

NASA Conference Publication 2401

1985 Get Away Special Experimenter's Symposium



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NASA Goddard Space Flight Center
Greenbelt, Maryland
October 8-9, 1985*

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1985 Get Away Special Experimenter's Symposium

Edited by
Lawrence R. Thomas and Frances L. Mosier
NASA Goddard Space Flight Center
Greenbelt, Maryland

Proceedings of a symposium held at
NASA Goddard Space Flight Center
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TO CATCH A COMET

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Technical Overview of CAN DO G-324

HALLEY'S COMET
SATELLITE BOARD PHOTOG.
ASTRONOMICAL PHOTOG.
IMAGE PROCESSOR

MR 118932

Edited by

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SPACE SHUTTLE MISSION 61-E
CAMERAS
TELEVISION EQUIP.
DETECTION
VIBRATION TESTING
PHOTODIODES
MICROPROCESSOR

Contributors:

- | | |
|--|--------------------------------|
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| Tom O'Brien (Heating, Electrical, MAPC) | Jim Nicholson (Photo, Optical) |
| Tom Collings (Intervalometers, Computer) | David Lee (Software) |

INTRODUCTION

The primary objective of the C.E. Williams Middle School Get Away Special "CAN DO" is the photographing of Comet Halley. As it is currently scheduled, G-324 will fly in March of 1986 aboard STS 61-E. The project will involve middle school students, grades 6 through 8, in the study and interpretation of astronomical photographs and techniques. The objectives of these studies are detailed in another paper to be presented at this symposium.

G-324 is contained in a 5 cubic foot G.A.S. Canister with an opening door and pyrex window for photography. It will be pressurized with one atmosphere of dry nitrogen. Three 35mm still cameras with 250 exposure film backs and different focal length lenses will be fired by a combination of automatic timer and an active comet detector. A lightweight 35mm movie camera will shoot single exposures at about 1/2 minute intervals to give an overlapping "skymap" of the mission. The fifth camera is a solid state television camera specially constructed for detection of the comet by the microprocessor. In addition to the photographic goals of CAN DO, the canister also contains a separate package of middle school student designed passive experiments. These projects will be described in detail in the accompanying paper on educational goals.

DESIGN PHILOSOPHY

The key word in any design is simplicity. The simpler you can make a working system, the more reliable it is. Particular attention should be paid to good construction practices. A clean neat layout not only looks good, but has a much better chance of working in flight. It's best to start with a basic mission definition and design a simple package to meet that definition. The bells and whistles can come later and they should in no way compromise the basic design by their failure. Use redundant high quality parts whenever possible. This is not the place to skew the cost/benefit ratio too far. Don't overlook a major resource in your Design Reviews, team members of other disciplines. Questions that require explanations of principles and concepts sometimes reveal a flaw that has been overlooked because of familiarity with the design. Perhaps, even a way to simplify it further can be found. When designing redundant protection, it's very easy to get paranoid about remote failures. It's not a black and white issue and the design review process can be most helpful here. You can bet that sooner or later something will fail. The trick is to design the system so the failure occurs after the mission is complete. Preferably LONG after.

With this in mind, we have designed CAN DO with several small independent control subsystems. These can continue the mission at a reduced capacity in the event of most failures short of the door not opening.

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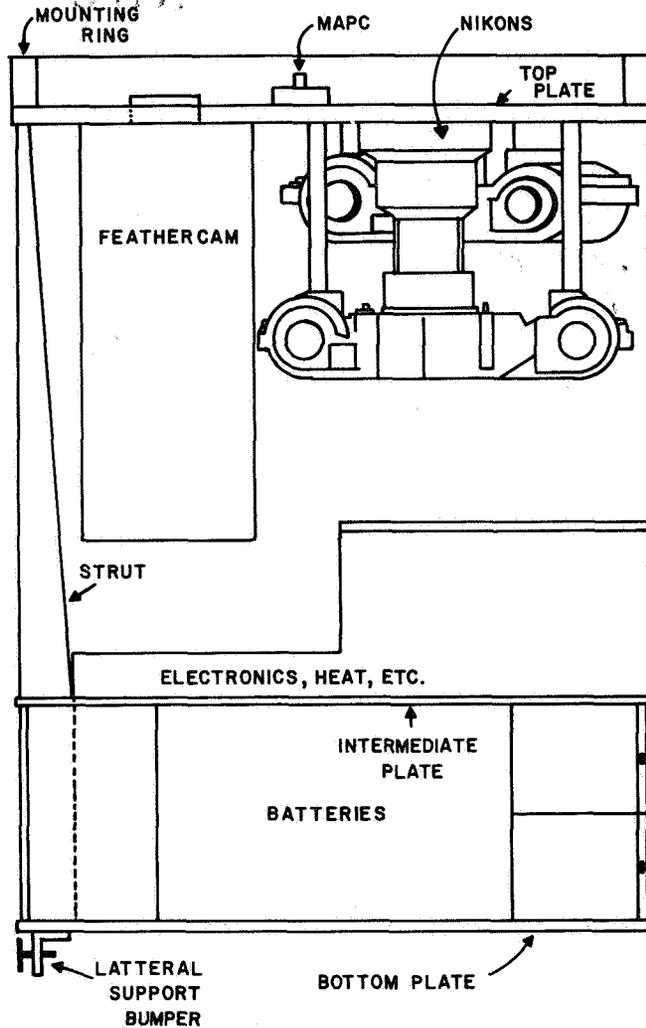


Fig. 1

STRUCTURAL

The G-324 payload structure (Fig. 1) consists of an upper and a lower plate at the top and bottom of the canister, separated by 3 struts. These struts are bolted at 120° intervals on the plates' perimeter. The top plate is bolted to the experiment mounting ring through a spacer ring which has a thermal barrier at the mounting ring and supplies clearance for the insulation and lens openings. The bottom plate will be laterally supported by three adjustable bumpers which will press on the canister wall. The cameras are mounted on the lower face of the top plate and their lenses protrude slightly through the holes in the top plate. Mechanical support for each lens is provided at its opening. The batteries are mounted in four battery boxes. Three are mounted between the struts and bolted to the bottom plate. The fourth is bolted to the top of the intermediate plate. All battery terminals are facing out for easy accessibility. The electronics and electrical systems are mounted on an intermediate plate which forms the top of the battery box assembly and is also attached to the three struts. All structural material is machined from alloy 6061-T6 aluminum. This alloy was chosen for its high resistance to stress corrosion, and it does not require a NASA Material Usage Agreement. Stainless steel bolts and lockwashers were chosen for their strength and compatibility with the aluminum. The stress analysis was done by hand calculation techniques and based on the published ultimate strengths of the material and load factors, as given in the NASA Safety Manual.

Extensive shake and vibration tests are scheduled for the structure with the cameras and batteries mounted before the electronics are added. Additional shake and vibration tests as well as environmental tests are scheduled for the completed canister.

THERMAL AND HEATING

In addition to the standard canister insulation of Dacron, aluminized Kapton and Beta cloth, G-324 will be wrapped in aluminized mylar to form the equivalent of a "vacuum bottle effect" at \emptyset g between the payload and the interior of the canister. The top plate will be insulated from the experiment mounting ring by an insulating ring on top of the 1 3/8" mounting ring to block heat conduction. All through bolts to the experiment mounting ring will also have insulating washers and sleeves to further reduce conductive heat loss.

Due to the insulating effects of the closed lid, thermal loss prior to the actuation of GCD-C (which opens the door) is considered to be at the rate of a container with an insulated lid. This, combined with the large thermal mass of the battery stacks should give us a starting ambient temperature several degrees above 0°C. As seen in Fig. 2 and 3, the projected mission life at the designed ambient temperature should exceed 30 hours.

The lack of convection at \emptyset g which enables us to form this "vacuum bottle" effect also causes several problems. Since STS-61E is a deep space oriented flight with no sun exposure for the cargo bay, temperatures can effectively form a -70°C heat sink. As a result, we will need to supply at least 22 watts of heat on a continuous basis to replenish the heat lost by conduction and radiation. G-324 has two heaters mounted on a finned heat exchanger and an atmospheric circulation system. If one heater fails, the remaining heater can supply the needed heat to maintain 0°C.

An EG&G Rotron Mil-80 DC fan provides the primary nitrogen atmosphere circulation through the heat exchanger and canister. This primary fan has an integral speed comparator. This comparator switches in an auxiliary fan in the event of a low or no speed condition. The reference speed was set below the fan speed running at the projected end of mission voltage level. Circulation volume at the end of the mission will still be more than required for thermal stability.

The dual heaters have both primary temperature and secondary overtemperature thermostatic control. Two paralleled Elmwood 3153 Hi-Rel thermostats with a 2.2°C differential provide a redundant primary temperature setpoint of 0°C. As overtemperature protection, two additional paralleled 3153 thermostats are in series, with a set point at 5.5°C. These will open on a temperature rise in the event a primary thermostat sticks closed with a welded contact.

ELECTRICAL

As seen in Fig. 5, power is supplied from five battery stacks. The four primary stacks supply 28 volts at 156 ma, 24 volts at 105 ma, 12 volts at 220 ma and 12 volts at 3.6 Amps. The fifth stack acts as a redundant source to both the 28 volt and 24 volt stacks through blocking diodes, and likewise, the 24 volt stack acts as a redundant source for the low current 12 volt stack. All battery stacks are isolated from each other by schottky diodes to prevent discharge into lower voltage stacks.

The 28 volt source supplies 60 ma to the door opening circuit and 96 ma to the main power relay. The Leach KD-2A 4PDT relay will drop out at 7 volts. The door will close when the source voltage drops to 24 volts.

The 24 volt source supplies 100 ma to the Matrix Array Photodiode Camera (MAPC) for comet detection, an averaged 25 ma to the Feathercam 35mm movie camera and 1 ma to the Feathercam's intervalometer. The MAPC will operate down to 12 volts and the Feathercam will drop out at 16 volts.

The low current 12 volt source supplies 176 ma to the fan, 40 ma to the microprocessor for detection of the comet and 4 ma to the intervalometers that automatically fire the three Nikon cameras. All loads on this supply will operate down to 5 volts.

The high current 12 volt source supplies 3.6 Amps to the thermostatically controlled heaters. Heat lost from the canister can be replaced by 22 watts of continuously supplied heat. At this rate, 0°C ambient can be maintained in the canister until the battery voltage reached 8.4 volts. At this point, the thermostats remain on and the canister begins to cool. As the temperature drops, the mission will end either by the door closing or thermal shut-down of the cameras.

Schottky diodes were chosen for blocking diodes because of their very low forward voltage drop that, in our application, is on the order of 300mV compared to standard silicon diodes at about 1V or better. Two are paralleled at each location for redundancy. The battery stacks are fused on the negative lead to ground by special Bussman GOV Hi-Rel fuses.

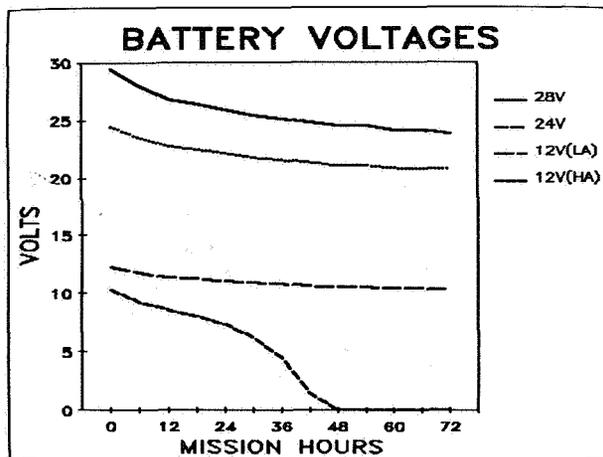


Fig. 2

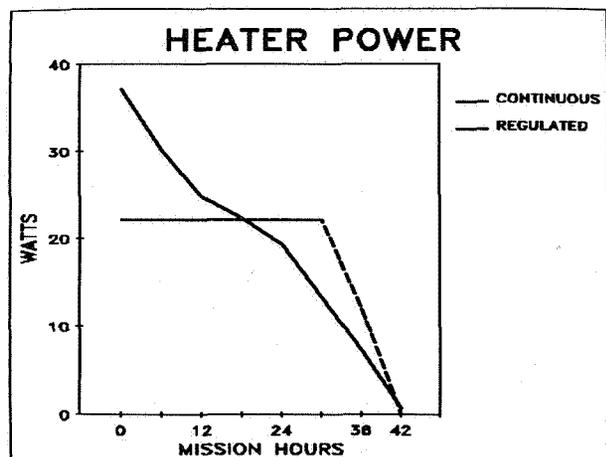


Fig. 3

BATTERIES

The Duracell Industrial Alkaline ID9150-6V and ID9260-12V 20 AH batteries used were extensively tested under expected and stressed conditions. To predict mission life, a set of batteries was chilled to +2°C for 5 days and then discharged through the projected loads for each supply. Graphs of the voltage drop vs. time were then used to estimate mission life and to set photo intervals. In addition, one chilled battery was shorted with two 14 ga. wire straps to determine a direct short hazard. Post-short examination revealed that pressure was released at the cell seal. There was no evidence of swelling of the cell can, so explosion under a direct short is unlikely. Pressure venting seems to occur above 90°C. Several batteries were then frozen to -70°C and showed no evidence of leakage or rupture. They performed quite well when brought up to 0°C. It now appears that the temperature drop due to heat loss and depletion of the heater batteries will determine the length of the mission.

PHOTOGRAPHIC LENSES

Plans for the fulfillment of the primary photographic mission of CAN DO were based on several assumptions. Most challenging were the somewhat disappointing estimates of the brightness expected from Halley on this apparition because of the relatively unfavorable viewing angle and distance from the Earth. In addition, the drift rate of the shuttle itself can not be accurately predicted but will limit the length of exposure possible without unacceptable image streaking. We chose full color pictures which ruled out electronic image intensification as an option. Consequently, tests were designed to find the fastest possible films and lenses that would produce images of adequate quality in a short enough exposure time to minimize shuttle-movement blurring.

After consultation with the National Geographic Society, three lenses were selected for testing and obtained on loan through the courtesy of Nikon Inc. The 35 mm f2 wide angle Nikkor lens with a 65 degree angle of view was selected to be mounted on an intervalometer fired Nikon F3 to provide a wide enough field to include the full visible tail of the comet with enough background stars for adequate mapping. While not the fastest 35mm lens available, NGS's experience indicated that it would give superior results. Our tests verified this impression, and the lens gives a bright, clear image free from noticeable distortion, even at the edges.

Two high speed 58mm f1.2 "Nocturnal" Nikkor lenses were selected for use, one on a Nikon F3 and the other on the Feathercam 35mm movie camera. These lenses were selected for their exceptional speed, their 40 degree field of view, and for their design which is optimized for full aperture use with high contrast subjects. Astrophotography tests verified the competence of these lenses, and although they show a slight "coma" distortion for extremely bright stars at the edge of the field, this is considered a small price to pay for their exceptional speed.

The final photographic lens chosen was a Nikkor 200mm f2 ED telephoto lens with a 12 degree field of view. Mounted on a Nikon F3, this lens with its narrow field is primarily designed to be fired by the microprocessor when the comet is actually in view. In the event of the MAPC failure, only a small percentage of shots will likely hit the comet. This lens also tested well for astronomical subjects and it is hoped that it will justify its weight and size by returning spectacular "close-up" views of the comet.

MAPC LENS

The hardest lens to find was the imaging lens for the MAPC comet detection solid state video camera. Image quality was not a primary consideration because of the low resolution (32x32) photodiode array, but high speed was, as tests indicated that the predicted brightness of the extended comet would stretch the sensitivity of the chip beyond the maximum design limits. Most difficult was finding a lens with a 20-25 degree angle of view that produced an image small enough to fit the 2.5mm square active area of the array. Early tests indicated that a 10x microscope objective used inverted would produce an adequate image but the f6.3 speed and a bad tendency to flair in the presence of a bright object were disappointing. We have shifted our attention to a Rolyn Optics Coranar series 12mm f1.9 lens. This lens has a 50 degree field of view but overcovers the chip, producing an effective field of view of approximately 20 degrees. It comes in a very tiny (13mm x 12.5 mm) and lightweight (1/2 ounce) package. The size is important as the MAPC camera will be mounted above the main plate to achieve maximum chilling to reduce dark current. An added benefit of this lens is that its relatively fast f2 "photographic" aperture should brighten extended objects like the comet while its tiny 6 mm clear or "astronomical" aperture will suppress point source objects like stars.

CAMERAS

A 35mm format was chosen as being large enough for image quality while being small enough for high capacity. Also considered was the large selection of high speed films available in this format. Nikon F3 High Eyepoint cameras were selected for their proven record in space and were obtained on loan from Nikon Inc., with the support of the National Geographic Society. Each Nikon is equipped with a MV-4 motor drive and a MF-4 250 exposure magazine back. The cameras will be powered by their internal (alkaline AA) battery packs. Tests indicate they have a capacity far in excess of requirements, even allowing for a 90 day waiting period and cold mission temperatures.

In addition to the three Nikons, there will also be one Continental Camera Co. Feathercam 35mm motion picture camera on loan from the manufacturer. This ultra lightweight movie camera has a 400 foot magazine (5000+ 35mm pictures) but is only slightly larger than a typical 16mm camera. Its capacity will allow it to be fired approximately once every half minute throughout the mission.

FILM

It was originally intended that at least two of the many new high speed films now coming onto the market would be flown, one positive and one negative. The films would be selected on the basis of thorough testing. It is beyond the scope for this paper to detail the results of this test, but the information is available on request. In brief, 10 films were selected, including four color slide films (ISO 400-1600), four color negative films (ISO 400-1600), and two black and white films (ISO 400-800). All major manufacturers were represented.

The main test was conducted with the cameras mounted "piggy-back" on a 12 inch Questar telescope equipped with an equatorial drive. The Orion constellation was selected for a target because of its combination of bright and easily identified stars and a prominent nebula to serve as an extended source. In addition it was noted that the Orion region is being used as a calibration standard by the International Halley Watch and other organizations. An evening was found when Orion was well situated against a relatively dark rural sky and near ideal "seeing" conditions prevailed. A complete series of exposures from 1 second to 5 minutes in length were executed with both the 58mm f1.2 and 35mm f2 lenses on each of the ten films. The range of exposure proved adequate for every film to cover the range from gross underexposure to an exposure where the sky "burned in".

All films were carefully developed to manufacturers' standards and then evaluated with a thorough battery of subjective and objective tests. These tests included enlargements up to 40x, microscopic examination for greatest magnitude detected, and evaluation for such criteria as grain, resolution and color fidelity. The most damning test for many films was found to be their inability to produce a dense saturated black background when printed to show the best detail in the stars and nebulae. This criteria, in fact, ruled out all of the color negative materials which were notably inferior in this regard. Surprisingly, the test was won "hands down" by

Ektachrome 800-1600. Even compared to slower films, it produced the best image quality, while easily capturing the title of the fastest color film tested. When enlarged as much as 20 times, this material showed brilliant color against a pure black, seemingly grainless background. The results were so conclusive, in our opinion, that we are going to fly it as our only film. This result is in direct contradiction to several other published reports, including one by NGS. These reports did not include astronomical subjects as part of their tests. We feel this emphasizes the necessity of conducting your own test under the particular conditions you will encounter. You should not rely on tests conducted by others, which may be irrelevant to your application.

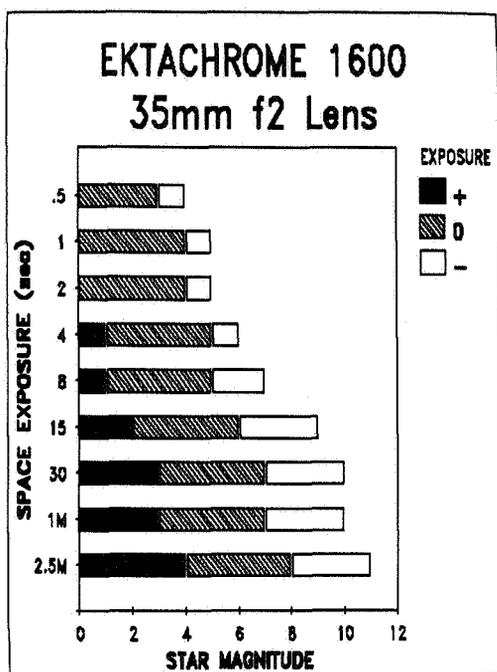


Fig. 4

Table 1
FILM TEST SUMMARY

FILM	SCORE
Ektachrome 1600	68
Ektachrome 400	52
Fujichrome 400	50
Kodacolor 1000	48
Agfacolor 400	48
Ilford XP-1	42
Fujicolor 1600	41
Kodak Tri-X Pan	41
Agfacolor 1000	40
Agfachrome 1000	35

Scoring System

Film Speed	(2-8) pts.
Sensitivity (1 sec).	(1-5)
Sensitivity (15 sec)	(5-8)
Range.	(2-5)
Resolution	(0-9)
Color.	(3-9)
Grain.	(3-9)
Black Sky.	(1-9)
Print Quality.	(1-9)

EXPOSURE

The photographic tests were also used to determine exposure times for the mission. It has proven difficult to get complete information on either the expected brightness of the comet, or the predicted drift rate of the orbiter. The final exposure will be set closer to flight time and will be based on the latest available information. At present, we are considering a baseline exposure of 10 seconds for the f1.2 lenses, and 15 seconds for the f2 lenses. These exposure times are felt to be the best compromise between adequate exposure to show significant tail detail while being short enough to minimize drift problems. This baseline exposure will be used for all "automatic" intervalometer fired photos. The same time will also be used for the first shot taken by the "smart" cameras in each comet window. Each additional photo taken during active MAPC comet detection will be doubled in time to produce more detail in the dimmer reaches of the tail, and to avoid unnecessary duplication. The final maximum exposure will be determined by the actual length of the comet window but it could reach as long as 10 minutes in the several 30+ minute windows currently shown in the Astro 1 mission profile. There is a relatively small chance that these extremely long exposures will not be spoiled by drift, but if successful, they will produce the most significant photos of the comet tail structure. The opportunity to obtain such photos is solely dependent on the ability of the MAPC detector to determine when the comet is, in fact, in the proper line of fire. This justifies the considerable effort in designing a system to add some 200-300 aimed shots to the possible 1100 shots of the comet which may be obtained by intervalometer alone. This figure, based on scheduled window time, is misleadingly encouraging because most of these photos will be nearly exact duplicates of each other and will offer little new information.

INTERVALOMETERS

In the pursuit of redundancy, each camera is controlled by its own independent timer or intervalometer. This allows both the interval time (the time between pictures) and the exposure time (the duration) to be "programmed" by simply changing a resistor value. Each intervalometer is composed of three integrated circuits. The 4060 functions as a multivibrator whose frequency is set with two resistors and a capacitor. This oscillator runs at a few hertz using small RC values. Because the chip also contains a divide by 16,384 counter, the output changes only every 15 to 30 minutes. This allows accurate timing into the hour range. Its output is used to trigger a 7555 which is operating in a monostable or one-shot mode. When the 7555 is triggered, its output becomes active until the exposure time is complete. This time is a function of its R.C. values. The output is buffered and inverted by a 74CO4 which controls the camera shutter directly. All of the timers are held in the reset condition while the door is closed. The 74CO4 is replaced with a 74CO2 in two of the intervalometers which allows either the sequencer or the Image Recognition Computer to fire the cameras and determine the exposure time.

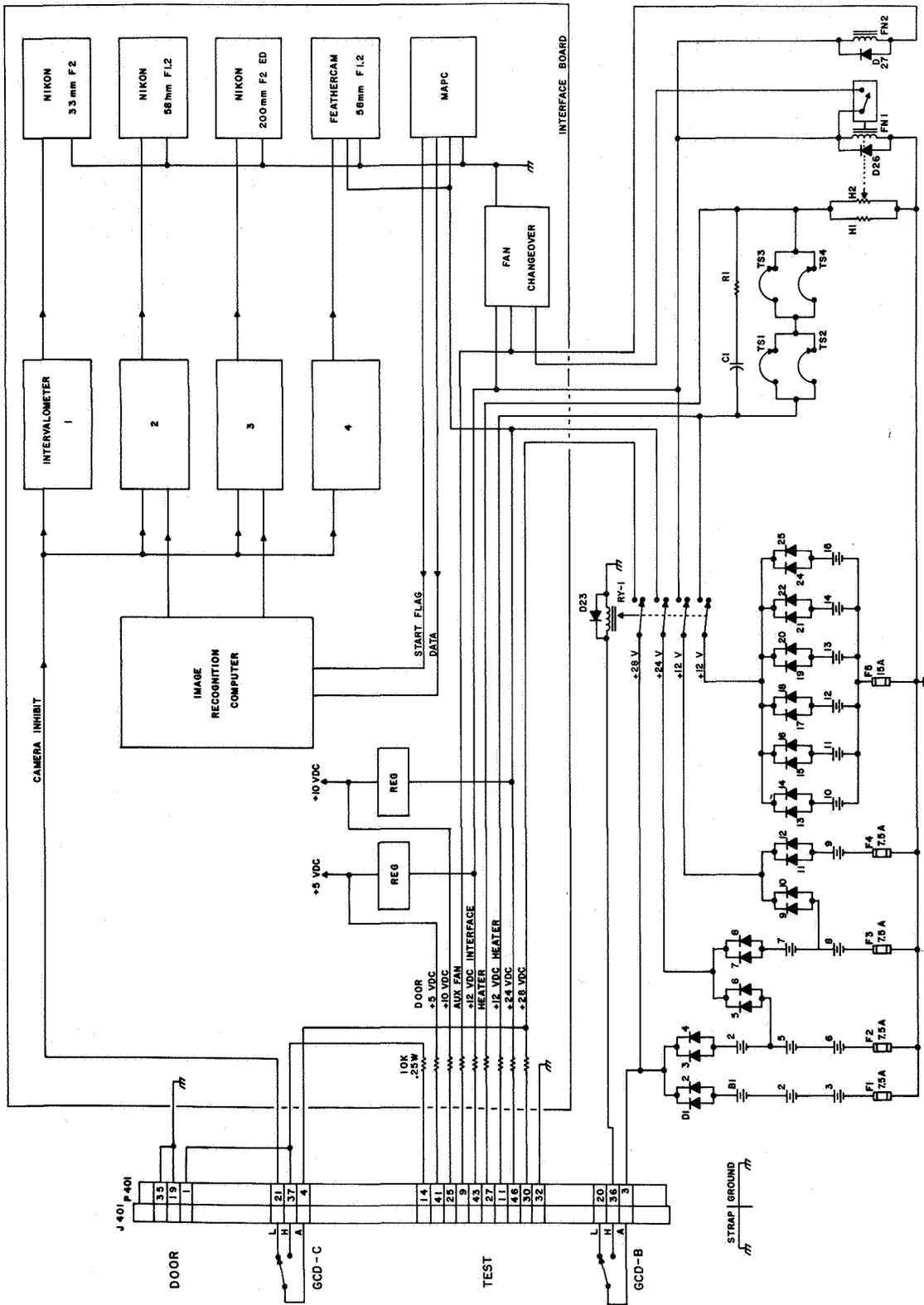
MAPC

The heart of the MAPC or Matric Array Photodiode Camera is an EG&G Reticon RA32x32A imaging chip. This chip has 1024 photodiodes arranged in a 32 by 32 array. Each photodiode is one small picture element or pixel. To run the array chip, a crystal controlled master clock oscillator of 4.9152 mhz is divided by 512 to produce a 9.6 khz clock. This clock is used to encode the data output for transmission to the Image Recognition Computer. It is further divided by 2 and the resulting 4.8 khz is gated to suppress every fifth pulse. This fifth pulse pause is used to insert the start and stop bits in the digital output. The gated 4.8 khz clock drives the sample and hold stage, digital latch and the X register of the RA32x32A. One pixel is clocked out of the array for each gated clock cycle. At the end of each line of pixels, a flag advances the Y register to the next line. As each group of four pixels is clocked out, they are buffered and sampled by the sample and hold stage. Each pixel level is held while it is converted to a 2-bit digital value. This value indicates four distinct video levels or picture brightnesses, white - gray - black and "blacker than black" or blanking. Each group of four Pixel values is then combined with a start and stop bit to make a digital word. The 10-bit words are transmitted serially at 9600 baud to the microprocessor along with a separate "start word" flag. At the end of each line and at the end of the field, a flag is sent to the Image Recognition Computer by transmitting a blanking bit pattern in place of a dummy pixel in the data stream. There are two dummy pixels provided at the end of each line for housekeeping, both in the chip and by the user. One blanking group indicates the end of the line (EOL) while two in a row indicates the end of the field or picture (EOF). These are used by the Image Recognition Computer to reconstruct the image for comet detection. Power from the 24 volt stack is pre-regulated to +12 VDC by a linear monolithic regulator. This feeds a +5 VDC and +7 VDC regulator as well as a DC-DC converter. The converter operates at 10 khz using a Siliconix Si7661 CMOS voltage converter to supply the -7 VDC needed by the camera.

IMAGE RECOGNITION COMPUTER

The Image Recognition Computer is built around the NCR 65C02 microprocessor. This processor was chosen because it is simple to program and the Apple II computer could be used as a development system, once a monitor and interface program were completed. The system is designed to have 8K of memory in blocks of 2K each. The memory can be static RAMs (61LL6s) or EPROMs (2716s). The firmware program, once completed, will reside in EPROM at the top of the 8K memory space. The main scratchpad memory starts at location zero. For program development, the EPROM is replaced by RAM. After the program is edited and assembled on the Apple, it is downloaded at 4800 baud to RAM and debugged. The interface to the Apple and the cameras is through the NCR 65C22, an interface IC with two eight bit ports and two timers. To isolate the cameras from any possible computer malfunction, the controlling signal from the NCR 65C22 triggers a monostable multivibrator (7555) instead of the cameras directly. This allows the camera to fire only once in a given period of time. The end of the exposure is signaled by the 65C22 resetting the monostable. A "heartbeat" circuit is also installed to assure correct operation of the processor. This circuit will reset the processor every 30 seconds if the processor "gets lost" for whatever reason, and continue to reset until the program starts again. The circuit is composed of a 7556 dual timer. The first timer operates in the monostable mode while the second timer is in the astable mode.

ORIGINAL PAGE IS
OF POOR QUALITY



G-324 MAINFRAME — ELECTRICAL DISTRIBUTION			
DESIGN	DRAWING	REV	DATE
THOMAS COLLINGS	THOMAS O'BRIEN	3A	06-01-85
THOMAS O'BRIEN			

Fig. 5

SOFTWARE

The software is divided into four areas or routines: data acquisition comet detection, system maintenance and a run-time system. The data acquisition routine is responsible for reconstruction of the image from the MAPC. After reception, the end of line and end of field flags used in the reconstruction are removed, and the image is passed to the comet detection routine.

The algorithms for digital image processing and pattern recognition will be empirically defined using simulated comet images in planetariums at South Carolina State University and the University of North Carolina. Our present research has centered on the use of a simple neighborhood averaging filter to suppress isolated bright values (stars) while reinforcing diffuse bright objects (comet). This is followed by a simple pattern recognition algorithm which checks for the presence of a large, relatively homogenous, bright object having a recognizable axis which is significantly brighter than the surrounding background. This simplistic routine is possible because no other astronomical target, including the sun and moon, is larger than .5 degrees, while the comet should cover at least 10 degrees. Possible sources of false positives could be the Milky Way or the illuminated edge of an approaching Earth sunrise. The Milky Way should be suppressed by the small "astronomical" aperture of the imaging lens and its spotty brightness variations should make it relatively susceptible to the averaging filter. No special protection has been planned against a terrestrial false positive as little or no Earth viewing time is scheduled in the current mission profile.

When the comet detection routine indicates that the comet has been acquired, the shutter control subroutine will proceed with an exposure sequence. This controls the shutters on the 58 and 200mm Nikons by generating pulses on two separate pins of the NCR 65C22. One will open the shutters by starting a monostable multivibrator while the other closes them by resetting it. The multivibrator will automatically reset after ten minutes if the computer should fail, returning control to the intervalometers.

System Maintenance software includes initialization routines, a real time clock and the Run Time System. At power up or after a reset cycle, the initialization software configures the timers and I/O ports, initializes the symbol table, starts the real time clock and finally enables interrupt requests. The real time clock is driven by timer T1 on the NCR 65C22. A countdown timer which pre-sets to 50,000 will request interrupts at 20 Hz. Counting these 20 Hz "ticks", the real time clock will maintain the seconds, minutes, and hours registers.

The Run Time System is an executive program, responsible for calling the acquisition and processing routines. It also monitors the system for error conditions and takes corrective measures. The Run Time System is responsible for generating the "heartbeat" pulse on one pin of the output port which indicates that the microprocessor has bootstrapped successfully, and that the software is functioning. Should the software fail (unsuccessful bootstrap, trapped in a loop, interrupt thrashing, etc.), the Run Time System would no longer produce the heartbeat. An external resettable multivibrator will sense the loss of the heartbeat and initiate a processor reset cycle and continue to reset until the heartbeat is recovered. The Run Time System can also detect failure modes in data transmission, image processing and film use.

POSTFLIGHT IMAGE PROCESSING

The CAN DO team is fortunate to have access to a Zeiss IBAS image processing and analysis system for postflight image enhancement and evaluation. We have included this potential in designing our photographic parameters. Most likely to be useful will be digital processing to suppress photographic grain, to mask excess contrast, to enhance faint detail, and to remove image streaking if necessary. If MAPC directed exposure series are available, we will use this apparatus to construct composite true color and density graded false color images of the comet. It will, in addition, be possible to conduct accurate astrometry measurements of comet structure using background stars for scaling.

FLIGHT DATA RECORDER

The original design included a separate microprocessor with non-volatile RAMs to serve as a flight data recorder. It was intended to record information to assist in photograph evaluation including camera firing times and sample digitized MAPC images. In addition, temperature and voltage data were to be logged to produce general information on the operation of a glass lid canister in an extremely cold (constant deep space orientation) environment. This independent microprocessor became a victim to the great weight purge following the very belated discovery that there is a 45 lb. weight penalty imposed on the use of an opening lid with the pyrex window.

Flight data recording functions in the final configuration will be handled as a secondary function of the image recognition computer.

TO CATCH A CHILD'S IMAGINATION

Educational Overview of CAN DO G-324

8955

MR 7-18932

by
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C.E. WILLIAMS
MIDDLE SCHOOL

EDUCATION
STUDENTS
SCHOOLS
CREATIVITY
LEARNING

SPACE SHUTTLES
HALLEY'S COMET
EXPERIENCE
TRAINING DEVICES
EXPERIMENT DESIGN
INTRODUCTION

When a public school district is involved in the flight of a GAS package, they have a serious responsibility, along with an unprecedented opportunity. The desire to produce an experiment with real scientific validity must be tempered by the schools' primary mission to make the adventure educationally meaningful to the greatest possible number of students. Such a project can easily become the personal plaything of the select few that are highly scientifically motivated, and technologically gifted. A public school system must, in addition, insure that the resources invested in such a project also reach out to the average student, and those whose interests lie in other directions. Such a broad reach is only possible if the GAS canister is but one part of a comprehensive and ongoing educational effort that involves as many disciplines and levels of student interest as possible.

The total program should include:

- 1) A background of previous programs which have trained teachers and prepared students to fully utilize special activities as part of the educational process.
- 2) Direct student involvement in the design of experimental packages to be flown, and in the evaluation of data derived.
- 3) Supplementary related activities that use a Get Away Special as a motivational catalyst to generate efforts in other disciplines, other schools and other student groups.
- 4) Future plans to fully exploit and maintain the momentum developed by this one dramatic event.

The one overriding goal of all these activities as articulated by one of our leading classroom teachers:

"TO CHOOSE AND DEVELOP HIGHLY MOTIVATIONAL ACTIVITIES WHICH WILL CAPTURE THE IMAGINATION AND INTEREST OF STUDENTS, AND TO TAKE ADVANTAGE OF THE "TEACHABLE MOMENT" WHICH IS SO ELUSIVE."

BACKGROUND

CAN DO is a Charleston County S.C. public school project with C.E. Williams Middle School serving as host. Students directly involved in experimental design are drawn from the 14 middle schools (grades 6-8) in the county. Other area schools, primarily in neighboring Berkeley and Dorchester Counties, have also developed independent enrichment activities surrounding the overall CAN DO effort. Organizations such as The Young Astronauts and The Trident Amateur Radio Society also have programs that complement the actual space flight.

For a project to achieve a broad base of support, the efforts of many talented and dedicated people are required. But no matter how large the team, it is always a few key "spark plugs" that make it happen. A proud record of special scientific activities has been achieved by this area. The greatest part of this record is the history of two individuals, a classroom teacher and a college professor who have tirelessly worked to enrich the educational process. The complete list of their efforts on behalf of area science students would take far too much room, but in the last seven years would include:

- Development of airplane field trips to study coastal ecology.
- Development of a student ham radio net of licensed operators.
- Participation in the development of NASA's telelecture program.
- Establishment of a regular program of astronomical sky parties.
- Student construction of a six-inch telescope including grinding the mirror.
- Construction and licensing of a student-run FM radio station.
- Construction of a portable planetarium.
- Tours of NASA moon rocks and space vans with educational training.
- Establishment of a Community Involvement Program with NASA
- Development and certification of the first (below college graduate level) program for saving stranded whales.
- Design and construction of a unique outdoor classroom surrounding two local ponds.
- Construction of a Get Away Special to photograph Comet Halley.

PRIMARY STUDENT INVOLVEMENT

Primary student involvement for the purposes of this paper, can be defined as student activities which are totally dependent on the GAS canister itself. These include design, student experiments, and evaluation of the data and photographs returned.

Junior Design Team

The original intent was for the experimental package to be largely student designed with the help and advice of expert adults. In practice, as the technical complexity needed to achieve mission goals became apparent, the actual student involvement changed. A junior design team was maintained to apprentice and observe the activities of the senior team, but their input never reached the levels hoped for. A few individual members made substantial contribution in the areas of computer programming and film test evaluation, but the bulk of the engineering was done by the senior team. In retrospect, it was probably unrealistic to expect sixth graders to have the ability or desire to tackle the technical minutia of approved alloys and low power Mil spec computer chips. It is highly questionable if such specialized knowledge would be particularly relevant to appropriate educational goals for this age group. Rather than compromise the ambitious goals of the mission, we decided to provide a powerful package designed on their behalf, and place emphasis on other areas such as data evaluation.

Student Experiments

A competition was held in all 14 county middle schools for student designed independent experiments. The criteria for selection were that the experiment must be passive in nature, light in weight and moderate in size. The experiments were then selected on the basis of scientific value and were to be flown in order of the least weight first. Eleven projects were chosen, representing 28 students at four schools. Most projects involved the effects of microgravity and the space environment on material including human blood cells, local seeds, mold spores, penicillin, a magnet, and even a Timex watch. Other experiments included passive dosimeter measurements, and a device to measure maximum acceleration experienced in flight.

Where appropriate, identical packets of control material will be prepared. These will be stored until flight time, and then exposed to a combination of cold, low radiation, and high radiation depending on the particular experiment. The students will then evaluate the flight and control material and tabulate the results, before knowing which packet actually flew. All of the student experiments will be housed in a single styrofoam block which will be cut to securely hold each experiment. This block will then be sealed and mounted to the upper battery plate in the canister.

These experiments are more appropriate to the age group and can be completely designed and evaluated by the students themselves with minimal supervision. The details and results will hopefully serve as the subject of a paper at next year's symposium.

Student Evaluation of Results

The most intensive involvement for the greatest number of students will begin after the actual flight. If all four cameras perform as planned, some 7000 photos will be taken of which up to 1400 may include Halley's Comet. The time of each comet picture must be identified and changes in the structure of the comet will be noted. Brightness values will be estimated by comparison to background stars and the size of the visible tail will be plotted as a function of photographic exposure time. This work will be performed in cooperation with and under the supervision of astronomers who will prepare packets of photos with appropriate background information.

Classes and clubs who participated in ground based photography will be supplied with matching space photos to compare. If proper calibration can be achieved, this information, for both space and ground-based efforts, will be supplied to the International Halley Watch.

The 5500 "misses" may well include other astronomical objects of considerable interest. Our experiments indicate that the cameras and lenses are capable of capturing stars down to magnitude 10-11 and diffuse nebulae such as the Rosette. A large project will be the identification and evaluation of these other photos.

We assume that some percentage of the time exposures will be spoiled by excess drift rates in the Orbiter. These photos will provide an accurate measure of these rates by the length and axis of the streaked star images. The constant 30 sec. firing of the Feathercam will allow the reconstruction of a drift rate history for the mission. This profile will be of value to future astrophotography missions.

SECONDARY STUDENT ACTIVITIES

Secondary activities are independent and self sufficient, but inspired or enhanced by CAN DO. They have the great advantage in that they are unaffected by contractual obligation or administrative boundary line. Any number can play, as long as they have interest and enthusiasm enough. These activities will be divided into the areas of historical perspective, technology, astronomy, and communications.

Historical Perspective

An active program of historical research has developed which includes library research on past apparitions. In addition, students have solicited letters and personal interviews from the elderly who remember Halley's 1910 visit. The interviews are on a standard form which asks questions about the impact of the Comet as reflected in what was said in their church and schools. This interaction between generations teaches far more than astronomy. It can teach a sense of history and a perspective on the changes in society.

This new perspective can then be exploited in turn to impress the students with the value of records that they leave for those that come after them. Several projects are under way to compile and archive the views and impressions that this generation receives.

Not to be left out, several English teachers are preparing to take advantage of the interest generated. Besides the library science entailed in the historical research, they plan to introduce books about space and science fiction into their reading programs.

Technology

Many local science teachers are integrating CAN DO into their teaching plans. This will be detailed in a sample curriculum given below. In addition, several extracurricular science activities are also involved. The NASA-supported Young Astronauts also have their own active programs of scientific activities, many of which complement the overall program. The Young Astronauts played an especially active role in the recent Spoleto Children's Festival which had space as a theme. This highly successful activity included scientific and education exhibits, space art competition and was capped by a live rocket firing.

Astronomy

Area schools have a well established astronomy program going back several years. Activities have included numerous sky parties and the actual construction of a telescope. It is a natural progression to continue these activities with Halley's comet and its related meteor showers as a subject. During the apparition period, regular visual and photographic observations will be made and compiled. During the flight period, the emphasis will be on concurrent photography with closely matched cameras and film. Active recruitment is underway to establish observation groups at as many locations as possible. Contacts have been made with groups in South American, who will have a much better viewing angle. Results will be compared with the space photos to evaluate the gains made. It is hoped that the results will be suitable for inclusion in the data being gathered by the International Halley Watch.

Communications

Another well established local program is the ham radio net. With students already trained and equipment in place, they are ready to render valuable communication service. During the flight of STS-61E, they will attempt to establish around-the-clock radio monitoring of Shuttle activities. In addition, they will serve as a vital communication link with the far flung astronomical observation teams. This is in addition to the service they have already done in the solicitation of long distance interviews with 1910 Halley veterans.

OPERATION CAN DO
LOW COUNTRY STUDENTS IN SPACE
STUDENT EXPERIMENT RECORD

Student name John Doe School C.E. Williams
 Sponsor Mrs. Smith Grade 7
 Student Address 01 Oak Drive Phone 000-0000
Charleston
 School Address 02 Oak Drive Phone 000-0000
Charleston
 Phone where you can be reached during summer 000-0000
 Name of experiment The Effect of Space on S.C. Seeds
 Nature of material Seeds from South Carolina
native plants such as wild flowers.
 Weight of 1 sample 1/2 oz
 Goal of experiment To see if microgravity or cold
or radiation in space effects the germination
of seeds.
 Any hazards? NO
 Special handling required? Controls to be exposed to
cold + radiation.
 (attach pages if necessary)
 Submit this form with two samples of non-biological and four samples of biological material. The extra samples will be used as blind controls. You will be informed on which sample actually flew in space after all experiments are concluded. Use separate forms for each experiment.

①


**HALLEY'S COMET
QUESTIONNAIRE**

BACKGROUND DATA

LAST NAME DOE FIRST NAME John MIDDLE NAME M.
 DATE OF BIRTH July 23, 1900 PLACE OF BIRTH Charleston RESIDENCE IN 1911 Charleston
 FATHER'S OCCUPATION Farmer FATHER'S EDUCATION 6th Grade
 IN WHICH OF THE FOLLOWING SETTINGS WERE YOU LIVING WHEN YOU VIEWED THE COMET? URBAN _____ RURAL
 YOUR PRESENT RELIGION Baptist
 YOUR RELIGION AT THE TIME YOU SAW THE COMET Baptist

COMET QUESTIONS

1. DO YOU REMEMBER OBSERVING HALLEY'S COMET? YES NO _____
 2. HOW DID YOU VIEW THE COMET? Through the door.

3. HOW MANY OBSERVATIONS DID YOU MAKE?
 A. ONE OR TWO
 B. SEVERAL
 C. MANY
 D. NONE

4. HOW VISIBLE WAS THE COMET WHEN YOU OBSERVED IT?
 A. BEARLY VISIBLE
 B. DIM
 C. BRIGHT
 D. VERY BRIGHT

5. WHICH OF THE FOLLOWING PARTS OF THE COMET WERE OBSERVED?
 A. HEAD
 B. TAIL
 C. ~~NECK~~
 D. ~~WINGS~~
 E. OTHER _____

6. CAN YOU REMEMBER ANY MYTHS ABOUT THE COMET MENTIONED AT THE TIME OF YOUR OBSERVATION?
 A. FEAR OF COLLISION WITH EARTH

③

B. FEAR OF POISON GASES IN THE TAIL
 C. TALES OF RELIGIOUS EXPERIENCES
 D. OTHER _____

7. IF YOU WERE IN SCHOOL, HOW WAS THE EVENT TREATED?
 A. IGNORED
 B. BRIEFLY DISCUSSED
 C. EXTENSIVELY DISCUSSED
 D. NOT IN SCHOOL

8. HOW WAS THE EVENT TREATED IN CHURCH?
 A. IGNORED
 B. BRIEFLY DISCUSSED
 C. USED AS PART OF A SERMON
 D. OTHER _____

9. BRIEFLY DESCRIBE YOUR OBSERVATION I remember,
it looked like a long stringy cloud.
A friend of mine had a dress that
was named after the comet because
it had a long tail.

④

Fig. 1: Example of "mini-PAR" used by student experimenters.
 Fig. 2-4: Example of one of the questionnaires used to interview the elderly. This questionnaire uses mostly short answers and is especially designed for younger grades and ham radio use.

CURRICULUM ACTIVITIES

The success of the overall project in reaching the average classroom student depends on how well it is integrated into the regular curriculum of the courses that these students normally take. Below is an example of the special activities planned by an eighth grade Earth Science teacher for the 1985/86 school year. Many of these are at least partially inspired by the Comet Halley activities while others are totally unrelated. The emphasis will be different for teachers of seventh grade Life Science and sixth grade General Science.

1985-1986 Earth Science Schedule

<u>Planned Date</u>	<u>Activity</u>
9/1	*Teams established and procedures defined for stranded whales rescue. *Comet Halley historical research begins with library research and distribution of questionnaires.
9/15	*Ham radio classes begin (2 nights per week) for students who wish to obtain novice license.
10/1	*Develop filing system for Comet Halley material gathered.
10/20	*Sky Party #1 to observe Comet Halley, Orionid Meteor shower and Orion nebulae. Also there will be computer and radio demonstrations.
11/1	*Field Trip to Planetarium.
11/15	*Select weather forecasting team and begin daily in-house forecast.
12/1	*Novice radio class graduates, radios are set up and first contacts are established.
12/15	*Use radio/computer equipment to capture satellite news programs and analyze them for frequency and propagation characteristics. Great circle routes are computed for stations.
12/20	*Sky Party #2 to continue observations of Halley and other astronomical targets. Radio and computer demonstrations included.
1/5	*Explore local area for likely fossil dig sites.
1/15	*Ham radio classes begin for those with novice license who wish to obtain a general license.
1/15	*Begin the regular photography and visual tracking of Halley and establish procedures for the correlation of comet data.
2/1	*Field trip to University library to do research on Comet Halley, prehistoric animals, coastal land formation, electromagnetic wave propagation and whales.
2/15	*First field trip to fossil dig site and the start of operations.
3/6	*CAN DO launch to be observed on television or in person.
3/6-3/13	<u>CAN DO IN ORBIT</u> *Monitor STS-61E by radio with assistance of ham radio net. *Use triangulation to compute height and speed data for shuttle and comet. *Photograph comet and assist in coordination of concurrent comet photography by other schools in Northern and Southern Hemispheres. *Conduct various radio exercises to establish educational ham net.
3/13	*CAN DO landing to be observed either on television or in person. *Begin analysis of data gathered during flight.
4/15	*Guests speakers to discuss local area geography conservation. Start gathering data on the impact of local industries.
5/1	*Students fly in chartered aircraft to observe local sea islands and to observe areas where industry and population are impacting on the natural order.
5/15	*Final compilation and archiving of Comet Halley historical and observational information.

(This particular schedule represents the plans of an exceptionally active teacher. Other teachers will adapt the activities to fit their own teaching goals.)

FUTURE PLANS

Looking down the road to a time when Halley's Comet is Pluto-bound and its pictures have all been sorted and digested, what next? We hope to keep the design team intact in anticipation of the arrival of the next crop of students. What's left after space flight? We would like to think it's more space flight.

One idea is "CAN DO II," a new flight with a new mission. Building on the lessons learned, we feel we could redesign and reprogram a canister to tackle deep space, solar, lunar or terrestrial targets. If the TV targeting system is successful, it can be easily adapted to a wide variety of mission profiles.

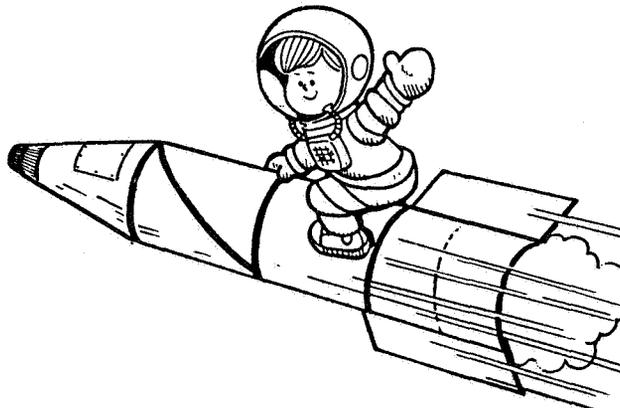
Even more ambitious plans are in the preliminary discussion stage. The NASA Hitch-hiker program offers two important assets--power and communications. A project could be developed to fly a powerful battery of cameras and optics, possibly including a telescope. Various cameras would be equipped with appropriate filters and films for terrestrial, solar, and deep space subjects. Digital TV targeting images would be transmitted back to Earth and forwarded to schools anywhere in the world. Each school would be responsible for developing their own research efforts, which could range from hurricane tracking to solar flare studies. They would have to develop communication and computer imaging facilities appropriate to their mission. When their target imaging indicated that they were in range, a fire signal would be relayed to the appropriate camera. Ambitious? Yes! But no insurmountable obstacles seem in the way except for a lot of hard work and some funding support. With hundreds of schools involved, the load would be made bearable. Without ambitious dreams, nothing worthwhile would ever be accomplished.

CONCLUSION

^{THE} Our experience has shown that a GAS experiment can be a valuable education tool. It can return results far in excess of the resources invested. ^{THE} Our best estimate on the financial investment per student indicates that it is somewhat less than the cost of a school lunch. That's a bargain in a time when educational bargains are hard to come by.

To reach this goal means reaching far beyond the students who could possibly design or fly experiments in a single canister. The greatest value of CAN DO is that it serves as a catalyst and inspiration for other activities. To not reach out would have turned it into an overblown, expensive "science fair project" for a few exceptional students.

To fully exploit the benefit of a GAS canister, you should build on a well established science enrichment program. As part of a comprehensive plan, a Get Away Special can be one of the most motivating educational tools available. It could touch a child's life and give him a special memory to carry with him always. It might just teach him that even the road to the stars is open to those that know how to dream. How much is a lesson like that worth?



N86-27300

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8956

SPACE MANUFACTURING UTILIZING THE
DIRECTIONAL ELECTROSTATIC ACCRETION PROCESS

Presented by .

Mr. Alan Mortensen

DM593208

The Ohio State University
Department of Aeronautical and Astronautical Engineering

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ELECTROSTATICS
DEPOSITION
SPACE MANUFACTURING
SPACECRAFT MAINTENANCE
COST EFFECTIVENESS

PORTABLE EQUIPMENT
EPITAXIATIONS
LIQUEFACTION
DROPS (LIQUIDS)
SURFACE FINISHING

ABSTRACT

The Directional Electrostatic Accretion Process (DEAP) is described with respect to both the physical process and its application to manufacturing in space. This high precision portable manufacturing method will revolutionize current practices in manufacturing and repair of spacecraft and space structures. The cost effectiveness of this process will be invaluable to future space manufacturing projects.

1.0 Introduction

Through scientific space exploration and development, newly developed technologies have generated considerable assets which are currently benefiting mankind in many ways. For example, minicomputer and composite material technologies have spun-off of earlier space missions into multi-billion dollar industries benefiting the entire humankind. Furthermore, these same technologies continue to permit further maturation of the space environment for human use.

Currently, NASA is planning a permanent presence in space through the development of the space station. In order to be cost effective on the economic returns from such development, scientists and engineers must develop resourceful techniques of space utilization. One way to achieve this end is to maximize productivity per each trip into space. This can be achieved by performing production and repair in space instead of on the earth. Thus, new manufacturing and repair techniques are being developed to achieve this end. The DEAP method is one such manufacturing and repair technique (ref. Fig. 1).

2.0 The DEAP Method

Scientists and engineers have been investigating possible space manufacturing techniques for the past two decades. Most construction techniques investigated to date are for large scale (from feet to miles in length) space structures with only minor attention given to smaller scale capabilities. The DEAP unit will greatly facilitate filling needs of small scale manufacturing and repair capabilities.

DEAP requires five major material processing steps:

- a. bulk material liquefaction
- b. droplet formation and charging
- c. droplet directional guidance

- d. target surface accretion
- e. three-dimensional build-up

Each of these steps will be described in the following text.

2.1 Bulk Material Liquefaction

Producing or repairing parts and structures through the DEAP method requires that a reasonably pure material be utilized. That is, microstructural inclusions which degrade the purity of the material must be minimized to preclude clogging of the DEAP unit. Liquefaction of bulk material by heating is performed to produce a molten flow of material which is then developed to the small orifice device via a mechanical displacement pump.

2.2 Droplet Formation and Charging

The small orifice device collects the initial supply of molten liquid in a chamber which supplies a convergent channel. Through control of this small orifice channel molten material droplets are formed. Utilizing a charging technique, a small surface charge is imparted to each droplet.

2.3 Droplet Directional Guidance

Since each droplet carries a small charge, electrostatic fields are utilized to impart accelerations to each droplet, thereby permitting directional control of the molten droplets. Through directional control, a variety of geometric shapes may be produced with considerable dimensional precision in the unfinished condition. Thus, finishing work is largely unnecessary and can be avoided.

2.4 Target Surface Accretion

Droplets are deposited on the target surface in a precise manner. As the droplets are deposited, two important phenomena must occur; surface wetting and solidification. Surface wetting must occur or the material droplets will simply rebound or be displaced from the target by newly arriving droplets. Solidification of the wetted droplets must be rapid upon contact with the target or splattering and loss of dimensional tolerance is likely to result.

2.5 Three-Dimensional Build-Up

To date, the accretion layering process is largely untried and will be an important factor in the success of this manufacturing technique. Through layering of desired two-dimensional geometries, three-dimensional build-up can be achieved to produce a completed part or to repair a damaged structure.

3.0 Cost Effectiveness

Although there are many ways to produce high precision parts in space, cost effective methods must be developed to preclude excessive expenses which diminish returns from investments in space development. The DEAP method is one such potential cost effective method due to the following cost effective criteria:

- a. material bulk packaging
- b. high precision finish tolerance
- c. facility mobility

3.1 Material Bulk Packaging

Due to packaging constraints, boosting cumbersome payloads into orbit is inefficient with respect to spatial arrangements in payload bays. However, if structures were manufactured and assembled in orbit, then bulk material need only be supplied to low earth orbit for use in processing. This approach would greatly ease packaging requirements and increase the efficiency of delivering needed materials to orbit. The DEAP method utilizes bulk material for manufacturing and repair.

3.2 High Precision Finish Tolerance

The DEAP method inherently produces a high quality finish, thereby minimizing, if not completely precluding, the need for finishing work. Thus, additional man-hours, machine-hours and materials will largely be deleted from the required production schedule. Also, repairs on present structures and spacecraft could be achieved without having large stores of spare parts on hand. Parts and repairs could be made on location in orbit resulting in largely improved efficiency of mobility and utilization of available resources and time. As a general result, further increases in cost effectiveness yield improved returns from investment in space.

3.3 Facility Mobility

Since the DEAP method can vary in size from small to large units, manufacturing facilities could be placed in stationary orbits or transported about to fulfill manufacturing and repair needs as they become determined. In fact, DEAP units as small as current NASA manned-manuevering units could be designed for mobile repair services on spacecraft and space structures.

4.0 Experimentation

The DEAP unit is currently being developed by engineering researchers and students at The Ohio State University (OSU) under research funds supported by AIAA, OSU and donations from the commercial sector. This experiment will be flown as a NASA Get Away Special payload on board a future shuttle mission.

5.0 Results

Experimental findings will be published after completion of the experiment.

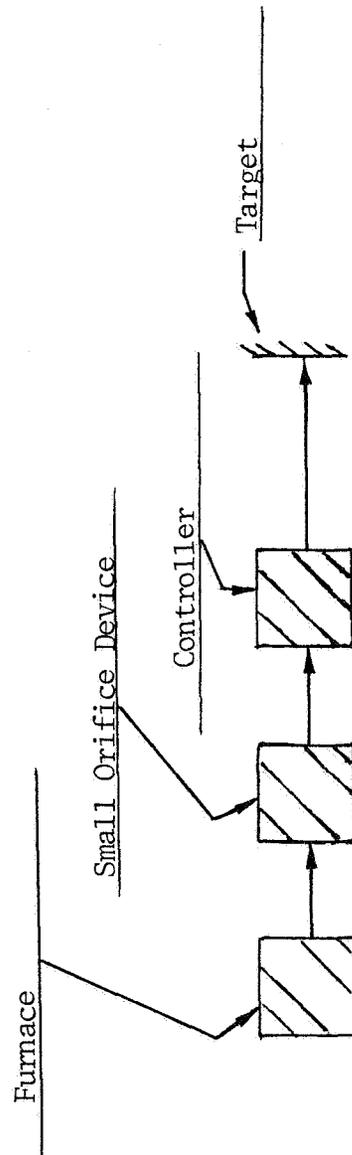


Figure 1. The Directional Electrostatic Accretion Process Unit

DATA LINKS
COMMUN. SATELLITE
RADIO FREQUENCIES
FREQU. ASSIGNMENT
ANTENNA DESIGN

CONTINUOUS RADIATION
SIGNAL TO NOISE RATIO
FREQ. MODULATION
FREQ. SHIFT KEYING
DOPPLER EFFECT
CODES
DECODER

RADIO RECEIVER DY-32
TRANSMITTER 8P

8957
N86-27301

THE NORTHERN UTAH SATELLITE (NUSAT) COMMUNICATIONS LINK

LEE BARRETT - COMPUTER SCIENCES CORPORATION (CSC)

CZ 785393

INTRODUCTION

During the planning stages of the NUSAT satellite, an obvious issue to be discussed was the method of communications to be used. The frequencies would have to be high enough to pass through the atmosphere relatively unattenuated but low enough that antennas and transmission lines would not be so critical in length and properties that unexperienced students would have difficulty with handling them. In conversations with the Federal Communications Commission (FCC) in Washington D.C., the frequencies of 450.000 MHz and 137.900 MHz were decided upon and applied for licensing. The 450.000 MHz up-link is used on a "non-interference" basis to control the satellite while the 137.900 MHz frequency down-links data from the satellite.

A reasonable amount of time was spent discussing the mode of transmission to be used in communication with the satellite. Representatives of the amateur radio satellite organization, AMSAT, were contacted for ideas. This organization seems to favor AM types of emissions such as CW to control their OSCAR series of satellites and also the current Phase III unit. NUSAT personnel felt, however, that there would be merit in the improved signal to noise ratio usually obtained in an FM mode. Doppler shift of the transmitted information on the NUSAT also had to be considered. The final decision was to use Audio Frequency Shift Keying (AFSK) modulated on an FM carrier. In this mode the audio tones used would not shift frequency with Doppler - only the carrier would shift in frequency. If necessary, the receivers could be fitted with a form of AFC control or simply have enough bandwidth to handle the Doppler shift with an acceptable increase in noise.

DATA ENCODER/DECODER

Initially a data rate of 2400 baud was planned. This high speed of data transmission created no small design problem for the communications group. In the worst case where every alternate bit is high and then low in the data stream, the fundamental frequency of the data rate will be half the baud rate or 1200 Hz. The length of a bit is then:

$$t=1/f$$

where:

t= time in seconds
f=frequency in Hertz

therefore:

$$t=1/1200 \text{ Hz} = 833 \mu\text{s}$$

It was felt that at least three cycles of the lowest AFSK tone would be needed to decode a data bit correctly. Applying the above equation in reverse:

$$f=1/t=1/(833 \mu\text{s}/3)= 3600 \text{ Hz}$$

A stable frequency source was needed to produce the AFSK tones. Both mechanical and temperature stability had to be considered. The first attempt was to use a color-TV 3.579545 MHz crystal in an oscillator circuit and divide the frequency by 600 and 400 to develop the audio frequencies. In this way, any drift of the oscillator would also be so divided. The frequencies of 5966 Hz and 8949 Hz resulted from the divisions and met the minimum 3600 Hz tone criteria for bit recognition. The lower tone was designated as the Mark (or binary 0) tone. The upper tone was the Space (or binary 1) tone.

The first encoder circuit simply gated the outputs of the two divider chains on or off with the respective logical input. The initial decoder used two "Biquad" active filters - one tuned to each of the respective tones. The filter outputs were rectified and applied to a comparator to regenerate the logical signal.

Severe aliasing problems were encountered with this encoding/decoding scheme. Several decoders were tested with minimal improvement in error output - including dedicated LC circuits. The best results were achieved with a Phased Locked Loop (PLL). The next step was to examine the encoding technique.

If a coherent method of detection was to be used, then it made sense to pay attention to the phase of the encoder signals. The encoder was modified to allow a change in logical input to change the AFSK tone only at a zero crossing of the tone. This seemingly simple change improved the error rate in decoding dramatically. It was further discovered that if even multiples of half the baud rate (the baud fundamental frequency) were used as AFSK tone frequencies, the error rate was further reduced an insignificant level.

The final encoder was constructed to produce phase coherent keying at tone zero crossings. Following the earlier argument about having enough cycles to decode for bit recognition, the AFSK frequencies were chosen to be 7200 Hz and 9600 Hz - both multiples of the 1200 Hz fundamental baud frequency for a data rate of 2400 baud. It was found that the varactor modulator of the FM transmitter to be used would not modulate 9600 Hz properly due to circuit capacitances. Rather than change the oscillator circuit and risk frequency instability in the transmitter, the tone frequencies were reduced to 4800 Hz and 7200 Hz. A 2.88 MHz crystal was used in a high speed C-MOS oscillator circuit and then divided to provide the desired frequencies. The remainder of the circuit was also designed using C-MOS gates to keep power consumption low. The final encoder circuit is illustrated in Figure 1.

The decoder used was based on the LM565 PLL. Audio from the receiver was sampled directly after the ratio detector and fed to the decoder circuit board. The original low pass filter in the receiver was disabled and replaced on the new circuit board with an RC filter with a sufficiently high cutoff frequency to allow the 7200 Hz AFSK tone to pass unattenuated.

Following the RC filter, a single bipolar transistor, with a gain of roughly 100, was used to hard limit the audio input against the positive voltage supply and ground to remove AM type noise. The limiter then drives a "Biquad" active filter with a Q of about 5 centered on 5879 Hz - the geometric mean frequency of the two AFSK tones. This active filter band limits the input signals to the region of interest.

The filter output was next connected to the LM565 PLL input. The free running frequency of the Voltage Controlled Oscillator (VCO) in the PLL was also adjusted to the 5879 Hz geometric mean frequency. A silvered mica capacitor was used to set the VCO frequency for temperature stability.

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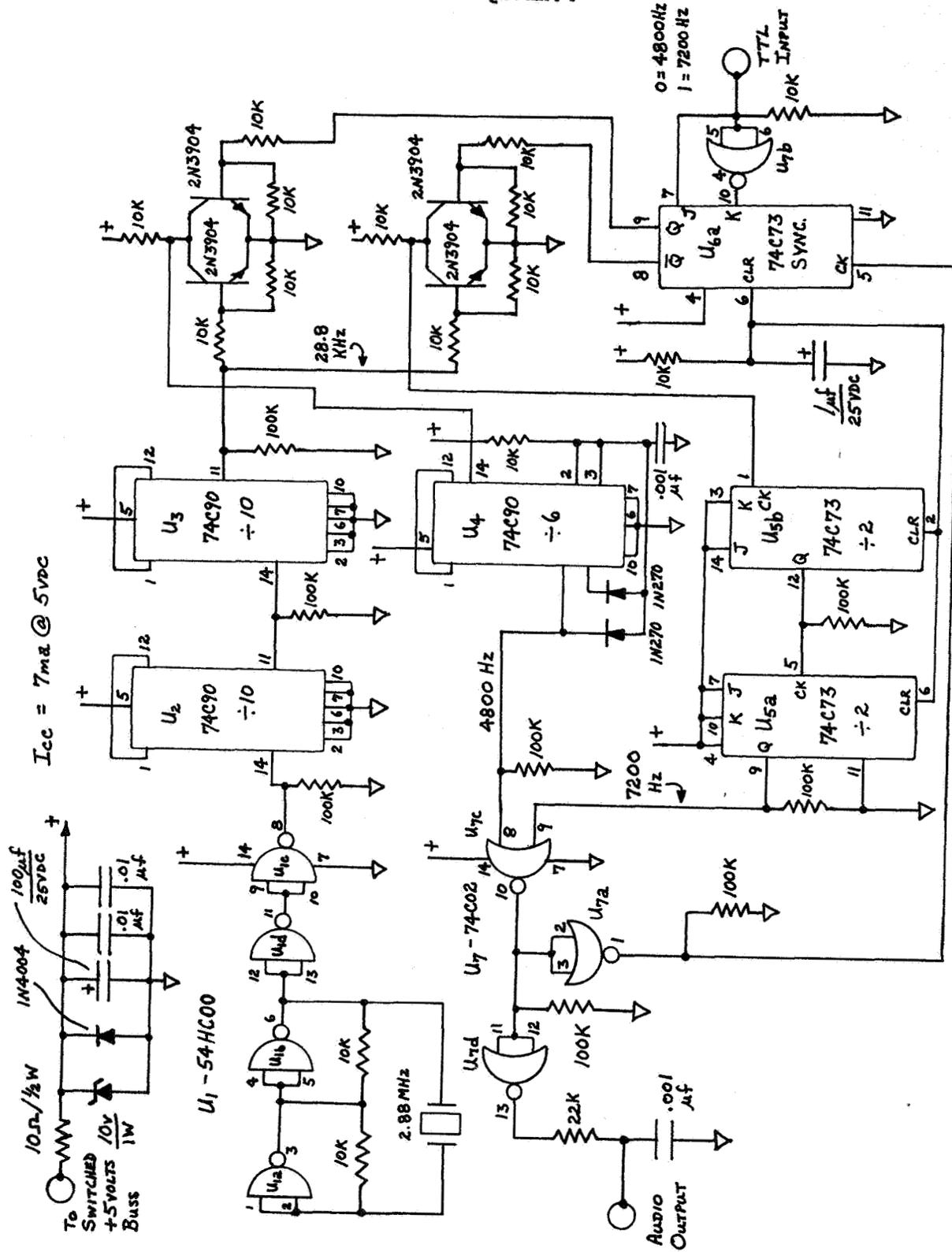


Figure 1. NUSAT AFSK Encoder

Baud rate is also a consideration in the low pass filter network of the PLL. The cutoff frequency of the filter had to be higher than 1200 Hz - the fundamental frequency of the 2400 baud rate. Following the filter, a high gain comparator was used. The comparator made use of an operational amplifier operating at open loop gain. A capacitor in the feedback loop was used to slow the slew rate of the comparator to prevent oscillations with inputs close to the switching point.

The comparator output was applied to a buffer transistor which interfaced with the computer. To prevent open squelch noise from driving the computer input randomly, a second transistor was connected to incorporate the receiver's built in noise squelch with the data circuit. When the receiver squelch is closed, the data line to the computer is held low (at a logical 0). The data line only follows the comparator output when the receiver squelch is open.

The final circuit for the decoder is shown in Figure 2. Tests on the encoder/decoder circuits demonstrated that a data rate as high as 4800 baud produced very few errors. At 2400 baud the circuits were tested extensively with no apparent errors occurring. The only abnormality worth mentioning has to do with "jitter". Jitter in the output wave form was observed because the AFSK tones derived from the crystal oscillator in the encoder were not exact multiples of the baud rate fundamental frequency. As a result, a beat note between the AFSK frequencies and the fundamental baud rate frequency occurs causing a cycle of the AFSK tone to "slip" between the data bits at the same rate as the beat note. The output data bits are then pulswidth modulated at the beat note rate and one AFSK tone cycle in width. However, since the computer samples the data bit at the center of the bit frame, this pulswidth modulation or jitter was not significant. One way to avoid jitter in future projects would be to use the computer data clock to generate the AFSK tones.

All data levels on the satellite are TTL while those in the ground station are RS-232.

SELECTING A TRANSMITTER AND RECEIVER

The next task was to select a transmitter and receiver to handle the selected data rate of 2400 baud. The main concern with the selection was the required bandwidth to handle such a data rate. Several sources indicate that in the transmission of Pulse Code Modulation that a deviation of (.7 X data rate) yields an acceptable signal to noise ratio on an FM carrier. If this criteria can be applied to the AFSK data to be applied to the FM carrier, then a deviation of only 1680 Hz is needed. It was decided to use twice the required deviation or 3.4 kHz. Using Carson's rule for FM bandwidth, the following relationship developed:

$$BW=2(F_m+\Delta F)$$

Where: F_m = the highest modulation frequency
 ΔF = the instantaneous carrier deviation

therefore: $BW=2(7200 \text{ Hz}+3400 \text{ Hz})=21.2 \text{ kHz}$

Several narrowband transmitter and receivers are commercially available with the capability of this modulation index and bandwidth.

The ground transmitter and receiver along with the satellite transmitter were all purchased from Spectrum Communications Corporation, Norristown, PA. Because of size constraints in the satellite, a smaller receiver had to be purchased. A handheld transceiver was purchased from ICOM Corporation, Bellevue, WA, for this purpose.

The Spectrum Communications transmitter, model SCT110, was ordered for 137.9 MHz. The circuitboard was physically too big to fit in the satellite at first. The transmitter was carefully cut in half and the two halves were stacked to allow for mounting. The speech clipper and microphone preamplifier were eliminated and the AFSK encoder was wired to drive the transmitter directly through a level setting resistor network. In final configuration, the spacecraft transmitter had an output of 10 watts using the onboard 10 volt power buss. Only minor tuning was required to achieve this output.

The ICOM transceiver, model IC-4AT, was purchased for the receiver onboard the satellite. The transmitter section was completely disabled and many of the components were removed. The IC-4AT is synthesized in frequency by means of thumbwheel switches. The switches were disabled and removed while the synthesizer was hard wired for the 450 MHz frequency. As mentioned previously, a data squelch transistor was added to the original receiver noise squelch to prevent random receiver noise from driving the computer when no carrier signal is received. The receiver had a final sensitivity of $.35 \mu\text{V}$ for 20dB quieting of the audio output. The squelch threshold was $.3 \mu\text{V}$. The receiver was tested for bandwidth before distortion. At a $10 \mu\text{V}$ input level, the receiver will work to $+5.7 \text{ kHz}$ and -6.4 kHz . The receiver can, therefore, accept a reasonable amount of Doppler shift and still function.

All electrolytic capacitors on the spacecraft radios were replaced with tantalum capacitors to prevent possible damage while exposed to the vacuum of space.

The station on the ground employs a model SCR200, 137.9 MHz receiver and a model SCT410, 450.0 MHz transmitter - both manufactured by Spectrum Communications. The two units are mounted in individually shielded boxes to prevent broadband noise interference from the transmitter into the receiver during duplex operation. Each have separate power supplies. The transmitter speech clipper and microphone preamplifier were also removed in this application and the AFSK unit was again wired directly in. A 100 watt output, in line power amplifier was added to boost the transmitter's 7 watts to a usable level on 450 MHz.

The ground receiver is fully metered for deviation level, signal strength, and center frequency. The sensitivity of the receiver is $.3 \mu\text{V}$ for 20dB audio quieting.

There was an original concern about whether or not there would be sufficient signal at the 10 watt power level of the spacecraft to establish reliable communications. Independent studies by NUSAT personnel using both the standard space loss and the radar equation methods of signal strength predictions confirmed that a good signal should exist. These calculations were further confirmed by one of the shuttle missions in which an amateur radio operator communicated easily with the ground with similar power and antenna. Communications with the NUSAT subsequent to its launch have also confirmed good signal levels exist.

ANTENNA CONSIDERATIONS

The final pieces of the communications puzzle were the antenna designs. Because the antenna orientations would not be known because of the unpredictable attitude of the satellite, the decision was made to use circular polarization on the satellite.

The first plan was to use a common pair of vertical antennas for both frequencies. The verticals would use the satellite chassis as a groundplane. A microstripline filter and matching network was designed and built but there was inadequate room for the unit in the satellite. To solve the problem, a second set of vertical antennas were added to the satellite and the matching was done using coaxial transformers and phasing sections.

Each vertical is mounted on a triangular panel of the satellite. All of the antennas are at right angles to each other. As a result, there should be no mutual coupling between them because they are in orthogonal planes. Two of the antennas are tuned to the 450.0 MHz frequency and the other two are tuned to the 137.9 MHz frequency. Because each pair of antennas are space separated by 90 degrees already, it is only necessary to shift the phase of the feed point signal another 90 degrees to produce circular polarization. Essentially the array is half of a turnstile antenna. In this configuration, a ground station should see at least one linear polarization or circular depending on the spacecraft orientation.

The antennas were tuned individually to the proper length and match. A phasing and matching harness was built for each antenna pair from coax. Figure 3 shows the method used. The quarterwave 72 Ohm sections transform the 50 Ohm impedances to about 102 Ohms. These impedances are in parallel at the tee connector producing a 50 Ohm match. The additional quarterwave length of 50 Ohm cable from one antenna is only to delay the phase to that antenna by the required 90 degrees without creating a mismatch. The final satellite antennas had a standing wave of 1.5.

The ground station antennas are cross polarized, multiple element yagi antennas. One for each frequency is mounted on an Elevation/Azimuth rotor assembly to point them. The ground antennas exhibit about a 30 degree beamwidth and are pointed by an Apple IIe computer loaded with orbital data. The ground antenna gains are approximately 10dB.

A block diagram of the entire communications link is illustrated in Figure 4.

CONCLUSION

Since the NUSAT launch the communications link has been tested. The concepts have been successfully demonstrated and some weaknesses have been discovered. The 450 MHz antennas on the satellite seem to be more shaded than originally anticipated because of their short size. As a result, the communications have been somewhat geometry sensitive. Currently it is felt that with additional improvements in the ground station such as lower loss cables to the antennas; additional shielding between the ground transmitter and receiver; and perhaps more ground transmitter power that the link can become more reliable and less geometry dependent. It is also possible that the ground station inadequacies have caused some illegal uploads to the computer on the satellite causing undue battery drain. As time goes on these ideas will be validated.

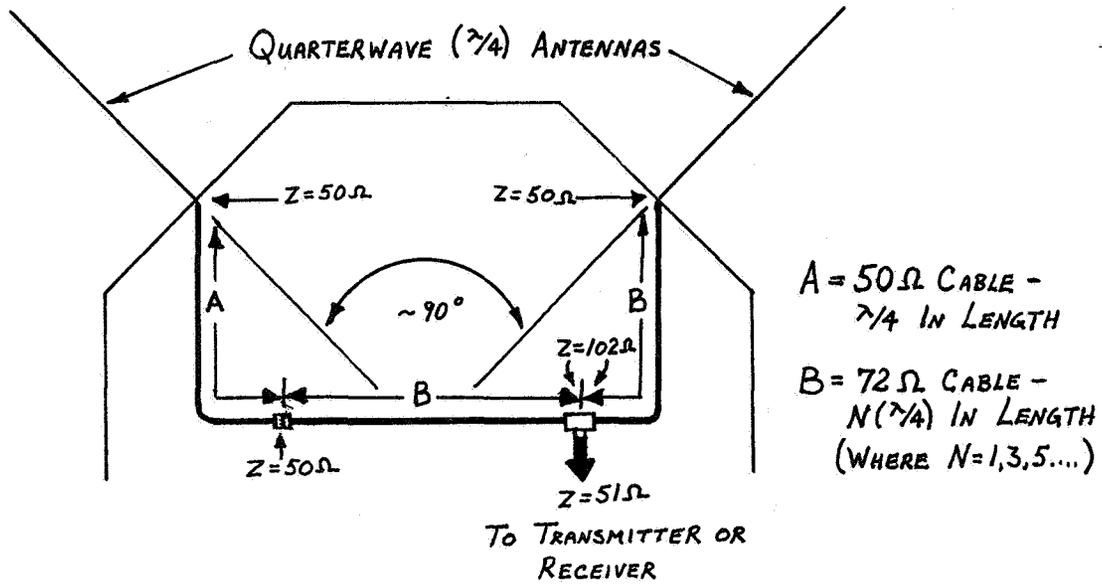


Figure 3. Antenna Matching and Phasing

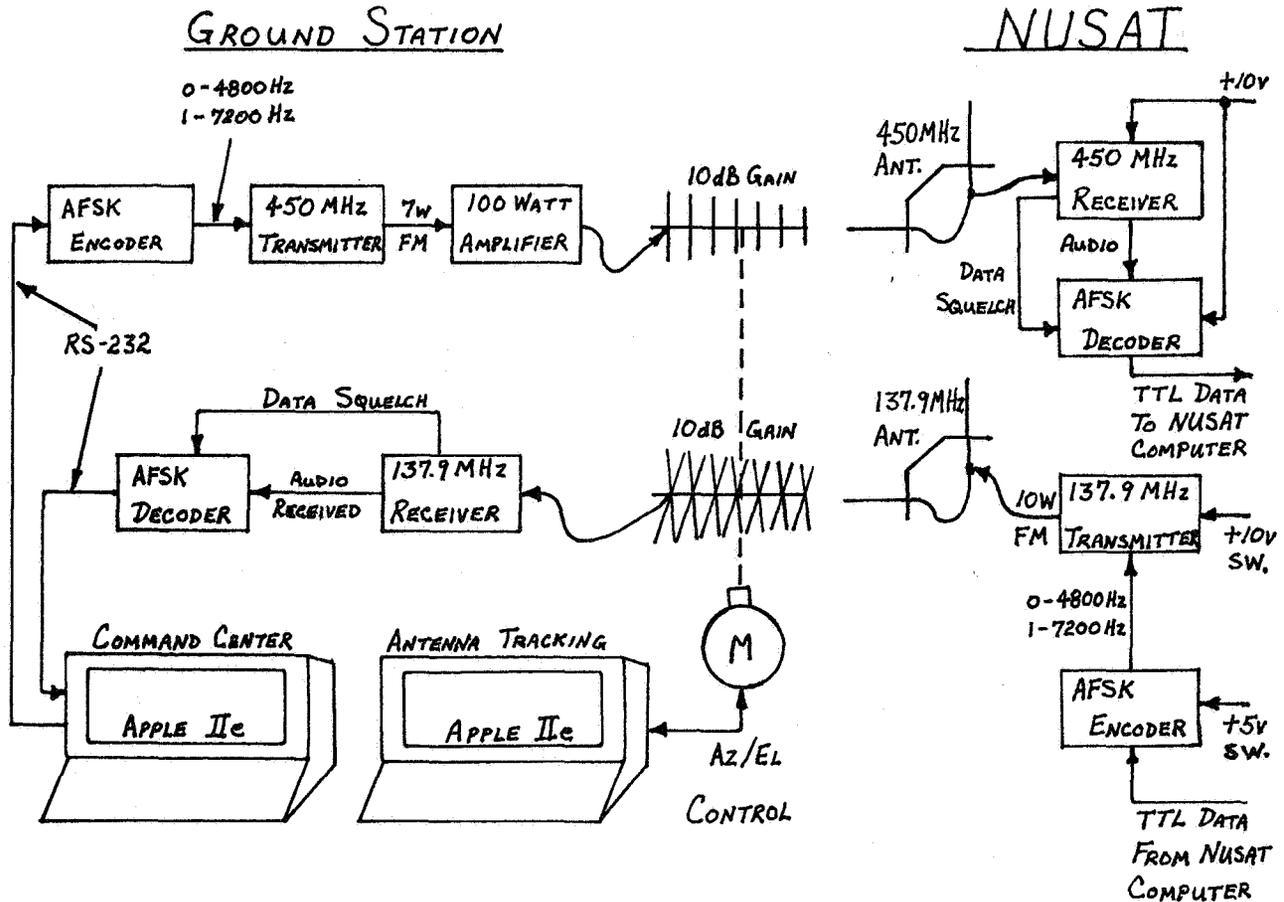


Figure 4. Communications System

REDUCED GRAVITY
FLUID DYNAMICS
WETTABILITY
FLUIDS FLOW
GAS-LIQUID INTERACT.

VAPORS
FILM THICKNESS
INTERFACIAL TENSION
GRAVITATIONAL EFFECTS
PERTURBATION

D5-34
88

8958

N86-27302

FLUID BEHAVIOR IN MICROGRAVITY

William W. Durgin and James W. Matthews

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W6934033

ABSTRACT

An experiment designed to study some fundamental aspects of microgravity fluid dynamics has been built and is scheduled for flight. The purpose of the experiment is to investigate differences in behavior of wetting and non-wetting fluid systems at low Bond numbers. Methods have been developed to determine liquid quantity, estimate vapor contact area and measure liquid layer thickness. Both the responses of the fluid systems to external perturbations and the transfer of liquid through a connection between two containers can be studied.

INTRODUCTION

The objective of the experiment is to study some aspects of microgravity fluid mechanics and quantity determination. In particular, the focus of the work is to develop measurement methods capable of quantitatively measuring the behavior of liquid-gas mixtures in microgravity [1,2]. The systems to be studied are dominated by surface tension effects as compared to gravitational effects and, so, are characterized by very small Bond numbers. Measurement techniques were developed to enable determination of liquid phase volume, estimation of vapor (or gas) phase contact area, and surface layer liquid thickness. In addition, provisions have been made to investigate liquid migration due to surface tension and perturbations such as g-jitter.

Two fundamentally different fluid systems are compared using identical experiment modules. One system utilizes an aluminum container with a FreonTM 11 fluid system. The freon is known to wet the aluminum strongly and will tend to coat interior surfaces somewhat uniformly in search of a minimum energy configuration. The Freon is maintained at saturation conditions so that the liquid and vapor phases can be the same substance. The second system utilizes a water/ethylene glycol liquid mixture with nitrogen "vapor". The container surface is TFE which forms a non-wetting system with the liquid. This system will tend toward a floating mass or masses of liquid not in contact with container walls.

Each experiment module is actually two containers connected by a motor activated valve as shown in Figure 1.

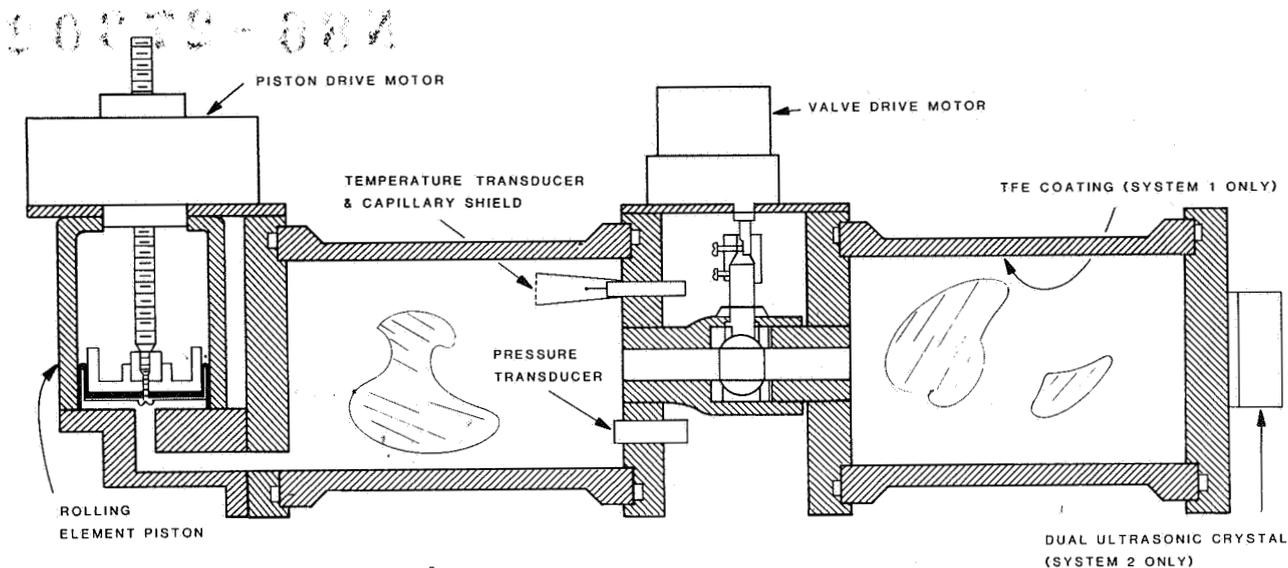


Figure 1 MODULE SCHEMATIC

Initially, most of the liquid will reside in one of the module containers. During the 2 day experiment program, the connecting valves will be periodically opened allowing liquid and vapor exchange between the containers. It is expected that the wetting system will quickly redistribute by capillary action to approximately equal liquid volume in each container. The only driving force for the nonwetting system to redistribute, however, will be the kinetics of the liquid globules driven by orbiter accelerations. The redistribution will thus be a probabilistic process depending on the acceleration spectrum encountered. The liquid present in one of the module containers will be periodically monitored so that the redistribution process can be analyzed after the flight.

METHODOLOGY

By perturbing the container volume a small amount, dV , in a short time, the vapor phase undergoes an essentially isentropic process. It can be shown that the corresponding vapor phase temperature excursion is

$$dT = -T(\gamma - 1)dV/V$$

where γ is the ratio of specific heats. Simultaneously, the container pressure undergoes an excursion

$$dP = -P \gamma dV/V$$

Thus, if the volume perturbation is prescribed while the temperature and/or the pressure response is measured, the volume, V , of the vapor phase can be determined

$$V = -(\gamma - 1) dV / dT$$

$$V = -P \gamma dV / dP$$

The liquid phase volume is thence found by subtraction from the total volume.

Typically a small volume change is desired in order to minimize the displacer mechanism size and power. The perturbation must, however, produce measurable temperature and/or pressure responses. It is, perhaps, easiest to measure the pressure response. In order to achieve reasonable resolution with existing transducer technology, it is necessary to use volume perturbations of the order of 5% of the total volume. While this can be achieved with nearly full tanks, it becomes a problem for nearly empty ones.

Greater resolution can be achieved through temperature measurement although the methodology is more complex. The sensor must be protected from liquid contact because the liquid phase remains fixed in temperature during the perturbation. The isolation can be accomplished with capillary control shields. The sensor must have a very small time constant and the measurement circuitry must be arranged to minimize self-heat error.

After the initial perturbation equilibrium is slowly re-established with heat transfer occurring between the vapor, the liquid, and the container. The process is complicated by mass transfer as well but in global terms, the time constant of the (exponential) return to equilibrium is proportional to the contact area of the vapor. In this sense, the time constant reflects the configuration of the liquid-vapor mixture. A dispersed system, such as many droplets or globules, would return to equilibrium much faster than a coherent film type system.

By measuring the temperature decay curve, the time constant can be established. The sampling methodology has been designed to take advantage of the statistics of acceleration events at selected levels. It is expected that accelerations will cause the liquid configurations to change thus producing measurable differences in the time constants of the decay curves. A companion accelerometer package will provide a time history of acceleration events for correlation with the heat transfer data. The sensitivity of the three-axis accelerometer package is 5×10^{-5} g/bit which is sufficient to characterize thruster firings and crew motions [3].

A relatively straight-forward method of ultrasonic film thickness determination is utilized. A dual-element transducer of 2.25 MHz natural frequency was selected in order to simplify the electronics and provide high resolution. The transducer was fitted against a PMMA window which was coated with epoxy to protect it from the FreonTM 11. In operation, one of the transducer elements is excited with a step voltage which causes about 7 cycles of 2.25 MHz compression waves to be generated. The waves propagate through the window which is impedance matched to the FreonTM 11. Wave reflection from the liquid-vapor interface is received by the second transducer element. The electronic circuitry directly measures the time elapsed from transmission to reception and thus is proportional to film thickness.

Depending on the transducer beam angle, the returning signal can only be detected for a range of surface angles to the transducer normal. In addition, wave reflections from the window/liquid interface will occur because the impedance match is not perfect and blanking must be used. The reso-

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lution, however, is excellent being 0.015 inch/per bit. An 8 bit counter was used giving a range of about 2 inches. It is not difficult, of course, to increase the range significantly.

IMPLEMENTATION

The two modules are shown schematically in Figure 1. Except for the fluid/vapor and surface treatment, they are identical. Each module consists of two 500 cc chambers separated by a motor operated ball valve with a 1 cm bore. A rolling diaphragm piston assembly driven by a linear displacement stepper motor is used to produce volume perturbations. The stroke is software controlled with a nominal displacement of 5 cc in 1 second.

