CLEMENTINE: FUTURE SPACE TECHNOLOGY..TODAY

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"Advanced technologies", "dual use", "commercial parts", "Faster, Cheaper, and Better"—if you're a spacecraft manufacturer, these are words to live (or die) by in the 90's. Everyone is using them, but what do they really mean? What would a satellite that employed these ideas look like? in January 1994, in an attempt to answer these questions, the Clementine spacecraft was launched to the Moon and the subsequent results were only a little short of amazing. Clementine, arguably the least expensive, most advanced, and shortest scheduled mission flown in a decade, was planned as a joint DoD/NASA mission to evaluate advanced military technologies and demonstrate the applicability of those technologies for civilian scientific uses. In the process, it took almost two million pictures of the Moon in a little over two months. At the same time, it flight tested nearly two dozen advanced technologies and opened up a new era in space microelectronics.

What did Clementine (also called the Deep Space Program Science Experiment, DSPSE) accomplish technically? In terms of its creators, it was "a desktop computer hooked up to some camcorders and a mobile phone". Of particular interest to the microelectronics industry are three aspects of this next generation desktop computer in space: the electronic systems and sensors that made up the basic spacecraft package, their performance, and the parts selection procedures employed. Several of the systems (Table 1) represent the most advanced prototypes ever flown in space (see Rustan, 1994a, 1995; Nozette and Garrett, 1994). They are of much more general interest, however, because they were based in many cases on so-called commercial (or "plastic") parts and "commercial off-the-shelf' (COTS) systems. For the first time, a program actively encouraged the utilization of such components- –a bold, innovative step that has opened up a whole new range of capabilities for the spacecraft designer. The success of the systems (not one significant hardware failure occurred during the mission) amply rewarded the perceived risks taken by the program.

What did Clementine accomplish scientifically? Clementine produced the first high fidelity photometric survey of an extraterrestrial body. Based on Clementine, we know more about the geology of the Moon's surface than any other extraterrestrial body (see special issue of <u>Science</u> for a comprehensive review of the scientific results--Nozette et al., 1994, and associated papers therein). This survey is enabling a truly global mapping of the lunar rock types and, for the first time, of the geology of the lunar polar caps and farside. The spacecraft measured one of the deepest basins in the Solar System, the -12-14 km deep South Pole Aitkin Basin (Figure 1), and showed that the lunar surface may have had volcanic activity as recent as a billion years ago. And, on the basis of a serendipitous experiment, Clementine may have discovered evidence for water ice at the lunar poles—a potentially important step towards colonizing the Moon (Nozette et al., 1994).

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The Spacecraft

The Clementine spacecraft was an unusual engineering achievement as it was taken from concept to launch by the Naval Research Laboratory in only 22 months (Rustan, 1994b). The final dry mass of the spacecraft was only 508 lbs. with an additional 420 lbs. of fuel added to accomplish the mission--both quite low for a mission of this complexity. As the weight of a spacecraft and the schedule duration often directly translate into cost, these were major drivers in keeping the total cost below a remarkable $80M ($20M for the launch plus $60M for the spacecraft and operations)! For most of the mission, the spacecraft was three-axis stabilized and deployed as shown in Figure 2. Clementine operated as a remote telecamera in this mode with a communications downlink rate of 128 kbps and 1 kbps command uplink. The four primary telecameras or sensors, integrated by the Lawrence Livermore National Laboratory, were capable of imaging the Moon and Earth over a wide spectrum in the ultraviolet and visible, the near and far infrared, and at high resolution. The latter, high resolution camera, was also part of a sophisticated Laser imaging, Detection, and Ranging System (LIDAR) which included a compact laser transmitter developed by McDonnell Douglas Corporation. These sensors together returned more than 1.8 million digital images and succeeded for the first time in mapping the entire 38 million square kilometer surface of the Moon during the 71 day lunar phase of the mission. Their combined weight was only 8 kg (a little heavier than a bowling ball) and required less power than a 75 W light bulb to operate! In the spirit of a true advanced technology demonstration and to further reduce the weight, the spacecraft incorporated several novel, light weight electronic attitude control and power systems: two CCD-based imaging Star Tracker cameras, two light weight inertial measurement units (IMUS), ultra-thin, lightweight gallium arsenide on germanium (GaAs/Ge) solar arrays, lightweight reaction wheels, and a nickel hydrogen (NiH_2) common pressure vessel (CPV) battery. The sheer rate of images being returned by the mission (a peak aggregate data collection rate of 15.7 Mbps) demanded the use of advanced electronic processing systems: an R-3081 32-bit microprocessor, a new CMOS-based image compression device, and a 2 Gbit Solid State Data Recorder (SSDR). Even the release mechanisms and the composite-based structures used were "first timers" in space. Overall, the Clementine spacecraft was a test bed for 23 newly-developed advanced lightweight technologies.

The Sensors

At the heart of this electronic cornucopia were the four spacecraft sensors—the UV/Visible (UV/Vis) Camera, the Near IR Camera, the Longwave IR Camera, and the High Resolution Camera (part of the Clementine LIDAR)—and the two Star Tracker Cameras. These systems, originally developed by the DoD for ballistic missile defense, represented a significant reduction in weight and power over earlier systems of similar capability. This was accomplished by using both novel design approaches and the latest focal plane and microelectronic components. Approximately 50% of the diodes, 80% of the transistors, and 60% of the integrated circuits that made up these systems were commercial parts. An example of the potential savings that this provides is the UV/Vis Camera. It weighed in at less than 0.5 kg (Figure 3) whereas previous space-qualified cameras for the same wavelength range weighed more than 50 kg. In operation, it used only ~5 W for imaging and 11 W when changing filters as compared with 30-40 W for previous systems covering the same wavelength range. As a practical matter, manufacturing efficiencies were attained by employing common electronic modules for the

U V/Vis, Star Tracker, and HiRes cameras and filter wheel assemblies. The base electronic modules, incorporating a Thomson Si Charge Coupled Device (CCD) and state of the art flex board electronics, were approximate] y the size of a pack of cigarettes. likewise, the two IR cameras had common integral dewar/cooler assemblies and imaging electronics with built-in power conditioning. They both incorporated state-of-[he-art Ricer K506B integral Stirling cyrocoolers capable of maintaining the optics at a very cool -65°-700 K.

As developed by LLNL, the Clementine UV/Vis Camera consisted of 3 modules: the optics module, the filter wheel, and the camera assembl y. The core focal plane array was a phosphor overcoated silicon CCD with UV and visible response between 0.3 and 1.1 pm. The weight, power requirements, filters, and format for this camera arc listed in Table 2. The optics were a catadioptric design with an aperture of 46 mm and a speed of F/1.96. The UV/VIS electronics operated at a maximum rate of 30 frames per second with an 8 bit analog to digital converter.

The High Resolution Camera (HRC), which also doubled as the receiver for the LIDAR, was developed by LLNL and the Optical Corporation of America. It consisted of an intensified Thomson S20 photocathode silicon CCD coupled to a microchannel plate image intensifier (see Table 2). The optics were a 13.1 cm aperture Cassegrain telescope with reimaging lens and beam splitter. When used as part of the. L] DAR, the return signal from the laser transmitter was directed to an avalanche photodiode by the telescope's beam splitter. The time difference between the laser firing and signal return provided range informat ion—the minimum range detection was 240 m and the maximum range detection was 640 km with a resolution of 40 m. The laser transmitter, which was designed and built by McDonnell Douglas in St. Louis, MO, had a mass of 1.1 kg and size of 4 x 10 x 22 cm. It was highly portable with many applications in the commercial and military sector. It consisted of a Nd:YAG diode pumped with a pulse energy of 180 millijoules (90% at 1064 nm; 10% at 532 nm) and a 10 ns pulse length. Although it could be operated continuously at 1 Hz or in bursts at a higher repetition rate, the laser was operated for 400 pulses at 8 Hz during the mission. The LIDAR proved to be a valuable tool in determining the topography of the Moon and provided accurate measurements of the deep impact basins on the Moon (Nozette et al., 1994).

By replacing the optics module and using similar camera assembly modules to the UV/Vis and HRC, LLNL was able to efficiently develop two ultra-miniature Star Trackers. Despite the small size of these Clementine sensors, they could perform as well as their larger cousins. Indeed, the two Clementine Star Tracker Cameras were capable of a 3-axis attitude determination with only one star field image. The wide field of view of the cameras, 28.4° by 43.2° (Table 2), allowed the. star catalog to be substantial y reduced over previous systems because several of the brightest stars would be guaranteed to appear in any frame. The Star Tracker Camera electronics operated at a maximum rate of 10 frames per second with 8 bits resolution in the analog/digital converter— -at such rates, the overall attitude determination rate (at less than 1 second) is 10x to 1 00x faster than previous systems. In operation, the Star Trackers proved to be not only excellent for determining spacecraft at lit ude but valuable imagers in their own right and took some of the more spectacular (though less "scientifically" valuable) images such as the sunrise in Figure 4.

The Near IR and Longwave IR Cameras, developed by the LLNL and Amber Engineering, inc., were designed to detect and track in the IR range (see Table 2 for filter wavelengths). The Near IR Camera optics are catadioptric with a 2 cm aperture. The camera electronics operate at a rate of 10 frames per second. The Longwave IR optics are a Cassegrain telescope with relay optics at an aperture of 13.1 cm. The electronics operate at a rate of 30 frames per second. Of particular interest for systems designers, these two sensors used an active, electromechanical cooling s ystem-- --t wo active Ricer K506B Stirling cycle cryocoolers with Hybrid-10 control electronics (to commutate the motor and control the focal plane array (FPA) temperature) were built as an integral part of both IR cameras----ancl marks the first use of these active cryocoolers in the space environment. Each cooler was part of a dewar-cooler assembly with the FPA mounted inside the dewar and the dcwar welded to the cooler. They were intended to extract 0.25 watts from the cameras' focal planes and maintain the focal planc arrays at their desired operating temperatures and within $\pm$0.5° K for the 7 month life of the mission. The NIR cryocooler operated successfully for a total of 840 hours with 360 on/off cycles. The H-10 electronics provided good temperature control with a typical FPA temperature of 70.3° K and a one-sigma variation of 0.46 K (the target was 70° K). The thermal environment of the cooler ranged from -23.7° C to 38.2° C. NIR cooldown times were typically less than 9 minutes. The LWIR unit ran for at total of 670 hours with 370 on/off cycles. Unfortunately, the H-10 electronics did not provide as good temperature control for the LWI R as for the NIR. This inability to operate at a fixed set point was believed to be. caused by a ground difference bet ween the H-10 controller electronics and the FPA temperature sensor diode. The LWIR FPA temperature during imaging varied from 66.9° K to over 85° K (the target was 65° K). Results for the two coolers are compared in Figure 5.

