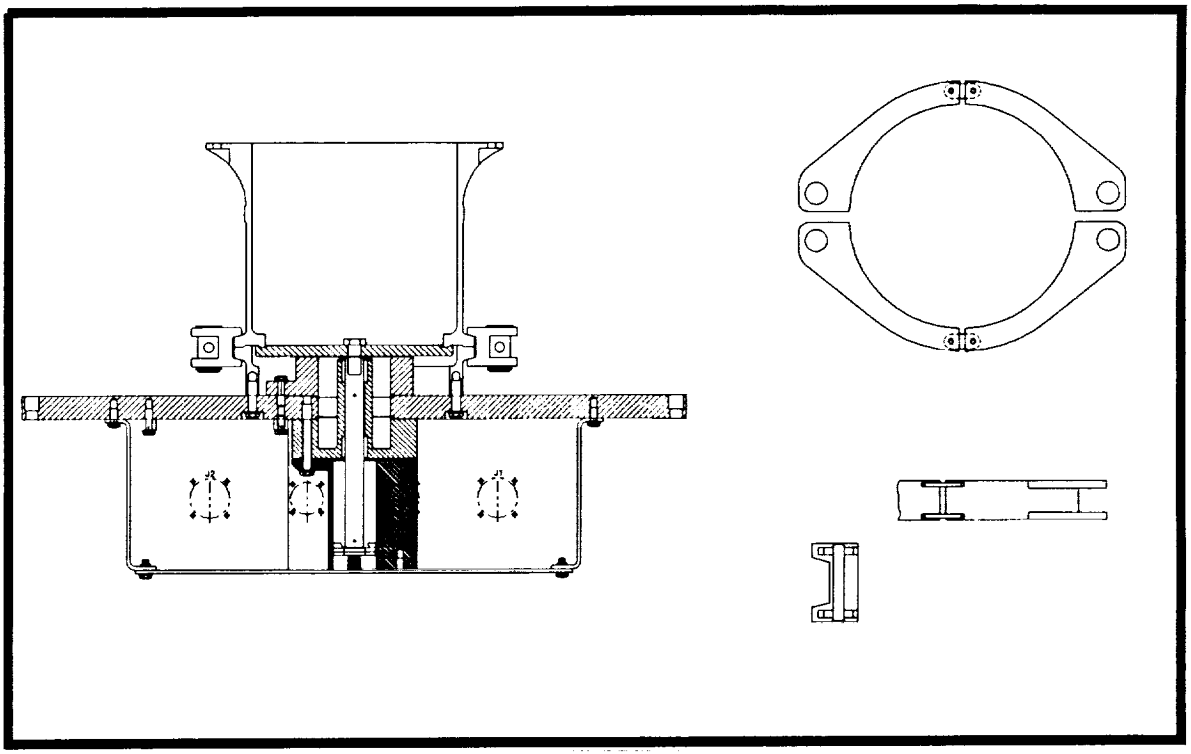

FIG 3. DSI Launcher Assembly With V-Band
(A) Side View (B) V-Band

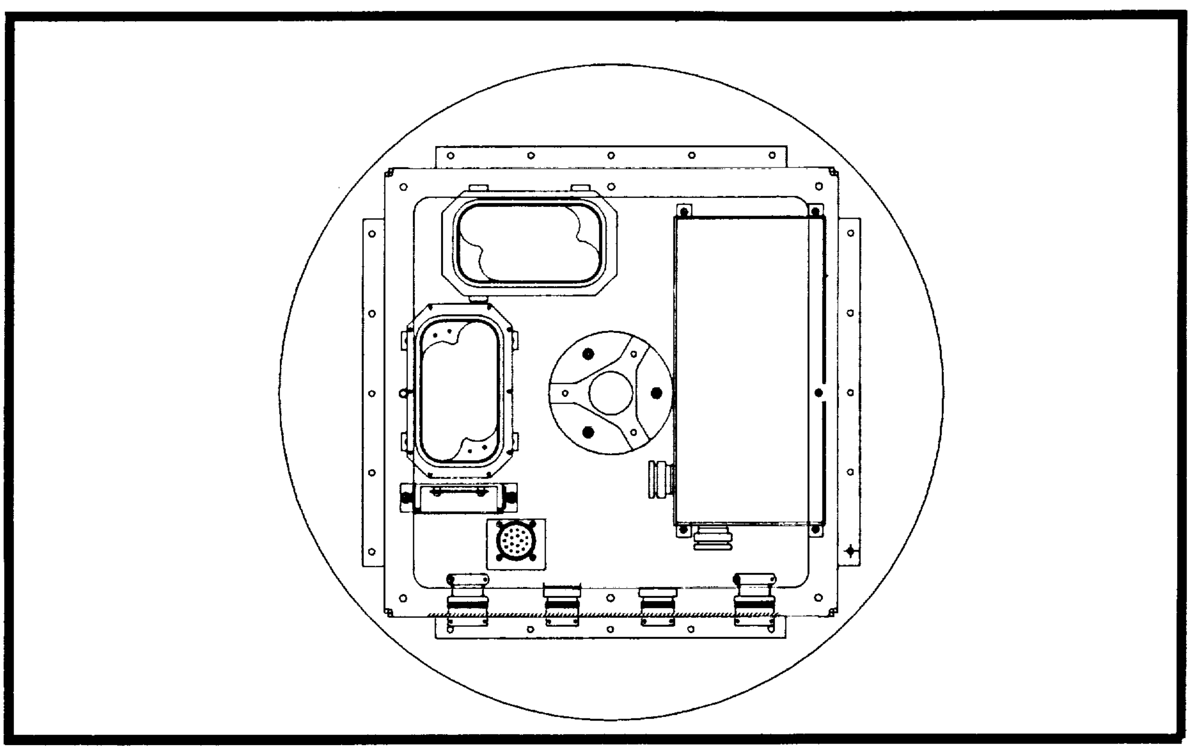

FIG 4. DSI Launcher Electronics Box