An Entran pressure transducer having 25 psia full scale range is used in conjunction with precision signal conditioning and a 12 bit Analog Devices A/D converter. The temperature is sensed with a Thermometrics "Fastip" thermistor probe and similar processing circuits. The thermistor bridge is pulsed for 40 ms to avoid inaccuracy due to self-generated heat. In addition, it is fitted with a conical capillary control shield and screen to ensure that the element will always be in the vapor and/or gas phase. On launch, very little liquid is in the left chamber so that the screen will not be defeated.

Water in conjunction with TFE coating constitutes the non-wetting system. Freon TM11 in conjunction with machined aluminum forms the wetting system. The ultrasonic film thickness system is fitted only to the wetting system. Figure 2 is a photograph of the TFE coated system.

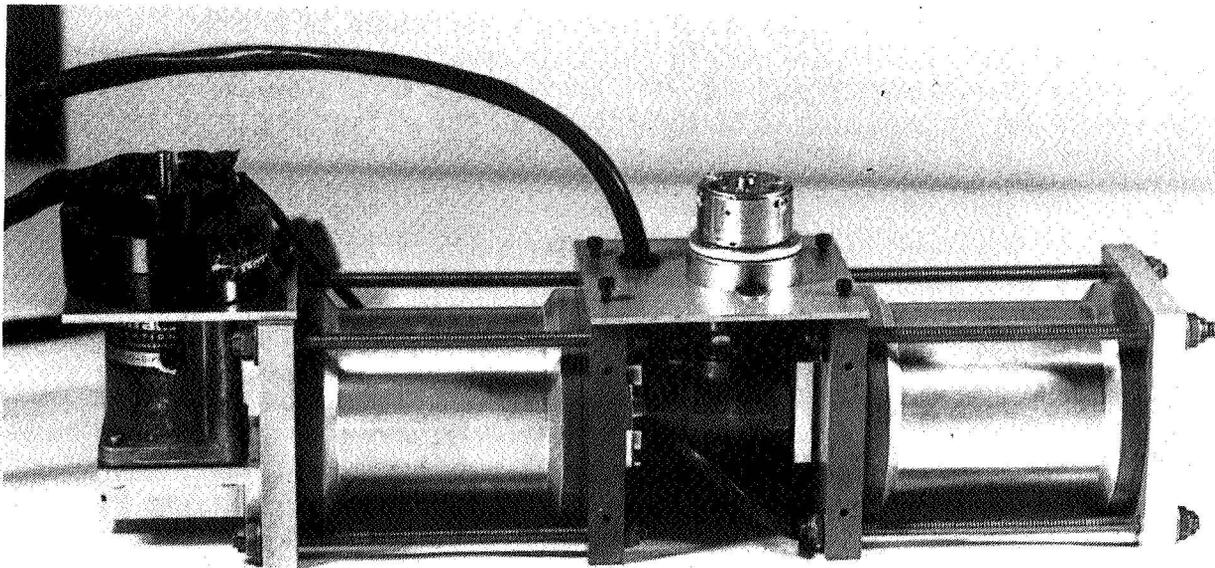


Figure 2 EXPERIMENT MODULE

Both experiment modules are controlled by an electronics assembly, Figure 3, which contains six circuit boards. The system block diagram is shown in

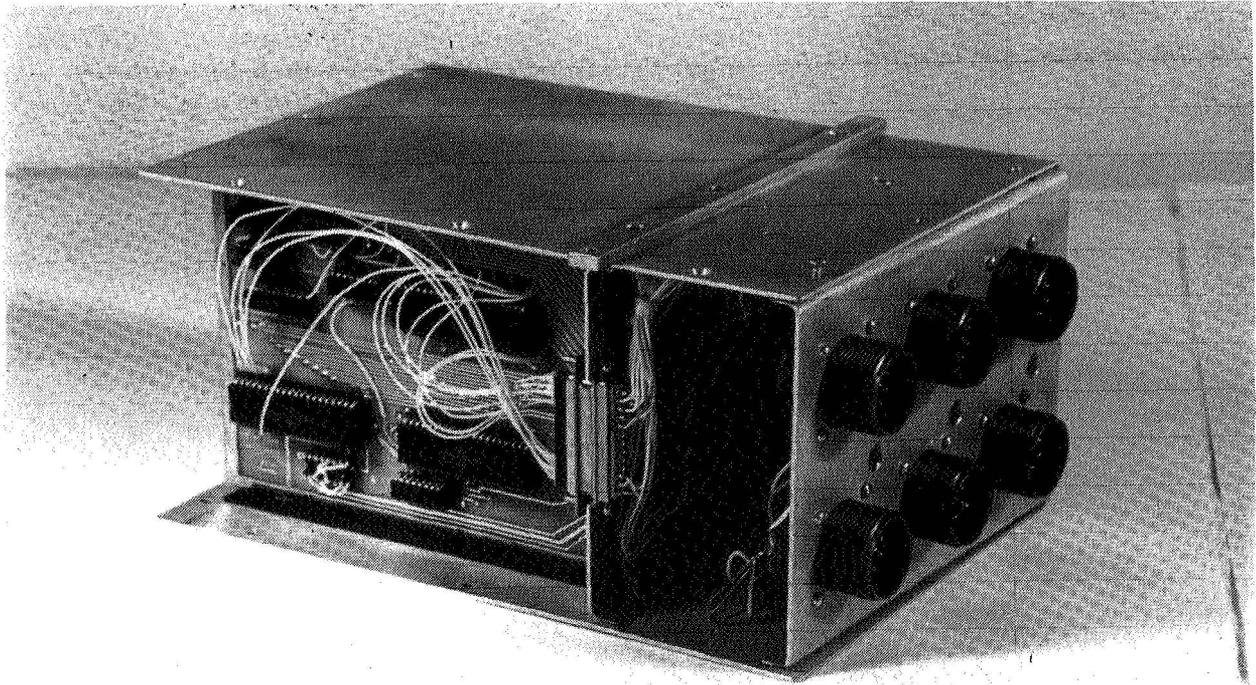


Figure 3 ELECTRONICS ASSEMBLY

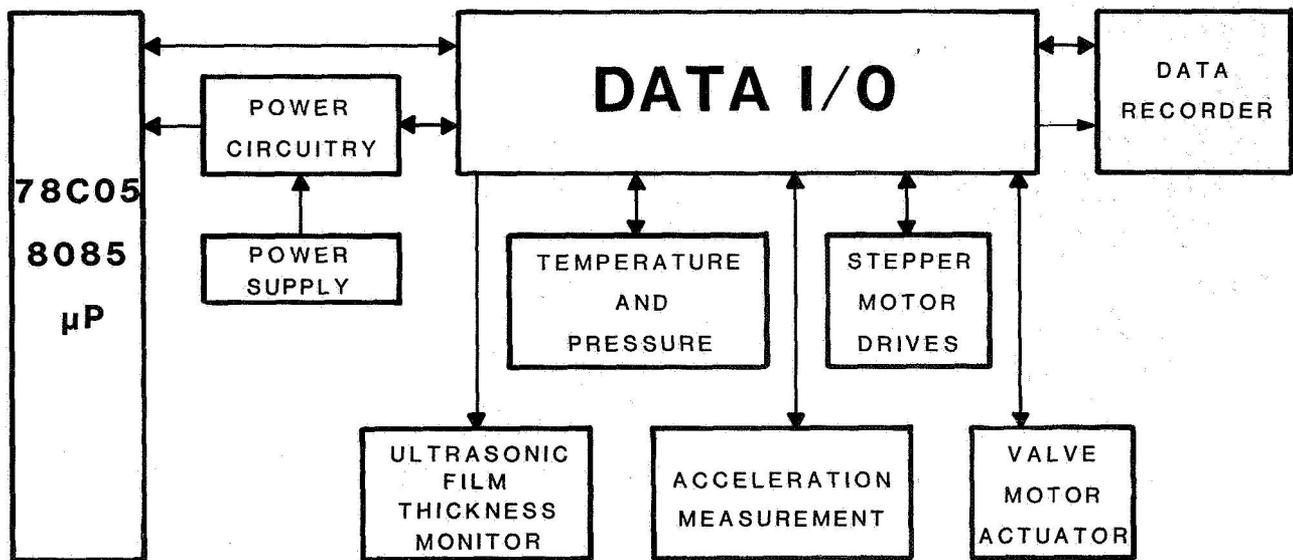


FIGURE 4 SYSTEM BLOCK DIAGRAM
MEASUREMENT AND CONTROL

Figure 4. A single board microcomputer is the basis for all measurement and control functions. It contains the operational software on PROMS which is booted when the GAS container is signaled by the crew when on orbit. A data I/O card provides all computer interfaces for measurements and controls. It utilizes both parallel and analog interfaces with other cards and performs serial data transmission to the (separate) tape recorder.

The primary signal processing and control functions utilize the remaining four circuit boards. These functions are:

1. Temperature and Pressure Measurement
2. Stepper Motor Drives
3. Ultrasonic Processor
4. Acceleration Measurement
5. Valve Motor Drives

In addition, auxillary functions are also included such as power switching reference voltage generation, clocks, and limit sensing. In order to maximize accuracy and resolution of temperature and pressure measurement, local signal conditioning was provided in each experiment module for the sensors. Figure 5 shows the Panametrics ultrasonic transducer fitted to the PMMA container endcap. The Wilcoxon accelerometers are housed in a separate box and utilize sample and hold circuits to establish the acceleration magnitudes over 5 second intervals.

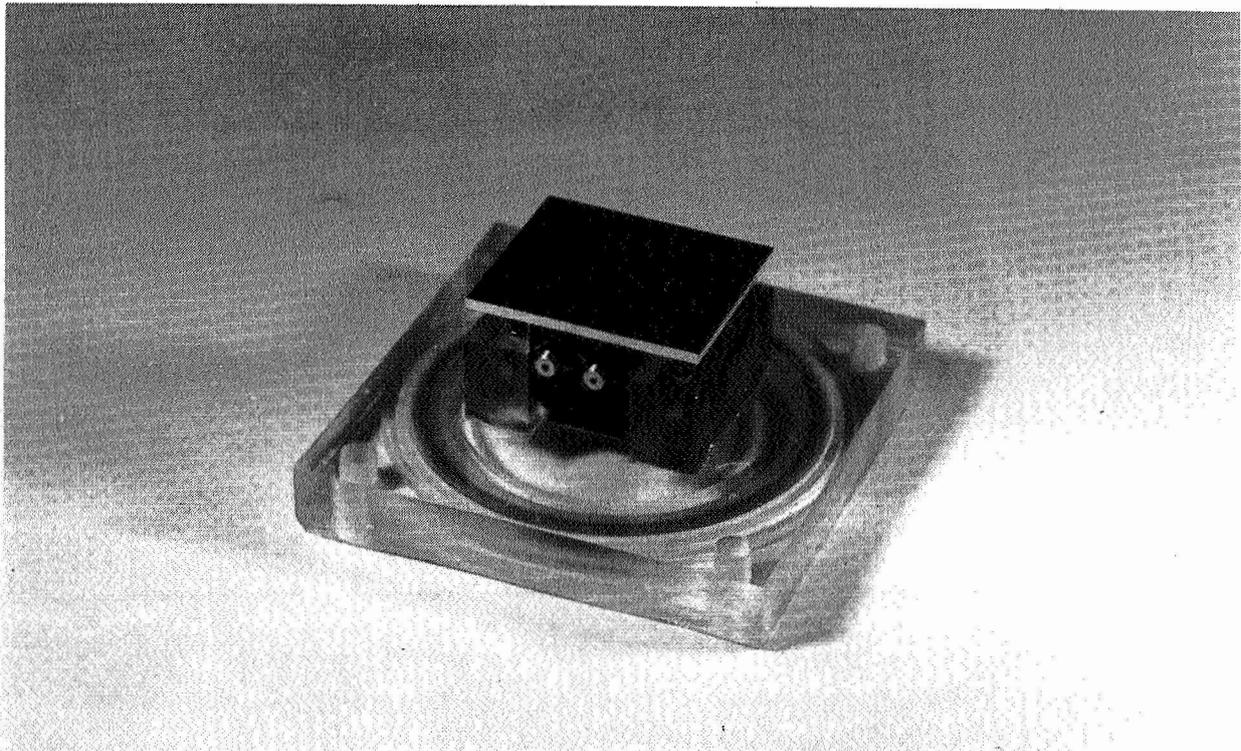


FIGURE 5 ULTRASONIC TRANSDUCER

TESTING

Environmental testing includes the acceleration spectrum specified for GAS payloads [4] on all three axes and function testing over the design range of 0-30°C. Vibration testing uncovered some unsuitable component mounting practices which were remedied with suitable fasteners and RTV sealant. In addition, stepper motors were found prone to drift so that software was added to "home" the motors before use. Because the pressures in the experiment modules vary differently than that in the GAS container with temperature the bias force on the stepper motors varies limiting the operational temperature range. The 0-30°C range was predicted from a canister thermal analysis and the piston assemblies spring biased for this range. The microprocessor checks the temperature and conducts experimental runs only when the range is satisfied.

Operational status was achieved using a Motorola 8085 emulator system to debug each board with its associated software. The software strategy was to utilize a subroutine for each measurement and control function. A supervisory program then called the subroutines in the desired sequence. The system executes a complete measurement cycle every half-hour including data storage from RAM to tape. The programmed duration is 72 hours unless powered down by the redundant actuator, experiencing low voltage, or signaled-off by the crew.

Considerable effort was applied to achieving very accurate temperature, pressure, and film thickness measurements and calibrations. The resolutions were ultimately limited by 1 bit for temperature and pressure and 1 ultrasonic wavelength for the thickness monitor. The final resolutions were 0.007°C, 0.006 psi, and 0.4 mm.

SUMMARY

This experiment was conceived as both an investigation of fundamental differences in the behavior of wetting and non-wetting systems and a demonstration of measurement techniques in the microgravity environment. It is one of a number of experiments to be flown in G-408 by the WPI/MITRE Space Shuttle Project program. It formed the MQP activity of two groups of students over a two year period. Eleven students majoring in Electrical Engineering, Mechanical Engineering, and Physics designed, fabricated, and tested the system.

The MITRE Corporation donated the GAS container and is providing overall project support. The Fluid Behavior experiment received substantial financial and engineering support from the Instrument Systems Division of Simmonds Precision. The contributions of these and other firms to the WPI effort are gratefully acknowledged.

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MEMBRANES
SPACE MANUFACTURING
REDUCED GRAVITY
PERMEABILITY
POLYMERIC FILMS
PHASE TRANSFER

REVERSE OSMOSIS
FILTRATION
ELECTRODIALYSIS
COMPOSITE MATERIALS
MORPHOLOGY
DENSITY (MASS/VOLUME)
CASTING

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THE CASTING OF SEMI - PERMEABLE MEMBRANES
IN A MICRO GRAVITY ENVIRONMENT

Ivan Vera
National Electric Company of Venezuela
C.A.D.A.F.E. - VENEZUELA

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I. Introduction

The National Electric Company of Venezuela, C.A.D.A.F.E., is sponsoring the development of this experiment which represents Venezuela's first scientific experiment in space. The experience can be classified as an educational one which could add knowledge to the fundamental study of polymeric membranes.

Presently, semi-permeable membranes are being manufactured from several different kind of polymers all over the world and specific applications have been identified in fluid separation processes such as reverse osmosis, ultrafiltration and electrolysis.

Although, the ultrastructure of asymmetric and composite membranes have been under intensive study, still there are many questions about the factors affecting this structure and their degree of correlation. Nevertheless, there is indication that the entire morphological structure of polymeric membranes could be affected by the differences in specific gravity between the cast solution and the coagulation liquid, normally used in the membranes preparation process.

The casting of semi-permeable membranes in space might help to identify the effect of gravity upon the structure of these membranes.

It is important to recognize that the casting process involves changes of states (liquid-gas, liquid-solid) and that in a micro gravity environment, there will be a reduction on buoyancy-driven natural convection and density gradients

II. Experiment Description

A schematic of the experiment concept is presented in Figure 1, along with the desired sequence. The experiment will be conducted in one-half of a longitudinal cylinder (5 cubic

0032-0000

feet) as shown by the sketch of figures 2 and 3. An engineering layout of the experiment mechanism is presented in Figure 4. The experiment will contain several solenoid valves, a motor, several actuators, a roller mechanism, and a reservoir for both water and the polymer solution to be cast. The payload has an active heating subsystem. The plan to maintain the temperature requirements of 0 to 10/30 degrees celsius is simply to allow the ambient thermal conditions in the Orbiter bay to drive the temperature down near 0 oC passively. If the temperature of the payload drops below 0 oC, a thermostat will activate strip heaters to elevate the temperature until 10 oC or 30 oC is reached at which time another thermostat will de-activate the heater.

The sequence after experiment activation is simply to activate a valve to open the polymer solution dispenser and simultaneously force the solution on to the roller device. The roller device is then linearly translated to coat the flat plate glass with a 10 mil thickness of the polymer. Sixty seconds after the linear traversal of the polymer dispenser has been completed, the timer will activate a valve which will flush the polymer with cold water (0-10oC). The experiment is complete at this time and will be de-integrated in the above condition. The membrane casting experiment has an absolute limit of 30 oC. Temperatures higher than this level will invalidate the experiment.

III. Integration and Tests

The membrane casting apparatus will be flown in the large 5 cubic feet cannister that corresponds to the GAS Payload No. 559, reserved by the National Electric Company of Venezuela, C.A.D.A.F.E. A secondary experiment of the Bio-processing Research Center of the Philadelphia University Center will also be integrated in the same cannister. This protein crystal growth experiment will share with the membrane casting experiment the large ITA standardized Experiment Module (ISEM) which has been modified to provide maximum linear flat plate distance to maximize the membrane yield as shown in Figure 5. Figure 5 also shows a small volume allocated to payload 3. This space will contain (if possible) photographic equipment to provide optical data on the membranes and protein experiments.

The modified ISEM will be fabricated of standard aircraft aluminum of the grade and quality shown in Figure 6, and will be analyzed and tested to the launch shuttle environment as is the standard ISEM. The modified design will have a structural margin greater than 1.5 which will be verified by test.

The avionics package to support both experiments, will consist of a power supply, recorder, programmer sequencer, heaters and thermostats, and instrumentation (pressure, temperature and accelerometers).

The whole system with the experiments, structure and avionics will be subject to acceleration, vibration and thermal-vacuum tests and results will be reported at a latter time. The acceleration and vibration tests will be as specified in the July 1984 issue of the NASA GAS Handbook (page 57).

IV. Special Design Considerations

Special design considerations were taken in order to ensure the construction of an apparatus which could handle, in a successful way the fluids involved in this experiment.

One of the major design concerns was the elimination or reduction of Air Bubbles in the surface of the membrane, at the time when the casted membrane is submerged in water. The solution to minimize bubble formation was to provide space for the water to filter through two screen areas as shown in Figure 3.

Another important concern was that the fluids have to be kept in a range of temperatures between 0 oC and 30 oC. According to data available of the more likely internal temperatures to be experienced by the GAS containers, temperatures below 0 oC should be expected during the space mission. Therefore, heaters were needed to ensure temperatures above this limit at all times. The probability of high temperatures (above 30 oC) are only expected some time after the landing of the Space Shuttle. However, the payload will have been cold prior to entry (0 oC to 10 oC) and the thermo-time lag in conjunction with the air conditioning equipment provided by NASA should eliminate this risk of high temperatures.

V. Important Observations

The GAS program of NASA presents certain requirements and limitations, some of which were important to recognize previous to the development of this experiment:

1. A minimum of 13 months is usually necessary for the completion of all the documentation, safety procedures and integration of the experiment.
2. It was necessary to ensure that the fluids involved in this experiment could stand the minimum of two months specified by NASA at the launching site, without losing chemical consistency and without allowing organic growing.
3. The internal temperature profiles that the container will probably experience were carefully studied in order to ensure that the experiment could be successfully performed.

Consultations

The following organizations-persons provided guidance, recommendation and encouragement:.

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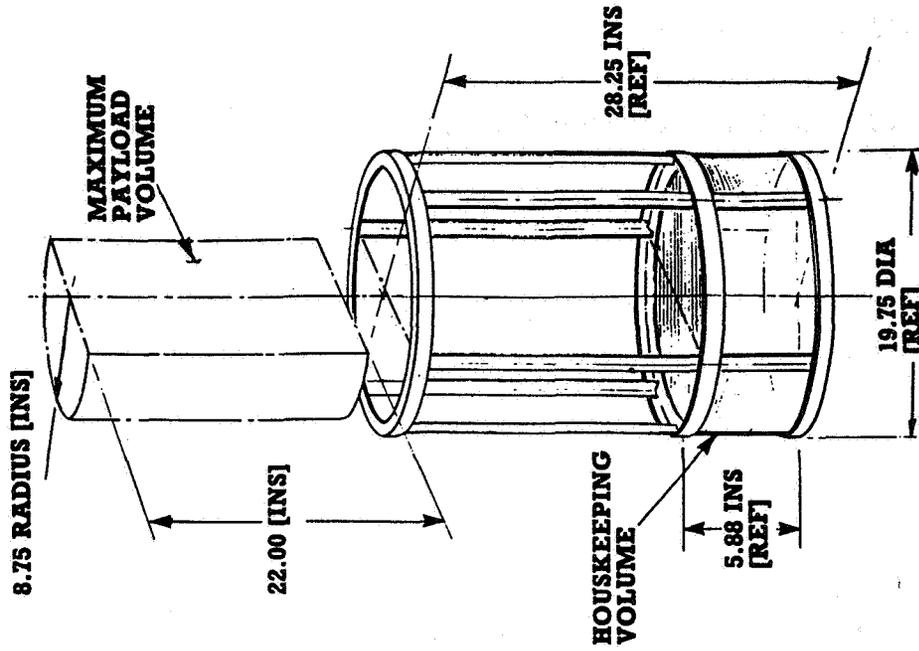
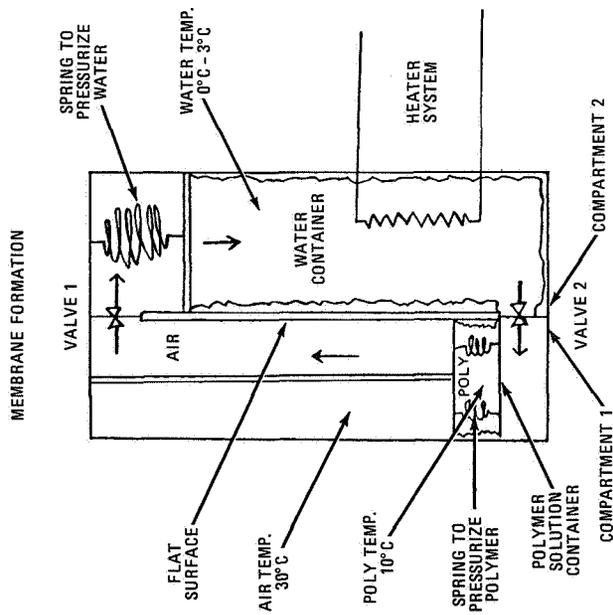


Figure 2. LONGITUDINAL CYLINDER



SEQUENCE

- 1) Motor will move polymer container across flat surface. At the same time pressurized system will force polymer solution out of container, casting membrane on flat surface.
- 2) After one minute Valve 1 and Valve 2 will open and water will be forced to go through Valve 2 to Compartment 1, while air will be forced to go through Valve 1 to Compartment 2.
- 3) Once water has been transferred to Compartment 1 and air to Compartment 2 Valve 1 and Valve 2 will close.

Figure 1. EXPERIMENT CONCEPT

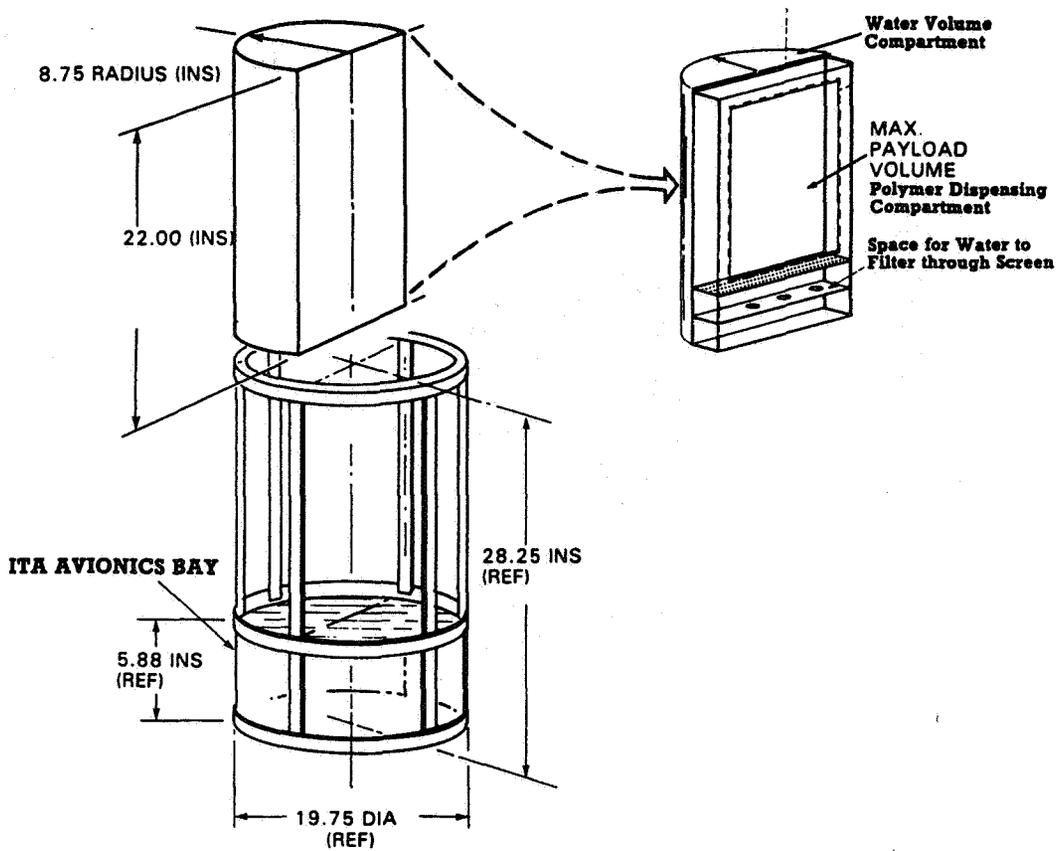


Figure 3. IVAN VERA PAYLOAD CONCEPT IN THE MODIFIED ITA EXPERIMENT MODULE

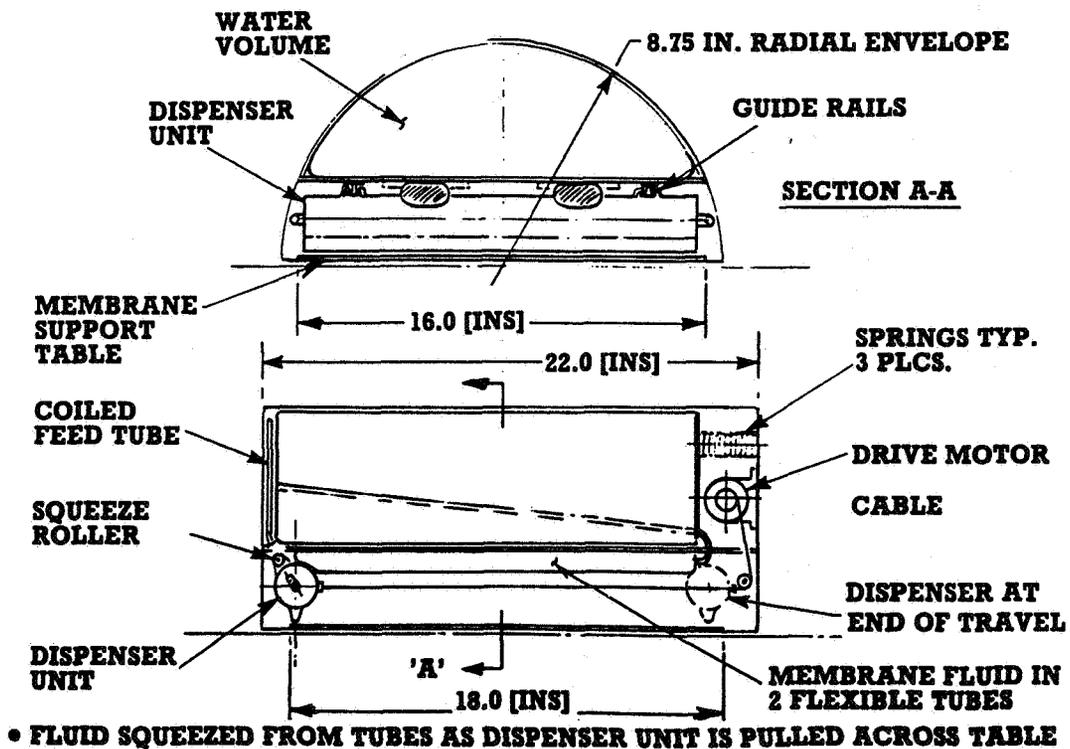


Figure 4. EXPERIMENT MECHANISM

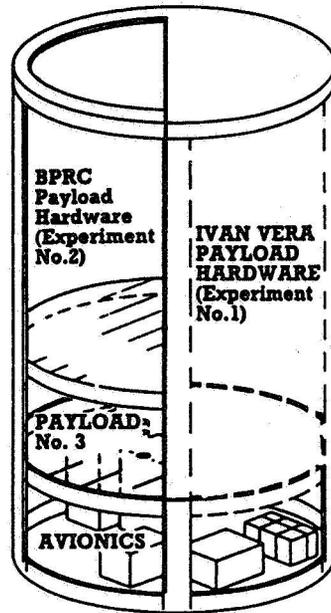


Figure 5. PRIME PAYLOAD NO. 1
IVAN VERA EXPERIMENT



Experiment bays tailored to client requirements

- Shuttle compatible for GAS program
- STRUCTURE - 6061 T6 A1 anodized
- Weight - 28 lbs. (for 5FT³ model)
- Adjustable lateral support snubber design
- Fasteners - Aerospace CRES stainless steel
- Adjustable experiment deck height
- User Experiment Bays accommodate
 - Payload weight ~ 140 lbs.
 - Payload volume ~ 3.5FT³

ITA Bay with Base-Line Avionics

- Power Supply
- Recorder
- Instrumentation
- Programmer/Sequencer

Figure 6. ITA STANDARDIZED EXPERIMENT
MODULE STRUCTURE CHARACTERISTICS

SHUTTLE
SPACE PAYLOADS
PAYLOAD INTEGRATION
SPECTROMETER
FLUID DYNAMICS
GRAVIRECEPTOR

LUMINOUS
NITROGEN DIOXIDE
CHEN. COMPOSITION
CENTRIFUGES
FMNOL
AIR GLOW
THERMAL ENVIRONMENT

N86-27304

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88

8960

CU IN SPACE: The University of Colorado
Get Away Special Payload G-285

Presented by

C4508845

Kent Tobiska
University of Colorado
Boulder, CO 80309



Getaway Special Team
University of Colorado

PROJECT OVERVIEW

The University of Colorado's first Get Away Special was conceived by a dozen science and engineering students in the spring of 1984. Since that time, the project has grown to include over 100 undergraduate and graduate students who have provided the experimental objectives, design, construction, integration, testing and management of the payload. Faculty and staff of the university and its affiliated laboratories as well as engineers from local industries have provided many useful suggestions at all stages of development.

The principal motivation for developing this payload is to provide university students with the opportunity to participate in space science and engineering beginning at the ground level. At the same time, they are contributing their results from new science and technological discoveries to several disciplines.

There are four separate experiments integrated into the single payload of G-285 which are supported by internal payload power, thermal environment control, command and data handling facilities and structural subsystems. The four experiments contained in this payload are: a single spectrometer performing two experiments- one to study night time shuttle glow phenomena in the ultraviolet and the other to observe NO2 concentrations over equatorial latitudes in the day; a fluids management experiment using centrifugation for particle/liquid/vapor separation; and an experiment using Phycomyces fungi to study the gravireceptor mechanism theory.

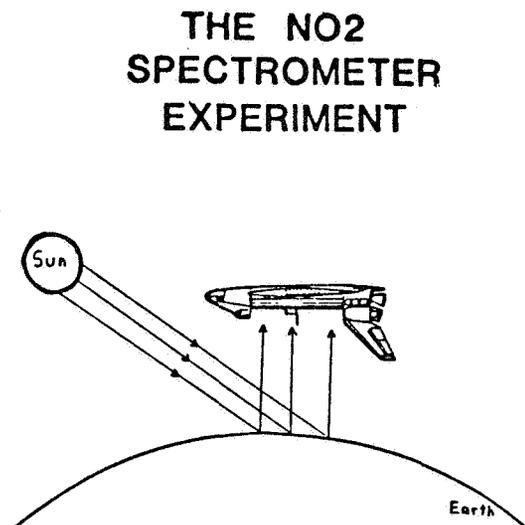
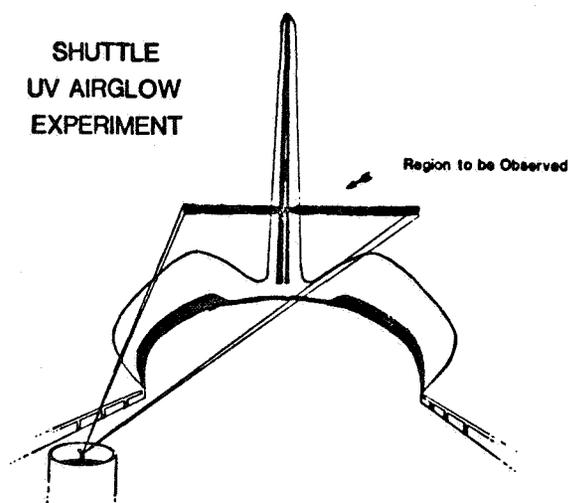
The experiments are supported by a microprocessor, data storage devices, power and thermal control subsystems. All these are located within a sealed aluminum container. The container, a refitted sounding rocket casing, has a top and bottom which bolt to the main body of the rocket section. The top section holds a five inch diameter quartz window through which light enters the spectrometer. A rotating mirror assembly and one of two battery boxes are mounted on the outside of the top section below the interface plane of the MDA. A second battery box is attached to the bottom section of the sealed container. All payload items except the battery boxes and mirror assembly are located inside the sealed rocket section to insure containment and to prevent outgassing of materials. The container will be sealed prior to launch and purged with one atmosphere of argon.

THE SCIENCE: Atmospheres, Fluids, Biology

A,B) UV/VIS Spectrometer Experiments

The objective of the shuttle glow experiment is to conduct a general survey of atmospheric emissions near moving surfaces during dark portions of each shuttle orbit. The 1900-3100 Å (UV) wavelength range of the refitted Ebert-Fastie spectrometer yields information about atomic and molecular interactions occurring in the near-shuttle environment. A rotating mirror assembly set at the correct tail-viewing angle focuses light into the spectrometer from the vicinity of the rear stabilizer (tail). The second spectrometer experiment makes use of the instrument's visible light sensitivity by taking nadir observations of tropospheric NO₂ over equatorial latitudes during sunlit portions of each shuttle orbit. With the mirror assembly used in the night-time glow study commanded to an upright position, continuum emission from visible NO₂ enters the spectrometer through the quartz window. Data for both atmospheric experiments are stored on the same storage device.

To successfully perform both spectrometer experiments, the instrument is mounted vertically with respect to the GAS can. This gives visible light direct access to the optical train of the spectrometer during daylight. During dark orbit segments, the instrument's field of view is directed onto the rear stabilizer by the mirror assembly, which rotates through 7.5 degrees of arc using ten positions, resulting in 30x30 cm spatial resolution. This configuration allows for the acquisition of data both on and off the shuttle tail surface with a spectral resolution of less than 10 Å.



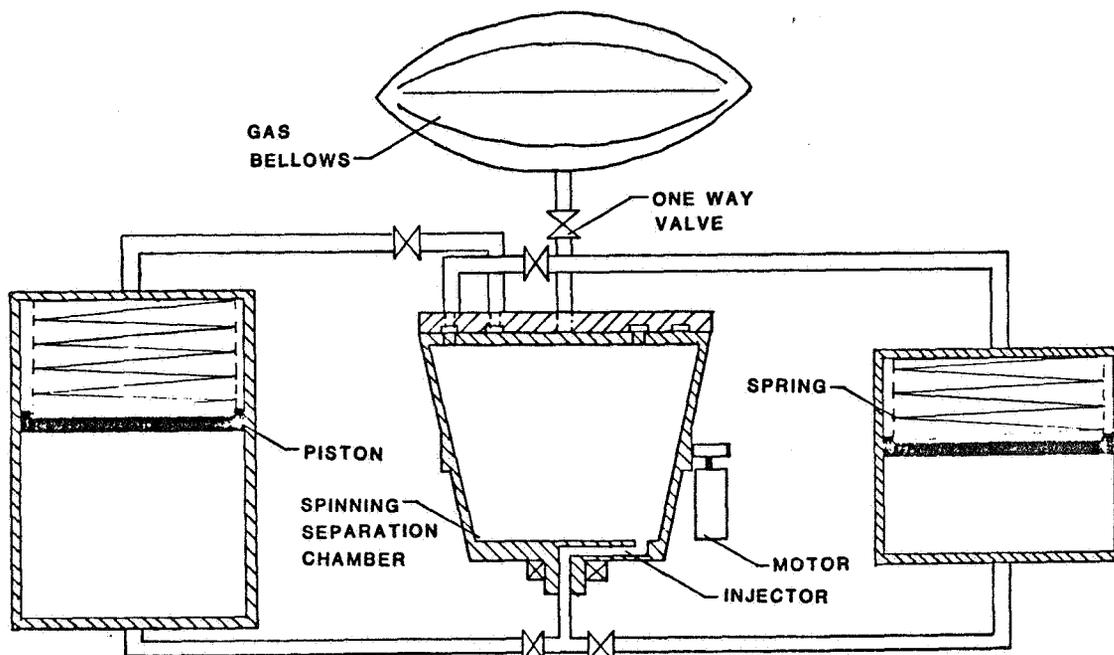
**OBJECTIVE : MEASUREMENT OF
TROPOSPHERIC NO₂ IN EQUATORIAL REGIONS**

C) Fluid Management Experiment

Management of fluids in space is complicated by the behavior of fluids in the absence of gravity as well as by the limited quantity of fluids that are typically available for use. Three specific model problems have directed the designing efforts: liquid degassification, solid-liquid-gas separation, and algae growth, separation and harvesting. To address these problems, a system has been designed that is conservative, requiring little energy, and can be applied to a wide variety of fluids management challenges.

The apparatus incorporates a centrifugal approach using a truncated cone-shaped spinning separation chamber, coupled with density-dependent valving. Degassification of fluids and the separation of multi-phased media is achieved while creating the pumping pressures needed to move the separated media to desired areas. Data is retrieved through photographic and pressure measurements. Upon return of the shuttle, the results should document the overall viability of fluids management by centrifugation in a microgravity environment.

SCHEMATIC OF FLUID MANAGEMENT EXPERIMENT



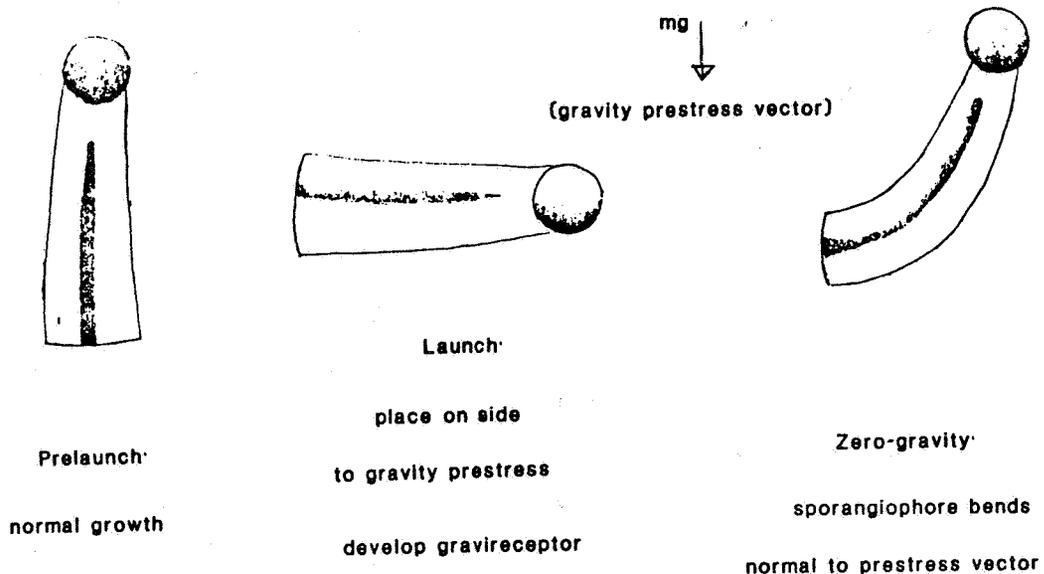
D) Phycomyces Fungi Experiment

The final experiment is designed to study the harmless, non-toxic single-celled Phycomyces fungi to verify a current theory on plant cell susceptibility to gravity and acceleration stresses. A zero-gravity environment is required to verify the proposed mechanism for cellular reorientation: it is anticipated that the sporangiophores will bend toward the normal with respect to the gravity forces generated by the shuttle takeoff acceleration vector.

Because the G-285 payload must sit unattended for 2-3 months prior to launch, students have performed experiments aimed at prolonging the dormancy of the fungus which otherwise has a shelf-life of only a few weeks. Five days prior to launch, growth is initiated in the fungi by dropping inactive spores onto a growth medium. At launch, the beds containing the growing phycomyces will be rotated on a gimbal assembly through 90 degrees and locked into place. Data pertaining to the subsequent growth patterns will be recorded on film for two hours after launch.

PHYCOMYCES FUNGI

EXPERIMENT



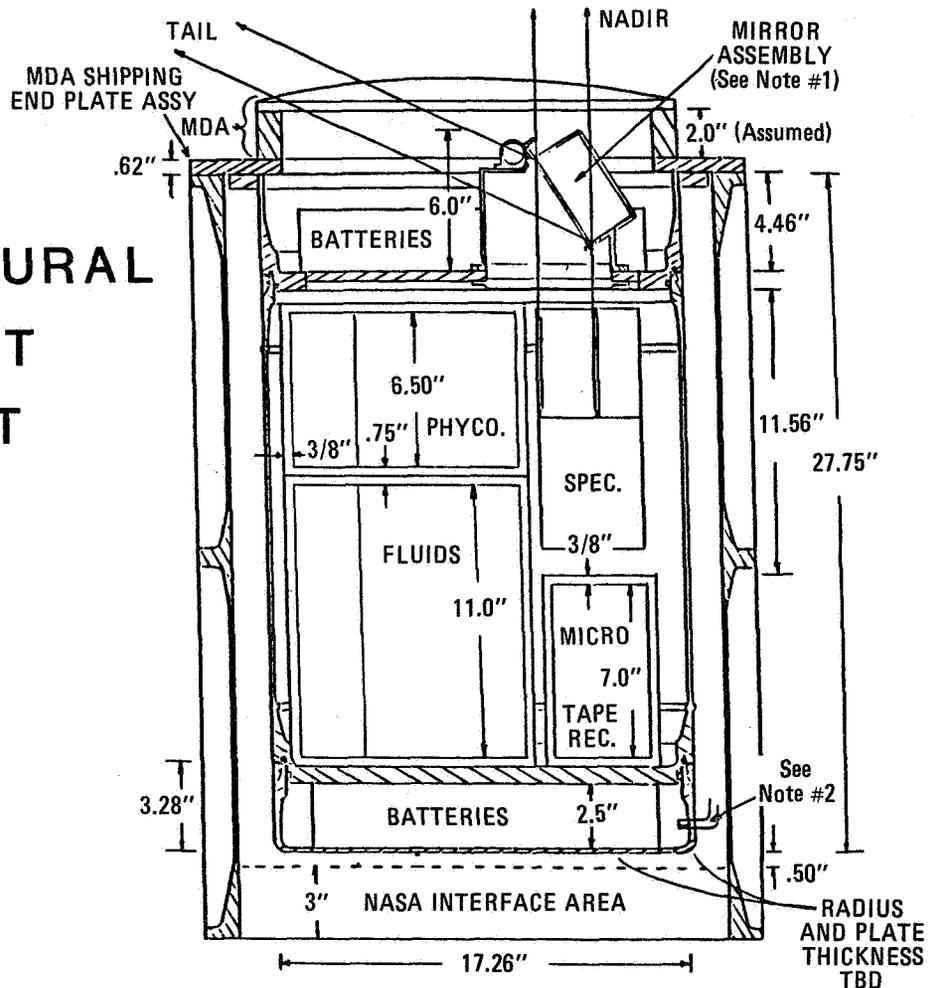
ENGINEERING SUBSYSTEMS

A) Support Structure

GAS experiments must be fully self-supported and self contained. The University of Colorado payload requires the five cubic feet user space of a cylinder 28.25 inches in height and 19.75 inches in diameter. G-285 requires the lid of the cannister to be opened during flight to accommodate the two atmospheric experiments, thus restricting any exposed surfaces to non-outgassing space qualified materials. For this reason, G-285 uses a sounding rocket casing (6061 T-6 Al) to house the experiments and their supporting equipment.

The bottom of the rocket casing is also fabricated out of 6061 aluminum and is designed to withstand the shuttle's extreme buffeting with only small deflections. The loading of the support structure has been analyzed using a worst-case scenario. At about 1000 Hz, the effective loading would be as much as 188 times its static weight. Even in this extreme case, the weight of the structure can be kept low enough to meet NASA's weight requirements while providing a safe environment for the internal science and engineering packages.

STRUCTURAL
SUPPORT
SIDE CUT



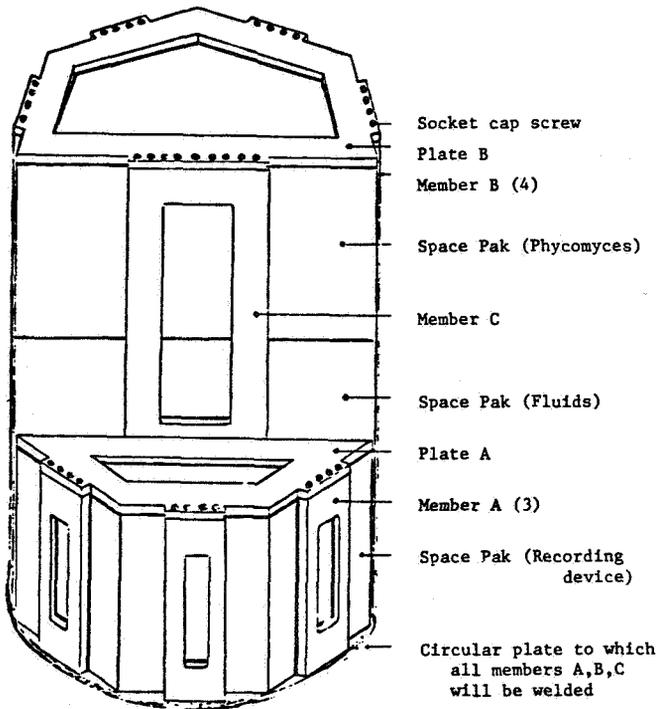
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B) Space Paks

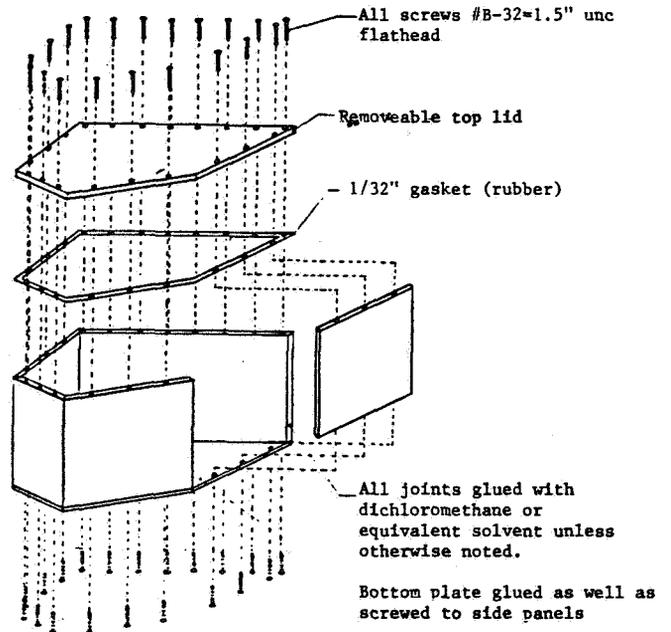
Internal to the rocket section, individual quasi-hexagonal space paks contain the fluids management experiment, fungus experiment, and the microprocessor/data storage units. (The spectrometer and bright object sensor are not housed within a space pak). The space paks are constructed of polycarbonate material (Lexan) to provide strength, low weight, visibility from the outside, and ease of construction.

Space paks serve two functions: to provide a mounting surface for the experiment apparatus and to isolate one system from another. Each space pak is sealed with an O-ring and contains one atmosphere of air at the time of installation. The rocket cannister as a whole is purged with argon prior to flight, in order to maintain a non-emitting environment for the spectrometer experiments.

SPACE PAKS within Structural Casing



Structural Material: Aluminum 2024-T4*



ALL PANELS MACHINED FROM 3/8" THICK POLYCARBONATE

*Rubber will be used between space paks and all plates and members

EXPLODED VIEW
 TYPICAL SEALED CONTAINER ASSEMBLY

C) Power Subsystem

The power subsystem provides electrical power for all four experiments and all support equipment in the canister. The G-285 payload has the somewhat prodigious power requirements of sequentially running the fluids experiment motor drive, data storage devices, MDA, spectrometer, and a series of thermistors that maintain an ambient thermal environment in the sealed rocket canister. Silver-zinc batteries will supply all power to the payload. For safety considerations, fuses have been inserted into most of the circuits.

The batteries are constrained both upon activation as well as during flight by two battery boxes that are attached top and bottom to the outside of the sealed container. Since the battery outgasses during discharge, it is necessary to vent any pressure buildup and possible combustion of gas mixture through purge valves located on the upper end plate.

D) Thermal Subsystem

The thermal environment of G-285 could be considered an exercise in extremes. While worst case scenarios of the cargo bay facing into direct sunlight and toward deep space occur infrequently, the sensitive design of the spectrometer and the fluids management vessels containing liquid require an active temperature monitoring system to prevent overheating and/or freezing of the experimental components. Maximum heat loss from the container occurs while performing the two atmospheric experiments with the open MDA. Maximum heat gain occurs when the shuttle cargo bay is illuminated by solar radiation.

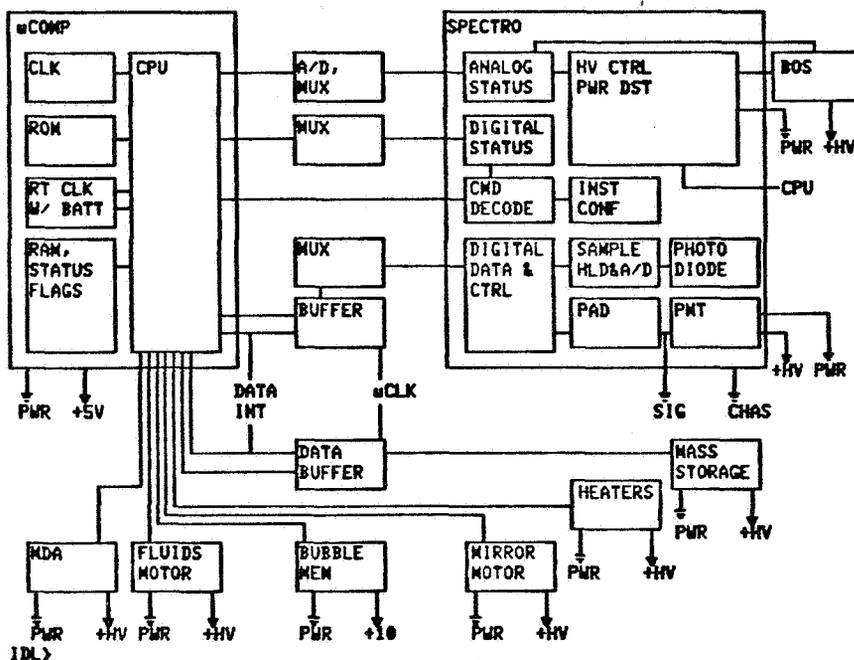
An operational scenario has been devised to maintain the most stable thermal environment possible in addition to insulating the can's interior. The biology experiment reaches completion two hours after the shuttle is launched, and has no residual thermal requirements beyond maintaining a suitable environment for the photographic film on which its results are stored. The fluids management experiment, time-lined to operate for approximately two hours, will be kept at a relatively constant temperature by means of small thermistors placed strategically throughout the rocket section. Power to run the centrifuge motor will add a small amount of heat to the system, but is not anticipated to significantly affect the results of the experiment. At the completion of the fluids study, the MDA is commanded open and held in an upright position. The spectrometer is powered up for the next 48 hours, during which time the shuttle glow and NO₂ observations are made. In the event that the thermistors are unable to keep the spectrometer sufficiently warm (as heat is being radiated outward to space), or that the can interior heats up due to solar illumination, the microprocessor will automatically close the MDA. Because the temperature is constantly monitored, in the event that interior temperatures stabilize within the preset limits, the MDA will be reopened and data collection will resume.

COMMAND AND DATA HANDLING SUBSYSTEM

The Command and Data Handling Subsystem of the University of Colorado Get Away Special project consist of a controlling microcomputer and two types of data storage units. The microcomputer system is comprised of National Semiconductor MA2000 Macrocomponent modules. The central processing unit module (MA2800) incorporates an 8 bit NSC 800 microprocessor operating at 2.5 MHz, 4K bytes of ROM, 2K bytes of RAM and a real time clock. These modules represent a family of stackable microcomputer function blocks which are configured in such a manner as to interface with the following systems:

- I. UV/NO2 Spectrometer
- II. Fluids management experiment
- III. Data storage devices
 - a. Bubble memory
 - b. Hard disk
- IV. Bright object sensor (to protect the UV experiment photomultiplier tube from direct sunlight)
- V. Network of pressure and temperature sensors

Data acquired from the UV/NO2 experiment, the fluids management experiment, the pressure and temperature sensors will be stored, along with time tags, in bubble memory. The hard disk system, which has never been used in this environment before, will be used as the secondary data storage system and will continue to collect data after the bubble memory has been completely filled.



METAL FORMS
TIN
SOLID SUSPENSIONS
SPACE MANUFACTURE

EXPERIMENT DESIGN
BUBBLES
ZINC COMPS.
CARBONATE
REDUCED GRAV.
RADIATION SHIELDING

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8961

Design of an Experiment for the Production
of a "Foamed" Tin Sample

by

Eric Wernimont
Purdue University
West Lafayette, Indiana 47907

89391092

Abstract

The purpose of this paper is to present to fellow GAS users a portion of Purdue University's efforts. One of the major experiments in our GAS container is concerned with the experimental production of a foamed metal. A foamed metal is one that contains a significant amount of gas bubbles suspended in its solid volume. Purdue's GAS team proposes to do this with the help of a solid zinc carbonate that gives off carbon dioxide at high temperatures. Because of low energy requirements, the metal used for this experiment is tin. It is hoped that the use of near zero environment will keep the suspended bubbles more uniform than in an earth based process, hence not depleting the physical strength of the material as greatly as is observed on earth.

1. Introduction

A major problem with construction of large space structures is that large amounts of material must be brought from either the earth or the moon. This is both expensive and time consuming. One method that has been proposed to overcome this problem is to produce at the construction site a material that has small gas bubbles uniformly distributed throughout the foamed metal's volume. This process greatly increases the useful material while decreasing the expense to bring the material to space. Not only could this type of material be used for structural purposes but also for radiation shielding from the sun. The amount and type of gas would largely depend on the purpose chosen for the foam metal.

The Purdue GAS team is planning to produce a sample of a foamed metal. This will be a cylindrical piece of tin, 3 inches in height and 1 inch in diameter, having 1/2 of the volume occupied by carbon dioxide. Placing the carbon dioxide into the tin will be done by a foaming agent (a chemical

that gives off a gas at high temperatures). For our purposes, a minimum energy foaming agent is desired, so zinc carbonate is chosen. At 575 kelvin, zinc carbonate decomposes to zinc oxide and carbon dioxide. Because of the small amount of foaming agent required, the impurity resulting from the zinc oxide is negligible.

This experiment is broken down into three distinct phases: (1) construction (preflight), (2) experimental (flight) and (3) testing (post-flight). Each of these are discussed below.

2. Construction (preflight)

In order to obtain the desired final geometry, a cylindrical containment vessel with moveable piston heads is chosen. A movable piston is necessary to allow expansion during the foaming process when the carbon dioxide gas is given off. The temperatures required for the decomposition of zinc carbonate (the foaming agent) suggested that a ceramic container would be suitable.

Another consideration in this selection is that the vessel has to be structurally strong, have high thermal conductivity (k), low specific heat (C_p) and low density. These requirements will minimize the chances of the vessel breaking on launch due to vibrations and provide for a low weight container. The high conductivity would insure that the heat given off by the nichrome heating coil around the container would quickly be transferred to the tin sample. The low capacity would insure minimal heat retention by the container. After much consideration an alumina ceramic container is chosen with dimensions given in Fig. 1 (and the physical properties of $k = 18.5 \text{ W/m K}$, $C_p = 906 \text{ J/kg K}$, density = $3850 \text{ kg/cubic meter}$). To heat the zinc carbonate, to decomposition temperature, a nichrome wire is used. For

the heat requirements, a total of 9.2 turns (.66 container height ohms/ft.) is needed, requiring one turn every 1/3 inch. To ensure that most of the heat given off is effectively used, a glass fiber blanket ($k = .038 \text{ W/m K}$, $C_p = 835 \text{ J/kg K}$, density = 32 kg/cubic meter) is used to insulate the nichrome wire from the environment.

To enable the now liquid tin to solidify, after the zinc carbonate has decomposed, an aluminum rod (1/8 inch diameter, density = 2700, kg/cubic meter, $k = 237 \text{ W/m K}$, $C_p = 903 \text{ J/kg K}$) is placed so that when the two containment pistons, seen in Figure 1, are expanded fully, one is in contact with one end of the rod. The other end of this rod is connected to the passive thermal control salt that is used to control the temperature of the whole payload. By doing this, the thermally "hot" region of the experiment is connected with the thermal sink of the passive heat cells. The heat cells absorb energy if not fully charged, which will be the case by the time the experiment is in operation. This speeds up the cooling process, which prevents the sample from having nonuniform carbon dioxide bubbles that can be caused by possible orbiter maneuvers during a long solidification period.

To insure that in a weightless environment the piston heads stay in contact with the tin-zinc carbonate mixture a "pseudovacuum" is created. This is done by first heating up the required amount of tin to the liquid phase, but below the decomposition temperature of zinc carbonate (575 k), then adding the zinc carbonate as a solid. This liquid mixture is poured in to the containment vessel with one end blocked by a piston head. After allowing it to solidify, the solid cylinder of tin-zinc carbonate mixture is removed from the container and the ends of the sample are ground to remove any oxidized tin as well as modifying a flat surface for contact

with the other piston head. Next, the solid tin-zinc carbonate and one piston head are held against one another while these two pieces (tin-zinc carbonate first) are pushed into the containment vessel until the uncovered end of the tin-zinc carbonate is flush with the centerline of the hole for the stopbar (see figure 1). The stopbars are used to prevent the piston heads from overextending during the foaming process. In this position the other piston can be put in place while the trapped air escapes out the stopbar hole. These three pieces are pushed down a little more and the stopbars are put in place. Because there is little space between the pistons and container wall, no air is able to separate the pistons and the tin-zinc carbonate sample, regardless of relative position of the piston tin-zinc carbonate pieces to the containment vessel. The nichrome heating wire is then wrapped around the containment vessel. The glass fiber insulation blanket is added leaving room for the exit of the aluminum heat sink rod and the nichrome heating wire leads to the power supply. The entire apparatus is now complete and ready for launch.

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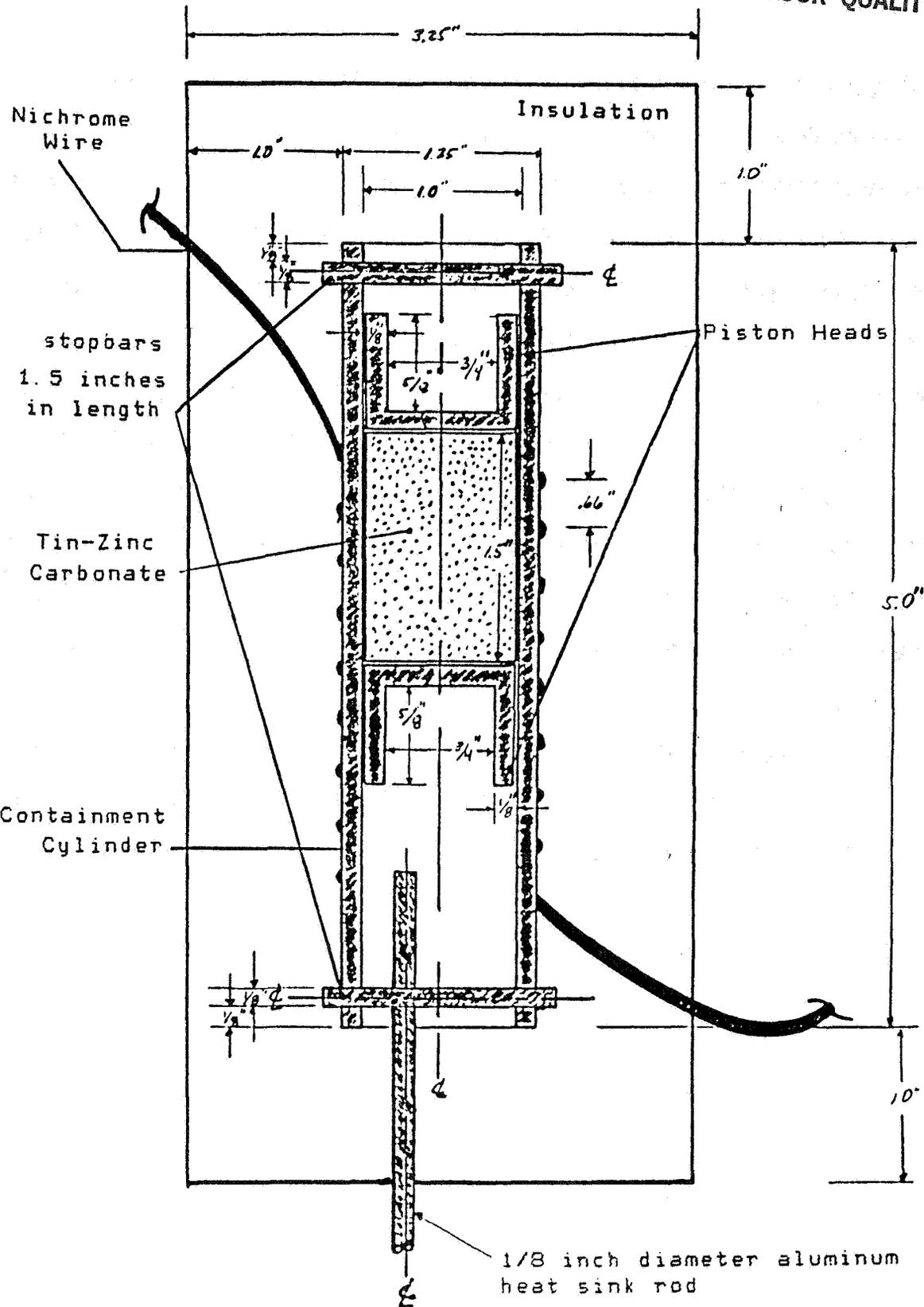


Figure 1 Cross Sectional view of Apparatus

3. Experimental (flight)

Once in orbit during a period when the Orbiter is not making any maneuvers, the nichrome heating wire is turned on by the canister's micro-controller. The micro-controller will hold the temperature of the wire at about 600 K which is approximately 25 K above the decomposition temperature of zinc carbonate. This is to be done by a direct temperature feedback system using a thermocouple placed on the outer containment vessel wall (not shown in Fig. 1). Once power is applied, there will be a short warm-up period for the container, then the tin sample goes into the liquid phase (595 K). Decomposition of zinc carbonate begins at about 575 K. During this decomposition the zinc carbonate becomes zinc oxide and carbon dioxide. Because carbon dioxide is a gas and zinc carbonate is a solid the volume increases greatly, moving the pistons away from each other until stopped by the stopbars. Using a finite difference thermal model the approximate total time required between warm-up and decomposition of zinc carbonate will be 10 seconds. Thus a timed power shut down after 15 seconds is sufficient. Because of the insulation almost all of the required 10.5 kJ of power will be dumped into the passive heat cell used by the entire canister. Using another finite difference thermal model, the estimated time required between power shut down and solidification will be less than 5.0 minutes.

4. Testing (postflight)

After the flight the foamed sample made in space will be strength tested against samples of solid tin and against other foamed samples made on earth. The sample foamed in space is tested against samples made on earth to determine if the carbon dioxide gas generation is more uniform in space and if so how uniformity effects the strength. Using strain gage analysis

the Purdue GAS team will be able to determine the full extent of the changes in material strengths under shear, torsion, compression and tension. It is hoped that the strength to weight ratio will greatly increase in all areas.

PROJECT PLANNING
PAYLOAD INTEGRATION
SPACE SHUTTLE INTEGRATION
PROJECT MANAGEMENT
MANAGEMENT PLANNING

MISSION PLANNING
SPECIAL MISSION
PROGRAM

D9-12
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THE DEVELOPMENT OF A PROJECT PLAN FOR THE GET AWAY SPECIAL PROGRAM

STEVEN J. BUTOW
PAYLOAD MANAGER G-480

SB413911

SPACE RESEARCH & DEVELOPMENT ORGANIZATION (SRDO)
SAN JOSE STATE UNIVERSITY

ABSTRACT

Trying to get a project started? Well, since the introduction of the Get Away Special Program, there have been 451* reservations placed by people who, just like yourself, are eager to send a small payload into space; and yet, only 33* of them have actually succeeded. Even more staggering, many of those who have flown have done so more than once; meaning that less than 10% of all GAS users have actually sent something into space. This paper approaches some of the problems that face GAS users and should be especially helpful to those who are new to the program. Some of the subject areas include selecting a project, and payload management.

SPECIAL NOTE: This paper is especially written for the "first time" Payload Manager. Due to the limited space in these proceedings I was only able to put a fraction of the information into this paper that I had originally intended; however, I have committed myself to documenting this topic in detail so as to provide GAS users with a better means to organizing their projects. So relax, this is not a technical paper. Your suggestions and comments would be appreciated.

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INTRODUCTION

NASA's Get Away Special Program has presented researchers with an opportunity to do what was once considered unthinkable, experiment in space at a relatively affordable price. In 1981, the Space Transportation Office established a user price of \$718.00 per pound for major payloads onboard the Shuttle. This is quite reasonable for the typical users of space such as the military, larger commercial entities, and foreign governments; however, in all reality, any one individual would have to be quite wealthy to afford to put a small payload into space. With the inception of the Small Self-Contained Payload Program (SSCP), NASA made space available to prospective small payload users for an exclusive price of \$50.00 per pound. Hence, the Get Away Special got it's name. Why is it, then, that NASA does not have GAS payloads booked through the end of the decade? This dilemma seems to stem from the rather complex nature of the GAS Program in general. This is by no means the fault of NASA; it is, however, an indication that the program is young and much of the support hardware and services that one would expect to have at his access simply do not exist, or are very limited.

In conversations with other GAS users, there seems to be some continuity in the type of problems that face many of them. There are GAS users who cannot decide what to put in their experiments; while, of the ones that have decided, many lack the engineering and technical support to turn their ideas into reality. The GAS user must be careful not to exceed his abilities. The first payload should be simple, utilizing those resources that are available. Before starting on a Get Away Special project, the user should familiarize himself with the basics of the GAS program.

PROJECT REQUIREMENTS

At its present level, the GAS Program requires an experimenter to have some basic engineering abilities; also, some familiarity with the Space Shuttle and its operational environment. Do not despair if you should lack these abilities. There are engineering support groups and small companies that are in the business of designing systems for GAS payloads. These engineering services can be costly; therefore, GAS users may be better off duplicating existing designs rather than paying big bucks to find new solutions to solved problems. Conserve your resources - know your costs. Publications geared towards the new users of space can be obtained through the Government Printing Office (GPO), the local library, or through a regional NASA center.

WHERE TO START

Following the placement of your GAS payload reservation, NASA will forward you all kinds of documentation pertaining to the Space Shuttle and the GAS Program. Read it, and become familiar with all the options and limitations of the Get Away Special. It will save you time in the long run.

DEVELOPING AN PAYLOAD CONCEPT

The Get Away Special begins with an idea. A realistic and, more importantly, an achievable idea that has merit. There are some considerations to take into account before you develop an idea:

- ◇ WHAT ARE MY RESOURCES?
- ◇ WHAT CAN I AFFORD?

Resources include education, experience or background in the area of your experimentation; adequate tools, hardware, and/or access to a machine shop; time to work on all the associated requirements for your payload; money to afford the project costs; and help from friends and family, after all, you might as well share the fun (and the work). Stick to what you know. There are an infinite number of possibilities for GAS experiments. First, consider your own area of expertise and how it can be applied to space. Then consider the environment of space, specifically weightlessness. Consult text and reference books for principles that rely heavily on the force of gravity, availability of air or the filtration provided by the Earth's atmosphere. A simple change of the environment can result in a dramatic change in your experiment findings. Also, consider the needs of everyday human existence, many things that we take for granted will not work in space, and therefore must be replaced. After the first flight, you will have an insight on the needs of other GAS users and can develop support equipment for various applications.

There are associated costs above and beyond the launch costs for your payload. Whether you build, buy or lease experiment hardware, it will represent the second major cost of your payload. Support equipment such as batteries, controllers, sensors and the like, all must be space qualified (meaning that they are very expensive). The purchase of space qualified products and services can cut a mighty chunk out of the project budget. As I mentioned previously, engineering is very expensive and, in some cases, can be avoided. Many GAS users have sought the assistance of local aerospace companies. Remember that your community is an additional resource worth tapping.

PUT YOUR PROJECT IDEAS TO THE TEST

- **Do you like it?** Be kind to yourself. There's no fun in working on a project that doesn't please you.
- **Has it been done before?** If so, what happened? Be sure to examine all the difficulties experienced by the previous experimenter(s). Research your idea. Try not to reinvent the wheel or, even worse, repeat someone else's failure.
- **What will it cost?** Might as well know in advance if you are going to exceed your financial capabilities.
- **Is it the best choice?** Perhaps there is another idea that is more suitable for you.
- **Is there available technology to support this type of payload?** If not, now is the time to look at another payload idea.
- **Will NASA allow you to fly it?** It all comes down to common sense really. Stick to legitimate research and development. NASA is not too enthusiastic about publicity stunts that exploit the Space Shuttle. None of us are.
- **Is it safe?** If not, good luck. NASA will require you to safety qualify your payload.
- **Do you have any experience in this field of research?** Your payload does not need to solve the problems of the world. Develop a project that you can understand. When in doubt, remember:

STICK TO WHAT YOU KNOW.

All of these variables increase the cost of the Project. How dramatically depends on how complex your experiment payload becomes. There are other associated cost for items such as NASA's GAS payload optional services; these include the Motorized Door Assembly, special payload preparation, special handling, etc. Also, there are extra costs such as travel, telephone calls, stationery supplies and reference texts which are fairly inexpensive but add up over a period of time.

PROJECT PLANNING

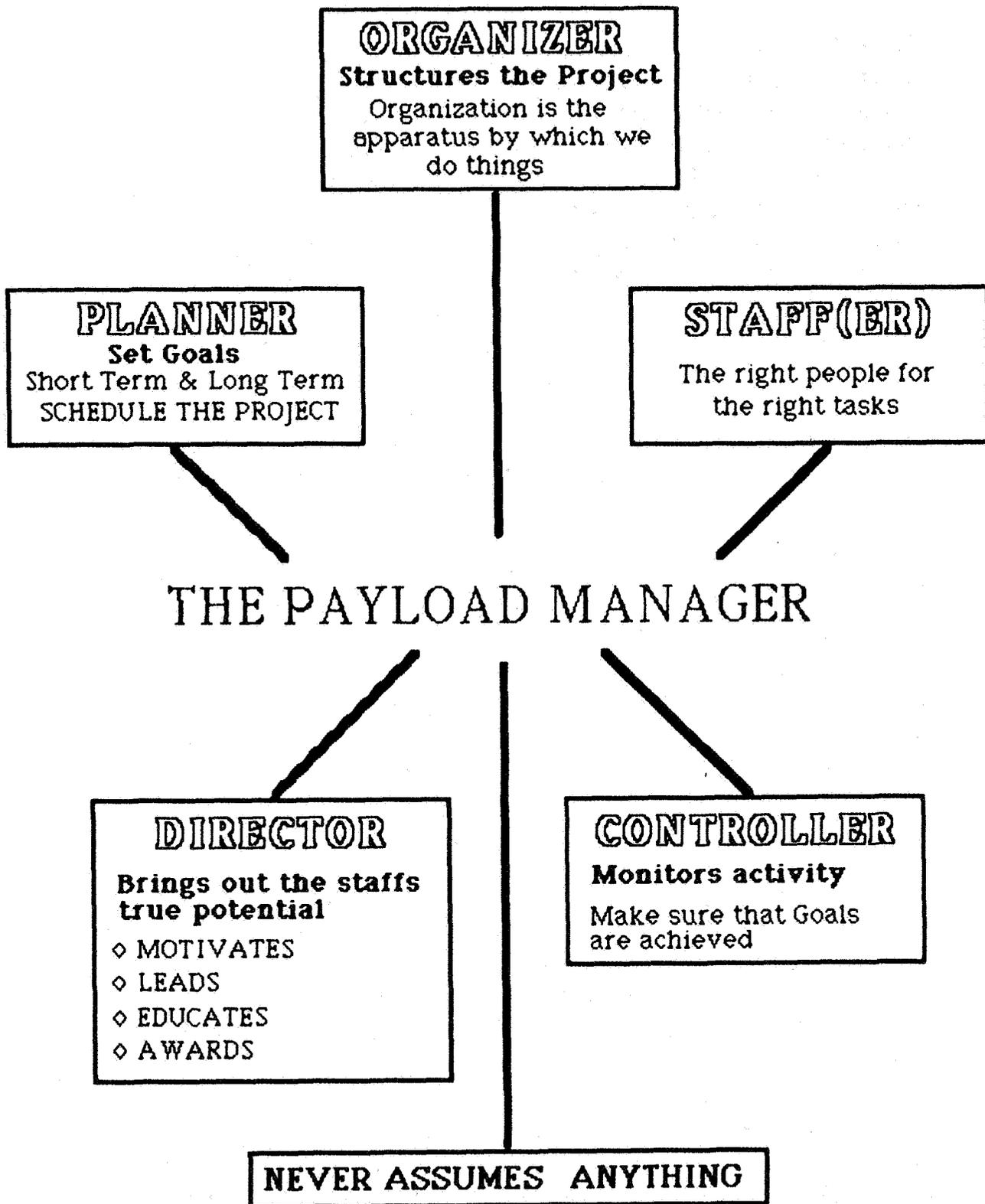
Once an experiment has been chosen, a project plan is the next item of order. Organization is essential; also, knowing "what to do" and "when to do it" can make the project run more smoothly. Before writing the project schedule, consult your GAS user's handbook for important information regarding the flow of documentation and information with NASA. If you are coordinating a group, be sure to delegate duties appropriately and to each person's interest. If you are working on a GAS project as an individual, pursue one aspect of the project at a time; it is all too easy to run circles around yourself and not get anything done.

DOCUMENTATION

Document all aspects of your project. Keep a log book of important developments and events. This provides a means of following the progress of the payload. Documentation will help you keep track of what's going on, especially when the pace of the project picks up. All experiment ideas and other payload concepts should be documented as well. As you become more familiar with the system, you will develop all sorts of ideas for projects to come.

THE FUTURE OF THE GET AWAY SPECIAL

In view of the many challenges presented by today's Get Away Special Program, we can easily compare these difficulties with those of the early personal computer industry. Imagine what it would be like if you wanted to write a simple program that would display seasonal star charts; but first, you had to build a computer on which to write it. This is what the personal computer world was like over a decade ago. Today, however, PCs are available everywhere and there are programs that help you develop programs. The personal computer has become "user friendly". The same can hold true for the Get Away Special. The greatest challenge is continuing the growth of the program so that, in the future, there will be an excess of space qualified hardware and qualified experts; thereby making the GAS Program accessible for everyone to use.



THE PAYLOAD MANAGER

To date, the role of the payload manager has never been very well defined beyond "...the one who has the responsibility for the development of the payload; while being the single point of contact with NASA for the exchange of technical information." The definition expands even more depending on the size of your group and the complexity of the payload. For a university program, such as San Jose State's, the GAS project is managed similar to the way that NASA manages it's projects. We have principle investigators who are in charge of researching and documenting the experiment operations; payload engineers who design the hardware components to support the experiments; and a support staff to work with these key people. A technical advisory board was established to obtain feedback on our design concepts and method of experimentation. They keep us from "reinventing the wheel". These people are the project group's greatest resource. It is up to the payload manager to coordinate their activities.

Communications are essential. Communicate with NASA as well as your people. Keep everyone posted on the status of the project. Motivate the staff and be sure to let them know how important they are to the project. The payload manager must also be a good listener. Know your staff and be aware of their personal goals. Be Objective, and most of all - be fair. Make sure that your project has an identity and a purpose. This was the driving force behind the successful Apollo Program. Everyone involved in the project shared the same feeling of pride and commitment because they knew that they were an integral part of the overall program.

The payload manager is also a controller. He not only must set goals, both short term and long term, but make sure that they are achieved. Making a plan such as this is easy. While scheduling, figure how much time a particular exercise should take, and then double it. Do this for each portion of your project. It is more satisfying to stay ahead of schedule than to always try to catch up.

The payload manager is also a director. Know your staff before assigning them duties. Put them where they want to be in order to bring out their true potential. Finally, never assume anything. Monitor activity regularly and prevent problems by approaching questionable items right away. A structured program is a well run program. It will help you achieve your ultimate goal more timely and efficiently - sending a payload into space.

CONCLUSION

This paper reflects the philosophy of SRDO's GAS Project management at San Jose State University. I am almost certain that there will be changes along the way that will refine and improve our operations for the better; and of course, we will be sure to share them with everyone at symposiums to come. When the Shuttle leaves the pad at KSC, it not only carries our modest payload; it carries us as well. The thrill of seeing the Shuttle takeoff is well worth the long hours of laboring over books, technical notes and the payload itself. Many people will never experience the emotional high of actually sending a payload into space; however, it is in our best interest to get as many of them to try.

Let us not forget that the Get Away Special Program is still a government program; being so, it is subject to the same political realities that have ended similar NASA programs of equal worth. It's a awful thought but, regretfully, it is a possibility. The GAS Program's future is dependent on the amount of use and interest generated from people such as ourselves. We can ensure its growth by spreading the word, sharing our resources and finding new solutions to the problems and deficiencies facing us today. There is a pressing need to help GAS users, who hold reservations, get into space. Those of us who work through organizations, or educational institutions need to establish an ongoing program within these entities in order to make this type of space research a more permanent activity for years to come. It will certainly be a great day when we are able to pack GAS experiment payloads into each Shuttle flight; thus, ensuring the longevity of the overall program.

The real benefit will come in the future, in the form of low cost user programs that utilize the Space Station and other Earth orbiting platforms. Although we have come a long way in the last few years, I believe that the true potential of the Get Away Special has yet to be tapped.

EXPLORER SATELLITES
EXPERIMENT DESIGN
RADIO TRANSMITTERS
PAYLOAD INTEGRATION

SPACE SHUTTLE MISSION SIG
DOWNLINKING
REAL TIME OPERATION
THERMAL SENSORS
VIBRATION TESTS
SITES: MARCE ET

D10-12

N86-27307

89631

GET AWAY SPECIAL
EXPERIMENTER'S SYMPOSIUM
OCTOBER 8-9, 1985
GODDARD SPACE FLIGHT CENTER
PROJECT EXPLORER GAS #007
MARSHALL AMATEUR RADIO CLUB EXPERIMENT (MARCE)

EDWARD F. STLUKA, W4QAU
MARCE PRINCIPAL INVESTIGATOR

ND 7-26801

INTRODUCTION

The effort involved in preparing the PROJECT EXPLORER, GAS #007 Experiments Package for the STS-41G flight, yielded a close knit team experience. The mission operations phase provided the "hands on" experiences which the space crews witness daily. This could not have happened without the cooperation and help of the GAS teams at the MSFC, GSFC, JSC, KSC, JPL, DFRF, the AIAA, the Alabama Space and Rocket Center, the University of Alabama-Huntsville, the Alabama A&M University and the many amateur radio operators and commercial organizations that supported the GAS #007 and MARCE efforts.

The Alabama Space and Rocket Center Director, Edward O. Buckbee, paid the GAS canister fee. Konrad Dannenberg is the GAS #007 Project Manager. The student science experimenter Principal Investigators are: Arthur Henderson, Experiment #1, Materials Solidification, A&M University, Huntsville, AL; Guy Smith, Experiment #2, Plant Biology, University of AL, Huntsville, AL (UAH); Jonathan Lee, Experiment #3, Superconductivity Crystal Growth, UAH. Experiment #4 is MARCE. See NASA Conference Publication 2324, August 1-2, 1984, pages 69-76.

The student experimenters (long graduated) have maintained interest and endured to keep the experiment team together and to complete their experiment.

Polls have been taken at the Project Explorer meetings regarding flying without the radio experiment transmitting. The radio downlinks require extra coordination and are sensitive to certain payloads. The poll results were unanimous. The radio down links are vital in providing data on the health and status of the total experiments package, in real time, during the flight. The amateur radio operators, prepared to receive the downlinks and OSCAR-10 relays, revealed that there was enormous interest throughout the world, to participate. This sets the stage for the reflight opportunities which the GAS PROGRAM has kindly provided.

BACKGROUND

Major activities, pertinent to the STS-41G flight preparations by the GAS #007 Team and Support Groups, are listed below.

The Functional Test or Simulated Flight Test was conducted for 120 hours, starting March 3 and was completed March 16, 1984. This was the first time the total experiments package was put through an integrated test. A fully charged SPAR battery was used. This battery had flown a SPAR mission in June 1983, returned and was put in cold storage. The SPAR battery has a 120 day activated life. No problems were encountered. The test was a great success. A cassette recorder recorded the voice messages during the three eight hour downlinks. Information and data is sent to the MARCE memory every 10 minutes, during the mission, after power-on. Strip chart recorders measured the battery voltage and current and the transmitter input voltage and current. This test confirmed the ambient prediction parameters and the experiment package integration.

The Thermal Test was conducted for 120 hours, starting April 7 and was

completed April 13, 1984. A new fully charged battery was used, although it was two years beyond the two year shelf life. The GAS Thermal Design Summary Report X-732-83-8, July 1983, revealed the maximum and minimum temperatures reached during the STS-3 GAS Flight Verification Payload. This was both a GAS flight test and qualification of the GAS carrier system. Table 4, in the report, shows the maximum and minimum internal canister temperatures were +35 and -3 degrees C. Using this data, the GAS #007 thermal test limits were set at +45 and -15 degrees C. At the -15 level, it was found that the Experiment #2 nutrient solution froze. The pump did not operate, when commanded. As a result, the following changes were made. The heater capacity was increased three times and the heat up time was increased by four. The nutrient pump was set to operate at four hours after GAS #007 power-on instead of 1 hour. Likewise, the Experiment #3 heater capacity was doubled. All other systems worked without any anomalies. The thermal test margins qualified the GAS #007 system, thermally.

The EMI test was conducted on June 13, 1984, in a screen room. The MIL-STD-462A, RE02 test was conducted on the total package from 14KHz-10GHz. A trial run was made with the GAS lid off. The radiated emission, electric field readings were within acceptable levels. With the lid on, the readings were ambient, like an empty room. The GAS canister was provided by the GSFC, for the EMI test.

The Vibration Test was conducted July 18-26, 1984. Normally, a package to be tested has a functional test performed prior to and after the vibration test. GAS #007's system was ON during all three axis of vibration. The intent was to find any intermittent connections. None were found. The systems worked with no anomalies. Recommended composite vibration criteria, for GAS payloads-based on STS-1 through STS-5 flight data: X axis = 4.9; Y axis = 7.8; Z axis = 8.0 grms. Test data shows that GAS #007 received the following: X = 4.935; Y = 7.861; Z = 8.204 grms for 60 seconds each axis. The antenna was attached to the vibration fixture.

A Stress Analysis, using a Finite Model of the GAS #007 internal structure, was created and run on SPAR. The model was subjected to the following accelerations:

<u>TYPE</u>	<u>X-DIRECTION(G's)</u>	<u>Y-DIRECTION(G's)</u>	<u>Z-DIRECTION(G's)</u>
<u>RANDOM</u>	14.7	23.4	24.0
<u>QUASI-STATIC (NASTRAN MODEL ANALYSIS)</u>			
<u>LIFT-OFF</u>	+3.8/-6.4	+/-6.5	+/-5.6
<u>LANDING</u>	+/-5.9	+/-5.0	+7.1/-4.9
<u>WORST CASE COMBINED TOTAL</u>			
	21.1	29.9	31.1

32 G's and 516 degrees of freedom were used to cover all canister orientations. The lowest frequency (47 Hz) found relates to the component mounting plate, in the direction perpendicular to the plates surface. The Shuttle's resonant frequency of concern is 3 Hz. The resulting stresses were subjected to the appropriate factors of safety (fos), for untested hardware, per MSFC-HDBK-505A. These are 2.0 for Ultimate and 1.25 for Yield. All structures have positive margins.

A Containment Analysis was performed on all of the components within GAS #007. All components were found to be adequately contained, which eliminated all

of the experiments, inside the canister, from the requirements of a fracture mechanics analysis.

The antenna is the only item attached outside the canister. The antenna is a polarized rod dipole consisting of two rods welded to two channels, which are separately welded to a base plate. The antenna structures have high margins of safety, when subjected to 55 G's, in all axis; the antenna rod = 14.4 fos, bending stress mode and the antenna base = 9.5, bending. It was determined that three of the four fasteners are adequate to support the antenna. If one weld fails, tuning brackets between the antenna halves will restrain the broken half. The antenna was, therefore, exempt from a fracture mechanics analysis. A load test to 2.0 times limit stress was performed on the antenna, with no anomaly.

EMI tests, on the 5 watt Motorola Transmitter, consisted of MIL-STD-462A, CE06-3, where emissions were measured from 2.0 MHz to 10.0 GHz. Emissions at 1,330, 1,770 and 2,180 MHz were above the -55 dBW limit. No broad band signals were observed - all were narrow CW signals. To suppress the three signals, a HP 630A low pass filter was added. All signals, above 860 MHz are now no higher than -64 dBW. A broadband transient emissions test was made by keying the transmitter, on and off, several times. The only large signals found were near the transmitter carrier, which uses a phase lock loop to determine the transmitter frequency. Signals observed were the transmitter carrier, as the loop locked. The emissions are present only for a few milliseconds, at the beginning of each transmission.

MARCE Antenna Tests were conducted using a GAS canister mockup, a HP 4191A Impedance Meter, a HP 85 Computer and a plotter. The computer program swept the impedance meter between 400 and 500 MHz. Reflection coefficient (Rc) was converted to Voltage Standing Wave Ratio (VSWR) by:

$$VSWR = \frac{1 + R_c}{1 - R_c}$$

The minimum Rc measured (0.2084) occurred at 435.033 MHz, the transmitter frequency and equals a VSWR of 1.52 to one. Tuning the antenna, by adjusting the shorting bars (tuning stub) distance, yielded a 1.2 to one VSWR, for the flight antenna #1. A transmission line length sensitivity test was conducted by adjusting the transmission line length, between the antenna and transmitter, over a range of 0 to 1.5 wavelengths. A one wavelength variation is required to simulate all possible impedances. The transmitter, as observed on the spectrum analyzer, was stable at all times during the test. The antenna and transmitter performed well. The conclusion was that the transmitter should have no problem completing a GAS #007 mission.

A MARCE Link Analysis was conducted, using an antenna gain = 1.7 dB, path loss = -151.4 dB (1,266 miles), receiver gain = +10.0 dB, polarization loss = -3.0 dB, receiver bandwidth = 20.0 KHz, receiver noise temperature = 290 deg. K, KBT = -131.00 dBm, carrier to noise = 5.0 dB and cable loss = -0.25 dB, with the following assessment:

<u>TRANSMITTER POWER</u>	+36.02 dBm(4w)	+30.00 dBm(1w)	+25.82 dBm(0.382w)
<u>LINK MARGIN</u>	+19.07 dB	+13.05 dB	+8.87 dB
<u>EFF. RADIATED PWR</u>	5.58 w	1.40 w	.53 w
<u>PWR DENSITY @ 1Mtr.</u>	.44 w/m ²	.11 w/m ²	.042 w/m ²
<u>FIELD STRENGTH</u>	12.94 v/m	6.48 v/m	4.00 v/m

Orbiter limit at 435.033 MHz = 4.00 v/m

During all testing, on GAS #007, the transmitter system never failed to respond to a turn-on command.