The Electrical Power System

The Clementine Electrical Power System, in keeping with the spirit of technological innovation, consisted of several new devices --specifically, the bat tery and the solar arrays. The battery was a unique common pressure vessel (('PV), nickel hydrogen (NiH_2) battery design and, even though nickel hydrogen batteries have been flown as an independent pressure vessel (IPV) since 1976, a more compact, higher energy per kilogram CPV nickel h ydrogen battery had not been flown previously. Batteries represent a major weight driver for spacecraft and this 9.5 kg, 15 Amp-hour (22 cell), 30$\pm$6 Vdc battery provided 47.1 W-hr/kg or about 80% more than previously flown NiH_2 IPV batteries. The battery, which is 12.7 cm in diameter and 32 cm long, employs a back-to-back nickel electrode configuration with three modules per cell and a potassium hydroxide electrolyte (Figure 6). It was developed under a Cooperative Research and Development Agreement (CRADA) between NRL and Johnson Controls Battery Group, inc. (now part of the Eagle-Picher Corp.). An excellent feat ure of the battery is that the charge is proportional to its internal pressure permitting the usc of a battery pressure transducer to independently determine the charge on the battery-this proved to be a very useful in-flight check of the charge level (Figure 7). Despite unanticipated discharges of the battery early in the mission (this could have caused bat tery failure in less advanced designs), the NiH_2 CPV performed perfectl y throughout the missi o]l—another testimony to the potential value (and unexpected benefits!) of the new technologies now becoming available. As a result of its performance on Clementine, this battery is already being integrated into other new spacecraft (i.e., IRIDIUM[TM/SM]).

The power system also incorporated an innovative solar array technology-a 5.5 mil (O. 14 mm) thick gallium arsenide on germanium (GaAs/Ge) solar cell array. The solar array had a peak capability of 280 watt/m^2 and was the thinnest GaAs/Ge solar cell array yet flown in space. Clementine had a total solar array cell area of 2 m^2 (1,2484 cm x 4 cm cells) with an additional 1.21 m^2 (95 W @ 30 Vdc) on the Interstage Adapter (ISA). To further reduce the array mass, the panels on which the solar cells were mounted were made up of composite lightweight graphite epoxy/aluminum honeycomb core substrates. The combination provided an array specific power of 53 W/kg. The spacecraft had two "wings" of solar arrays that were independently gimbaled in a single axis and could track the Sun autonomously. During low Earth orbit (LEO) and the lunar transfer orbit burn, the solar arrays were stowed and provided only 160 to 268 W of power (the ISA array was necessary to provide the additional power requ i red to operate the spacecraft) . After solar array deployment, they provided up to 400 W @ 33 Vdc/28° C (-75 W each for the inboard panels and -105 W each for the outboard panels)— well within planned margins. Although the arrays experienced over 330 thermal cycles from -1 60° C to +60° C while in orbit at the Moon, there had been no observable degradat ion fron ı radiation or other causes in these arrays after one year of operat ion.

The Attitude Control System

Clementine had a wide range of attitude control requirements---ranging from 3° during the spin-stabilized low Earth orbit phase to 0.03° for the. 3-axis stabilized image data collection phase. 10 achieve this, the spacecraft employed a combination of the Star Trackers and inertial measurement units (lMU). Again, however, to make the most use of the Clementine "technology test platform", several previously untried but revolutionary components were incorporated. These included two of the lightest IMUs ever flown. These were a 0.5 kg ring laser gyroscope (RLG) provided by 1 Ioneywell, inc. (a Lightweight Advanced Implementation Technology (1.Ail) IMU GG1308 with Sundstrand RA-500 Accelerometers) and a 0.65 kg interferometric fiber optic gyro (IFOG) provided by Litton Guidance and Control Systems, inc. (a Litton LN200 IMU with Silicon Accelerometers) (Figure 8). These 1 MUS incorporated lightweight packaging and applications specific integrated circuits (ASIC) for a significant weight reduction over existing IMUs. The sizes of the units were comparable: 7.7 cm dia x 12.7 cm and 8.9 cm dia x 7.9 cm for the RI .G and IFOG respectively. The units were radiation hardened to about 10 krads (Si) and were not susceptible to destructive latchup. Both units demonstrated a drift rate of about one degree pcr hour and operated at a power of -10 watts. The RLG had a random walk of O. 125°/square-root-hr while the I FOG was 0.0050/square-root -hr. Whereas both units provided data at 400 Hz, for Clementine, the lMU data were taken at 100 H7.. Although this undersampling increased the noise for both IMUS, both easily met the mission requirements. Indeed, despite a fcw minor operational upsets, the I M Us performed extreme] y well throughout the mission and should, based on the Clementine results, be space qualified for future spacecraft.

Another "first time in space" electromechanical Attitude Control System component was a lightweight reaction wheel developed by Ball Aerospace Corporation for line pointing and for low acceleration slews. Four wheels, each consisting of a flywheel, the case, and the internal electronics, were employed. Unique to these reactions wheels, the drive electronics were integrated with the wheel assembly to provide a 30% weight reduction and an increase in reliability. The Clementine reaction wheels each weighed only 2 kg and used about 10 watts of power. They provided ± 2 Nms @ 2500 rpm and were mounted such that three arc orthogonal and a fourth is a skewed backup.

Processor Power for the Future

Clementine had many of the most sophisticated microelectronics processing components ever flown in space. Ranging from a rad-hard 1750 architecture main processor to the state of the art R3081 32-bit image processor and Matra JPEG compression chip set, the Clementine spacecraft made the most of modern processor power. Indeed, Clementine was the first spaceflight of the Honeywell, Inc. Ml[.-STD-1750A microprocessor. This 16 bit (1.67 M IP/0.7 MFLOPS) processor was used for standard spacecraft housekeeping operations and was part of the DSPSE Spacecraft Controller (DSC). This main housekeeping processor (HKP) was an advanced Very High Speed Integrated Circuit (VHSIC) device fabricated using a "stretch" electronics module, type 'E' (SEM-E). With a mass of 0.5 kg and volume of only 575 cm^3, it was responsible for the spacecraft command and telemetry, housekeeping, and Star Tracker image processing. The ASC 1750 processor, which supported 'C' and 'ADA', featured a 64 Kword start up ROM, 256 Kwords local memory, single error detection and correction, double error detection, and processor reset conditions. For reference, there were seven user interrupts and eight input/output discretes. Maximum power consumption for the HKP was 5 W at 9 MHz. There were no known in-flight hardware problems (the SEU rate was estimated to be <10-7 bit upsets per day). The HKP did, however, have several resets during the, mission attributed to software problems (indeed one of these upsets has been blamed for the loss of contact with the spacecraft that led to a loss of all the attitude control fuel). In addition to the HKP, another 1750, the UTMC69-R000 (based on the rad-hard, >100 krad, UT1 750 AR) was used as part of the Data Handling Unit (DHU) for data formatting and sequencing. It controlled the chipset for image compression and the optical sensors and was operated at 8 MIPS in RISC mode.

A major microelectronic advance demonstrated on Clementine was an R3081 commercial 32-bit RISC processor developed by Telenetics, inc. that was used as the primary image processing system. Implemented with Field Programmable Gate Arrays (FPGAs), the logic represented 10:1 packaging density gain over SSI/LSI logic. The Clementine Sensor image Processor (SIP) utilized the 20 MHz microprocessor for image processing and as a "backup" to the HKP. This was the first application of a 32-bit RISC processor architecture in a spacecraft, The Clementine IDT 3081 (R3000 ISA) processor was capable of 18 MIPS and 3.5 MFLOPS and had a software selectable, multi-speed clock of 20, 10, and 5 h4Hz with 18, 9, and 4.5 integer MIPS respectively. The CPU, floating point unit (FPU), and cache were integrated on a single chip. The floating point processor had single (32-bit) and double (64-bit) precision acid, subtract, multiply and divide at 3.5, 1.7 and 0.8 MFLOPS. It used both a 16 Kbyte instruction cache and a 4 Kbyte data cache with a 4 deep read/write buffer with a memory management unit. There were 5 available direct interrupts with a multiplexed address/data bus. The 40 bit wide internal memory provided 512 Kwords of EDAC SRAM while the 40 bit wide EDAC PROM was divided into 256 Kwords of E^2 1'ROM. The PROMs were power strobed to conserve power. The sensor interface provided dual SASI bus and dual multiplexed camera data interfaces to a 1 Mbyte image storage buffer. The internal timing was provided by a timer to generate 125 μs resolution for backup time tagging and watchdog timeout generation. The watchdog timer used triple timeout to detect latchup—the first timeout interrupted the processor, the second reset the processor, and the third requested SIP power to be turned off. TheR3081 met Mil-Std-883 Class B screening and was radiation hardened to about 20 krads (Si)—-thc unit was tested with >25 Krads total dose with no unit failure. The quality and quantity of images returned by Clementine amply attest to the outstanding success of this processor.