FIG 5.   DSI Launcher in Flight Configuration

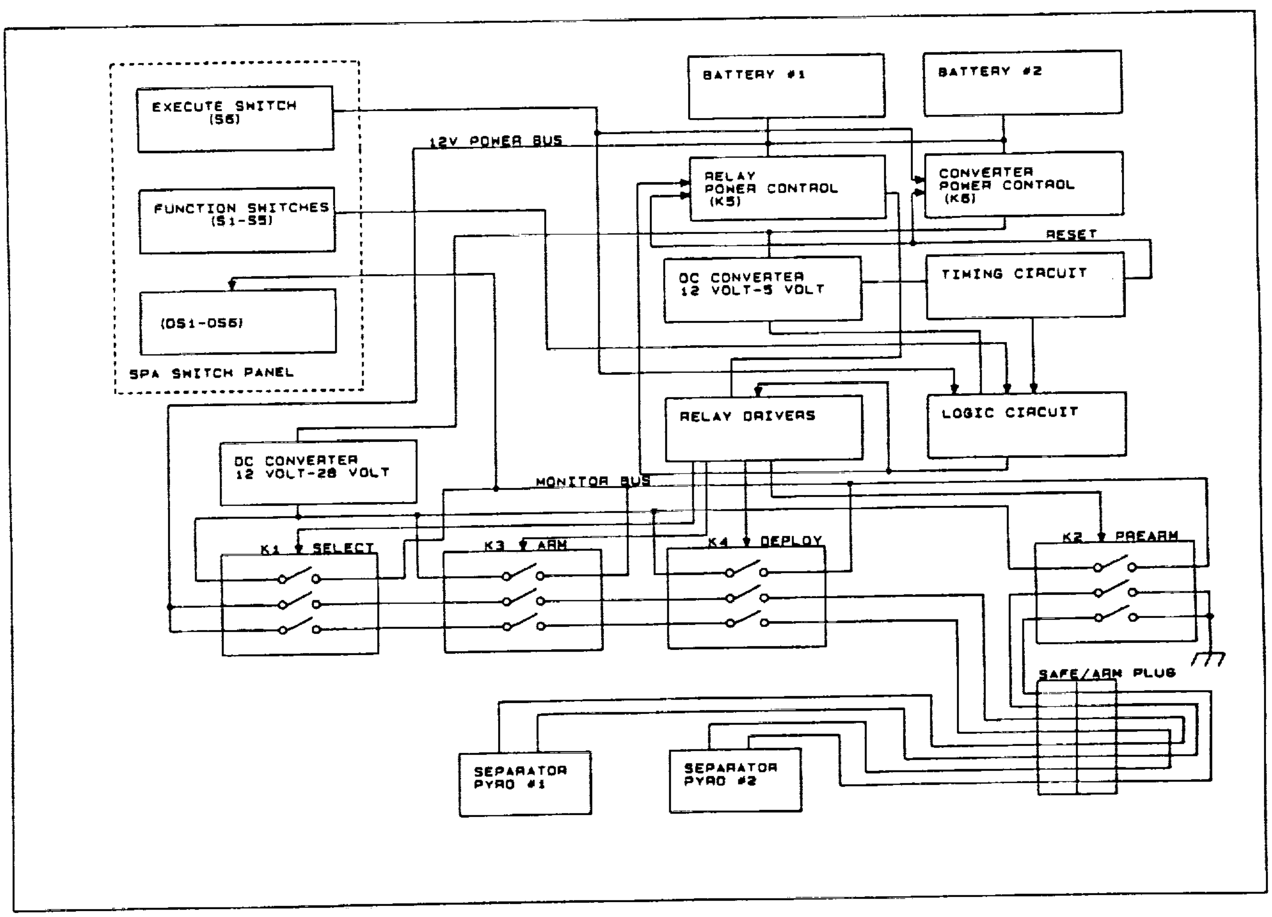

FIG 6.   Electronics Block Diagram

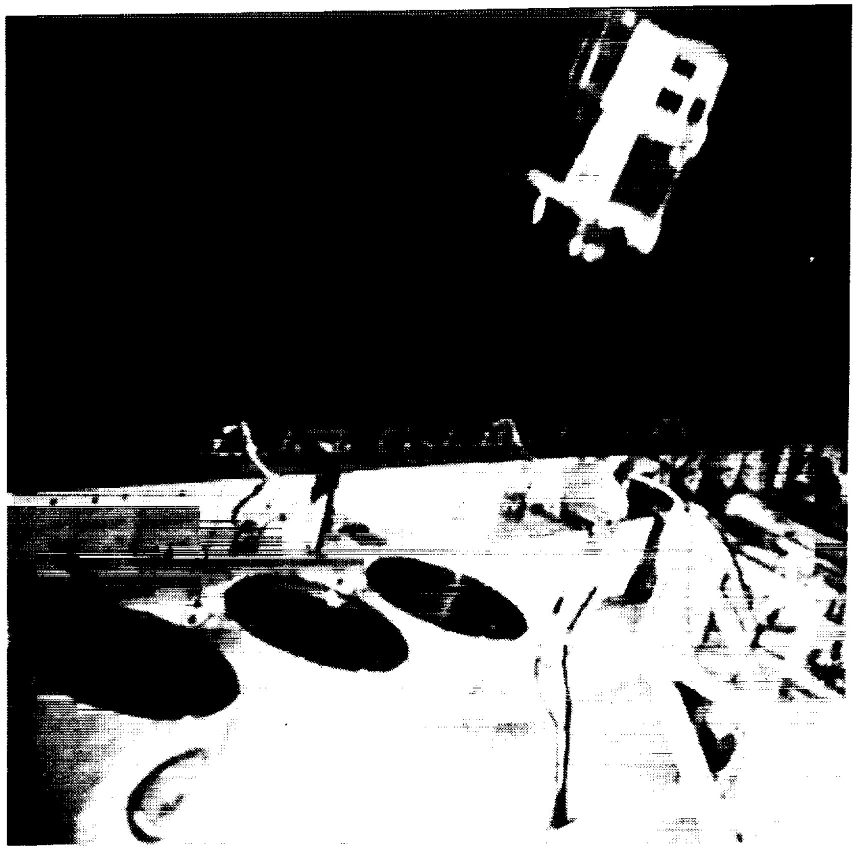

FIG 7. CRO Deployment from Shuttle

