2.2.5

LOW COST SPACE EXPERIMENTS

STUDY REPORT

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Abstract: This is part of the ALTAIR study.
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1.0 Introduction.

1.1 Background. Over the course of the last several months of FY91, a re-stacking of funding priorities took place at the Strategic Defense Initiative Organization (SDIO). This reordering of priorities was the result, in part at least, to the changing nature of the ballistic missile threat posed by the Soviet Union, recognition of the theater ballistic missile threat as the Persian Gulf war with Iraq clearly brought forward, and the desire by Congress for the United States to again field a limited anti-ballistic missile capability. The out-growth of these developments was a reduced interest in deploying a space-based directed energy weapons system in the near term. The consequence of this reduced support was reflected in the SDIO budget by reduced funding in the acquisition, tracking, and pointing (ATP) programs.

ALTAIR was initiated in January 1991 as a key ATP technology development and demonstration program and an essential experiment enroute to an Advanced Technology Transfer Demonstrator (ATTD) planned for the last half of the decade. The ALTAIR satellite based experiment teamed the United States Air Force Phillips Laboratory with Johns Hopkins University Applied Physics Laboratory. The goals of ALTAIR were developed within the framework laid out in SDIO's ALTAIR Experiment Planning Document (EPD) dated 23 April 1991. The program had progressed through the Systems Requirements Review (SRR) in June 1991 and final preparations were underway for the Conceptual Design Review (CoDR) in October 1991, when SDIO began investigating alternative lower cost programs. Guidance for FY92 provided under Continuing Resolution Authority indicated ALTAIR funding would fall at least $25M below the level necessary to maintain schedule and all procurement activities were put on hold. Shortly after the comprehensive CoDR on 22-23 October 1991, SDIO requested the Phillips Lab to initiate two study activities to develop and evaluate lower cost experiments as potential replacements to ALTAIR. The two categories for such experiments were balloon and space based experiments.

On 28 October, a kickoff meeting was held at Phillips Lab with SDIO/TND providing general guidelines on the study efforts as well as report milestones. PL/SXL provided additional guidance to the study panels and outlined the role of a third team, the Measures of Effectiveness (MOE) panel.

1.2 Study Guidelines.

1.2.1 General. The guidance initially provided by SDIO was to develop concepts for a series of High Altitude Balloon Experiments (HABE) and a set of Low Cost Space Experiments (LCSE) alternatives that would demonstrate ATP functions and/or collect critical data necessary for developing full scale DEW ATP systems. The critical issues that these experiment concepts must address are in part described in the ALTAIR EPD but are also necessary functions that must be dealt with before progressing to an ATTD. The specific charter of the LCSE study was to: "Investigate the feasibility of resolving ATP critical issues through low cost orbital and/or sub-orbital space experiments and to develop technical approaches and cost and schedule data for viable tests."

1.2.2 Cost and Schedule. The HABE and LCSE programs should be developed within the cost constraints of approximately $50M annually over a period of three years with authority to proceed expected on 6 January 1992.

1.2.3 Hardware and Software. Hardware and software assets from the STARLAB and ALTAIR programs would be available for either option.

1.2.4 Briefings. A status briefing to SDIO would be provided on 22 November 1991 at Phillips Lab and a final briefing would be given on 18 December 1991 at SDIO.

1.3 LCSE Panel. The LCSE study panel was formed on 28 October 1991 with the following membership:

Lt Col David Williamson ALTAIR Deputy Program Manager	USAF/Phillips Lab
Dr Russ Butts ALTAIR Payload Chief Scientist	USAF/Phillips Lab
Mr John Dassoulas Chief Engineer Midcourse Space Experiment	Johns Hopkins University Applied Physics Lab (APL)
Dr Jack Hammond	WJ Shaffer Associates
Dr Sherm Seltzer ALTAIR Associate Engineer	ACE Logicon/Control Dynamics

Although not on the original team, Mr Denny Bosen of Logicon/RDA and Dr Pete Bythrow of APL both played key roles in developing payload concepts and their work is reflected in the experiment descriptions that follow.

1.4 Study Approach. The geographic separation and primary job responsibilities of the team members dictated that the normal integrated, face-to-face study approach be modified. The nature of the study complimented this modification in that there was no single experiment that the entire team focused on. Instead, potential or candidate experiment concepts were identified by the team and then individuals were assigned the responsibility of developing a particular concept. The approaches were shared via telecon as they matured providing the opportunity for exchanging ideas and identifying weaknesses in the concept. In parallel, launcher data was collected and assimilated into a table of capabilities which provides performance information and payload constraints to assist in concept development.

Examining potential experiments and the cost and availability of launch vehicles led to the definition of a set of experiment types. They were:

1.4.1 Fly-Alongs. This experiment class consists of payloads that make observations or measurements from aboard a non-dedicated launch vehicle.

1.4.2 Sub-Orbital Experiments. Concepts in this category employ one or multiple rockets (simultaneously launched) to conduct ATP experiments and/or collect phenomenology data.

1.4.3 Secondary Payload. This category was identified to explore experiments which would operate on-orbit but would be piggybacked on a platform that was already scheduled to fly.