In addition to the principle microprocessors, Clementine employed an advanced CMOS ASIC-based data compression circuit developed by Matra Marconi Space of France. The ASICs used were the ASP17 (29 C80F) 8x8 Discrete Cosine Transform (DCT) Processor and the ASP16 Compressor/Formatter (COFO). These had a simple pipeline hardware architecture (1 μm gate arrays) that employed single pass coding and used an efficient 1 SO Joint Photographic Expert Group (JPEG) algorithm wit 1 ı a compression range of from 4 to 20. The MATRA MI IS chip sets were radiation tolerant to 40 Krads and were low power at 700 mW/(Mpixels/s) maximum. Physically, the devices were a small (<6000 mm^2), 132 pin CERQUAD flat pack design. They were latchup resistant with a cross section of less than 3x10-7 cm^{-2} at 116 $MeV\text{-}mg^{-1}\text{-}cm^2$. The SEU cross section was 10^{-6} cm^2/bit at 60 $MeV\text{-}mg^{-1}\text{-}cm^2$. The parts were high reliability, high quality having been screened using 883C Class B, SCC9000 Level B criteria. The design was also fast (4 Mpixels/s at 14 MHz) and flexible with the algorithmic and transmission format parameters stored in programmable memory.

Although the Matra configuration did not permit quantization values less than four, it provided four image compression options. Tables supported compression ratios of 4:1 (for lunar images and stars), 8-10:1 (for lunar pictures taken with the Hi-Res camera), and 16:1 JPEG. The four algorithm modes were stored in four 512 byte pages in a programmable memory (these could be the JPEG standard, a user defined version, Y, or color). in-flight results indicated that the on-board compression was identical to the on-ground compression simulations. By camera, the findings and recommendations of the imaging team were: UV/VIS (compression ratio 6.0) adequate, no change; NIR (compression ratio 2.2) adequate, no change; LWIR (compression ratio 1.6) adequate, no change; H RC (compression ratio 1 to 10) change quantization matrix; Star Tracker (compression 12 to 30) change 1 luffman table (note: only the HRC was operated in JPEG mode). An average compression ratio for the mission was 5.5.

Memori es...

The on-board image compression proved to be extremely valuable and yielded high quality results throughout the mission. The nearly two million images in 2 months would have swamped the data handling/storage capabilities of earlier spacecraft. Fortunately, another new electronic technology was available to solve this data saturation problem—a next generation, solid state data recorder (SSDR). The Clementine SSDR, developed by SEAKR, inc., was more than up to the challenge. It had over 2 Gbits of usable storage capacity (actually 2.9 Gbits of which 786 Mbits wei e for error detection and correction (EDAC))—four times the capacity of any previous flight qualified SSDR. The novel design incorporated redundant EDAC with active fault management and built-in test capabilities. The recorder employed commercially available 4 MB x 1 Hitachi Dynamic Random Access Memories (DRAM) and had a data thı oughput greater than 20 Mb/s with a bit error rate of less than 1 part in 10 billion. The weight (volume,) of the SSDR was 4.1 kg (2,048 cm^3) and it used a maximum of 15 W power. The ASIC used in the design provided a 5:1 power/area and cost savings over programmable logic arrays used in other designs.

The SSDR was evaluated during the very demanding mission timeline that included passages through the Earth's radiation belts and through a solar proton event. Operationally, to limit the effects of single event effects duc to high energy radiation, the SSDR memory was continually scrubbed (checked for bit errors) at 1.3 to 1.5 Mbytes/s and error detection and correction applied (EDAC)—single bit errors were corrected and counted while double bit errors were only counted. The single event upset (SEU) count was kept as a part of the real time telemetry (the memory address of' the upsets could be determined and used to create an upset memory map but this was not done in flight). As illustrated in Figure 9, over 7500 single bit errors were detected and corrected (no double bit errors were observed) by the SSDR EDAC software. Amazingly, not one SEU slipped by and the SSDR performed flawlessly throughout the mission. The SSDR was capable of mapping around problem areas or damaged memory by ground command if it detected non-correctable single bit errors (this turned out not to be necessary in flight). Although the actual memory size was over 2 Gbits of usable memory, the mission storage requirement was ~1.6 Gbits leaving -400 Mbits for replacement of such damaged memory.

Of Frangibolts™ and Composites

In addition to advanced electronic systems, Clementine tested several electromechanical and mechanical assemblies. Although not strictly an electronic technology, the devices used to release the Clementine solar arrays arc an example of the innovative electromechanical technologies test cd. The solar array deployment system employed two pairs of Frangibolt s™ developed by 'TiNi Corporal ion to release the two solar pane] wings. These were lightweight, low-shock, non-pyrotechnic release devices (typically, spacecraft use pyrotechnic devices called squibs). The bolts, which retained the solar arrays during launch, contained a shape memory alloy held within their grip length. When the alloy was heated (using two 100 W heaters for 90 seconds), the material expanded to its original shape breaking the bolt. Since. the actuation mechanism contained no moving parts, the Fran gibolts™ were particular y useful in space. B y eliminating the dangers associated with pyrotechnic devices, the Frangibolt™ provided safe handling, allowed pre-flight and in situ testing, were insensitivity to transient electrical signals, avoided fragmentation and other explosion by-products, and were relatively inexpensive. They allowed a -40% reduction over conventional pyrotechnic ordnance and harness. Clementine was the first operational space use of these Frangibolts™.

An important mechanical structure for any mission is that connecting the spacecraft to the last stage of the booster. In the case of Clementine, this structure, the "interstage adapter" or ISA, had to carry launch and kick motor loads from an aft attachment ring to a forward ring that was mated with the spacecraft primary structure. The ISA structure was a lightweight g,raphite-epoxy composite, conical shell structure (8.8 kg) with aluminum forward and aft rings. It and the main Clementine lightweight solar array structures were based on advanced materials and structures technologies developed under the cognizance of the USAF Phillips Laboratory by McDonnell Douglas Corporation and Composite Optics. The ISA'S total mass (excluding electronics and SRM components) was 47,0 kg and represented a mass savings of 8.3 kg over a normal aluminum ISA. In fact, sufficient mass was saved to allow the ISA to carry engineering instrumentat ion for the Clementine Interstage Adapter Satellite (ISAS). This additional experimental platform, described in the next section, was another unexpected bonus of the usc of advanced technologies!

The Engineering Experiments

Spacecraft electronic systems must survive and operate reliably in intense space environments for long periods of time. in addition to radiation, temperature variations, spacecraft charging, ,surface contamination, and mitt ometeoroicls/space debris all present environmental challenges to the long term survivability of a spacecraft. To obtain data on specific microelectronic and sensor systems in the space environment, several engineering experiments were specially developed for the Clementine spacecraft and the Clementine ISAS. The latter was the result of a series of serendipitous events. Early planning for the Clementine mission indicated the requirement for a lunar transfer booster. however, for stabilization it was required that the spacecraft and its SRM be spin stabilized during the burn. This in turn required that the Clementine solar panels be stowed. As the solar arrays provided too little power in the stowed position, it was necessary to wrap additional solar cells around the ISA ring structure to make up the difference during this time period. The Clementine program team realized that even though this module would be left behind in Earth orbit, it would still be a valuable resource as -100 W of power were available. To utilize this capability, the ISA was fitted with a small instrument pallet (total weight -5 Kg) that fit on the inside adapter wall above the SRM. From this pallet grew the concept of the ISAS for studying the effects of the Earth's space environment on individual Clementine components. The ISAS was left in a -400 km x --120,000 km, 67° inclination orbit for about 100 days following Clementine's lunar injection permitting it to repeatedly sample the near Earth environment. As shown in Figure 10, the ISAS consisted of the interstate vehicle and its solar arrays, the 1 SAS engineering experiment package, and a simple communications package (the vehicle had no attitude control or determination and was left in a random orientation).

Table 3 lists the instruments and devices under test (DUTs) flown as part of the Clementine engineering program, The majority of the radiation instruments are physically grouped in terms of two major packages—the Jet Propulsion Laboratory Radiation and Reliability Assurance Experiment (JPL RRELAX) on the Clementine spacecraft and the ISAS and the NRL Data Acquisition System (DAS) on the IS AS. in addition to these packages and the other instruments in the table, data from the Clementine optical instruments, its computer systems, IMUs, and housekeeping functions arc available for specific studies. The two Clementine vehicles have created a unique opportunity to simultaneously measure the effects of the space environment on identical or similar electronic systems and components under substantially different conditions. The Clementine engineering results are being made available to the spacecraft engineering community for future design and testing of advanced microelectronic systems.

Onc additional point is that the Clementine spacecraft design deviated from the normal practice of using military standard components and, instead, introduced commercial plastic parts which were not hermetically sealed. Most of the commercial plastic parts were screened, however, to Military Specification 883B ancl selected so as to prevent destructive latchup and to operate in the mission space environment for a full year. The importance of these components to the microelectronics community is that they forced the requirement that the effects of the space environment on their operation be carefully evaluated before Clementine was launched. The validation of their performance and the ground testing in turn needed to be determined so that future missions could learn from the Clementine model. Universally, the performance of all parts met or exceeded our expectations.

An ultra-compact, microelectronic radiation monitor and device test facility was developed for the mission. Weighing only 624 g but containing over 160 test devices, the JPL RRELAX was flown on the Clementine spacecraft and the IS AS to evaluate the effects of radiation and the thermal environments on microelectronics. The RRELAX contained a miniature Radiation Monitor (RADMON) to record the total dose and SEU environments necessary to calibrate the engineering measurements, profile the Earth's trapped proton environment, and characterize the solar proton and Cosmic Ray fluxes; a Linear Charge Coupled Device (LCCD) to study the effects of space radiation on CCDs; and sample CMOS and inverter devices to determine the damage coefficients in situ for Honeywell RICMOS-III (radiation hard) and MOSIS (radiation soft) CMOS test structures and the effects of feature size— 1.2 μ and 0.8 pm, respectively-on total dose effects. The use of wire resistor test structures for "on-chip" temperature measurement was also tested. The RRELAX was flown on both the Clementine and ISAS behind minimal shielding (-1 mil and -7 roils). The RADMON used an inexpensive, low power SRAM as an SEU monitor and a pFET dosimeter as a low-cost, low power total dose detector. The CCD experiment, on the other hand, made repeated measurements of the charge transfer efficiency (CTE), dark current (I_{dark}), and flat band voltage shift (V_{fb}) of a linear CCD (LCCD). Comparisons with ground tests and theoretical models will allow the evaluation of techniques for extrapolating laboratory data to the space environment and accurately predicting in-flight degradation of CCDs. Radiation effects on the CMOS test structures are currently being used to establish the limits for using radiation-soft quick-fab CMOS for digital logic in spacecraft and to establish inverter parameter spreads. Figure 11 compares the proton upset rates measured on the Clementine and ISAS RRELAX experiments for the 1.2 μ and 0.8 μm devices during a solar proton event on 21 February 1994 (note: solar proton events are often identified as "solar flares" though they are not technically the same thing). This dual measurement is another unique "first" for Clementine. Such data will aid system designers in designing optimized fault tolerant systems and allow the characterization of FET and inverter array threshold voltage shifts in space which will be compared to ground based QML flight qualification tests.