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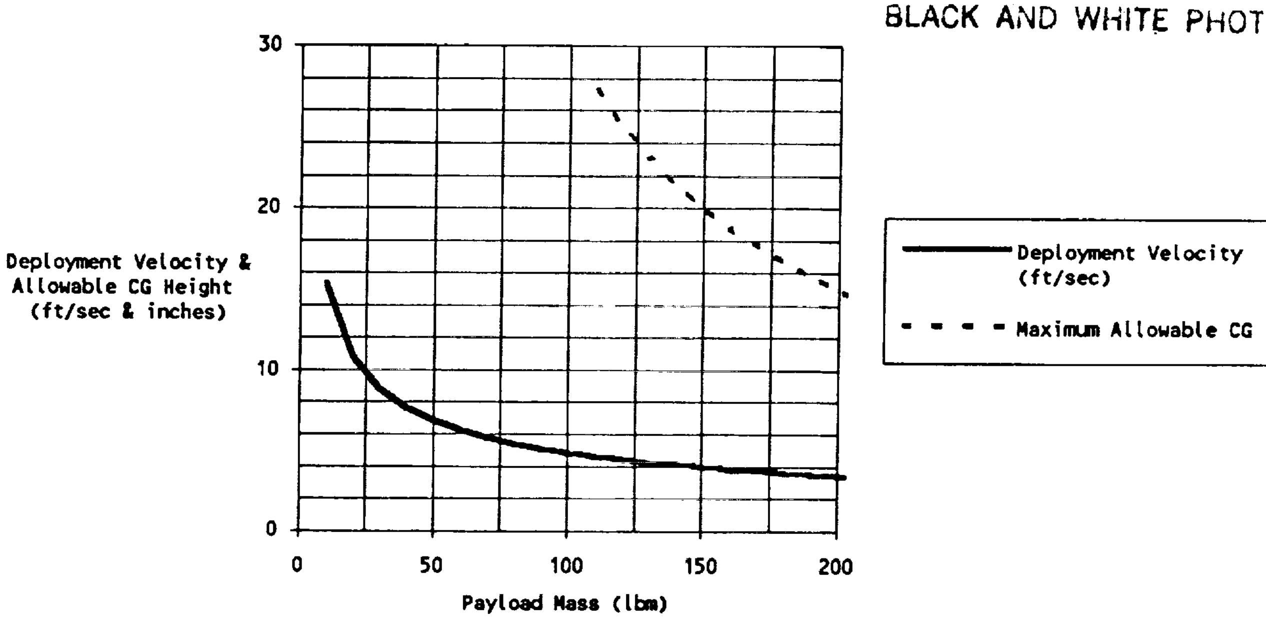

FIG 8. DSI Launcher Predicted Deployment Velocities and Allowable CG Heights as a Function of Payload Mass