STS-41G CONFIGURATION

Figure 1 shows the student experiments. Note the battery vent line connections into the GAS provided redundant pressure relief valve assembly. The coax cable connector is connected to the lid coax feed through. The internal support structure is a rounded plate (1/2 X 19 3/4 in. Dia.), with a machined rib. The rectangular experiment mounting plate (1/2 X 27 3/4 X 19 3/4 in.) is attached to the machined rib, on the plate. Both plates are Aluminum 2219-T87. The mounting holes are on 2 inch centers over the total plate area. 1/4 X 24 303 cres bolts, with 303 cres Keenserts, are used to attach the experiment hardware to the plate. The heater batteries, around the experiment #2 crystal chamber, are to keep the potassium cyanide solution above freezing.

Figure 2 shows the MARCE experiment #4. Pressure transducer P1, measures canister pressure, thermisters T5, mounted on the DC-DC converter, measures canister temperature and T6 measures the SPAR battery case temperature. The two malfunction thermal switches, mounted on the battery brackets, provide safety control switch closures, should a battery cell short circuit heat the battery. The 10 dB attenuator is connected between the transmitter output and the low pass filter input. The fuse holders, on the electronic support assembly, protect the positive and negative power circuits. The double shielded cables are attached to the plate with self sticking pads. The battery mounting bracket was added to make sure the battery cells were not inverted during launch or landing. Quality Control stickers were attached after final inspection.

Figure 3 shows the battery vent line installation.

In Figure 4, the flight antenna is being disassembled. The antenna coax must be connected to the lid feed through connector first. The antenna is then assembled around the coax. Next, the coax is connected to the antenna and final installation of the antenna mount is made to the lid, after the experiment package is installed into the GAS canister.

Figure 5 shows the experiment #2 camera and crystal growth chamber installation. Figures 1 through 5 pictures were taken August 6, 1984, immediately prior to GAS #007 installation into the flight canister.

FLIGHT OPERATIONS PREPARATIONS

Extensive coordination was required to inform the amateur radio community of the timelines, locations of the downlinks, how to read the down link messages, the RF link parameters, ground track data for antenna pointing, OSCAR-10 tracking and receiving parameters and data handling techniques. Continual updates were necessary, due to the Shuttle schedule changes and the resultant timeline changes. The American Radio Relay League (ARRL) provided the service of being the link to the amateur radio operators, around the world, for information and updates. The American Radio Satellite Corp. (AMSAT) provided the information link on the OSCAR-10 relay information, to the radio community. The ARRL and AMSAT members, plus Hams in general and Short Wave Listners are the volunteer ground stations for receiving the experiment data. The Johnson Space Center Amateur Radio Club (JARC) provided updates and general information, on regularly scheduled nets, to the world amateur radio community. The JARC also provided Orbiter tracking data and charts, on timeline and position information. This allowed the radio community to know the exact time and location for the downlinks over the Orbiter ground track, around the world.

STS-41G FLIGHT

ORIGINAL PAGE IS
OF POOR QUALITY

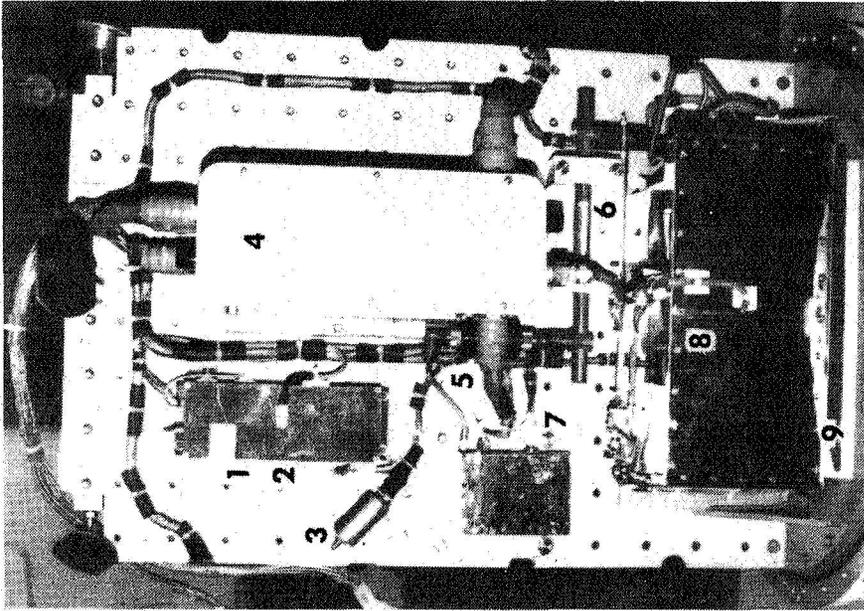


FIGURE 2. MARCE RADIO EXPERIMENT

1. DC-DC CONVERTER
2. T6
3. P1
4. ELECTRONIC SUPPORT ASSEMBLY
5. 10dB ATTENUATOR
6. LOW PASS FILTER
7. 5W TRANSMITTER
8. T6
9. BATTERY

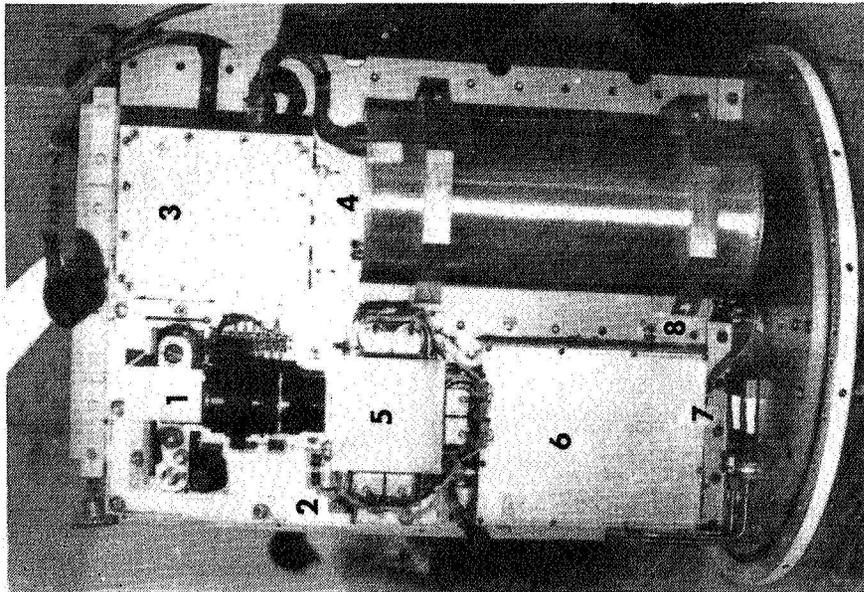


FIGURE 1. STUDENT EXPERIMENTS

1. CAMERA
2. HEATER BATTERIES
3. EXP. 1 ELECTRONICS
4. OVENS 1 & 2 CASE
5. EXP. 3 CRYSTAL CHAMBER
6. EXP. 2 ROOT CHAMBER
7. BATTERY PRESSURE RELIEF ASSEMBLY
8. LID FEED THROUGH

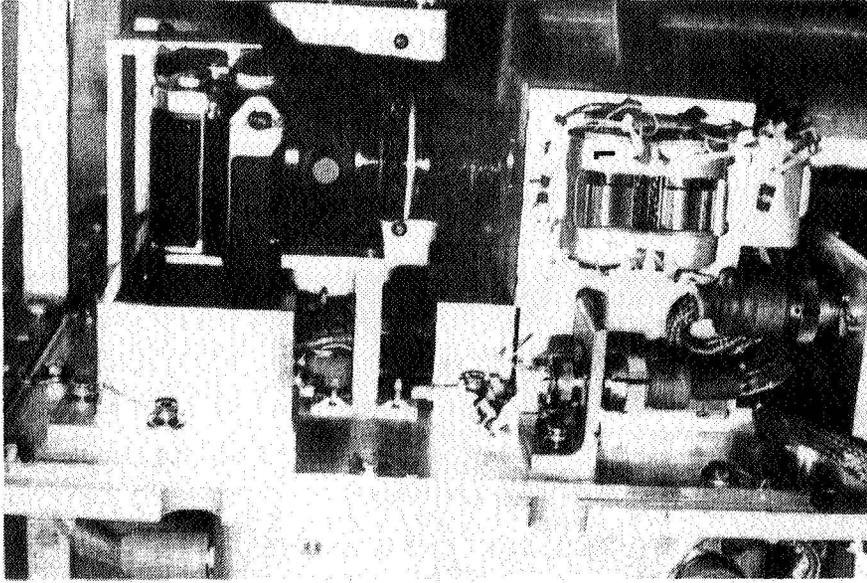


FIGURE 5. EXPERIMENT 2 CRYSTAL
GROWTH CHAMBER AND
CAMERA INSTALLATION

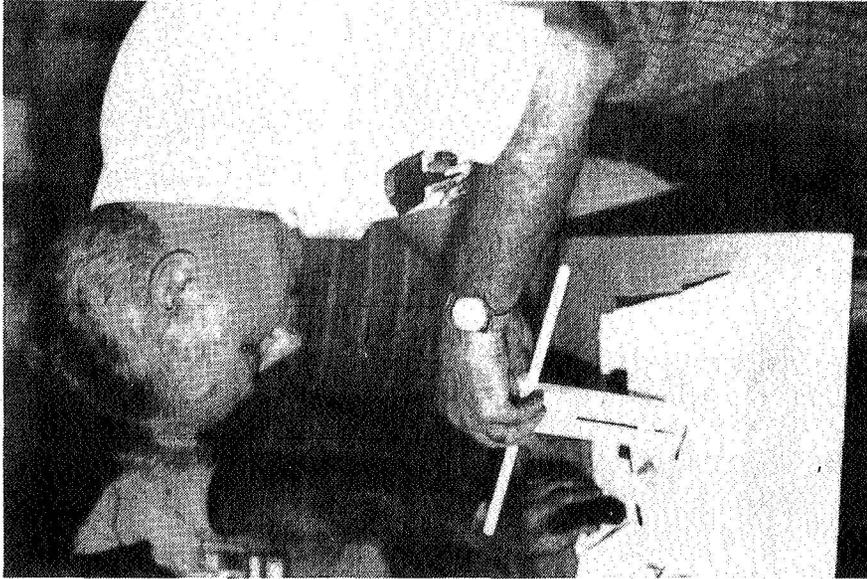


FIGURE 4. MARCE ANTENNA INSTAL-
LATION

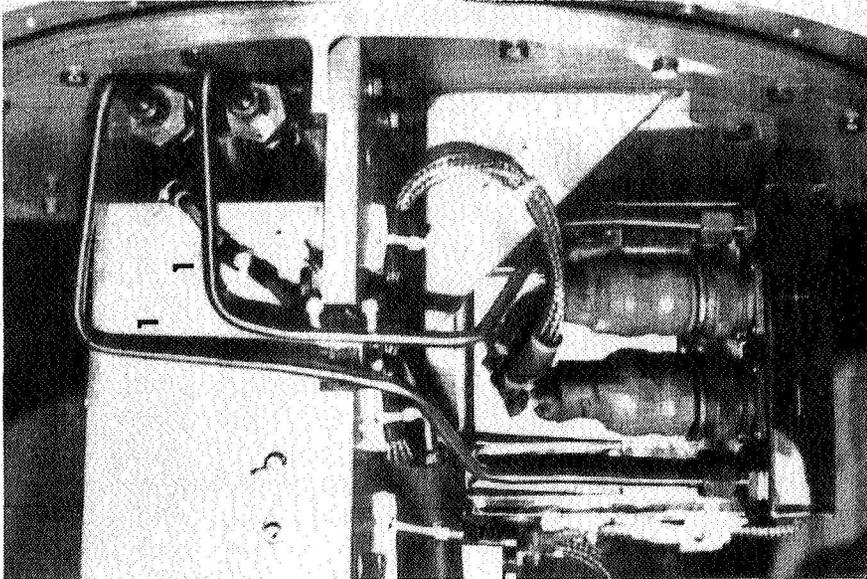


FIGURE 3. 28VDC BATTERY INSTAL-
LATION

1. HEATER BATTERIES

1. VENT TUBES

GAS #007 completed the 7 day mission on Challenger (October 5, 1984 launch) in the starboard, bay 6, forward position. The 57 degree inclination and an earth facing mission were ideal conditions for the radio downlinks. The RF power was limited to 0.5 watts due to the 4 volts per meter payload to payload interface limitation. A 10 dB attenuator and slight adjustment of the transmitter output produced the 0.5 watt RF level. 5 watts would provide a comfortable margin for extra distance and for OSCAR-10 relay opportunities. Since the mid and eastern United States would not have any direct downlink passes, OSCAR-10 was the only alternative. There was high controversy as to whether an OSCAR-10 relay could be completed. There was a large group of radio amateurs around the United States, Canada and Ireland that were on the OSCAR-10 relay frequency (145.972 MHz) during the October 6,7 and 8, eight hour, downlink periods.

An October 4, 1984 cablegram, from Dr. John Kennewell, Principal Physicist, Learmonth Solar Observatory, Australia, requested that GAS #007 be turned ON two orbits (3 hours) early, to ensure maximum USA & Australia coverage. Some very interested and capable Australian experimenters wanted a chance to participate in this exciting venture. No orbits are favorable to receiving MARCE, in Australia.

Likewise, in a letter from Hans Ngfzger, HB9AQZ, Kloten, Switzerland was very upset that the European hams would not be able to receive MARCE.

The GAS #007 power was not turned on, due to an operational error.

STS-51G REFLIGHT OPPORTUNITY

The post flight testing revealed no anomalies in GAS #007. Therefore no changes were made, although some were considered, to improve experimentation. 5 watts RF transmitter power output was approved for this flight. Transmit times included good passes over the USA. Only two SPAR batteries remained, for this flight. After activation of the primary flight battery, a ground terminal to case leakage resistance was found, indicating a cracked cell case. At about the same time, the back-up battery showed a lower than normal voltage on one cell. The decision was immediately made to cancel GAS #007 on STS-51G. Previous SPAR battery experience, as noted in BACKGROUND, above, indicated such batteries could perform without problems.

REFLIGHT PLANNING

A new battery is planned for the reflight. A 50 AH silver-zinc battery will be installed, which provides 150 percent more power. The eight hour down link is being changed to transmit voice data once each minute instead of every four minutes. Heater power, on experiment #2, will be increased, for added confidence. Since STS-61B is only a 5 day mission, less than 120 hours are available, for experimentation. With a higher capacity heater, Pump A will be turned ON one hour after GAS #007 Power ON, for maximum growth time. A warmer chamber would expedite root growth.

Reflight approval has been received. GAS #007 is now assigned to STS-61B, scheduled for launch on November 8, 1985.

CONTRIBUTORS

The following companies and individuals made significant contributions of hardware and effort to MARCE and GAS #007.

MOTOROLA-5 watt UHF MX300 transmitter and a telpager receiver. Jim Warsham,

WA4KXY, Bill Pence, KI4UF and Norman Alexander, VP, Ft. Lauderdale, FL; NATIONAL SEMICONDUCTOR CORP.-2 Data System Module sets. Peter Lami, Interep Associates, Huntsville, AL; RCA-Data System Parts, including an 1802 micro processor system. Ivars Lauzums, Micro Systems, Somerville, NJ; ZERO CORP.-Electronic Support Assembly enclosure. Jay Shorett, Monson, MA; SPAR PROGRAM-20 AH battery; 7.5 vdc converter regulator; measurement sensors; connectors; UNIVERSITY OF ALABAMA, HUNTSVILLE-Fabrication, Assembly, Testing in the Environmental Lab., Guy Smith; MIDWEST COMPONENTS INC.-Thermal Sensitive Switches. John Saling, Muskegon, MI; ICOM AMERICA INC.-IC-271A 2 meter transceiver; IC-471A 70 cm transeiver, for OSCAR-10. Evelyn Garrison, Bellvue, WA; KLM & MIRAGE-2M-22C and 435-18C OSCAR antennas. Evert Gracey, Gilroy, CA; TRIO-KENWOOD COMMUNICATIONS-TS430S transceiver, power supply and speaker; TL-922A linear amplifier. Tom Wineland, Compton, CA; BDM Corp.-GAS #007 transportation to KSC, Stanley E. Harrison, Washington, DC.

VOLUNTEERS

MARCE could not have been completed without the following who responded to the request to contribute their time and talents on STS-41G and STS-51G Flight preparations.

EXPERIMENT MANAGER-Leigh Du Pre', WB4WCX; ASSISTANT EXP. MANAGER-Ed Clark, K4KFH; DATA SYSTEMS & SOFTWARE-Chris Rupp, W4HIY; TRANSMITTER, RF & EMI TESTING, LINK ANALYSIS-Leon Bell, WB4LTT; FABRICATION & PLANNING-BILL RICHARDSON, WA4LRE; POWER, NETWORKS, CONTROL, INSTRUMENTATION-Art Davis, WB4KKA; BATTERY-Al Henry; ANTENNA DESIGN AND TESTING-Reggie Inman and Ed Martin; ASSEMBLY & OPERATIONS-Tom Poole, WA4RRA; Joe Appling, W4WIA; Guy Smith, U of AL; INSPECTION & QUALITY-Wiley Bunn, NO4S; STRESS ANALYSIS-Tim Stinson; MECHANICAL DESIGN-Ken Anthony & Jerry Hudgins; VIBRATION TESTING-Clif Kirby; VIBRATION CRITERIA-Jim Mc Bride; EMI TESTING-Jimmy Rees.

AMATEUR RADIO SUPPORT

ARRL-Bernie Glasameyer, W9KDR; Dale Clift, WA3NLO; AMSAT-John Champa, K8OCL; Vern (Rip) Riportella, WA2LQQ; John McDonald, WB4ZXS; Rich Zwirko, K1HTV; Gordon Hardman, KE3D; Bill Tynan, W3XO; Dr. Perry Kline, K3KP; Doug Loughmiller, K0SI; Art Feller, KB4ZJ; FCC-Robert Foosaner, John Johnston & James McKinney; NASA HQ-Chet Lee; Donna Miller.

NASA AMATEUR RADIO CLUBs, MARCE TECHNICAL REPRESENTATIVES and key GAS #007 support:

NASA HQ-Richard Daniels, W4PUJ; GSFC-Frank Bauer, KA3HDO; Jack Gottlieb; Clark Prouty; Jim Barrowman; John Annen, KB3DN; Susan Oldin; Dennis Roth, N3AZB; Gary Walters, David Miller and Steve Granillo installed GAS #007 into the flight canister. JSC-Dick Fenner, WA5AVI; Gil Carman, WA5NOM, provided all of the Orbiter ground track and timelines, Keplerian Element Sets, Orbiter Ascending Nodes for tracking during the 3 downlinks and the OSCAR-10 relay tracking and timelines, for the primary receiving stations; Dale Martin, KG5U, coordinated the NET CONTROL STATION at the JARC, prior to and during the mission; Art Reubens. KSC-J.D. Collner, W4GNC; Eric Olseen, WB4BNQ, coordinated the SPAR battery activation activities; Al Belsky conducted the primary and backup flight battery activation and tests; Andy Wheeler, WB4ZLW; Haley Rushing. JPL-Jim Lumsden, WA6MYJ; Stan Sanders, N6MP. DFRF-Gary Barr, WA6TWT.

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G-035

STS51-D

WATER BALL COLLISION

June 12, 1985.

Payload Manager:

Kazuo Fujimoto
Asahi National Broadcasting Co.,
Tokyo, Japan

A 3291816
~~AA000000~~

REDUCED GRAVITY
TRANSITION EFFECTS
COLLISIONS
INTERFACIAL TENSION
FLUID-SOLID INTERACTION

BALLS
STAINLESS STEELS
WATER

80075-887

INTRODUCTION

What happens if a stainless steel ball hits a water ball in the weightless space of the Universe? In other words, it was the objective of our experiments in the Space to observe the surface tension of liquid by means of making a solid collide with a liquid. Place a small volume of water between 2 glass sheets to make a thin water membrane: the 2 glass sheets cannot be separated unless an enormous force is applied.

Also, fill a cup with water to the brim, place a sheet of paper on top of it, and turn the cup upside down gently: the water inside the cup does not spill out. It is obvious from these phenomena that the surface tension of water is far greater than presumed. On the earth, however, it is impossible in most cases to observe only the surface tension of liquid, because gravity always acts on the surface tension.

METHODS

Water and stainless steel balls were chosen the liquid and solids for our experiments. Because water is the liquid most familiar to us, its properties are well known. And it is also of great interest to compare its properties on the earth with those in the weightless Space.

Stainless steel balls were chosen the solids because they have perfect rigidity and also because they are affinitive for water. In choosing the solids for our experiments, we primarily sought for materials having the following 2 properties: affinity for water and water repellency. In other words, it was presumed that the collision of 2 different kinds of solids having opposite properties with water would result in the revelation of different phenomena. However, because it was impossible to produce solid balls having perfect water repellency under our tight schedule for the Space experiments, we abandoned the idea of using the 2 kinds of solids.

The dimension of the water ball was determined to be 16 mm in diameter from the volume of the GETAWAY SPECIAL, the dimensions of the apparatus used and the number of experiments on condition that the experiments would be recorded on a $\frac{1}{2}$ " VTR with a CCD color camera.

The dimension of the stainless steel balls is 4 mm in diameter.

Our experiments were aimed at observing and recording the following 3 phenomena:

- (1) A phenomenon that a stainless steel ball passes through the water ball completely.
- (2) A phenomenon that a stainless steel ball shot at the water ball breaks into the water ball but, failing to break through the latter, stays inside it.

- (3) A phenomenon that a stainless steel ball shot at the water ball cannot break down the surface tension of the water ball.

The stainless steel balls were shot out at 20 varied rates ranging from 50 to 1050 mm/second. The rates were those presumed from the results of preliminary experiments on the earth to cause the phenomena in (1) to (3) above.

For details of the experimental apparatus, the reader is referred to PAR.

RESULTS

A total of 20 experiments were made. Although 6 of them turned out to be failures, the other 14 experiments gave very interesting results:

The successful experiments were made at the following rates of stainless steel balls:

50, 65, 120, 140, 155, 175, 200, 210, 230, 300, 380, 600, 900, and 1050 mm/second.

These rates, slightly deviating from the specifications, were calculated from the recorded images.

At the rates of 50 to 300 mm/second, the steel balls appeared as if breaking into the water ball. However, they failed to break into the water ball, but bounced back from the surface tension of water, to revolve around the water ball along the surface of the latter.

At a higher rate of 380 mm/second, the stainless steel ball broke into the water ball, but failing to pass through the surface on the opposite side, was caught on the surface of the water ball, and as at the lower rates, revolved around the water ball along the surface of the latter. It was a very interesting phenomenon revealing the drastic force of the surface tension of water by which the stainless steel ball was drawn back. On seeing this image, a physicist said, "I did not think water appears as if covered with such a strong membrane."

At a rate of 690 mm/second or more, the stainless steel balls passed through the water ball. At the rate of 900 mm/second, however, an interesting phenomenon was observed. The water ball, given high energy by the colliding stainless steel ball, shivered vigorously, and when the stainless steel ball left the water ball, a portion of water was separated from the water ball, to form 2 small water balls. One of the daughter water balls flew away in the direction at 90° to the direction in which the stainless steel ball left.

An incidental erroneous operation of the experimental apparatus resulted in an unexpected phenomenon. The apparatus were originally designed to shoot the stainless steel balls into the center of the water ball. At the shooting rate of 230 mm/second, however, the stainless

steel ball deviated from its orbit, in a way as if scratching the water ball. The stainless steel ball was about to be caught by the water ball for its affinity for water, but the higher kinetic energy of the stainless steel ball overcame the affinity, to tear off the water ball, though its orbit changed greatly.

Professor Fuke at the National Laboratory for High Energy Physics, Japanese Ministry of Education, commented that "this phenomenon is an exact copy of the Rutherford scattering that occurs on collision of an α -particle with the atomic nucleus."

Nuclear physicists are greatly interested in the results of our collision experiments, because the experiments presented the reactions of atomic nuclei macroscopically. For example, they used to think only in mind that when an atomic nucleus composed of protons and neutrons collides with another nucleus, part of the former is attached to and revolves around the latter, while our experiments have visualized this phenomenon.

The stainless steel ball, when caught by the water ball, becomes stable in a state that the one-half portion of it is sunk in the water ball so long as it is observed in the recorded image. Why does it behave that way? When a stainless steel ball has touched the water ball, the surface tension of the water ball applies a force to draw the stainless steel ball into the water ball, while the water ball itself tries to push out the stainless steel ball that has sunk into it with its water pressure. The directions of these 2 forces are opposite to each other, and they were balanced when the one-half portion of the stainless steel ball sunk. This reasoning was demonstrated by calculation. This balancing point does not vary with the specific density of the liquid and that of the metal ball. In other words, this phenomenon has resulted from only the surface tension of water.

CONCLUSION

Analysis of the results of our experiments are still under way. I would like to conclude this report, believing that the imaged results of our experiments in the Space will provide further interesting facts.

REDUCED GRAVITY
SOLIDIFICATION
GRAVITATIONAL EFFECT
SOLUBILITY

SPACE SHUTTLE MISSION
SPACEBORNE EXPERIMENT
FURNACE
ZINC ALLOYS
LEAD ALLOYS

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**ABSTRACT FOR THE 1985
GET AWAY SPECIAL EXPERIMENTER'S SYMPOSIUM**

N86-27309

**TITLE: United States Air Force Academy's Micro-Gravity Research Using
G-0307**

**SPEAKER: Major Glon D Turner
Assistant Professor
Department of Engineering Mechanics
United States Air Force Academy, CO 80840-5861
(303) 472-4396 or x2531**

AH658739

Introduction

The purpose of this discussion is to report on the current materials research being done in micro-gravity solidification and future experimentation planned on-board a space shuttle mission. is REPORTED

The Department of Engineering Mechanics at the USAF Academy is developing a micro-gravity furnace to be used on board the space shuttle. The micro-g furnace will be used to conduct materials research dealing with such topics as immiscible alloy solidification. The purpose behind this research project is three-fold: first, to develop a simple, inexpensive, an easy to use furnace to conduct space materials research; second, to conduct a solidification experiment on a lead-zinc alloy in space that macrosegregates due to gravity; and third, to provide a research mechanism for students to get involved with space materials research.

The Micro-G Experiment

Besides developing this low cost, simple-to-use materials experimental device, we anticipate this solidification experiment in space will provide data to resolve an interesting debate concerning the feasibility of materials processing in space. There are differing views on whether it is possible to solidify useable alloys in a low gravity environment. One view believes that surface tension forces will dominate over the very low gravity forces and will cause each phase to segregate into spheroidal phases. The opposing viewpoint believes that there are sufficient forces to cause mixing due to laminar thermal currents, surface tension mixing, and acceleration forces.

To show macrosegregation effects as well as surface tension effects, we have tentively selected a lead-zinc alloy. This alloy is of a monotectic binary system that does not mix on Earth. Due to gravitational forces, lead sediments from the zinc mixture resulting in a nearly pure lead at the bottom and nearly pure zinc at the top of the alloy. The specimen will be made up of alternating thin cylindrical disks of pure lead and pure zinc. This preliminary set-up will allow certain parameters to be initially defined: distribution and composition of phases, surface area and surface area per volume, and a characteristic shape and radius.

The experimental analysis will include a thermodynamic study of the cooling

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curve from tape recorders. At each phase change, latent heat of fusion will be given off during solidification. The degree of disorder or bonding will be compared with on-Earth experiments to deduce the relative amount of bonding. Micro and macrostructural analysis will be done to determine distribution of phases, composition of phases, any eutectic structure present, characteristic shapes, mean phase radius and surface area. The experimental evidence obtained should determine the degree of bonding as well as surface tension and gravity effects.

This research has been funded in part by the Frank J. Seiler Research Lab, United States Air Force Systems Command, United States Air Force Academy, Colorado.

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N86-27310



"NUSAT 1 - The First Ejectable Getaway Special"

by

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Salt Lake City, Utah

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AA000000

and

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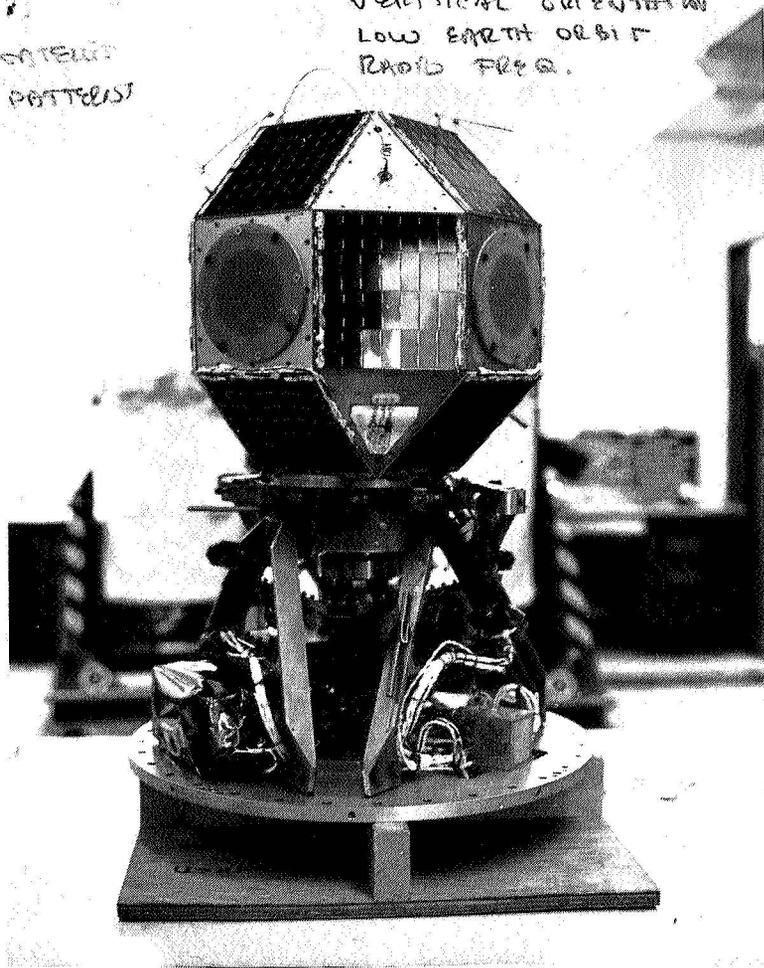
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presented at the

Get Away Special Experimenters Symposium
Goddard Space Flight Center
October 8 & 9, 1985

(1) SPACECRAFT LAUNCHER
GROUND STATION
ORBITAL LAUNCHER
COMMUNICATIONS SATELLITE
ANTENNA RADIATION PATTERN

SPACE SHUTTLE MISSION 51-B
EJECTION
VERTICAL ORIENTATION
LOW EARTH ORBIT
RADIO FREQ.



Abstract:

Four hours, fourteen minutes and forty-two seconds after the space shuttle, Challenger, was launched on mission 51B, a small satellite was ejected into independent orbit from a Getaway Special canister. This event was an exciting milestone in a project conceived over seven years ago. It is hoped that the story of NUSAT (Northern Utah SATellite) will be an inspiration for other experimentors to exploit this new service of the Getaway Special program. This paper describes the purpose and history of the project, the NUSAT spacecraft, the ground station and its operation, and some future directions envisioned by the participants.

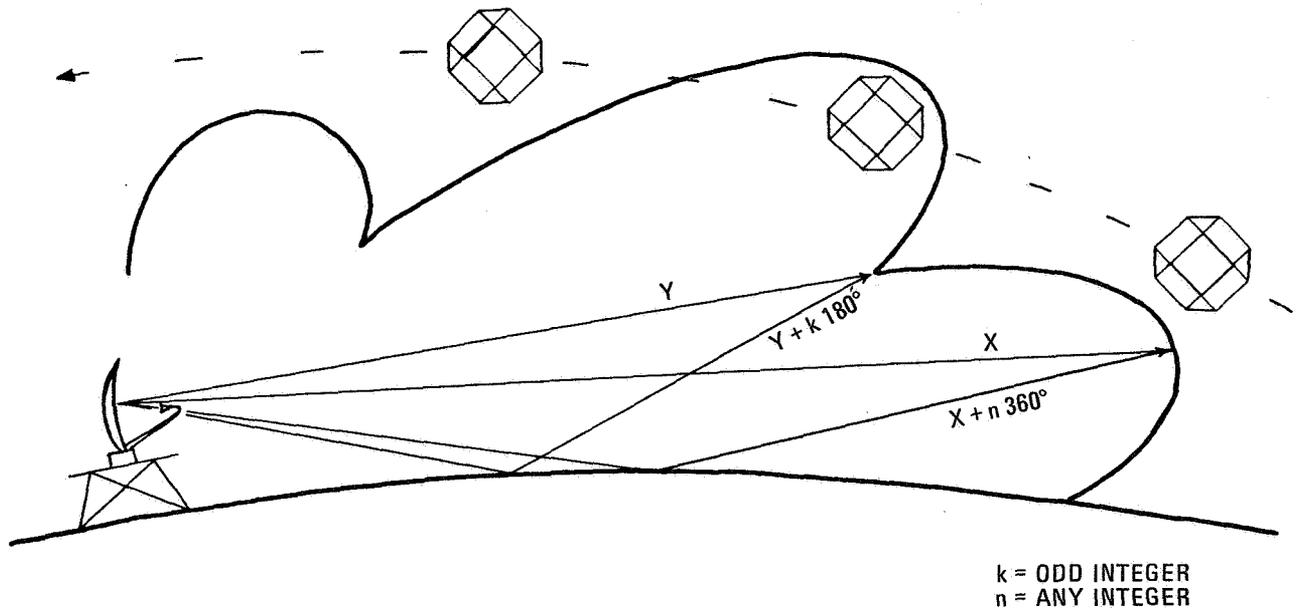
Purpose:

The project was organized with three stated goals:

- to provide an exciting real life educational experience for student participants;
- to demonstrate an efficient technique for optimizing coverage of FAA air traffic control radars; and
- to experiment with the GAS canister as a platform for ejecting small satellites into independent orbit.

Among implied goals also considered were:

- to test the effectiveness of an ad hoc, all volunteer organization made up of individuals from education, industry, and government;
- to enhance public and industry interest and support in Northern Utah as an environment conducive to aerospace activities;
- to facilitate networking of interested aerospace technologists.



The Problem:

The Air Traffic Control Radar Beacon Systems (ATCRBS), are the primary systems used by air traffic controllers to identify and locate aircraft in their control volume. These radars, an evolution of the old military Identification Friend or Foe (IFF) system, provide the controller with azimuth, distance, altitude and identity information for each target, making it possible for them to maintain a safe and efficient flow of air traffic. These radars, in excess of 2,000 stations around the globe, all operate at the same uplink frequency of 1030 MHz and downlink frequency of 1090 MHz.

The ideal radar would provide uniform coverage of a cylindrical volume of air space to a distance and altitude of interest to the air traffic controller. However, low angle reflections in the environment of the radar interrogator cause interference patterns in space, creating nulls and lobes. To combat this problem the FAA began installing radar antennas which reduced low-angle radiation and, therefore, reduced the blind spots in the coverage volume.

Because of their vertical radiation pattern geometry the tilt adjustment of these new antennas is critical. The only way to determine the existence and magnitude of this interference pattern is the use of flight check aircraft at a cost now in excess of \$1,700 per hour. A technique called the "solar" uses the sun as a source of incoherent radio frequency energy to determine antenna tilt angle, but has many drawbacks and cannot determine the interference pattern.

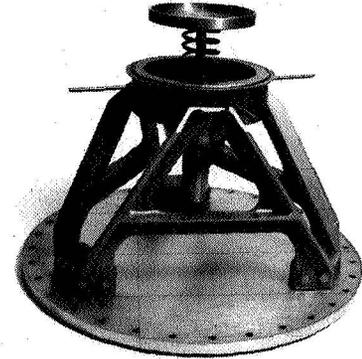
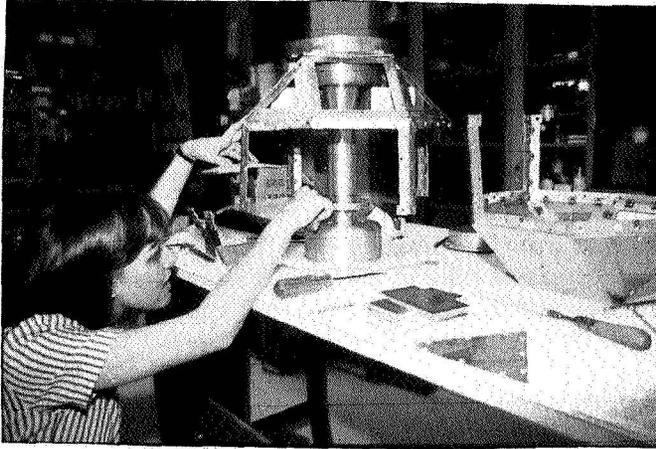
It was proposed that a device in low earth orbit could be used to measure the vertical radiation pattern of these antennas, and that this could be done more completely and at lower cost than any other technique.

Evolution:

The original idea was to have a 1090 MHz orbiting transmitter continuously transmitting a unique RF pulse code at a predetermined duty cycle. Although this would have been a very simple device, there were several problems. Since all airborne transceivers use this same frequency, there was the nagging worry that some failure mode could result in interference with the global air traffic control system. The peak RF power required exceeded the state of the art available at the time. And, finally, the power possible from solar cells on a craft small enough to launch from a Getaway Special canister could not support the orbiting transmitter envisioned.

After several iterations, a consensus was reached to build a receiver that could be remotely programmed to come on at a future time, recognize a particular radar and store the received data (amplitude and time) for later transmission to the ground station.

Another consideration was the effect of the satellite's antenna pattern upon the signals received. Either the antenna system should be isotropic and circularly polarized, or else the spacecraft would have to be stabilized. Several stabilization schemes were considered then rejected, including spin stabilization, gravity gradient stabilization, aerodynamic stabilization and photon pressure stabilization. It was finally decided to build an antenna system as nearly isotropic as possible. This was achieved with six orthogonally mounted L-Band receiver antennas. Each antenna was a circularly polarized, double Archimedean spiral, back-loaded cavity, designed so that the half-power points coincided with the same points of adjacent antennas.



Mechanical structure of the spacecraft (contributed by John Boyer, Weber State College):

The decision to eject an isotropic receiving system from a Getaway Special Canister also constrained the size and shape of NUSAT 1. It would have to be nearly spherical and no greater than 20" in diameter. It was decided to construct a 26-sided polygon with 18 square sides and 8 triangular panels. Six of the 8" square sides would support the L-band radar antennas and the remaining 12 would support solar cell arrays. Four of the triangular panels would support the VHF and UHF communications antennas, two would support Xenon flash bulbs, one a Langmuir probe experiment and one a power connector for ground testing and battery charging.

The satellite was assembled with five major components:

- The central post is a single dumbbell-shaped piece of 7075-T73 aluminum. The top is a cylindrical cup large enough to allow mounting of the top L-Band antenna and rigid enough so that lifting screws can be attached for manipulation in 1 g. The bar of the dumbbell provides the major structural strength and support for the module shelves. The bottom is another cylindrical cup similar to the top, but with the addition of a beveled ring to be held by the V-band clamp.
- The two octagonal shelves of $\frac{1}{2}$ " thick 7075-T73 aluminum provide rigid support for the electronic circuit modules, the battery pack and the outer shell.
- The outer shell of the satellite was fabricated in two pieces from 14 gauge 2024-T6 sheet aluminum, formed into a 26-sided polygon. Each seam and joint is fastened by aircraft type backing plates and nut plates. This outer shell provides support for the solar cell arrays, antennas, strobe lights, Langmuir probe and connector.

Although this structure is exceptionally strong and a good thermal conductor, a major flaw is the lack of accessibility to internal components. Students are now designing a structure that will retain the advantages of NUSAT 1's mechanical design, but provide greater accessibility to all internal areas.

Ejection Mechanism (contributed by John Boyer, Weber State College):

The dimensions of this mechanism were constrained by the volume of the satellite, the canister and the space needed for the ejection control system.

The three "A" frame supports were machined from 7075-T73 aluminum. They support a single-piece, multicavity aluminum structure that houses the ejection spring and plunger, and provides the bevel ring that matches the satellite ring and to which it is held by the Y-band clamp. They were designed to withstand the cantilever stress of supporting the satellite during shipping and launch accelerations.

The spring characteristics and plunger length were designed to allow full spring compression by the weight of a 150 pound satellite in 1 g and to provide an ejection velocity of about 3.5 ft/sec. at \emptyset g.

Four small, steel radial rods were installed below the beveled ring to limit motion of the V-band clamp towards the base by springs after ejection.

The V-band clamp, which has been used before on other launch systems, held the beveled rings together until released by pyrotechnic bolt cutters.

Power System (contributed by Robert Twiggs, Weber State College):

The source of electrical power for the spacecraft is a system of solar cells, twelve panels of 28 cells each, which charge a battery pack via isolation diodes. The 10.5 v, 5 ampere-hour battery pack is made up of five 2.1 v Gates X-cells.

Several weeks before launch the batteries were charged and then isolated from both the solar cell arrays and the satellite electrical systems by microswitches on the base ring. Upon ejection the microswitches closed, connecting the solar cell arrays to both the battery pack and the satellite electrical systems.

Since these systems require ± 15 volts, +5 volts, and the +10.5 volts of the battery, a +5 volt switching regulator and three ± 15 volt DC converters were needed. These voltages are distributed to the various circuit groups via electronic switches under control of the microcomputer. These groups provide the following modes of operation:

- Mode \emptyset : the default, or "sleep", mode, in which only the microcomputer and the uplink communications receiver are powered.
- Mode 1: the RAM or, program loading mode.
- Mode 2: the L-Band data collection mode, in which all radar receivers and associated processing circuits are powered.
- Mode 3: the down link transmit mode.
- Mode 4: the on board sensor monitor mode, in which all light sensors, Langmuir probe and voltage monitors are on.

Although some of these modes, or combinations of them, require much more power than the solar cell arrays can provide (0.6 ampere/array), the duty cycle of these modes is low enough to allow satisfactory operation of the satellite.

Control System (contributed by Chris Williams, Hill Air Force Base):

On-board management of the satellite is under the control of an NSC 800 National Semiconductor microcomputer. This is an eight-bit, Z80 comparable, microprocessor system. On-board memory includes 2K of ROM and 2K of RAM with the microcomputer, and 48K of RAM which are turned on under software control.

The operating system is characterized by exceptional flexibility, with very little program in ROM and most software loaded into RAM from the ground station when needed. This allows the ground system operators to change procedures and variables and to experiment in ways not even considered before launch.

The microcomputer is capable of monitoring and controlling many functions and parameters of the spacecraft. These include monitoring of: battery bus voltage (1), temperature sensors (12), light sensors, for attitude determination (8), Langmuir probe voltage (1) and L-band receiver outputs (6). And the control of: antenna release (1), Xenon strobe lights on/off (1), 48K RAM on/off (1), power group switching on/off (6), radar pulse pair spacing (1), start conversion command (1), as well as analog outputs for setting the radar receiver noise threshold and determining the sun direction.

Upon ejection, the microcomputer was turned on and went through an initialization sequence which included deployment of the communications antennas 18 seconds after ejection and resetting of software clocks. The operating system then entered an "IDLE" program loop which keeps the system in mode 0, polls the communications receiver system for incoming messages (the "COMM" routine) and does an unconditional battery check every 10 minutes.

If upon polling the uplink communications system a valid query message, "WSC return", is present, the control changes to mode 3 and replies to the ground station "NUSAT TO WEBER STATE GROUND". It then enters mode 1 and awaits a program upload for up to 10 minutes. If it receives a valid program it responds "SUCCESSFUL UPLOAD" and then executes the program. If it receives an invalid program, such as one with check sum data errors, it responds "UNSUCCESSFUL UPLOAD" and returns to the IDLE routine.

Every 16 milliseconds a non-maskable interrupt is received from a crystal-controlled clock circuit. The interrupt processing routine then updates the software clock, checks if an uploaded program is being executed and if the COMM routine has been called in the last 10 minutes. If an uploaded program is being executed and it has been less than 10 minutes since COMM was called, control returns to the executing program. But if no RAM-based program is executing or 10 minutes have elapsed, the interrupt processing routine returns the satellite to IDLE.

There are at least three levels of protection against hardware or software failures which would result in loss of control of the on board system. These include:

- the non-maskable interrupt routine described above which automatically returns the system to its IDLE mode if 10 minutes have elapsed since the last call of COMM.
- an eighteen minute hardware reset timer that is reset to zero whenever the COMM routine is executed. In other words, the system would reinitialize from a hardware reset if COMM had not been called within eighteen minutes.
- a battery bus voltage check every 10 minutes which executes a software reset if the voltage falls below 9.75 volts.

Of course, long duration programs such as these which call up L-Band collection during future orbits can remain in RAM and operate for many hours as long as they periodically call up "COMM" to reset the software clocks.

Communications System (contributed by Lee Barrett, Computer Science Corp):

Another paper, "The Northern Utah Satellite (NUSAT) Communications Link" by Lee Barrett, will present a detailed description of the design and evolution of the NUSAT communications system. In the present paper the reader will find a general description of the system used to send control and data messages between the ground station and the satellite. The antenna tracking control system is included as part of the communications system.

The ground station employs two Apple IIE computers. One computer controls the orientation of the antennas by driving elevation and azimuth control rotors according to a predicted plan for each pass of NUSAT over the ground station. The other computer handles the communications duties of uploading programs and downloading data.

The first Apple IIE contains a table of predicted azimuths, elevations and times for the period NUSAT will be in sight of the ground station. This table is generated by another computer from orbital parameters provided by NORAD. When commanded to start tracking by an operator, this computer outputs signals to rotors which control the elevation and azimuth of an antenna array. This array includes a KLM model 435-18C/CS-2 UHF antenna with 12dbC gain and 33° beamwidth, and a KLM model 2M-14C cut for 137.9 MHz with 11dBd gain and 38° beamwidth. Both cross-polarized multiple-element yagi antennas are mounted side by side on a fiberglass crossbar.

The second computer is connected via an RS232 interface to an audio frequency shift keying (AFSK) encoder/decoder. Although designed for a 2400 baud rate, the system usually is operated at 600 baud. The encoder and decoder circuits are connected to the 450 MHz transmitter and 137.9 MHz receiver, respectively. The operator selects the message or program to be uploaded to the satellite from a disc-loaded menu.

The VHF and UHF antenna systems on the satellite are similar in design. Two vertical antennas are mounted at right angles to each other, resulting in no cross coupling. The two antennas are then fed with another 90° phase shift to create a circularly polarized antenna system. The antennas are all flexible whip-type elements that are tied down to Nichrome wire retainers by nylon lines while in the GAS canister. Eighteen seconds after ejection a large current pulse vaporizes the wire and releases the antennas.

The microcomputer is interfaced to the UHF receiver and the VHF transmitter on the satellite via another AFSK encoder/decoder circuit. The spacecraft radios are standard commercial models modified for this application.

The 450 MHz transmitter is a Spectrum Communications model SCT410, with the 7 watt output boosted to 100 watts by an external linear power amplifier.

The 450 MHz receiver is an ICOM model IC-4AT transceiver, with the transmitter section removed. This receiver has a sensitivity of 0.35 uv for 20 dB noise quieting. The amount of doppler shift either the ground or satellite system will tolerate depends upon the level of signal received.

The 137.9 MHz transmitter is a modified Spectrum Communications model SCT110, with an output power of 10 watts.

The 137.9 MHz receiver is a Spectrum Communications model SCR-200.

L-Band System:

This system consists of the circuits needed to receive, recognize and measure the signals from a particular air traffic control radar. There are six separate, identical RF channels, each associated with one of the six orthogonally oriented antennas.

The output of each antenna is amplified approximately 60 db by a broadly tuned amplifier before passing through a 1030 ± 1 MHz multi-cavity filter. The output of this filter is amplified 30 dB more prior to detection. The detected signal, a 0.7 microsecond rectangular pulse, is compared to a noise threshold level generated under software control and sent to a peak-and-hold circuit if the signal exceeds the threshold.

This signal is sent to a coincidence circuit, along with output signals from the other five channels, where it is applied to a pulse pair discriminator circuit. The purpose of this circuit is to determine when radar pulses of a unique time interval have been received. This interval is determined by control software. When radar pulse pairs of the correct interval and sufficient amplitude have been received, the microcomputer then commands the analog-to-digital converter to measure the amplitude of the receiver output signal in the peak-and-hold circuit of each channel at a future time. This time is the predicted time of the next radar pulse, based upon the known pulse repetition frequency of the radar under test.

The microcomputer then processes the six received signals, rejecting those from antennas aimed at the sun and determining the actual pulse amplitude, based upon the signal component seen by each receiver. The calculated pulse amplitude along with the time of the measurement is stored in RAM for transmission to the ground station upon demand.

Auxiliary Experiments:

There are two auxiliary experiments included on NUSAT. The Langmuir probe is designed to measure ambient electron density in orbit. The result is a voltage included in the sensor readings. The two Xenon flash lamps can be commanded on to facilitate optical tracking of the spacecraft.

Operation:

The planned operational strategy is to progress slowly from shorter to longer program operations. The first operation after launch was to send the "WSC return" query at the zero doppler location in the pass over the ground station and receive the reply message. The second operation was to upload a program called "Beddy-bye" which simply returned the satellite to the IDLE mode. This operation was to be repeated a number of times per pass to determine the length of time and doppler range within which the system can reliably operate. After reliable operation of two minutes or more was achieved, then the sensor monitor and data down load were to be executed. However, this last step has not been tried as of June 6, 1985, due to cyclical variations in the reliability of system communications. The next operation will be a test to discriminate one radar signal successfully. When this has been achieved, full radar antenna pattern measurements will be scheduled.

A fully successful operation will follow this scenario: On a pass over the ground station, a program will be uploaded which will command NUSAT to start the L-Band data collection process one or two minutes before appearing on the local horizon of the radar to be tested. For example, to test the FAA radar at Pico del Este, Puerto Rico, the program will start execution while NUSAT was still southwest of Bogota, Colombia. At this time, technicians at the radar will start transmitting a unique test signal interlaced with the normal radar signals.

NUSAT will measure and store the peak amplitude and time of the strongest signal received during each scan of the radar antenna. Five to nine minutes later, after the satellite has descended over the radar horizon, the data collection routine will stop and the data will be held in RAM until commanded to down load on a subsequent pass over the ground station at Weber State College.

It is hoped that the vertical radiation patterns of several radars can be measured during the lifetime of the satellite.

History:

In 1978 the authors presented a proposal to carry a 1090 MHz transmitter into low earth orbit in a Getaway Special canister

In 1982 a group was formed at Weber State College in Ogden including representatives from Weber State College, Utah State University, New Mexico State University, the Federal Aviation Administration, the United States Air Force and several aerospace and electronics companies.

In January of 1983 the concept was still that of a transmitter in orbit. In March of 1983 it was modified to be a transponder replying to radar interrogations. By September of 1983 the final design concept was agreed upon, and system design and fabrication was started.

In August of 1984 the mechanical design and construction was complete and a thermal analysis was performed. This was done by computer model and indicated that in-orbit temperature extremes would be between 44°F and 88°F (excluding the antennas).

An initial vibration test was performed on satellite and support structure components only in the summer of 1984.

Later in the fall, the entire satellite, mounted upon the ejection mechanism in its launch configuration, was vibration tested. In both of these test series three procedures were performed: a resonance search from 0 to 50 hertz, which was negative; a transportation test to simulate the stresses experienced during shipment and launch; and a random high frequency noise vibration test. Based upon the second series of tests, three modifications were performed; installation of foam backing on the circuit board enclosure cover of the computer to prevent boards from shaking loose from their sockets, removal of the brass spheres from the ends of the antennas and moving of the antenna tie-down locations. The latter two modifications were performed to avoid damage to the solar panels by motion of the antenna tips.

Thorough testing and debugging of the electronic systems continued through the winter of 1985 until delivery to the Goddard Space Flight Center in March, at which time the satellite was integrated with the Getaway Special canister and tested by NUSAT and NASA personnel.

In early April the final charge was applied to the batteries and the package was installed in the orbiter cargo bay at the Kennedy Space Flight Center.

The Future:

The success of the NUSAT project was due to the intense dedication and vision of a handful of skilled volunteers, the support of many cooperations, academic institutions, students, government groups, the press and families. Looking back upon the project to date, it is obvious that there were numerous errors, oversights and inefficiencies. Some efforts were duplicated and some things forgotten.

The NUSAT team is now in the delicate process of forming a new organization that, it is hoped, will assure greater efficiency and opportunity to all, while continuing to promote the spirit of volunteerism and camaraderie that marked this project from its birth. This organization is called the Center for Aero-Space Technology (CAST) and is really just a formal coalescing of most of the original participants.

Its Statement of Purpose and Goals is:

"The Center for Aero-Space Technology is a non-profit organization of individuals from industry, education, and government in association with Weber State College. The purpose of the Center is to propose, solicit, design and manufacture useful aero-space experiments, devices or systems, or to support similar enterprises in other Utah schools and organizations.

The goals of the Center are:

- to generate significant, practical and realistic technical experiences for students;
- to provide a local center in which aero-space technologists can connect with others of similar interests, share their expertise with students and associates, and achieve other personal goals;
- to create an environment in Northern Utah which will support aero-space industries;
- and to facilitate public aero-space education in Utah.

It is envisioned that future projects will be modeled after the NUSAT 1 project, in which its goals were achieved through a combination of student and volunteer efforts, donated resources, and contracts from other organizations."

It soon became apparent that organizations can budget donations to projects like NUSAT on an annual basis, but it is difficult to obtain one-time donations on short notice to an informal group. Therefore, it is hoped that CAST can more readily solicit donations for sponsored projects. Also, such projects need the logistics support made possible by association with an educational institution, its physical facilities and contract office.

CAST is already considering at least three proposals to launch satellites into independent orbit, and several projects involving non-ejectable Getaway Special payloads. These include: a standard ejectable vehicle with a reliable communications control and sensor system for carrying other experiments; a NUSAT 2 satellite for the purpose of checking FAA radars; assisting a high school experiment germinating seeds in an artificially created 1 g environment; and an experiment concerning the broadcast of high frequency (27 MHz) signals from a satellite.

Humanity is on the threshold of developing an infrastructure that will allow exploration, commerce and recreation throughout the solar system and eventually to the stars. The NUSAT participants are enthusiastically doing their small part to develop the technologies and technologists to exploit this infrastructure, and they thank NASA and its Getaway Special program for making this possible.

The following corporations made donations of money, material or time to the NUSAT 1 project:

Morton Thiokol, Inc.
TRW, Inc.
Rockwell International Corp.
National Semiconductor Corp.
Apple Computer, Inc.
Sperry Corp.
Boeing, Inc.
Microtech Research, Inc.
Globosat, Inc.
McDonnell Douglas Astronautics Co.
Spectrum Communications
Consolidated Air Freight
United Technology
Maggione Electronics
ICOM, Inc.

Weber State College
Utah State University
New Mexico State University
Federal Aviation Administration
NASA, Goddard Space Flight Center
Hill Air Force Base
AMSAT, Inc.
Western Airlines
Applied Solar Energy Corp.
Pacific Chromalox
South Western Data Systems
Flamenco Engineering, Inc.
Computer Science Corp.
Wall Industries

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MATERIAL SCIENCES EXPERIMENTS UNDER MICROGRAVITY CONDITIONS WITH M*A*U*S

G.H. OTTO and D. BAUM

Deutsche Forschungs- und Versuchsanstalt für Luft- und Raumfahrt (DFVLR),
Köln-Porz, F.R. Germany

D0701453

REDUCED GRAVITY
GRAVITATION EFFECTS
SEDIMENTATION
CONVECTION

SPACE SHUTTLE PAYLOADS
DISPERSION
TEMP. GRADIENT
X RAY
RADIOGRAPHY
LIQUID-SOLID INTERFACIAL CASTING

Abstract

LIQUID-SOLID INTERFACIAL CASTING

Project MAUS is a part of the German material sciences program and provides autonomous payloads for the Space Shuttle. These payloads are housed in canisters which are identical with those of NASA's Get-Away-Special program. The main components of the hardware are: a standard system consisting of power supply, experiment control, data acquisition and the experiment modules containing experiment specific hardware. So far, three MAUS modules with experiments from the area of material sciences have been flown as GAS payloads.

Introduction

NASA's announcement to provide flight opportunities in the Space Shuttle on a low-cost, space-available basis induced the German government, represented by the Minister for Research and Technology (BMFT), to book 25 options in the Get-Away-Special Program. For optimal utilization of these flight opportunities project MAUS (Materialwissenschaftliche autonome Experimente unter Schwereelosigkeit) was initiated with project management assigned to DFVLR and MBB/ERNO company selected as the main contractor. This project is part of the German material science program supporting experiment development for the German Spacelab Missions (D1 and D2), a sounding rocket project (TEXUS) and ground based research.

As a result of a phase A study it was decided that a MAUS payload should consist of a standard system, developed and manufactured by a main contractor and an experiment module containing experiment specific hardware. In general this hardware is supplied by the principal investigator with support being also provided by the main contractor.

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In addition, a MAUS payload should not only be compatible with the requirements of the GAS program but also should be designed for use in dedicated Shuttle Missions as payload complement of partial payloads. In the past years ten MAUS experiments have been flown among others with the structure OSTA-2 (1983) and the satellite SPAS-01 (1983, 1984).

In December of 1979 a contract was placed with MBB/ERNO for development and fabrication of ten MAUS standard systems and preparation of flight experiments was started at several institutions.

MAUS Standard System

A MAUS Payload consists of the experiment mounting structure (EMS), the batteries, the standard electronics for experiment control and data acquisition, the house-keeping systems, and the experiment hardware.

The experiment mounting structure is built of an adapter ring, 6 posts and 2 experiment platforms with brackets. Three different batteries are providing power for experiments and electronics. The total capacity of the experiment battery is 1.8 kWh. The data acquisition system consists of a microprocessor controlled multiplexer unit with digital and analog inputs. The data are stored via an intermediate memory on tape. The capacity is 10 Mbits. To allow for long measurement phases a data reduction system is provided. A detailed description of the standard service system can be found in Ref. 1, a photo is given in Fig. 1.

Experiment Interface

For the experiments to be accommodated in a MAUS module two platforms are available one of which is adjustable in height by 25 mm-steps. The maximum height for the experiments is 400 mm and the maximum mass about 20 kg. Space for 6 cards is available in the standard electronic boxes for the experiment dedicated electronics.

Experiments

It is a policy of the project to assign experiments to a MAUS mission rather late in order to maintain a high degree of flexibility. About one year is con-

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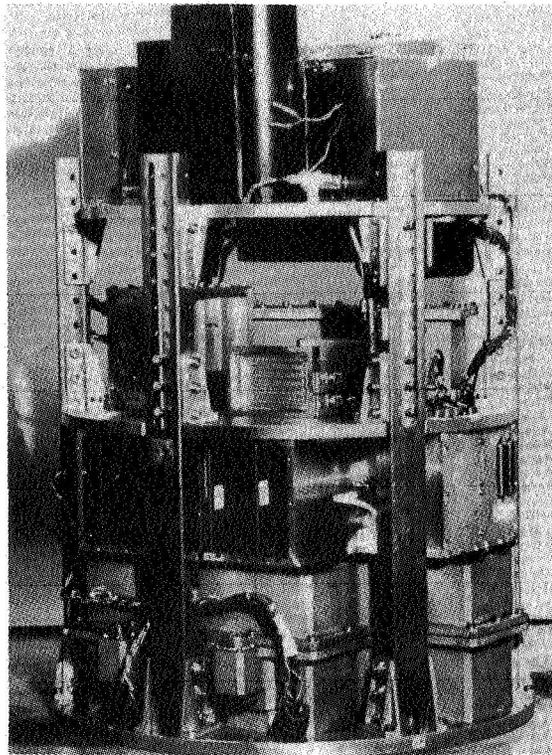


Fig. 1: Total view of MAUS Standard System with dispersion experiment at top during integration. The battery housing can be seen at the lower part of the photo, above the electronics for experiment control and data acquisition.

sidered for the development of the experiment specific flight hardware by the experimenter. Preliminary flight assignments are made from a pool of experiments consisting of microgravity relevant proposals which fit into the limitations set by the MAUS project with regard to power-consumption, volume, weight etc.

The objectives of the different proposals are generally of a scientific nature and do not attempt manufacturing processes in space. It has been one of the results of earlier experiments that "secondary" phenomena become important which are normally masked by Earth's gravity. These effects should be understood first in order to perform meaningful future experiments and to be able to exploit the benefits of microgravity. In this regard MAUS experiments from the area of material sciences so far dealt with the effects of reduced sedimentation, convection phenomena caused by temperature gradients and the behaviour of dispersed particles ahead of a solid/liquid interface.

The first flight of a MAUS payload occurred within the GAS program on Shuttle STS-5 (1982). The objectives were twofold: to test the function of the MAUS standard system and to perform a scientific metallurgy experiment using X-ray radiography as a diagnostic tool. A short summary of objectives follows:

"Stability of Metallic Dispersions"

Investigator: Dr. G. Otto, DFVLR Köln-Porz

There are several combinations of two different metals which show solubility in their liquid state above a certain temperature (consolute temperature) and immiscibility below this temperature. Such a combination consisting of gallium and mercury was used to investigate the dissolution process above the consolute temperature and the time-dependent stability of the resulting dispersion, composed of mercury droplets in gallium. For the first time, this experiment employed X-rays to penetrate the metallic sample and to provide a series of real-time data during different states of the experiment sequence. The sample could be recycled into its starting conditions by repeated thermal treatment. The experiment was planned to run for a duration of 3 days.

After return of the payload to the MAUS team it was found that the experiment had not worked. A failure analysis yielded that a leak in a silver-zinc electronic battery had developed during the several weeks of waiting time on ground. Because of no voltage conditions the electronics of the standard system could not be activated by the "on" signal given by the crew.

The cause of the leak, a simple O-ring seal, was corrected and the same payload successfully reflown on the NASA structure OSTA-2 with STS-7. This experiment then yielded the first X-ray photos from a metallic dispersion by cooling a homogeneous solution into the miscibility gap. Some of the observations made are the following:

- Homogenisation appears to be completed after 4 hours at 190°C. This can be concluded from the constant grey scale value of the sample when measuring across the X-ray film. In the laboratory at least 8 hours are needed for worst case conditions when the heavier mercury is on the bottom of the container.
- When cooling the sample into the miscibility gap with a rate of 30 K/min the precipitation of Hg-droplets occurs rapidly. However, no finely dispersed state with a particle size of about 0.3 mm diameter (resolution limit of the X-ray photos) can be observed. Hg-droplets seem to be generated by heteroge-

neous nucleation at the gallium surface. Droplets seem to be stationary once they achieve the visibility limit and do not show any blurring by movement despite an exposure time of 20 s.

- Supercooling of the melt appears small and if present should be less than 20 °C.
- When cooling into the gap the growth of precipitated droplets is rather fast (Fig. 2). Within one minute (30 K into the gap) the particles have already grown to an average diameter of 0.8 μ m. Anticipating growth by diffusion only, the diameters increase too fast by at least a factor of five. Other processes like convective material transport or coalescence are likely to contribute to growth. A detailed discussion of the data is given in Ref. 2

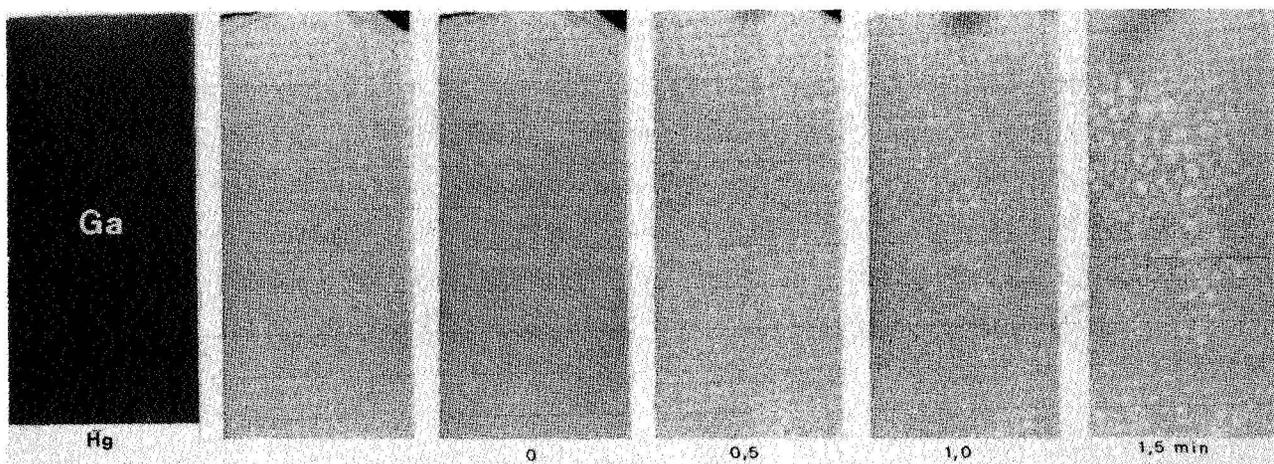


Fig. 2: Precipitation of mercury during cooling into the miscibility gap with a rate of 30 K/min. Shown are from left to right the ground based state, the appearance of the sample after 8 hours at 190°C in microgravity and the cooling sequence. Time interval between each of the last four photographs is 30 sec. Color representation: Ga - Dark; Hg - Light.

The housekeeping systems also provided information about the payload from which the acceleration data taken over a period of three days are the most interesting (Ref. 3). Crew activities and activation of the robotic arm can be seen clearly on the record. It should be stated that g-sensitive runs of the X-ray experiment were programmed to happen during the sleeping time of the crew.

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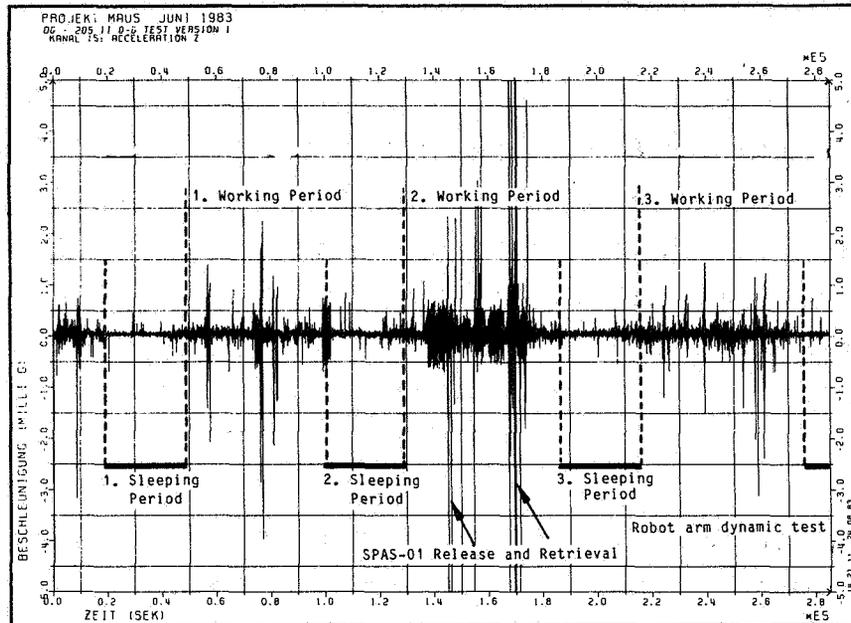


Fig. 3: Acceleration data acquired during the run of the X-ray experiment for a period of three days during mission STS-7. The generation of g-jitter by different crew activities and the activation of the robotic arm during this mission can be seen clearly. Saturation of the g-sensor occurs at 5×10^{-3} g. Each MAUS payload carries its own acceleration sensor which is activated with the "on" signal.

Two additional MAUS payloads to be flown aboard STS 51-G in the GAS program were turned over to NASA on April 26, 1985. A short description of the experiments and their scientific objectives are given below.

"Fundamental Studies in the System Manganese-Bismuth"

Investigator: P. Pant, Krupp Research Institute, Essen

The main objective of this experiment is the synthesis of the intermetallic compound MnBi in an isothermal furnace. The compound forms by a peritectic reaction at 450°C involving liquid bismuth and solid manganese. This kind of reaction is diffusion controlled and requires a longer time for completion as when both components were in a liquid state. On Earth, such reactions are incomplete when both components exhibit different densities and become separated by sedimenta -

tion and buoyancy. The compound MnBi has promising applications as a magnet material because of its high theoretical coercitive strength which so far could not be achieved with ground based specimens.

First experiments on TEXUS (Technology Experiments und Reduced Gravity, a German program using sounding rockets) showed that during melting and solidification under microgravity separation of the components did not occur and forces promoting segregation, e.g. surface tension gradients did not become effective. The MnBi-phase in the flight samples was found in form of micron-sized particles uniformly distributed. The small size is thought to be responsible for the good magnetic properties of the flight samples when compared with the ground base samples.

During the course of this MAUS experiment 8 MnBi samples will be processed in a two-chamber isothermal furnace. Sequentially, each furnace chamber containing 2 samples will follow a pre-programmed temperature profile of heating and cooling. Since the processing time under reduced gravity in MAUS is about 3 hours for each sample the question may be answered whether the peritectic reaction to form MnBi can run its full course. This hasn't been possible with the TEXUS experiments which were performed during only 6 minutes of microgravity conditions.

"Slip Casting Under Microgravity Conditions"

Investigator: Dr. K. Schweitzer, Motoren und Turbinen Union (MTU), München

The process of slip casting employs a ceramic slurry to form complicated shapes of hollow bodies. On Earth, this process is limited in applications because of gravitational influences on the dispersed particles in the slurry. Sedimentation can only be avoided by the use of materials with equal densities or by the utilization of stabilizing additives. However, the latter may be harmful to the desired properties of the slip cast product. Goal of this experiment is to demonstrate with model materials that slip casting is possible in microgravity even with unstabilized suspensions. Using mixtures of powders with different density, grain-size and concentration. For this reason ceramic and/or metal powders are homogeneously mixed in solid paraffin by kneading. Rods of these solid slurries are pressed into cartridges against the ends of porous ceramic rods which are mounted in the lower halves of these cartridges.

During weightlessness thirteen samples of these solid slurries will be melted by heating the upper part of the cartridge in a furnace. Then the slip casting process will be started by additionally heating the lower part of the cartridge containing the suction bodies made of porous ceramic. These will slowly absorb paraffin but not the dispersed particles. The casting process will be stopped by turning off the furnace and cooling of the samples. Solidification of the paraffin will preserve the slip cast layers as well as the residual slurries for later examination on Earth in respect to their structure and particle distribution. This experiment will lead to a better understanding of the slip cast process and evaluate the possibilities for the casting of delicate shapes under microgravity conditions.

Future Planning

MAUS missions will continue with flights of dual payloads using the remaining 22 GAS options. Launch Services Agreements with NASA have been signed. In principle, international participation in this project is possible. The MAUS standard system is also available by MBB/ERNO on a commercial basis.

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THE DESIGN OF FLIGHT HARDWARE
Organizational and Technical Ideas From The
MITRE/WPI Shuttle Program

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ABSTRACT

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The MITRE Corporation of Bedford Massachusetts and Worcester Polytechnic Institute are developing several experiments for a future Shuttle flight. We have standardized upon several design practices for the development of the electrical equipment for our flight hardware. The purpose of this paper is to present some of our ideas, not as hard and fast rules, but rather in the interest of stimulating discussions for sharing such ideas.

ELECTRICAL
ELECTRIC BATTERIES
BATTERY CHARGERS
DATA STORAGE
HEAT TRANSMISSION

SPACE SHUTTLE MISSIONS
PAYLOADS
ELECTRONIC EQUIPMENT
COOLING
VIBRATION TESTING
SOFTWARE ENGINEERING

INTRODUCTION

Since late 1982, The MITRE Corporation and Worcester Polytechnic Institute [1] (WPI) have been working together to develop and build experiments for a future GASCAN payload [2] (6-408). Although the experiments have not been launched yet, in the course of developing the experiments many ideas have been implemented concerning the organization of the program and the design and implementation of the systems necessary for experiment control.

The purpose of this document is to present several of our ideas. We believe that this is particularly important for those who have either never attempted to develop space-flight hardware for the GAS program or for those who are in the early stages of experiment development.

PROGRAM ORGANIZATION AND DECISION PROCESSES

Upon initiation of our program, a Technical Steering Committee (TSC) composed of five WPI faculty and a MITRE engineer was appointed. The responsibilities of this Committee are to insure that: (a) the GAS canister is filled with experiments, (b) the experiments are chosen fairly from those developed, (c) the experiments comply with applicable safety, scheduling, budgetary

and other logistic requirements, and (d) that all decisions relating to the program are made in an orderly manner.

The detailed concerns of the committee have included (a) soliciting project ideas, (b) selecting students to work on the projects, (c) committing WPI faculty to projects as advisors, (d) setting standards for reports, documentation and presentations, (e) acting as a liaison between MITRE, WPI and NASA, (f) controlling the disbursement of funds and other resources, (g) resolving technical issues involving all aspects of the program and of course, (h) setting deadlines for and evaluating the progress of the development of the experiments.

The presence of the MITRE engineer/scientist on the TSC has been of particular help to the program. In addition to providing an industrial perspective on the organization and development of the program, for WPI he functions as the point-of-contact with MITRE. For example, through him all visits, equipment requests, and requests for professional help are directed.

Industrial Interactions

The faculty and students at WPI interact regularly with local corporate engineers. For example, in an effort to develop a low friction mounting system for a custom accelerometer, the student project team members contacted a manufacturer of jeweled bearings and bearing assemblies. After an initial phone conversation, the students arranged for a plant visit to meet with an applications engineer. As a result of their efforts, the electrical and mechanical engineering students not only learned about the use of jeweled bearings, but also convinced the company to donate the necessary components.

In general, each project team has the responsibility of identifying the system design areas they are having problems with and, if they so choose, contacting a local company which they believe can help them solve their problems. The corporations, in turn, have generally been very receptive to the student requests and have committed many hours of staff time toward helping our students.

Design Reviews

In addition to the individual corporate engineering contacts to help with system designs and design reviews, we have had project group design reviews. Most recently, five MITRE scientist/engineers spent a full week at WPI. Each project team had to meet with the MITRE engineers for a full morning or afternoon. During that time the students presented their experimental equipment, calculations, theory and so forth for review. Prior to the meetings the student teams were, as expected, quite nervous about what the engineers might think of their efforts. After the meetings, the students felt that the expert comments from an outside source, the instructional information and suggestions and just as importantly, the praise they received for their work was well worth the three to four hours spent explaining

details and operational aspects of their projects.

After the student meetings, the TSC met with the MITRE staff members to review their impressions of the projects. Comments ranged from minor suggestions concerning the strengthening of mounted components to discussions of major problems such as the trade-offs between expected battery power and anticipated canister weight. In short, the meetings served as a focal point where both the students, the WPI faculty and the MITRE staff could critically evaluate the experiments and as a result, improve the chances of an experiment working properly.

It has become apparent that regularly scheduled design reviews are crucial for any GASCAN project program. We have found that these reviews identify problems in designs, can help find flaws in basic system approaches to experiments, allow one to make changes to systems before they become difficult to change or a safety problem, help identify safety related issues and provide a good educational experience for the students. We intend to schedule these intense reviews for twice a year for our GAS II program.

SYSTEM DESIGN GOALS

All of the experiments the student groups are working on are based on the use of a microprocessor based system controller and the use of auxillary electronic and electro-mechanical equipment. As a result of this dependence we have spent considerable time selecting components and systems that we believe are suitable for the GASCAN environment.

Components And Logic

Almost universally, CMOS devices are the logic elements of choice for any design expected to survive the temperature extremes that can be encountered in the canister environment. While standard TTL devices can withstand a temperature range of 0 to 70 C, the 54xx series TTL family can work in an environment from -55 to +125 C. However, components in the 54xx logic family are not readily available, are expensive and have a high power consumption. Other MOS devices are generally restricted to a temperature range similar to that of standard TTL devices.

CMOS devices inherently have an advantage over other families for two major reasons; first, their operational temperature range and second, their tolerance to supply voltage fluctuations. The nominal operational temperature range for CMOS devices is -40 to +85 C for the 74Cxx family. The supply tolerance for these devices ranges from 2 to 6 volts for some high-speed families to approximately 4 to 15 volts for most of the 4xxx series of CMOS components. In short, the devices are ideal for harsh environments and have a minimal current draw.

The choice of analog devices is not as clearly defined for 'off-the-shelf' components. We have had considerable success using the Texas Instruments TL061 series of biFET operational amplifiers. These devices have the advantages of having low power supply

requirements (+/-3V min), very low current draw (200uA, typ), relatively high slew rates (2-3 V/usec typ) and are available in several package arrangements.

Other components, such as analog multiplexers, converters, sensors and so forth are chosen first for their temperature range and low power consumption and second for the type or configuration available. In all cases, availability is of prime concern and to some extent, is more important than power draw.

Finally, we have provided our students with a list of components that we 'guarantee' that we will have available for their use. This list includes approximately twenty types of CMOS 74Cxx devices and approximately fifteen analog ICs such as operational amplifiers, multiplexers and analog-to-digital converters. Some system components such as 80C85's and associated support chips, and other items such as sensors, development hardware and connectors are also included in the list. The purpose of this list is to constrain our students to components we believe are reliable, easy to obtain, somewhat inexpensive and easy to replace if found to be defective. The result is that many of the controller designs are very simple and incorporate standard components.

Batteries And Powering

There are several interrelated problems associated with the choice of a battery power source. These problems include the following:

- The peak current draw required.
- The self-discharge of the battery over time.
- The available energy in a cell with temperature.
- The desirability of recharging a cell.
- The total energy capacity versus size/weight of a cell.

There is no single battery that is appropriate for all requirements. Generally, there are two basic types of cells that can be used, those that are rechargeable and those that are not.

Rechargeable cells have the obvious advantage that they can be reused. Two common types that are readily available include the standard nickle-cadmium (nicads) cell and the gelled lead-acid cells. The lead-acid cells generally have a high self-discharge rate, can be recycled to any discharge level without a memory effect, have exceptionally high peak-current capabilities, must be vented and are moderately affected by temperature. Our current canister is based on the use of GATES, sealed lead-acid (gelled) cells.

Nicads are rechargeable, have a low self-discharge rate, have good peak current capabilities, and are generally more energy dense than the gelled lead-acid cells. These cells have the problem, however, of having a memory for discharge cycles. Unless these cells are fully discharged and charged each time they are used, they will exhibit a memory effect and will either not deliver their full energy when discharged or will not accept a full charge.

There is some experience and test data indicating that 'heavy-duty' cells such as the Duracells and Eveready Alkaline

cells are excellent for short duration space experiments [3]. Their features include exceptional energy density at low current draws and a relatively stable output voltage with time. In addition, they have a very long shelf life and do not need to be vented. The primary problem is that they cannot provide very high peak currents and their energy availability is strongly dependent on the average current draw, dropping as much as 40% in tests we have performed at 70ma (13.6wh at 1.2V) and 300ma (8wh at 1.1V). Finally, there is evidence that these cells are strongly affected by the ambient temperature, losing much of their energy at low temperatures but regaining that capacity if warmed up [3].

Data Storage

Two reasonable approaches exist; storage on tape using a ruggedized data recorder or storage in memory using some form of non-volatile memory device. Tape storage is ideal for mass data storage. Because of the cost of such units (around \$5000 minimum), however, other alternatives may be more appropriate for smaller amounts of data storage.

One attractive approach to storing data is the use of battery backed-up CMOS RAM or the use of an inherently stable PROM such as an EEROM or an EPROM. CMOS RAM is low power, available in relative dense configurations and is readily interfaced to any microprocessor system. In addition, many manufacturers have 32K and 64K standard bus CMOS boards available at reasonable prices. In addition, some of the commercial boards have a built-in battery back-up capability based on the use of a Lithium battery. Since NASA will not allow lithium cells to be used in a canister, the batteries must be removed or replaced with NASA acceptable units.

EEROM and EPROM need no such battery support but may require unusual programming timing and/or voltages. The latter requirement can often be mitigated through the appropriate choice of a device with a 5V-only programming mode. Although these devices are attractive from a storage viewpoint, they suffer from two primary problems; storage capacity and operational temperature range. EPROMS and EEROMS are generally available in 2, 4 and 8K sizes with byte-wide data paths and are, primarily, based on NMOS technology. CMOS devices, with their corresponding wider operational temperature range, are becoming more available however.

One form of data storage that should not be overlooked is the use of a 35mm camera with a close-up lens and an autowinder. Such an arrangement has been used before and reported on during the first GAS User's Symposium [4]. The film is used to record numerical data via BCD displays in the field of view of the camera. The advantage of such an approach is primarily that film is a simple and permanent form of data storage. To enhance the amount of data storage required, the user need only add a larger film pack.

Two other forms of data storage include the use of a movie camera and a video camera and recorder. Several portable video camera units are now available that are exceptionally small and are completely self-contained. All necessary power requirements are provided through the use of the battery pack included with the

camera. The user need only find a way to mount the unit and either pad the unit or ruggedize it against vibrational damages. The movie (film) camera, like a 35mm camera, is easy to use and generally only requires a close-up lens. Such a camera has the advantage that it can take single frame pictures. Based on a three minute roll of film at 24 frames-per-second, a movie camera could be used to take more than 4000 individual frames of picture data. This is, obviously, many more than a 35mm camera can handle.

Heat Dissipation

It is imperative that GAS experimenters recognize that heat dissipation in an environment where there is no air convection presents a serious problem for heat dissipation from electronic components. Although one could add a low power fan, a more direct approach and certainly a more reliable and lower power approach is to properly heat sink the target devices. Prime candidates are power transistors for driving control motors, voltage regulators and possibly special analog signal processing IC's.

The solution to the heat sink problem is simply to mount the devices on a solid metal wall such as the side of the electronic housing. A less attractive alternative, but acceptable if the heat generation problem is less severe, is to securely mount a device to a wide copper ground plane on the host printed circuit board. If there is sufficient metal in the ground plane, the device may be kept in a reasonable temperature range.

Proper attention to this problem will also solve a second, equally important design problem, that of mounting components so that they do not vibrate and fail during launch. As an example of problem consider the typical 3-pin flat pack voltage regulator. Proper mounting for heat dissipation of this device will also provide a secure mounting mechanism.

DEVELOPMENT SUPPORT

With any program such as a GAS experiment development effort, the host organization should endeavor to provide the design teams with the proper development tools. In an educational environment, except for general laboratory equipment, design and evaluation tools are often not readily available. However, we have found that there are a number of aids that are useful for developing quality experimental equipment. Some of these items will be discussed below.

Vibrational Testing

Although not strictly a design tool, we have found that a shaker table crucial to insure that a system will survive the launch environment. In particular, we have calibrated our table by adding accelerometers and have developed a special, white-noise, spectrally shaped signal generator to drive the table controller. Based on preliminary shaker tests, we have already found problems with some of our designs and have found vibrational effects that,

although not considered a structural problem, have forced us to modify our design approach for a given experiment. An example of this is a situation where we have used a stepper motor to activate a piston in a pressure vessel. The shaker tests showed that the motor would move from its 'home' position to an unknown position. To counteract this action, our software routines now include an initialization procedure to force the motor to a 'home' position, regardless of whether it was there originally.

Software

Software support items include uP based development aids, word processing, drafting aids, printed circuit board layout tools and analytical modeling aids. Although some of the tools were not available for our first canister, we have found that a significant amount of time is wasted in tasks that could be better handled with the help of a design tool. Since there are a number of inexpensive tools available for an IBM PC or equivalent system, we have purchased a number of these tools for student use.

A word processor is available to the project students. This system allows the students to generate professional looking reports. This is an asset to the individual who must interpret those reports and generate the FAR, PSDP and so forth.

Drafting aids are provided in the form of general PC based picture drawing tools. These tools can be used to generate figures, mechanical drawings, schematics, system drawings and other types of graphical pictures. The software also finds use as an aid for developing overheads for presentations and discussion sessions. In particular, like the word processor, the aids are invaluable in the generation of graphics for NASA documents.

The Wintek (tm) printed circuit board design software is available for student use. For GAS I, all of our PCB designs were done manually. For GAS II and new GAS I boards, we will use the Wintek software to design boards. This software, while not providing true auto-design capabilities, does handle auto-routing and double sided boards. In addition, the output is a 2X layout that is camera ready.

Finally, several analytical modeling aids are available to the students. These aids range from P-SPICE for general circuit analysis, to specialized (non-PC based) CAD packages for thermal and mechanical modeling.

SUMMARY

An organization considering the development of a GAS experiment should closely examine the support required to properly carry out such a development. We were rather naive when we started with our GAS I program. As a result, it is likely that of the original experiments we developed for our first canister, only a few of them will actually fly.

For our GAS II program, we believe that we are much better prepared to develop experiments that will almost certainly function properly in the canister environment. In part, this is a result of

the development and evaluation mechanisms we have outlined above.

ACKNOWLEDGEMENTS

No program is the result of a single individual. I would like to thank, first, the approximately 150 students who have worked on the GAS I and GAS II experiments. I would also like to thank the MITRE scientist/engineers, and particularly Mr. R. Labonte', for their committment to WPI and our students. Finally, I would like to thank the faculty and staff of WPI for giving of their time to support this intricate program.

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MICROELECTRONIC
CONTROLLER
DATA
INTEGRATED CIRCUIT
RELIABILITY ENGINEER

SPACE SIMULATED PAYLOADS
DATA STORAGE
ACQUISITION
POWER EFFICIENCY

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Design Considerations for A Gas Microcontroller

by

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Abstract

This paper discusses some of the design problems that we are now addressing in considering a microcontroller for our upcoming GAS payload. We will be using microcontrollers to run our experiments and collect and store the data from those experiments. Some of the requirements for a microcontroller are to be small, lightweight, have low power consumption, and high reliability. This paper will discuss some of the solutions that we have developed to meet these design requirements. At present we are still in the design stage and our final design may change from what we have now. We are continuing to look for new integrated circuits that will do what we need all in one package.

1. Introduction

Microcontrollers will be used in our GAS payload to automatically run our experiments. There are only 3 controls, operated by the astronauts, that can be used to control the payload. When the command is received to start an experiment the microcontroller must prepare the experiment and then run it. The microcontroller must also collect the data from the experiment and store it for use back on earth. The microcontroller will be in control of almost everything inside the payload.

In designing our microcontroller we considered a number of design criteria. Some of these were reliability, power consumption, weight and compactness. High reliability was very important to us and played an important part in the decision for the basic design. There are two ways of assuring high reliability. One is to have three or more parallel processors, all checking one another. This was determined as too difficult to implement on our scale. The next best alternative was to have a separate microcontroller for each experiment. If one controller failed it would not

jeopardize all of the other experiments. Our design goal was to develop a microcontroller that with a few modifications or additions could be used with many different experiments.

2. Power Consumption

Designing a microcontroller with a low power was very important, as batteries are heavy and take up space. By limiting power consumption we can reduce the size of batteries that are required and utilize the weight and space saved for the experiments.

In considering available integrated circuit technologies, it was determined that CMOS was the best choice. Most of the popular microprocessors and microcontrollers are available in a CMOS version. Several companies are also making the 74HC series of digital integrated circuits which will also be needed in the microcontroller. This new generation of integrated circuits combine the low power requirements of CMOS with the high speed of the older TTL circuits.

There is, however, one major drawback to CMOS circuits. They are very static sensitive. If the devices are mishandled or if the circuit takes a static shock, the electronics can easily be damaged. We decided that with proper circuit board and enclosure design, static electricity should not be a problem for us.

3. Size

Because of the small size of the GAS canister, space is at a premium. Reducing the size of the electronics would leave valuable space for the experiments. Besides making the layout of the circuit boards as efficient as possible, reducing the number of integrated circuits used is the best

way to reduce the size of the electronics. Reducing the device count was the major factor in selecting the circuits we plan to use.

A side benefit of reducing device count was that it would also reduce power consumption. A device that replaces several integrated circuits will have a lower power consumption than the total of all the individual devices, so wherever possible, we tried to find circuits that would do as much as possible in a single package.

4. Device Selection

Taking these design criteria into consideration, we had to make our selection of the devices that we were to use for our microcontroller. Most of the popular microprocessors are available in CMOS versions, so power consumption is not much of a problem. However, microprocessors require support chips for timers, input and output, and memory, which leads to increased device count. We then looked at microcontrollers. These devices have RAM, I/O, and timers all in one package. The particular microcontroller that we looked at in detail was the INTEL 80C31. This device is of the CMOS variety, has 4 8 bit I/O ports, and 128 bits of random access memory for temporary data storage. This seemed to fit the requirements for what we needed.

However, we identified several problems with this device. It has on-board mask programmable read only memory which we could not use. The chip can be programmed to use external RAM, but we lose the use of 3 I/O ports for address, data and control lines. This left us with 1 I/O port and internal timers, so we still have a reduced device count.

For our program memory, we have chosen a CMOS PROM device from Harris. It is an HM-6616 CMOS PROM package with 2048 by 8 bit memory arrangement.

On our previous GAS package that we sent up (GAS 009), EEPROMS were used for data storage. When the data was read out after the return of the canister, some of it seemed to have been altered and it was suspected that cosmic radiation could change a few bits here and there. This is very undesirable in the program memory because if a bit is changed, it could result in the microcontroller locking up. We decided, therefore, to use a PROM device. In a PROM, selected conductors are actually destroyed like a fuse to create the ones and zeros and it would not be possible for cosmic radiation to change them.

To store data collected during the flight, we decided to use an EEPROM device. We wanted to keep the power supply at 5 volts and some of the new EEPROM chips come with single 5 volt supply requirements. The device we are testing is a NCR 52832, a 4096 by 8 bit low power EEPROM memory package. One thing to note about EEPROMS is it takes as long as 10 milliseconds to permanently write data. This is much longer than the clock cycle of the microcontroller device, so the chip must be buffered from the microcontroller. Further, the microcontroller must go into a delay loop while the device burns in the data. The particular device we chose has a 16 word register, so 16 words can be written at one time. During the write cycle the EEPROM requires a lot of power, so writing 16 words at a time will save power.