In addition to the RRELAX, NRL instrumented both the Clementine spacecraft and the ISAS with PMOS semiconductor dosimeters. These devices are susceptible to radiation as their threshold voltages shift with total ionizing dose. As they need to be sampled only once per orbit, their data and power requirements were very minimal. These factors made them ideal lightweight, low power sensors for measuring total ionizing dose. For Clementine, varying thickness of shielding was used to obtain radiation dose/depth curves at selected locations on the spacecraft and the IS AS. NRL also provided sample SRAM, Electrically Erasable Programmable Read Only Memories (EEPROM), and Field Programmable Gate Array (FPGA) devices for testing on the IS AS. in addition, as already discussed, NRL monitored the Clementine Solid State Recorder for single event upsets (SEUs).

A joint team from Aerospace Corp., the NASA Goddard Spaceflight Center, and NRL provided a small (20 cm x 7.6 cm x 2.5 cm), lightweight (270 g), low power (0.75 W) Charged Particle Telescope (CPT) for Clementine. The CPT sensor consisted of a single 3 mm silicon detector collimated to a 10° half angle field of view. Its primary objective was to measure solar protons between 3 and 80 MeV and energetic electrons between 0.3- 1.0 MeV. The lower energy electron channels provided data on spacecraft charging and on interactions between the Moon and the Earth's geomagnetic tail plasma. To date, the CPT has seen the plasma wake created by the Moon and observed numerous bursts of energetic electrons during geomagnetic storms and substorms. Like the RRELAX, the CPT clearly detected the solar particle event cm 20-25 February 1994 (Figure 12) and the associated interplanetary shock wave allowing a remarkable multi-point study of the event between Clementine, the ISAS, and several other spacecraft.

in addition to radiation effects, the ISAS carried the Orbiting Meteoroid and Debris Counter (OMDC) provided by the NASA Langley Research Center. The OMDC provided data on the near Earth natural meteoroid and manmade debris environments—data critically needed to plan and design future NASA and DoD spacecraft. The experiment also demonstrated the performance of an ultralightweight microelectronic particle impact detector that can be used on spacecraft in the future. to provide continuous monitoring of the manmade debris environments in space. Besides space debris, the OMDC measured the near Earth natural meteoroid environment at high altitudes where manmade debris (hopefully) do not yet exist in significant amounts thus allowing the contribution of the meteoroid impacts to be separated from measurements of the combined meteoroid and debris impacts. The experiment also made measurements of the smaller debris populations over a much greater altitude range than previous] y attempted and, by doing so, provided the first data on the manmade debris population to expect at altitudes up to and including synchronous orbits. The OMDC Experiment utilized ultrasensitive, thin MOS (metal-oxide-silicon) capacitors that are partially discharged when impacted by hypervelocity part icles. The OMDC instrument consisted of -1400 cm^2 of these MOS capacitor impact detectors, The detectors were mounted around the periphery of the 1 SAS (Figure 10). Approximatel y 80 impacts were detected overt he 100 day life of the IS AS. The effects of such impacts by microscopic particles on sensors and spacecraft surfaces may be important in the design of future systems just as radiation has been a design driver in the past.

The Final Word

This article has summarized the key aspects of the Clementine program from the standpoint of the new advanced technology systems and their performance. The program was designed to evaluate advanced technologies and sensor systems in the stressing space environment. We believe that Clementine has been extreme] y successful in meeting this goal. To conclude, we hope that Clementine will continue both as a mission and as a concept for quickly evaluating future technologies and systems at l ow cost. The results of the mission both scientifically and technically have been, arguably, astonishing and mark a new beginning in the design and construction of spacecraft, the era of "FCB" (faster, cheaper, better).

NOTE: After successful completion of the lunar mapping mission, a software error during the Earth gravity assist phase of the mission led to a loss of attitude control fuel and spun the spacecraft to 80 revolutions per minute. After 18 months of operation, all spacecraft systems/subsystems continue to work properly. By using the fuel in the main engine, the spacecraft rotation rate has been reduced to about 40 revolutions per minute. However, without fuel in the attitude control system thrusters, no additional scientific mission can be performed. The long duration test of the Clementine advanced lightweight system is the main goal being accomplished by monitoring the. spacecraft. Clementine is not yet lost and gone forever.

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Table 1. Clementine Technologies and Sources

TECHNOLOGY	O
Star Tracker Cameras Optics for Cameras	Lawrence Livermore National Laboratory (LLNL); Livermore, CA Optical Corp. of America, Garden Grove, CA
Ultraviolet/Visible Camera	T17TLziiZRGFFX
Laser imaging Detection and Tracking System High Resolution Camera Optics for Camera Laser Transmitter	 LLNL; Livermore, CA Optical Corp. of America, Garden Grove, CA McDonnell Douglas; St. Louis. MO
CCDs for Visible Cameras	Thomson, CSF; Totowa, NJ
Long Wave and Near Infrared Cameras	LLNL; Livermore CA Amber Engineering; Goleta, CA
infrared Focal Plane Arrays	Fermionics, Inc.; Santa Clara, CA
Stirling-Ricer Cycle Cryocooler	Ricor; Israel
Ring Laser Gyro Inertial Measurement Unit	Honeywell, Inc.; Clearwater, FL
Internal Fiber Optic Gyroscope Inertial Measurement Unit	Litton Guidance and Control Systems, Inc.; woodland Hills, CA
Reaction Wheels with Internal Drive Electronics	Ball Aerospace Corp.; Broomfield, CO
15 Amp-Hr NiH_2 Common Pressure Vessel Battery	Johnson Controls, Inc.; Milwaukee, WI
5.5 mil Gallium Arsenide on Germanium Solar Cells	Applied Solar Energy Corp.; City of industry, CA
4-Ply P75. 1999 Graphite-Epoxy Prepreg Tape Solar Array Substrate	Composite Optics; San Diego, CA with materials from Phillips Lab; Edwards AFB, CA
8-Ply T650-35/R6376 Graphite-Epoxy Fabric Composite Structure for the Interstage Adapter	McDonnell Douglas; Huntington Beach, CA
Low-Shock, No-Pyrotechnic Frangibolt™ Release Devices	TiNi Alloy Company, inc.; San Leandro, CA
32 Bit RISC Sensor Image Processor	Telenetics; Landover, MD
1.9 Gbit Capacity Solid State Data Recorder	SEAKR Engineering, inc.; Torrance, CA
Data Compression Integrated Chip Set	Matra Marconi Space; France
integrated DSN Compatible S-Band Transmitter	Loral Corp.; San Diego, CA
Composite Honeycomb Fixed Parabolic High Gain Antenna	Pacific Raydoms; Santa Clara, CA

Table 2. Clementine Sensors

Camera	Wavelength (microns)	(kg)	Power (w)	PIXELS Format	Size (microns	FOV (deg)	Resolution at Moon* Periselene (m)	Pole (m)
Star Tracker Thomson Si CCD	0.4-1.:	**0.28**	**4.5-6.5**	**384** x 576	23 x 23	28.4 x 43.2	(550)	(1650)
UV/Visible Thomson Si CCD	0.40—0.95, 0.415 ±0.02, 0.75*0.005, 0.90*0.01, 0.95±0.015, 1.0*0.015	0.41	4.5-11.0	384 x 288	23 x 23	4.2 x 5.6	115	306
High Resolution Thomson Si CCD	0.41510.02, 0.56 ±0.025, 0.65±0.025, 0.75 ±0.025, 0.4-0.8	1.12	9.5-11.0	384 X 288	23 X 23	0.4 x 0.3	30	90
Near infrared Amber InSb FPA	1.1±0.03, 1.25±0.03, 1.5±0.03, 2.0±0.03, 2.6±0.03, 2.78±0.15	1.92	11.0-24.0	256 x 256	38 x 38	5.6 x 5.6	178	475
Long Wave Infrared Amber HgCdTe FPA	8-9.5	2.10	24.0	128 x 128	50 x 50	1.0 x 1.0	65	195

***Approximate resolutions of the Clementine sensors. Lunar periselene values are** for an altitude of 425 km and **lunar polar values are for 1275** km

Table 3.

CLEMENTINE ENGINEERING INSTRUMENTS

NAME	LOCATION	INSTITUTION
RRELAX		
RADMON	Clementine, ISAS	JPL
LCCD	ISAS	JPL
CMOS/INVERTERS	Clementine, ISAS	JPL.
DAS		
EEPROM	ISAS	**NRL**
FPGA	**ISAS**	**NRL**
SRAM	ISAS	NRL
DOSIMETERS	Clementine, ISAS	NRL
CPT	Clementine	Aerospace, GSFC, NRL
OMDC	ISAS	LaRC
SOLID STATE DATA RECORDER (SEU RATE)	Clementine	NRL

FIGURES

1. The lunar South Pole between 70° and 90° showing a permanently shadowed area imaged by Clementine for the first time that may contain ice.

2. The Clementine spacecraft in its 3-axis stabilized, flight configuration.

3 The Clementine advanced camera/sensor suite. Shown are the Star Tracker, the ~~V/Visible Camera, the Near Infrared Camera, the High Resolution Camera/LIDAR Receiver, the LIDAR Transmitter, and the Long Wave infrared Camera.

4. A spectacular image taken by the Clementine Star Tracker showing the Sun rising from behind the Moon. The lunar surface is lighted by sunlight reflected from the Earth.

5. Data from lunar orbit 295 comparing the NIR and LWIR cryocooler performance. As discussed in the text, the NIR unit performed significantly better than the LWIR unit, easily maintaining its temperature near 70° K in this example.

6. Photograph of the Clementine NiH_2 CPV battery, Although discharged on several occasions, the battery showed no ill effects from this often damaging, procedure.

7. Plot of the Clementine battery pressure (psi) as a function of charge time in hours. Note the line.ar increase of pressure with charge. This allowed the. usc of a simple pressure transducer to monitor the battery charge level.

8. Photographs of the ring laser gyroscope (RLG) provided by Honeywell, Inc. (a I lightweight Advanced Implementation Technology (LAIT) IMU GG 1308 with Sundstrand RA-500 Accelerometers) and an interferometric fiber optic gyro (I FOG) provided by Litton Guidance and Control Systems, inc. (a Litton LN2001 MU with Silicon Accelerometers).

9. Plots of the Clementine SSDR integral SEU count and daily SEU rate over the mission. January 1 = day 1.

10. The Clementine Interstate Adapter Satellite (ISAS)—built around the cast-off, ancillary Clement inc solar array and lunar transfer stage, the 1 SAS was a very low cost test platform for several Clement inc engineering experiments. Shown are the Interstage Adapter ring and its solar array, the omni-directional antennas, the band of micrometeoroid detectors, and the kick motor used to inject Clementine into a lunar transfer orbit.

11. Plots of the Clementine and ISAS RRELAX proton upsets for 0.8 μm and 1.2 μm radiation soft SRAMS for a large February 1994 solar proton event. This is a unique measurement set as the identical sensors were at two very different 1 ocati ens. When orientation effects arc included, the 1.2 μm technology is less sensitive to upset than the 0.8 μm SRAM technology.

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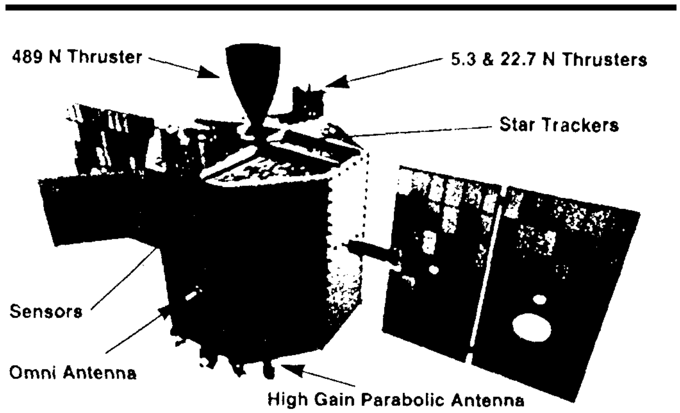
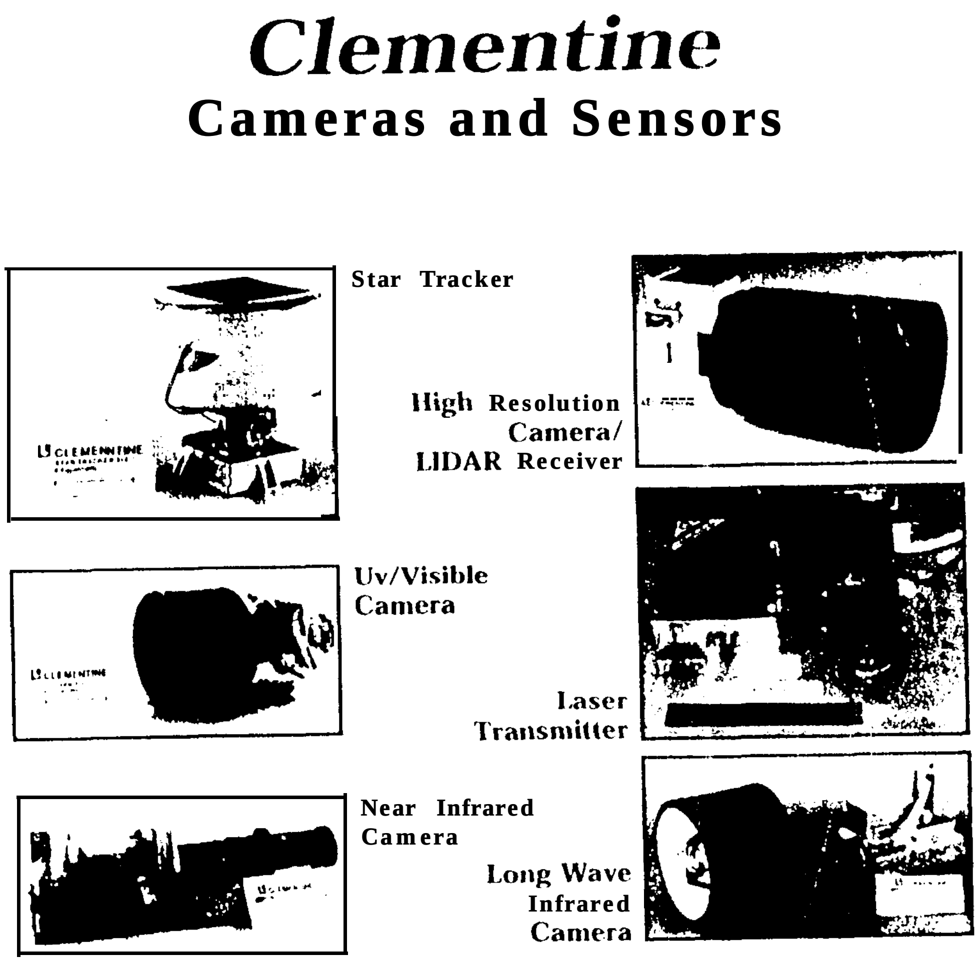


2. **The** Clementine spacecraft in its 3-axis stablizied, flight configuration.



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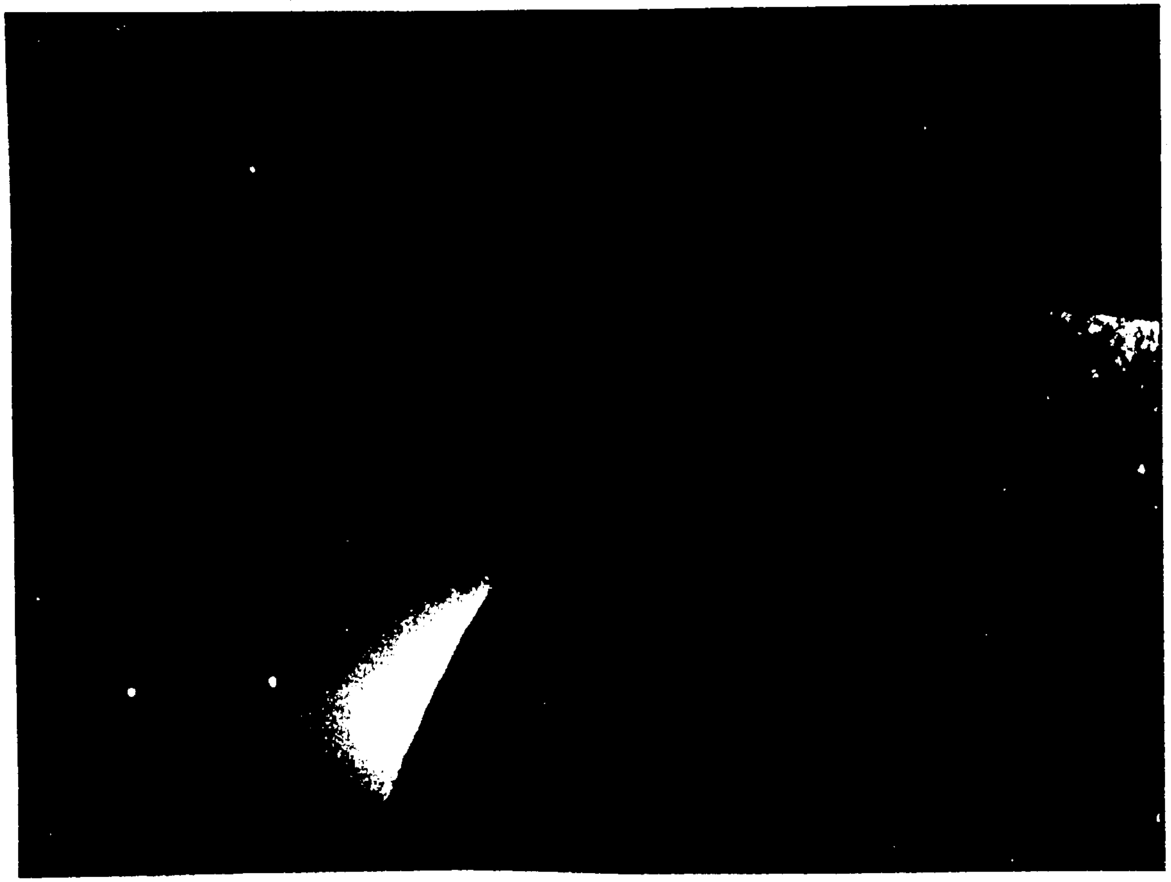
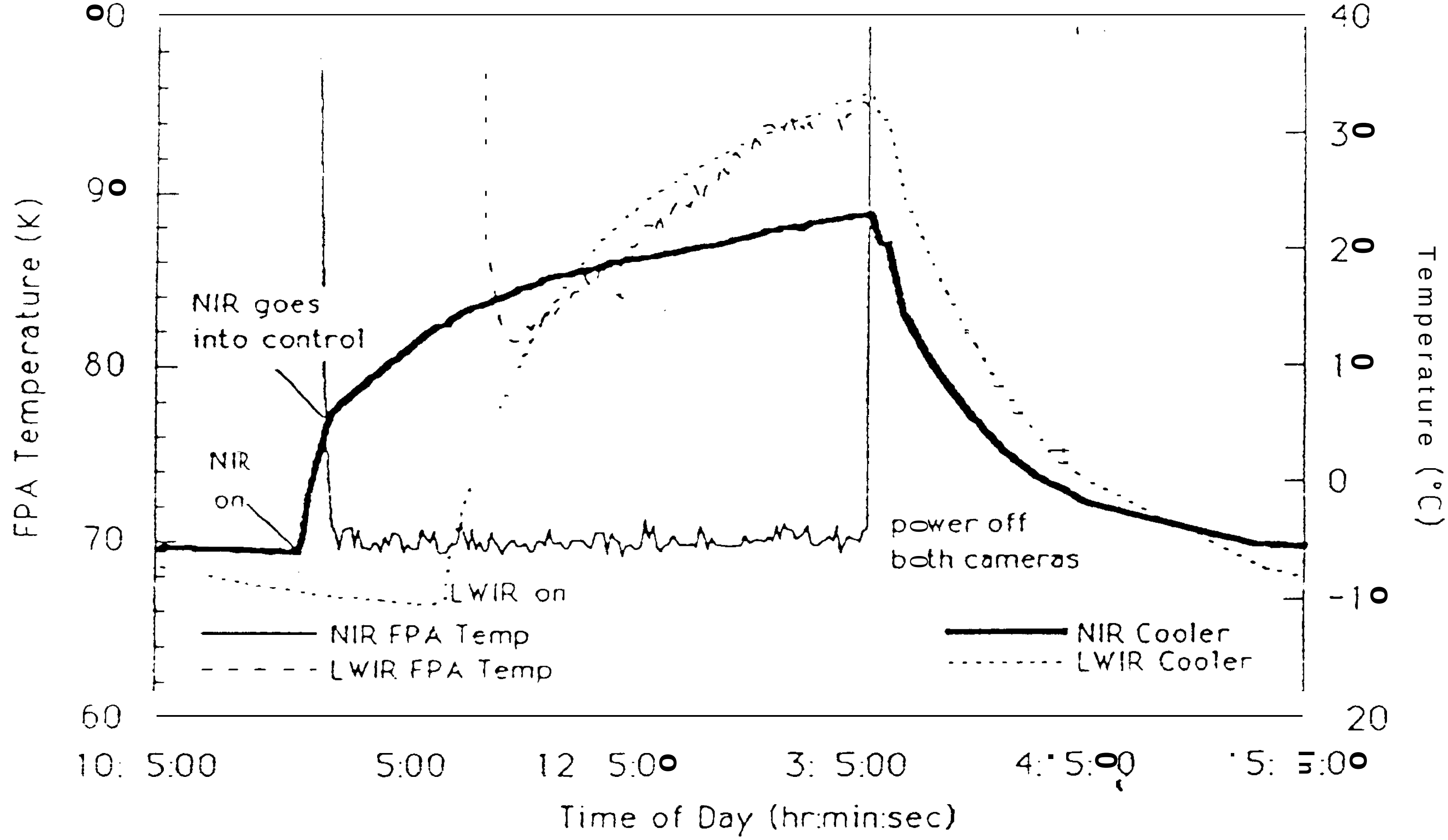