We are now looking at several analog to digital converter packages. We are looking at a few CMOS A/D devices that have an eight channel multiplexer on board. This will allow 1 circuit to read eight different analog inputs. This again combines several chips into one and reduces the device count. This chip will be used to collect temperature data or voltage readings throughout the canister.

The rest of the support devices that we will use are from the 74HC series of integrated circuits. These are CMOS versions of the 7400 and 74LS series of TTL chips.

5. The Microcontroller Design

We are presently working on 4 experiments that require micro-controllers. There will be two other experiments in our payload, but they are passive. For reliability each experiment will have its own microcontroller. The basic design of our microcontroller that will be common for all the experiments will include a microcontroller circuit, a PROM for program memory, and an EEPROM for data storage. This will serve as the foundation of our microcontrollers. Other circuits will be added as required by the different experiments.

The first experiment we are working on will measure the noise, shock, and vibrations that our payload experiences during launch. We are planning on using a new kind of structural support for our canister using graphite fibers. We intend to measure the stresses involved during lift off to determine the effectiveness of our new design. We will need to add a high speed analog to digital converter to the basic microcontroller to measure the outputs from the accelerometers.

For the second experiment, the microcontroller will be taking and storing temperature data taken at various points in the canister. The purpose is to measure the effectiveness of the thermal blocks whose job it is to maintain the temperature in the canister. The microcontroller will require an A/D converter, but because the sampling rate will be slow it does not need to be high speed. We may need to use a high resolution converter depending on the accuracy of the measurements that we need.

The third experiment is the materials processing experiment. In this experiment tin and zinc carbonate are melted to form a "foam metal". The microcontroller for this experiment will be interfaced with a switching circuit to control the heating of the metal. It will also have an A/D converter circuit for temperature measurement inside the oven.

The fourth experiment will be this data telemetry experiment. The microcontroller for this experiment will collect data and status information from the other microcontrollers through a serial communication network. This microcontroller will be interfaced to a speech synthesizer device to produce English speech. The output from this circuit will go to a radio transmitter to transmit the data back to Earth. This is so we can get data and the status of the canister on a real time basis. This will be the only microcontroller that does not require the EEPROM in the basic design because no data will be stored by this microcontroller.

Not all of the experiments will run immediately when turned on. The data telemetry and the materials processing experiments will be run only after certain prerequisites are met. The astronauts will determine when these prerequisites are met and will signal for the experiments to commence through the use of the GCD relays. The microcontrollers will have to be able to read these switches. The I/O port on the microcontroller circuit will be used for this.

6. Conclusion

We recognize that the design described here is probably not the best design possible. Design requirements will probably change from payload to payload. It is hoped, however, that what we have described are the problems to be solved and how we have chosen to meet them. We would like

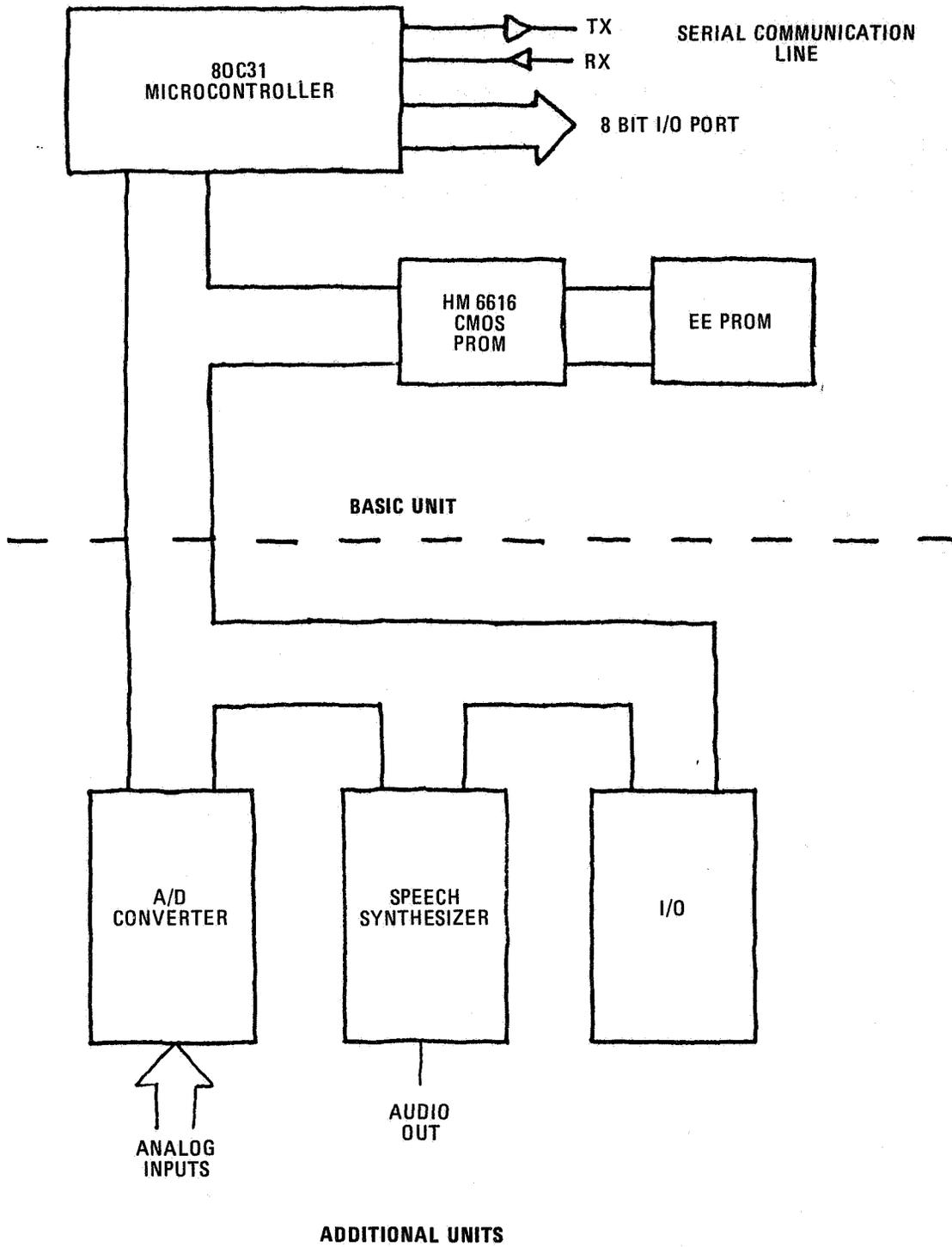
to hear about any ideas on how to meet these requirements in a better fashion. We are always open to ways of doing things better.

One of the new ideas that we are considering is that a new microcontroller device that has an auto reset feature. It is a timer that counts down. If this timer is not reset by the program (because, for example, the microcontroller has locked up), the timer will force the device to reset itself. This may be a desirable feature in the event that the cosmic radiation changed a bit inside the microcontroller circuit, or if there was just noise on the line causing a lock up. The search goes on.

TABLE OF DEVICES USED

<u>Device</u>	<u>Manufacturer Number</u>	<u>Manufacturer</u>
Microcontroller	80C31	INTEL
CMOS PROM	HM-6616	HARRIS
EEPROM	52832	NCR CORP
A/D Converter	AD7581	ANALOG DEVICES

MICROCONTROLLER BLOCK DIAGRAM



N86 - 27314

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PROJECT EXPLORER'S UNIQUE EXPERIMENTS:
GET AWAY SPECIAL #007

by

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ND776801

GAS EXPERIMENTERS SYMPOSIUM AT GSFC
October 8-9, 1985

ABSTRACT

The Project Explorer payload represents the first attempt at broadcasting digitized voice signals via a Space Shuttle flight on amateur radio frequencies. These amateur ham-radio frequencies will be transmitting real time data while the experiments are operating. Experiments 1, 2, and 3 represent the work of students ranging from materials processing to the science of biology. Experiment #1 will study the solidification of two hypereutectic alloys, lead-antimony and aluminum-copper. Experiment #2 will investigate the examination and growth of radish seeds in space. Experiment #3 will examine the electrochemical growth process of potassium tetrocyonoplatinate hydrate crystals and Experiment #4 involves amateur radio transmissions, monitoring and support of the entire GAS #007 payload.

INTRODUCTION

G #007 was flown in early October, 1984 on Space Shuttle "Challenger," STS-41G. However, there was substantial evidence that this payload was never turned on during the otherwise most successful mission. Because of this procedural error, G #007 did not yield the desired results. Currently, Project Explorer is tentatively rescheduled to fly in November, 1985, on STS-61B. In any event, a unique feature of this GAS payload will be its ability to demonstrate transmitting amateur radio transmissions to global ground stations around the world, real-time and in the English language.

Obviously, where there are new problems, changes in the paperwork must be amended accordingly. The one and only spare battery we had available unfortunately developed a crack in its outer casing which allowed the electrolyte substance to leak out. In addition, the only two battery vent valves we had were damaged during shipment of the GAS flight lid. G #007 was sealed and ready, we thought, for its June 1985 launch date, but last minute problems indicated the need to scrub, repair and replace vital components. The entire group inspected the overall package and found that a new SRB (Solid Rocket Booster) battery would fit into the same space in which the SPAR (Space Applications Rocket) battery was being accommodated. And with this new installation, only two components of the radio experiment had to be relocated. No other apparent problems exist.

This GAS experiment package consists of three micro-gravity experiments and an amateur radio experiment which permits data relay to receiving radio ground stations. The concept of the project is to design, develop and fly selected student experiments in the Space Shuttle's cargo bay and to obtain scientific data on the unique conditions of space flight, especially in the area of low-gravity conditions.

In 1978, the co-sponsoring agencies, the Alabama Space and Rocket Center, the Alabama-Mississippi Section of the American Institute of Aeronautics and Astronautics, Alabama A&M University, and the University of Alabama-Huntsville, from which it has required extensive technical support, undertook this project to encourage high school students to become involved in space-oriented engineering efforts. A brochure was developed and distributed nationwide to high schools throughout the United States soliciting proposals by students. The captivating brochure read: "Students, Can You See Your Ideas in Space?" Over 150 proposals were submitted and only thirteen students were selected. Today, only two of the original thirteen students remain with one other being a research associate, and the other being an invited guest.

Experiment Nos. 1, 2, and 3 use the micro-gravity of space flight to study the solidification of hypereutectic lead-antimony and aluminum-copper alloys, the germination of *Raphanus Sativus* radish seeds, and the growth of potassium-tetracyanoplatinate hydrate crystals in an aqueous solution. Flight results will be carefully compared with earth-based equivalents.

Experiment No. 4 features the amateur radio transmitter. It will also provide timing for the start of all other experiments. A microprocessor will obtain real-time data from all experiments as well as temperature and pressure measurements taken within the canister. These data will be transmitted on previously announced amateur radio frequencies after they have been converted into the "English Language" by a digi-talker for general reception.

OPERATIONAL SCENARIO

The G #007 payload will require a duration of five full days and a "turn-on" signal for the experimental package as early in the Shuttle mission as possible. Other operational requirements of the individual experiments are as follows:

Experiment No. 1: The solidification of alloys experiment will be timed so that solidification will occur under the best available micro-gravity conditions. At that time, a signal from the GAS Operations Panel within the crew compartment will trigger the operation of this experiment, via Experiment No. 4, for the first solidification sequence. Subsequent operations will be started by built-in controls and will not require additional signals from the crew. Another period of low-gravity operations for the second solidification sequence of this experiment will occur about a day later.

Experiment No. 2: The radish seed germination and growth experiment must be initiated as soon as feasible, i.e., at about the time the Shuttle reaches its orbit, to obtain the longest possible growth period for the seeds. Operational control will be provided by Experiment No. 4. Upon the initial G #007 power-up, a relay will activate pump "A" to supply the water-fertilizer solution needed to generate the seed growth. Upon power-down, approximately 120 hours later, another relay will activate pump "B" and freeze any further seed development by the application of a buffered formaldehyde solution to the seeds.

Experiment No. 3: Operational control will be provided by Experiment No. 4. Crystal growth will begin when the best micro-gravity conditions exist. At the beginning of the first available low-gravity period lasting 4 hours or more, Experiment No. 4 will power-up the electrolysis cell by a 1.3 Vdc power supply, and crystals will start to form on the anode. A 35 mm camera and its electronic flash will have been activated at the same time with a built-in time delay to take pictures every 40 minutes for at least 9 hours.

Experiment No. 4: The Marshall Amateur Radio Club Experiment (MARCE) will control all other experiments in accordance with individual requirements. The "ORBITER'S AUTONOMOUS PAYLOAD CONTROLLER (APC)" provides AFT-Flight Deck Control for experiments "toggle-on" and "toggle-off" and can also terminate all GAS operations if SAFETY NEEDS require such premature cessation of the experimentation.

The radio experiment's dipole antenna installed on the NASA-furnished GAS container lid will transmit from the Orbiter Cargo Bay during flight at the frequency of 435.033 MHz which has been obtained from AMSAT. This frequency will also be compatible with the OSCAR-10 (Orbiting Satellite Carrying Amateur Radio) for a possible relay link to the ground. A direct down-link requires the ground station to have an FM receiver with 16 KHz RF bandwidth and a 0.1 microvolt sensitivity. Ground antennas should have right circular polarization with at least 10 dB gain. To relay through the OSCAR-10 satellite, a 145 to 146 MHz two-meter receiving station will be required. After initialization by the Orbiter crew via APC, the transmitter "ON" times will be controlled by the MARCE-CPU for maximum experiment data downlink times and efficient power usage.

Three transmission cycles of eight hours each are planned. However, because of our "new" energy capability (about double the amp-hour of the old battery) it would be good to transmit every minute and also extend the cycle life to 12-16 hours, if possible. A transmission cycle consists of a 30 second transmission every minute. When Experiment No. 1 is active, transmissions will last 45 seconds. The first cycle will be activated by a Shuttle crew member at G #007 initial "toggle-on." The second, third and etc. cycles will be subsequently activated during the second crew "toggle-on" command. Furthermore, we highly recommend that HAM radio operators use a cassette recorder as well as our Data Log Sheets to record these data transmissions.

DESCRIPTION

All experiment packages and/or their components are mounted on a rectangular mounting plate, which is in turn bolted to the rib of a round plate which is bolted directly to the GAS canister top lid.

Experiment No. 1 will be a solidification study of two hypereutectic alloy samples (Lead-Antimony and Aluminum-Copper) melted and solidified inside an internally insulated aluminum cylinder (15 in. long and 6 in. diameter). It will house two small oven cylinders that will encompass two miniature furnaces (4½ in. high and ¾ in. diameter). The wall thickness of the oven cylinders is 1/16 in. thick. The melting furnaces are made of lava cores and are wrapped with Nichrome wire (spring coils). The alloy samples are centrally located in the middle of each furnace core, occupying a total volume of 1.0 cm³, and can heat up to temperatures as high as 800°C. The heat is generated by a 28 Vdc electric current from a central

power supply. Moreover, while the inner most part of the small oven is around 750°C, the outer most surface of the oven cylinder is around 60°C.

These alloys will be molten while contained in high purity alumina crucibles. A micro-structural as well as chemical analysis of the obtained alloys will be compared with samples processed under similar environmental conditions on the ground. In addition, comparative mechanical tests will be performed on each specimen to determine the influential parameters that may or may not have altered the solidification process.

The DAQ2-K (Data Acquisitions and Control Unit) exterior is also made of aluminum and it contains the experiment control system which supervises the two separate metallurgical experiments. Its primary functions include measuring temperatures of the two experimental vessels, storing measured values for later recall, operating the two ovens used in the experiment and sending experimental data to the external telemetry equipment in Experiment No. 4. The experiment control system has two modes of operation, normal and test. The normal mode is used during flight to run the experiments solely under control of the internal experiment system (DAQ2-K). The test mode is used in the laboratory to insert and display the experiments' parameters via the DAQ2-Console Control Package (a KAYPRO II portable computer).

During operation of the experiment, data is sent to the external telemetry equipment (located in Experiment No. 4) once a minute via a TTL serial data port in Experiment No. 1. If the system is operating in the test mode, identical data is also routed to the RS232 port. Telemetry data consists of identification codes currently executing the current time as maintained by the system and the measured values from the eight thermocouples. These messages are formatted on a computer as shown below and are terminated by a carriage return and line feed:

```
AA T=HHMM TC1=RR TC2=RR TC3=RR TC4=RR TC5=RR TC6=RR TC7=RR TC8=RR
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Where AA is a two character abbreviation of the current mode, HH is the current hour in 24 hour format, MM is the current minute, and RR is the actual temperature reading – all in hexadecimal notation. This experiment weighs less than 30 lbs.

Experiment No. 2 will be a comparative morphological and anatomical study of the primary root system of radish seeds conducted inside an aluminum rectangular container, about 270 in.³ in size. This experiment will demonstrate the ability of rapid germination and subsequent seed structure growth. Differences in tissue orientation and organization will be examined under the microscope after return from the mission and will be compared with normally grown seeds. Intercellular adjustments occurring during best available micro-gravity conditions will thus be determined.

The radish seeds will be held in place by filter paper as the growth substrate. Gear-type pumps will initially deliver a water-fertilizer solution to the seeds, and at the end of the mission supply a buffered solution of 5% formaldehyde in isopropylalcohol to stop any further growth during descent and disassembly after return. The radio experiment will provide control of the radish seed operations as follows: One hour after initial power-up, a control relay will be closed for 30 seconds and thus activate pump "A" which supplies the water/fertilizer solution to the seeds. Prior to GAS power-down (approximately 120 hours later) a second control relay will close for 30 seconds and thus activate pump "B", which supplies the buffered formaldehyde/alcohol solution.

This experiment will require a temperature of 30°C plus/minus 10°C. Should temperatures inside the growth chamber go below 20°C, a small heater will be activated to maintain required temperatures. There will be no cooling provisions for temperatures above 40°C.

A turn-on signal as early in flight as possible is desired to assure an experiment operations time in orbit as close to five full days as possible, or more. Again, best available micro-gravity levels are desired, but 10⁻³ g's is acceptable. This experiment will weigh not more than 15 lbs.

Experiment No. 3 will be a crystal growth study of metallic appearing needle crystals in an aqueous solution of Potassium Tetracyanoplatinate. This experiment will be conducted in an electrolysis cell of 6 ml in volume. The solution carried is a 0.3M concentration. The cell is made of plexiglass and fitted with two platinum electrodes. The overall experiment volume will occupy 0.5 ft³. Upon application of an electrical potential, nucleation and crystal growth are effected at the anode. A very minute amount of hydrogen gas is released at the cathode. The optical system consists of a 35 mm NIKON F-2 camera with a 50 mm close-up lens; camera autowinder MD-3, with camera battery pack MB-1; and a small NIKON SB-E electronic flash. The battery pack holds 10 AA size batteries (15 Vdc) and is used to power the camera. The electronic flash holds 4 AAA size (6 Vdc) batteries. This experiment requires a micro-gravity duration of approximately 72 hours for completion. Camera operation will be synchronized with the flash to photograph the crystals at a rate of one exposure per forty minutes.

A thermister is attached to the electrolytic cell to monitor the temperature fluctuations of the experiments' environment. This experiment requires a temperature of 20°C plus/minus 10°C. Should temperature inside the electrolysis cell go below 10°C, several small heaters will be activated to maintain required temperatures. The precision reference supply and a control timer will be furnished by Experiment No. 4. This experiment will weigh less than 10 lbs.

Experiment No. 4 will be an amateur radio experiment that will provide information on the "Project-Explorer-Payload-elapsed-time" and the operational status of experiments (such as measurements in canister) during flight by down-linking data to all amateur radio stations and short-wave listeners (SWL) around the world. The data will be transmitted by voice in English at 435.033 MHz, so that amateur radio operators and SWL's around the globe can participate. The Marshall Amateur Radio Club (MARC) identification signal "WA4NZD" will be included at the beginning and end of each transmission. This experiment will acquire these data for input to analog-to-digital converters through signal (hand-shake) lines. A voice synthesizer Digitalker system will convert the experimental data into "English" and will modulate the transmitter.

A surplus SRB battery will furnish the power at a nominal 28 V and 50 Amp-hours. All measurements will be stored in a microprocessor memory for post-flight analysis. All data will be taken every 10 minutes for a total operation time of 120 hours. Receiving amateur radio stations will record the measurement data and relay the information via High Frequency amateur radio channels to MARC at the Marshall Space Flight Center in Huntsville, Alabama, and other interested parties (to be determined). It may also be possible to utilize an existing OSCAR satellite in orbit for this purpose. In the event of objectional radio interference with Shuttle operations, the radio transmitter can be turned-off to assure safety of Shuttle operations.

The data system uses a CMOS micro-processor central processing unit (CPU) with an 8k byte memory, a 2k byte EPROM program memory, Input/Output/Timer (OT) and a 128 bytes

scratch-pad memory for temporary (volatile) storage. A 4 MHz crystal provides the base for a CPU and a timer. A back-up battery (2 D cells) is used to retain memory when the CPU power is removed.

The weight of all equipment for Experiment No. 4 aside from the battery will be about 10 lbs. Total volume this experiment takes up is only 0.8 ft³. A 45 lb surplus SRB battery will provide the overall power at 28 Vdc and 50 amp-hours. It has a total volume of 0.4 ft³.

SUPPORT STRUCTURE

The support structure for all G #007 experiments consists of two primary plates and four "bumper" assemblies. One of the primary plates is a round plate which mounts to the GAS canister top lid. This round plate has a machined rib along a diameter, to which the second rectangular plate divides the GAS canister volume into two equal halves along the longitudinal axis. The bottom of the rectangular plate, which supports all experiments, is supported by four "bumpers" contacting the inside of the canister. Two of these "bumpers" are mounted on the lower corners of the rectangular plate. The other two "bumpers" are mounted on "T" mounts perpendicular to the faces of the rectangular plate.

TECHNICAL LESSONS LEARNED

On November 16, 1984, the Explorer Team officially learned from Goddard Space Flight Center that the G #007 payload was never turned on during its scheduled 8 day Shuttle flight October 5-13, 1984, STS-41G. Because of this error, neither of the four experiments obtained the long awaited results. Furthermore, based on the post flight inspections and analysis, the Explorer Team concluded; all four experiments, the structural assembly, and the integration hardware withstood the launch and Shuttle environment in all respects from liftoff to landing. The Principal Investigators, as well as the entire Explorer Team learned a great deal about flying and re-flying a Shuttle GAS payload and are continually learning day by day that things do not always go according to plans. These lessons will be gladly passed on to other interested parties and future GAS payload users.

Moreover, even from the simplest of experimental ideas, there is much, much more than meets the eye. The main lesson learned was the complexity involved in preparing a simple GAS payload – in addition to the reams of paper work. From the initial ideas, design and fabrication, to integrating the four experiments into one functional package, made the whole endeavor invaluable for the Explorer Team.

For Experiment No. 1, the major problem was finding a way to monitor and store the experimental parameters during operation. This problem prompted the development of the DAQ2-K.

For Experiment Nos. 2 and 3, the main problem was maintaining a constant temperature for an extended period of time. Miniature heaters were designed to facilitate this problem. In addition, for Experiment No. 3, recording the KCP crystal's growth was questionable until a modified 35 mm camera was acquired.

Experiment No. 4's major hurdle was obtaining approval to mount an external antenna on the outside of the GAS canister. In addition, damage to the battery vent valves and the battery case cracking developments also presented some unforeseen problems. However, these problems were also reworked and resolved.

Overall, Project Explorer has come a long way from its initial beginning and has accomplished 95% of its original objectives. The real test, "during Flight," is yet to come.

ACKNOWLEDGMENT

The G#007 payload could not have been completed without the dedication and total commitment of the student P.I.'s and the MARC P.I. who also made gallant achievements in preparing their individual experiments. Also, thanks and appreciation to the Co-Director, Project Manager, the Integration Team, and countless others who have contributed immensely in making this payload a success.

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GROWTH OF ZEOLITE CRYSTALS IN THE
MICROGRAVITY ENVIRONMENT OF SPACE

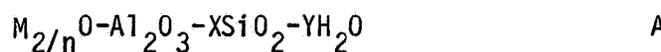
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1. Introduction

Zeolites are hydrated, crystalline aluminosilicates with alkali and alkaline earth metals substituted into cation vacancies (Equation A)



They have a uniform internal channel system which makes them ideal for use as catalysts and adsorbents. Presently zeolite materials constitute a 1-2 million dollar/year business. Typically zeolite crystals are 3-8 μm along their characteristic dimension (e.g., edge). Larger zeolite crystals are desirable. Large crystals will allow more detail characterization of their complex crystal structures. Also, they will help improve reactor performance, may be used as membrane separators, and they should enhance basic studies in molecular diffusion in zeolites.

Large zeolite crystals have been produced (100-200 μm); however, they have taken restrictively long times to grow. It has been proposed if the rate of nucleation or in some other way the number of nuclei can be lowered, fewer, larger crystals will be formed. The microgravity environment of space may provide an ideal condition to achieve rapid growth of large zeolite crystals (e.g., >100 μm). Since settling crystals are believed to cause secondary nucleation, by limiting crystal movement by suspension in the zero gravity environment, fewer crystals will form, but they will grow larger. The objective of the project is to establish if large zeolite crystals can be formed rapidly in space.

2. Experimental Procedure

The experimental procedure for making zeolites is straight forward. An example of this procedure is illustrated in Figure 1. An aqueous Sodium Metasilicate solution is mixed with an aqueous Sodium Aluminate solution which

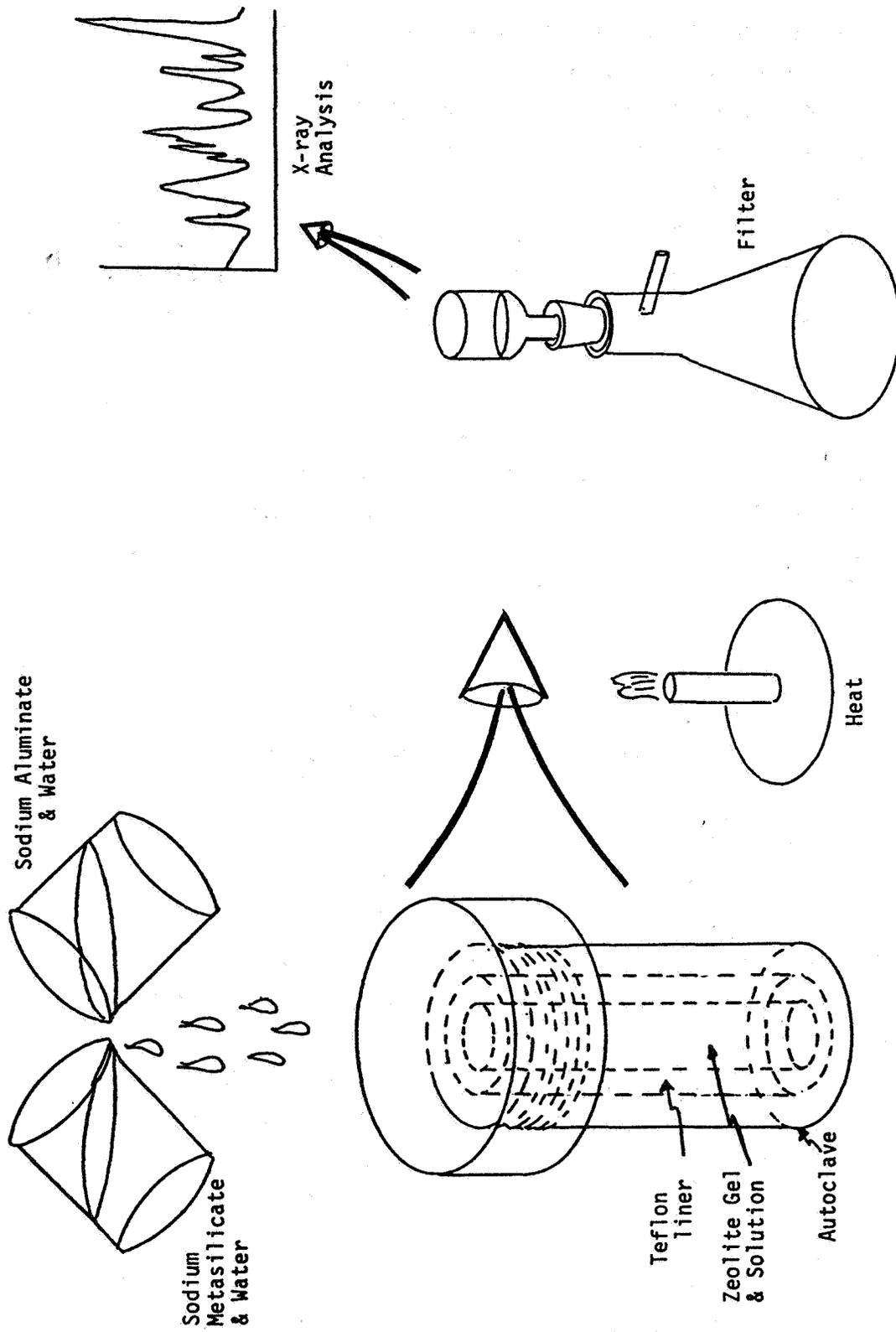


Figure 1. Experimental Procedure

forms a "gel". This gel is poured into a teflon lined stainless steel or aluminum vessel (autoclave). The autoclave is then heated to the desired temperature and held at the temperature for the required time. The resulting mixture is filtered and the crystal structure of the crystals formed are characterized using X-ray diffraction. For the space shuttle, this procedure is only moderately altered. The solutions are premixed and stored in a specially designed autoclave. This system is designed to heat up the reaction vessel rapidly to the desired temperature and maintain the temperature for up to 72 hours. Analysis of the product (crystals) will be performed on return to earth.

3. Zeolite Selection and Preliminary Results

There are 44 species of natural zeolites and as many as 150 synthetic zeolites. The initial selection criterion used to determine which zeolite should fly was as follows:

- A. Material of industrial interest
- B. Must achieve at least 40% crystallization within 72 hours
- C. Reaction temperature must not exceed 393K
- D. The "gel" must be unreactive at temperatures below 308K for up to 90 days.
- E. No rapid phase transformations at 393K or on slow cooling

Criteria B-E were the direct or indirect result of NASA requirements. For example, NASA will only guarantee a flight of 72 hours duration. Thus, the experiment must be complete in that time. The internal pressure was not to exceed one atmosphere. This limited the mostly water solution to a maximum temperature of 393K. Since launch delays could be as long as 90 days, the premixed system must not substantially settle or react in that time while at ambient temperature. Also, post flight delays require the typically metastable zeolites not to transform rapidly to a more stable form on cooling. After a detailed literature survey and some experimentation, zeolite A was chosen.

Figure 2 is a plot of percentage crystallization versus reaction time for zeolite A at three temperatures. As shown, at all three temperatures, 100% crystallization is reached in 72 hours. The intermediate temperature of 369K was chosen. At this temperature the system's vapor pressure is less than 1

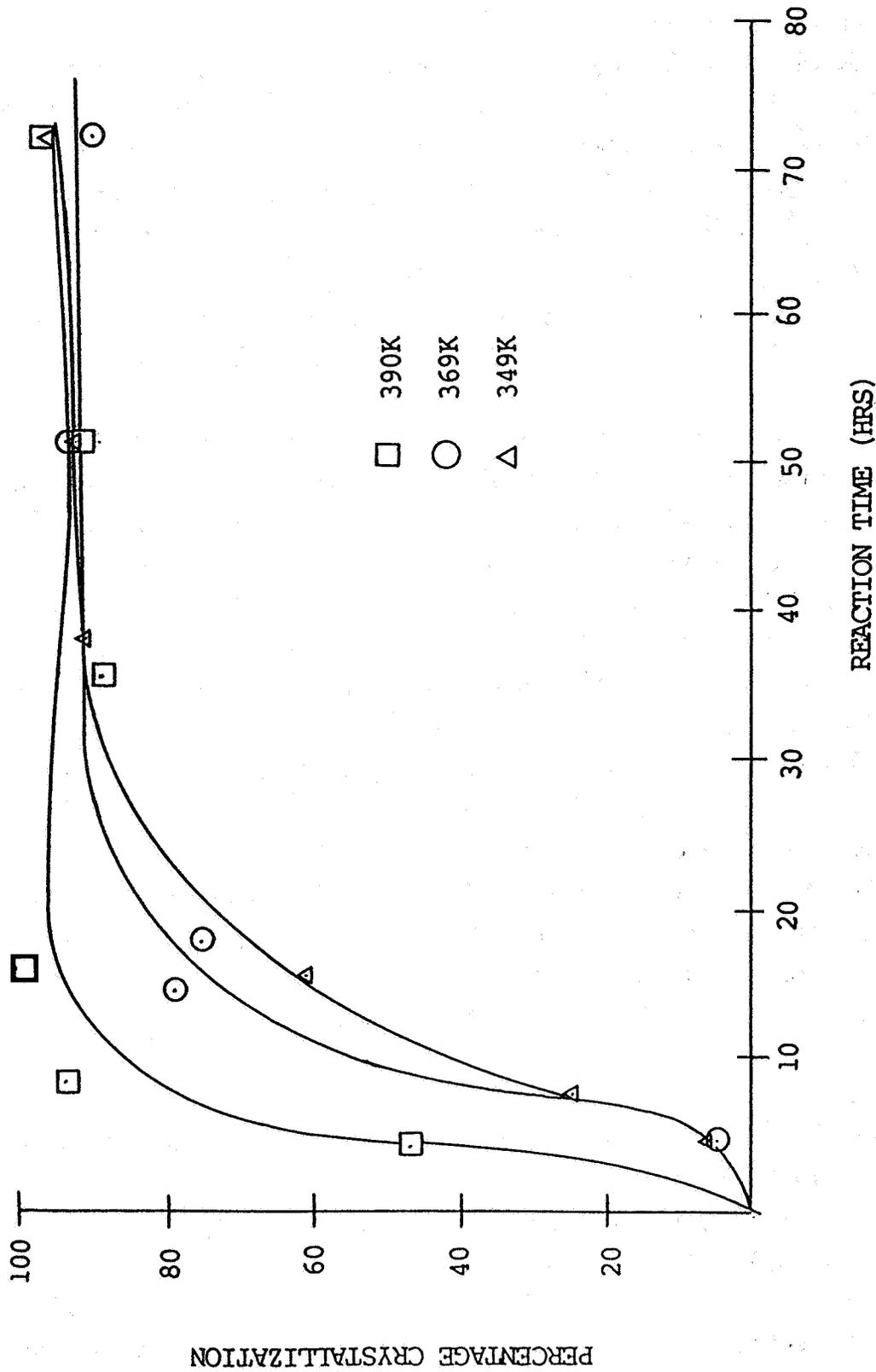


Figure 2. Percentage Crystallization versus Reaction Time for Zeolite A
 (2.05 Na₂O - Al₂O₃ - 1.05 SiO₂ - 138 H₂O)

atmosphere and the reaction rate is still high.

It is known that if one suspends a crystal in solution, it depletes local nutrients and the growth rate is slowed. The effect of this phenomenon was tested on zeolite A by systematically adding more and more water (diluting the reaction mixture). Figure 3 is a plot of crystal size versus the moles H_2O /moles Al_2O_3 ratio for zeolite A at 369K for 24 hours. As shown, the more dilute the solution, the greater the average and maximum crystal size. The rate is linear for both the average and the maximum crystal sizes, and it is about 1-1.5 μm /100 moles H_2O . At the present time, this is not anticipated as a problem.

Figure 4 shows two plots. One illustrates the systematic addition of Triethanolamine (TEA) to zeolite A solutions. This was done to enhance the growth of large zeolites as well as to provide a suspending mixture to help stop settling during the prelaunch phase. As indicated in Figure 4(a), both the maximum and average crystal size were increased by adding TEA up to a TEA/ H_2O ratio of 1.83×10^{-2} . After this value crystal size decreased. The reason for the maximum is still under investigation. However, growth is definitely enhanced when settling is hindered. Also, the dark symbols indicate the effect of filtering the initial reactants. Upon filtering, secondary nucleation points are eliminated resulting in fewer larger crystals. Also shown in Figure 4(b) is the effect of a prelaunch delay. A similar set of experiments to those presented in Figure 4(a) was performed. However, these solutions were held at room temperature for 45 days prior to reaction at 369K for 72 hours. As shown settling is hindered but not eliminated. Additional work to improve this performance is presently being done.

4. Conclusions

These results as well as others, not included for the sake of brevity, suggest the following conclusions:

- A. Zeolite A is a viable candidate for testing in space.
- B. Triethanolamine, TEA, helps in the growth of large crystals.
- C. Filtering the reactants helps in the growth of large crystals.
- D. Shelf-life experiments suggest TEA/zeolite A solutions may not be substantially affected by a 90 day delay to launch.
- E. At projected growth rates, zeolite crystals over 100 μm should form.

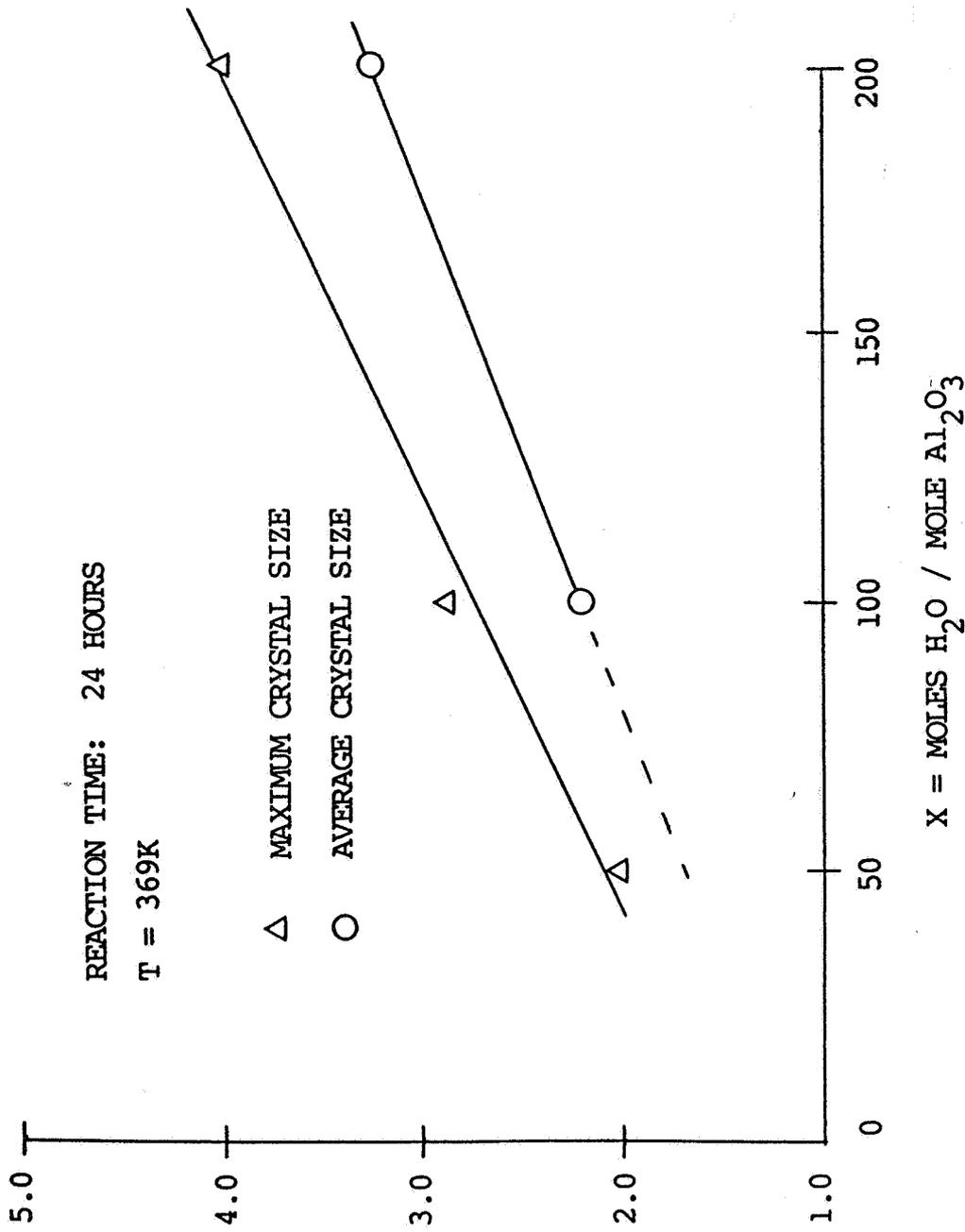
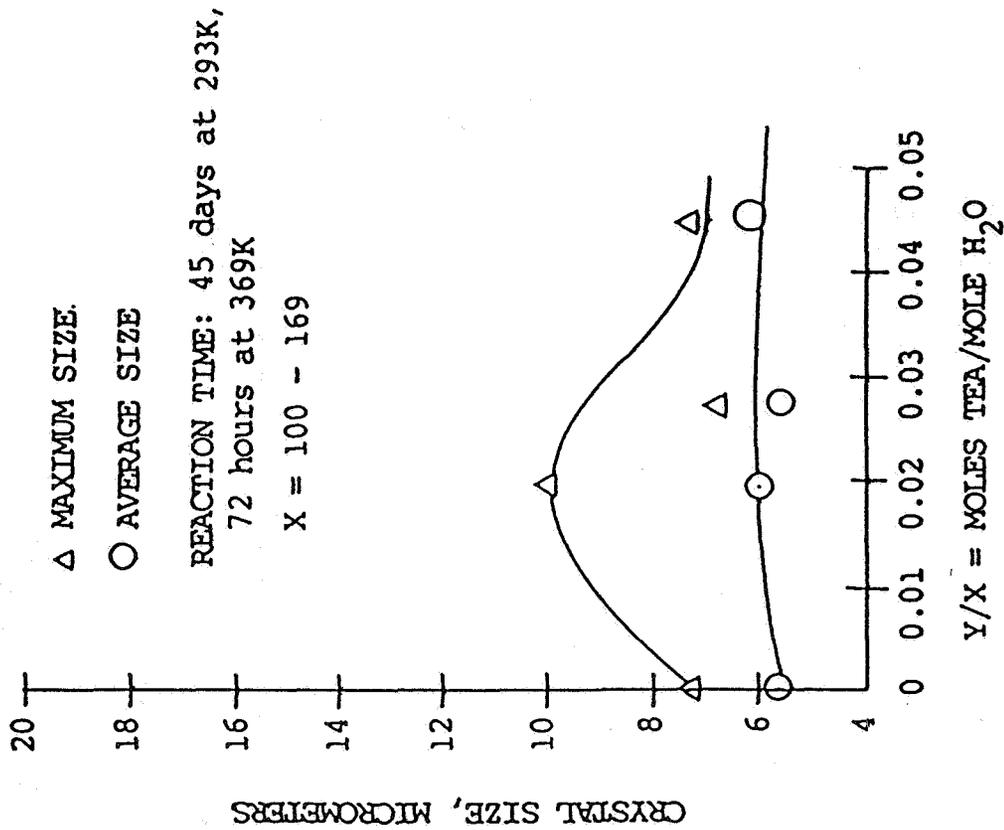
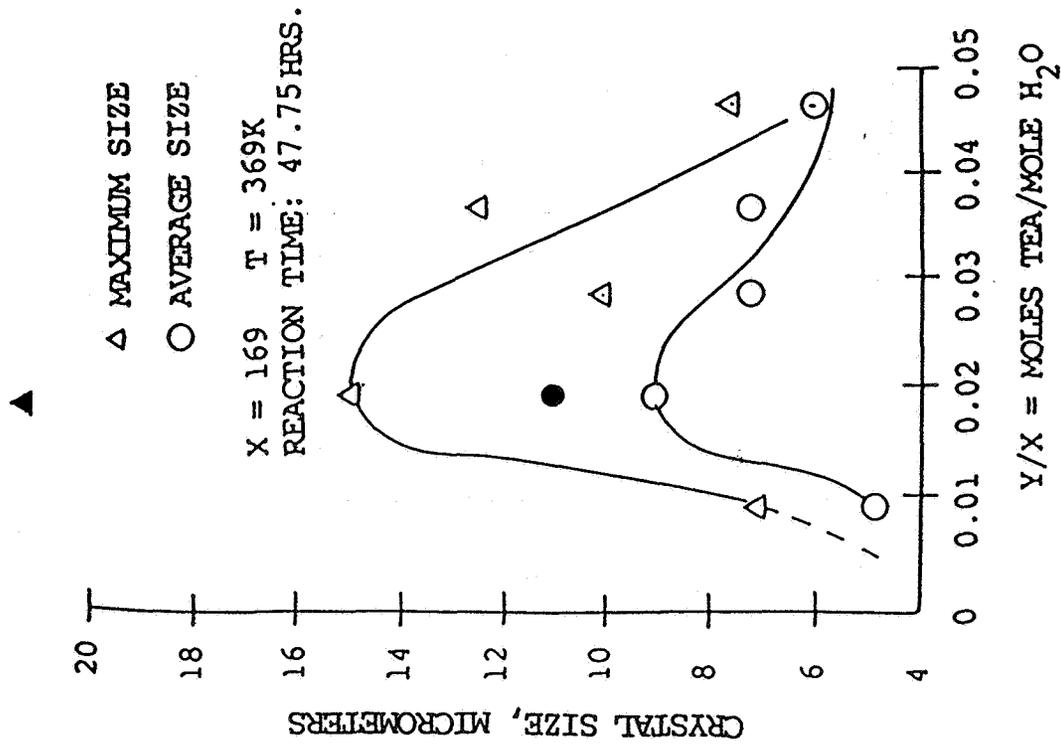


Figure 3. Crystal Size Versus H₂O/Al₂O₃ Ratio for Zeolite A
 2.03 Na₂O - Al₂O₃ - 1.5 SiO₂ - xH₂O



a) dark samples represent filtered samples b) reaction after 45 days at 293K

Figure 4. Systematic Addition of TEA to Zeolite A (2.04 Na₂O - Al₂O₃ - 1.04 SiO₂ - xH₂O - YTEA)

ACKNOWLEDGEMENT

We gratefully acknowledge the technical and financial help of the MITRE Corporation in this work.

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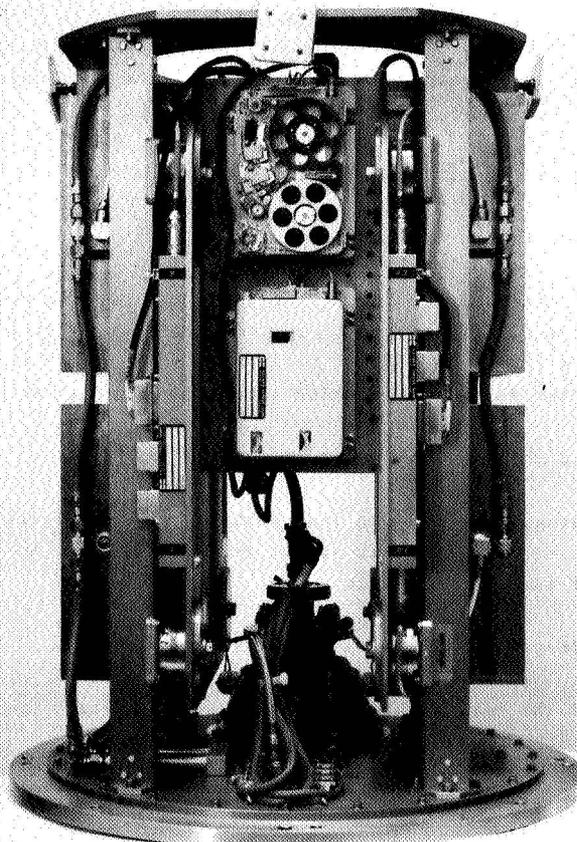
HALOGEN LAMP EXPERIMENT, HALEX

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HALEX PAYLOAD

INTRODUCTION

The main purpose of the Halogen Lamp Experiment HALEX was to investigate the operation of a Halogen lamp during an extended period in a microgravity environment and to prove its behavior in space.

Hallex was funded by the European Space Agency under two separate contracts. The payload was designed and built by Kayser-Threde, who also coordinated the mission with NASA. The experiment was defined, built and maintained by Dornier System.

OBJECTIVES

Mirror Heating Facilities for Crystal Growth and Material Science Experiments in space rely on Halogen lamps for their source of heat.

Dornier System developed and built a MHF Mirror Heating Facility which was flown on FSLP in 1983, the MEDEA-ELLI will be used during the upcoming D-1 Spacelab Mission 1985 also the 1988 EURECA Mission will be furnished with an AMF Automatic-Monoellipsoidal Mirror Furnace.

All these furnaces use one or two Halogen lamps where the radiation is focussed for melting a specimen.

With the AMF a long-term operation of a fully powered Halogen lamp is planned.

The HALEX aim is to verify:

- o Full-power operation of a Halogen lamp for a period of about 60 hours
- o Achievement of about 10% of its terrestrial life span for a particular type of Halogen lamp
- o Operation of that Halogen lamp under conditions similar to the furnace operation

Therefore HALEX should show:

- o Radiative behavior of a Halogen lamp during long-term operation in space
- o Tungsten deposition inside bulb if not retransported on to filament
- o Performance of the Halogen cycle

Therefore HALEX should prove:

- o Feasibility of the mirror furnace concept for long-term operation in space by using Halogen lamps

Therefore HALEX should give:

- o Design inputs for lamp improvement with respect to gas filling, gas pressure and dimension of lamp.

These goals should be verified in a relatively simple and cost effective way by using for the most part existing and proved hardware for a payload which can be carried out within NASA's GAS program.

PAYLOAD CONCEPT

The payload design is governed by the experiment requirements (table 1).

Table 1
HALEX EXPERIMENT REQUIREMENTS:

- VOLUME : ~ 7.5 LITRES
- WEIGHT : ~ 4 KG
- EVACUATED MIRROR COMPARTMENT DURING MISSION
- LAMP CURRENT : ~ 7 A
- LAMP VOLTAGE, CONTROLLED
 - UPPER LIMIT : 8.2 VDC, ± 1%
 - LOWER LIMIT : 7.0 VDC, ± 1%
- OPERATION TIME : AS LONG AS POSSIBLE,
 - ENVISAGED ~ 60 HOURS
- PERMANENT HEAT DISSIPATION OF 65 W
- DATA RECORDING:
 - 8 MIN CONTINUOUSLY AFTER ACTIVATION,
 - THEN 1 SECOND DATA BLOCKS EVERY MINUTE
 - TILL DEACTIVATION
- TIME REFERENCE IN MINUTES

The experiment hardware (figure 1) mainly consists of:

- o Sealed ellipsoidal mirror shell with vacuum port
- o Halogen lamp and lamp holder.
 - The HALEX 45 W flight lamp has been "designed" by OSRAM to ensure a transferability of experiment results to a 300 W lamp. The lamp is filled with 10 cm³ Xenon and admixtures at a pressure of approx. 4,5 bars.
- o 2 photocells for light detection
- o Temperature sensors at photocells, lamp base, heat pipes and intermediate plate
- o 2 heat pipes for heat transfer from lamp base to intermediate plate

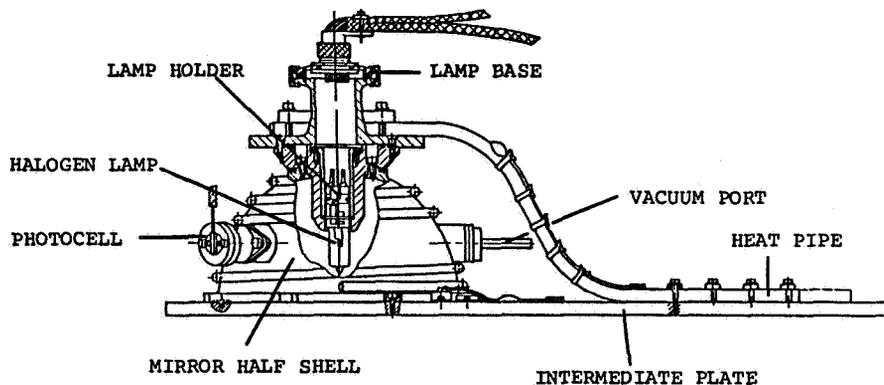


Figure 1

To economically fulfil the experiment requirements, the payload design (figure 2) is based on Kayser-Threde's concept of a standardized and modular GAS payload service system.

Attention was paid especially to:

- o Maximum capacity of weight and volume of batteries in order to obtain the longest possible operation time
- o Data recording in an intermittent mode for approx. 120 hours of operation
- o Evacuation of mirror compartment during flight
- o Sufficient heat transfer by attaching the experiment compartment to the intermediate plate underneath the GAS Experiment Mounting Plate

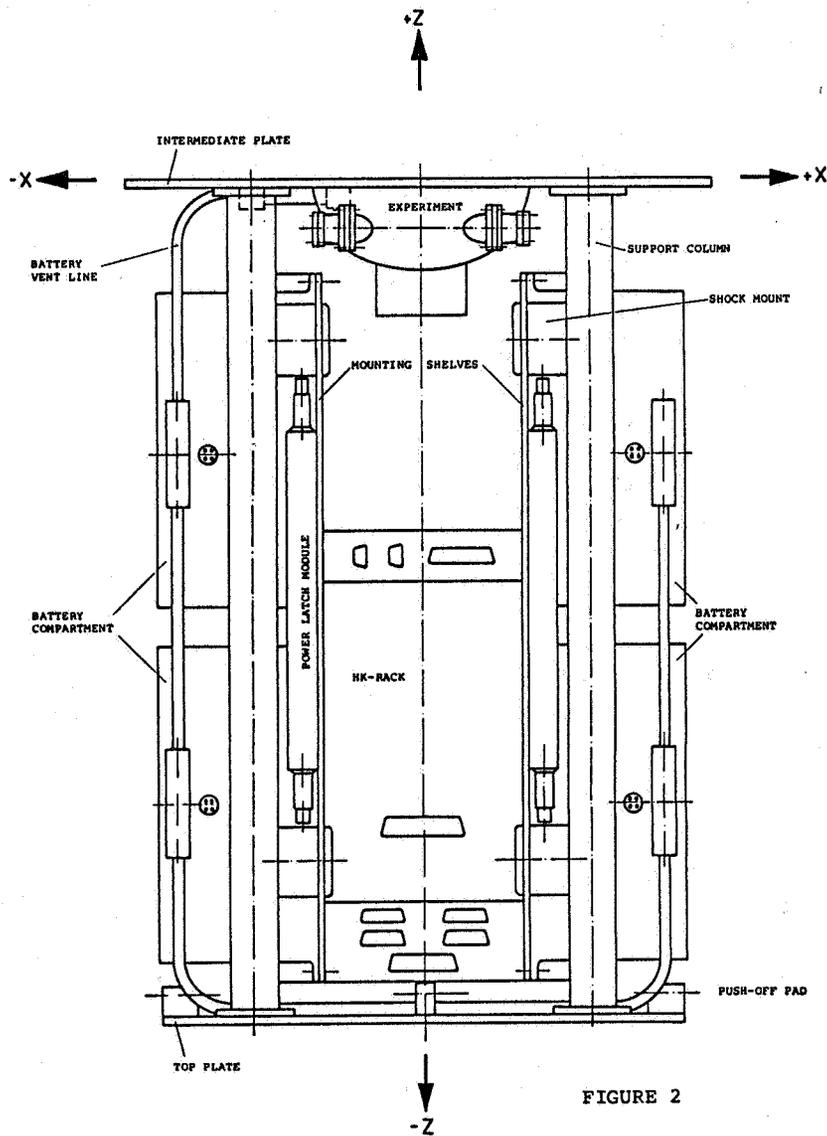


FIGURE 2

Figure 2

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The mechanical support structure consists of:

- o Intermediate plate (with attached experiment)
- o Top plate and attached push-off pads
- o 4 support columns
- o 2 mounting shelves, each attached to 2 support columns via 4 shock absorbers
- o HK or electronics rack

Payload power is supplied by:

- o 8 batteries, 27 VDC at a typical capacity of 18 Ah consisting of
- o 144 silver zinc cells SHV 01500, capacity matched, total energy of 4 KWH, integrated into
- o 4 battery housings (figure 3), pressure tight, nickel plated, temperature switches, H₂ outgassing capability

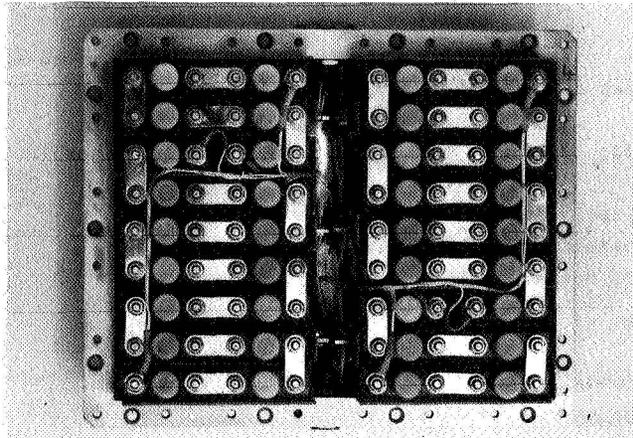


Figure 3

For safety a low voltage power cut-off at 22,3 VDC is incorporated. Regulated DC power is applied to the data acquisition and recording system and the sequencer. Controlled DC power is applied to the Halogen lamp only.

Furthermore the payload consists of:

- o Control subsystem
- o Housekeeping subsystem
- o 12 bit PCM data acquisition system (figure 4) for 12 analog and 4 digital channels at a data rate 5 kBit/sec
- o 2 redundant tape recorders with data capacity of 56 Mbit each

The ground support equipment for payload operation includes:

- o GAS interface simulator and battery simulator
- o PCM decoder K1180
- o Data channel output and display unit
- o HP power supply
- o External sequence stepper
- o Kayser Quick Apple Check out system



Figure 4

OPERATIONAL SCENARIO

As HALEX requires an operation time of approx. 60 hours the payload had to be activated in the very early part of the STS 41 G mission.

The experiment profile (figure 5) shows the three different experiment periods

- o Warm up (180 sec) with soft start (2 sec)
- o Setting of operation point (200 sec)
- o Long term lamp operation (57,9 hours actual)

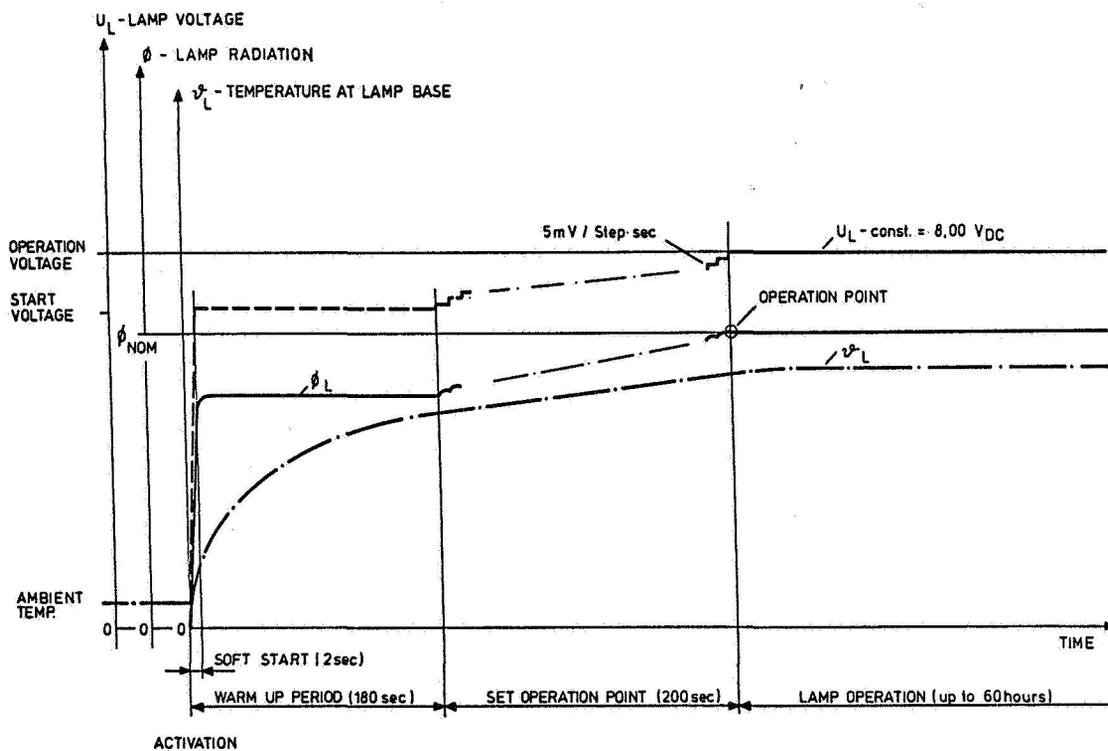


Figure 5. Experiment Profile

The mission profile (table 2) contains major events.

Table 2
MISSION PROFILE STS 41 G

- LAUNCH : OCT/05/84
 - HALEX ACTIVATION : T + 34 H : 13 MIN
 - DEPLOYMENT SIR-B ANTENNA : T + 50 H
 - LAMP SWITCHED OFF AUTOMATICALLY
DUE TO EXITATION OF UPPER TEMP-
ERATURE LIMIT AT HEAT PIPES : T + 89 H : 55 MIN
 - LAMP SWITCHED ON AUTOMATICALLY
AFTER TEMPERATURE DROP : T + 90 H : 25 MIN
 - AUTOMATIC PAYLOAD SWITCH OFF
DUE TO LOW VOLTAGE POWER CUT-
OFF : T + 92 H : 7 MIN
 - HALEX DEACTIVATION : T + 167 H : 48 MIN
 - ORBITER RETURN : OCT/12/84
- TOTAL PAYLOAD OPERATION TIME: 57,9 HOURS

Due to the permanent heat dissipation of approx. 65 Watts a GAS can with a non insulated end cap was used to provide sufficient temperature environment (figure 7).

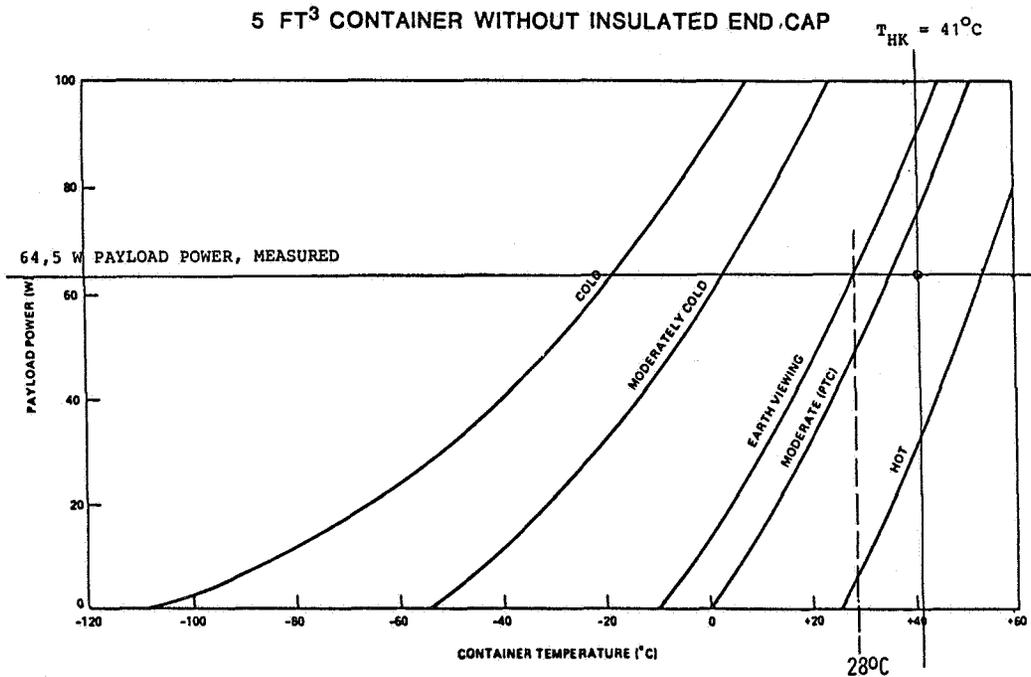


Figure 7. Get Away Special Small Self-Contained Payloads

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However, the unexpected coverage of HALEX by the deployed SIR-B antenna caused a significant temperature increase (figure 8). This lead approx. 56 hours after payload activation to an automatic shut down of the lamp power for roughly 30 minutes before it came back on again. After a total payload operation time of 57,9 hours HALEX was automatically switched off by the low voltage power cut-off.

The mirror compartment was evacuated during ascent and repressurized during descent via a ventline.

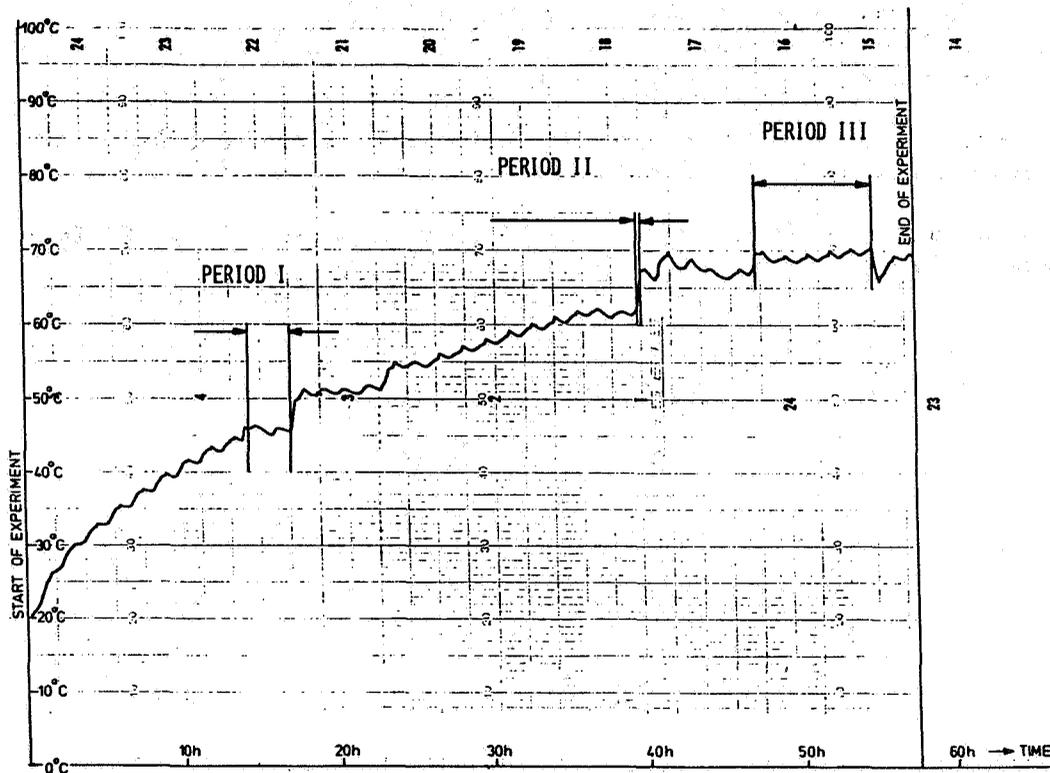


Figure 8

SUMMARY

HALEX performed as predicted within its specified limits and all experiment goals could be fulfilled. The experiment was not affected by the significantly higher temperature environment and the related 30 minutes interruption of lamp activation nor by the extension of the sequencer created time intervals for data acquisition and recording during the last eight hours of operation.

From the evaluation of the flight data it was learnt that:

- o Lamp voltage was constant over the whole experiment period
- o Lamp current was constant
- o Resistance of lamp filament did not change ($<0,1\%$)
- o Photo signals were constant with respect to the radiation input
- o Lamp base temperature showed that lamp bulb temperature was as expected
- o Heat pipe temperatures showed proper function

The inspection of the flight lamp indicated:

- o No detectable disturbances of the Halogen cycle (i.e. no deposit of Tungsten on the bulb)
- o Surface characteristics of the filament as expected (microscopic inspection)

Therefore it can be concluded that:

- o Absence of convection (under microgravity) inside the lamp bulb results in a reduction of convective heat transfer from 5% to about 2%
- o Due to this reduction the filament temperature rises about 20 K resulting in an increase of light efficiency of about 8,8%

The duration of the HALEX flight experiment of 57,9 hours corresponds to approx. 10% of the expected life span for the envisaged future flight lamp.

Concerning all results of the HALEX experiment it can be assumed that a 300 W lamp with similar design will operate in space as well as the HALEX lamp did.

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RESPONSE OF GAS PAYLOAD (G 345 AND G 347)
TEMPERATURES TO VARIOUS ORBITER FLIGHT ATTITUDES

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I. INTRODUCTION

At the beginning of the Get-Away Special (GAS) flight program, the GAS Project flew a Flight Verification Payload on STS-3 to make measurements of the vibrations, acoustic and magnetic environments of the canister, and to obtain thermal profiles of internal and external components of the GAS system. These data were used to verify pre-flight thermal models of the GAS system and results have been published in a GSFC Technical Note, X-732-83-8 (Butler, 1983). On somewhat later flights of the Orbiter, STS-7 and STS-8, the Naval Research Laboratory and the Goddard Space Flight Center jointly developed and flew a GAS payload whose primary objective was to evaluate the performance of ultraviolet-sensitive (Schumann) photographic emulsions in the Orbiter environment, including pre-flight integration, on-orbit exposure to the ambient environment of the Orbiter bay and post-flight conditions before removal of the experiment from the vehicle. These emulsions, used in spectrographs to record solar radiations with wavelengths between 100 Å and 2000 Å, have low gelatin content and no protective gelatin overcoating in order to maximize their UV sensitivity. Consequently, they are extremely sensitive to environmental conditions. Furthermore, they are exposed directly to space during the observations because any intervening protective window or lense would completely absorb the radiations to be studied. Our instrument, which was prepared at the Naval Research Laboratory under the direction of Robert Kreplin (Kreplin et al., 1984 a) also included two thermal sensors, whose output was recorded once an hour with a precision of 0.7° C. The experiment operated nominally on both missions and provided important data confirming that UV - sensitive emulsions could tolerate integration and flight on the Shuttle with relatively little deterioration. The results of these studies have already been reported (Kreplin et al, 1984 b). The temperature measurements also were recorded successfully and are the basis for this paper. They are a useful complement to the thermal observations made on STS-3 and provide additional insight into the reaction of a payload to typical thermal environments that GAS experiments are subject to.

Our GAS experiment and the circumstances of its flights were particularly useful in terms of modeling the thermal response of the GAS canister and our instrument. The Orbiter was held (except for interruptions for star tracker alignment, satellite deployments, etc., in specific attitudes for sufficiently long periods of time that the thermal response of the instrument to each attitude was well measured. Two types of GAS canister end cap, one, the insulated end cap and the other a silverized teflon covered end-cap, were used on the two flights so that a direct comparison of the thermal performance of the two types was obtained.

This report is an attempt to compare these observations with a simple thermal model of the instrument in the GAS canister in order to assess whether such simple models can be useful to experimenters in predicting the thermal response of their payloads.

II. Instrumentation and Observations

A crosssectional view of the instrument is shown in Figure 1. The EMP is at the top of the diagram. The experiment was directly mounted to it using two thick aluminum support rings. The major components of the experiment, in terms of thermal mass, were the doughnut - shaped battery container mounted directly to the support rings, the cylindrical Main Experiment Container, carrying the photographic emulsion samples, and also mounted directly to a support ring, and the cylindrical electronics box attached to the Main Experiment Container.

Temperatures were measured at two locations using thermistor networks (thermilinear component YSI No. 44203 incorporating a bead thermistor (YSI 44018)). These were supplied by the Yellow Springs Instrument Co. One sensor was located inside the Main Experiment container on the flange supporting the inner film cylinders. The second was located on one of the brackets supporting a film cylinder on the outside of the Main Experiment Container. The location of these two sensors, effectively at each end of the Main Experiment Container, provided a means of measuring temperatures and temperature gradients along the Main Experiment Container and inner film cylinders.

The data obtained for flights of this instrumentation on STS-7 and STS-8 are shown in Figures 2 and 3 (See also Kreplin et al, (1984). Data from the two sensors (shown by solid and dashed lines) are very nearly the same. The small differences are not consistent, and may be due to errors of roundoff in the digitization, etc. On each flight the experiment power was turned on internally using altitude switches (Altitude/ Absolute Pressure Switch E45C-72 (50,000 ft. turn-in) backed-up with E45C-73 (70,000 ft turn-on). These switches were obtained from Precision Sensors, Inc.

The first flight of our instrument, designated G 345, began with an extended period of about 70 hours during which the Orbiter flew with its bay facing the earth (-Z Local Vertical, abbreviated to -Z LV). This orientation was held for 70-75% of the time, being interrupted by scheduled satellite deployments, etc. In this attitude the payload cooled rapidly -8.4⁰C, consistent with but slightly lower than the nominal - 5⁰C equivalent sink temperature expected for this orbiter orientation. Following this period, the Orbiter attitude was changed so that the bay faced deep space (equivalent sink temperature of - 100⁰C), for about 2.5 hrs during deployment of the European SPAS subsatellite and the payload temperature dropped. A more substantial temperature drop was recorded between 92 and 100 hours Mission Elapsed Time (MET) when the Orbiter bay again faced deep space during the SPAS operations and retrieval. The payload temperature dropped for about 4 hours after the Orbiter returned to -Z Local Vertical attitude, then began to

increase again, approaching the steady-state temperature of -8.4°C recorded earlier in this flight.

Eventually, at landing, the temperature rose, with no overshoot, and showed a daily variation about 5° before the temperature recording was terminated.

The flight of STS-8, carrying the same experiment but a different, insulating, GAS canister end-cap and designated G 347, also began with an extended period of 34 hrs. of $-Z$ Local Vertical orientation. The payload temperature dropped much more slowly than for G-345, although the flight began with the same Orbiter attitude. This was the result of adding the insulating end cap. After 34 hours MET the Orbiter was re-oriented to a tail to sun attitude during which the bay of the Orbiter always faced away from the earth, i.e. into deep space. This attitude is called $-X$ Solar Inertial, Orb Rate, abbreviated to $-X$ SI, Orb Rate, as the Orbiter rotates once an orbit about its longitudinal (X) axis. Neither direct nor earth-reflected sunlight illuminated the Orbiter bay during this attitude for which the sink temperature again was -100°C . The payload temperature dropped rapidly, continuing to do so for about 20 hours after the $-Z$ Local Vertical attitude was re-established at 48 hours MET. The remainder of the flight was spent for the most part at $-Z$ and $+X$ Local Vertical (Orbiter nose $(+X)$ or tail $(-X)$ pointing toward the earth) and the payload equilibrated at about -5°C , the nominal sink temperature for the $-Z$ Local Vertical attitude.

III. Thermal Modeling of the Payload

Thermal modeling of the payload, for purpose of obtaining a comparison with the in-orbit temperature measurements, began by adopting a simplified thermal model, shown in Figure 4. The major structural components of the instrument were identified as:

1. The battery pack baseplate of T6061 aluminum, and attached to the GAS Experiment Mounting Plate (EMP) via two heavy aluminum (T6061) spacer rings.
2. The battery pack itself, the housing of aluminum and carrying 34 alkaline "D" cells held in a fixed assembly by machined teflon rings.
3. The Main Experiment Container, including all inner and outer film cylinders, all control valves, etc., made primarily of stainless steel.
4. The electronics box, the housing of which was made of T6061 aluminum, together with an electronics module containing components, mounting boards, etc. For purpose of the model, the electronics box was assumed to be 50% aluminum and 50% fiberglass.

All aluminum experiment surfaces were iridited (for which we assumed an emittance of 0.09, the mean of values given by Butler) and the stainless steel was passivated. An emittance of 0.17, typical for mechanically polished stainless steel, was used (Wolfe, 1965).

As an initial step, in an attempt to simplify further calculations, the heat loss of the individual major components was estimated using standard equations for conductive and radiative transfer given by Butler (Figure 5). Modeling of the thermal response of the instrument was simplified because internal power dissipation was extremely low (12 - 13 milliwatts) and could be neglected, and internal convection was not present, as the experiment was exposed to the vacuum of space through a purge port in the Experiment Mounting Plate (EMP). Radiative view factors for infinite concentric cylinders were used as being most appropriate for this experiment. For this geometry the effective emittance is given by:

$$\epsilon_{1,eff} = \epsilon_1 F_{2-1} = \frac{1}{\frac{1}{\epsilon_1} + \frac{A_1}{A_2} \left(\frac{1}{\epsilon_2} - 1 \right)}$$

where A_1 and A_2 are the areas of the two concentric surfaces and ϵ_1 and ϵ_2 are their emittances. (Wolfe, 1965). The estimated heat loss and consequent temperature drop per hour for two typical temperature differentials is shown in Figure 6. The very good conductive coupling between the GAS EMP and the base of the battery pack baseplate via the very heavy aluminum spacer rings means that the baseplate temperature will very closely track the EMP temperature. For subsequent calculations we therefore assumed that this baseplate was effectively a portion of the GAS canister and increased its effective mass accordingly. Likewise, the battery pack temperature itself will also track the EMP temperature quite well and can be assumed to be thermally a part of the EMP.

The remaining components of the experiment were less well thermally coupled to the EMP and it became clear from the model that both conductive and radiative losses to the GAS canister had to be considered. Radiative losses by the electronics box were particularly important for cooling that module. Conductive coupling to the EMP was low because of the low conductance of the thin walled stainless steel Main Experiment Container. The conductive losses given in Figure 6 are an upper limit as the contact resistance of the various mechanical joints has not been included - Note that the temperature drops in the Main Experiment container and electronic box are predicted to be within a degree of one another as is confirmed by actual temperature readings of the two probes, so that net heat transport between the Main Experiment Container and electronics box could be considered to be negligible.

Using these preliminary results, we simplified our thermal model, i.e the battery pack and its baseplate became a part of the GAS canister and the thermal contribution of the electronics box to the temperature of the Main Experiment Container was assumed to be negligible. Hence, a final thermal model of the Main Experiment Container included only radiative and conductive interaction with the GAS canister.

IV. Results

This simple model of the experiment was set up on a desk calculator using external boundary conditions for the flight of G 347 (Figure 7). A sink temperature of -5°C (268K) was used for the -Z LV attitude and -100°C (173K) during -X SI with bay facing away from the earth. Temperatures were calculated at hourly intervals and sink temperatures changed in accordance with the as-flown flight plan (short intervals of orbiter maneuvering being disregarded). Criteria used to evaluate the accuracy of the model were agreements between calculated and observed temperatures at the end of the -Z LV interval at 34 hr MET, at the end of the -X SI interval at 48 hr MET, and the timing and value of the minimum temperature after return to the -Z LV attitude.

Using nominal thermal properties of the experiment materials the calculated temperature profile for the G 347 flight (Figure 8) was substantially below the measured values. It appears, from Figure 8, that the Main Experiment Container is not as tightly coupled to the GAS canister as we had calculated. If we decouple it (reduce both radiative and conductive losses to the canister by a factor of 3.3) we can then match the observed temperatures at the end of -Z LV and -X SI, Orb Rate, but not the timing of the temperature minimum, which now occurs later in time because of the increased thermal isolation of the experiment. This case is shown in Figure 9.

We can now use this thermal model, with reduced experiment conductance and emittance to the GAS canister, to predict the thermal behavior of the experiment during the STS-7 flight. The results are shown in Figure 10. The calculated temperatures are appreciably higher than observed values. Two cases are shown in Figure 10. Case A was obtained using a nominal sink temperature of -5°C . A more appropriate sink temperature appears to be -8.4°C , the value used in calculating case B. In either case, the experiment temperature does not decrease rapidly enough because of the decoupling of the experiment from the GAS canister assumed (in Figure 9) to force a match to G 347 temperatures.

We can explore the possibilities of altering the emittance of the GAS canister as a way of improving the agreement between model and observations. One such combination of reduced experiment emittance and conductance reduced by a factor of 2.5) together with slightly reduced canister emittance (from 0.065 to 0.055) is shown in Figure 11. The agreement with experimental data is not appreciably better than for our earlier case (Figure 9). However, if this second model is used for the G 345 flight, the results, in Figure 12, are appreciably closer to the observations than the earlier calculations were.

There may be several other reasons for the discrepancy between G 345 data and predictions. For one, the Orbiter was flown in the thermally benign -Z Local Vertical attitude only about 70-75% of the tracker/navigational base re-calibrations, etc., could have been at less benign, i.e., colder thermal orientations, producing a faster than expected temperature drop on average. Yet another reason might be the crudity of the thermal model that we have used here.

In conclusion, the procedures outlined in the GAS Thermal Design Summary appear to give valid results when applied to an actual flight experience. They will provide useful predictions and should be used by any GAS experiments with a critical thermal requirement for his or her payload. In addition, we found some evidence that the average canister emittances may be slightly different from values given in the GAS Thermal Design Summary, but more data must be taken and evaluated before this can be established with certainty.

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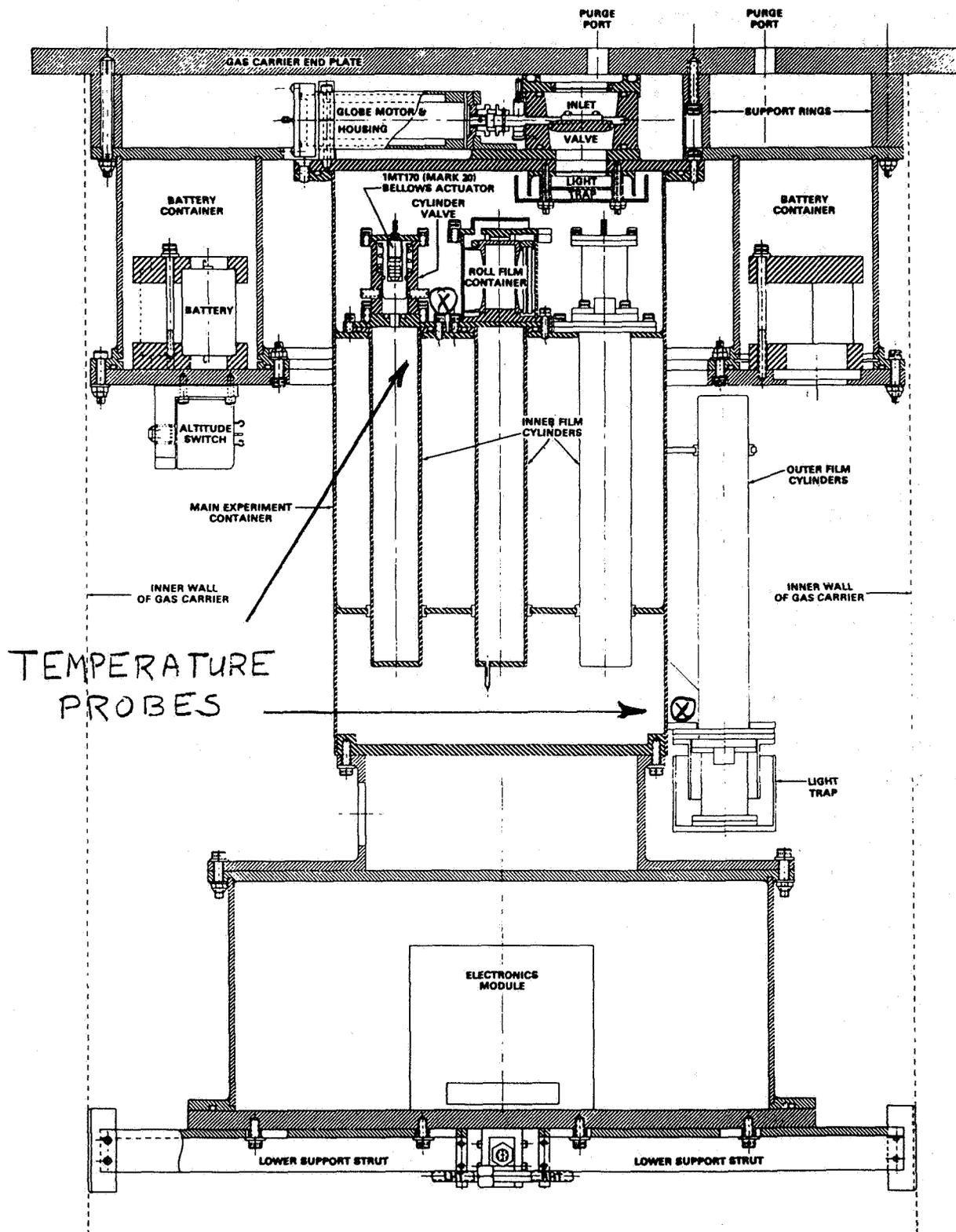


Figure 1. Cross-sectional view of the G.S.F.C./N.R.L. GAS Experiment for evaluating UV-sensitive photographic emulsions in the Orbiter environment.

TEMPERATURE PROFILE FOR GAS PAYLOAD G345 USING AN END CAP WITH SILVER TEFLON COATING

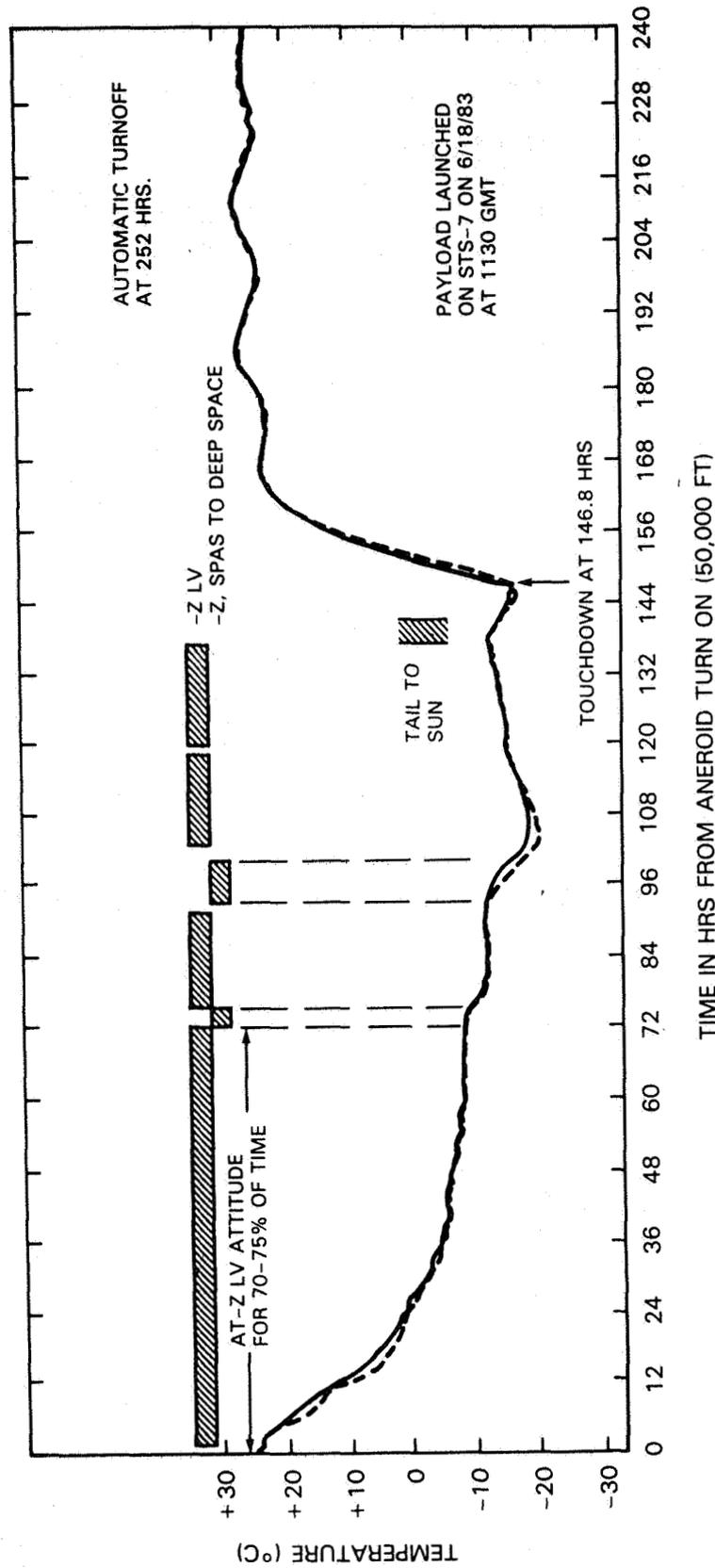


Figure 2. Experiment temperature data obtained from two sensors during the flight of GAS Payload G 345 on STS-7. Orbiter attitudes maintained during the flight are also shown.

TEMPERATURE PROFILE FOR GAS PAYLOAD G347 USING AN INSULATING END CAP

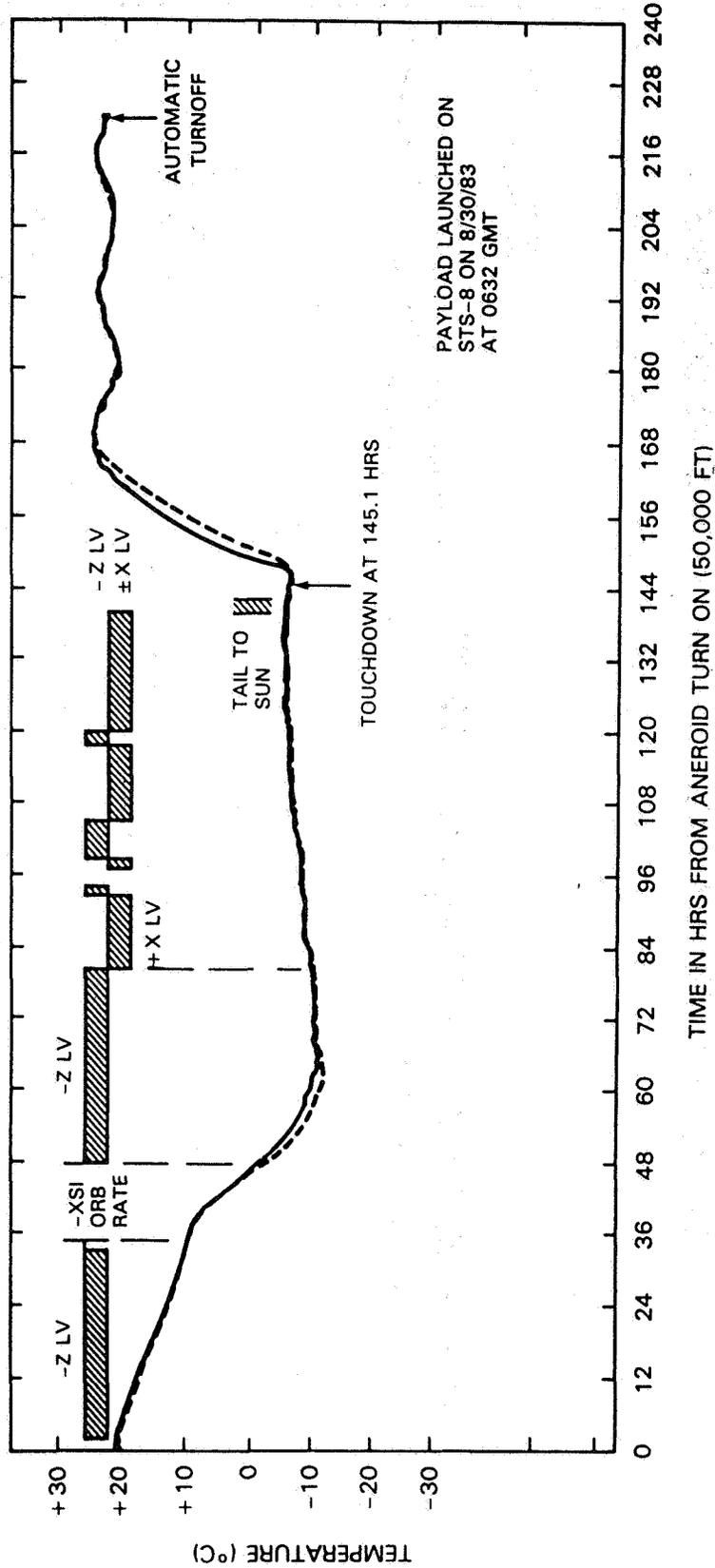


Figure 3. Experiment temperature data obtained from two sensors during the flight of GAS Payload G 347 on STS-8. Orbiter attitudes maintained during the flight are also shown.

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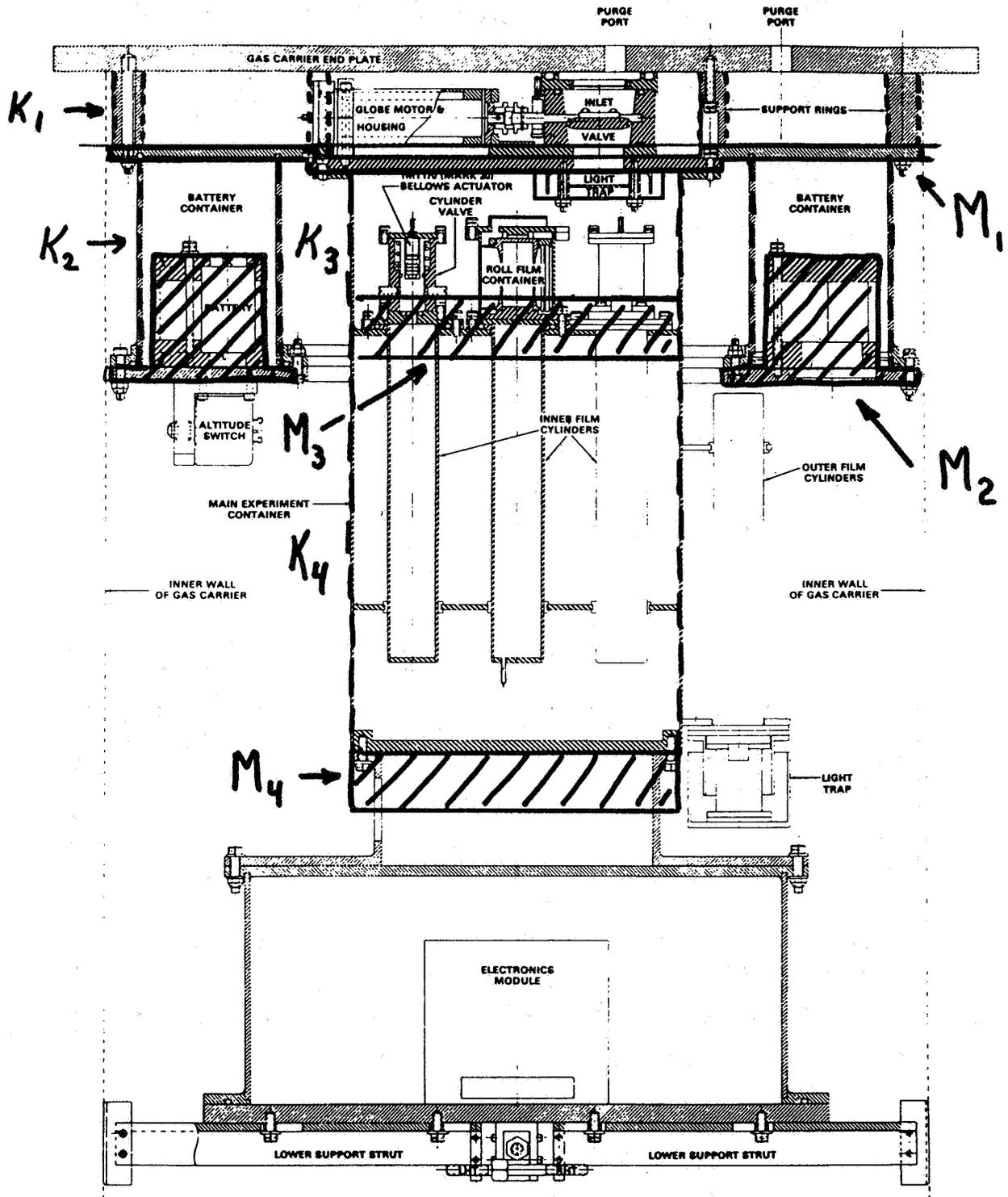


Figure 4. Simplified model of GAS experiment showing lumped subsystem masses used in our analysis.

$$Q_{\text{LOST}} = Q_{\text{COND}} + Q_{\text{RAD}} + Q_{\text{CONV}} = MC_p \Delta T$$

$$Q_{\text{COND}} = \frac{K_i A_i}{L_i} (T_i - T_{\text{GAS}}) \Delta t = K_i (T_i - T_{\text{GAS}}) \Delta t$$

$$Q_{\text{RAD}} = \sigma A_i \epsilon_i F_{2-1} (T_i^4 - T_{\text{GAS}}^4) \Delta t$$

$$Q_{\text{CONV}} \equiv 0$$

$$K(\text{Al}) = 155.8 \text{ W/m K}$$

$$K(\text{S.S.}) = 16.3$$

$$\epsilon(\text{Al}) = 0.09 \text{ (iridited)}$$

$$\epsilon(\text{S.S.}) = 0.17 \text{ (passivated)}$$

$$C_p(\text{Al}) = 0.21 \text{ cal/gm K}$$

$$C_p(\text{S.S.}) = 0.11$$

Figure 5. Equations and data used in estimating thermal losses of experiment subsystems.

i	Subsystem	K_i	$(MC_p)_i$	$T_i - T_{\text{GAS}} = 5^\circ$			$T_i - T_{\text{GAS}} = 10^\circ$		
				Q_{COND}	Q_{RAD}	ΔT	Q_{COND}	Q_{RAD}	ΔT
1	Battery Pack Base plate	34.6	160	6.2×10^5	-	5°	12.4×10^5	-	10°
2	Battery Pack	0.74	2700	13,300	290	$\approx 5^\circ$	26,600	600	$\approx 10^\circ$
3	Main Experiment Container	0.042	1220	770	460	$\approx 1^\circ$	1550	980	$\approx 2^\circ$
4	Electronics	0.017	3500	290	360	$\approx 0.2^\circ$	580	765	$\approx 0.4^\circ$

UNITS

K : calories / K

Q : calories / hr

M : grams

T : $^\circ\text{C}$

C_p : calories / gram K

Figure 6. Estimates of thermal losses and temperature changes of GAS Subsystems for two typical experiment/canister temperature differentials.

$$T_{GAS}(t+\Delta t) = T_{GAS}(t) - \frac{3600}{(MC_p)_{GAS}} \left\{ (\sigma A \epsilon_{eff})_{GAS} (T_{GAS}^4(t) - T_{EXT}^4(t)) \right. \\ \left. - (T_{EXP}(t) - T_{GAS}(t)) (K_3 + K_4) \right. \\ \left. - (T_{EXP}^4(t) - T_{GAS}^4(t)) (\sigma A \epsilon_{eff})_3 + (\sigma A \epsilon_{eff})_4 \right\}$$

$$T_{EXP}(t+\Delta t) \equiv T_3(t+\Delta t) = T_{EXP}(t) - \frac{3600}{(MC_p)_3} \left[K_3 (T_{EXP} - T_{GAS}) \right. \\ \left. + (\sigma A \epsilon_{eff})_3 (T_{EXP}^4(t) - T_{GAS}^4(t)) \right]$$

Where subscripts are defined in Fig. 6 , and

$$T_{EXT}(-Z LV) = 268 K ; T_{EXT}(-X SI) = 173 K$$

Figure 7. Equations used in modeling thermal performance of Main Experiment Container and GAS canister for G 345 and G 347.

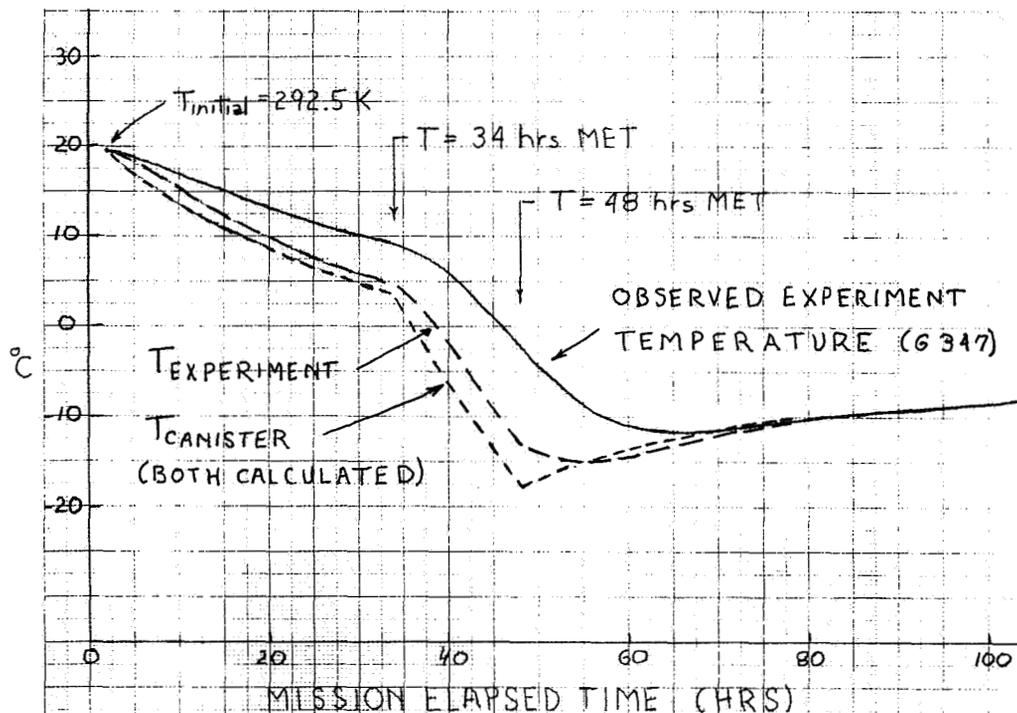


Figure 8. Comparison of calculated experiment and canister temperatures with observed G 347 experiment temperatures using nominal thermal properties for experiment and canister.

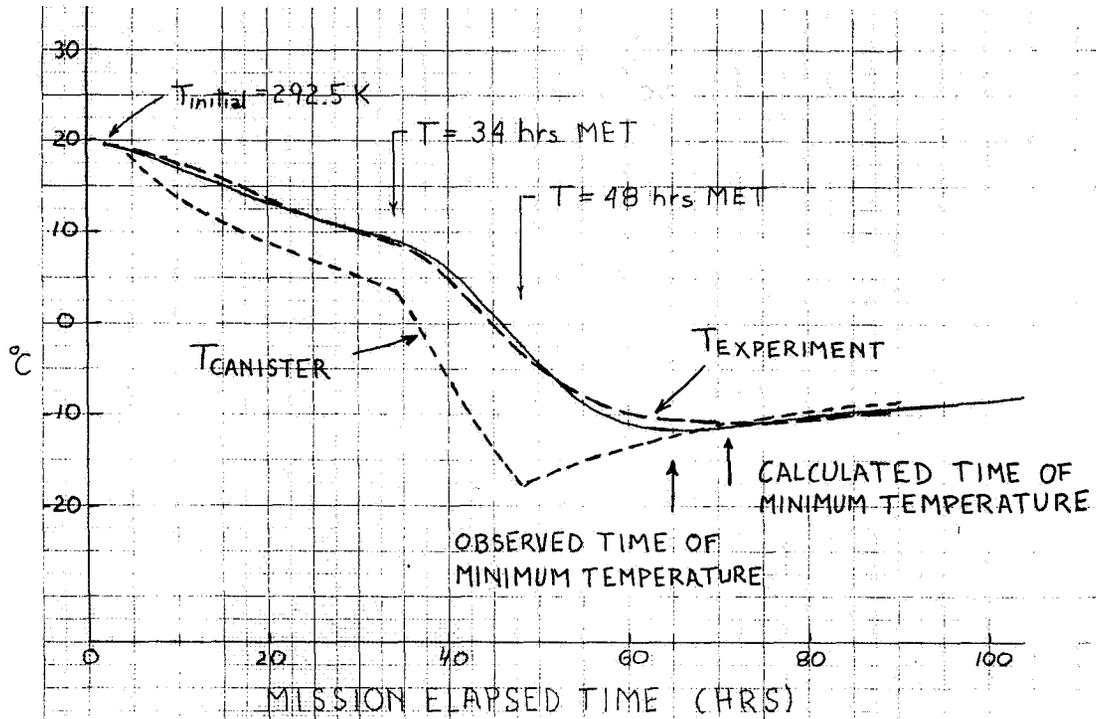


Figure 9. Comparison of G 347 calculated and observed temperatures using reduced experiment thermal conductance and emittance (0.3 of nominal values) but nominal canister emittance.

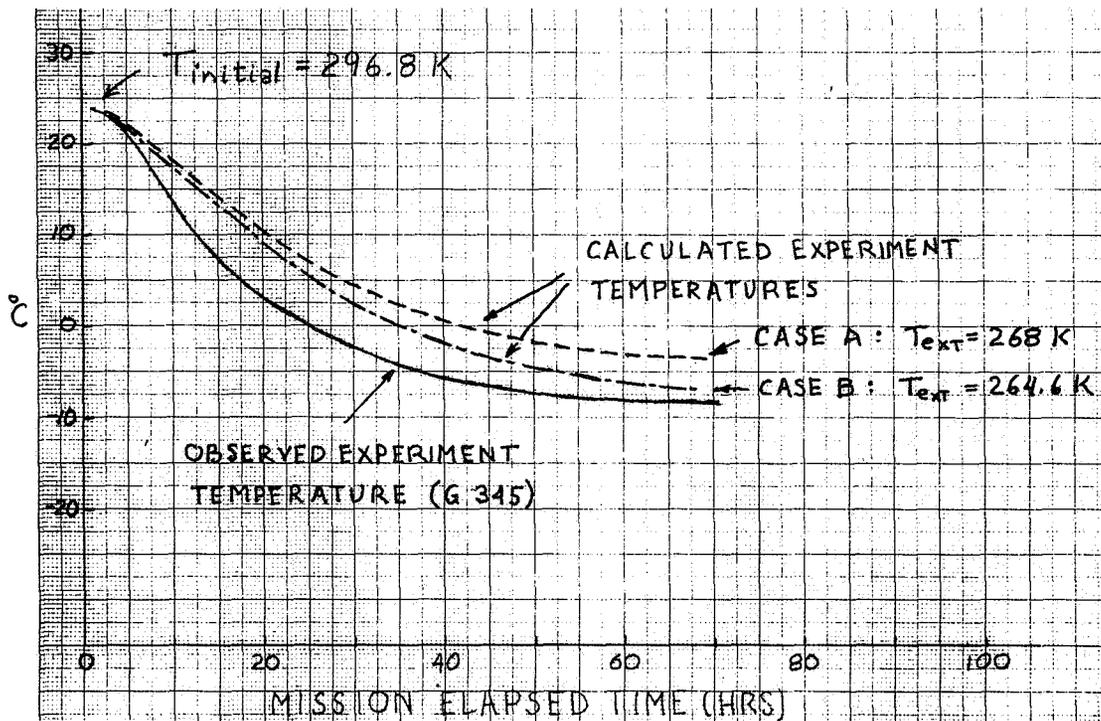


Figure 10. Comparison of G 345 calculated and observed temperatures using reduced thermal conductance and emittance (0.3 of nominal values) and nominal canister emittance (0.16).

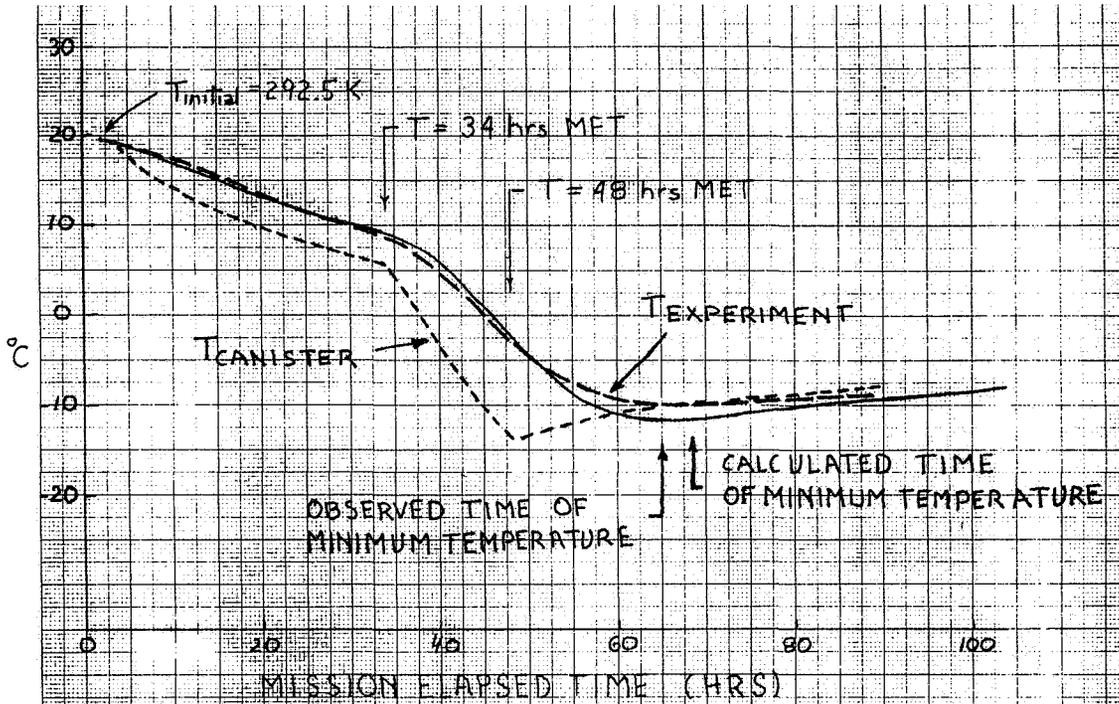


Figure 11. Comparison of G 347 calculated and observed temperatures using reduced experiment thermal conductance and emittance (0.4 of nominal values) and slightly reduced canister emittance (0.055 rather than 0.065).

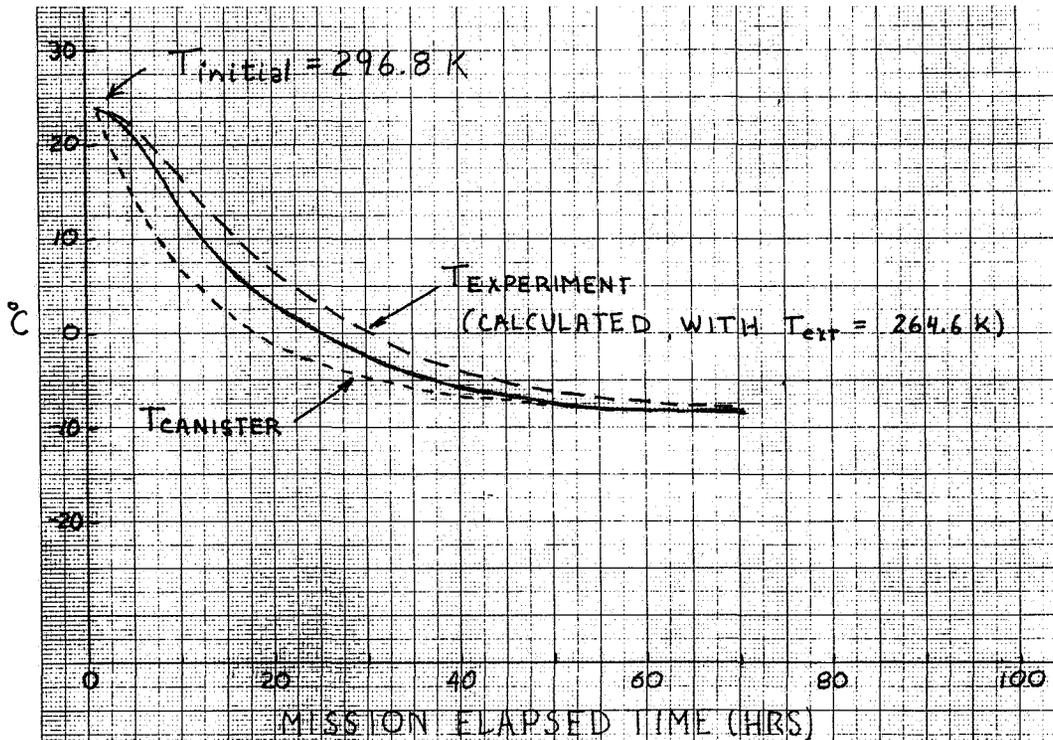


Figure 12. Comparison of G 345 calculated and observed temperatures using reduced experiment thermal conductance and emittance (0.4 of nominal values) and nominal canister emittance (0.16).

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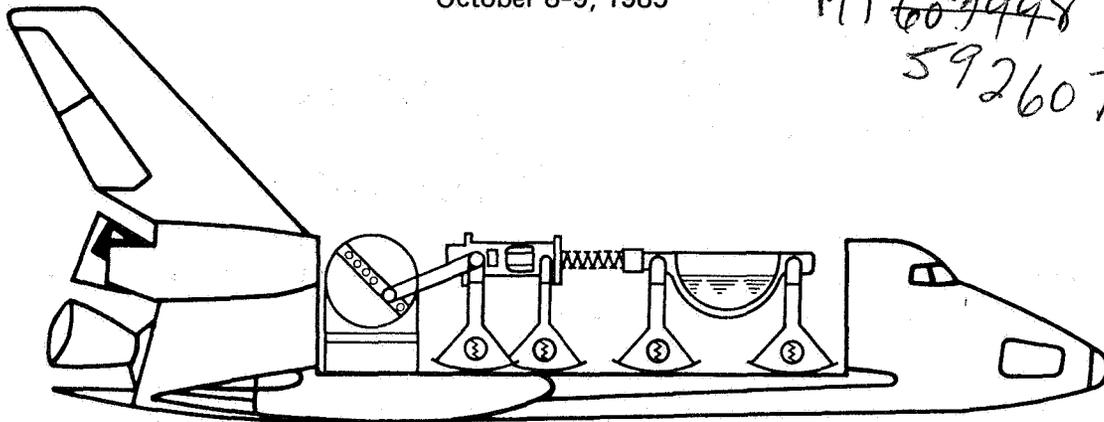
GAS PAYLOAD NO. G-025
STUDY OF LIQUID SLOSHING BEHAVIOUR IN MICROGRAVITY

8974

Prepared by C. R. Gilbert
MBB-ERNO Bremen

Presentation at the Second Get Away Special (GAS) Experimenters Symposium
NASA/Goddard Space Flight Center
October 8-9, 1985

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1. SUMMARY

GAS Payload No. G-025, which flew on Shuttle Mission No. 51-G, examined the behaviour of a liquid in a tank under micro-gravity conditions. The experiment is representative of phenomena occurring in satellite tanks with liquid propellants. A reference fluid in a hemispherical model tank will be subjected to linear acceleration inputs of known levels and frequencies, and the dynamic response of the tank liquid system was recorded.

Preliminary analysis of the flight data indicates that the experiment functioned perfectly. The results will validate and refine mathematical models describing the dynamic characteristics of tank-fluid systems. This will in turn support the development of future spacecraft tanks, in particular the design of propellant management devices for surface tension tanks.

The experiment was mounted on a Payload Support System (PASS) flight unit, identical to the system designed and developed by MBB/ERNO for the Federal German materials science project MAUS. PASS is a standardized structure, power supply, and data processing unit available commercially to GAS users.

2. INTRODUCTION AND OBJECTIVES

The trend to increased use of surface tension as a mechanism for control of liquid propellants in spacecraft has stimulated development of mathematical models which can adequately describe the behaviour of the propellant under a variety of environmental conditions. Of special interest are the behavioural characteristics of fluids under micro-gravity conditions, and the effect of operationally induced disturbances on propellant management. GAS Payload No. G-025 utilized the opportunities of the Get Away Special program to the full, to make a real contribution to product research and development being carried out by MBB/ERNO.

The purpose of this experiment was to generate data regarding liquid sloshing in partially filled tanks under micro-gravity conditions. A transparent, hemispherical tank was subjected to oscillatory disturbances corresponding to inputs used for theoretical simulation and analysis. The validity of such mathematical models will be significantly reinforced by comparison with empirical data, and areas not amenable to modelling will benefit from a review of their behaviour under realistic conditions.