Earth Shine

Star Tracker
Camera B

Filter: 450-1000 nm
Altitude: 2358 km
Lat: 12°S
Long: 99°E
Orbit#t: 65
Time: 21:10:29 UT (Z)
Date: 5 Mar 1994

4. A spectacular image taken by the Clementine Star Tracker showing the Sun rising from behind the Moon. The lunar surface is lighted by Earthshine (sunlight reflected from the Earth).



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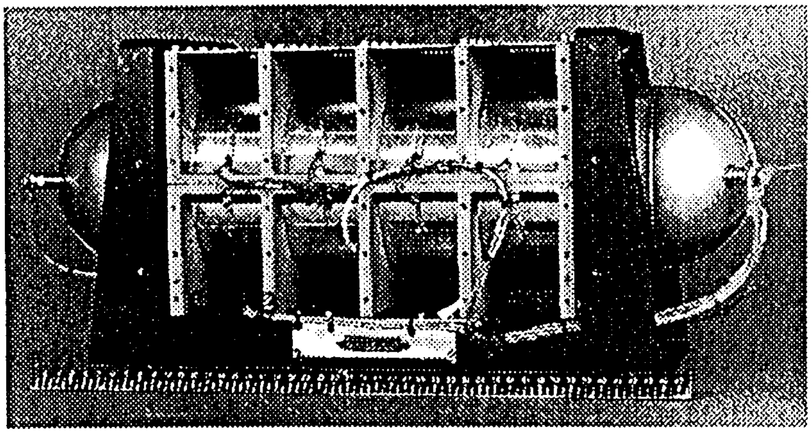

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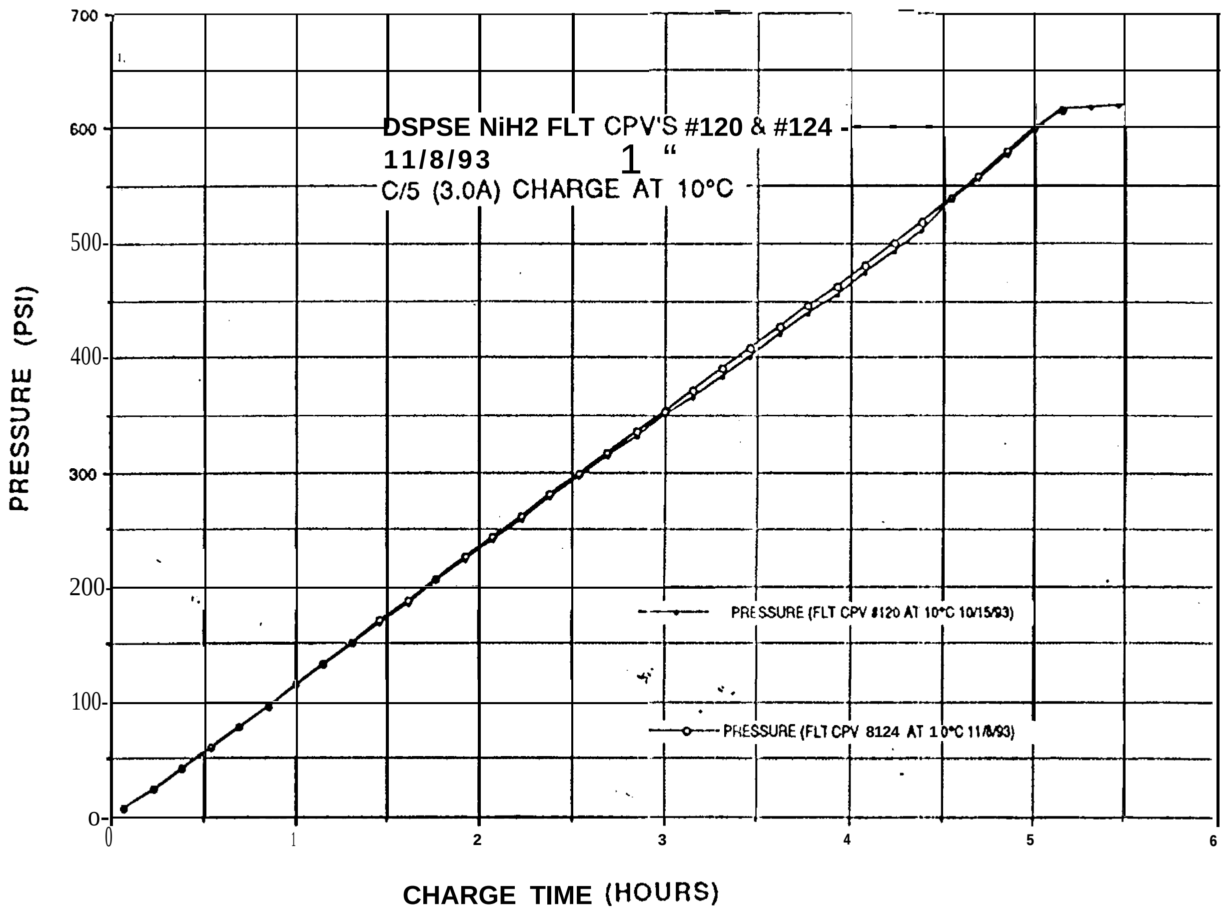

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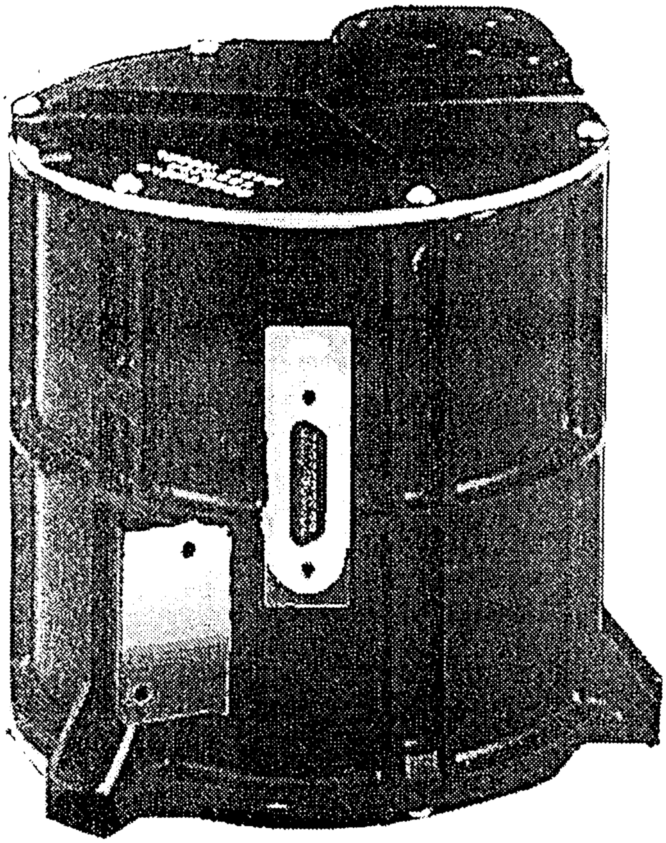
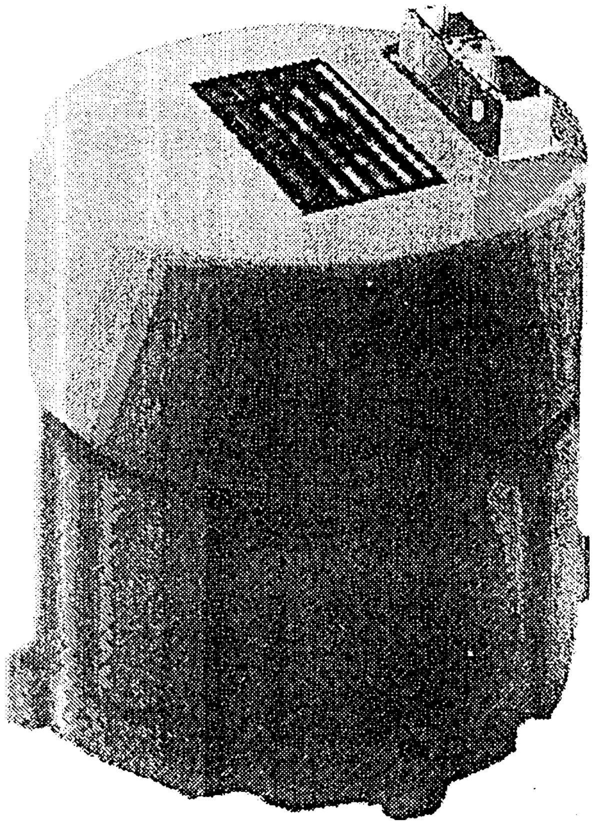


Ring Laser Gyro (RLG).



Interferometric Fiber Optic Gyro (IFOG)

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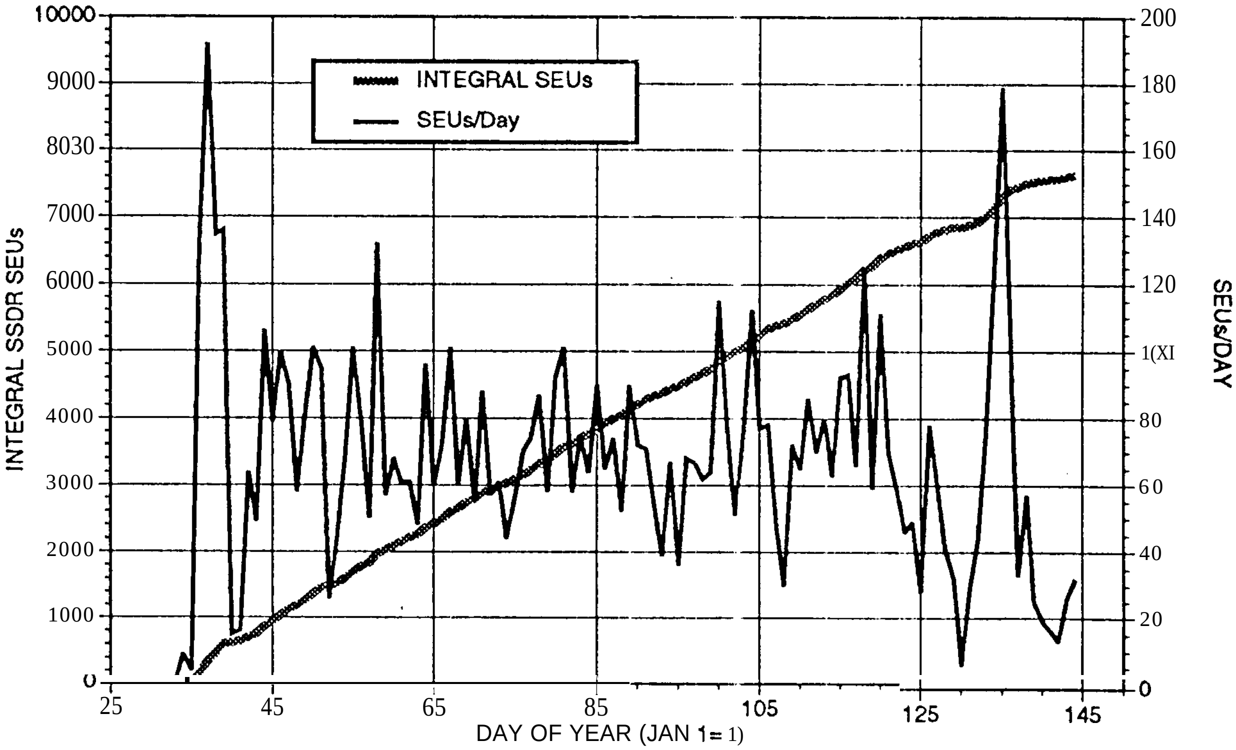

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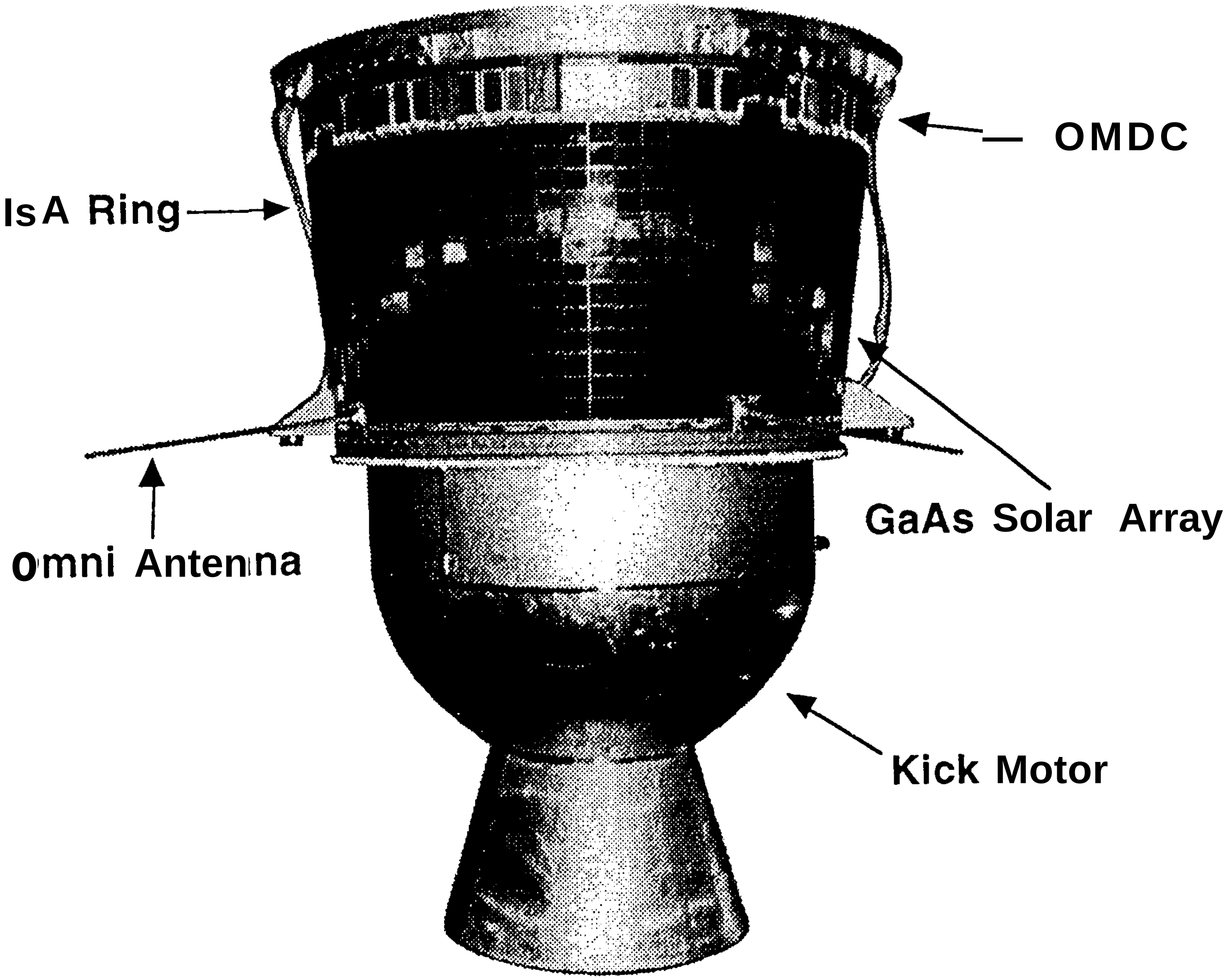
Interstage Adapter Satellite