The overall objectives consisted therefore of two parts:

- 1) investigation of the dynamic behaviour of the tank fluid system, and subsequent correlation with mathematical models, and
- 2) investigation of the orientation and stabilization of the fluid (propellant) under the influence of a propellant management device (PMD) especially with respect to outflow cases.

Specific aims were:

- o definition of all significant natural frequencies and damping of the fluid in the low frequency range to 6 Hz;

- o determination of the generalized propellant mass, to facilitate necessary analysis assumptions;
- o determination of sloshing forces and the pressure distribution in the fluid and tank;
- o observation of the dynamic behaviour of the propellant under quasi zero-gravity conditions;
- o observation of the fluid orientation stability effect, including the shape of flow during critical operational phases.

3. PAYLOAD CONFIGURATION

This GAS payload is of modular construction. A standardized support system provides physical accommodation, electrical power, experiment command and control functions, and data recording. Space is assigned, in the form of two horizontal mounting platforms, for the integration of experiment hardware. The volume so provided is about half that of the standard 5 cu.ft. GAS container.

3.1 Payload Support System

The tank experiment assembly is mounted on a standard payload support system (PASS) which meets NASA requirements for payloads flying in Get Away Special containers. PASS can accommodate a wide variety of GAS-type experiments, and offers a range of services to experimenters within a framework of standardized interfaces.

A unitary structure provides primary physical location and support for experiments. Within this structure are provisions for an electrical power supply, in the form of batteries, and electronic assemblies which regulate the operation of the experiment and record the data produced.

The structure is attached to the GAS container experiment mounting plate via an adapter ring. Six longitudinal posts are located equidistant from each other around the adapter ring and carry two further platforms for equipment and experiments.

The battery assembly is also attached to the adapter ring, and contains 80 silver-zinc cells providing 1.8 kWh of energy at 30 V min. Main power supplied to the experiment is unregulated but limited by fuses. A power conditioning unit provides regulated outputs at ± 15 V and + 5 V, 100 mA, which can be utilized, for example, for sensor operation and command signals.

Experiment control and data management is performed by a centralized electronic assembly, mounted on the underside of the lower experiment platform. The overall function of this assembly is carried out by two subassemblies: the experiment command unit, and the data acquisition unit.

The command unit produces software-controlled on/off commands for up to 12 experiment-dedicated channels, acts as a function generator (4 analog channels available), surveys experiment-generated parameters and initiates ongoing incremental changes to the experiment as required.

The experiment run is executed and controlled by the sequential switching of pre-selected command channels. Besides this pre-programmed switching function, certain events can be initiated if, for example, pre-determined boundary conditions are exceeded.

The control sequences can run for up to 200 hours. The program operates on an 'action point' basis, in which the data status is reviewed and the appropriate commands are issued. The shortest time interval between action points is 1 second, and the total sequence can handle up to 599 action points, with a corresponding command storage capacity of 4 Kbytes. If a particularly complex program is required, the memory available can be best utilized by grouping as many control activities together as possible at the same action point.

The data acquisition unit consists of a microprocessor-controlled multiplexer unit with digital and analog inputs. The analog signals are converted internally into digital data with a 10-bit word format for subsequent processing and storage. Of the 32 analog inputs, 16 are available for the experiment. An additional thirty digital status channels are available for suitable data inputs.

The data acquisition unit also carries out the selection of measurement data for special control activities, together with the surveillance of limit-sensitive data. Where necessary, data is transferred to the command unit, which then issues the appropriate control signals to the experiment.

Data which is to be retained is coded into PCM format and assembled to the channel designation, making a complete word of 16 bits. It is then fed into a buffer and held there, together with a time signal, until the buffer is full. At that time the buffer contents are read into a serializer, which in turn triggers a tape recorder and feeds the serialized data stream to the recording head at 2.5 kHz. The buffer continues to re-fill itself with incoming data.

The payload support system also provides a housekeeping subsystem, which monitors a range of signals relating to the well-being of the system itself and basic services.

The signals cover

- o all battery voltage, 10 channels
- o pressure in the GAS container
- o pressures in the Silver-Zinc battery housings, 2 channels
- o temperature within the GAS container
- o 3-axis acceleration levels on the upper experiment platform.

The accelerometer range is $\pm 5 \times 10^{-3}$ g, with a resolution of 10^{-5} g.

The acquisition rate for housekeeping data is variable. Typical values are 1 measurement per 10 seconds for voltage and temperature, 1 measurement per second for pressure and acceleration. High-rate bursts up to 30 Hz are possible over short, pre-determined periods. However, a balance has to be struck between the experiment data requirements and the need for housekeeping data.

Changes of acquisition rates for selected channels can of course be pre-programmed to occur at selected phases in the experiment run.

The physical space available for experiments is related to the position of the experiment platforms. A maximum height of 8.5 inches (218 mm) is available between the platforms; the corresponding height above the upper platform is then 6.7 inches (170 mm). In this position the upper floor is not obstructed by the support posts, and the full dynamic envelope is available, that is 19.75 inches (501 mm) diameter. Due to clearance requirements, however, the allowable hardware diameter is limited to 18.8 inches (478 mm).

The upper experiment platform height is adjustable, to provide the optimum accommodation for individual experiments. The adjustment range is nearly four inches in 1-inch increments. When the upper floor is lowered, the envelope is additionally limited by the support posts projecting above the plane of the platform. Cut-outs can be made in the upper platform to accommodate experiments with particular height requirements, provided that the approach is structurally feasible.

3.2 Experiment Assembly

The central feature of the experiment is a hemispherical plexiglas tank suspended in a rocking mechanism such that purely linear, nearly frictionless oscillation can be induced within a limited displacement range. Mechanical motion is initiated by a stepping motor and crankshaft connected via a pivot assembly to a transmission coupling system. The speed of the motor and hence the oscillation frequency can be varied during operation of

the experiment. Similarly, the moment arm of the pivot assembly and hence the linear displacement can be varied.

The transmission coupling system, which transfers the linear oscillation generated by the drive train to the tank, comprises a concentric rod and helical spring assembly. One end of the rod is permanently attached to the tank, but the other end can either be held fast or left free. This is achieved by an electro-magnetic clutch. For continuous operation (i.e. sinusoidal oscillation), the clutch is engaged; for pulsed operation, the clutch is released following the pulsed input allowing the tank oscillations to decay.

The hemispherical plexiglas tank and the transmission coupling system are supported by rocking assemblies which assure that a one-dimensional oscillation is experienced by the tank and contents, with negligible transverse components. The foot of each rocker has a cylindrical profile which provides a translation amplitude of ± 20 mm to the tank. Each rocker is held in place on the support system experiment platform by tangential leaf-spring yoke fittings. The rockers are additionally held down to the platform with helical springs, which also serve to give the assembly a neutral bias. The tension in one pair of these springs can be adjusted during the experiment to change the natural frequency of the moving assembly.

Besides the hemispherical test tank, initially 50 % filled with fluid, a second evacuated aluminum container is provided, connected to the test tank by a tube and solenoid valve. Outflow of propellant from the test tank is achieved by opening the valve at specific times.

A high-speed camera is used to make a visual record of the fluid behaviour at critical phases. The camera is mounted on the propellant transfer tank and views the test tank and contents via a mirror. A source of diffuse light is provided for illumination of the fluid.

Generation of the necessary data is accomplished by a variety of sensors located at strategic points on the experiment assembly. The instrumentation consists of the following sensors:

<u>Parameter</u>	<u>Sensor</u>
Output force of the transmission coupling system	force transducer
Input force to the tank	force transducer
Acceleration of the transmission coupling system	accelerometer
Acceleration of the tank	accelerometer
Pressure within the tank (4 locations)	piezzo resistive transducer
Temperature of the fluid	thermocouple

In addition to the basic program control and data processing facilities provided by the PASS flight unit, an experiment dedicated electronics unit is used for control of the three electric motors, the camera and illumination source, the electro-magnetic transmission coupling actuator, the solenoid fluid transfer valve, the instrumentation sensors and position switches. This unit is driven by the pre-programmed commands issued by the PASS control unit, and data signals generated by the instrumentation sensors are conditioned and relayed back to the support system for processing and storage. The assembly is mounted on the lower experiment platform.

4. EXPERIMENT OPERATION

In-flight operation of the experiment was controlled by a program in the PASS command module. The experiment run was divided into five distinct phases each with specific operating modes and objectives, as mentioned in the introduction. An important requirement for the experiment is the condition that the fluid in the tank be in a state of equilibrium, and stationary

at the start of vibration inputs. Therefore time was included between operations for the fluid to reach this condition. Of the total run time of just over 3 hours, 60 % was occupied by active operations, the rest being reserved for fluid stabilization requirements.

The five operation phases were of course designed to provide an overview of the natural behaviour of the tank/fluid system in micro-gravity, and to enable a systematic examination of the fluid dynamics under the combination of several variable parameters. The variables were:

- o oscillation/pulse frequency
- o oscillation/pulse amplitude
- o tank vibration stiffness
- o fluid quantity

The major portion of the active operations was assigned to data production by transducers and sensors. The time allowed for filming was very limited, because the high speed employed rapidly used up the film; hence only critical phases were filmed, where sensor data could not provide an adequate picture of the fluid behaviour.

Although the film returned is therefore of limited duration, the opportunity to view the fluid characteristics is a valuable supplement to the considerable amount of sensor data generated.

5. CONCLUSIONS

The operation operated perfectly, and generated a considerable quantity of data which has still to be analysed in depth. In principal, the underlying mathematical models developed by MBB/ERNO for the design of surface tension tanks have been supported by the experiment results, and the validity of current product design features has been confirmed. The results will provide additional insight into the effect of fluid sloshing on spacecraft attitude control systems.

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The considerable investment represented by this experiment has been amply rewarded by these flight results. At least one re-flight is planned, with a modified experiment run profile, to examine certain aspects in more detail.

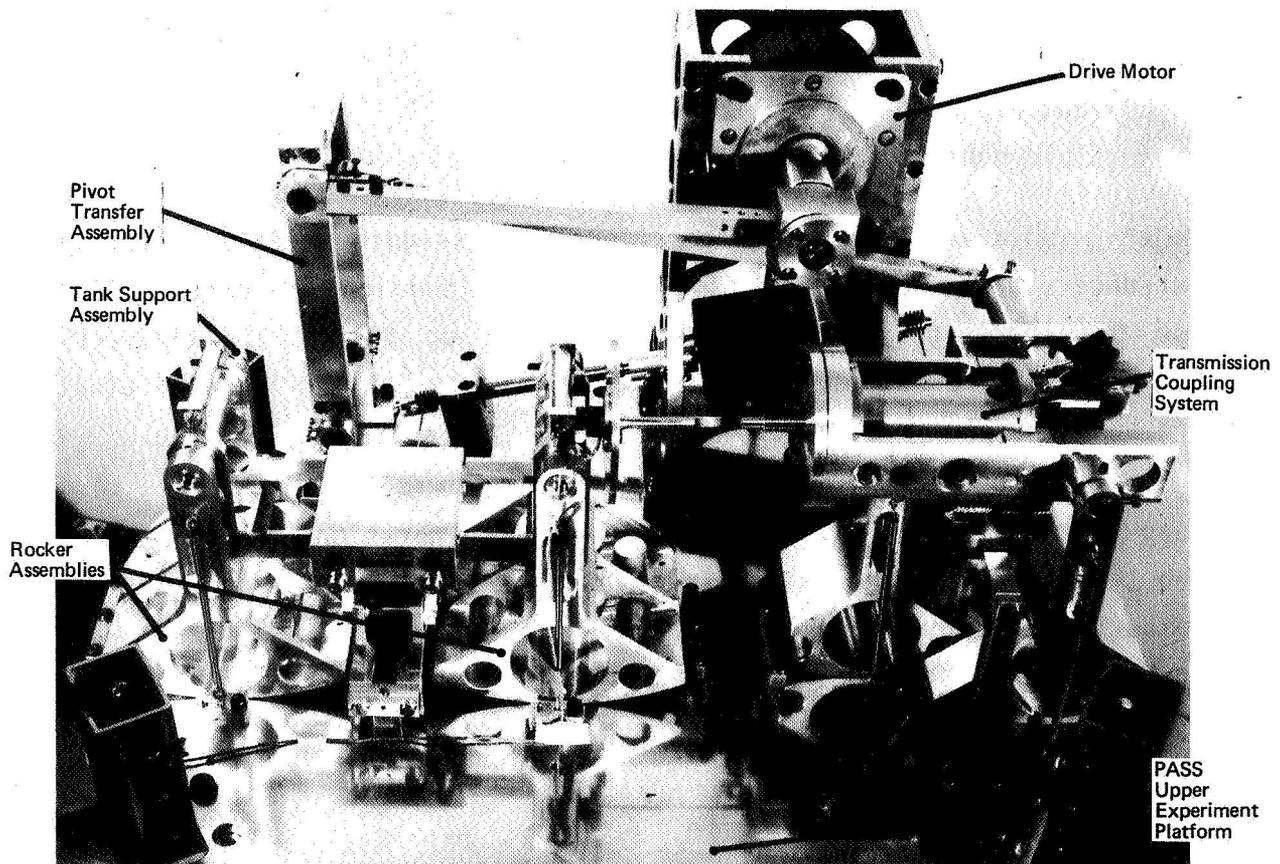


Figure 1. Experiment Basic Mechanical Assembly

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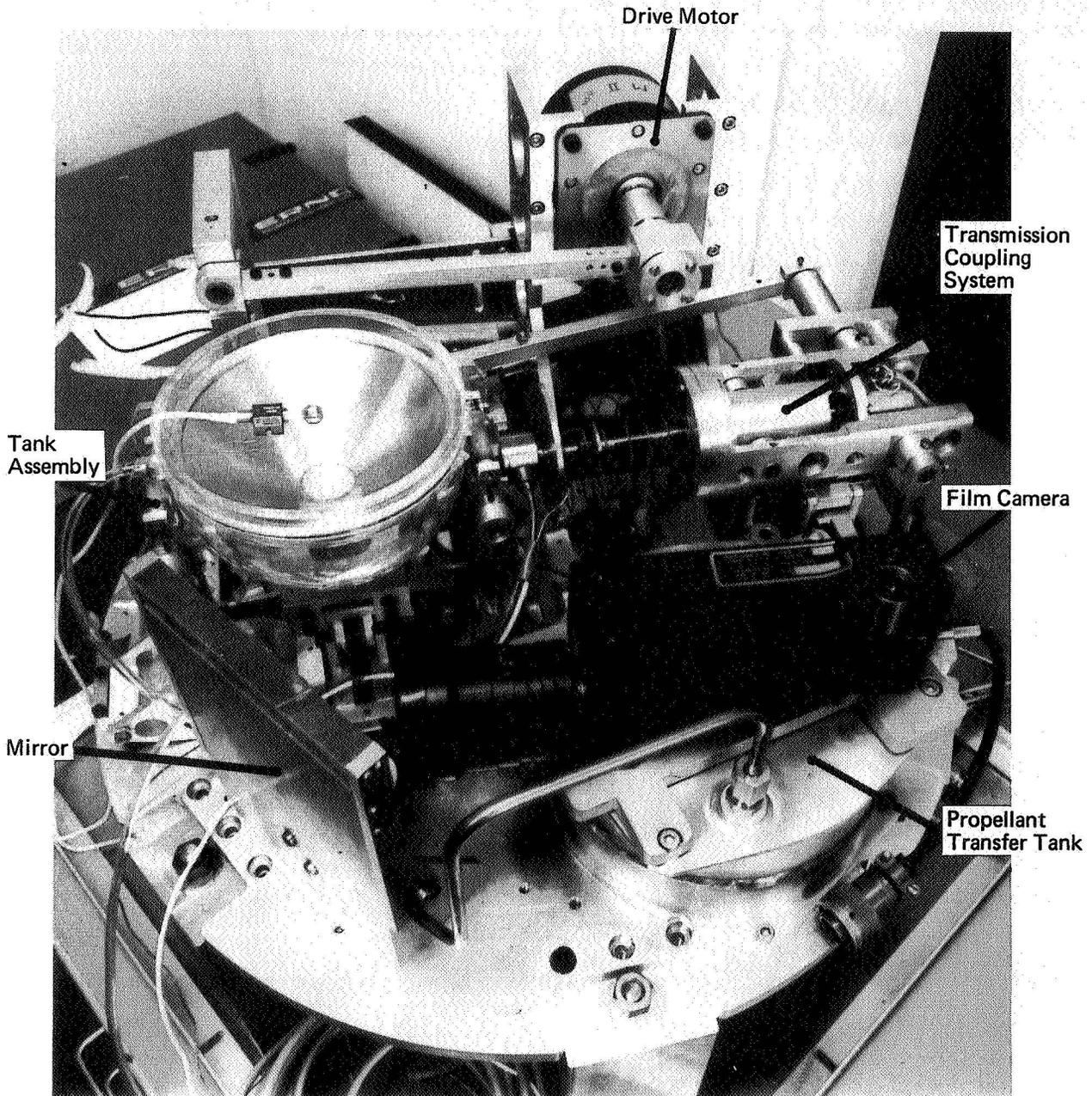


Figure 2. Complete Experiment Assembly

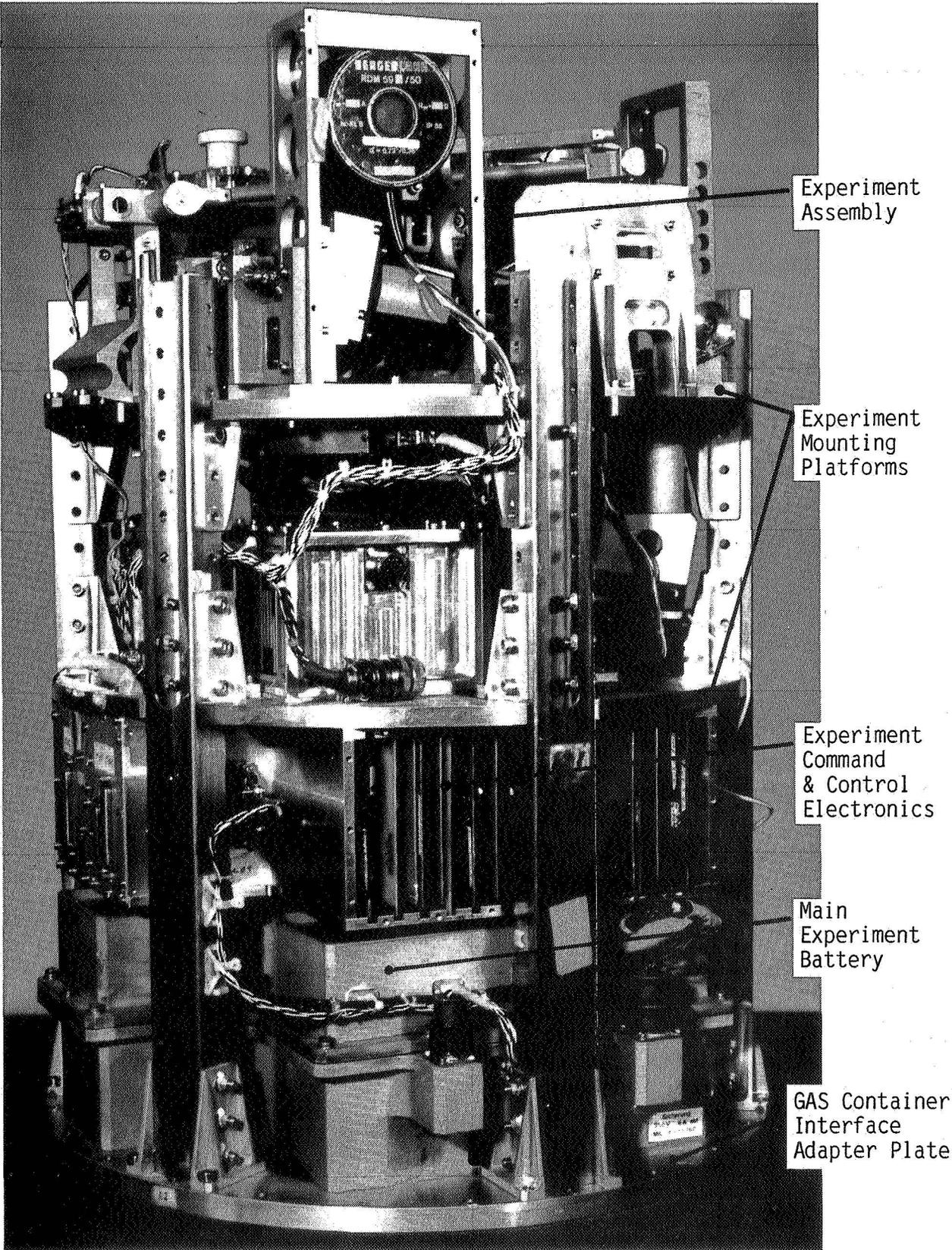


Figure 3. Complete Payload Assembly

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MEASUREMENT OF THE HEAT PRODUCED INTERNALLY DURING DISCHARGE
BY THE 6V EVEREADY ENERGIZER BATTERY, NO. 528

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ABSTRACT

When a 6V Eveready Energizer Battery, No.528, is discharged, heat is produced internally within the battery itself. Detailed measurements of this battery heat were made for a five day discharge period using a load resistance of 20 ohms and an ambient temperature of 10 Deg C. The battery heat produced per quarter day varied considerably, ranging from 1.08 watt-hrs to 2.63 watt-hrs. The total battery heat produced, using a 3.0 volt cutoff, is significant (37 watt-hrs) when compared to the total load heat produced (70 watt-hrs).

INTRODUCTION

In order to perform a thermal analysis of a payload, one needs to know all sources of heat. During the thermal analysis of our payload, the question was raised as to whether or not our batteries, the Eveready Energizer No. 528, should be considered a source of heat. To answer this question, we decided to measure the heat produced internally during discharge (Battery Heat) by a single No. 528, under load and temperature conditions similiar to those we expect during our actual flight. The basic approach that was chosen was to thermally insulate the battery, place it in a freezer and then heat it up to a fixed temperature. By comparing the heat that had to be supplied when the battery was not being discharged, to the heat that had to be supplied when the battery was being discharged, one could determine the Battery Heat. To make valid comparisons a fixed time interval was chosen.

APPARATUS

The apparatus used is shown in Fig. 1. There are two major systems, the discharge system and the thermal control system. The discharge system consists of the battery under test, a decade resistance box for the load, and one channel of a chart recorder to monitor the terminal voltage of the battery during the entire discharge. The decade resistance box was set to 20.0 ohms to simulate expected payload loads.

The remainder of the apparatus makes up the thermal control system designed to keep the battery under test at 10 Deg C. The desired temperature was achieved by using a freezer/heater combination. The freezer was set at its warmest setting, -15 Deg C, and its temperature was monitored through a thermocouple attached to one channel of a chart recorder. A heater and insulation were then placed around the battery and used to raise the battery temperature to +10 Deg C after being placed in the freezer. A second thermocouple attached to the other channel of the chart recorder monitored the battery temperature. The heater itself consisted of no. 30 insulated wire wrapped around the battery, and the insulation was of polyurethane foam. The heater was turned on and off automatically by a temperature control circuit which consisted of a thermistor, a controller and a power supply. Power for the heater came from a separate power supply. The heater voltage was monitored by a continuously connected voltmeter and a chart recorder. The heater current was monitored by a continuously connected ammeter. Actual heater on time was monitored by a clock turned on and off by a relay in the heater circuit.

PROCEDURE

Using the apparatus described above, data was taken in three steps:

1. Pre-Run calibration: Without the discharge circuit connected the thermal control circuit was activated and run for over a day. During this time the following variables were monitored and recorded: actual clock time, heater on time, heater voltage, heater current, battery temperature, freezer temperature and battery terminal voltage.
2. Discharge Run: Without changing anything the discharge circuit was connected. The same variables as listed above in step 1 were again monitored and recorded. This time readings were taken every quarter day (6 hours). The entire battery discharge was monitored for a total of 20 quarter days (5 full days).
3. Post-Run calibration: Same as pre-run calibration.

RESULTS AND DISCUSSION

Using the output from the chart recorder, which was several feet long, the battery terminal voltage was replotted using a compressed time scale (Fig. 2). As one typically finds with a 6V alkaline battery, the voltage quickly drops off after reaching around 3.0 volts.

During all three steps of the experiment, the following variables changed by less than 2%: heater voltage, heater current, battery temperature, and average freezer temperature. Thus, for all practical purposes they may be considered as constants. For all analysis a quarter day time interval was used.

Using the data obtained during the pre-run and post-run calibrations, we determined the average heat supplied per quarter day to keep the non-discharging battery at 10 Deg C. The pre-run calibration gave 4.58 watt-hrs and the post-run calibration gave 4.51 watt-hrs for an average of 4.54 watt-hrs. These values are indicated in Fig. 3 for later reference.

Using the data obtained during the discharge, we determined the heat supplied per quarter day to keep a discharging battery at 10 Deg C. The results are plotted in Fig. 3. It should be noted that the heat that had to be supplied varied considerably over the 20 quarter day period, reaching a low about half way through the discharge period when the battery terminal voltage was approximately 3.5 volts.

The difference in heat supplied per quarter day to the non-discharging and the discharging battery to keep it at 10 Deg C is the Battery Heat. More specifically, for any fixed time interval such as a quarter day, one can write:

$$\begin{array}{rcl} \text{BATTERY} & & \text{HEAT SUPPLIED BY} & & \text{HEAT SUPPLIED BY} \\ \text{HEAT} & = & \text{OUTSIDE HEATER} & - & \text{OUTSIDE HEATER} \\ & & \text{DURING NON-DISCHARGE} & & \text{DURING DISCHARGE} \\ & & \text{CALIBRATION RUNS} & & \end{array}$$

Using the results plotted in Fig. 3 and the above formula the Battery Heat was determined. These results are plotted in Fig. 4. Consistent with the earlier observation that the heat supplied by the outside heater each quarter day varied considerably and reached a low at around 11 quarter days, the Battery Heat also varies considerably, ranging from a low of 1.08 watt-hrs during the 1st quarter day to a high of 2.63 watt-hrs during the 11th quarter day. To see how significant the Battery Heat was, we chose to compare it with the heat produced by the load (Load Heat).

The Load Heat for each quarter day was calculated by going back to the original chart recorder output of the terminal voltage and determining the average voltage for each quarter day. Using this average voltage, the known fixed value for the load resistance, and the common relationship $P = V^2 / R$, the Load Heat for each quarter day was determined. These results are plotted in Fig. 5. As one would expect its shape is similiar to the plot of the terminal voltage seen earlier in Fig. 2.

The Battery Heat per quarter day and Load Heat per quarter day are compared in Figs. 6 and 7. In Fig. 6 we see that early in the discharge, for the first several quarter days, the Battery Heat per quarter day is quickly rising, while the Load Heat per quarter day is quickly dropping. Both variables then generally level off somewhat, finally intersecting after about 18 quarter days when the terminal voltage is between 2 and 3 volts. From Fig. 7 we can clearly see that for much of the discharge the Battery Heat per quarter day is over 50% of the Load Heat per quarter day.

Cumulative Battery Heat and cumulative Load Heat are compared in Figs. 8 and 9. In Fig. 8 we see that both totals grow almost linearly with time. By the time the terminal voltage has dropped to 3.0 volts during the 16th quarter day, the cumulative Battery Heat is about 37 watt-hrs and the cumulative Load Heat is about 70 watt-hrs (Fig. 8) for a ratio slightly over 50% (Fig. 9).

CONCLUSION

The heat produced internally during discharge by an Eveready Energizer No. 528, is significant when compared with the heat produced at the load by the same battery. Any thorough payload thermal analysis should incorporate this heat produced internally. For many payload designs it may even be a welcomed additional heat source used to keep the payload warm.

ACKNOWLEDGEMENTS

The authors are grateful to the other members of the A&T Student Space Shuttle's Electrical Support Team for their help with preparing the experiment, collecting data, and analyzing the data: A. Abul-Fadl (Assoc. Prof., Elec. Engr.) and D. Hood (Undergrad student, Elec. Engr.)

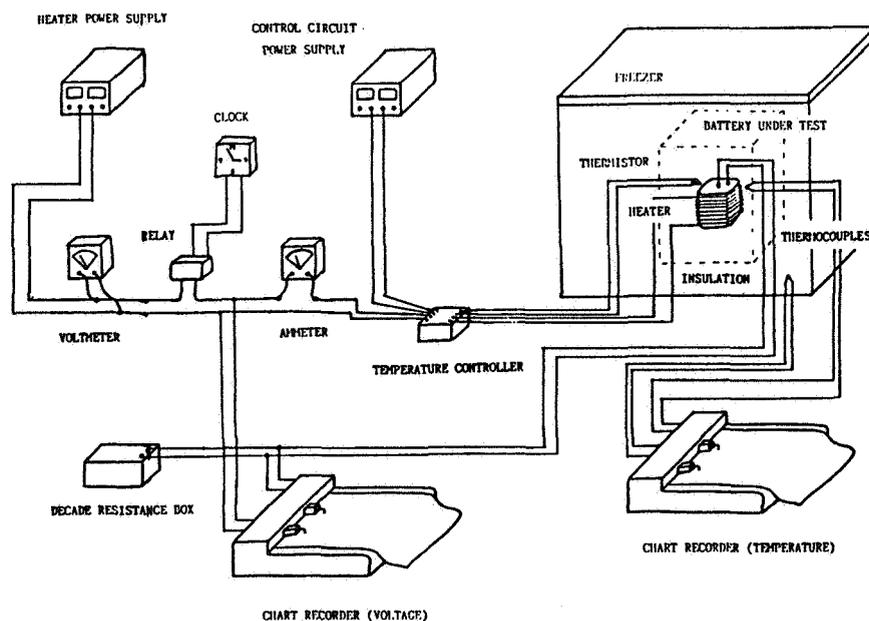


Figure 1. Battery Testing Apparatus

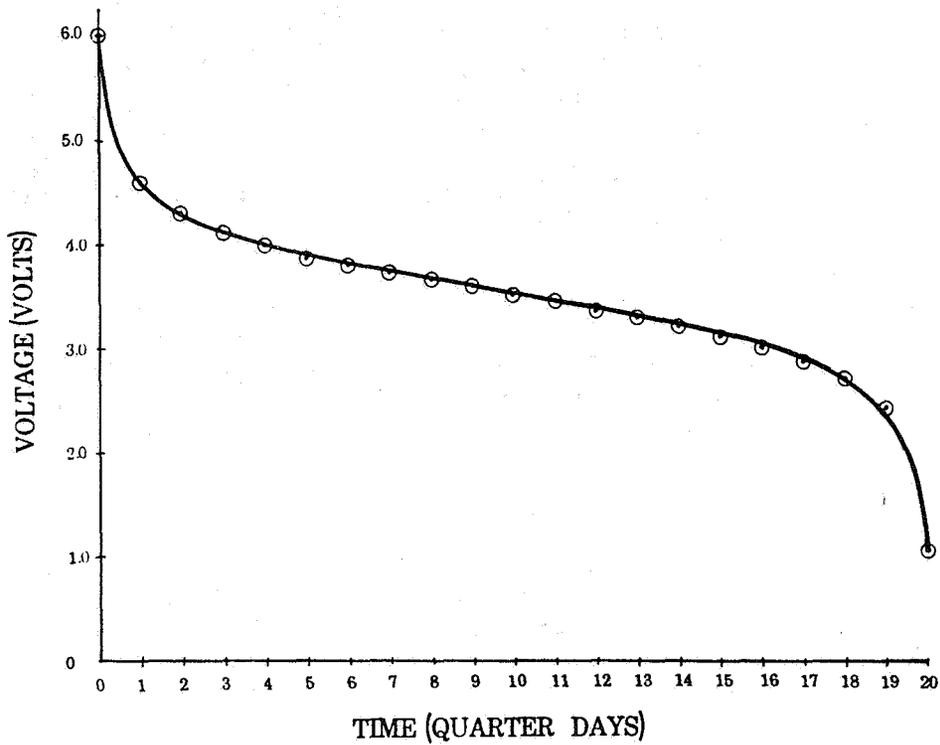


Figure 2. Battery Voltage vs. Elapsed Time

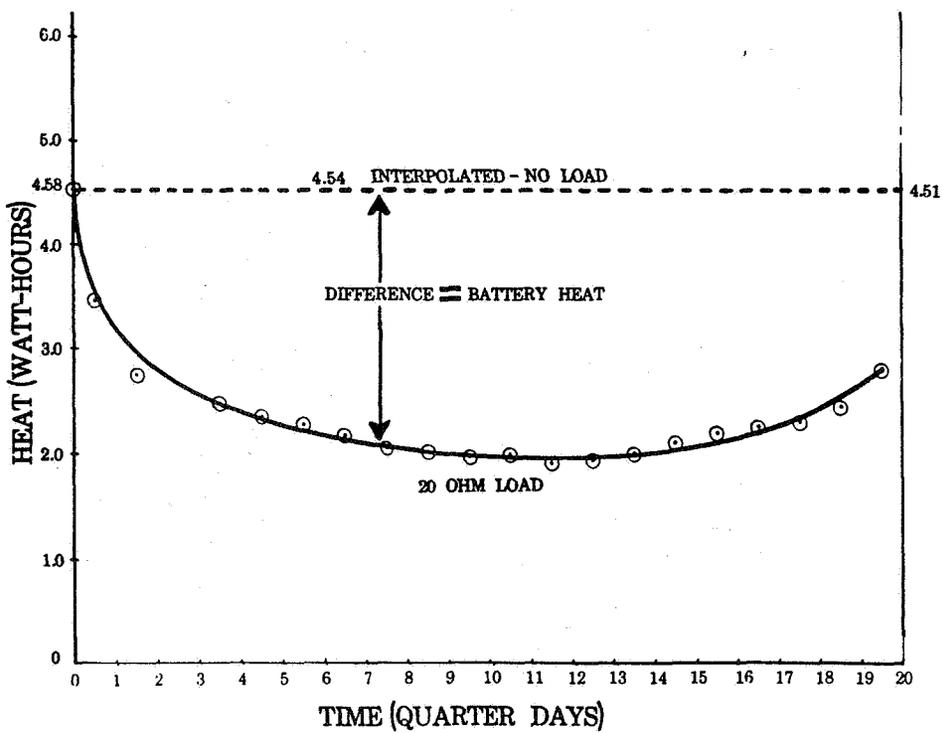


Figure 3. Heat Supplied to Battery per Quarter Day vs. Elapsed Time

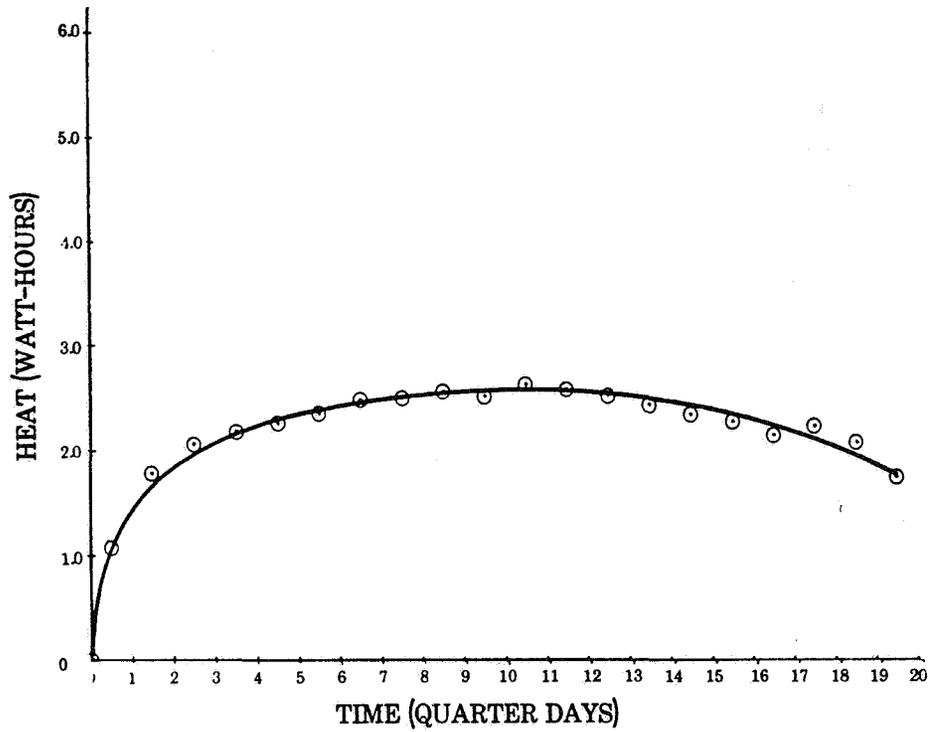


Figure 4. Battery Heat per Quarter Day vs. Elapsed Time

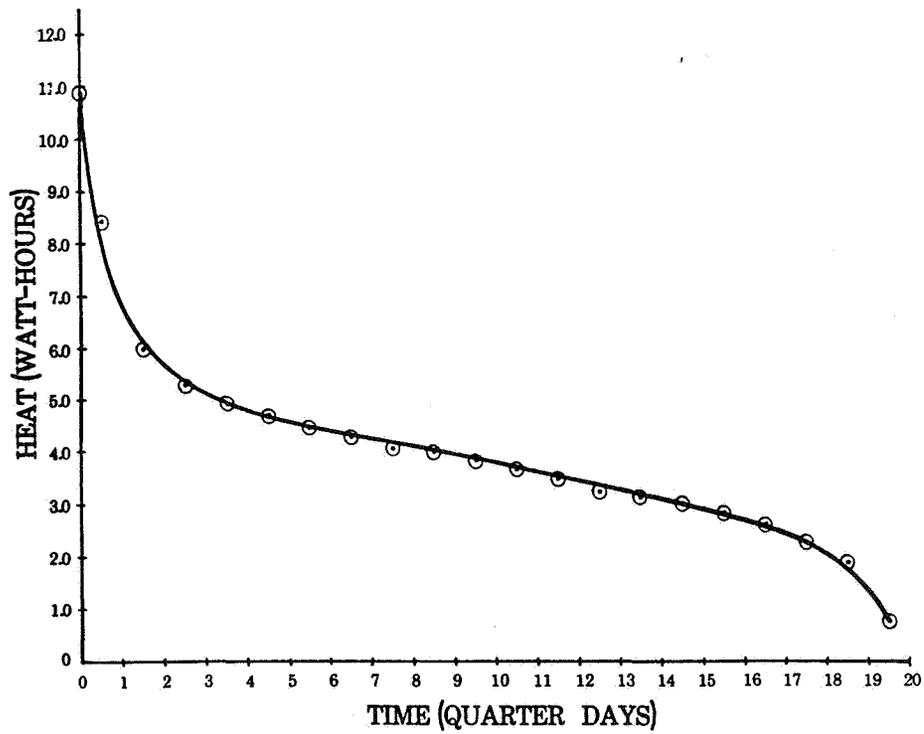


Figure 5. Load Heat per Quarter Day vs. Elapsed Time

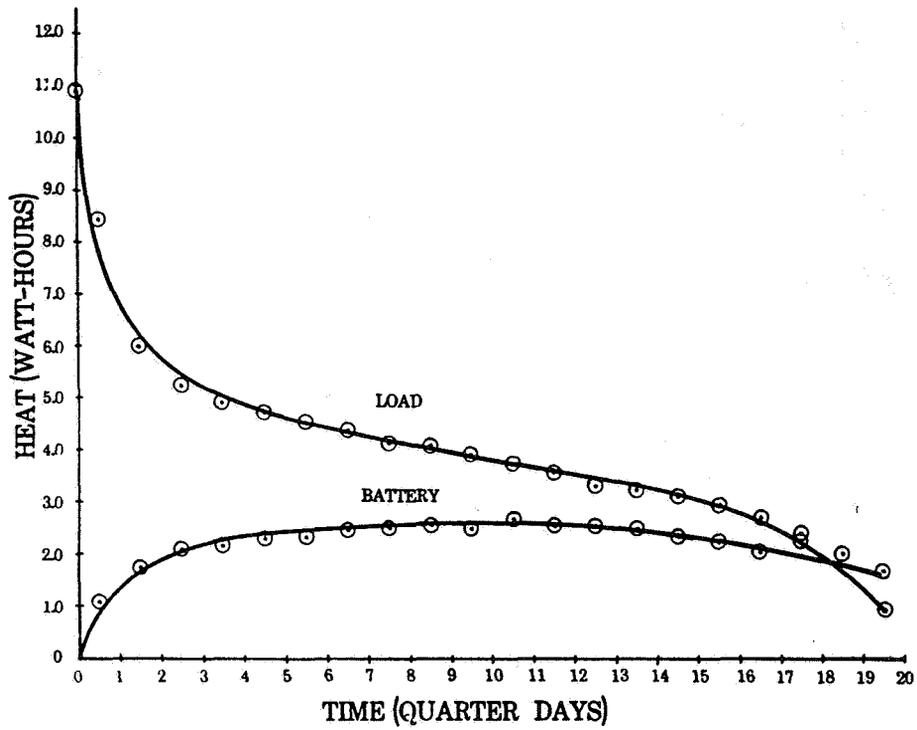


Figure 6. Battery and Load Heat per Quarter Day vs. Elapsed Time

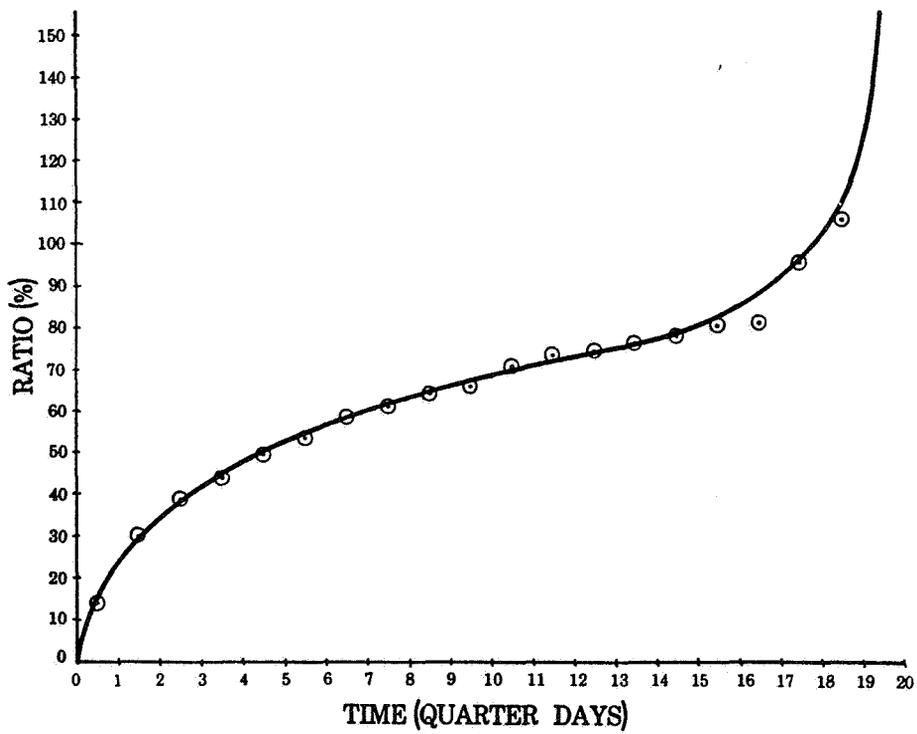


Figure 7. Heat Ratio (Battery to Load) vs. Elapsed Time

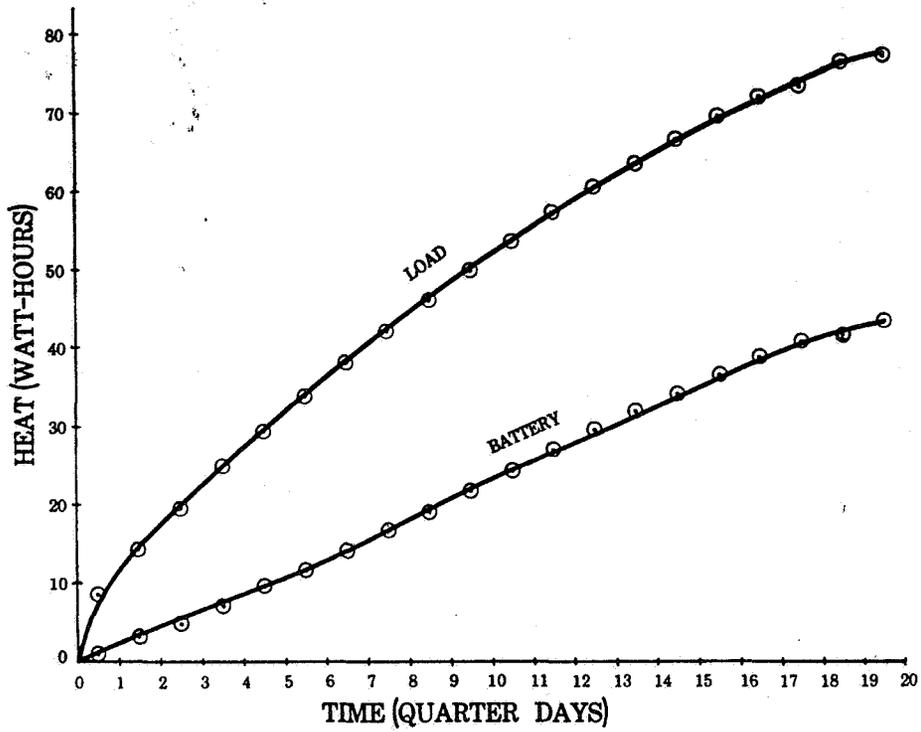


Figure 8. Cumulative Battery and Load Heat vs. Elapsed Time

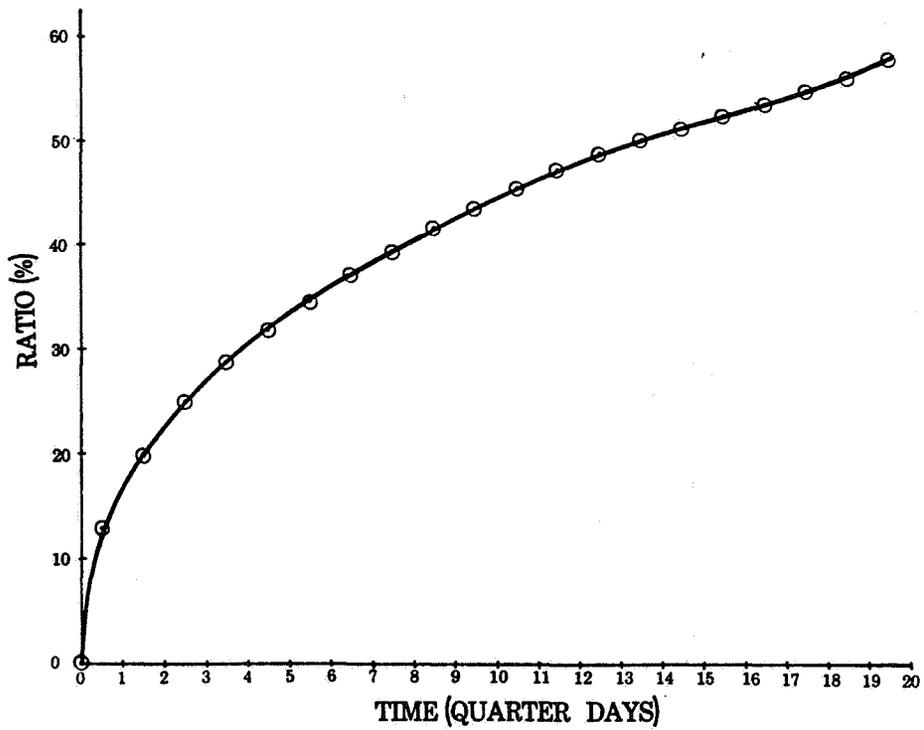


Figure 9. Cumulative Heat Ratio (Battery to Load) vs. Elapsed Time

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GAS EJECTION SYSTEM OVERVIEW AND DESIGN ENHANCEMENTS

Pat Miller
TSI

TV450050

Get Away Special Symposium 1985
October 8-9, 1985

GAS EJECTION SYSTEM

- Payload mounts on ejection system which is attached to the LEP (Lower End Plate).
- Provides a 20" opening – same as GAS canister.
- Ejection force by a spring after two bolts on a marmon clamp are cut by pyrotechnics.
- Lid serves as a barrier to contain loose satellite.

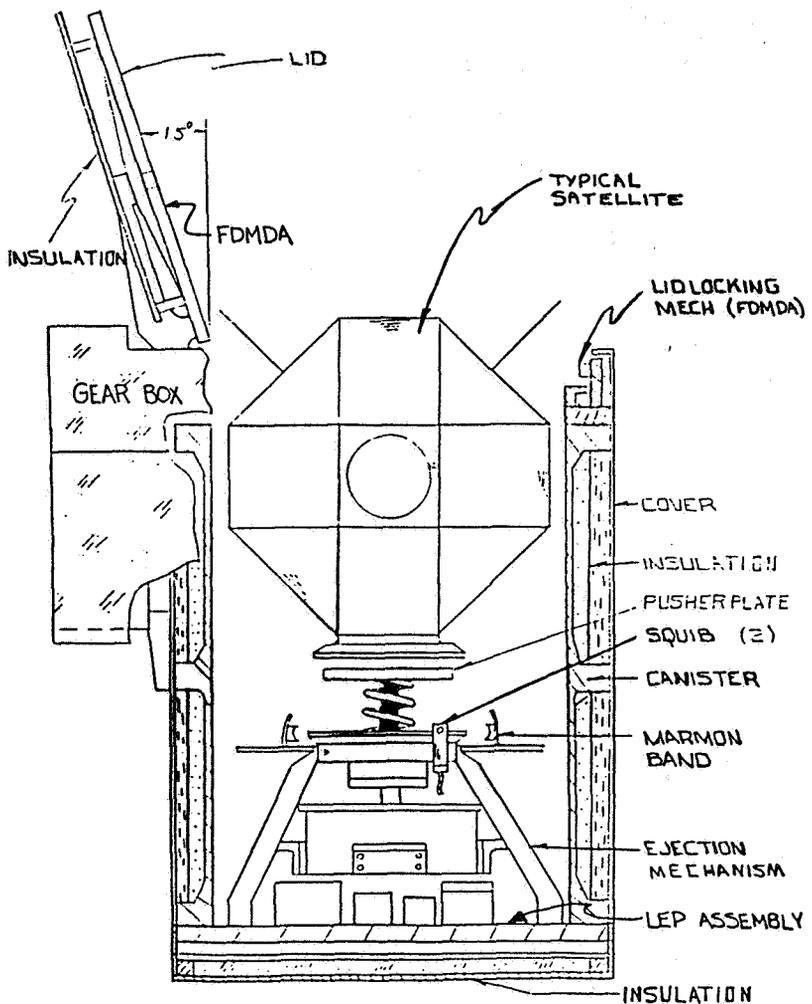


Figure 1. GAS Ejection System

OS: 75-087

SECOND GAS EJECTION SYSTEM

- First System - FDMDA Design and ready for shipping - 9 months.
- Second System - Ready for shipping in 8 weeks.
 - Started with GAS can.
 - Delivered in 9 weeks.

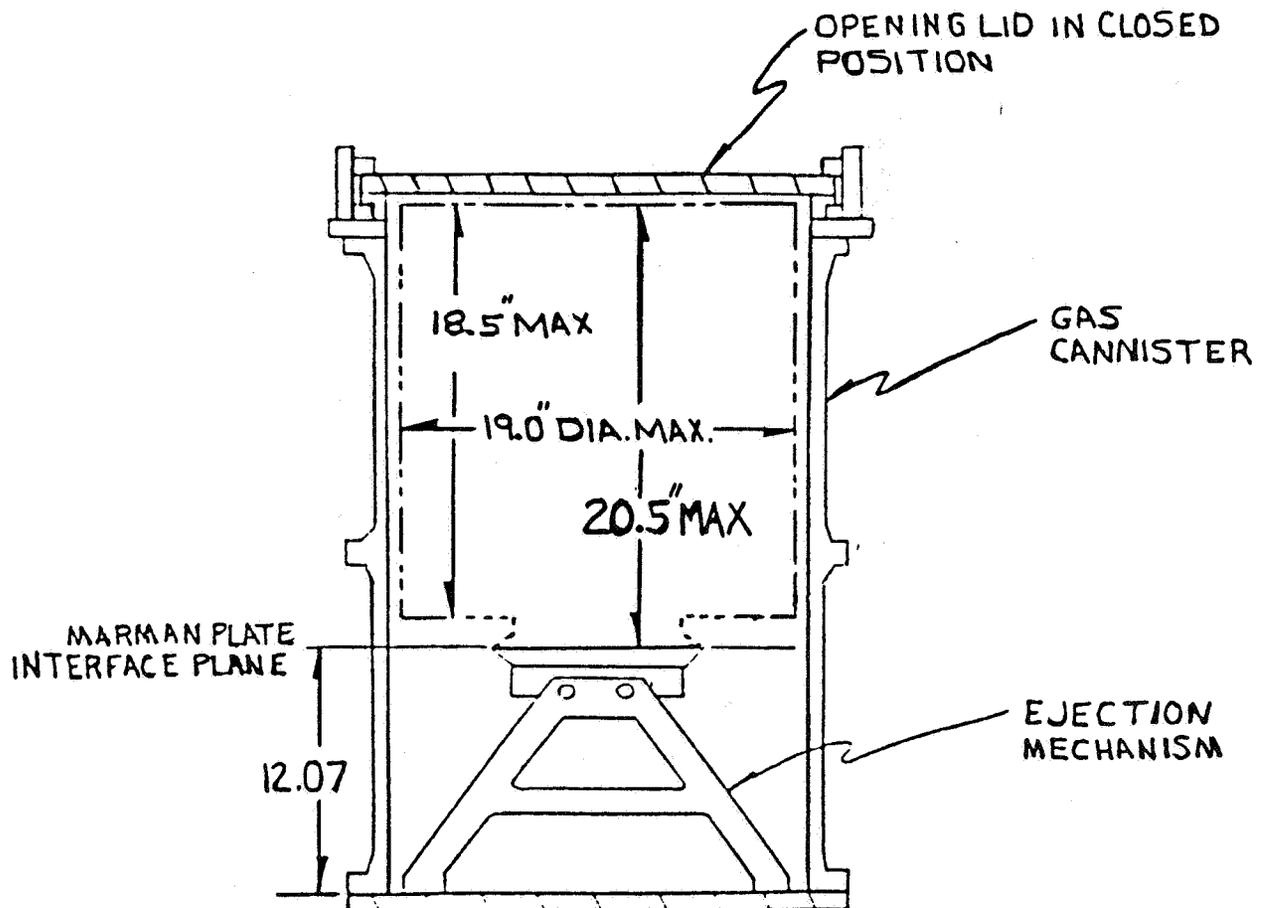
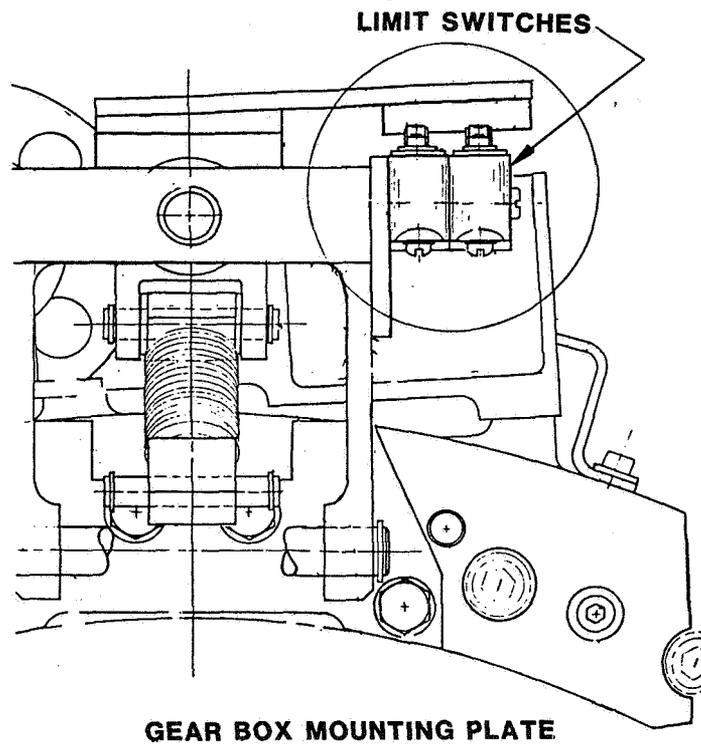
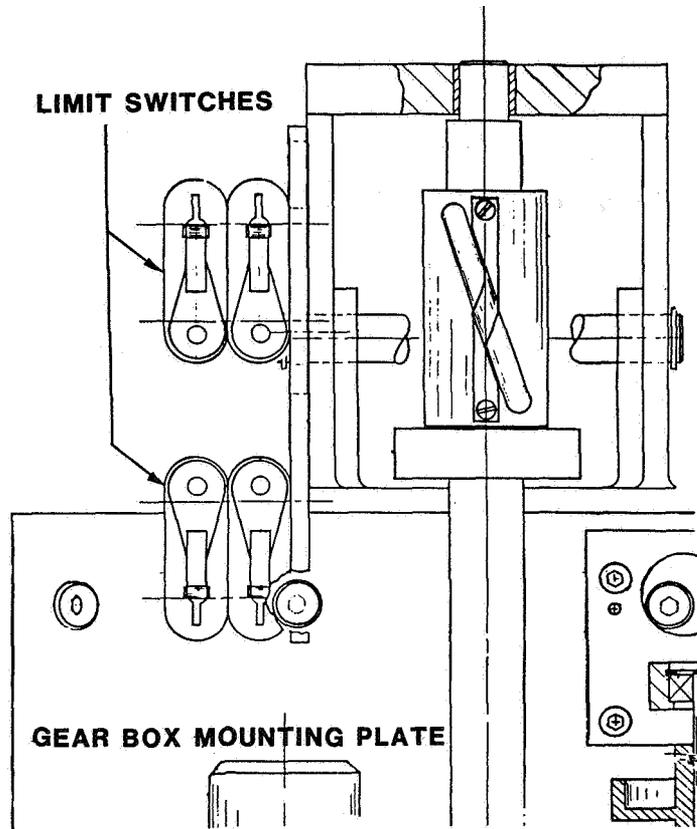


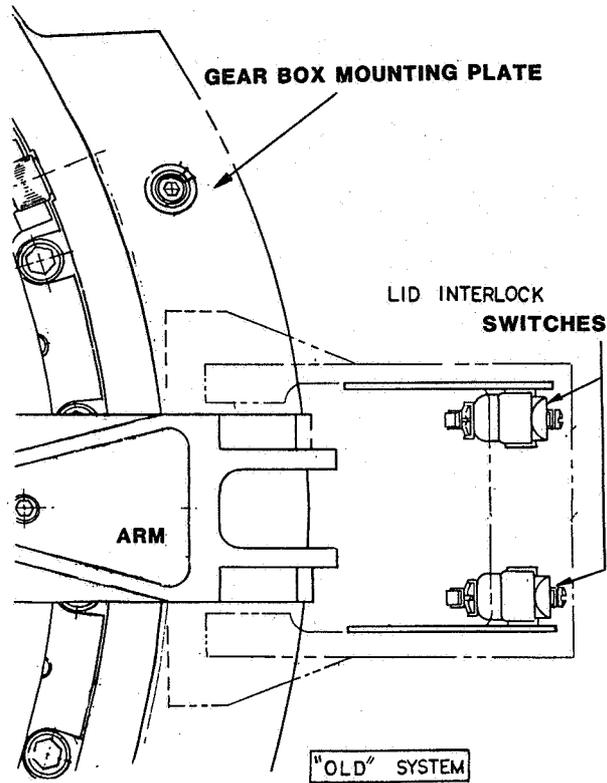
Figure 2. GAS Ejection System - Satellite Envelope

MISSION STS 51B (4/29/85)

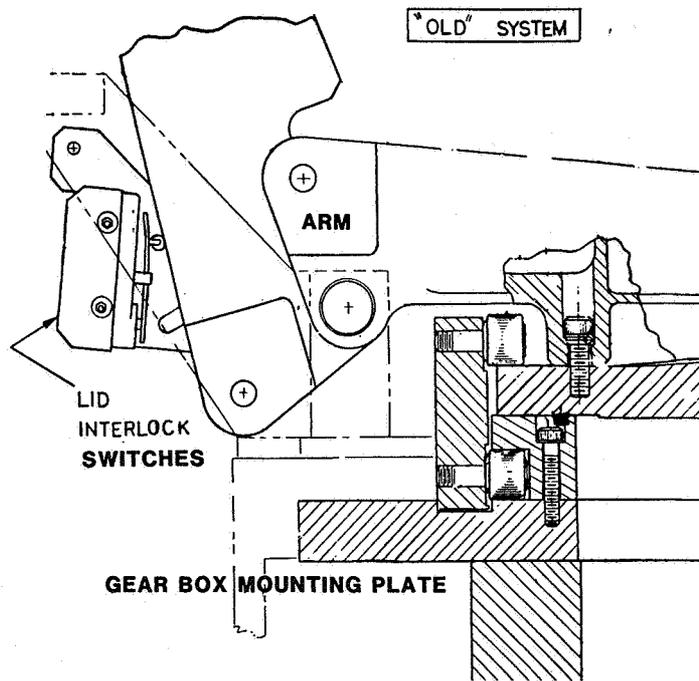
- GES #1 (NUSAT): Successfully ejected.
- GES #2 (GLOMR): Unsuccessful - returned intact.



Top View

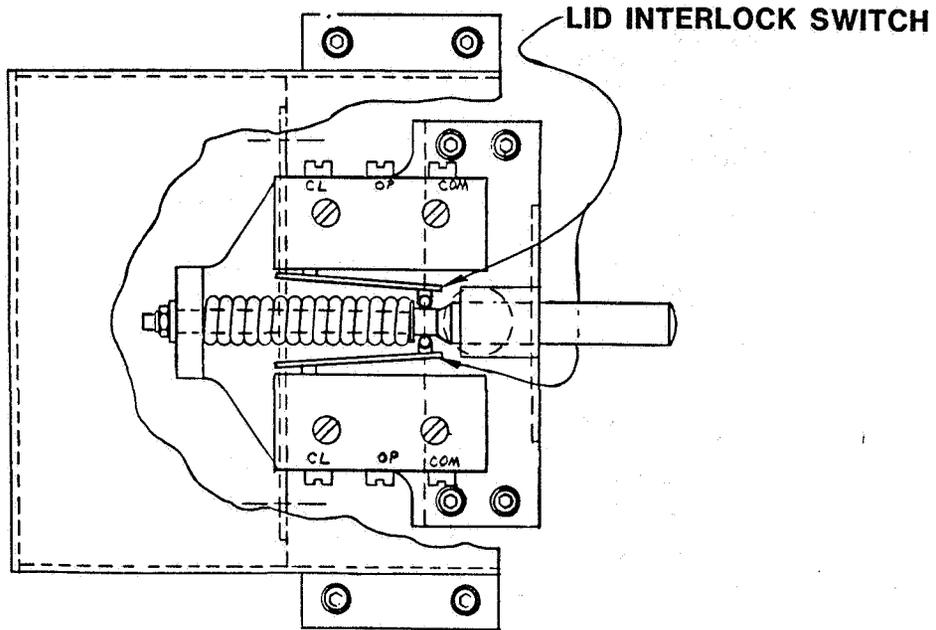


Top View



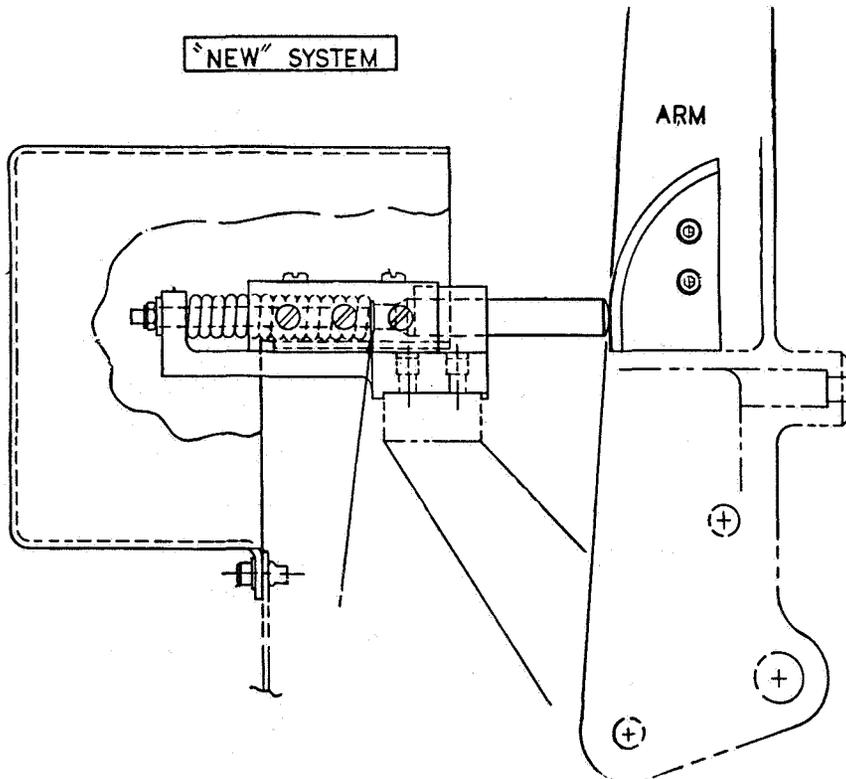
Side View

"NEW" SYSTEM



Top View

"NEW" SYSTEM



Side View

FUTURE OF GAS EJECTION SYSTEM

- Pressure tight can.
- Larger experiment volume.
- Possible attitude adjustment system.

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AN ADVANCED MATERIAL SCIENCE PAYLOAD FOR GAS

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Swedish Space Corporation

1985 GAS Experimenters Symposium

56191467

1 Materials Science in Sweden

Materials Science has a traditionally strong position in Sweden. Therefore it was natural that the Swedish Space Corporation when starting up Microgravity Activities concentrated on Materials Science. Experiments within this field have been performed on sounding rockets and also on NASA-aircraft F-104 in parabolic trajectory.

2 GAS experiment

When the interest turned towards long-duration zero-g flights and the possibilities opened up with the NASA GAS program, SSC rented two flight possibilities. The first of the two this payloads contains - in line with the scientific tradition - the first foundry in space. Four samples will be melted in three furnaces. The samples have a significant size and weight. As the material to be processed is lead-tin of different composition, the weight is quite high: 8 kg. Thus, nearly 10% of the payload weight is sample weight.

The scientific requirements have put heavy requirements on systems like power, heat storage and rejection, and mechanical support system. Some of the features is presented in this paper.

3 Scientific objectives

The aim of the experiments is to study solidification phenomena in metal alloys. Especially the dendritic growth and the effect of the absence of natural convection are of particular interest for the scientist, professor Hasse Fredriksson at the Royal Institute of Technology (RIT) in Stockholm. The results from the flight processed samples will be compared with results from a lot of earth-processed samples in order to investigate the influence of the natural convection on the solidification process.

4 Experimental techniques

a. Number of experiments and sample dimensions

Four samples of different size of a lead-tin alloy (composition 10-90 and 90-10 per cent) will be solidified unidirectionally. Two samples have the dimensions 100 x 100 x 25 mm, each placed in a graphite crucible also containing heaters and a cooler. A third crucible containing two samples with the dimensions 100 x 50 x 25 mm and 100 x 25 x 25 mm is used for simultaneous processing of the two smaller samples.

b. Experimental idea

The idea is to heat and melt the sample(s) and after that create a temperature gradient across one of the long sides while keeping the short ends at isothermal conditions (see figure 1).

The temperature is then decreased while maintaining the temperature gradient across the sample(s), until the whole alloy is solidified. This arrangement gives a unidirectional solidification assuming that the insulation on other surfaces is sufficiently good.

The temperatures are monitored with nine Chromel-Alumel thermocouples positioned in the sample according to the sketch in figure 2. In this way some information on the isothermality across the surfaces A_h and A_c is obtained as well as the temperature gradient from A_h to A_c .

In order to keep the isothermal conditions over the surfaces A_h and A_c during the cooling phase (the specifically interesting experimental phase) the contraction of the alloy during cooling and solidification must be taken into account. To compensate for contraction of the sample a pressurized alloy melt forces a piston to decrease the sample volume (see figure 3).

5 Payload description

5.1 General

The payload consists of the following subsystems.

- Experimental furnaces (3)
Crucible containing the sample(s) with three heating elements and insulation
- Cooling system:
 - pumps
 - plumbing
 - a cooler for each crucible, attached to the cold side of the sample
 - a heat sink with phase change material
- Structure
- Electronic system for control of the experiments and monitoring of experiment and housekeeping signals.

- Data handling and storage system
- Energy system
 - Batteries with control electronics
 - Power electronic unit
- NASA Interface electronics

Figure 4 shows the disposition of the subsystems of the payload.

5.2 Experimental furnaces

Four samples will be processed:

- two large samples of dimensions 100 x 100 x 25 mm each contained in a graphite crucible.
- two smaller samples of dimensions 100 x 50 x 25 mm and 100 x 25 x 25 mm, both contained in one graphite crucible of the same size as used for each of the two large samples (see figure 5).

The graphite crucible (outer dimensions 160 x 120 x 47 mm) holds three heating elements and a cooler (see figure 6).

The heaters are insulated coaxial resistive elements with a sheath of stainless steel (Thermocoax, Philips). They are positioned according to the diagram in figure 7.

The following experiment sequences delineate the function of the furnace:

- Heating and melting phase

All three heaters are switched on and the cooler is passive (no gas flow). The sample is melted and the "hot" surface A_h brought up to 340°C (T_h).

- Establishment of the temperature gradient

When the temperature of the hot surface T_h is equal to 340°C the heaters H2 and H3 are switched off while the heater H1 tries to establish the desired temperature gradient $T_h - T_c$ of around 20°C , and activates the gas cooling.

- Cooling phase

When the desired value of the temperature gradient $T_h - T_c$ is reached, the hot surface temperature T_h is decreased according to a preprogrammed profile giving a decrease of around $1^{\circ}\text{C}/\text{minute}$.

The temperature gradient signal, ($T_h - T_c$) will control the gas flow through the cooler in order to maintain a constant gradient.

- Post-experiment phase

When the hot surface temperature is below 180°C , the sample is totally solidified and the heater H1 is switched off. Also the nitrogen gas flow through the cooler is shut off.

The contraction of the sample during the cooling phase has to be taken into account. A melted alloy is pressurized by the bellow and forces the piston to decrease the sample volume, thus compensating for the contraction of the

sample. In this way a good thermal contact between the graphite surface A_h and the contracting sample is preserved.

We have chosen an alloy of lead-tin-bismuth. It is contained in a small cylindrical compartment in the graphite crucible and next to the heater H1.

The graphite crucibles are surrounded by insulation blocks 25 mm thick. The insulation material is Fiberfrax Duroboard 1200 (72% Al_2O_3 , 25% SiO_2 , manufacturer Carborundum, USA).

The three experiments are placed besides each other on a honeycomb aluminium deck (see also figure 8).

5.3 Cooling system

A principle for the cooling system is shown in figure 9.

Three pumps P1-P3 - one for each experiment furnace - operate by pressing nitrogen gas through the plumbing system to the experiment coolers.

There are three check valves (V1, V2, V3) - one for each pump - controlling the gas flow direction in the plumbing system.

The heated gas is transported from the experiment cooler to an energy buffer store. This is a cylindrical box of diameter 456 mm and height 50 mm with copper pipe plumbing (see figure 10) and the free volume filled with a phase change material. In this case we have chosen paraffin with melting point 56-58°C because of its high latent heat and rather low weight (around 6 kg is needed). The energy stored in paraffin buffer is radiated to space via the GAS experiment mounting plate.

The experiments are run in sequence. When experiment no 1 is on pump P1 forces nitrogen gas via check valve V1 to the cooler C1. The gas flows further on through cooler C2 and C3 and then through the energy buffer store, where the gas is cooled down to around 50°C before flowing back into the free volume of the cannister.

The coolers in the experiment furnaces are built of copper plates with grooves for the gas flow according to figure 11. The plates are stacked onto each other. With this design of the cooler the optimum performance regarding isothermality and high gas flow capacity is achieved.

5.4 Structure

The structure is built with three aluminium struts (fig. 4 and 12) positioned between the energy buffer store (EBS) and the battery package. The GAS experiment mounting plate is screwed on to the EBS container in the 24 holes spaced on 19" diameter.

The struts are made of hollow aluminium type 6063 profiles (75 x 25 mm, wall thickness 2 mm). The experiment furnaces together with the cooling system and some of the electronic systems will be mounted between the battery box and the EBS. The furnaces package with the insulation are mounted between two honeycomb aluminium (Metawell 05-02-05, Al6063) decks. The temperature monitoring electronics, the pumps and a power box are situated on opposite side of one support deck.

On the cover of the battery box the data handling and storage system boxes as well as the control electronic box are placed. These boxes will be available from outside without any necessity to disassemble other parts in the container.

5.5 Electronic System

To control the experiments and to monitor experiment and house-keeping signals the following electronic blocks are built (see block diagram in figure 13):

- Experiment temperature sensing systems: TE1, TE2, TE3
- Control Electronic system: CE
- Micro-g sensor electronic system; μ -g
- Temperature sensing system for the Energy Buffer Store (EBS): TEB

During the cooling phase the control system (CE) regulates the power to the hot side heater and the gas flow, i.e. the number of revolutions of the gas pumps. The signal from the thermocouple on the hot side (T_h) in the sample controls the hot side heater. The signal from the thermocouple on the cold side (T_c) is compared with T_h . The difference $T_h - T_c$ is the temperature gradient controlling the speed of the pump, and thus the gas flow, towards the desired gradient of 20°C.

The microgravity sensing system has three accelerometers placed in three orthogonal directions. The accelerometers used are purchased from Sundstrand and have the designation servo accelerometer type QA 2000-001 with a triaxial adapter type MB2000 Q-flex. The accelerometer signals are amplified and sent to the data handling unit. Also ambient temperature signals in the accelerometer unit are sensed and registered in the data handling unit.

The temperature in the energy buffer store (EBS) is sensed by two thermal sensors. The signals from these sensors are fed to the electronic control system, where it may delay the switching on of the experiments. The paraffin in the EBS has to be solidified before an experiment is started up, meaning that the temperature must be below 45°C.

5.6 Data handling and storage system

The experiment and house-keeping data will be fed to the four data acquisition units, one for each experiment and one for house-keeping.

The timers

The data acquisition system is controlled by four timers. Which are started on the power-on command.

Data acquisition system

The data acquisition system (DAS) consists of four independent subsystems. Each subsystem has 32 single ended analog inputs. Each system delivers 14 bits data to the memory unit.

In each subsystem the channel sampling pattern is determined by a programmable memory. The memory contains four different formats, which are changed during the experiment process to increase the sampling rate on channels of special interest.

Data storage unit

The data storage unit (DSU) is a solid state device using static N-channel MOS erasable and electrically programmable read only memories (EPROM's).

In read and erase mode the DSU is removed and inserted into a Data Readout unit. After reading an automatic erase- and check sequence can be activated.

The timers, DAS:s and DSU:s for experiment 1 and 2 are integrated into one electronic box placed on the battery deck; the corresponding units for experiment 3/4 and the housekeeping systems are built into the second of the three electronic boxes (figure 4).

5.7 Energy system

The Swedish Space Corporation (SSC) proposes to use a special sort of Lithium batteries for this payload. The type of Lithium batteries are superior to other battery systems with respect to energy density (as a function of volume or weight), ease of handling, low self discharge, and high safety standards.

Lithium for the anode, polycarbonmonofluoride for the cathode and an organic electrolyte form the battery cell. Polycarbonmonofluoride is a chemically and thermally stable solid material. The batteries are highly safe compared with other lithium batteries which use corrosive and toxic active liquid material (SOCl_2) or active gaseous material (SO_2).

Due to the high power requirements (around 3300 Wh) of the payload a great effort was put into an investigation of available batteries. The National/Panasonic batteries $\text{Li}(\text{CF})_n$, model BR-C, were selected because they are inherently safe and meet the experiment power requirements. This battery is not to be confused with Lithium/Sulphuroxychloride batteries which are referred to as hazardous in NASA Reference Publication 1099 (dated November 1982).

The battery consists of 378 primary lithium-polycarbonmonofluoride cells connected in 14 parallel branches with 27 cells in each branch. Mechanically the cells are contained in 7 cylindrical boxes mounted on a common deck with full payload diameter 501.6 mm (or 19.75"). The protection and current limiting electronics is mounted in this deck. The upper side of the deck serves also as a mounting surface for some experiment subsystems (see figure 14).

Inside the battery a number of protection devices are used in such a way that no possible failure mode or combinations of different failures could cause a safety hazard.

In the table below we have made comparison between different battery systems. For each system we have given the energy density (capacity per weight unit) and the required weight of the cells (N.B. the cover and venting system, when required, are not included) to fulfil the energy demands from the payload. The data comes from information from the different battery manufacturers. The data is consistent with the battery data given in the GAS ("red book") handbook.

Battery type	Energy density Wh/kg	Necessary cell weight (kg)
Lead-acid	25	128
Ni-Cd (high capacity)	35	91
Ag-Zn	60	53
Alkaline (primary)	50	64
Lithium (Li/CF) _n		
low discharge rate	260	12
high discharge rate	180	18

our case →

It is clear that none of the cell types, except the lithium Li/CF_n batteries, are a realistic alternative in our case.

The reasons to choose the BR-C lithium-polycarbonmono-fluoride cells are summarized here:

- low weight and volume per capacity unit (best when compared to other battery systems),
- simplified handling compared to Ag/Zn cells,
- lower safety risks than Ag/Zn cells,
- no storage problems
The storage time for the selected lithium cells is 10 years; to be compared to 3 months for electrolyte-filled Ag/Zn cells.

The drawbacks of the selected lithium cells are

- no recharge capability (but instead 10 years life cycle time which means a very low self discharge),
- rather low current capability (this drawback is compensated for by connecting in parallel, in our case, 14 cell packages).

5.8 Weight budget

The following table gives the weight budget for the payload:

Items	Weight kg	Total
<u>Energy system</u>		
(Battery with box and cover)	36.0	36.0
<u>Energy Buffer Store</u>		
Paraffin	5.0	
Box and cover	2.0	9.0
Cooler	2.0	

<u>Struts</u> (3 pcs)	3 x 0.5	1.5
<u>Experiment system</u>		
Sample	8.0	
Graphite crucibles	4.5	
Insulation	3.0	20.0
Experiment cooler	2.5	
Mounting plate	1.0	
Mounting details	1.0	
<u>Gas cooling system</u>	3.0	3.0
<u>Zero-g sensors</u> with electronics	1.0	1.0
<u>Temperature sensing electronics</u>	1.0	1.0
<u>Data handling system</u>		
DSU	2.0	
DAS	2.0	7.0
Power unit	3.0	
<u>Control electronics</u>	3.0	3.0
<u>Miscellaneous</u>		
(Nuts, bolts, cabling, piping)	2.0	<u>2.0</u>
	Total	83.5
	Margin	<u>7.3</u>
	Allowed weight	90.8

6

A complicated payload

This first mini-foundry in space is a complicated payload with high power and heat rejection requirements. However the developed concept seems to be in accordance with the experiment requirements. SSC is now waiting for acceptance of the final safety data package. The payload is on the tentative primary flight list for STS61-C, currently scheduled for December 20, 1985.

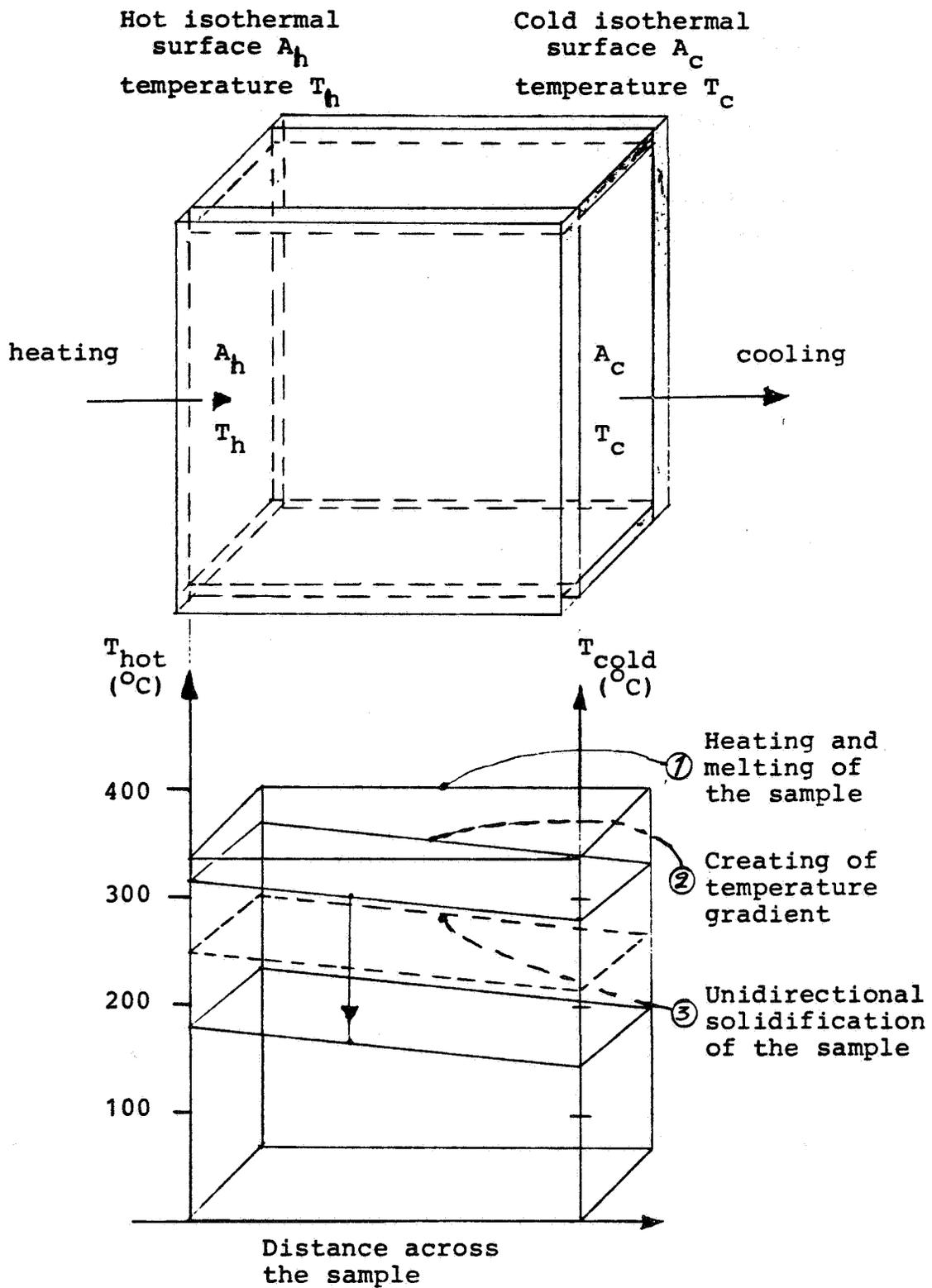
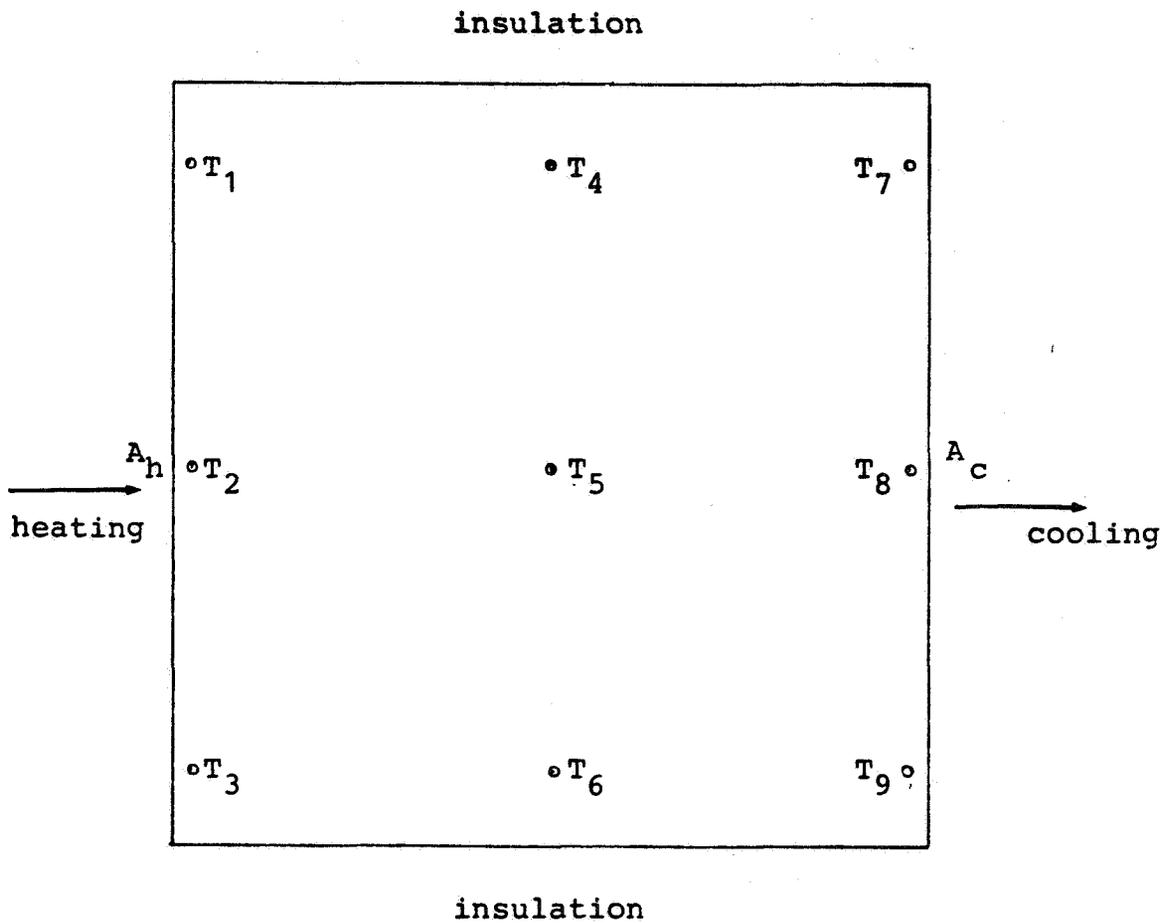


Figure 1. Principal experimental idea: solidification studies.



T_1 - T_2 - T_3 monitor the temperature distribution at the hot side A_h

T_7 - T_8 - T_9 monitor the temperature distribution at the cold side A_c

T_2 - T_8 gives the temperature gradient

Figure 2. Distribution within the sample of temperature monitoring Chromel-Alumel thermocouples.

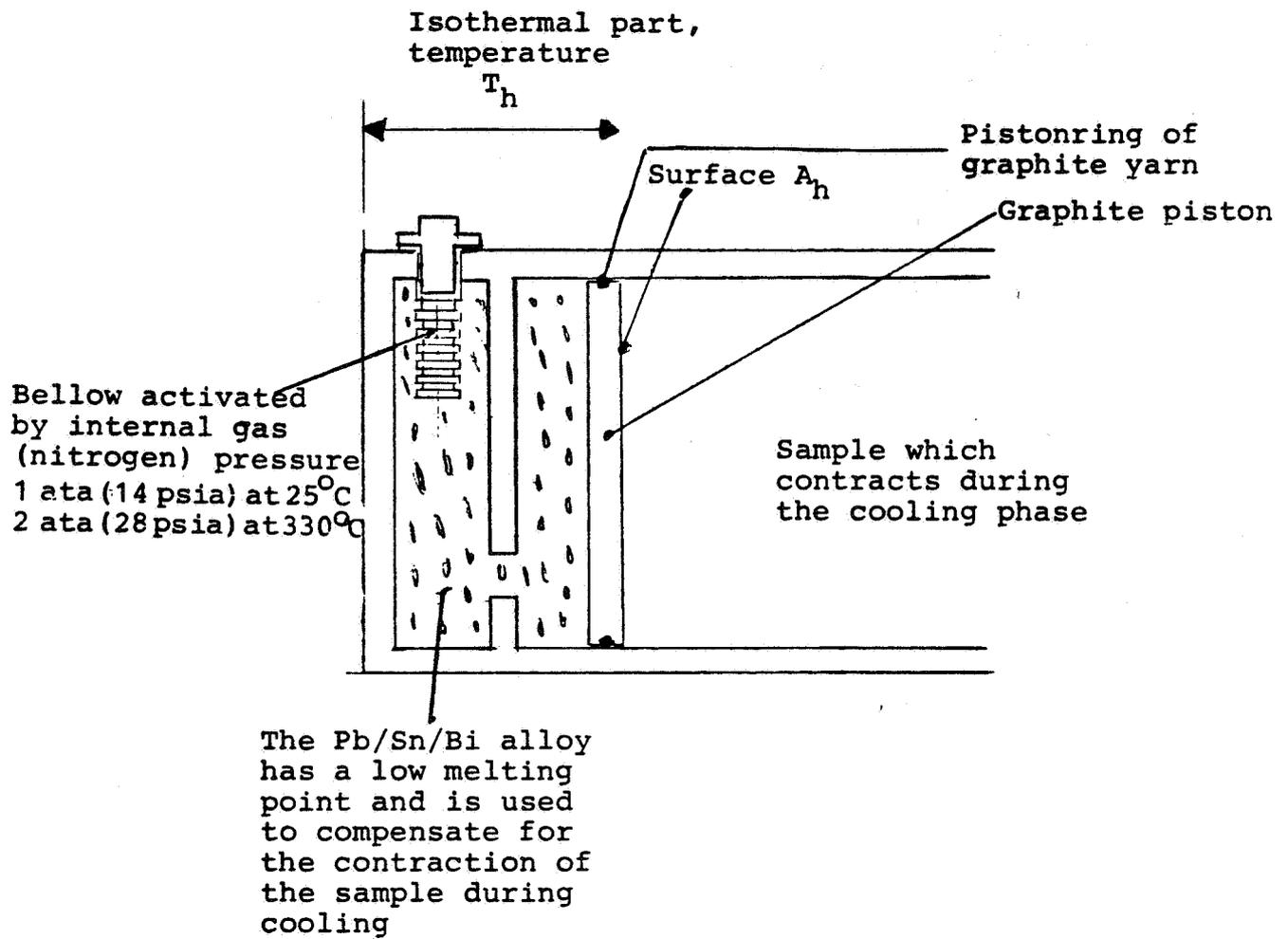


Figure 3. Mechanism, compensating for contraction of the sample - principal view.

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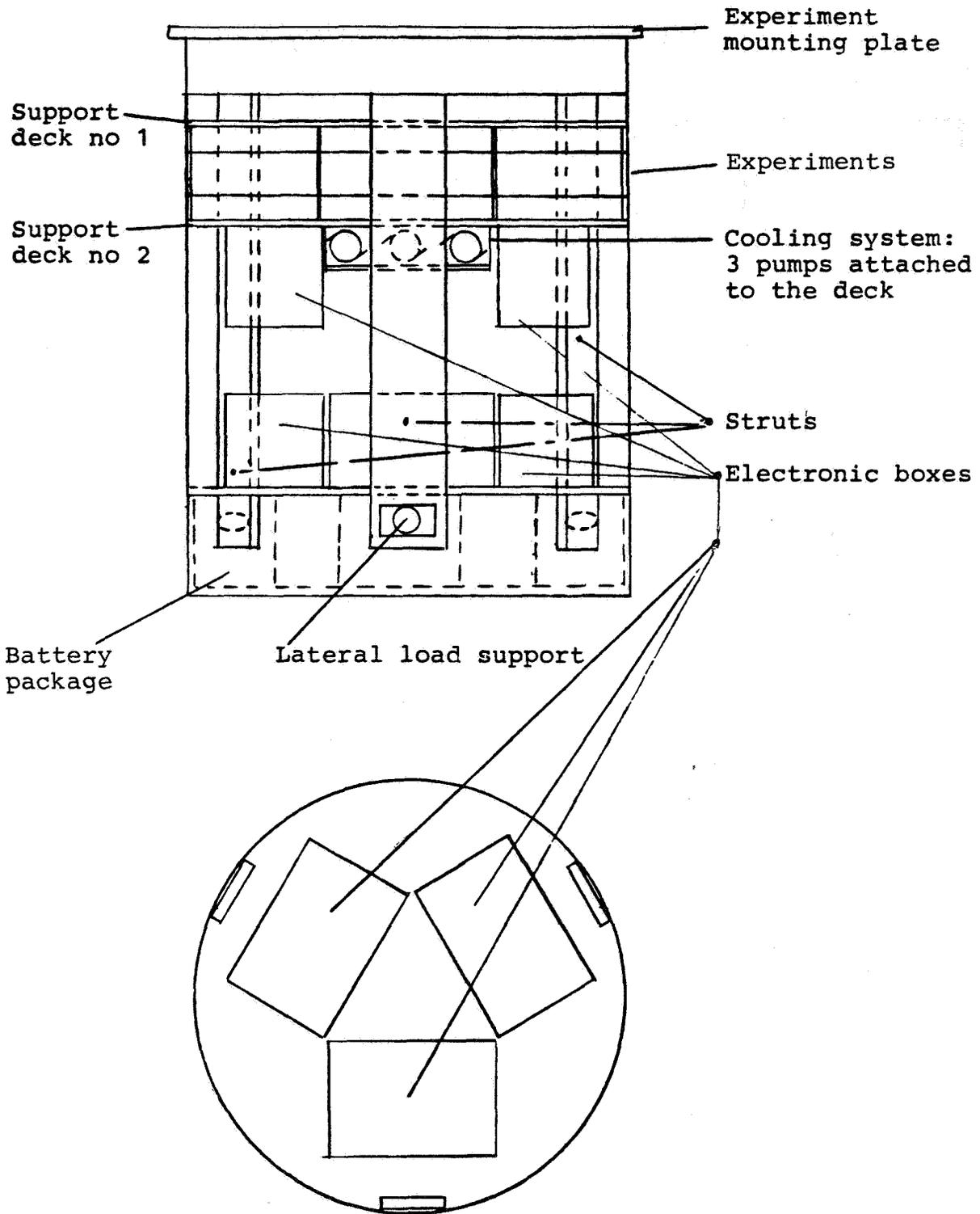
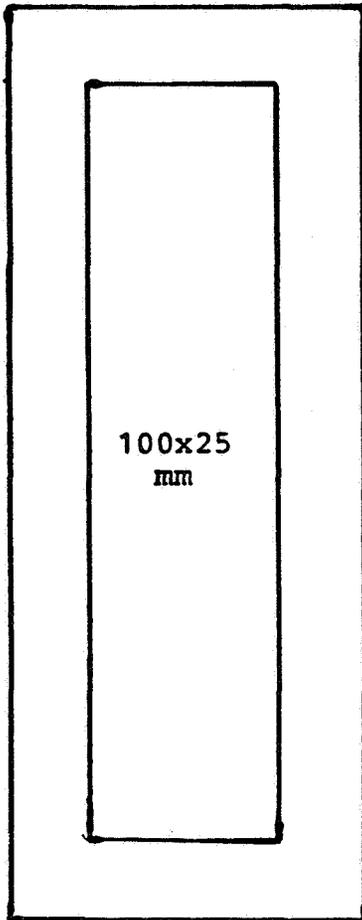
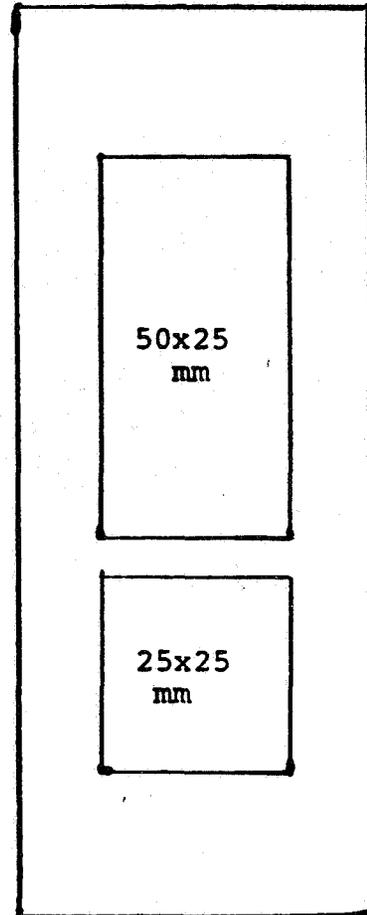


Figure 4. The configuration of the subsystems of the payload.

Sample length 100 mm



Crucible used
for processing of
a 100 x 100 x 25 mm
sample



Crucible used for
simultaneous processing
of two samples of the
dimensions

100 x 50 x 25 mm

100 x 25 x 25 mm

Figure 5. Cross section view of the two types of graphite crucibles used for the experiment samples.

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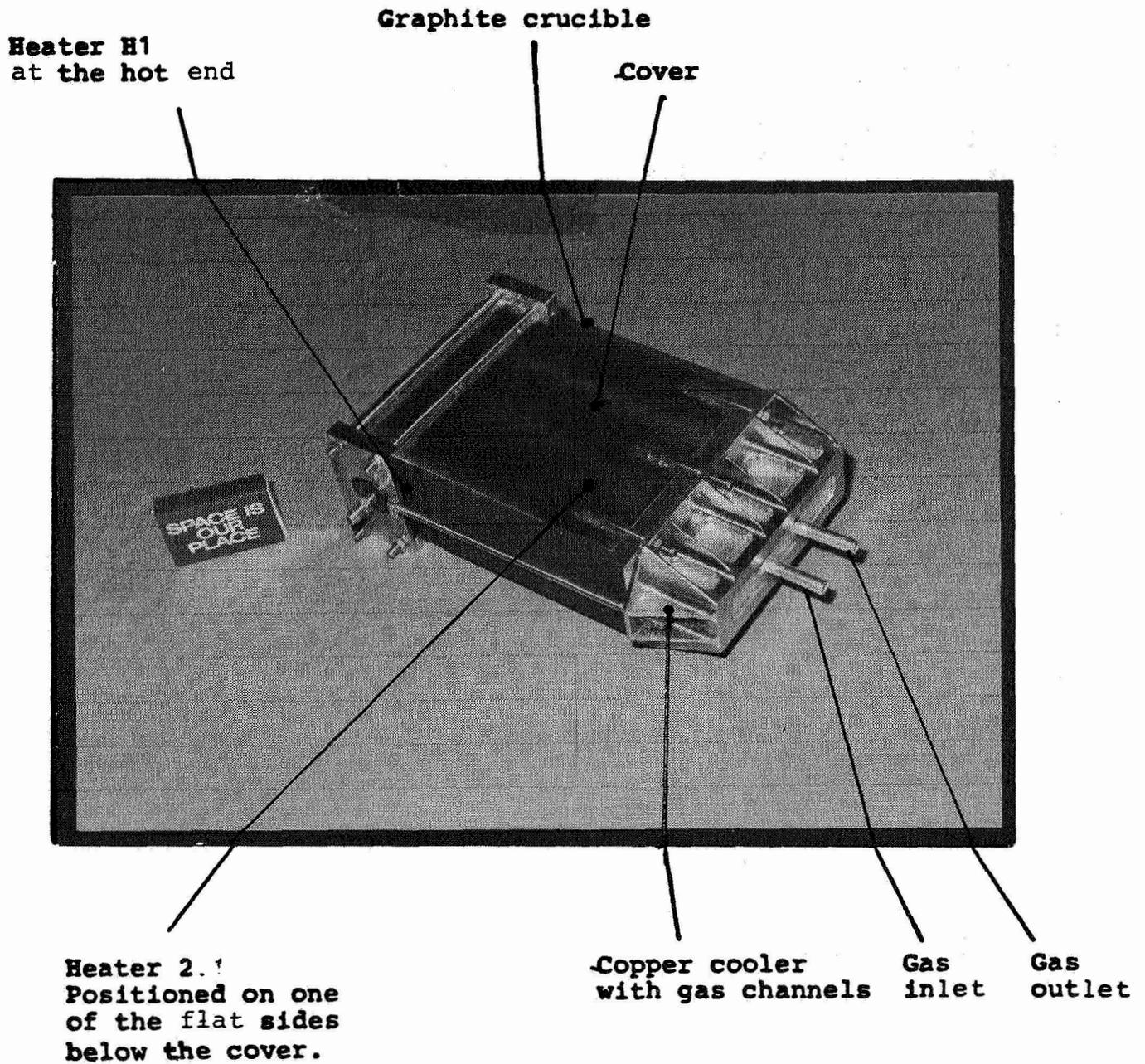


Figure 6. Experiment furnace.

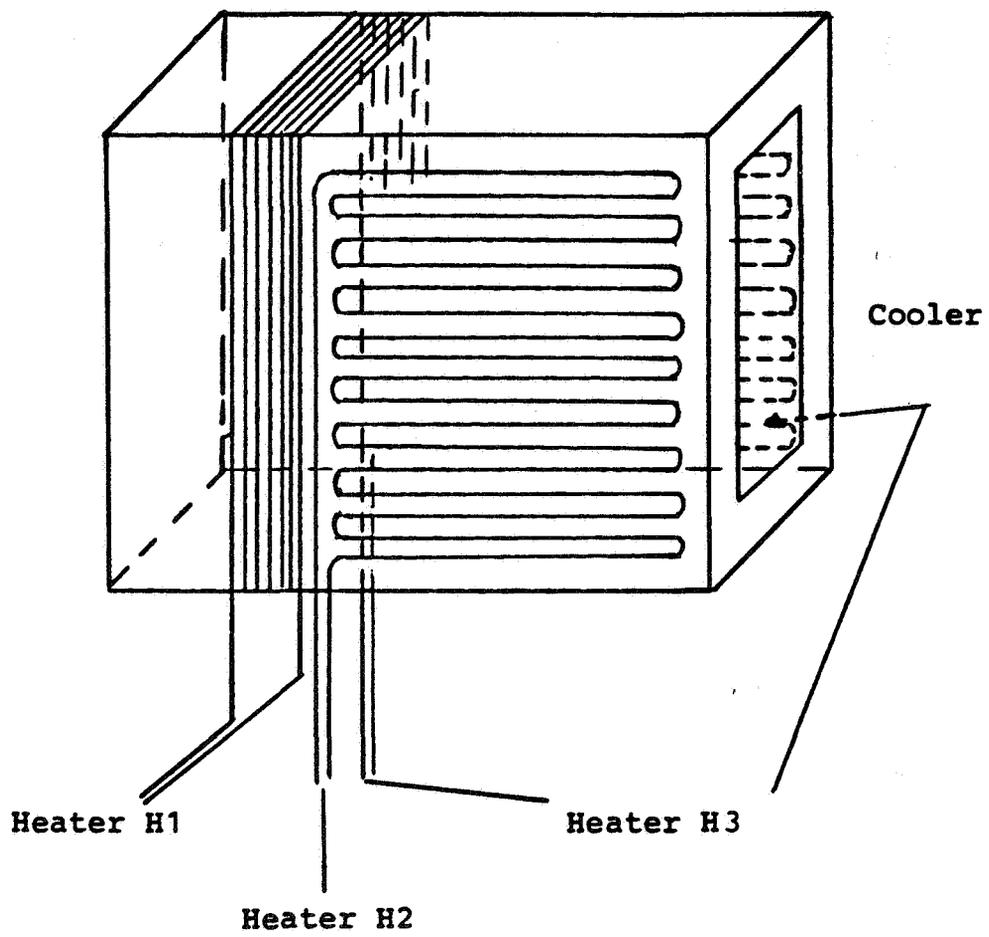


Figure 7. Heater Positioning.

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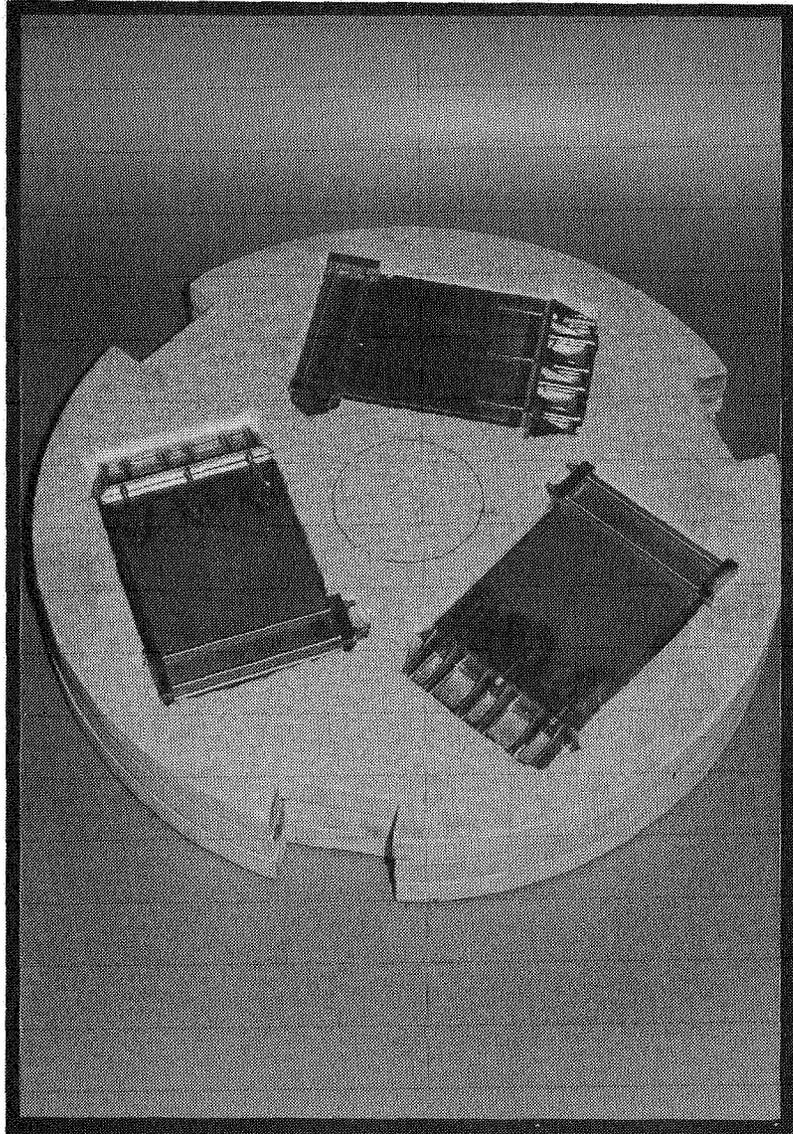


Figure 8. Insulation material around the three furnaces
(four blocks are used, three are visible on the photo).

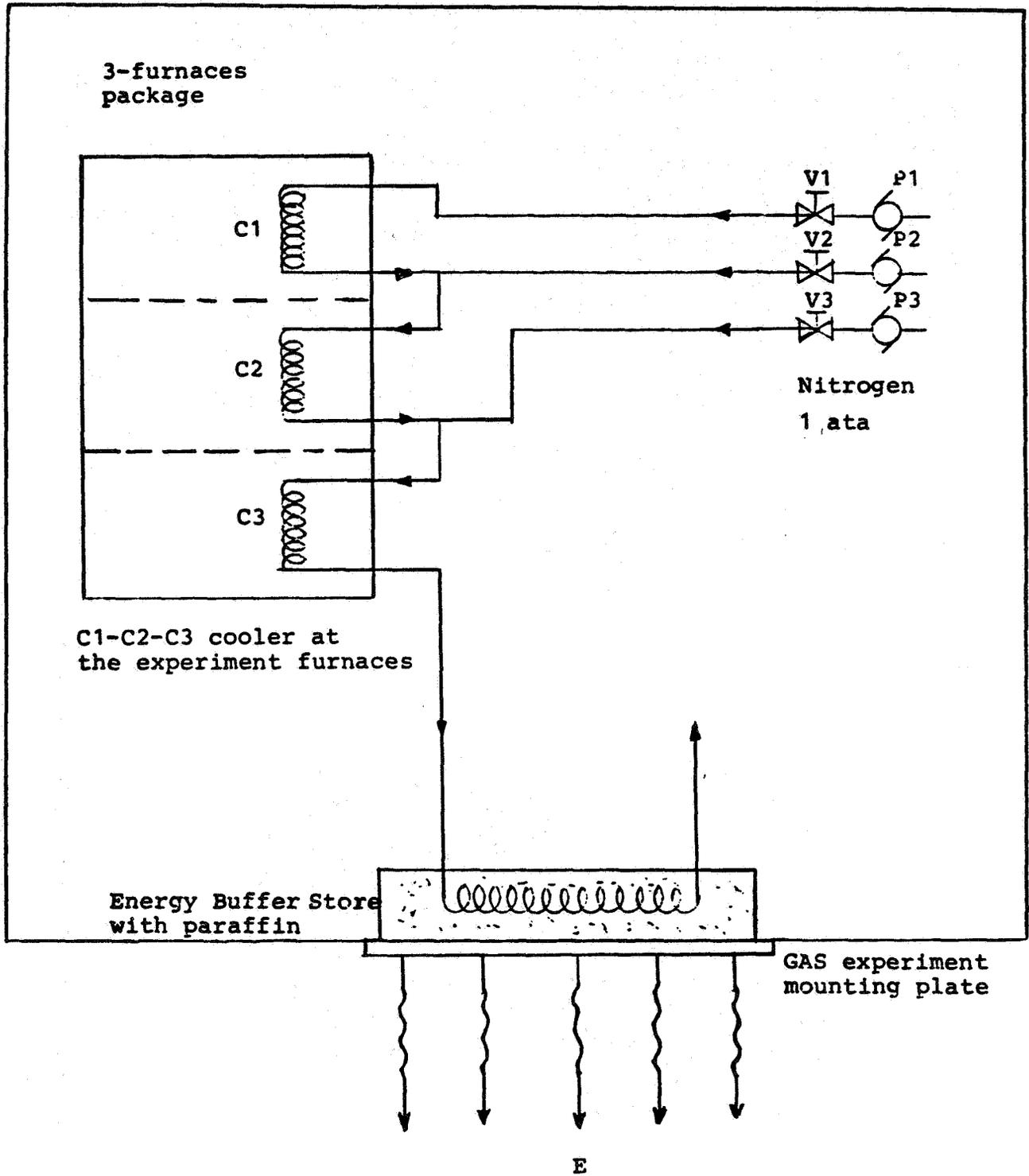
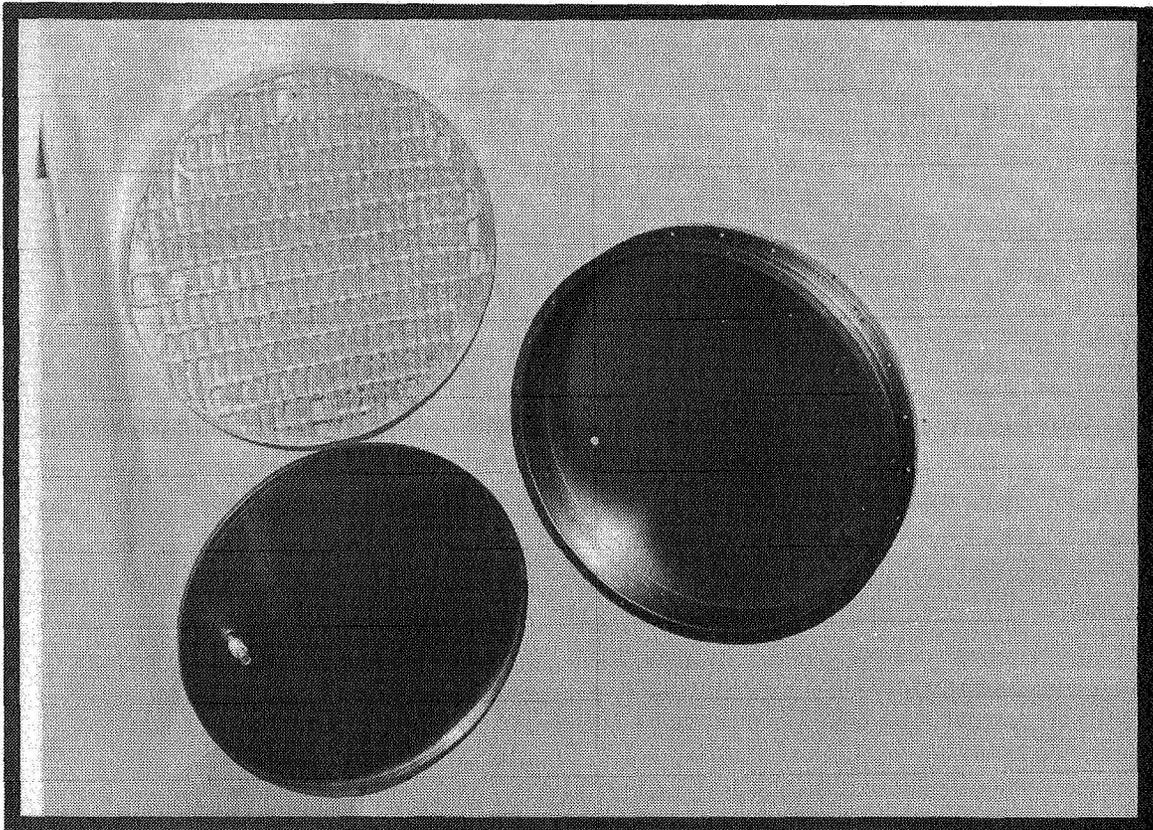
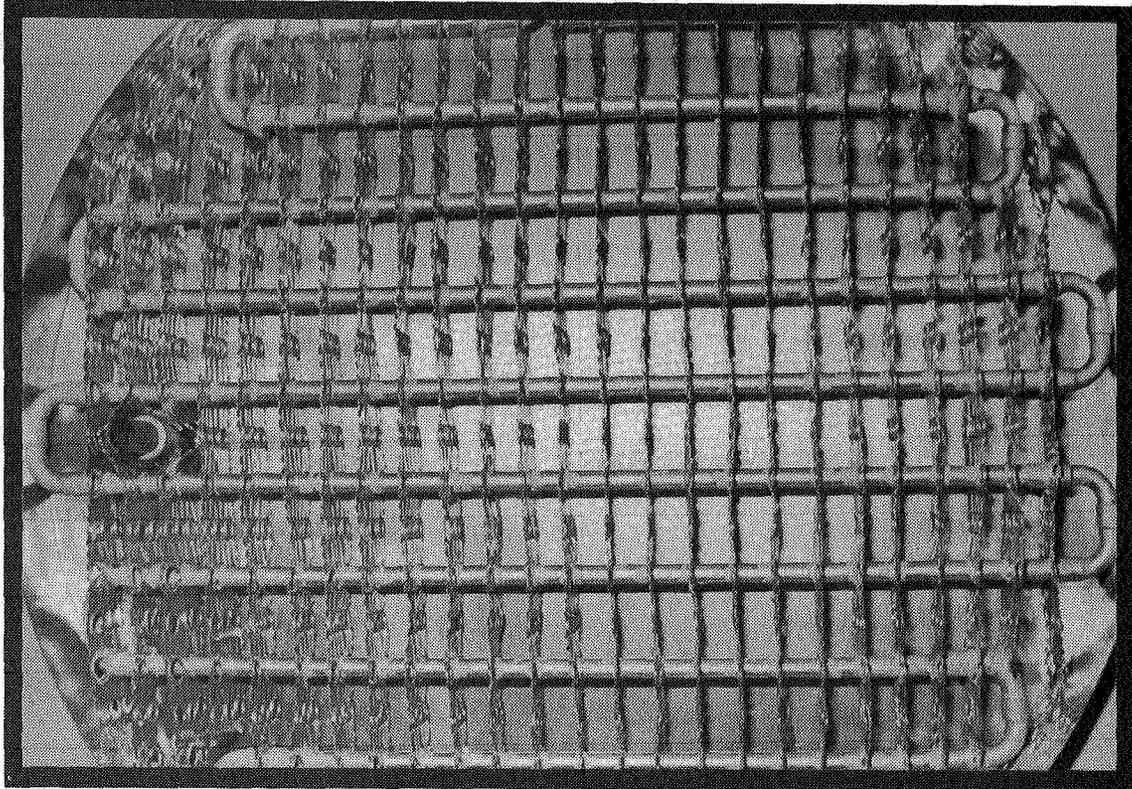


Figure 9. Principle for the cooling system.



The plumbing system with copper tubes and flanges. The box is not filled with the phase change material (paraffin).

Figure 10. Energy buffer store.

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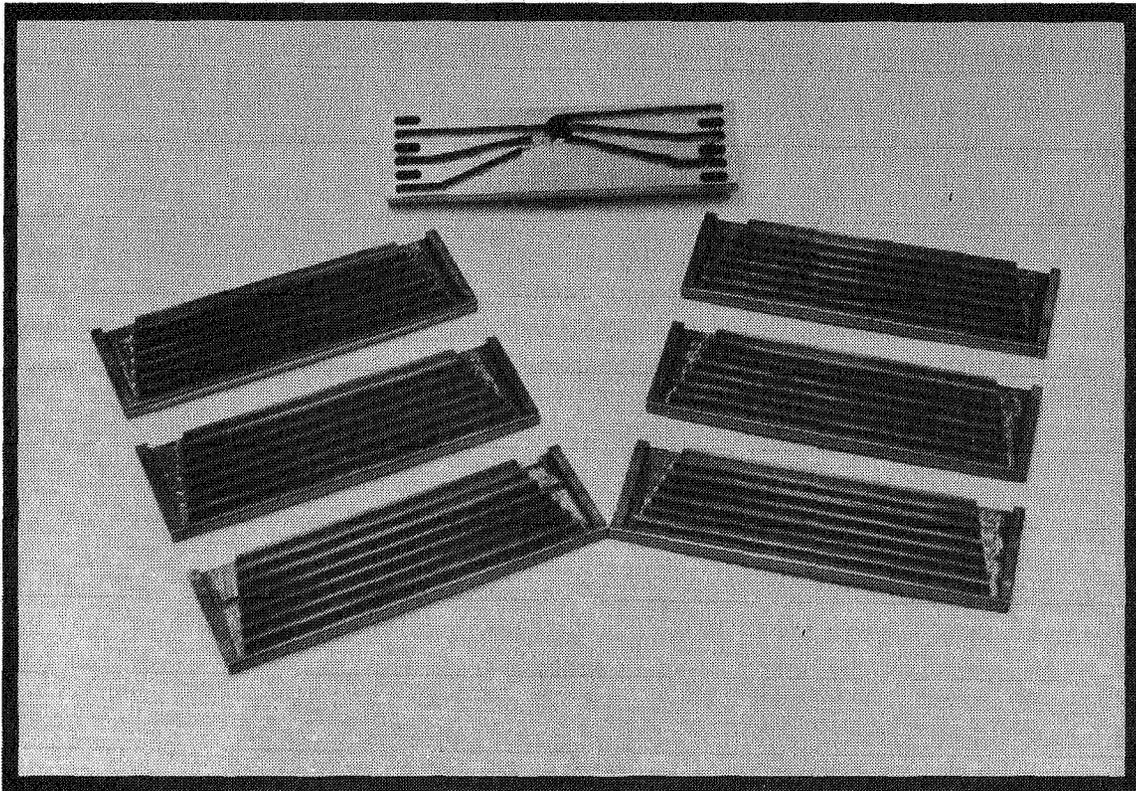


Figure 11. Cooler components showing the copper plates with grooves.

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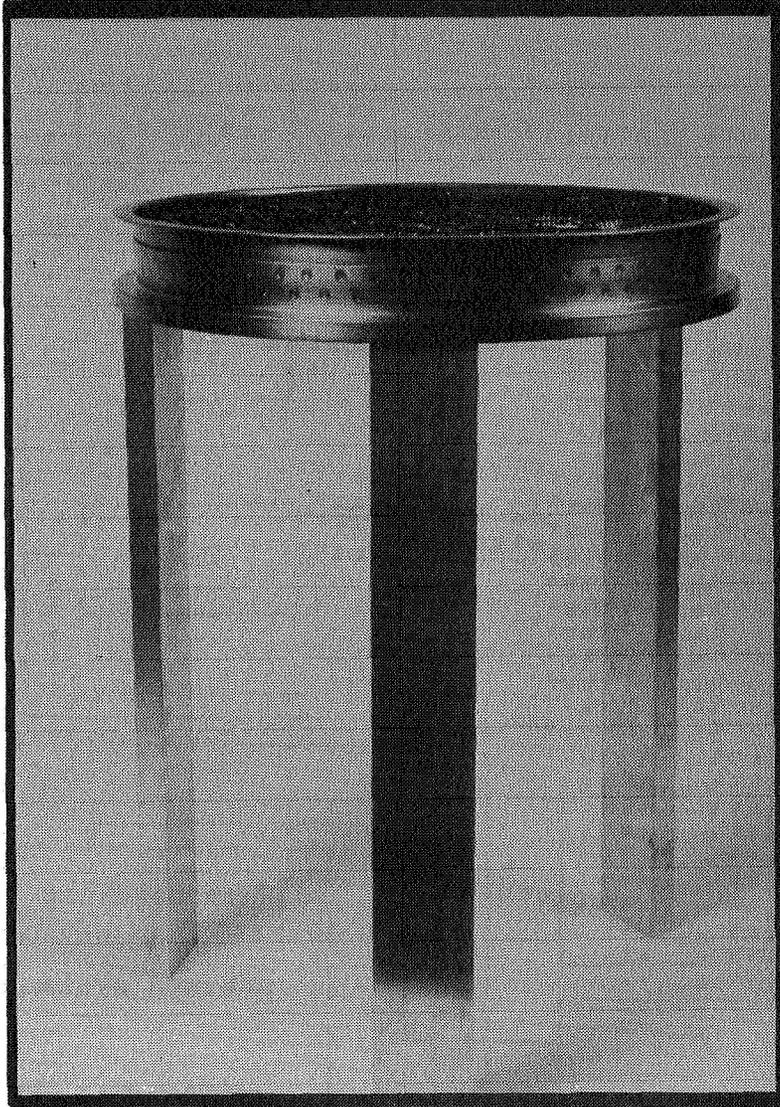
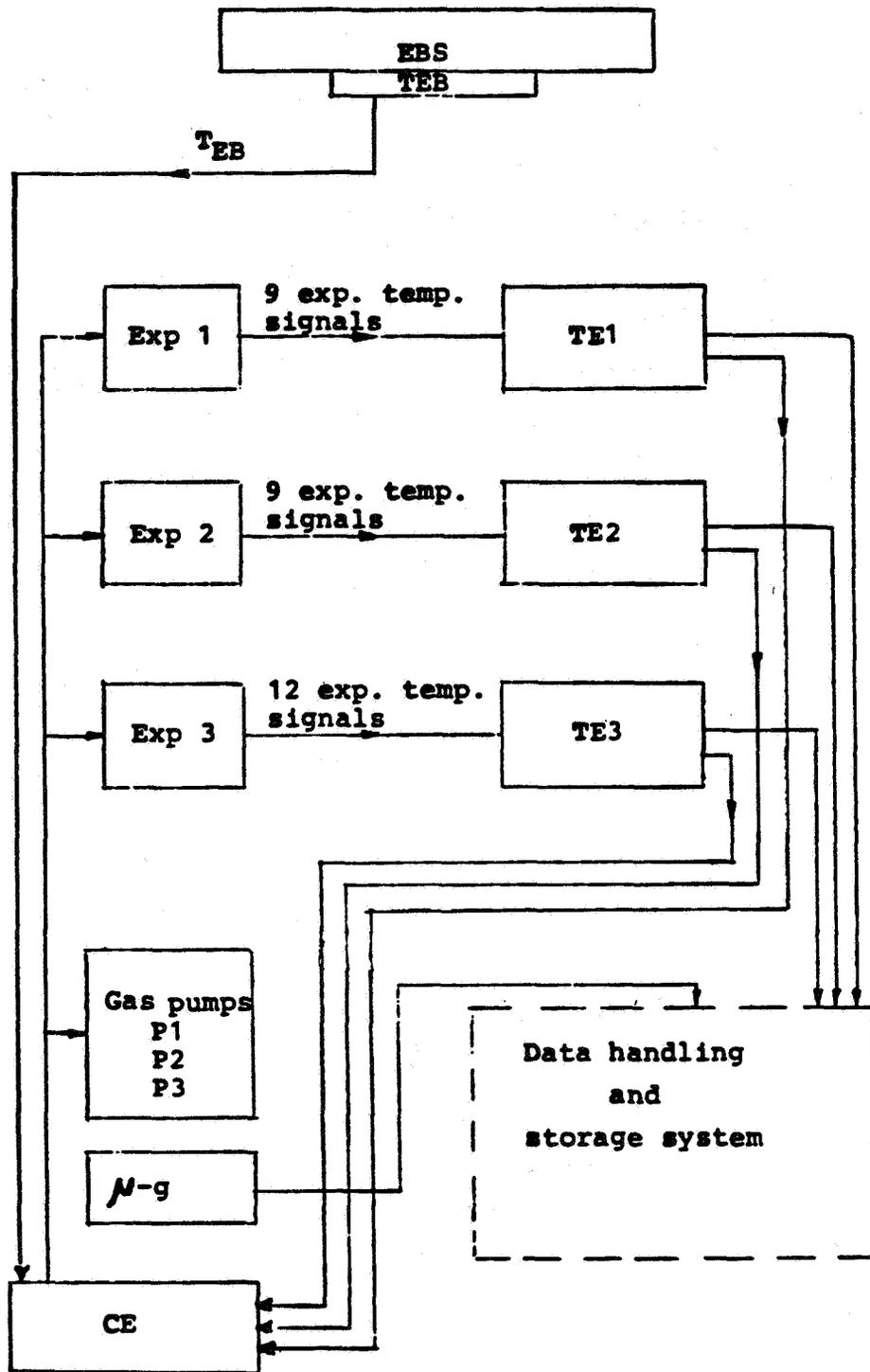


Figure 12. The three struts attached to the energy buffer store (EBS).



- TE Temperature sensing Experiment
- TEB Temperature sensing Energy Buffer
- μ-g Micro-g sensing system
- CE Control Electronic system

Figure 13. Electronic system.

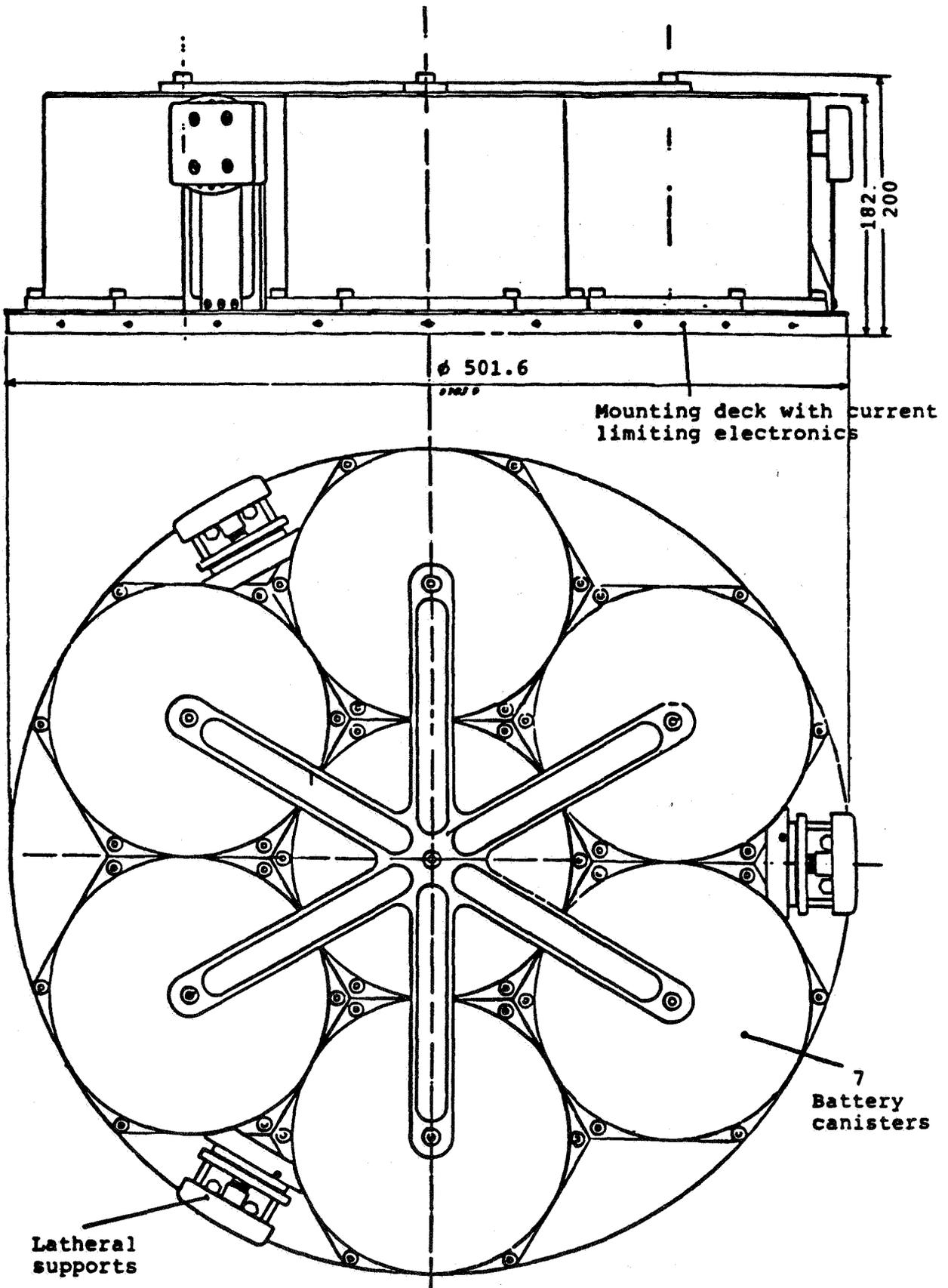


Figure 14. Outline drawing giving the dimensions of the battery system.

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AN UPDATE OF UTAH STATE UNIVERSITY'S GAS ACTIVITIES

8928

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U 8171875

Presented at the
1985 Getaway Special Experimenters Symposium
NASA/Goddard Space Flight Center
Greenbelt, Maryland
October 9, 1985

~~U 8171875~~

INTRODUCTION

In Oct. 1976 NASA announced the GAS program, a concept which allowed the non specialist access to space. This was an event which may well have ushered a new era into space activities. We have all paid lip service to the concept of the availability of space to all people rather than just to the "professionals". This program is one small step along that lengthy path. In the creation of the GAS program NASA has created a mechanism which is rare indeed for any bureaucracy. It has created a program which is designed to be responsive to the needs of those outside the system to institute change. In my interaction with the personnel of the GAS program I have met many individuals who have grasped the concept of change and encouraged it. In this paper I wish to present my version of the impact this program has had on Utah State University, the institution at which I am a professor.

USU SPACE INVOLVEMENT

Utah State University has had a fairly long involvement in the space program. The program as it now exists had it's initial roots in a small atmospheric research program, which grew into a rocket program in the early 1960's.

Table 1 shows some of the highlights of the space program at USU.

TABLE 1

1960--1970	ELECTRO-DYNAMICS LAB ROCKETRY---AURORAL AND IONOSPHERIC PHYSICS
1970--1980	CENTER FOR ATMOSPHERIC AND SPACE SCIENCES THEORETICAL PLASMAS--IONOSPHERIC PHYSICS MIDDLE ATMOSPHERE CHEMISTRY AND PHYSICS ROCKETRY--BALLOONS--AIRCRAFT--SATELLITES--RADARS
1970--1980	SPACE DYNAMICS LABORATORY CRYOGENIC IR TECHNOLOGY--INTERFEROMETRY INTERFEROMETERS--CRYOGENICS--ROCKETS
1980--1984	CASS, SDL & SHUTTLE VCAP-----STS-3 & 51F VEHICLE CHARGING AND POTENTIAL ISO-----STS-9 & 20 IMAGING SPECTROMETRIC OBSERVATORY FACILITIES CLASS INSTRUMENT CIRRIS-----STS-C & ? CRYOGENIC INTERFEROMETER(S) DOD INSTRUMENTATION UARS-----1989-90 UPPER ATMOSPHERE RESEARCH SATELLITE

Given the existence of a vigorous space program at the university, it was fairly natural that USU should get involved in the GAS program at an early stage. My own involvement began when Gil Moore, whom I had known for several years prior to that time, called me, after having committed to the first of the GAS payloads and said, after describing the program briefly, "What are you going to do with this if I give you half of it?" At that time I was concerned by the lack of hands-on opportunities in space for undergraduate students, and out of Gil's offer and my concerns at the time came the present USU GAS program.

ORGANIZATIONAL CONFIGURATION OF A STUDENT GAS PROGRAM

The program at USU is centered around a scholarship program. In this program, seniors in high school are encouraged to write a proposal to do an experiment in space. As you may imagine this requirement in itself represents a considerable selection in the student. Only those students who have been interested in science and space will have very much to propose. We choose three students per year. Each will have a four year tuition scholarship provided he or she maintains an adequate grade point average. Where possible we help them find part time jobs in some of the space related research areas at the university. The success of these students has been such that it is not difficult to find work for them. The vice president for research supplies \$500.00 per year per student for supplies and other costs associated with their experiments. It is a source of pride to me that we have a significant number of "walk-ons" in this program. Students fairly often come and say that they want to be involved. After explaining that there are no direct resources for them and that there is an enormous amount of extracurricular work involved, they are encouraged to participate in the program in a variety of ways. To date several of these students have flown experiments.

The essence of the program as it is designed at USU is that it is a "hands-on" program. Each student is required to do a large amount of the construction of the experiment. Certain items, to be discussed later are supplied but most items flown to date have been constructed by the students themselves. In many cases trades are made between the students for services in which one or the other of the students is particularly skilled.

Some resources must be made available to the students if one wishes to develop an effective team of students. Perhaps the most important of these is interaction with each other and suitable faculty members. To encourage this interaction one needs several items:

Space must be made available where the students can see each other frequently. This is probably the single most difficult item to supply in a university environment. We have been able to make available to the students, several rather spartan offices and a small amount of laboratory space.

A weekly meeting of the students and faculty is held during the school year and on occasion during the summer. The weeks before delivery, these meetings often occur every night.

A design philosophy must be established. If this is not done an inordinate amount of time is spent in negotiating for resources. The philosophy should be flexible enough to accommodate many students, but must represent the constraints of the GAS environment.

Certain of the major hurdles which any experiment presents should be either eliminated or minimized for the students. The student should be made aware that this is being done however. The hurdles which are minimized at USU are the physical configuration, the controllers and the power supply choices. A moderate amount of faculty help is supplied and modest resources are supplied.

WHAT HAVE WE DONE?

Given this philosophy, what has been accomplished? Table 2 shows what has been done so far in the program.

TABLE 2

STS-4 ONE PAYLOAD G-001 TEN EXPERIMENTS
This was the first GAS payload. See Fig. 1 for a picture of the payload and the participants.

41-B (STS-11) TWO PAYLOADS G-004&8 EIGHT EXPERIMENTS
These two payloads represented our first use of the 60 lb canister. The best bargain in space! Three of the eight experiments represented opportunities extended to other institutions including another university in the state, a high school and a foreign university. See Fig 2.

41-G (STS-17) ONE PAYLOAD G-518 FOUR EXPERIMENTS
This payload may represent the fastest turn-around in history. The payload was delivered six weeks after recovery of the payloads from the previous flight.

51-B Participation in the NUSAT program. Langmuir probe, orientation sensor, and flasher units were all USU student experiments. Assembly at USU.

Participation in a canister being flown by personnel from University of Mexico. One USU student experiment on board.

DESIGN PHILOSOPHIES

For the type of operation we are running, in which several individuals or groups are flying in a single GAS can it is important to keep each experiment as isolated from the rest of the pack as possible. In order to accomplish this we have designed a standardized "spacepak" for use by each experimenter. These will be described later.

A standardized controller is supplied for each spacepak. This controller was the only item not developed by an undergraduate. A masters degree candidate designed and built our controller and at present programs the ROM which contains the program for each of the students. This design incorporates a 16 channel multiplexer connected to an eight bit A/D converter, up to 16K bytes of program space and the ability to "go to sleep" during times when no action is required. This is required because of the need to conserve power. Data are stored in ROM using an on-board ROM burner. Up to 32K Bytes are available for storage. Control is accomplished through eight output lines, each capable of supplying three amperes.

Power sources are an ever present problem. We have found that lead acid batteries in the 2.5 and 5 ampere-hour sizes are adequate for most of our experiments. Because weight is seldom the most severe constraint in a GAS can, the weight penalty for using the lead acid technology is tolerable. It should be noted that the lead acid batteries, while quite good, do not satisfy ones needs in all cases. Alkaline batteries, can supply more energy in some cases, but since the batteries can be used only once, there is no chance for test of the individual cells.

One should always maximize the thermal isolation of the experiments from each other to the extent possible. For this reason we have encouraged the use of foam and epoxy structures for containers. Care should also be taken to eliminate radiative coupling between various experiments and the walls of the can. At some temperatures radiative energy transfer can dominate over conductive transfer unless care is taken.

We try to minimize inter-pak communication. On most of our experiments we are able to keep the interaction limited to that necessary to turn the paks on and off.

OUR CURRENT CONFIGURATION

After our first attempt to build a large number of experiments onto a single frame, it was clear that the tri-wall construction, while a very good, stiff framework, presented substantial difficulty when several experiments were to be flown together. Our aim in the program is to enable as many students as possible to fly experiments and the ability to incorporate as many experiments into a GAS can as possible. An analysis of the experiments which were flown on G-001 and a look at the experiments proposed for G-004 and G-008, showed that a large

percentage of the experiments really required a fairly small volume. As a consequence a standard "spacepak" was designed. This pak, which is shown in Fig 3, is based on a hexagon which will fit inside the 19 inch circle allowed for the GAS canister. The length allowed can easily be varied for different experiments, but it has been found that a four inch height (outside dimensions) will fit almost all experiments, especially if the allotted shape is known in advance of the initial design.

IMMEDIATE PLANS

At the present time there are a total of 12 experiments under development at USU. We anticipate that at least six of these, or one five cubic canisters worth will be ready for integration by Dec. 1 and that the remainder will be ready in by late spring.

The proposed experiments include:

A study of the velocity of a bubble in water, under the influence of a temperature gradient.

A reflight of the Scott Thomas's experiment on surface tension driven convective flow. (Marangoni flow)

The study of surface waves in zero-G. (Capillary waves)

Crystallization in Zero-G. (Vapor phase and liquid phase)

Bio gas generation.

Penicillum growth.

A study of undamped oscillations in a vacuum and Zero-G.

Several other experiments still being formed.

SPIN OFFS

The GAS program has had several spin off effects at USU. One of the most recent of these is the creation of a new program in space engineering which has created a Center for Space Engineering. For this program we have created several new courses and combined them with course work currently been given and created a degree specialty in space engineering. This specialty can be obtained at the masters degree level in the departments of Mechanical, Civil and Electrical engineering. The program has created considerable interest among students and there are already enough students for our first class, even though we have not formally announced the program. Experimentally the program will be based on having each Masters degree student involved in some activity in which a space experiment is flown. Some of these will be free flyers ejected from GAS canisters. A variety of experiments have been designed which include structural damping

in zero-G and vacuum, environmental studies and reentry vehicles to study the lower atmosphere. In Figures 4 & 5 we show some of the structures in which the experiments will be housed and a proposed additional "boosted" version of the satellite. Fig 6 shows a computer created for control and housekeeping functions. This computer and up to 30 megabytes of battery backed RAM can be housed in less than one third of the satellite structure.

You will hear today from two of the groups that have taken advantage of the USU expertise to start their own programs, which will hopefully continue into the future. These are the Weber State College group that was involved in NUSAT and the group from the University of Mexico which is currently building a payload at USU. The activities of these groups makes it clear that the GAS philosophy of space science is spreading.

LONG RANGE PLANS

Every enterprise needs goals. This is especially true of a program like the USU GAS program. In addition to planning the continuance of the existing and expanded programs, we feel the need for a major central effort. We have therefore created a five year goal, which has many hurdles in front of it before it can become a reality. If we accomplish this goal, we will construct a small radio controlled satellite with enough aerodynamic control and enough propulsion on board that we can command partial re-entry into the atmosphere, control the space craft to execute a plane change using aerodynamic forces, and then recircularize the orbit. This program will teach the students who participate in it not only the ordinary structural and environmental material, but will require knowledge of aerodynamics, orbital mechanics and control theory as well. Each us luck.



Figure 1. First USU Payload with Participants

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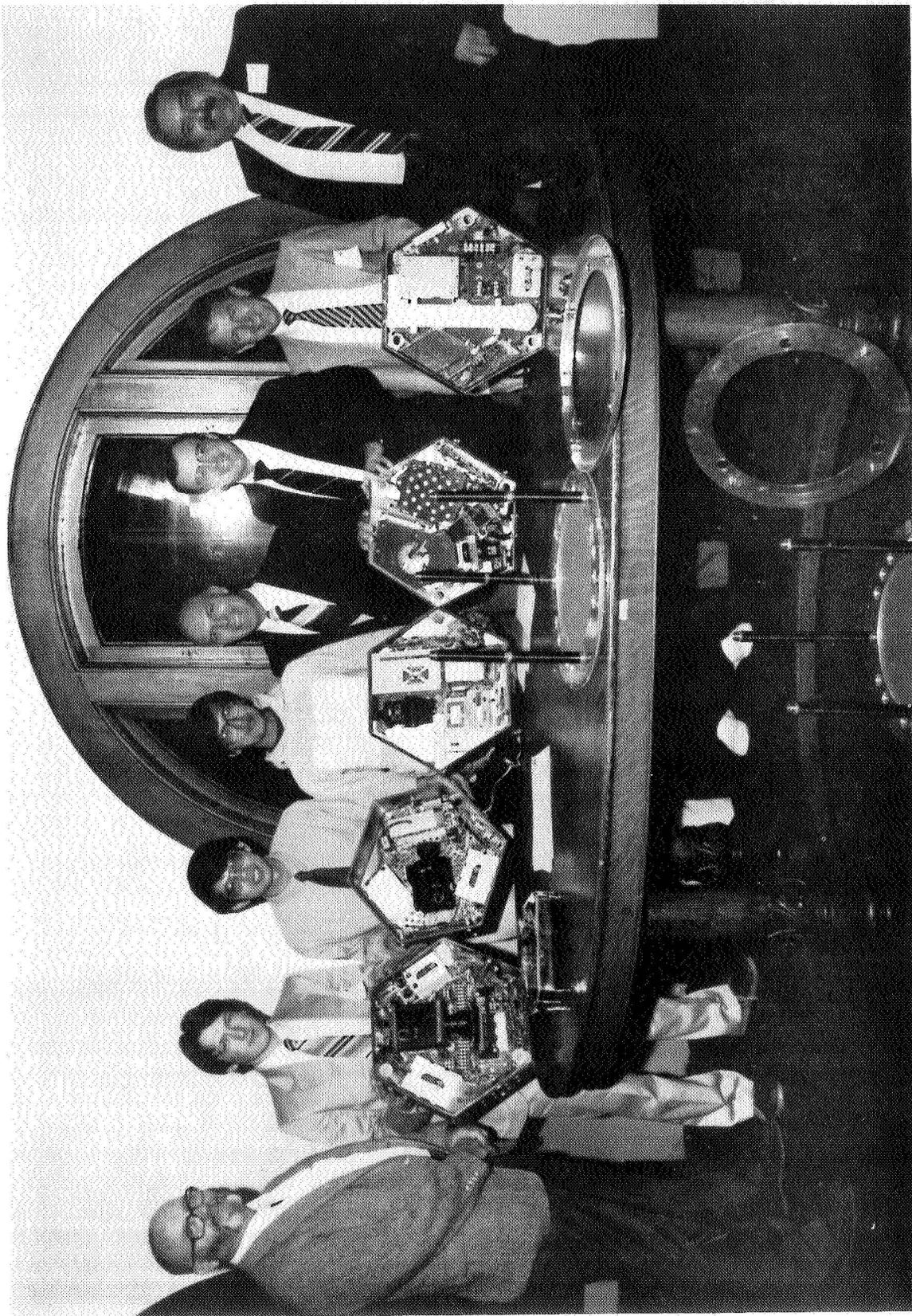


Figure 2. Second Payloads

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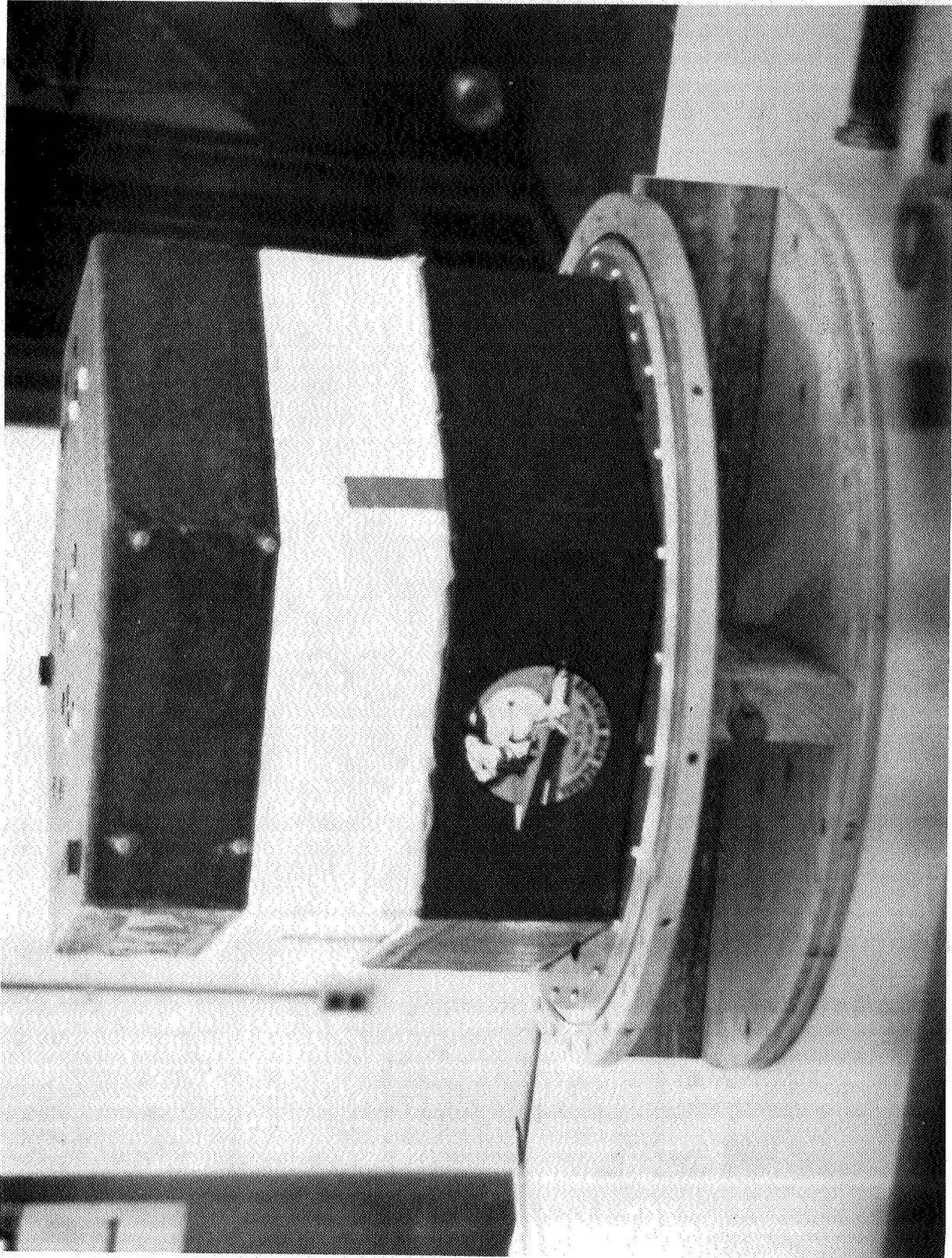


Figure 3. Space Pak Configurations

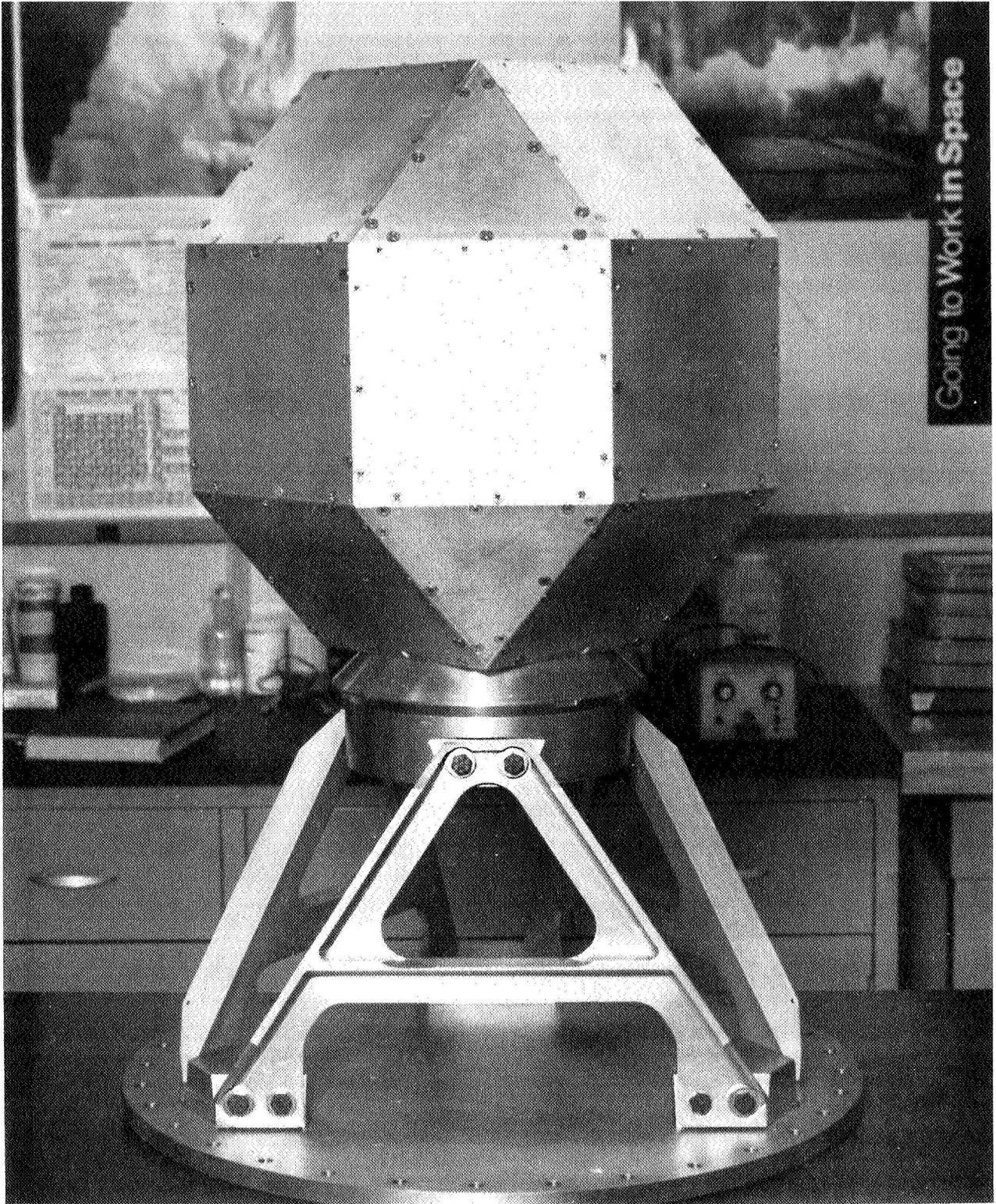


Figure 4. Satellite Structure

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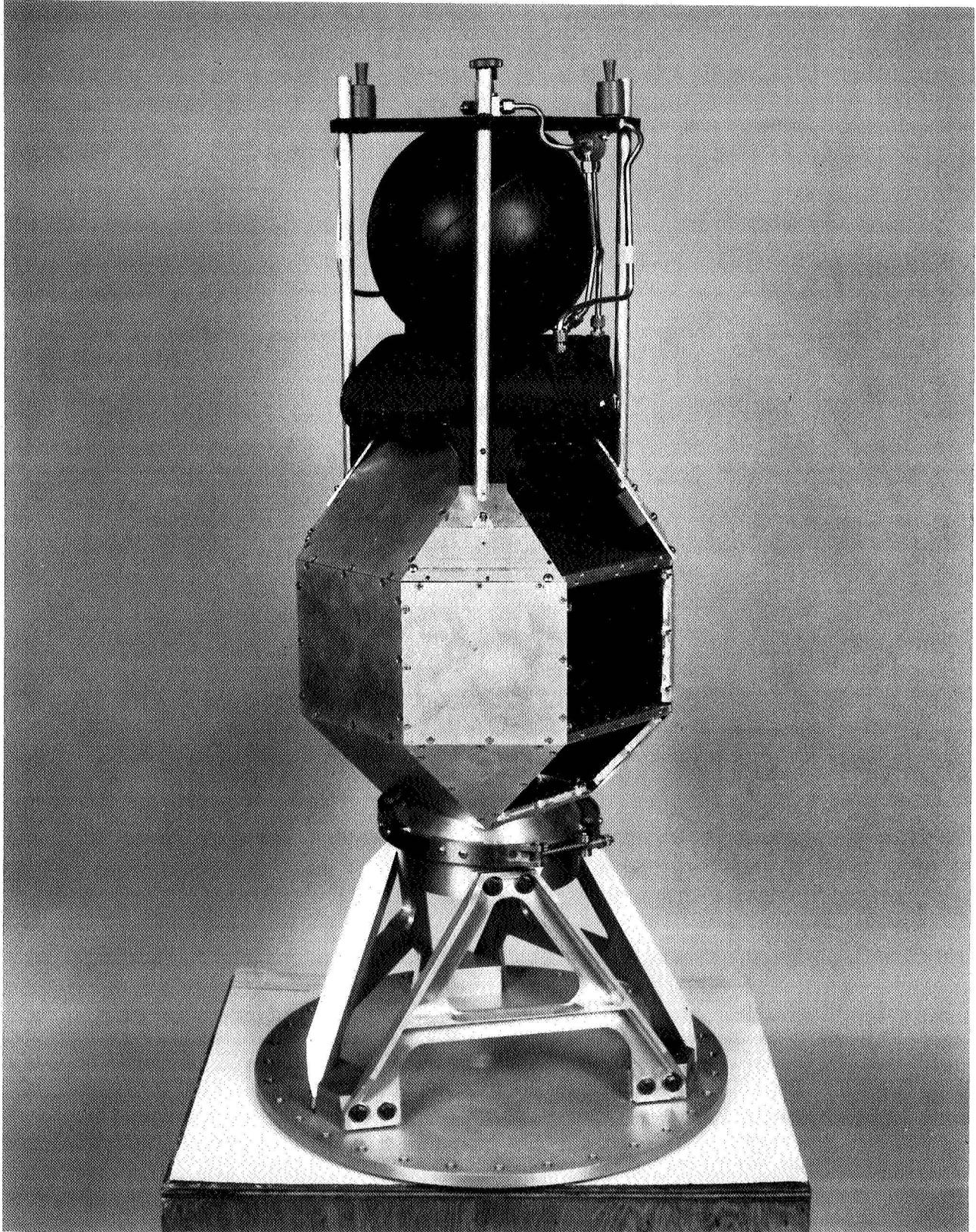


Figure 5. Boosted Satellite

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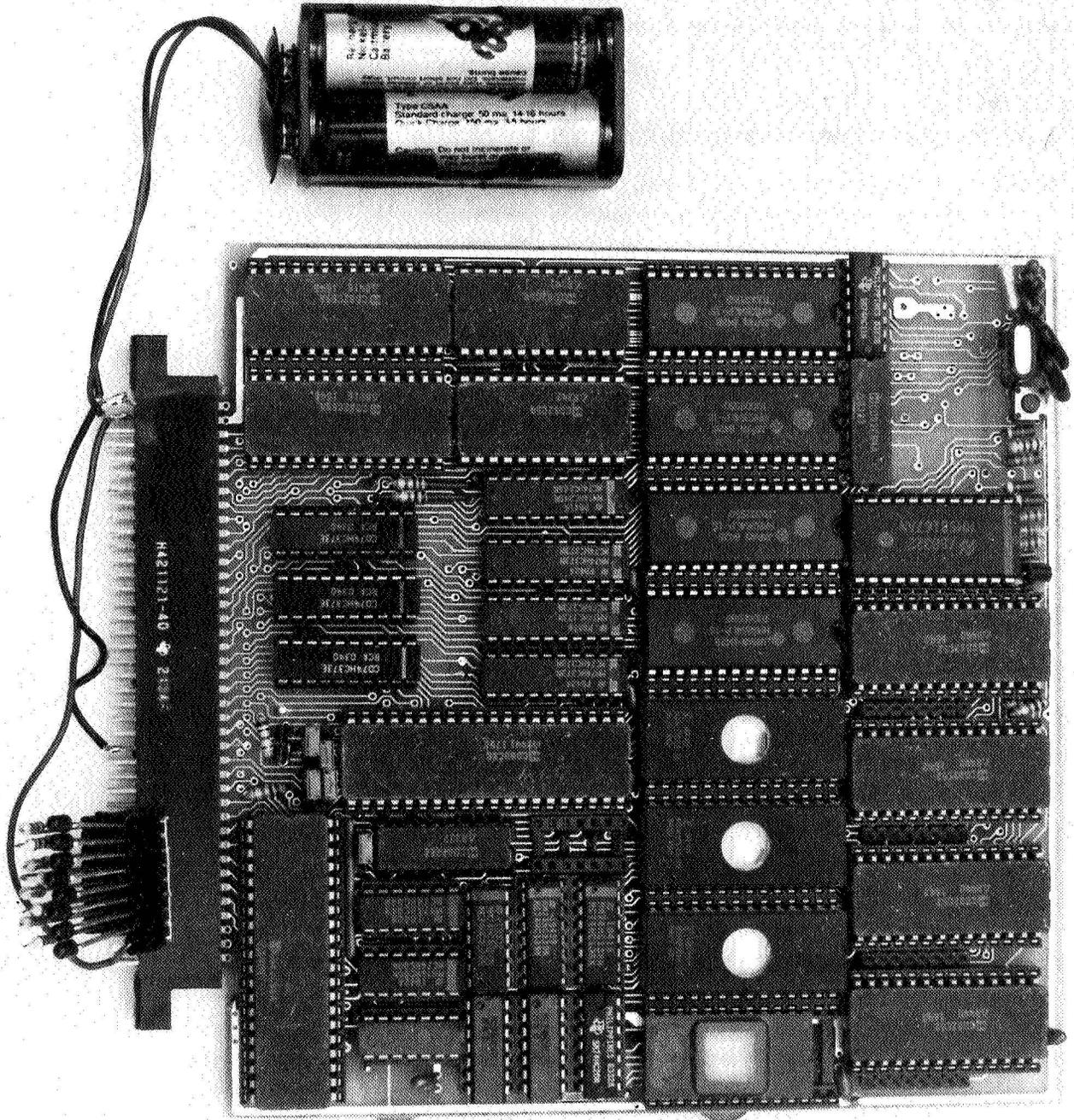


Figure 6. Computer

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6P.

STRUCTURAL INTEGRITY OF GAS EJECTION SYSTEM

Mark Cascia
TSI

TV850050

8979

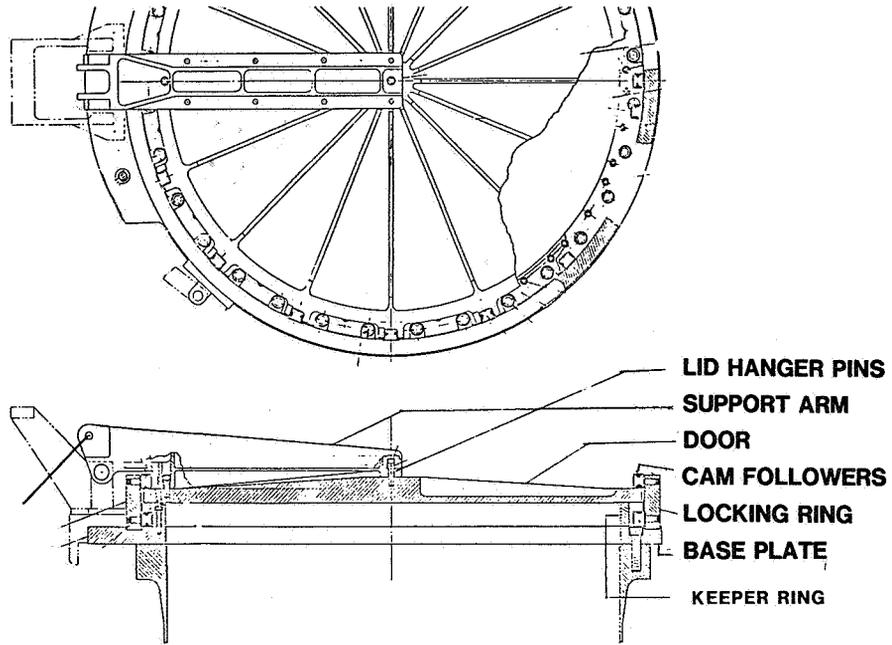
Get Away Special (GAS) Experimenters Symposium 1985
October 8-9, 1985

- **COMPONENTS ANALYZED**
- **APPLICABLE ENVIRONMENTS**
- **LOADS**
- **SPECIAL CONSIDERATIONS**
- **EXPERIMENTER CONSTRAINTS**

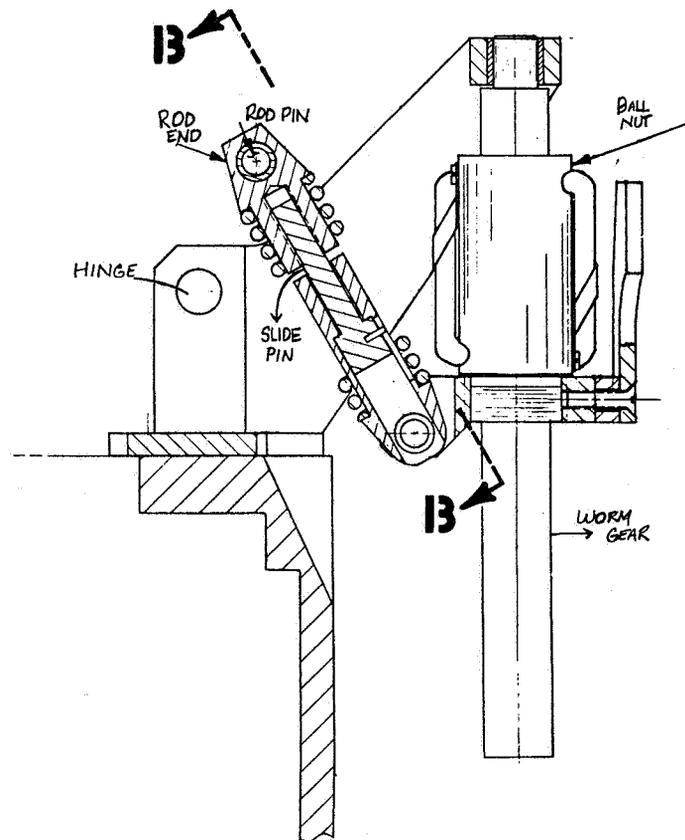
COMPONENTS ANALYZED

- **FULL DIAMETER MOTORIZED DOOR ASSEMBLY (FDMDA)**
- **ACTUATING ASSEMBLY**
- **GAS EJECTION SYSTEM**
- **LOWER END PLATE**
- **BATTERY BOXES**

8000-042

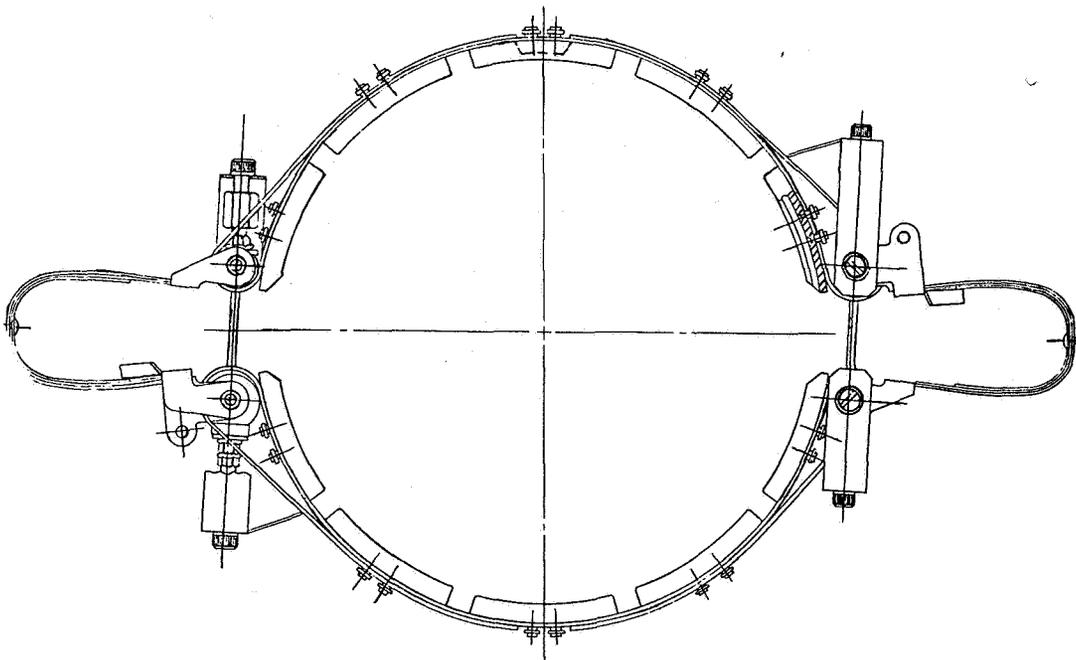
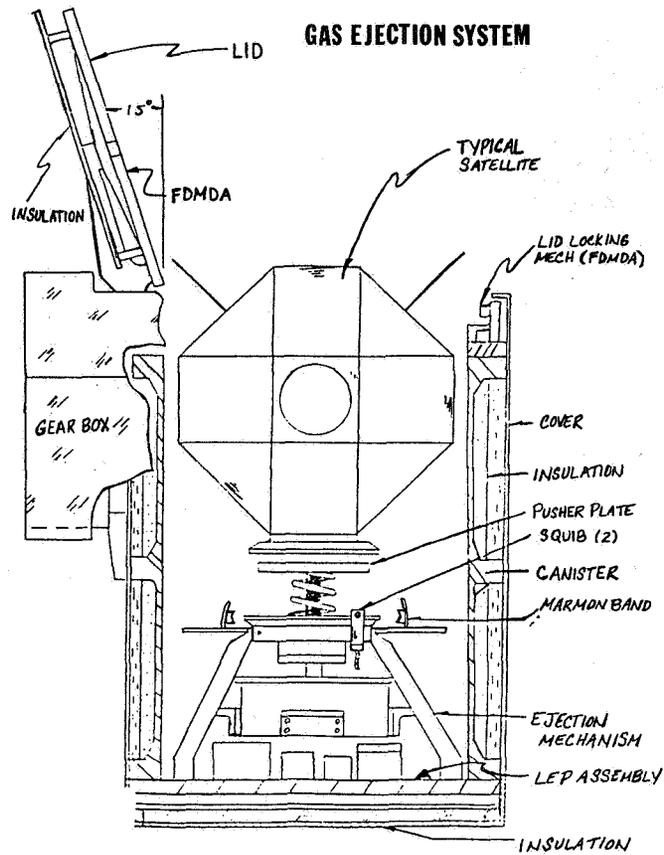


FDMDA ASSEMBLY



ACTUATING ASSEMBLY

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**MARMON BAND
CLAMP**

ENVIRONMENTS

- **LIFTOFF**
 - **LOW FREQUENCY TRANSIENTS (QUASI-STATIC)**
 - **HIGH FREQUENCY RANDOM/ACOUSTIC LOADS**
- **LANDING**
 - **EMERGENCY LANDING LOADS (EXTREME CASE)**
 - **THERMAL GRADIENTS**
 - **ANOMALIES**
- **ON ORBIT**
 - **THERMAL GRADIENTS**
 - **ANOMALY**

LOADS

- **OBTAINED FROM COUPLED LOADS ANALYSIS OF ORBITER**
- **UPDATED FROM ACTUAL FLIGHT DATA**

**LOAD FACTORS FOR STRUCTURAL ASSESSMENT OF
APC, EAPC, AND GAS BEAM MOUNTED PAYLOADS**

EVENT	LOAD FACTOR, G			ANGULAR ACCELERATION, rd/s ²		
	NX	NY	NZ	$\ddot{\theta}_X$	$\ddot{\theta}_Y$	$\ddot{\theta}_Z$
LIFTOFF						
LOW FEQUENCY	±7	±7	±6	±75	±20	±55
VIBRATION	±5.4	±8.0	±5.4			
COMBINATION (RSS ON ONE AXIS AT A TIME)						
1	±8.8	±7	±6	±75	±20	±55
2	±7	±10.6	±6	±75	±20	±55
3	±7	±7	±8.1	±75	±20	±55
LANDING	±6	±7	±8	±85	±30	±50

SPECIAL CONSIDERATIONS

- **BATTERY BOXES**
 - LEAKAGE IN VACUUM
 - SEALED TO MAINTAIN 1 ATMOSPHERE
 - MET ADDITIONAL SAFETY REQUIREMENTS OF NHB 1700.7A FOR SEALED CONTAINERS/PRESSURE VESSELS

- **ANOMALIES**
 - DOOR FAILED OPEN (LANDING)
 - - SAFE IN MOST UNFAVORABLE ORIENTATION
 - SATELLITE LOOSE IN CAN (LANDING, LIFTOFF)
 - - DOOR DESIGNED FOR CONTAINMENT
 - SATELLITE EJECTION WITH DOOR CLOSED (ON ORBIT)
 - - IMPACT STRENGTH DETERMINED

EXPERIMENTER CONSTRAINTS

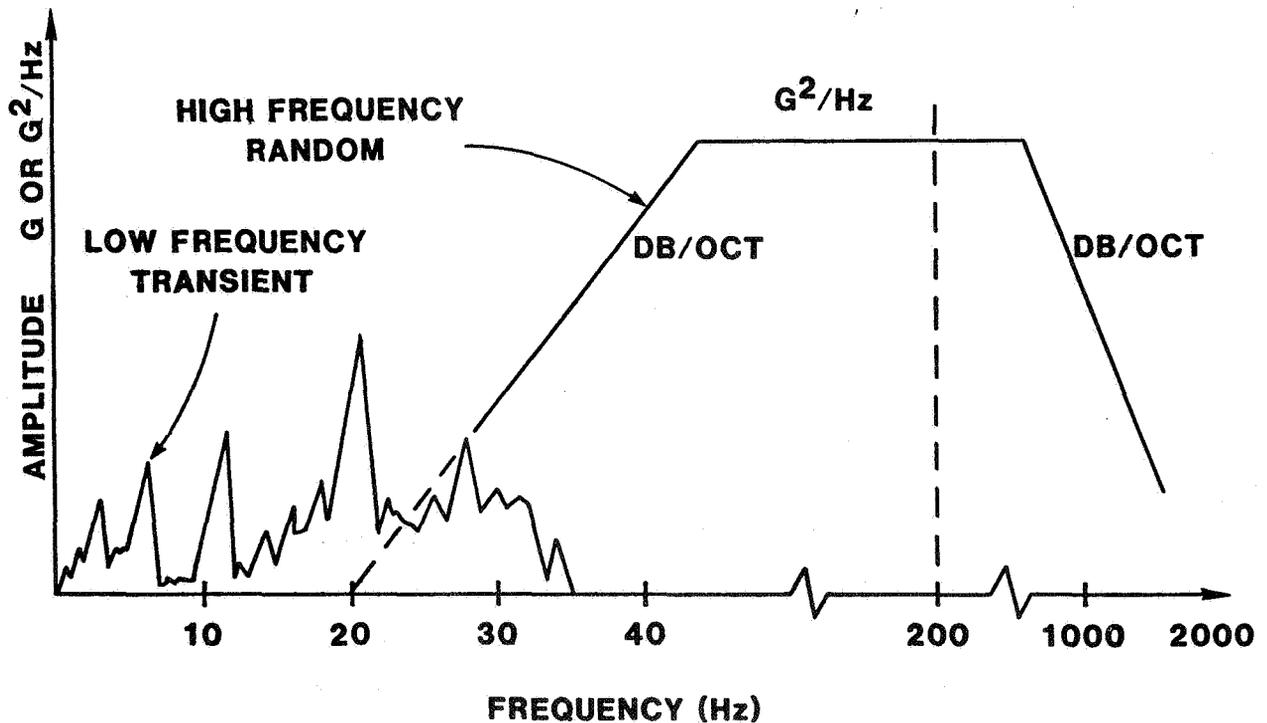
- **WEIGHT LIMITATIONS (150 LB SATELLITE)**
 - BASED ON 2 CANS PER GAS BEAM
 - LIMIT IMPOSED BY INTERFACE (MOUNTING) TO ORBITER
- **C.G. LIMITATIONS (1/2 INCH FROM CENTER OF CAN)**
 - PREVENTS EXCESSIVE MOMENTS ON MOUNTING HARDWARE
- **RESONANT FREQUENCY**
 - MAIN ASSEMBLY > 25 HZ
 - SUB ASSEMBLIES > 35 HZ
 - - DERIVED FROM FREQUENCY CONTENT OF QUASI-STATIC (LOW FREQUENCY) LOADS
 - - REDUCES EFFORT AND EXPENSE REQUIRED IN PERFORMING COUPLED LOADS ANALYSIS

EXPERIMENTER CONSTRAINTS CONTINUED

● EJECTION VELOCITY

- < 2 FT/SEC
 - WOULD NOT CLEAR ORBITER
- > 7 FT/SEC @ 150 LB SATELLITE WEIGHT
 - DOOR WOULD NOT CONTAIN SATELLITE IF EJECTED PREMATURELY
 - BASED ON RUPTURE STRENGTH OF DOOR
- TYPICAL EJECTION VELOCITIES
 - 4 ± 5 FT/SEC

TOTAL LIFTOFF DYNAMIC LOADS



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D27-34
17P.

THE CAPILLARY PUMPED LOOP (CPL) GAS EXPERIMENT G-471

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Greenbelt, Maryland

October 9, 1985

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NC 999967

INTRODUCTION

The Capillary Pumped Loop (CPL) experiment, G-471 is a thermal control system with high density heat acquisition and transport capability. The CPL consists of two capillary pumped evaporators with integral heaters, a fluid loop charged with ammonia (NH₃), a condenser plate (heat sink), and various control electronics. The purpose of the experiment is to demonstrate the capability of a capillary pumped system under zero gravity conditions for use in the thermal control of large scientific instruments, advanced orbiting spacecraft, and space station components.

A unique feature of the CPL is the capillary pumps, which contain no moving parts. Each pump contains a wick of porous material which is saturated with the working fluid (anhydrous ammonia). As heat is added to the fluid, it evaporates and travels to the condenser, thus transporting the heat (via the latent heat of vaporization) from the heat source to its sink at nearly a constant temperature.

The evaporation process produces the pressure gradient or pumping action that circulates the fluid. This is the same principal that plants and trees use to transport water and nutrients from their roots to their leaves against gravity. The difference is that the CPL employs a closed system to return the fluid directly to the pumps, whereas "Mother Nature" has an open system where the fluid is indirectly returned to the roots by condensation of water from the clouds in the form of rain. It should be noted that the CPL experiment was the first flight of a thermal control system of this type. It was also the first shuttle experiment from the Space Station Advanced Development Program.

The principal investigator of the CPL is Roy McIntosh of NASA's Goddard Space Flight Center, Greenbelt, Maryland.

BACKGROUND

The concept of a capillary pumped loop (CPL) was pioneered by F.J. Stenger of NASA/Lewis in the mid 1960's (ref.1). The design of the CPL is shown in Figure 1. The evaporator contains a porous wick material which produces the pumping action in the closed loop system via capillary forces. The liquid is drawn through the wick to the metallic shell of the evaporator where it vaporizes and then travels to the condenser. The heat is removed at the condenser and the vapor is thus condensed back to a liquid. The liquid is then returned to the evaporator to repeat the cycle. A cross section of the evaporator pump is shown in Figure 2 (from reference 2).

The CPL system has been undergoing development and ground testing at GSFC for the past several years. An engineering model was built for GSFC by the OAO Corporation. It uses ammonia as the working fluid and has demonstrated heat carrying capabilities of up to 6.4 Kilowatts. The CPL ground test engineering model is depicted in Figure 3. This system features eight capillary pumps mounted in parallel (left side of figure). The multiple evaporator pump feature demonstrates the ability to accommodate multiple users on a single thermal control loop. It also provides for heat sharing, whereby heat can be removed from the the system as well as added. Another important part of this system is the two-phase reservoir. By controlling the reservoir temperature, the loop temperature is controlled as well, since the saturation temperature of the working fluid is controlled at the reservoir. This means that the pumps (evaporators) stay at a relatively constant temperature regardless of the heat load or heat sink temperature variation, thus establishing temperature control in the loop. The loop temperature can be varied simply by raising or lowering the reservoir temperature to the desired level. Another salient feature of the reservoir is fluid inventory control. The reservoir can also be used for pressure priming of the pumps during startup operations.

The CPL system also includes a condenser zone (right side of Figure 3). The vapor travels from the evaporator pumps to the condenser where the heat is

removed and the vapor is returned to the liquid state. A chiller (refrigerator) is used on the ground system for the heat removal, while a heat rejection radiator is used for space applications. After the vapor is condensed, the fluid is returned to the pumps through an isolator to complete the cycle. The isolator allows the pumps to operate individually within the loop. It also prevents vapor from travelling the "wrong way" or backing up, thus providing direction to the flow within the loop.

CPL-GAS

The next step in CPL development is zero-g verification of its performance. Since the capillary pumps are sensitive to the effects of gravity, space flight experiments are required. The GAS system was chosen for the initial experiment due to its low cost, ease of integration, and frequent flight opportunities.

The CPL-GAS experiment was developed utilizing existing hardware where possible (see Figure 4). The support structure and battery are identical to those used for the STS-3 GAS Flight Verification Payload. The electronics and tape recorder were flown previously on the Atomic Oxygen Monitor GAS experiment flown on STS-8 and STS-11. Unfortunately, the space available for the CPL experiment is constrained due to the volume requirements of the battery and electronics. Nonetheless, a working mini-CPL has been developed that mounts directly to the GAS top plate between the structural support struts. It measures approximately 14" by 14" by 4" high and maintains most of the operating features of the large ground system. Figure 5 shows the mini-CPL experiment both before and after installation of the heaters, wiring, instrumentation, and electronics.

The mini-CPL has two evaporator pumps mounted in parallel, with heaters attached directly to their outer surfaces. A temperature controlled, two-phase reservoir and an isolator are also included in this system. The boxes seen in Figure 5 contain additional electronics for reservoir temperature control and over-temperature limitstats to prevent overheating of the experiment (via heater cutoff). The entire system is mounted on the condenser plate with fiberglass thermal isolators. The condenser plate is

then mounted directly to a CPL unique GAS top plate (supplied by the CPL project). The majority of the mini-CPL is constructed of aluminum, with the exception of the reservoir and isolators, which are stainless steel.

The CPL-GAS experiment is shown fully assembled in Figure 6. The mini-CPL is covered with a multi-layered insulation blanket (MLI) in order to thermally isolate it from the battery and electronics. A thermostatically controlled heater is used on the battery to maintain its temperature above its lower limit of 0°C during potential cold case operations. The electronics box is covered with high emittance Kapton tape to radiatively dissipate its internally generated heat, which is approximately 12 watts. The container was flown without the insulating end cap since the GAS top plate is used as the heat rejection radiator for the experiment. The top plate exterior surface was coated with silver teflon tape.

Since the mini-CPL is a closed system containing ammonia, it is a pressure vessel and therefore subject to special design requirements as required by the NASA safety office. These include design to a 2400 psi pressure for all of the pressure system components. The design pressure level is unique for each system, depending on the pressurant and the predicted maximum system pressure.

Also, a second identical mini-CPL was fabricated and then burst tested in order to prove the design. This unit, which burst at 3700 psi, now serves as a display model of the CPL. It should be pointed out that there is more than one possible method to flight qualify pressure vessels. We chose the ASME Boiler Code as the least expensive of the options available. Reference 3 describes the other alternatives as well as the generic safety requirements for shuttle payloads.

TESTING

The CPL-GAS was subjected to testing which included a vibration test, thermal vacuum tests, and extensive functional tests. A workmanship level vibration test was conducted in order to verify that we didn't have any loose screws and that everything would hang together during the flight. Structural qualification testing was not required since the support structure was previously qualified on earlier GAS flights. Pressure testing was performed

on the mini-CPL as already described.

A thermal vacuum test was performed to insure proper operation of the CPL under extreme temperature conditions and the vacuum environment of space. Figure 7 shows the test setup in the vacuum chamber. The CPL-GAS container was situated upside-down in the chamber to allow for proper operation of the CPL in the gravity environment. A thermally controlled cold plate served as a direct radiative heat sink for the GAS container top plate, which is also the heat sink for the mini-CPL experiment. The GAS container also had to be levelled so that gravity effects on the CPL would be minimized. Cabling from the experiment to a data and control system was routed through the GAS container bottom end plate and insulating end cap.

The chamber temperature profile for the thermal vacuum test is shown in Figure 8. The first part of the test (A) was a cooldown and cold case startup. The electronics were allowed to cool to -2 C and the mini-CPL cooled to -20 C for a cold start check. These levels correspond to the predicted cold case startup conditions. The next portion of the test (B) was a flight mission simulation with the thermal environment (chamber and cold plate temperature) set at -10 C , corresponding to the expected shuttle payload bay temperatures for the earth viewing case. The mission profile included experiment heater cycles of up to 220 watts total (110 watts on each pump) for operating times of up to one hour, followed by a cooldown period lasting approximately 10 hours. These operation times were based on the thermal analysis of the CPL-GAS. The experiment condenser was initially allowed to cool to approximately 5°C , then the experiment heaters were activated. Since the power input exceeds the instantaneous heat rejection capability of the GAS top plate, the condenser temperature increases. When the condenser temperature approaches the CPL operating temperature of 29°C , it can no longer absorb any more heat and the system is shut down and again allowed to cool down. The heater cycle is then repeated after the condenser cools back down to about 5°C . During flight, the cycles are repeated for the total mission time, approximately 120 hours. This portion of the test verified the heatup and cooldown times predicted by the thermal analysis. The last part of the thermal vacuum test was a hot case operational check (C), with the environment set at 30°C . This test was conducted to verify the electronics system

performance under the predicted hot case conditions.

As in many tests, things don't always go as planned. Problems were encountered when one of the experiment heaters wouldn't turn off. Difficulty also occurred during the cold start attempts of the electronics. The test had to be stopped to correct these problems and then restarted. However, further difficulties arose with the operation of the mini-CPL experiment itself, and with the understanding of its capabilities and limitations. The planned operational profile as proposed by the OAO Corporation (reference 4) started out with low power on the pumps (25 watts each), with 25 watt step increases to 100 watts each at the end of 45 minutes. When this power profile was tried, the evaporator pumps deprimed soon after startup and would no longer carry the applied heat load, as evidenced by a sudden rise in their temperature. Several other startup or priming techniques were tried, but none were entirely successful. After two weeks in the thermal vacuum chamber, testing was abandoned due to the high costs of the thermal vacuum facility and schedule conflicts with other experiments.

A low cost functional test setup was then pursued so that the mini-CPL could be further evaluated "at leisure". Reference to Figure 6 reveals the functional test setup. The GAS container was again oriented upside-down, but now the GAS top plate rested on a continuously cooled cold plate that removed the heat from the experiment via conduction. This provided more test time since the top plate cooled down in a couple of hours with this setup as compared to 6-10 hours for the thermal vacuum test setup. Although this setup was not a realistic simulation of the Shuttle environment, it did allow for low cost functional testing in our own laboratory.

The functional testing started with little more success than the thermal vacuum testing. The reservoir did not have enough heater power to maintain the system temperature during cold startup, thus leading to flow instability in the loop. Additional heaters were added to reduce the transient thermal effects on the reservoir temperature. Although the reservoir was indeed a problem, unfortunately it wasn't the only problem. The mini-CPL still encountered evaporator pump deprime during startup in many instances. Comparison with the larger ground system showed that the lower power levels

used on the mini-CPL had not been tried on the larger unit. Furthermore, miniaturization of the system had increased thermal "crosstalk" between the various components of the mini-CPL. Mr. John Ripple of NASA/GSFC finally pinpointed the solution to the startup problem. He suggested that the fluid flow rate was very low at low power since the pumps were designed to operate at power levels of up to 700 watts each. The thermal and power limitations of the GAS system forced the 110 watt per pump limit on the mini-CPL. Since the flow rate was so low, heat was leaking down the inlet tubes and pre-heating the fluid prior to entry to the evaporator pumps. This resulted in vapor flow into the pumps instead of liquid, thus causing the pumps to deprime.

Mr. Ripple's solution was to increase the amount of heater power to the evaporator pumps at startup rather than decrease it, as conventional wisdom might suggest. The increase in power resulted in an increase in flow rate, thus reducing the amount of liquid pre-heating. In other words, it worked! This new power profile was called the "100 watt zapp" (figure 9). Rather than proceeding through a gradual warm-up period, the evaporator pumps are given their maximum power level immediately. Additional functional testing showed that this higher power level can be maintained indefinitely, provided that adequate cooling of the condenser can be maintained. Lower power operation of the mini-CPL would require redesign and isolation of the fluid inlet tubes on the evaporator pumps.

The functional testing continued for a total run time of approximately 8 weeks. Additional power profiles were developed and flight simulations were conducted. The value of testing cannot be overstated, especially in experiments dealing with new systems and technology development.

CPL-GAS FLIGHT

The mission profile for the CPL was established based on the shuttle bay-Earth thermal environment, since this is the primary orientation for most shuttle missions. Each power cycle consists of an experiment heater on period followed by a cooldown period, as previously described in the testing section. The first power cycle (figure 9) was initiated four hours after payload

activation. The delay was built in to allow the battery to heat up, if its temperature was below 0°C, since its capacity significantly degrades below 0°C. Nine hour cooldown times were allotted between the cycles, which provided ample thermal margin above the predicted nominal requirement. The majority of the power cycles were the 100 watt zapp, with other types of cycles interspersed, for a total of 13 cycles in a 120 hour period. The other power cycles included heat sharing (power in one evaporator pump only), power stepdown profiles, low power steady-state, and induced deprime (intentional deprime of one of the pumps with inlet heaters).

Experiment sequencing was accomplished with the use of an electronic clock and a pre-programmed hardwired memory. The memory was a bi-polar fusible link 8K ROM built by the Harris Corporation to military specifications. Experiment data was written onto a lockhead tape recorder that is contained in the larger electronics box. The data includes thermistor readings, experiment heater power levels, battery and calibration voltages, and command status. The data was taken at approximately one minute intervals. The data collection system was built by TS Infosystems Incorporated, and the command sequencing and data storage system was built by ITE Incorporated.

The first flight of the CPL-GAS experiment occurred in April 1985 on STS-51D. Unfortunately, the GAS batteries that actuate the relay to activate the experiment failed, so the CPL could not be turned on during flight. The failure was apparently due to a bad batch of batteries that failed under a combination of vacuum and cold temperatures, even though they had passed qualification testing. The GAS project has solved the problem by enclosing the batteries in a hermetically sealed box; a fix that will be incorporated in future GAS and SPARTAN flights. Fortunately, a reflight opportunity was available on STS-51G in June, 1985. On this flight the GAS relays operated satisfactorily, and we had a very successful experiment. The mini-CPL operated for the planned 120 hours and 13 power cycles were run. The experiment worked even better than expected; all of the power profiles worked, even the low power cycle that wouldn't work on the ground. As of this writing, the data is being reduced and will be made available in later reports.

CONCLUSIONS

One of the important aspects of any project deals with how it is presented--public relations. It was suggested by Mr. John Krehbiel of GSFC that a logo should be designed for our two-phase technology development program. A logo design contest was sponsored within the Thermal Engineering branch at GSFC, and the winning entry is shown in Figure 10. This entry was submitted by Mathew Jarrell, a high school student whose father, Bill Jarrell, works for the NASA/GSFC thermal branch. Decals of the logo will be made and distributed to all interested parties.

Future plans for the two-phase flow heat transfer project include four more shuttle flight experiments over the next three years. The next flight in December, 1985 will be a reflight of the mini-CPL on the Hitchhiker-G carrier system. It will again be located in a GAS container, but power and real time data and command capability will be available from the STS. This will allow for higher experiment power levels (up to 800 watts) and control of the experiment during the flight. A specially designed 140 pound GAS container top plate will be used to absorb the large power dissipations from the CPL. The new top plate as well as the GAS container will not be insulated in order to enhance the heat rejection capability of the GAS system. The GAS container will be painted white, while the top plate will be coated with silver teflon tape.

The mission profile will resemble that used for the CPL-GAS, with short 30 minute power cycles followed by longer cooldown times. A larger, full-scale version of the CPL will be flown in December, 1986. It will have a large 100 square foot radiator that will allow for continuous operation of the experiment at a 1000 watt power level.

A Pumped Two-Phase system (PTP) is also being developed that uses a small mechanical pump instead of capillary pumps to transport the two-phase working fluid. A Hitchhiker flight is planned for July, 1986 for the first flight demonstration of a PTP system. It will be followed by a larger full-scale flight experiment utilizing the 100 square foot radiator in December, 1987.

What is the purpose of all this development work on two-phase systems? Figure 11 depicts a proposed design of the Space Station. The Space Station requires high capacity heat transport systems that can carry tens of kilowatts of heat over distances of tens of meters or more. Present single phase loops cannot do the job unless large, massive, and costly systems are built. Two-phase systems offer the potential of doing the job much more efficiently at a fraction of the cost. This technology will undoubtedly be applied in other areas as well, after it's development matures and more people become aware of it's potential.

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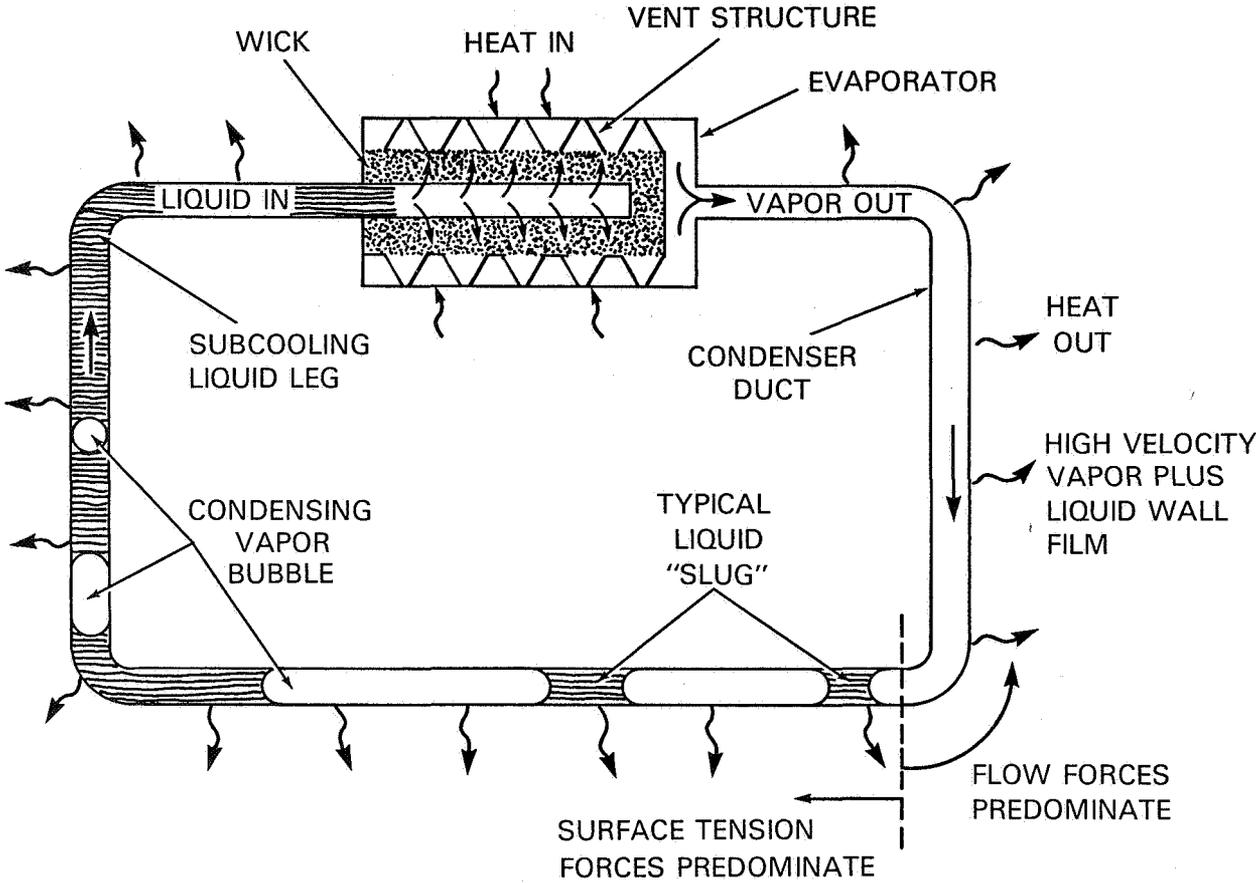


Figure 1. Model of a Capillary Pumped Heat Transfer Loop

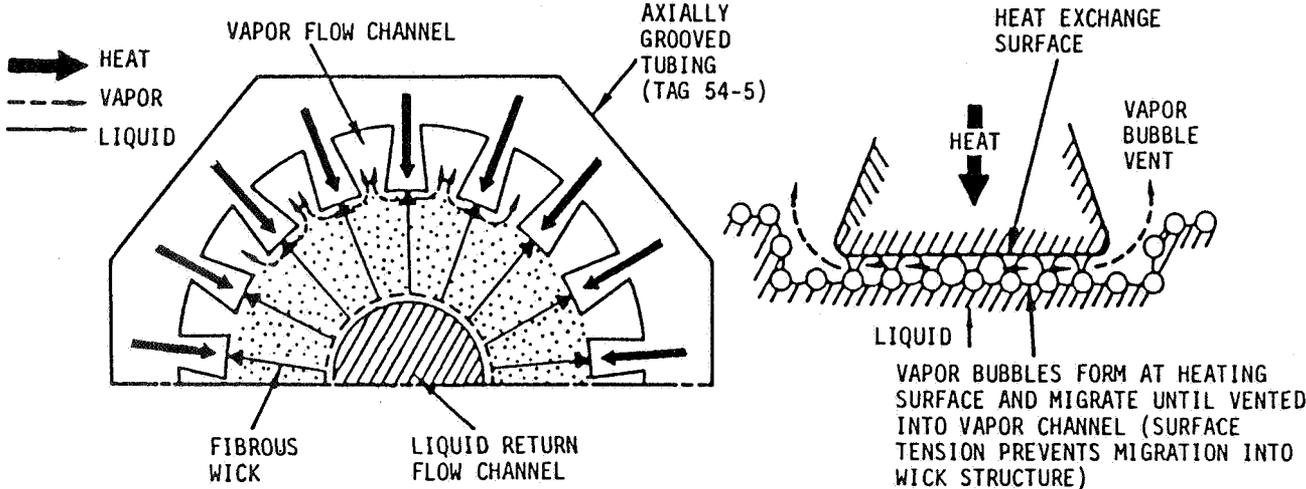


Figure 2. Heat and Fluid Transport in CPL Evaporator

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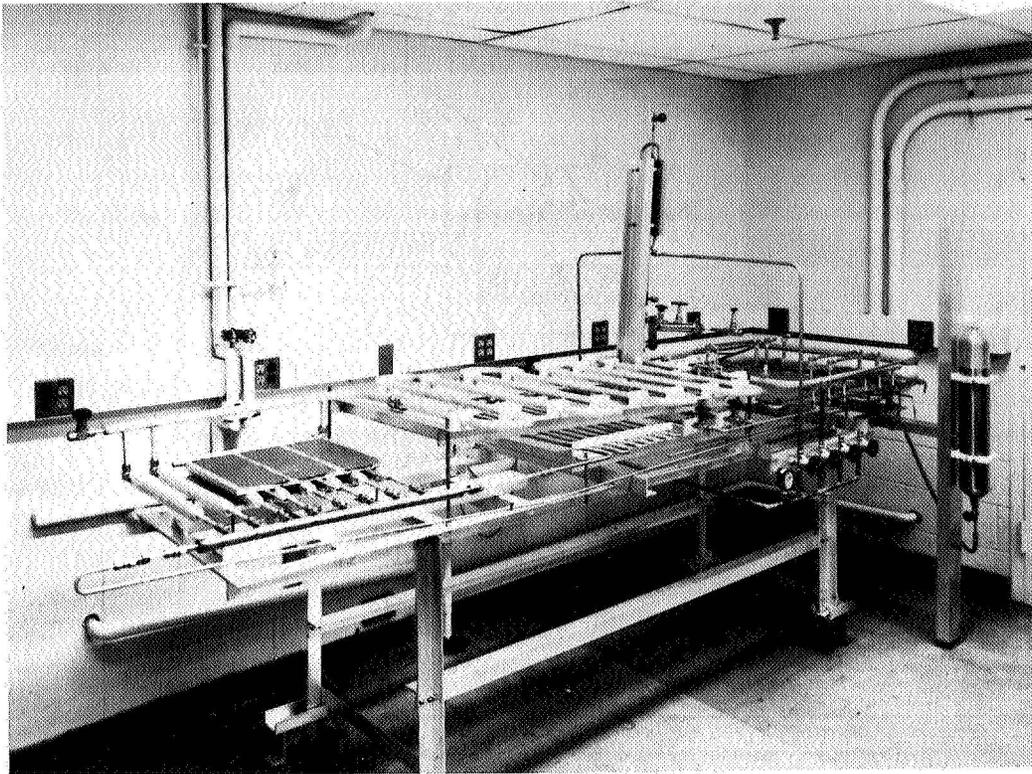


Figure 3

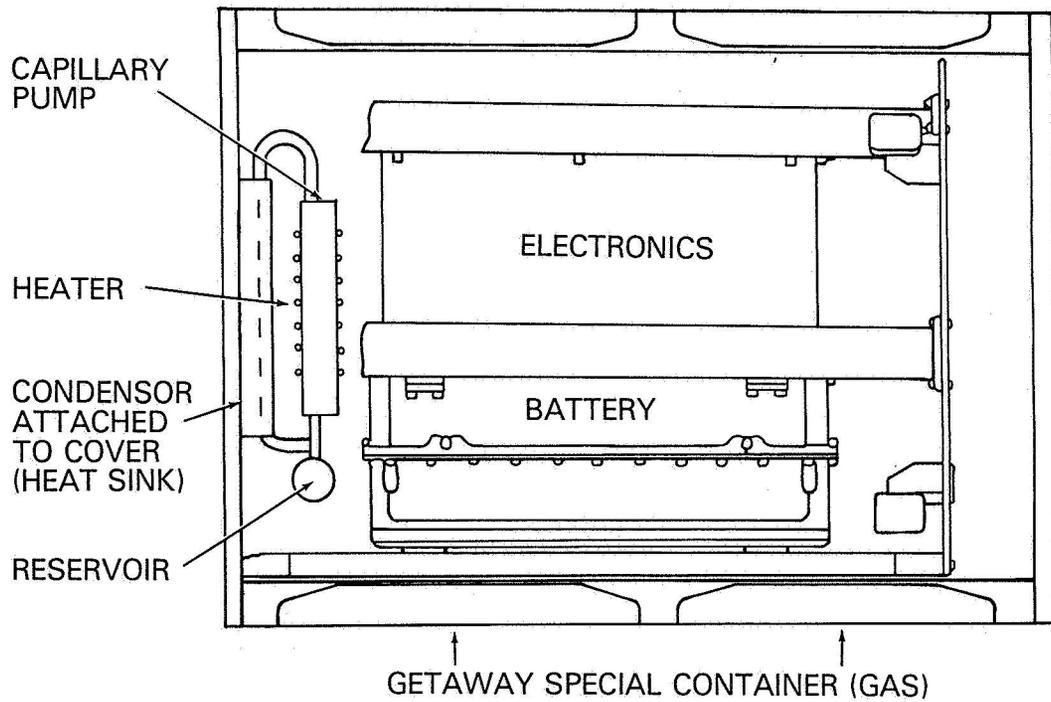


Figure 4. Capillary Pump Priming Experiment in GAS Container

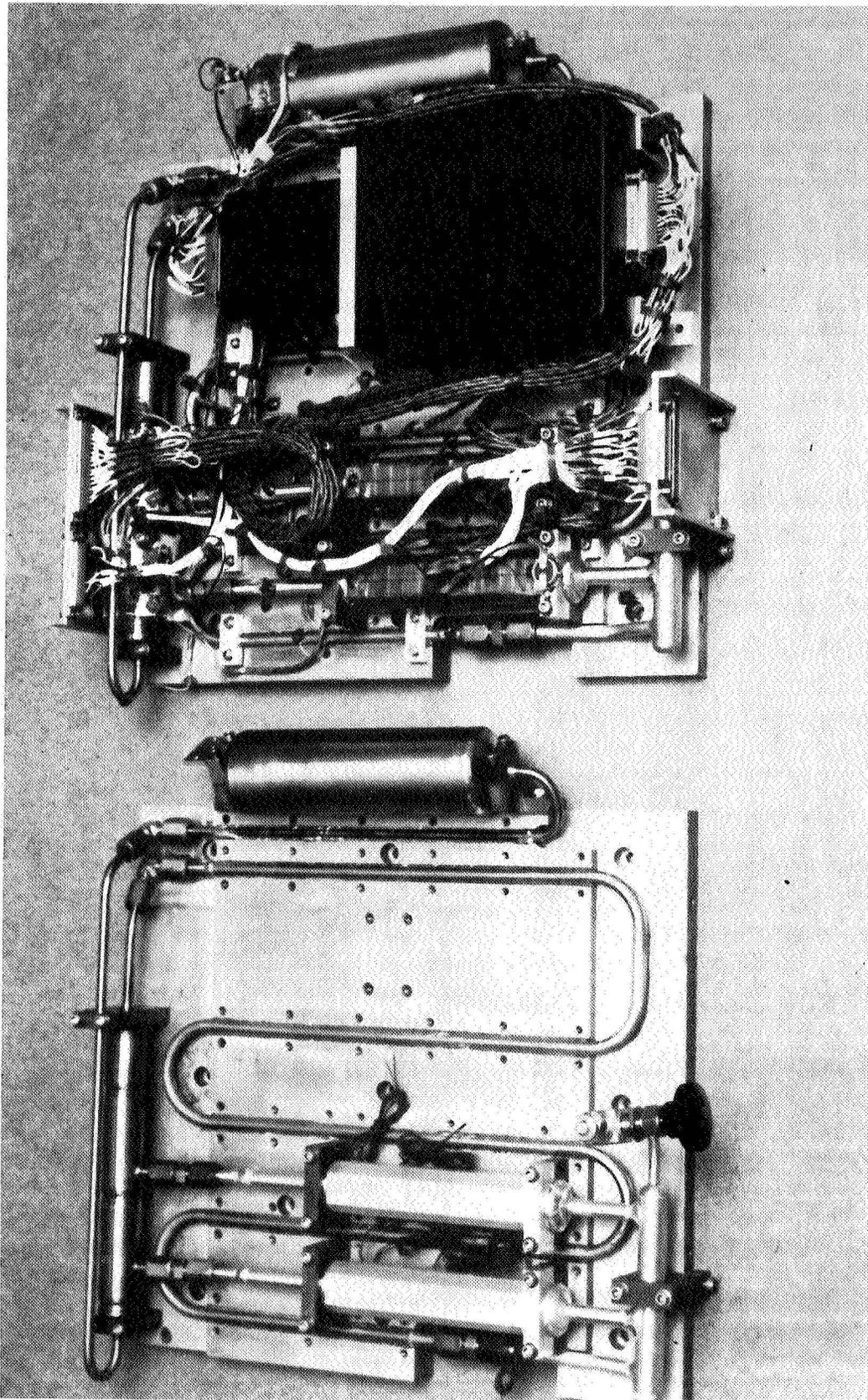


Figure 5

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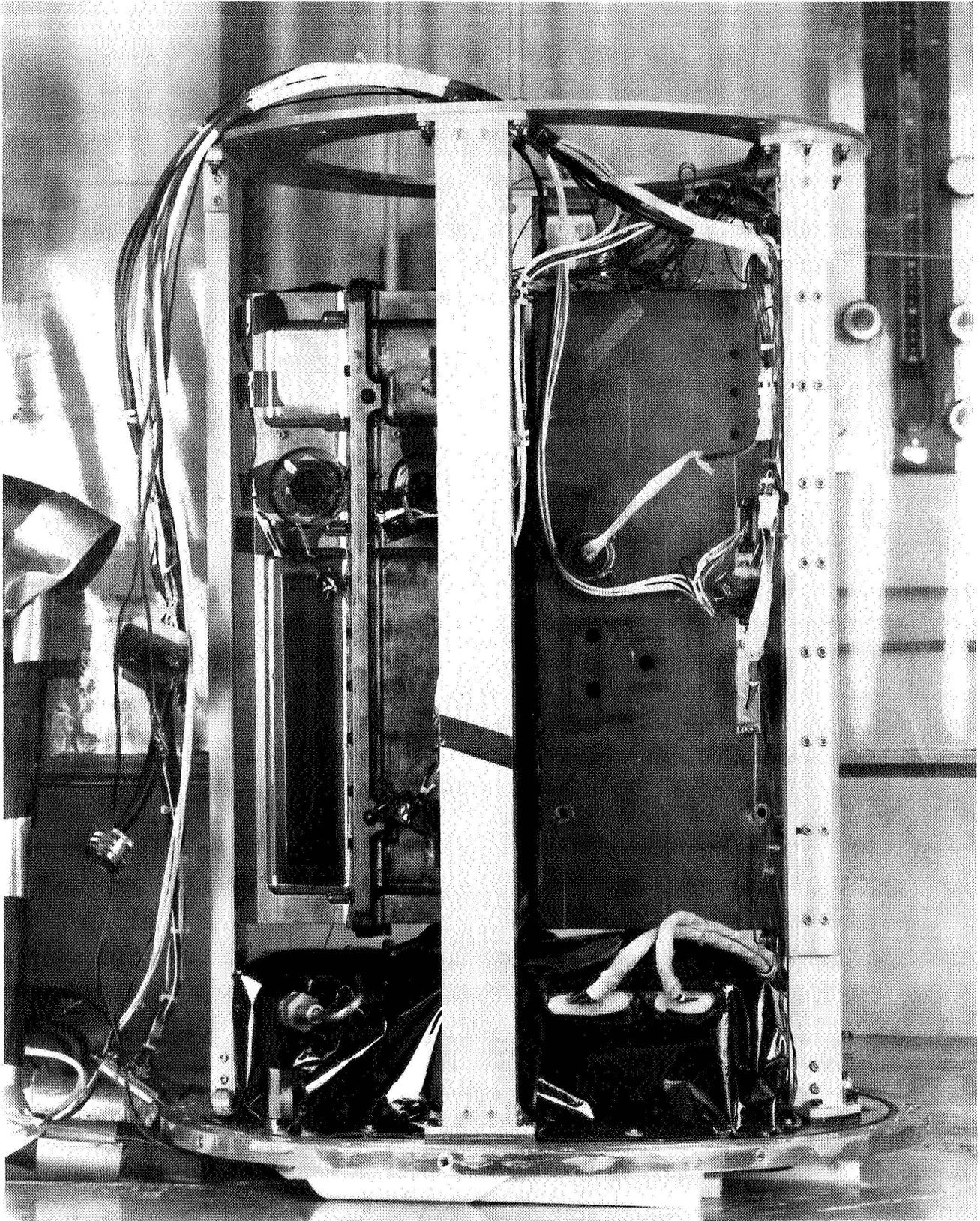


Figure 6

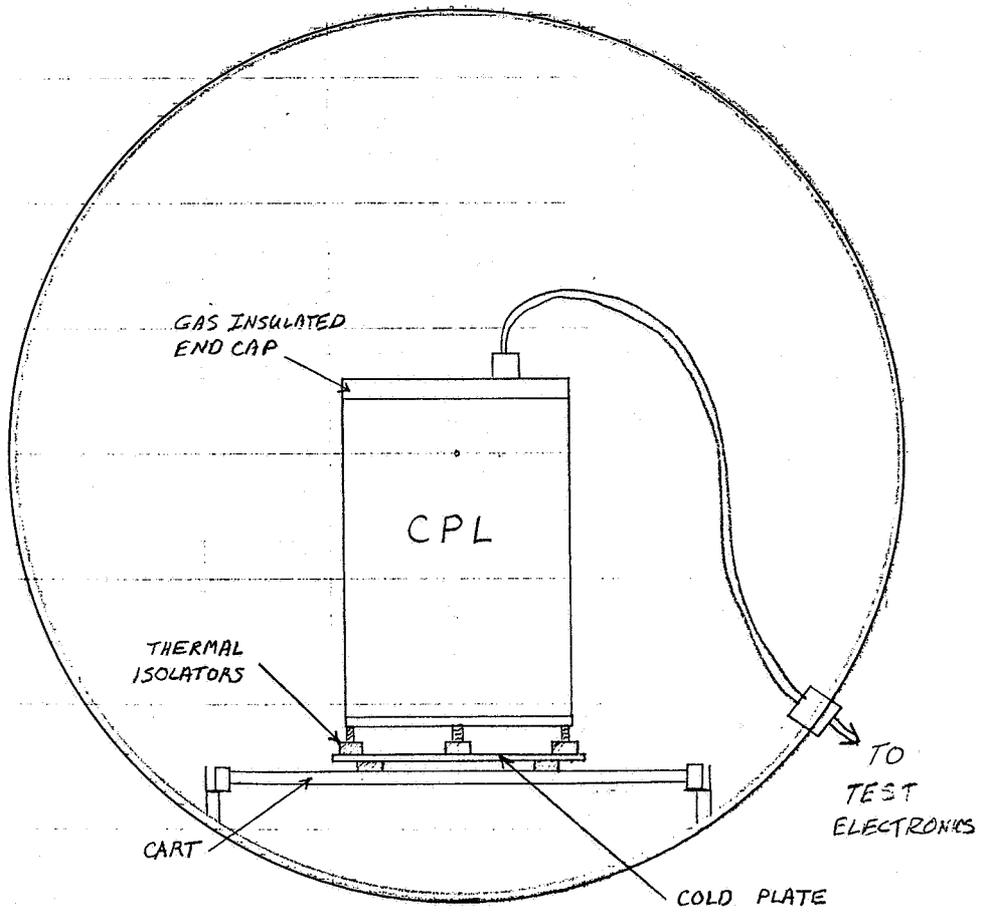


Figure 7. TV Test Set-up

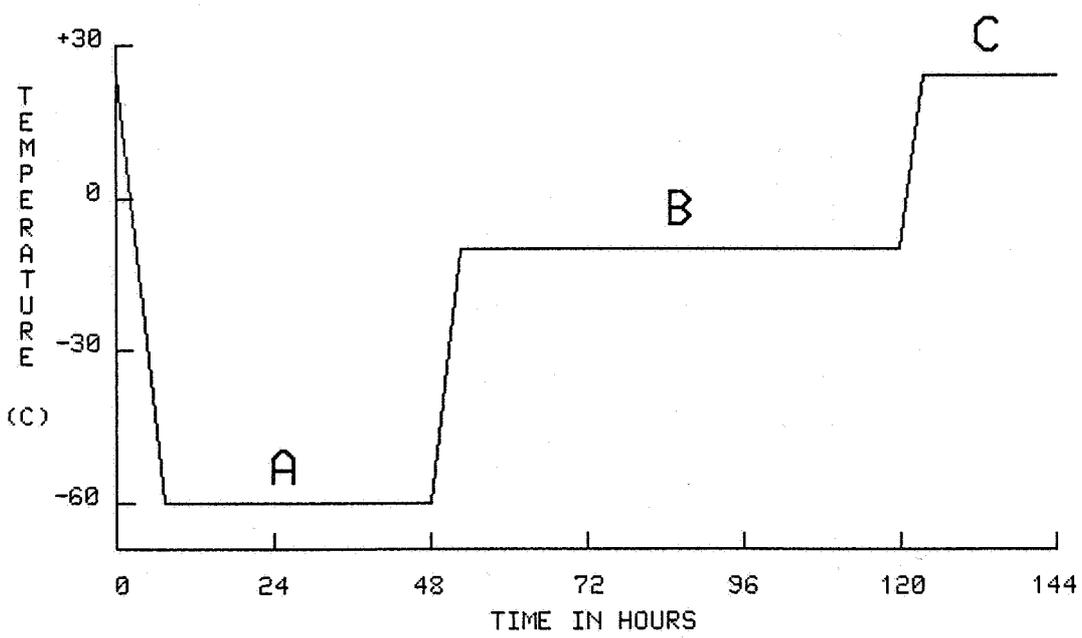


Figure 8. CPL TV and Mission Simulation

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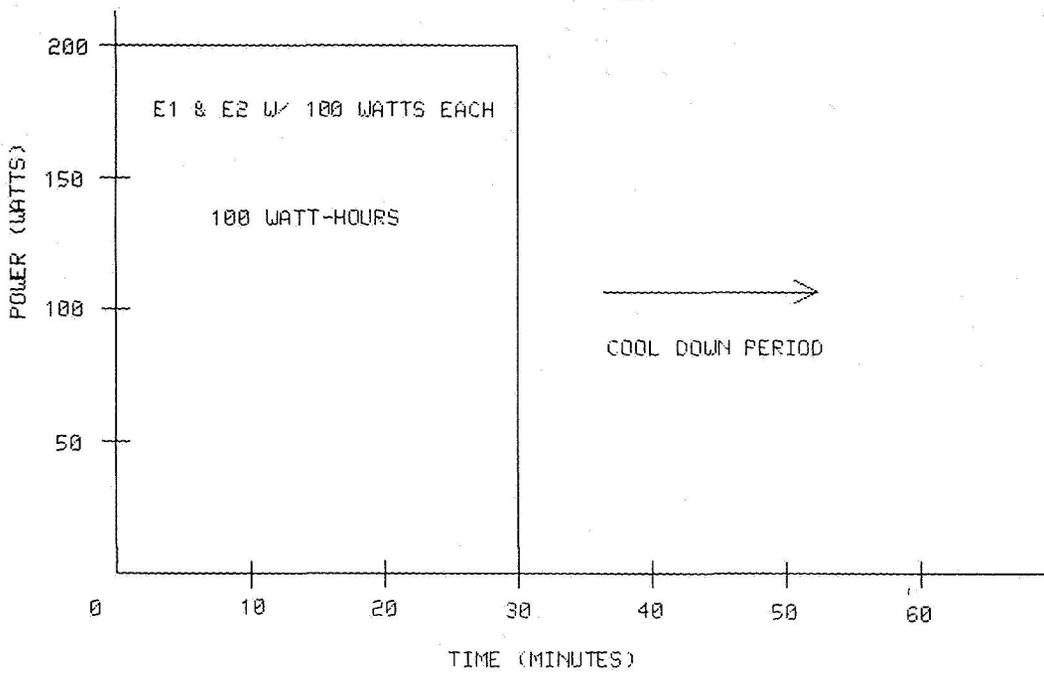


Figure 9. Cycle A (100 watt zapp)

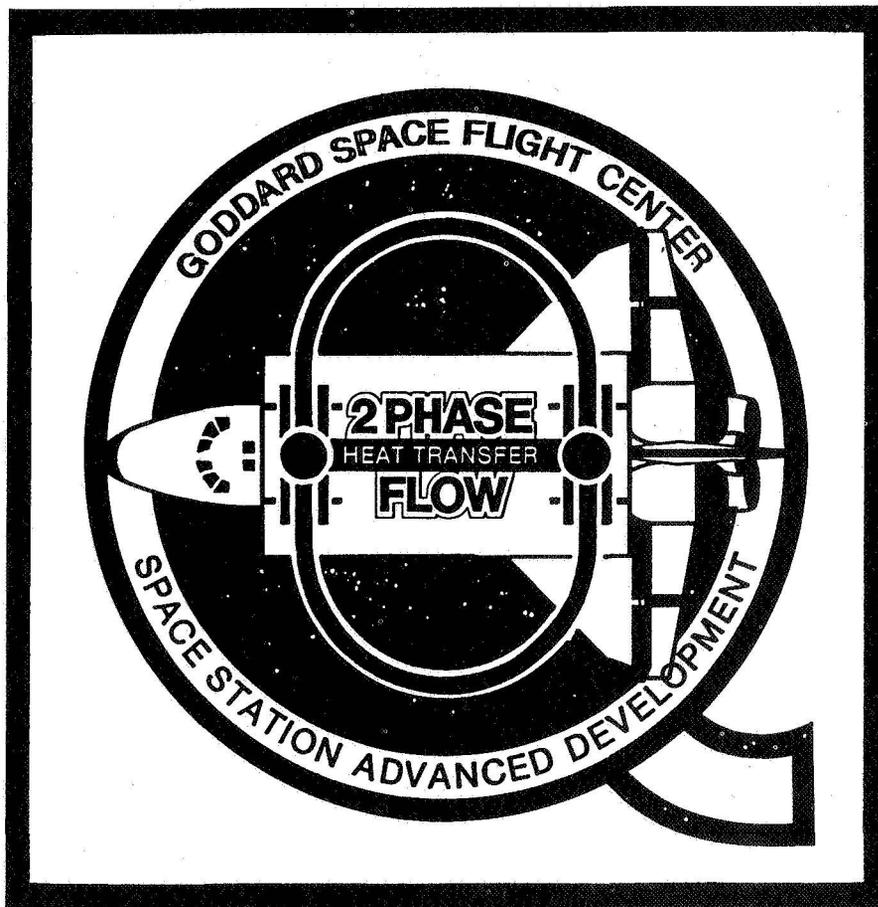


Figure 10

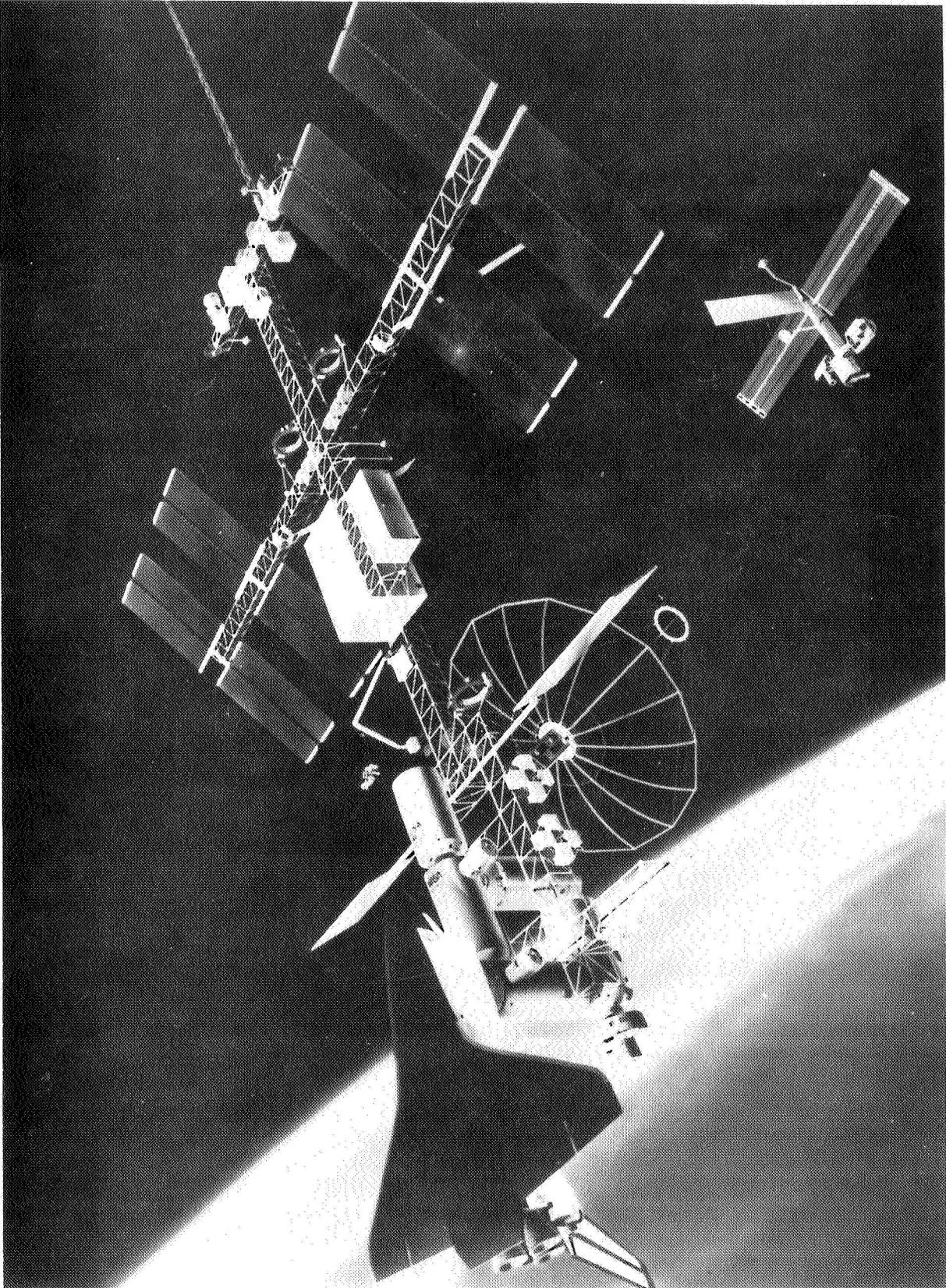


Figure 11

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LESSONS LEARNT FROM 11 GAS PAYLOADS
FLOWN ON 5 SHUTTLE MISSIONS

Dr. H. Stolze and Dr. P. Vits
MBB-ERNO
Bremen, FRG

592607
MT603998

Second GAS Experimenters Symposium
NASA/Goddard Space Flight Center
Greenbelt, Maryland
October 8-9, 1985

MBB-ERNO'S INVOLVEMENT IN THE GAS PROGRAM

- o INDUSTRIAL CONTRACTOR FOR THE MAUS-PROJECT (MATERIAL SCIENCE EXPERIMENTS)
 - PHASE A/B STUDY (PARALLEL STUDIES)
 - PHASE C/D PROJECT
 - 10 UNITS FOR 5 FLIGHTS EACH
 - 10 PAYLOADS FLOWN ON 5 MISSIONS (STS 5, 7, 11, 51G)

- o RELATED STUDY PERFORMANCES
 - MAUS CLUSTER
 - MAUS ON HITCHHIKER

- o COMPANY FUNDED GAS-PAYLOAD
 - EMTE I SURFACE TENSION TANK EXPERIMENT (STS 51G)
 - EMTE II REFLIGHT WITH NEW EXPERIMENT-PROFILE
 - BIOLOGY EXPERIMENT QUALIFICATION (BIO-MAUS)

- o COMMERCIAL GAS PAYLOAD SUPPORT SYSTEM PASS
 - HARDWARE LEASING
 - PAYLOAD INTEGRATION AND MISSION SUPPORT

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LESSONS LEARNT

- o SELECTED HINTS DERIVED FROM 5 YEARS EXPERIENCE IN:
 - EXPERIMENT HARDWARE DESIGN
 - HARDWARE QUALIFICATION
 - SAFETY PROVISIONS
 - EXPERIMENT TESTING
 - PAYLOAD INTEGRATION
 - FLIGHT PREPARATION
 - BATTERY TRANSPORT
 - POST-MISSION CHECK-OUT

- o THIS IS NOT A PRESENTATION OF A COMPLETE DESIGN PHILOSOPHY,
BUT ONLY SELECTED HINTS FOR FUTURE GAS-USERS

- o BASIC RULE
„MAKE IT SIMPLE AND OVERSIZED“

OVERVIEW OF MAUS-SYSTEMS

(IN ORDER TO UNDERSTAND THE LESSONS LEARNT)

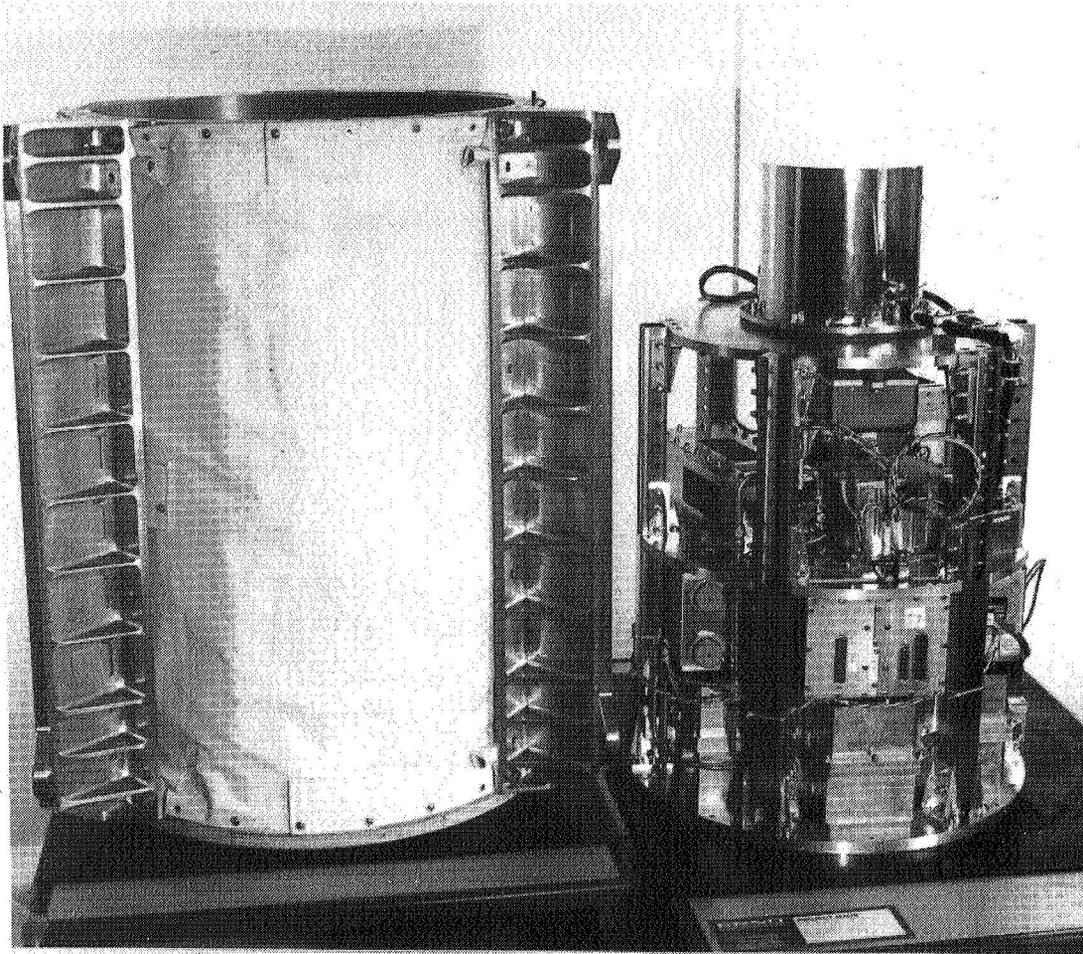
- o MAUS-CONCEPT VIEWGRAPH 1

- o SUBSYSTEMS VIEWGRAPH 2

- o PAYLOADS
 - MAUS DG-XXX VIEWGRAPH 3
 - EMTE I VIEWGRAPH 4

- o MAUS-MISSIONS
 - GAS VIEWGRAPH 5
 - SPAS VIEWGRAPH 6
 - OSTA II VIEWGRAPH 7
 - D1/NAVEX VIEWGRAPH 8

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PAYLOAD

- o VOLUME 140 l
- o ENERGY 2 kWh
- o WEIGHT 91 kg

EXPERIMENT

- o VOLUME 70 l
- o ENERGY 1,8 kWh
- o WEIGHT 20 kg

SERVICE MODULE

- o EXPERIMENT CONTROL
- o DATA ACQUISITION
- o DATA STORAGE
- o SIGNAL CONDITIONING
- o POWER CONDITIONING
- o POWER SUPPLY
 - ELECTRONICS
 - EXPERIMENT

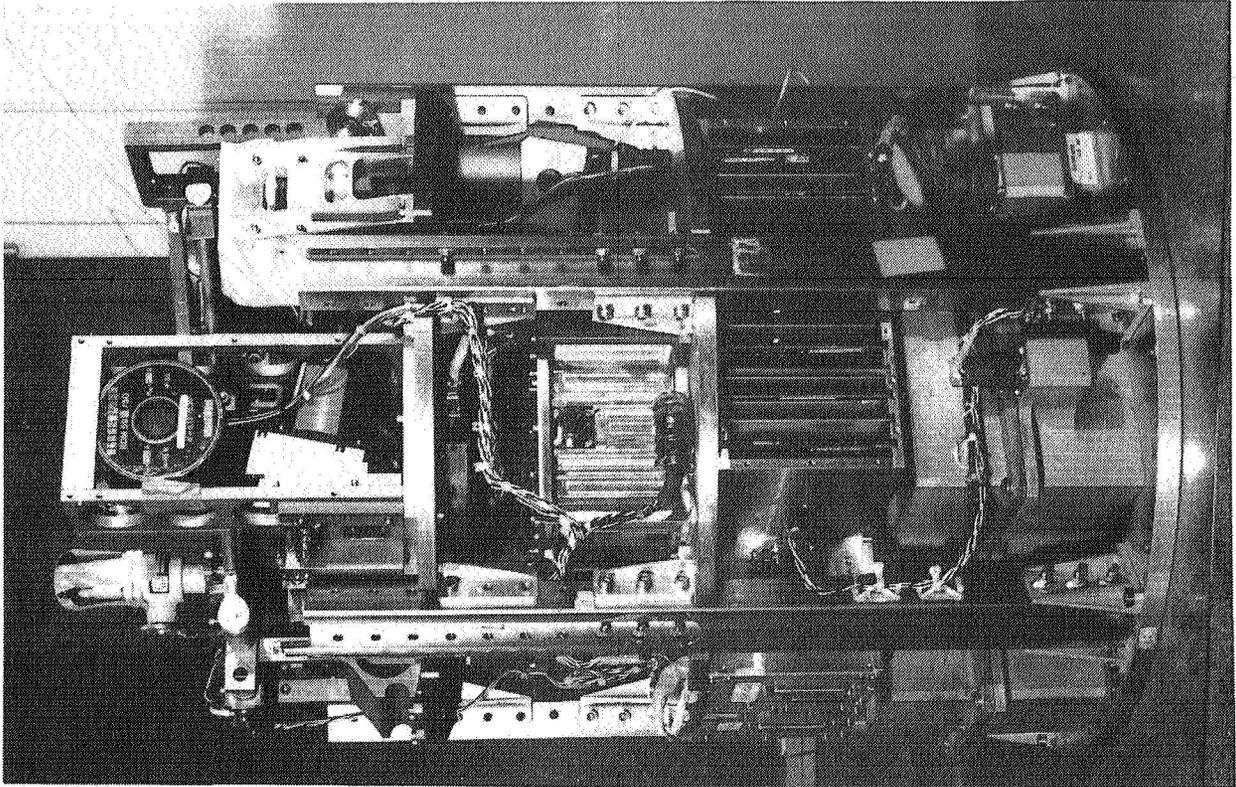
- o GSE-INTERFACE
- o EXPERIMENT MOUNTING STRUCTURE
- o HOUSEKEEPING SYSTEM
 - PRESSURE
 - CONTAINER
 - BATTERIES
 - TEMPERATURE
 - BATTERY - VOLTAGES
 - ACCELERATION (3-AXIS)

AND

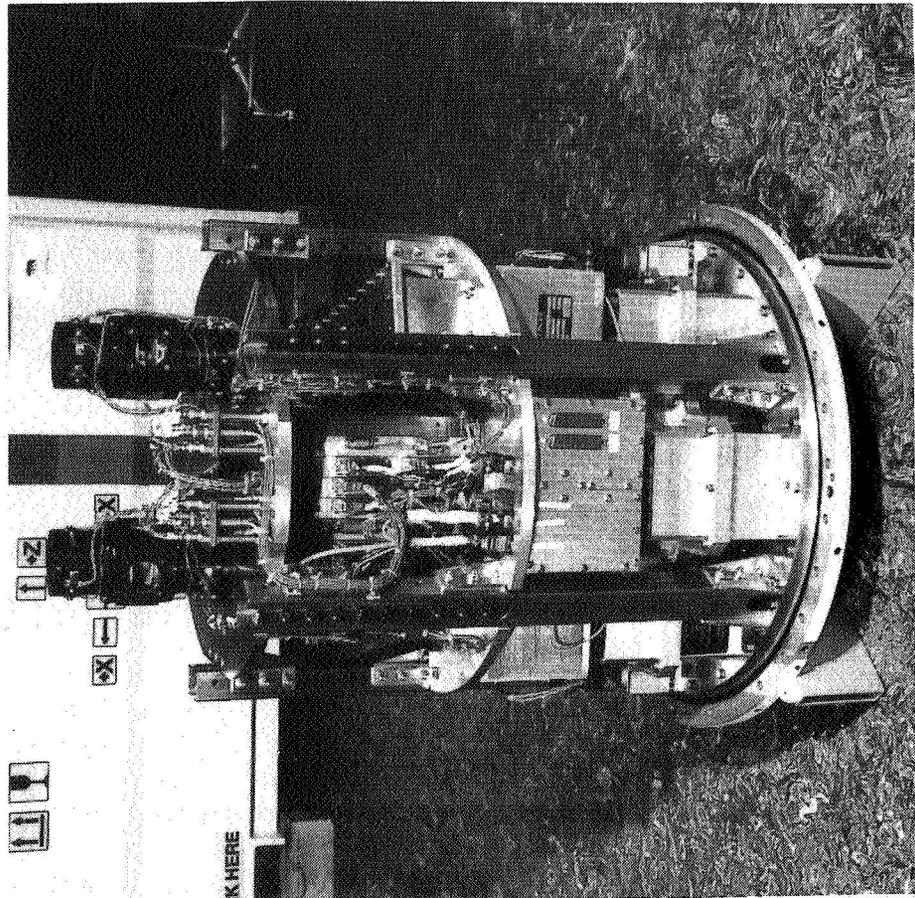
CONTAINER, ON/OFF-INTERFACE, ADAPTER BRACKET, MGSE, EGSE

MAUS-Concept and Subsystem Definition

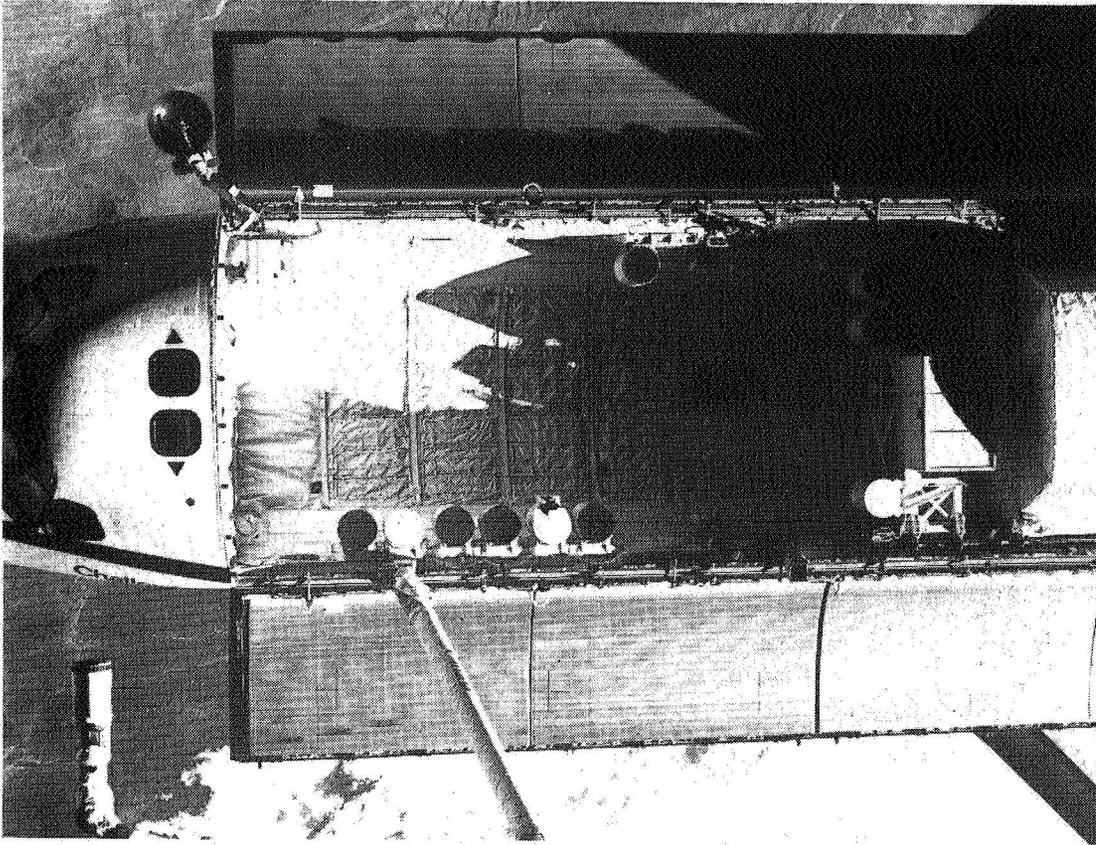
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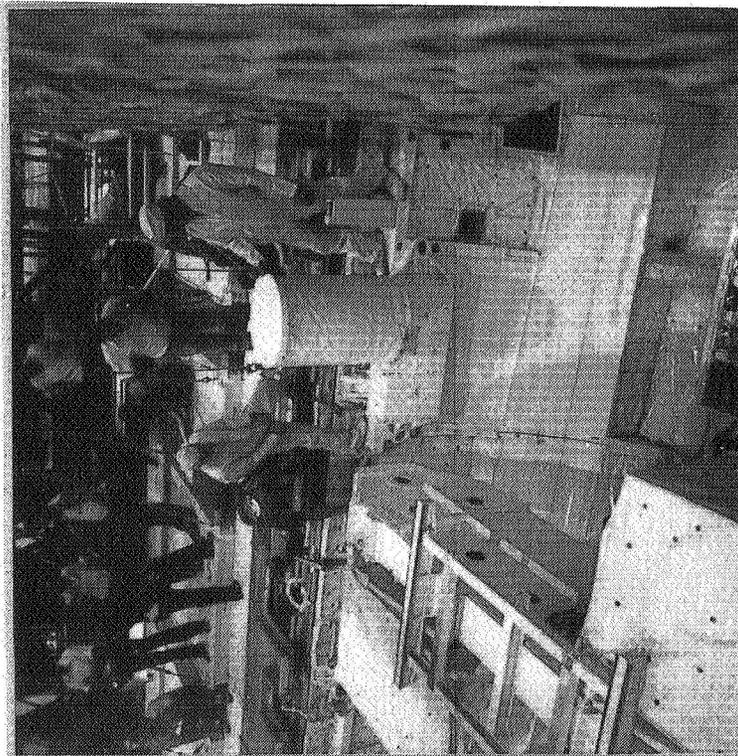
MAUS Internal Payloads



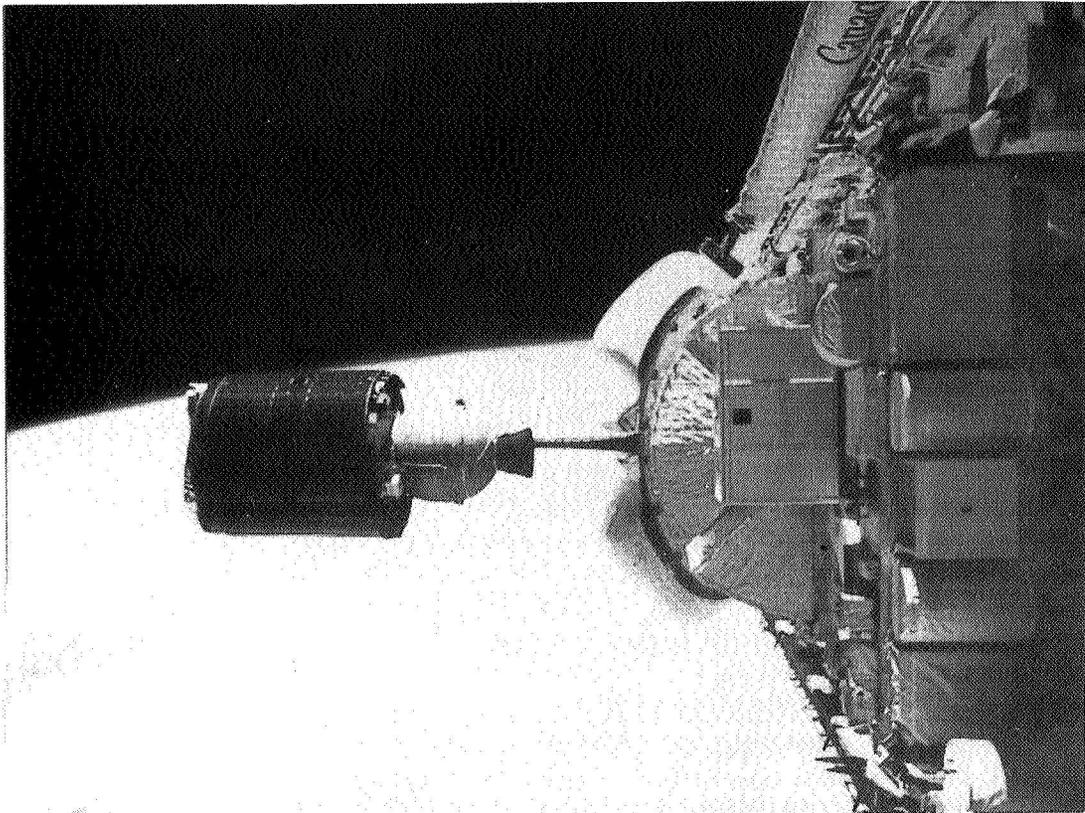
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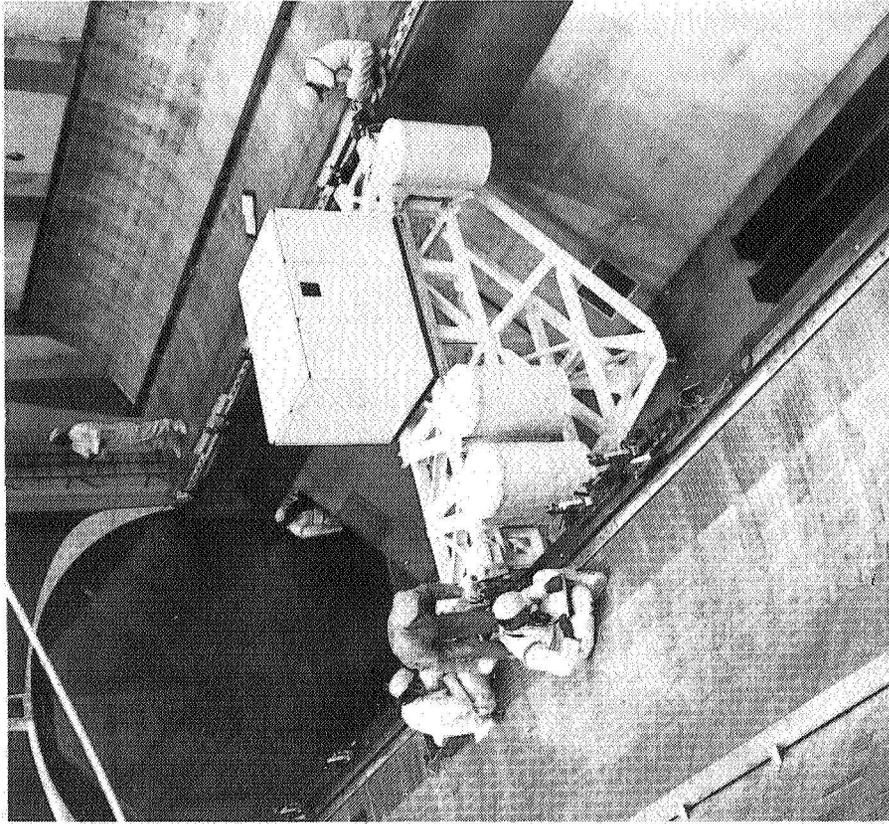
GAS-Configuration STS-7



GAS-Configuration STS-5

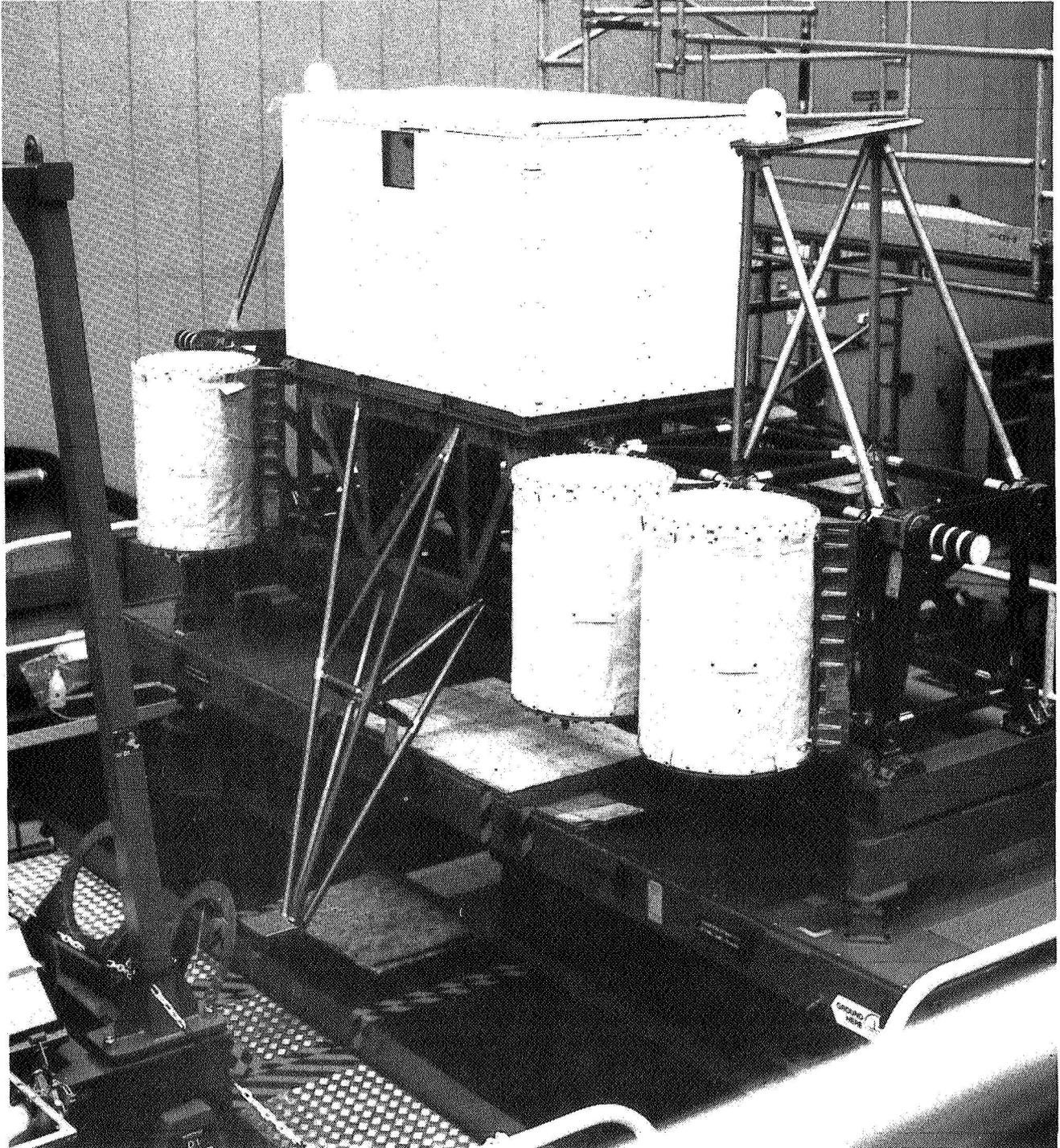


MAUS-Payloads on SPAS-01



MAUS-Payloads on OSTA 2

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NAVEX/D1-Configuration

LESSONS LEARNT (I)

1. ENSURE, YOUR EXPERIMENT RUNS DURING SLEEPING PERIOD OF CREW TO MINIMIZE DISTURBANCES CAUSED BY CREW MOVEMENTS AND ORBITER OPERATIONS
2. TEST THE INFLUENCE OF YOUR OWN EXPERIMENT ON THE μ -G ENVIRONMENT (CAMERA WINDOWS, SHUTTER, MOTORS, ETC.)
3. USE A 3-AXIS ACCELEROMETER AND RECORD THE OUTPUT TO PROVE YOUR EXPERIMENT IS TURNED ON IN SPACE AND NOT ON THE GROUND, AND TO VERIFY THE QUALITY OF THE μ -G ENVIRONMENT
4. SELECT OFF-THE-SHELF HARDWARE, BUT VERIFY AT LEAST
 - VIBRATION (QUALIFICATION LEVEL)
 - TEMPERATURE RANGE
5. PREPARE A LOGICAL QUALIFICATION PHILOSOPHY
 - PROTOTYPE SYSTEM QUALIFICATION
 - COMPONENT QUALIFICATION
 - INTEGRATED SYSTEM QUALIFICATION

LESSONS LEARNT (II)

6. CONSIDER THE CONSEQUENCES OF PRESSURE CHANGES DURING LEAKAGE TESTS AND/OR PURGING (FOAM, TIGHT BOXES, INTEGRATED CIRCUITS, RELAYS, ETC.)
7. SILVER/ZINC BATTERY EXPERIENCE
 - VERIFY ALL BATTERY SAFETY PROVISIONS (PRESSURE SWITCH OFF, UNDER VOLTAGE SWITCH OFF, FUSES, PRESSURE RELIEF VALVE FUNCTION, LEAKAGE)
 - TRANSPORT THE FILLED (HOT) BATTERY ONLY BY „CARGO ONLY AIRPLANES“ OR BY „COOLED TRUCK“ IN WOODEN BOXES
 - BE PREPARED FOR EXCHANGE OF WEAK CELLS
 - TEST YOUR BATTERY PACK FOR LEAKAGE IN ALL 3 AXES BEFORE INTEGRATION
 - SEAL THE POLE-BOLTS WITH INSULATING PAINT
8. TAKE INTO ACCOUNT THE EXTREME CAPACITY LOSS OF NiCd-BATTERIES BEFORE LAUNCH UNDER WORST CASE CONDITIONS (TIME, TEMPERATURE, ALL DATA TOLERANCES)

LESSONS LEARNT (III)

9. EMERGENCY SWITCH-OFF-FUNCTIONS (TEMP.-INCREASE, PRESSURE LOSS, BATTERY UNDER VOLTAGE, PRESSURE RISE, ETC.)
 - DETECT SEVERAL „OUT OF LIMITS“ BEFORE INTRODUCING ANY ACTION IN ORDER TO AVOID EXP. SHUT-OFF BY VOLTAGE PEAKS
 - THINK ABOUT PARTIAL SWITCH-OFF FUNCTIONS (E.G. WITHIN A PAYLOAD CONSISTING OF SEVERAL FURNACES)
 - THINK ABOUT GIVING YOUR μ -PROCESSOR THE POSSIBILITY TO
 - JUMP INTO SUBPROGRAMS DUE TO SPECIAL EVENTS (MODIFIED EXP.-PROFILE)
 - JUMP INTO DEDICATED SWITCH-OFF PROCEDURES (E.G. MAGNETIC COILS)
 - THINK ABOUT DATA REGISTRATION BEFORE ABNORMAL SWITCH-OFF FOR FAILURE DIAGNOSIS AFTER MISSION

10. VERIFY SENSOR CALIBRATION AT KSC BEFORE AND AFTER MISSION IN ORDER TO ELIMINATE TRANSPORT EFFECTS

LESSONS LEARNT (IV)

11. RUN COMPLETE EXPERIMENT BEFORE AND AFTER MISSION AT KSC AS REFERENCE EXPERIMENT FOR COMPARISON AND FOR DETECTION OF ANOMALIES
12. RUN REFERENCE EXPERIMENTS WITH BATTERY POWER SUPPLY AND NOT WITH EXTERNAL POWER SUPPLY
13. THINK ABOUT PROGRAMMING YOUR μ -PROCESSOR TO INCLUDE AUTOMATIC TEST RUNS AND SELF-CONTAINED FAILURE DIAGNOSTIC CAPABILITY
14. CONSIDER THE CONSEQUENCES THAT EVERY COMPONENT OR PART WITHIN YOUR EXPERIMENT MIGHT FAIL
15. USE SIMPLE TEMPERATURE SENSING STRIPS (COLOUR CHANGING MATERIAL) FOR
 - TEMPERATURE CONTROL OF SELECTED COMPONENTS
 - ENVIRONMENTAL CONTROL (MISSION DEPENDENT ENVIRONMENT)
 - QUICK-LOOK DIAGNOSIS AFTER FLIGHT

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G-38, G-39 AND G-40
ART IN SPACE — A DIVERGENT EXPLORATION

✓
Joseph W. McShane
Sculptor/Payload Manager
Marshall-McShane, Prescott, Arizona

MH 808304

A report on: (1) The results of GAS Arts-Science payload G-38, processed in orbit on board the Space Shuttle Challenger during mission 41-G STS 17, October 5 to 13, 1984. (2) An overview of future GAS payloads G-39 and G-40.

"The most beautiful experience we can have is the mysterious.

It is the fundamental emotion which stands at the cradle
of true art and true science." Albert Einstein

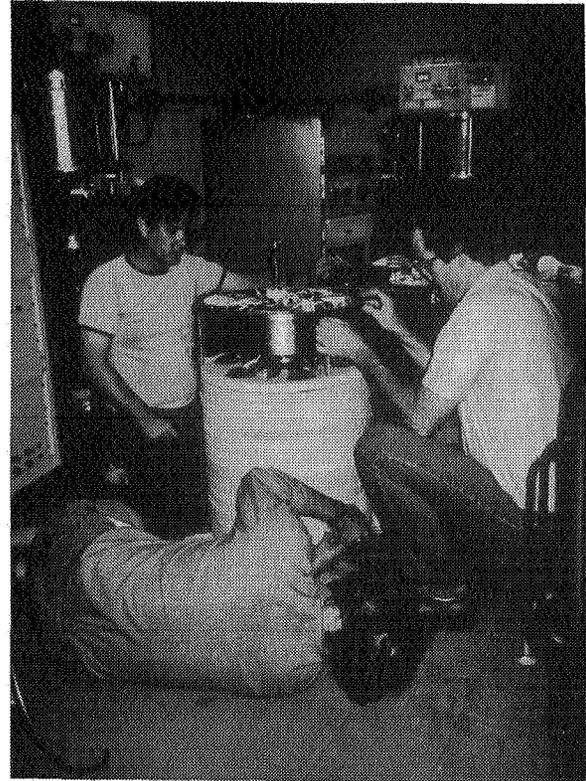
Payload G-38 generates a new definition of space art, as art created in and about space. The artist can now use the tools and processes of space technology to participate in and comment on man's adventure in space. G-38 was created as a unified Arts-Science payload that simultaneously explored the process of vapor deposition in the vacuum and weightlessness of the shuttle environment and created a series of space sculptures utilizing this process; seeking to experience the mysteries of space.*

The methods and results of G-38 supply useful data for simplified coatings of large antennae, heat shields, solar collectors and optical mirrors in space; where size is not limited to the confines of a vacuum chamber.

* For more specific information on vapor deposition and the operation of payload G-38, please refer to "G-38, 39 and 40/An Artist's Exploration of Space", 1984 Get Away Special Experimenter's Symposium Handbook, NASA C.P. 2324.

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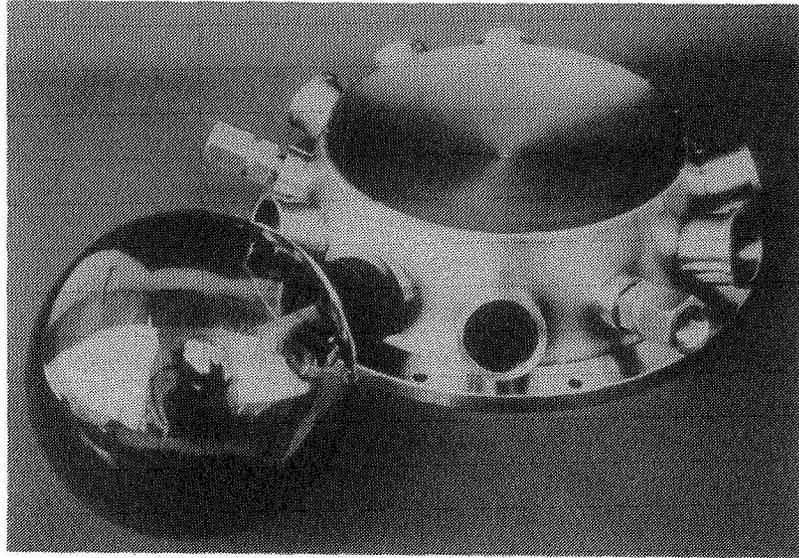
The final assembly of G-38, showing stainless steel control housing and expanded polyimide foam support structure encasing glass spheres.

Two separate and distinctly different thin film deposition processes were used in G-38. First, a sputter deposition process was performed on the interiors of five 500ml glass spheres. By accelerating positively charged argon ions into the surface of a negatively charged metal target mounted in the center of each sphere, target molecules were ejected by the impact of the argon ions and formed a coating on the inside of each sphere over a period of hours.

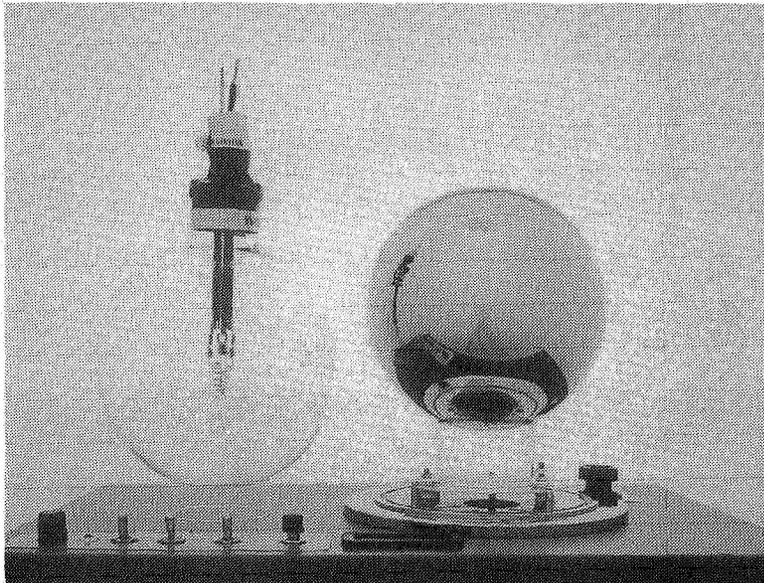
The purpose of the experiment was to test the sputter deposition process in space and to create five subtle spherical sculptures with metallic coatings of gold, silver, platinum and chrome. These five experiments all functioned in the expected manner but did not create coatings as dense as anticipated. This was traced to apparent arching in the vacuum manifold during the ionization process, which limited the charge being transmitted to the target. Thus, sculptures that were intended to have metal coatings applied in space ranging from opaque to semitransparent, returned with all semitransparent coatings instead; an unexpected but not displeasing result.

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Partially coated sputter
deposition sphere with target,
mounted on the vacuum manifold
of payload G-38.



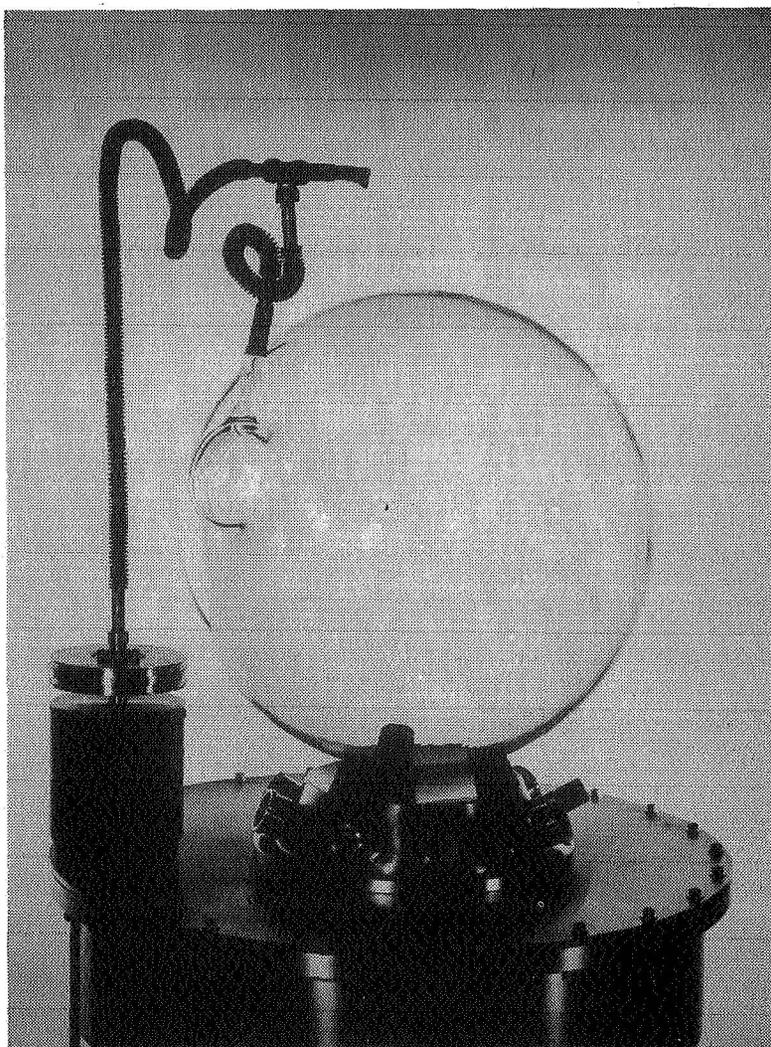
The second deposition process used was the more widely known vapor deposition, where a tungsten filament, coated or wrapped with a metal, is heated in a vacuum, causing the metal to vaporize, instantly covering the substrate of the object being coated. This process was performed inside of three 3,000ml glass spheres.



Uncoated sphere and filament
in pre-evaporation configuration
and completed sculpture with
gold deposition.

The three vapor deposition experiments performed as expected, creating two sculptures with very clean opaque coatings of gold and aluminum and one sculpture with a coating deposited from a two stage filament coated with aluminum and silicon monoxide, creating an iridescent effect.

The ninth and largest sphere (22,000ml) served as a sampling system, allowing the measurement of the working vacuum when the payload returned to earth. This sphere was connected to space via a high vacuum valve. Over a three day period the interior of the sphere attained an equilibrium with the vacuum of the shuttle orbit, becoming one with the vacuum of space. A copper tube connecting the sphere to the valve was cold welded, permanently sealing the sphere, creating the sculpture "S.P.A.C.E.". Attached to the sphere is a Baratron capacitance manometer, a vacuum gauge capable of a digital reading of the vacuum forming the sculpture inside the sphere.



The sculpture "S.P.A.C.E.",
with sphere and attached
capacitance manometer.

The sculpture "S.P.A.C.E." is not the glass, but the outer space contained within. The sphere serves only to keep the one-g earth atmosphere from

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intruding on the space within, creating an anomaly of our common experience; a sculpture to observe and stimulate wonder about the nature and meaning of space, a sculpture to touch and know that only an 1/8" of glass separates one from space.

Payload 3-39 will explore space with sculptural concepts, creating art on a scale not previously accessible; using materials and processes in a way specific to, and possible, only, in the vacuum and weightlessness of the shuttle orbit. Two separate methods of forming structure in space with inflated objects are being considered for payload G-39. One method will use thin polyimide films coated by vapor deposition with a reflective aluminum coating and ejected from the shuttle and inflated in orbit. A series of inflated aluminized shapes have been created and tested on earth for use as structure in space. When used in Space Art exhibitions this past year these forms have been filled with helium so they float. To draw attention to the Get Away Special program, they have, in addition to their reference as Space Sculptures, been referred to as "GAS" Sculptures. The second method would be a continuing exploration of the technology created and being developed by Gilbert Moore for the formation of inflated polymer bubbles in space as structure. This would allow the continuation in space of a series of bubble sculptures that I have blown for years on earth and for the eventual metallic coating of these forms in space, utilizing vapor deposition processes explored and developed in payload G-38.



Moore Stars in Space

Photo collage depicting the inflation of bubbles in orbit. Titled in honor of the creator of the space bubble process.

As sculpture either process is intended to create a subtle reordering of our sense of scale and place; the sculpture being not the objects themselves, but conceptually the 30,000 mile circumference of the orbit they travel, creating a sculpture on a scale not achievable on earth. The earth will form the center of this art work and instead of our walking around a sculpture resting on a base, the sculpture will orbit around us and we will exist inside of it and look out at it. G-39 is intended as a peaceful celebration of man's venture into space; an art work possessed by no one and equally accessible to all.

G-40, like the other payloads, is designed to prompt viewers to toss out old definitions of space and distance and to permit terrestrial observers to experience space in as direct a way as possible. The purpose of this experiment is to test new concepts of signal transmission and directional stabilization for orbiting transmitters.

The transmitter, designed for ejection from a GAS canister, will be activated in orbit and transmit signals to a series of ground based receiving sculptures. These receivers, located at selected sites around the earth, will establish a symbiotic relationship with the transmitter. The receivers simultaneously monitor the signal from the satellite and serve as objects that permit the viewer to experience the passing of the satellite and the nature of its orbit. As the transmitting satellite/sculpture comes into range of a receiving sculpture, the viewer will see the receiving sculpture slowly come to life with light and sound. This observable change in the status of the receiver will increase in intensity as the satellite approaches the receiver, reaching a crescendo at the moment the satellite is closest to the receiver and then diminishing until the satellite is out of range. Like the sensation experienced when observing a train passing in the night, the viewer will experience the passing of the satellite through the phase change in the receiving sculpture; similar to the observation of a Doppler effect as an object moves toward or away from the earth in space. A few minutes after the signal diminishes from a receiving sculpture in New York, the phenomenon will be experienced by the viewers of a receiving sculpture outside the