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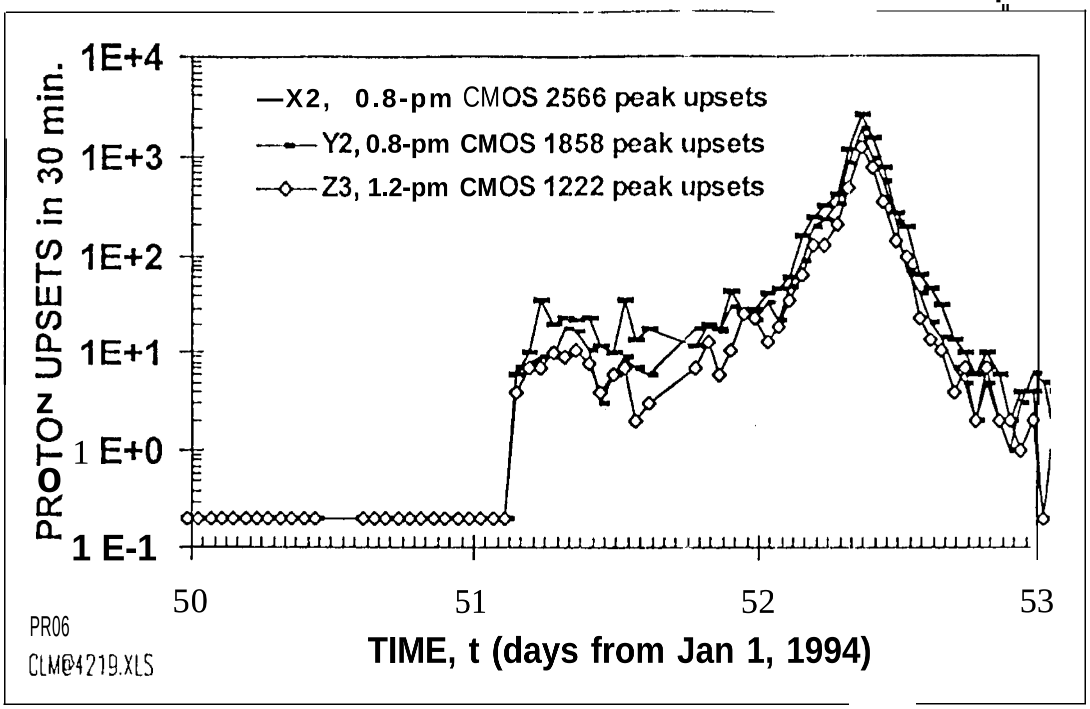
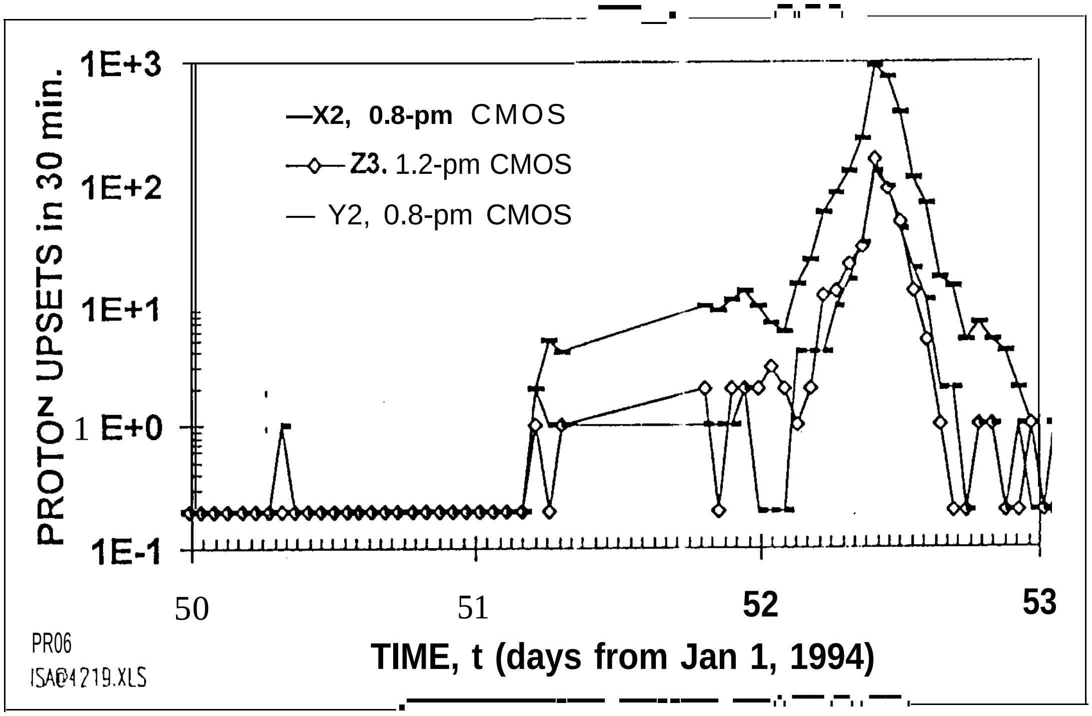
SIC RRELAX



- CMOS SRAMs have the same 31-mil Al shield.
- 0.8-pm SRAMS have more upsets than the 1.2-pm SRAM.

ISA RRELAX



- Devices have the same 37-mil Al shield.
- Must include orientation dependence but 0.8-μm CMOS SRAMs have more upsets.

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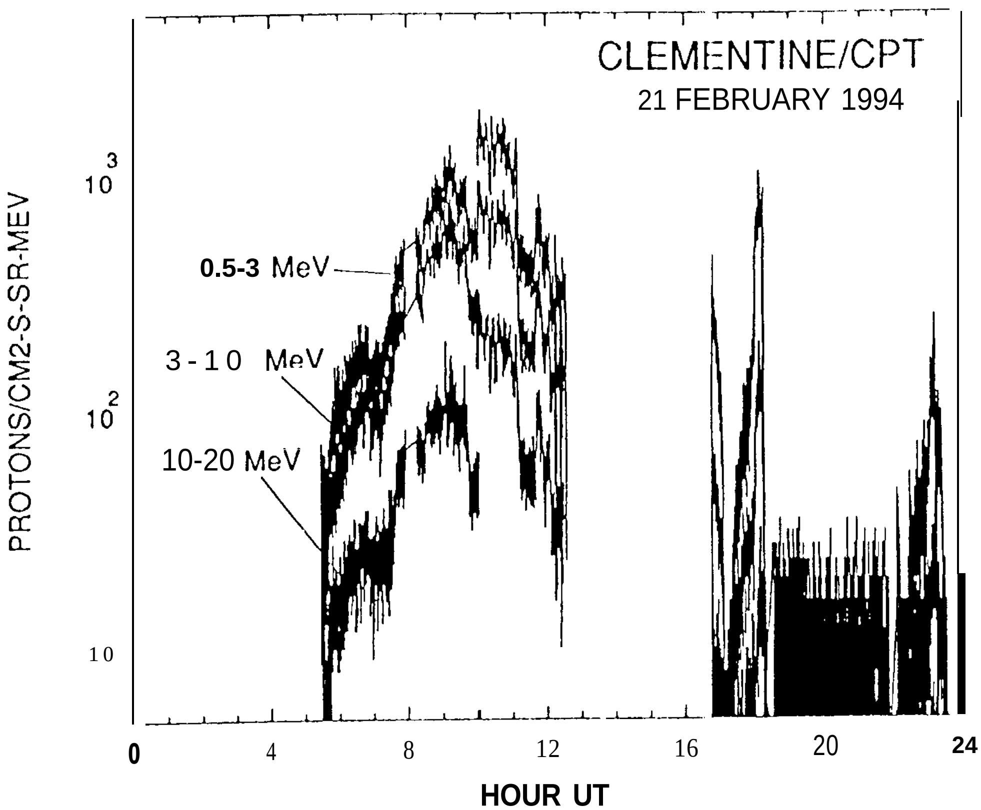

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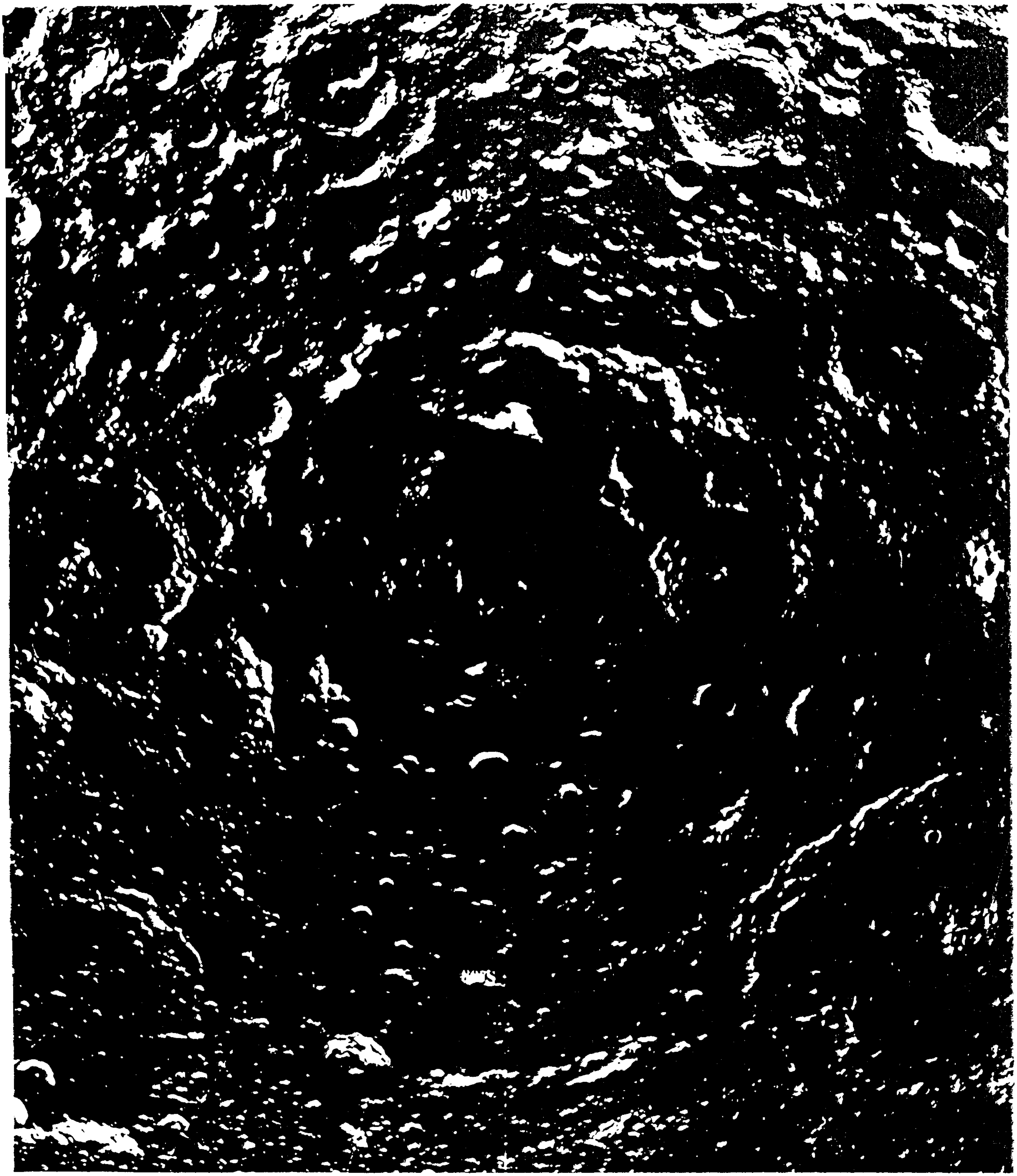

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