Beaubourg in Paris, and so on around the earth in correlation with the satellite's orbit, connecting observers around the earth in a common experience; a traveling exhibition about spatial relationships that circumnavigates the earth every 90 minutes.

During this past year I participated in a Space Art show in Paris with two French artists, Pierre Comte and Jean Marc Philippe, both preparing art payloads for flight on board the European space rocket. I would hope that they, Joe Davis (also preparing a GAS payload) and others will soon join in the creation of art in space; a new and humanistic perspective seeking understanding of man's journey.

"Science explains the world,
but only art can reconcile us to it."

Stanislaw Lem

In conclusion I would like to express my appreciation to everyone involved with the GAS program that I have worked with during the past eight years. All have provided invaluable assistance.

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10 P.

DESIGN AND DEVELOPMENT OF A WINDOW ASSEMBLY
FOR A G.A.S. PAYLOAD

8983

Bev Christie and Bob Tough
Bristol Aerospace Limited
Winnipeg, Manitoba

BW741890

ABSTRACT

A requirement exists for a sealed window assembly for G.A.S. payloads. Bristol Aerospace Limited has designed and developed a synthetic fused silica window assembly for the National Research Council of Canada's G.A.S. payload 'PHOTONS' G-494. The details of the design and development of this window assembly are given in this paper.

1. INTRODUCTION

For some G.A.S. payloads utilizing the Motorized Door Assembly (MDA) option, a requirement exists for a sealed window assembly. In addition to satisfying the scientific design requirements, the window assembly must also meet the NASA safety related requirements for payload flight-qualification.

Bristol Aerospace Limited has designed and developed a synthetic fused silica window assembly for the National Research Council of Canada's G.A.S. payload 'PHOTONS' G-494 (refer to Figure 1 for general payload layout). This paper will describe the requirements and details of the actual design and development phases of this window assembly. It should be noted that at the date of publication of this paper, the 'PHOTONS' payload has not been flown.

2. REQUIREMENTS

2.1 Scientific Requirements

The experiment, to be conducted by Dr. F. Harris of the Herzberg Institute of Astrophysics, NRC, is named PHOTOMETRIC OXYGEN NIGHT GLOW STUDY (PHOTONS), and it will measure terrestrial night glow emissions from atomic and molecular oxygen atmospheric band. The experiment will also make observations of the shuttle RAM glow, quantify the spectrum, and evaluate its significance to measurements made from the shuttle bay.

There were four primary scientific requirements which established the design of the window assembly, these were:

- 1) Optical Requirements - The windows must allow transmission of O and O₂ emissions from the U.V. to the near I.R. (wavelengths ranging from 2860 to 8653 Angstroms). Windows must be provided for:

Two 3-barrel photometers
One single-barrel photometer
One bright light sensing diode

- 2) Field-of-View Requirements - Uni-directional measurements to be taken by photometers (8° maximum conical field-of-view).
- 3) Thermal Constraints - The payload thermal limits are -20°C to +40°C while the temperature limits of the photometer filters, which are immediately behind the windows, are -30°C to +30°C.
- 4) Leak Rate - Photometers are sealed to prevent leakage and subsequent high voltage breakdown. However, for redundancy, it is required that the payload be sealed and, based on typical shuttle flight duration times of 3 to 4 days, not leak at a rate greater than 1 psi every 24 hours based on a nominal one atmosphere pressure inside the G.A.S. container.

2.2 NASA Requirements

NASA requirements consist largely of safety considerations. It also includes physical size and weight constraints, in accordance with the G.A.S. concept. The safety considerations include structural criteria, and material selection criteria.

Detail structural design requirements, which are presented in subsequent sections, are obtained from the G.A.S. payloads safety manual (Reference 1). Included are design loads and factors of safety which are to be used in structural analyses.

The NASA materials selection criteria is contained in Reference 2. Materials used on G.A.S payloads should have low susceptibility to corrosion and pitting, high resistance to stress-corrosion cracking and resistance to brittle crack propagation. The use of dissimilar metals in contact should be avoided, wherever possible.

The physical size and weight constraints of G.A.S payloads utilizing the MDA option are as follows:

Maximum diameter:	19.75 inches
Maximum length:	28.25 inches
Maximum weight:	165 lbs

3. DESIGN

3.1 General Configuration

The design of the top plate assembly involved trade-offs between weight, manufacturability, and cost. The layout of the window assembly is shown in Figure 2. It consists of an aluminum plate having four high purity fused silica windows - two large windows (6.25 inch diameter) for the 3-barrel photometers, and two small windows (3.0 inch diameter) for the single-barrel photometer and the bright light sensing photo diode. All four windows are sealed with neoprene rubber gaskets and held in place by retainers which are bolted to the top plate. The outer edge of the plate is clamped between two rings which are part of the G.A.S container assembly. The plate is sealed at the outer edge by O-rings. To reduce the heat transfer through the plate, it is highly polished on the outer surface and is alodined on all surfaces on the payload side. A weight breakdown of the window assembly is given in Table 1.

TABLE 1. Weight Breakdown of Top Plate Assembly

COMPONENT	WEIGHT (LBS)
Top Plate Machining	21.0
Window Retainers - Large 2 @ 0.383 lbs each	0.77
- Small 1 @ 0.173 lbs	0.17
- Diode 1 @ 0.433 lbs	0.43
Gaskets (Neoprene), and diode window retainer O-ring	0.60
Windows - Large 1 @ 1.278 lbs and 1 @ 1.277 lbs	2.56
- Small 1 @ 0.292 lbs and 1 @ 0.287 lbs	0.58
TOTAL	26.11 lbs

3.2 Materials Selection

The following is a list of the window assembly components, the materials selected, and the rationale.

- 1) Windows - The following window specification was derived to meet the requirements for spectral transmission and optical flatness:

Material - Dynasil UV-1000 synthetic fused silica
 Flatness - Both faces flat to $\lambda/4$ @ 546 nm
 Parallel - Faces parallel to 5 arc minutes
 Surface Finish - Scratch and dig 40-20

The windows were supplied by Interoptics Ltd. of Ottawa.

- 2) Structure - In order to meet weight constraints, aluminum alloy 6061-T6 was chosen for the structure based on its high strength to weight ratio and for its high resistance to stress-corrosion and brittle fracture.
- 3) Seals - Neoprene AMS3207 rubber was chosen as a gasket material because of its low static seal leak rate, low outgassing, and its ability to retain required physical and mechanical properties of temperatures as low as -40°C.
- 4) Hardware - Stainless steel hardware was chosen for mechanical fasteners because of its high resistance to corrosion and stress-corrosion cracking.

3.3 Structural Design Criteria

Pressure Loads

A G.A.S. canister pressure of 1.5X atmospheric (22 psi) was established as a conservative pressure for the fracture analysis of the windows. This constituted the maximum pressure loading applied to the windows during flight, including a factor of safety. In order to account for uncertainties in the structural analysis of the top lid assembly, NASA specified a design case of 4X atmospheric pressure (60 psi). This ultimate factor of safety was used in stressing the windows and the aluminum plate.

Flight Loads

The inertial flight loads were obtained from the G.A.S. Payload Safety Manual (Reference 1) and are summarized in Table 2.

TABLE 2. Design Flight Loads (Acceleration in g's)

Direction	Limit Load	Yield Load	Ultimate Load
X	± 6.0	± 9.0	±12.0
Y	± 6.0	± 9.0	±12.0
Z	±10.0	±15.0	±20.0

The axis orientation is shown in Figure 1. The loads are combined using the X, Y and Z loads in the worst possible combination. The Yield Loads give a factor of safety of 1.5 on the Limit Loads and the Ultimate Loads give a factor of safety of 2.0. These loads are for design analysis which is not verified by testing.

Temperature Limits

The temperature design limits for the payload during the mission are -20°C to +40°C.

3.4 Structural Analysis of Lid Assembly

Preliminary calculations indicated that the stresses and deflections in the lid assembly were of small magnitude, and a finite element analysis of the lid was not warranted. The stress levels were estimated with sufficient accuracy using conservative, classical analysis techniques.

In the top plate assembly, discontinuity stresses occur at locations of abrupt change in geometry, such as adjacent to the window mounts and the holes in the plate. In order to estimate the maximum stresses in the plate, a discontinuity stress analysis was performed on these areas. The analysis consisted of separating the structure into a number of elements of simple geometry, such as flat plates, rings and cylinders, which approximated the lid and structural discontinuities. A system of redundant forces and moments was calculated at each element edge. These are required to maintain structural continuity when the structural loads are applied. The redundant forces and moments are found by solving a system of simultaneous equations which express the compatibility of deformations at adjacent element edges. Stresses due to the system of redundant forces and moments are then calculated for each element and superimposed on the "free-body" pressure stresses in each element.

In performing the analysis, conservative, simplifying approximations are made for the actual top plate geometry. The modelled geometry is less stiff than the actual geometry, therefore, stresses predicted by this method are conservatively high.

The structural analysis performed addressed pressure loads, thermal loads, and flight loads. Vibration stresses were verified by testing, the details of which are covered in subsequent sections. Stresses in the lid and synthetic fused silica windows were calculated for the design pressure of 60 psi, and were found to give positive margins of safety based on the material yield strength of 6061-T6 aluminum, and ultimate strength values for UV 1000 synthetic fused silica. Pressure stresses in the windows were also calculated for the maximum service pressure of 22 psi. These stress levels were used in conducting the fracture analysis of the windows. The stresses due to differential thermal expansion between the windows and aluminum were considered negligible since the windows are isolated from the top plate by neoprene rubber gaskets at the edges. Stresses due to thermal gradient across the window thickness were found to be negligible. Flight loads were found to produce substantially lower stresses in the lid and windows than the pressure loads and were therefore considered negligible.

3.5 Fracture Analysis of Glass Windows

Analysis Method

Brittle materials, such as glass, which are subjected to static loading exhibit a decrease in strength with time known as static fatigue. Brittle fracture is preceded by subcritical crack growth which results in delayed failure. The principles of fracture mechanics can be used to predict the structural lifetimes and to develop acceptance tests for glass subjected to load. (Reference 3).

The time-to-failure of glass under external load is determined as the time necessary for surface flaws to grow from an initial, sub-critical length to a final critical size at which spontaneous failure occurs. The time-to-failure can be calculated from the crack velocity, $V = \frac{dL}{dt}$, and the stress intensity

factor, K_I , which is a measure of the stress field at the crack tip. The stress intensity factor is defined by the Griffith criterion as,

$$K_I = \sigma Y \sqrt{L} \quad (1)$$

where

Y = crack geometry factor

L = crack length

σ = surface stress in the vicinity of the crack tip

For a constant stress, the failure time, t, is found by substituting equation 1 into the expression for crack velocity.

$$t = \frac{2}{\sigma^2 Y^2} \int_{K_i}^{K_{IC}} \left(\frac{K_I}{V} \right) dK_I \quad (2)$$

where

K_i = initial stress intensity at the most critical initial flaw

The fracture calculations indicated essentially infinite lives for the single-barrel photometer window and the diode window and a lifetime of 167 years for the three-barrel photometer window, at a continuous service pressure of 22 psi.

3.6 Thermal Design

A thermal model has been developed to simulate the modes of heat transfer of the G.A.S. payload assembly (conduction and radiation). The results of the thermal analysis have determined that low transmission of energy through the top plate assembly is required in order to achieve the design temperature limits. A highly polished surface was chosen for the top of the aluminum plate for its very low emittance and absorptance of thermal energy. The surfaces on the payload side of the top plate assembly could not be easily polished, and were therefore alodined. This surface finish has a low emittance and absorptance. The fused silica windows have a high transmittance in the infra-red spectrum, but are required for the experiment.

4. TEST PROGRAM

4.1 Scope of Work

The test program included leak, vibration, thermal cycle, qualification proof pressure, and flight acceptance proof pressure tests. The leak test was performed to ensure that the pressure of the G.A.S. container would not drop below the critical level. The vibration test was required to verify structural integrity and pressure seal function under anticipated vibration loads. The thermal cycle test was also performed to verify pressure seal serviceability over the temperature design range. The qualification proof pressure test was performed to verify structural integrity of both the plate and the windows. The flight acceptance proof pressure test was essentially a repeat of the qualification proof pressure test; its purpose was to ensure that the actual flight windows were structurally acceptable. This test was conducted immediately prior to shipment of these components.

4.2 Leak Test

The acceptable leak rate was established as a drop of less than 5 psia from the nominal container pressure of 15 psia over the 5 day flight duration. The test fixture shown in Figure 4 was connected to a pressure transducer which monitored the internal test fixture pressure. The test fixture was pressurized to the maximum anticipated pressure (1.1 atmos or 16.5 psig, this pressure accounts for pressurization due to thermal effects) at 20°C ambient temperature. All connections were checked for leaks with OXYTEC, and the pressure was monitored for 24 hours, during which a drop of less than 1 psig was measured; this met the leak rate requirement noted in para 2.1.

4.3 Vibration Test

For the vibration test, the test fixture (Figure 4) was mounted to a vibration mounting plate which was in turn mounted to a shaker. The test fixture was again pressurized to 16.5 psig at 20°C ambient temperature and all connections checked for leaks. The top plate assembly was subjected to the following vibration levels (Black Brant V Sounding Rocket Subsystem Qualification Levels):

K_{IC} = critical stress intensity factor at which failure occurs

The crack velocity, V , is established as a function of the stress intensity factor, K_I , by fracture testing. For fused silica, a least squares fit of experimental data gives a function of the form

$$V = e^{\left(\frac{K_I - a}{b}\right)}, \text{ where } a \text{ and } b \text{ are constants}$$

In performing the analysis, experimental values for K_{IC} , a and b were obtained from published data for fused silica (Reference 4).

Proof-Pressure Testing

Typically, the initial surface flaws which exist in glass after grinding and polishing operations are too small to be measured by conventional non-destructive testing techniques. The most effective approach to determine K_i is by proof testing. Proof testing imposes a load on the component which is higher than the maximum expected service load. This establishes an upper limit for the maximum size of flaw that can be present after the proof test has been completed. Survival of the proof test guarantees that the stress intensity at the most serious flaw is less than the critical stress intensity factor because fracture is almost instantaneous when $K_i = K_c$. Thus by the Griffith criterion,

$$K_{IC} \geq \sigma_p Y \sqrt{L_i} \quad (3)$$

where

σ_p = proof stress

L_i = initial flaw size

For the service stress, σ , the initial flaw size results in a stress intensity of:

$$K_i = \sigma Y \sqrt{L_i}$$

From equation 3, $L_i = \left(\frac{K_{IC}}{\sigma_p Y}\right)^2$

Therefore,

$$K_i = \left(\frac{\sigma}{\sigma_p}\right) K_{IC}$$

Lifetime Estimates for Glass Windows

Using fracture properties for fused silica from Reference 4, the times-to-failure were calculated from equation 2. The glass windows were assumed to be subjected to a continuous service pressure of 22 psi. The proof test pressure used was 60 psi, which is in agreement with recommended values from literature (Reference 4).

Sinusoidal Vibration

Longitudinal

7.5 g peak, 2000 to 27 Hz,
0.2 in peak to peak, 27 to 15 Hz

Lateral

7.5 g peak, 2000 to 38 Hz,
0.1 in peak to peak, 38 to 15 Hz

NOTE: Each vibration test is a single logarithmic sweep with a total duration of 115 seconds (3.7 octaves/min) starting at the high frequency end and having no dwell time at any frequency other than the starting point.

The test was repeated in each of the three principal axes. After vibration, the assembly was maintained at 20°C ambient temperature for 24 hours. The acceptance criteria was that the test fixture internal pressure did not drop more than 1 psi. In fact, no measurable drop in pressure was observed.

4.4 Thermal Cycle Test

The thermal cycle test set-up had the test fixture assembly (Figure 4) inside an environmental control chamber (Convicon). The test fixture assembly was pressurized to 16.5 psig at 20°C ambient temperature and subjected to the temperature profile shown in Figure 3. During testing, the top plate temperature and test fixture internal pressure were monitored. The test acceptance criteria was that a pressure drop of less than 1 psi occur over the test duration. Again, no measureable pressure drop was observed.

4.5 Qualification Proof Pressure Test

The test fixture (Figure 4) was connected to a mechanical pressure gauge, and in steps of 5 psi, was pressurized to 60 psig. Strain gauges, which were attached to various places of concern on the top plate, recorded strains at each step. At maximum pressure, strain and pressure readings were taken every minute for 10 minutes, at which time the pressure was reduced in steps of 10 psi to zero, recording strain at each step. The acceptance criteria was based on achieving the following factors of safety:

Yield Reserve factor = 1.25
Ultimate Reserve factor = 1.5

The results of the qualification proof pressure test indicated a maximum stress of 8630 psi occurring adjacent to the diode window on the central axis towards the centre. This gives a yield reserve factor of 4.1 and an ultimate reserve factor of 4.9, both well above the acceptance criteria and the theoretical results (theoretical yield reserve factor = 1.99).

4.6 Flight Acceptance Proof Pressure Test

The set-up for the flight acceptance proof pressure test was similar to that of the qualification test but did not include the strain gauges. The internal test fixture pressure was increased to 60 psig at a rate of 10

psi/minute. The pressure was maintained at 60 psig for 1 minute and then decreased at a rate of 10 psi/minute until 0 psig was reached. The acceptance criteria was based quite simply on the windows carrying the maximum pressure load without failure. This test was successful.

5. CONCLUSIONS

Based on the design and development phase described above, the following conclusions may be stated.

- 1) Structural integrity has been demonstrated in accordance with NASA safety requirements.
- 2) Nominal pressure seal requirements have been satisfied for normal operating conditions.
- 3) The scientific optical requirements have been met.
- 4) The top plate assembly has been designed to satisfy the payload temperature design limits.

In summary, all tests and analyses have demonstrated that the top plate assembly is flight-ready.

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- 2 - JSC11123 - Space Transportation System Payload Safety Guidelines Handbook.
- 3 - Wiederhorn, S.M.; "Prevention of Failure in Glass by Proof-Testing"; Journal of the American Ceramic Society, Vol. 56, No. 4, April 1973.
- 4 - Evans, A.G.; Roberts, D.E.; Wiederhorn, S.M.; "A Fracture Mechanics Study of the Skylab Windows"; Fracture Mechanics of Ceramics, Vol. 2.

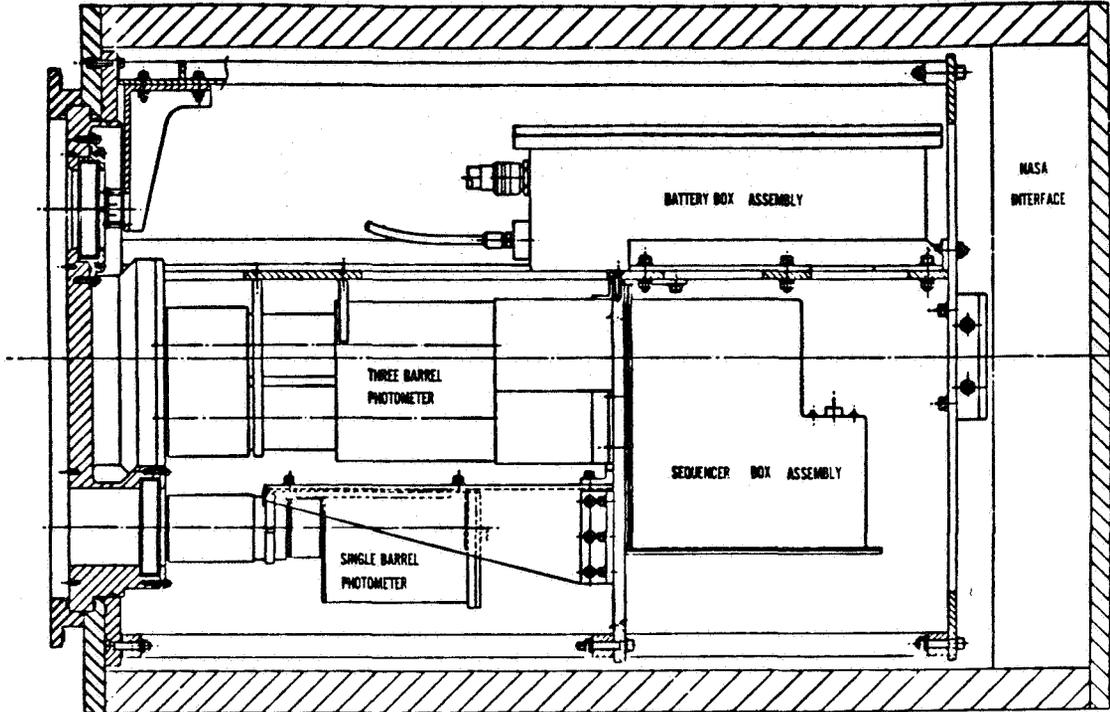


Figure 1

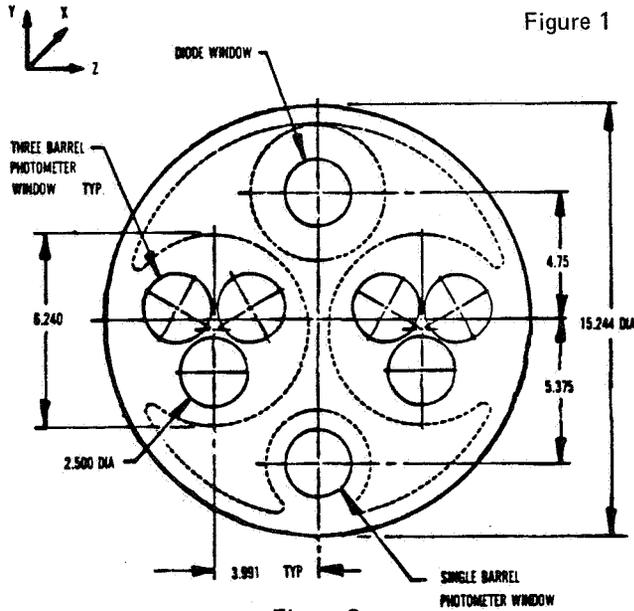


Figure 2

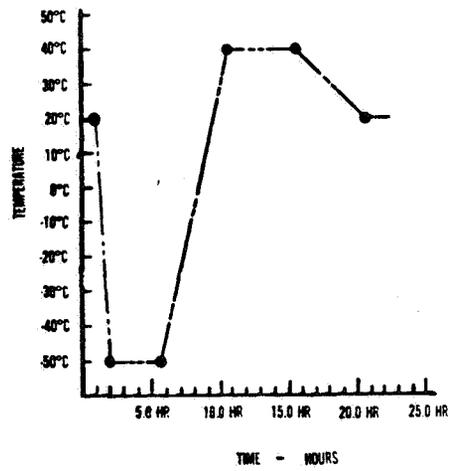


Figure 3

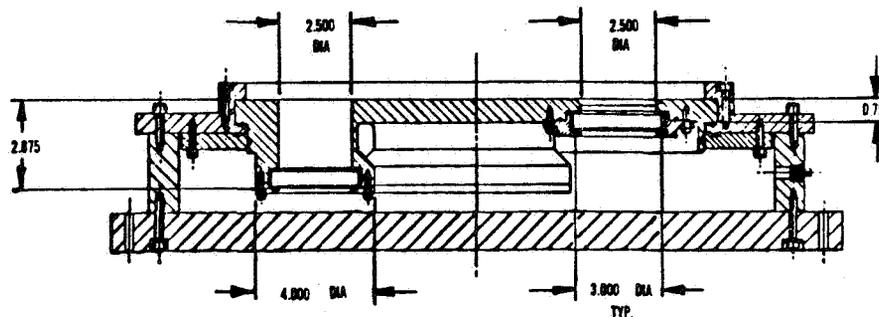


Figure 4

THE GLOMR SATELLITE
PAYLOAD G-308

NC999967

Abstract

This paper describes the Global Low Orbiting Message Relay (GLOMR) Satellite and its participation in the pioneering development of a Get-Away-Special (GAS) satellite launch capability. The GLOMR is a data relay communication satellite designed and built by Defense Systems, Inc. (DSI), a private company located in the Washington, D.C. area. Experiment objectives are to demonstrate the store-forward relay of data/messages and the communication to ground user equipment. The 150 pound brass satellite, powered by lead-acid batteries and solar cells, includes processing, control and RF electronics for data message handling and storage. GLOMR is uniquely mounted through a Marman clamp ejection/retention pedestal to a five cubic foot GAS canister bottom plate. The GAS canister includes a Full Diameter Motorized Door Assembly (FDMDA) lid which is opened remotely by astronauts for satellite launch.

Introduction

The GLOMR Satellite design was driven by experiment objective requirements to package the electronics that would incorporate a dual frequency transmission and receiving capability, data/message handling and storage, and DC power conversion and control (timing). Physical constraints were imposed by the NASA GAS canister volume, shown in Figure 1. In addition, safety requirements were imposed to preclude jeopardizing the Shuttle flight mission. A detailed listing of design issues and their solutions would be prohibitive in this forum, but significant activities addressed in this paper include:

1. Optimum weight to volume
2. Maximum solar cell exposure with random attitude
3. Structural integrity
4. Operational safety and hazard control

Satellite Description

Figure 2 is a photograph of the GLOMR Satellite mounted on the NASA ejection/retention pedestal. The polyhedron shaped shell is machined from naval brass. The solar panels form three orthogonal belts, including the bottom surface which interfaces with the pedestal. Satellite retention on the spring loaded pedestal is provided by a Marman clamp that is released on command by two pyrotechnic cutters. The compression spring thrusts the satellite out of the GAS canister at approximately 3.5 ft/sec after the cutters sever the Marman clamp bolts. The four dipole antennas are made of copper clad printed circuit board material and provide essentially an omni-directional beam pattern.

The internal electronics, shown in the two views of Figure 3, are mounted on a single aluminum central plate which also acts as a heat sink. Functional elements shown include dual transmitters and receivers, antenna phasing and control electronics, a digital processor and data storage memory system, dc-to-dc converters, and Gates D-size lead-acid batteries. The batteries are housed in two separately sealed containers. Housekeeping sensors include pressure and several temperature sensors.

Effective radiation power is approximately 10 watts.

Design Issues

The NASA supplied ejection/retention pedestal, previously used on Delta launch vehicles, limited the GLOMR Satellite weight to a maximum of 150 pounds. Brass was selected as the basic shell material because of its high density property, which provided a greater ability to approach the maximum weight constraint. In low orbits, higher weights for a given size improve the orbital life. In addition, brass has excellent thermal properties, an important consideration for electronic component performance. The exposed portions of the satellite's shell is painted with black absorbing paint to improve the solar heat input.

After ejection from the GAS canister, the satellite assumes a random attitude. To provide a predictable and optimum charge to the batteries while tumbling in orbit, the solar panels are installed in an array of three continuous orthogonal belts. This arrangement optimizes solar cell exposure to sun light regardless of the satellite's attitude. In addition, the omni-directional antenna arrangement also provides full coverage at random attitudes.

The satellite structure was designed (and tested) to withstand up to 12.9 g's RMS. The unique material selection and orthogonal shell configuration necessitated an extensive analysis and vibration testing at the NASA T & E facility. The critical area requiring the most attention, both in structural design and in validation testing centered on the mechanical interface of the satellite to the pedestal. The Marman clamp retains the satellite on a 9-inch diameter base. To provide the same 9-inch diameter continuous ring on the satellite would require the elimination of solar panels, or at least a structural ring that would shadow the cells. Both conditions were unacceptable. The solution was a unique segmented ring, four parts (feet) affixed to the satellite and the other four parts retained by the pedestal. The four feet on the satellite must, however, provide all the strength necessary to retain the satellite on the pedestal with the clamp while withstanding the shuttle launch and landing loads specified by NASA. Brass could not, but 4130 steel, hardened to 180 ksi could and did.

An hermetically-sealed microswitch was mounted integral with each of the four steel feet. The normally closed switches, redundantly wired in series and in parallel, "awaken" the GLOMR Satellites after release from the ejection/retention pedestal. In order to address NASA's safety concerns (discussed further in the next section), hermetically-sealed switches were required to preclude sparking in a potentially volatile atmosphere (e.g., propellant fumes, hydrogen gas, etc.). In addition, with the satellite attached to the pedestal and the switches in an "open" mode, inadvertant radiation is prohibited. Once launch occurs, the satellite's internal control circuitry must "see" two day-night cycles, plus 90 minutes, before transmission is allowed. This approach provides the time for natural separation in orbit between the GLOMR and the Shuttle.

GLOMR Qualification Process

To ensure maximum protection to the Orbiter, crew and other satellites, all GAS payloads must meet rigorous safety requirements. Detailed safety plans must be accepted by NASA authorities, usually after several joint sessions concerning test results. With respect to the GLOMR satellite, the following hazards were analyzed in detail:

- o Satellite structural failure
- o Antenna collision with GAS can during ejection
- o Emission of RF Radiation
- o Battery caused explosion or electrolyte leakage

The above hazards are typical of the many analyses and subsystem testing required to prove a gas satellite flightworthy. Defense Systems, Inc. (DSI), provided the following analyses during the GLOMR Satellite's qualification process:

- o Antenna patterns
- o Structural test for pressurization (final satellite was unpressurized)
- o Launcher release tests
- o Vibration tests
- o Thermal tests of each printed circuit board
- o Antenna blade fracture tests
- o Foot-mounted microswitch testing and alignment
- o Battery housing checks
- o Post ejection radiation control

These were the subject of many meetings and correspondence between DSI and NASA engineers to qualify the GLOMR for launch. While a launch anomaly prevented the April 1985 deployment of the satellite, the rigors of the 7 day mission have confirmed the findings of the extensive analyses.

The development of a unique, segmented Marman plate interface, typifies the exacting, time-consuming qualification process which must precede every flight. Table 1 summarizes many of the areas requiring close coordination between the GAS Project Office and the experimenters.

Conclusion

The GLOMR satellite was built in less than 1 year. This would have been impossible without valuable assistance provided by the GSFC GAS Project Office. Not only did the Office provide assistance in component selection and qualification testing, but the GLOMR exists today as a forerunner of low cost access to space. Formidable technical and procedural obstacles have been overcome to allow this experiment to play a significant part in the NASA mission of Space Commercialization.

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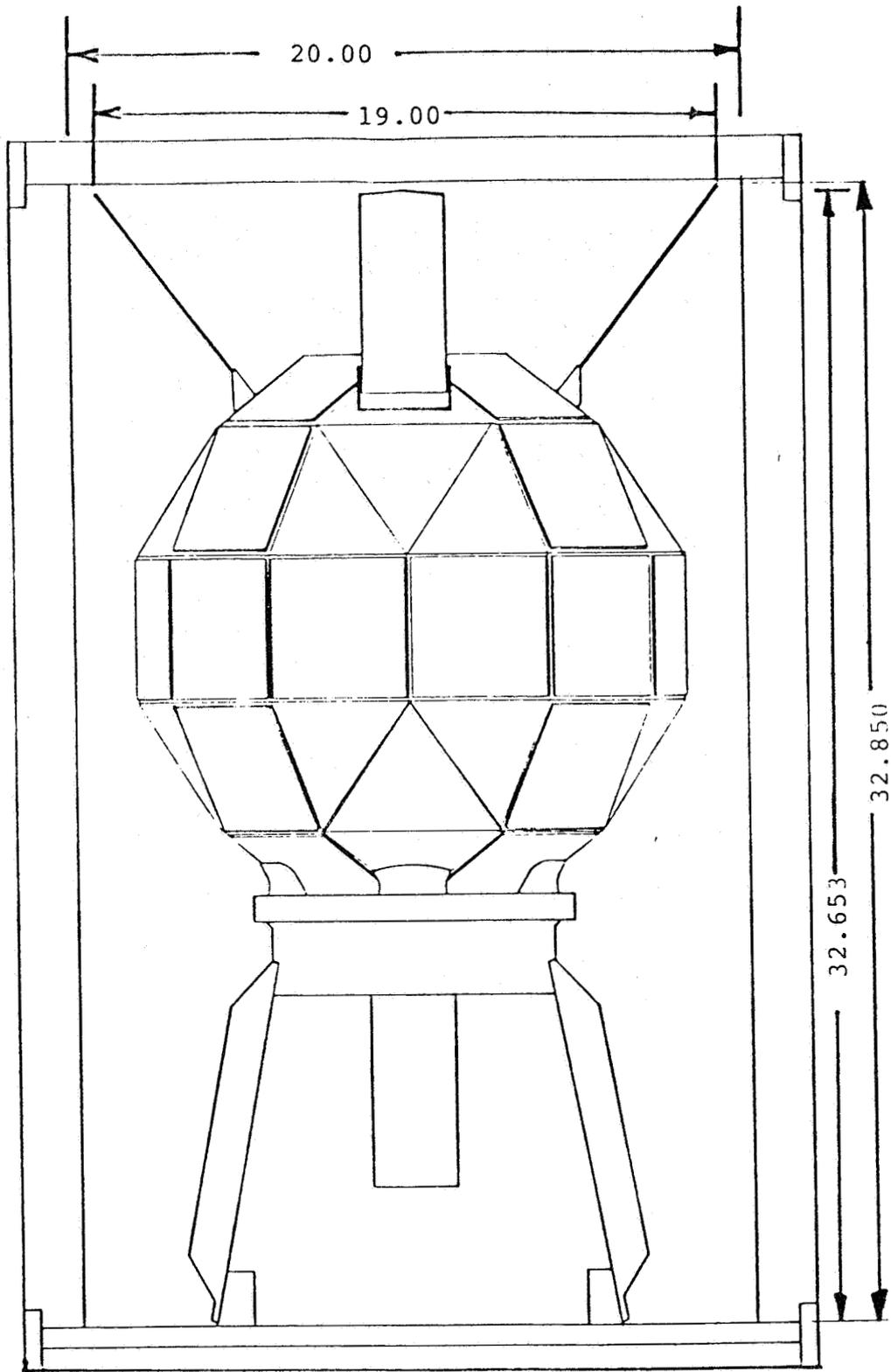


Figure 1. GLOMR in GAS Canister

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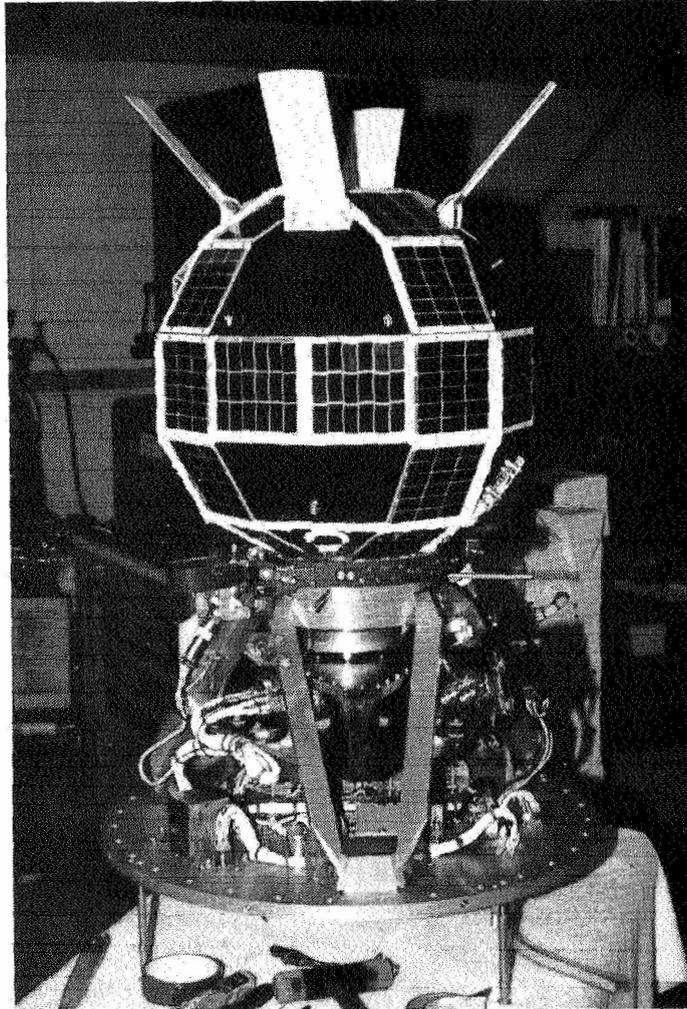


Figure 2. GLOMR Satellite and Launcher

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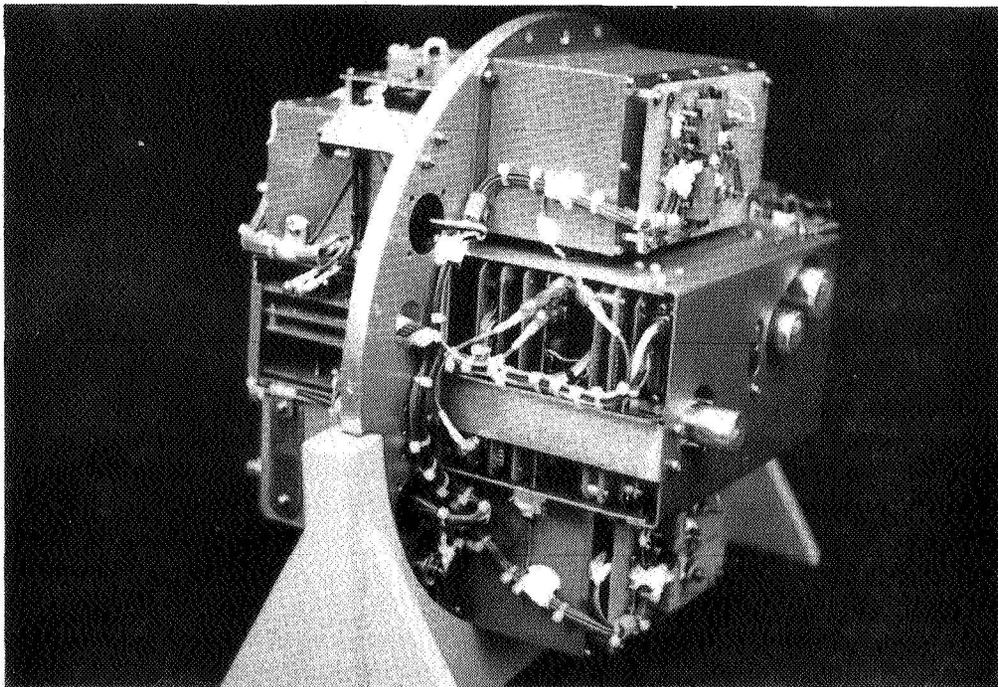
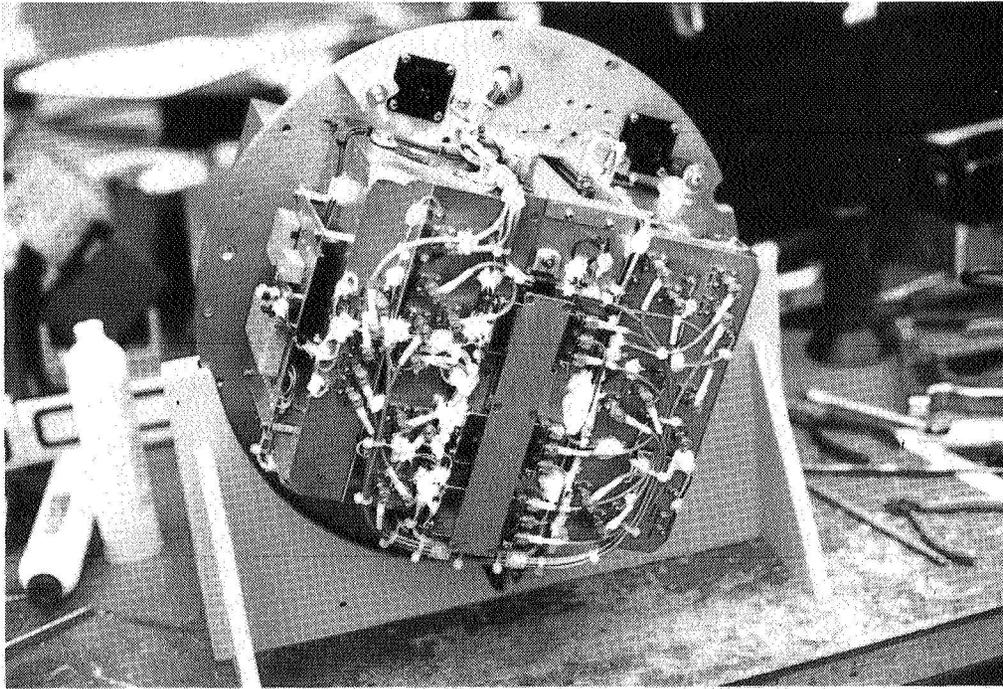


Figure 3. GLOMR Chassis, side view

Table 1
 GAS Payload Qualification Requirements (A partial listing)

QUALIFICATION REQUIREMENT	NASA AND EXPERIMENTER ACTIVITIES
o Material stress factors	o Pressurization test to insure structural integrity of cast shell
o Satellite feet/ejection system compatibility	o Vibration and deployment tests
o Battery caused explosion (control hydrogen gas development and/or electrolyte leakage)	o Implementation and test of hermetically-sealed microswitches
o Antenna design	o Use NASA-approved batteries o Install pressurized battery housing o Use hermetically-sealed microswitches o Fuzing in negative lead
o Solar cell breakage	o Flexible material avoids GAS can damage o Design features to preclude lateral motion o Non-interference with Shuttle communications
o Other (corrosion, electrical shock, temperature extremes, out gassing materials)	o Handling fixtures o Detailed integration handling procedures o Experiment-unique. Coordinate with GAS Technical Liason

OSCILLATORY FLOW
MARANGONI FLOW
REDUCED GRAVITY
PHASE TRANSFORMATION
LIQUID FLOW

SPACE SHUTTLE MISSION 41-D
PARAFFIN
MARANGONI CONVECTION
THERMAL BOUNDARY LAYER
THERMAL EXPANSION
OSCILLATING FLOW

D32-34
N86-27329
8985

THERMOCAPILLARY FLOW AND GASEOUS CONVECTION IN MICROGRAVITY: RESULTS FROM GAS PAYLOAD G-0518

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Abstract: Thermocapillary flow and gaseous convection in microgravity were investigated in GAS payload G-0518 during Space Shuttle mission 41-D. A cylinder of paraffin was supported and heated differentially from its ends to induce a melt from solid to liquid and drive thermocapillary flow in the resulting liquid phase. Laminar thermocapillary flow was observed in the liquid paraffin and found to show a transition to time-dependent oscillatory motion at a Marangoni number of about $Ma=34000$ with a period of approximately $T=8$ seconds. In addition, free convection in a gas in microgravity has been observed for the first time. The gaseous convection was caused by the thermal and/or velocity boundary layers present at the heater-liquid interface. Oscillations occurred in the gaseous convection simultaneously with those in the liquid, implying the two are strongly coupled. The gaseous convection may be driven by coupled thermocapillary flow / thermal expansion convection or microgravity buoyancy convection.

Introduction

The technology of processing materials in space requires a knowledge of the convective flows that can occur in a fluid in microgravity. Probably the most significant type of convective flow in microgravity is that caused by thermally induced surface tension gradients along a liquid-gas interface, thermocapillary flow. Other types of convection in microgravity may also prove to be important to materials processing (ref. 1). The experimental study of thermocapillary flow (or any type of convection important to materials processing in space) in 1-g on the earth is impeded by the simultaneous existence of buoyancy convection caused by thermally induced density gradients. Experiments in microgravity are therefore desirable to study thermocapillary flow and its applications to materials processing in space. This experiment was designed to study thermocapillary

flow in a cylinder of paraffin supported and differentially heated at its ends (fig. 1); a system analogous to a float zone used in crystal growth (fig. 2).

The surface tension of a liquid decreases with temperature so a temperature gradient along a free surface causes a surface tension gradient which in turn causes surface tractions. These surface tractions are balanced by viscous shear stresses which induce motion in the liquid. The dimensionless number describing the magnitude of this phenomenon is the Marangoni number

$$Ma = \frac{dS/dT}{DT} \frac{d}{uK}$$

where dS/dT is the temperature coefficient of surface tension, $DT = (T_2 - T_1)$ is the temperature difference across the surface, d is the length of free surface from T_1 to T_2 , u is the absolute viscosity, and K is the thermal diffusivity. It is known experimentally (ref. 2) and theoretically (ref. 3) that for small Ma simple time independent laminar flow occurs. It is also known experimentally, both from ground experiments (ref. 4) and from sounding rocket experiments (ref. 5), that for large Ma ($Ma >= 10000$) the flow shows a transition to time dependent oscillatory flow (fig. 3). It is not known exactly how or why this instability from laminar to oscillatory flow occurs and the precise planform of the instability in microgravity is not known although preliminary results have been obtained (ref. 5 & 6).

As mentioned above other types of fluid convection in microgravity may be important or significant. These include, among others, microgravity buoyancy convection, g-jitter convection, and thermal expansion convection. These have been studied theoretically and experimentally for simple geometries and boundary conditions (ref. 7) and found to be insignificant. But as Ostrach points out (ref. 8) boundary conditions and geometric configuration may be a dominant factor in determining significance of some types of convection in microgravity. The boundary conditions which exist between a cylinder of liquid undergoing high Ma thermocapillary flow and its surrounding gas (discussed below) have not previously been applied to the types of convection mentioned above. The author has carried out a preliminary analysis of possible convection in the surrounding gas using the pertinent boundary conditions to the system studied here and found significant flow can occur in the surrounding gas. (ref. 9).

This paper reports on the observation of laminar and time dependent thermocapillary flow and gaseous convection in microgravity for a differentially heated cylinder of paraffin surrounded by a gas.

Apparatus

The apparatus consists of support and control electronics and hardware for the basic experiment mounted in a fiberglass epoxy hexagonal tray. The hexagonal tray is mated to two others and bolted to the upper lid of a half size GAS canister. Electrical power for the various experiment systems is supplied by 2.5 amp-hour Gates lead acid cells. The experiment is controlled by a

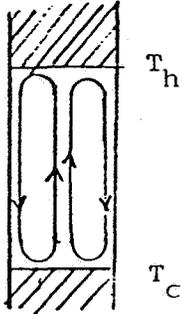


Fig. 1 Thermocapillary flow in a differentially heated free floating cylinder of liquid supported at its ends

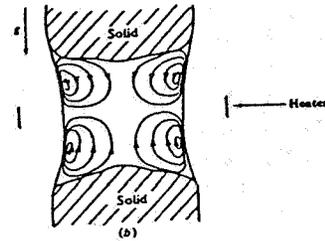


Fig. 2 Thermocapillary flow in a float zone

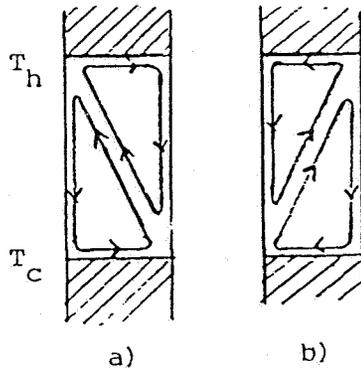


Fig. 3 Oscillatory thermocapillary flow in which the flow pattern oscillates between a) and b)

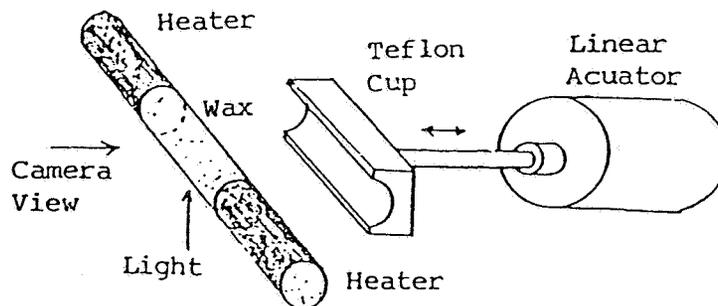


Fig. 4 Schematic representation of experiment

C6502 microprocessor based controller/sequencer with a 16 channel 8 bit A/D converter, 8 bit digital input, 8 digital control outputs, and 32k of onboard EPROM memory for data storage (ref.10).

The basic experiment consists of a solid cylinder of paraffin (length = 1.5 cm diameter = .635 cm) supported and heated differentially at its ends. The cylinder is supported for launch by two teflon cups which are retracted by linear actuators once in orbit. The liquid flows are visualized by means of suspended tracer particles illuminated by a light cut and recorded by an 8mm movie camera (fig. 4). Four different framing rates are used for the camera to allow more pictures during "interesting" periods of the experiment. An LED light in the field of view of the camera is activated during a framing rate change to allow an experiment real time to play back time conversion of the movie film. Ambient and heater/support temperatures, battery voltages, light level, and time are recorded by the controller; frame number is also recorded with the time to allow a correspondence to be made between frame number and temperature. Framing rate and heater power are actively controlled by the controller within time limits on the basis of heater temperature and frame number, e.g. if temperature too low after 2 min., switch to a higher heating rate, e.g. if frame number too large (near end of film), switch to slower framing rate. The use of a smart controller allows more and selective data to be obtained.

Results

The GAS canister was turned on at 34 hours 9 min. mission elapsed time. After an initial half hour "warm up" period two separate experiment runs, with a period of one hour cool down between, were to be executed. Digital data, i.e. temperature, voltage, and light level, was obtained for both runs. Visual data was recorded only for the second run. The failure has since been traced to an intermittently faulty relay which apparently failed to actuate the camera during the first run. Ambient temperature data was recorded for about 35 hours before the controller battery went dead (fig. 5). All the planned data for the second experiment run and ambient temperature data were obtained and so the experiment is considered a success.

For the second run 6 watts of power was applied to heater #2 and heater #1 was left at 0 power. The paraffin melted with the solidliquid interface advancing as anticipated. No boundary layer flow near the solidliquid interface was observed, implying that no significant phase change convection occurred. The total length of the liquid bridge was extracted from the film as a function of time and is given in fig. 6. The total length of paraffin did not melt during the experiment but grew asymptotically to 1.27 cm leaving .23 cm of solid paraffin at the cold end by the end of the experiment. The heater/support temperatures as recorded by the controller are shown in fig. 7 as a function of time. Since paraffin melts at 52 C the temperature at the solidliquid boundary is 52 C; the temperature difference across the liquid length is therefore just the heater temperature

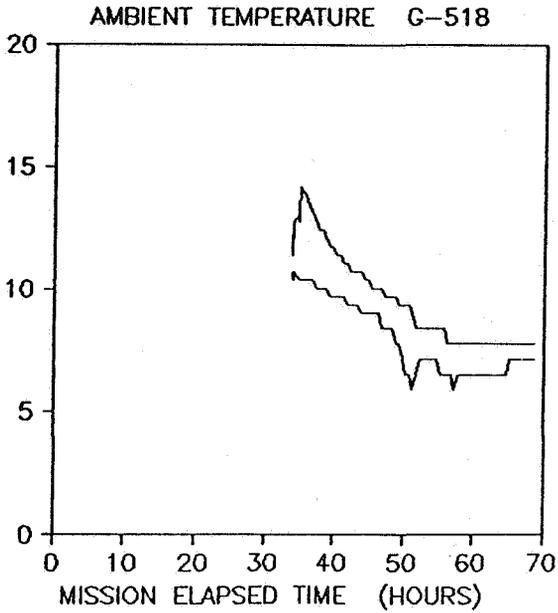


Fig. 5 Ambient temperature
top line - center of experiment
bottom line - outer wall of
experiment

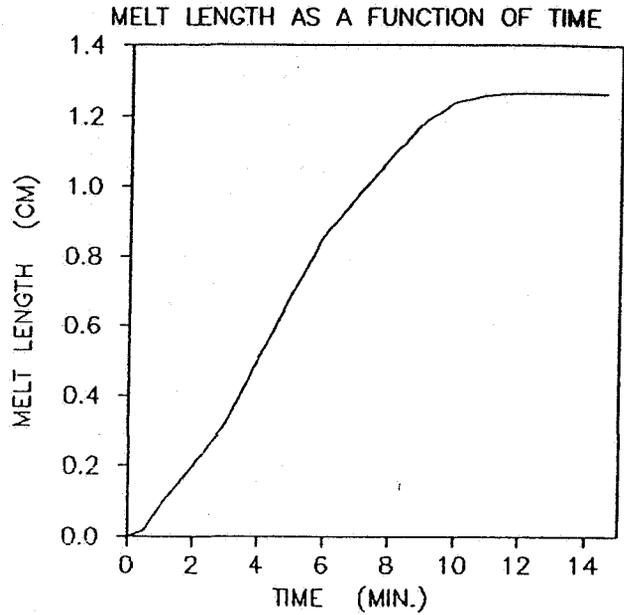


Fig. 6 Melt length as a function of time

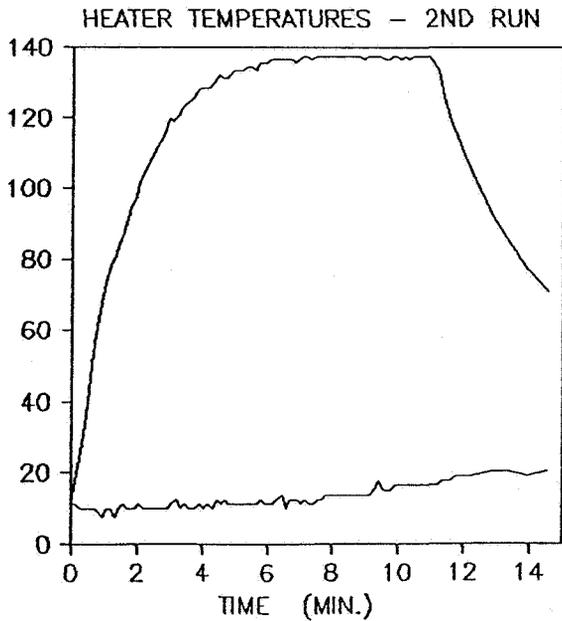


Fig. 7 Heater temperature as a
function of time

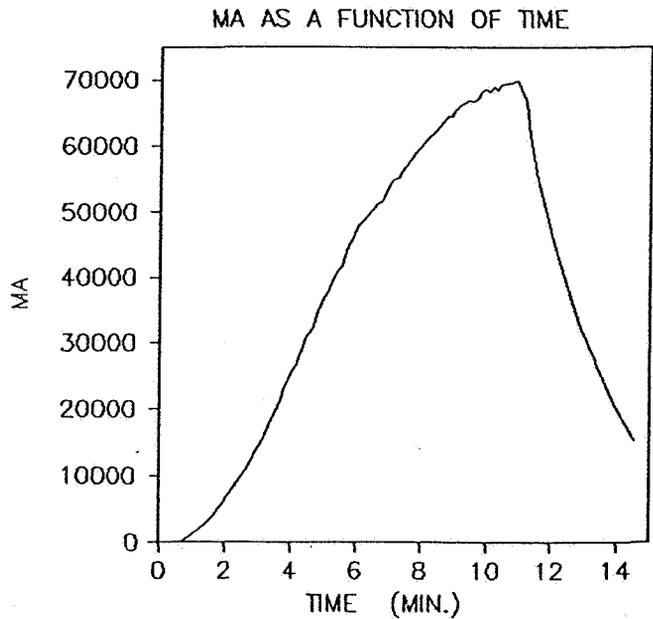


Fig. 8 Marangoni number as a function
of time

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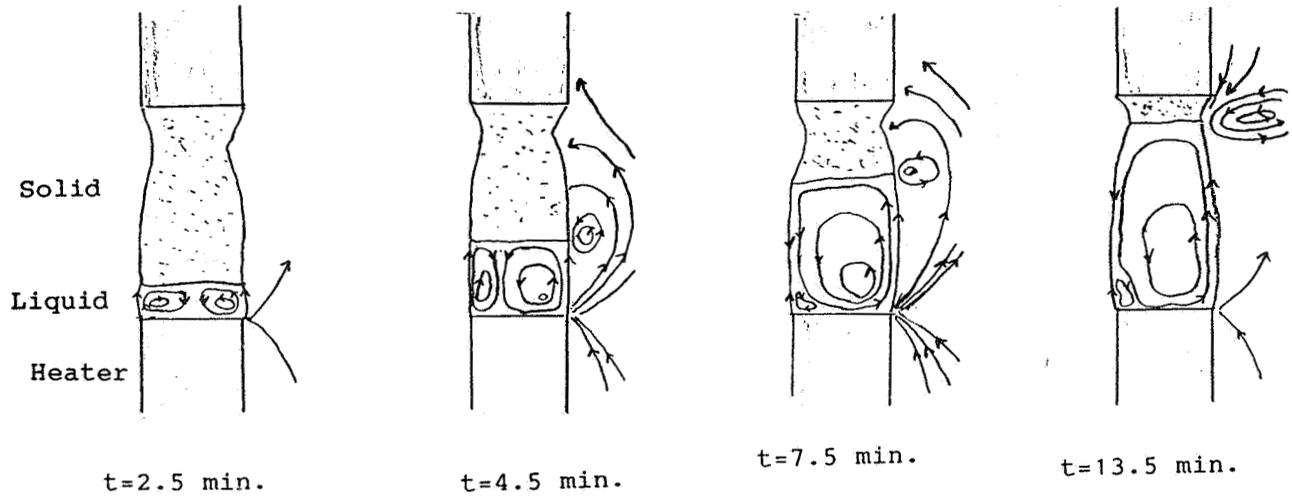


Fig. 9 Schematic of velocity field

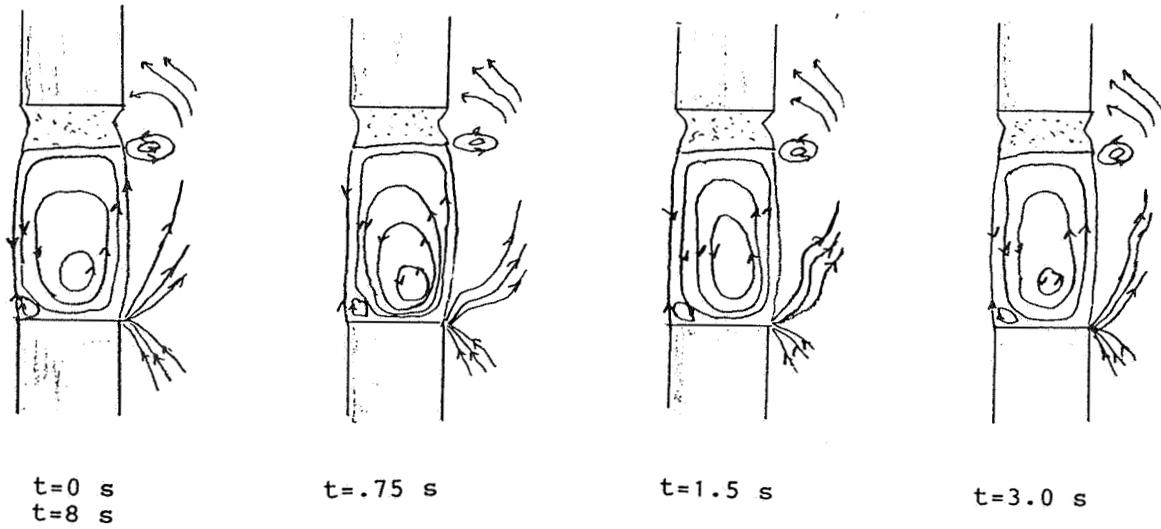


Fig. 10 Planform of oscillations

- 52 C. The temperature difference and liquid length determine the Marangoni number, Ma , which is shown in fig. 8 as a function of time.

The thermistor attached to heater #2 was secured in place before flight with a nylon string. When the heater was activated the nylon melted and produced smoke. This smoke serendipitously acted as tracer particles for convection which unexpectedly occurred in the gas to one side of the liquid column.

A schematic of the velocity field as extracted from the film is shown in fig. 9 for four different times. At about 2 min. there is noticeable convection in the gas to the right of the liquid bridge. The flow moves in from below and right and meets the heater-liquid interface exactly and moves back out away from the liquid. A small vortex also exists at the paraffin solid-liquid interface, rotating in a counter clockwise direction; this vortex tracks the solidliquid interface as it advances. The basic character of the flow in the air remains unchanged throughout the experiment but becomes more intense with time. No flow in the air to the left of the liquid is ever observed. Up to about 2.5 min. the flow within the liquid melt is axisymmetric and as expected for thermocapillary flow, a single symmetric roll with the surface moving towards the cold end. After about 2.5 min. though the longitudinal axis of the roll begins to move to the left until at about 6.5 min. the flow is a lopsided cell with its axis of rotation at the lower right and with a very small return cell at the lower left. This lopsided cell persists for the remainder of the experiment. After the heater is turned off at 13 min. the flow in the liquid and the flow in the gas near the heater subsided fairly quickly, the vortex in the air near the solidliquid interface, which had grown quite large by this time, persisted until at least 14.5 min.

Until 4.75 min. the convection in the liquid and gas was laminar and slowly changing as the liquid bridge grew in length. At 4.75 min., corresponding to a Marangoni number of $Ma=34000$, oscillations (pulsations) occurred in the velocity fields of the liquid and gas simultaneously. The period of the oscillations was about $T=8$ seconds. The oscillations are represented in fig. 10. They occur as a short pulse in the liquid with a wave that travels outward in the gas, followed by a long period of nearly laminar flow in both the liquid and the gas.

b Velocities measured for the flows at 10 min. were as follows; in the liquid at the lower right about 1mm from the surface, .28 cm/s; in the gas at the lower right about 1mm from the surface, .47 cm/s; in the gas in the vortex near the solid-liquid interface about 1mm from surface, .095 cm/s.

Discussion

There are three major results which are surprising and unexpected, the convection in the gas, the lopsided nature of the thermocapillary flow, and the odd (as compared to 1-g) oscillations which occur for $Ma > 34000$. All these are probably related.

First consider the convection in the gas. Continuity implies that for a velocity boundary condition which is large for

a short distance and small elsewhere along a surface (driving force in a small region) fluid flow will move from large distances into the driving region and back out as shown in fig. 11. Since the flow in the air is of this nature and meets the liquid-heater interface exactly it is certainly being driven by something in this region. An analysis of the region near the heater-liquid interface indicates that strong thermal and velocity gradients exist along the liquid surface. First consider the thermal transport from heater to liquid. The liquid velocity must be zero at the heater wall. This implies that at the wall only conduction can take heat from the heater. For near steady conditions this implies that the ratio of temperature gradient in the liquid to that in the heater will be equal to the ratio of thermal conductivities, about 1000 ! The temperature gradient will be very strong in the liquid near the wall and drop off rapidly due to convective heat transport. This of course causes a strong surface tension gradient which implies the velocity will be very high some short distance from the wall (fig. 12). The sharp boundary layers have been noticed previously in numerical simulations (ref. 11).

Given the existence of these sharp boundary layers at the driving region (liquid-heater interface) the gaseous convection must be caused by one or both of these boundary layers. The vortex in the gas would then be caused by similar boundary layers at the solid-liquid interface. The problem is then to identify what type of convection is making use of the boundary layers provided by thermocapillary flow and driving convection in the gas.

Preliminary analysis of the convection in the gas has been completed (ref. 9) and will be presented elsewhere. The most promising driving forces are coupled thermocapillary / thermal expansion convection and microgravity buoyancy convection. Coupled thermocapillary / thermal expansion convection would occur in the following way; a small amount of gas is pulled into the liquid-heater interface by thermocapillary flow as in fig. 12, the gas is heated very quickly and expands driving it outward. A simple 1-dimensional analytical solution has been obtained of the compressible Navier-Stokes equations for this model and a small flow is possible. This type of convection can not explain the vortex at the solid-liquid interface.

More promising is microgravity buoyancy convection which occurs as follows; a small residual gravity field exists and points approximately to the left and down in fig. 9. Gas near the thermal boundary layers is heated and "rises" to the right and up. The flow pattern observed in the air fits this explanation qualitatively, in addition an analysis of a gas "falling" through the thermal boundary layers expected with a $g =$ observed in the experiment. The geometric stability of the liquid bridge (ref.) implies $g < .009 g$ earth; the centripetal acceleration about the Shuttle center of mass is probably at least $g = .0001 g$ earth. So the limits on the g environment are consistent with the g implied by the order of magnitude analysis.

The lopsided nature of the flows is easily explained if there is a residual g pointing to the left and down. The flow in the liquid would "rise" to the right and a lopsided cell of the

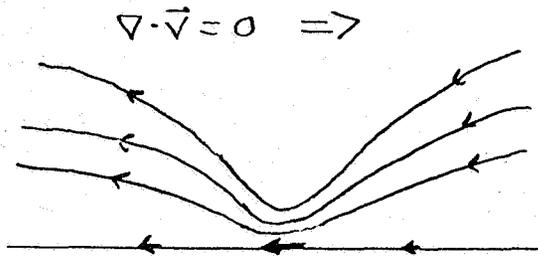


Fig. 11 Continuity effect on confined driving force

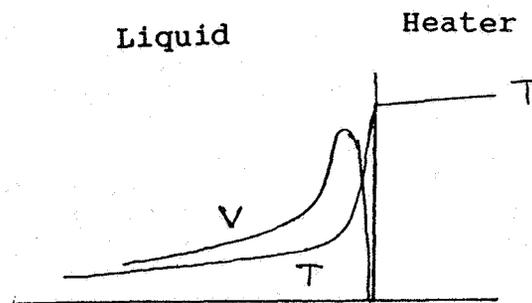


Fig. 12 Thermal and velocity boundary layers at the heater-liquid interface

type observed would form. In addition flow in the air would only be expected on the "high" side (right side of the cylinder). The lopsided flow patterns could also be explained if smoke was only introduced on the right side and if the smoke seriously affected the surface tension. The surface tension gradient would then be non-axi-symmetric and the observed flows could arise.

The peculiar oscillations which occurred could simply be do the influence of the convection in the gas which was apparently strongly coupled to the thermocapillary flow. The Shuttle was in an Earth viewing attitude during the experiment and vernier thrusters may have been in use in a pulsed mode. If the gaseous convection was driven by microgravity the the short pulses from the thrusters may have caused the oscillations. The thruster and/or pulse period have not been defined.

Further analysis of the convection in the gas is being pursued and a 2-dimensional numerical model is being developed in which effects can be added or subtracted so as to pin point the cause of the gaseous convection. A reflight is also planned to further investigate these phenomena and to see how repeatable the results are.

Conclusions

Thermocapillary flow and gaseous convection in microgravity have been observed and investigated. An oscillatory instability at $Ma=34000$ was observed in both flows. The oscillations were strongly coupled. The gaseous convection could be driven by microgravity convection or coupled thermocapillary flow and thermal expansion convection. The gaseous convection had a strong influence on the thermocapillary flow and warrants further investigation.

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