1.4.4 Satellites. This set includes dedicated space platforms for conducting ATP relevant experiments as the primary mission of the spacecraft while on-orbit.

Outside of the products developed by the study panel were concepts developed by industry. Most major spacecraft manufacturers were contacted and informed of the study activity the government was conducting. They were provided background data on ALTAIR and the EPD and offered the opportunity to develop concepts for conducting ATP experiments within the cost and schedule guidelines described earlier. The contractor approaches were provided to the MOE panel for evaluation and are described later. Their full study products are provided as appendices.

2.0 Platforms.

2.1 Candidate Launchers. Given the assortment of possible experiments, a comprehensive list of potential launchers was developed. Table 2.1-1 includes all launchers realistically considered. The key criteria in assessing the vehicles were cost, lift capability (mass and volume), and availability. The stated cost constraint of $150M on the entire program precluded consideration of rockets which cost more than $25M.

2.1.1 Fly-along Launchers. By design the experiment will be of minimal impact to the launcher therefore it is anticipated that there will be no difficulty finding a ride. An interesting potential is the 2nd to 3rd stage interstage on the Delta II. NASA recently flew approximately 200 pound of payload in this area. With the frequency of Air Force Delta 7925 flights carrying Global Positioning System (GPS) satellites, multiple opportunities would exist. The Midcourse Space Experiment (MSX) target rockets could provide other potential vehicles and have the advantage of being under SDIO control.

2.1.2 Sub-orbital Rockets. A large variety of such rockets have flown or are on the drawing boards to fly. as a result, no firm decisions have been made on exactly which rocket best suits the planned application. Since most cost well under $5M, they are not considered major cost drivers.

2.1.3 Satellite Launchers. After evaluating the rockets in this category, five emerged as realistic candidates.

2.1.3.1 Pegasus. Versions of this vehicle provide satisfactory performance at an affordable cost. These launchers are being procured as the Air Force Low Cost Launch Vehicle (LCLV) by the Space Test Program Office at Space Systems Division and would be available for a FY95 ILC.

2.1.3.2 Taurus. DARPA is sponsoring development of this vehicle and its first launch is scheduled for late 1992 with the Phillips Lab managed TAOS experiment as payload. The Taurus Critical Design Review (CDR) was held 20-23 November 1991 and there are some concerns over the program schedule. The launcher offers good performance but costs approximately twice as much as Pegasus. A Taurus vehicle would probably be available through DARPA but given its current development status and concurrent launch support infrastructure, there is some reluctance by the panel to plan on its use.

2.1.3.3 Titan II. Approximately 40 Titan II rockets exist as a result of their phase out in the US ICBM force. The great majority of these are Titan II-Bs and require substantial refurbishment to be converted to a spacecraft launcher. There are also several Titan II-Gs which have already been converted by an Air Force customer who has since elected not to use them. The Air Force is considering using these G models to launch DMSP satellites in the late 90's but it is quite likely that one could be made available for a near-term customer. The performance of the Titan II, as it exists without solid rocket motor strap-ons, is quite adequate for LCSEs and a good track record and reasonable cost make it an attractive launcher.

2.1.3.4 Atlas II and Delta II. The cost of purchasing an Atlas or Delta rocket is too high for a LCSE, however, the opportunity may exist for sharing a ride with another satellite. The Air Forces P-91-1 spacecraft is scheduled for a Delta II launch in early 1995. The weight of the spacecraft is projected to be 5800 pounds which is only 60-70% of the lifted capability of the launcher. McDonnell Douglas has also identified the potential opportunity of piggybacking with a communication satellite launch in late 1994. The General Dynamics' Atlas companion payloads opportunities could also provide a ride for a secondary satellite. Piggyback type launches would be more fully evaluated after a

F.A.W.G. PLANNING MANIFEST

THIS MANIFEST IS A FAWG ASSESSMENT OF THE NASA BASELINE AND SHOULD NOT BE CONSIDERED AN OFFICIAL SSP MANIFEST.

Orbiter	Date	Flight	Duration	Payloads
102 COLUMBIA	[3 JUN]	50	13	<USML-1> CONCAP-IV IPMP
102 COLUMBIA	[24 SEP]	52	9	LAGEOS-II USMP-1 CANEX-2 IOCM ASP HPP PSE PCG BLOCK - II
102 COLUMBIA	[27 JAN]	55	9	<SL-D2>
102 COLUMBIA	[22 JUN]	58	13	(SLS-2)
102 COLUMBIA	8 DEC	62	9	USMP-2 SPTN-201-2 DEE
102 COLUMBIA	7 JUN	66	13	<IML-2>
103 DISCOVERY	[22 JAN]	42	7	(IML-1) GAS BRIDGE GOSAMR-1 IPMP RME-III SE 83-2 SE 81-9
103 DISCOVERY	[15 OCT]	53	4	27 OCT* DOD-1 GCP
103 DISCOVERY	[22 FEB]	51	8	ACTS ORFEUS -SPAS
103 DISCOVERY	1 SEP	59	7	SPACEHAB-2 WSF-1 OAET-1 CAPL-1 GAS BRIDGE
103 DISCOVERY	5 FEB	63	7	SPACEHAB-3 SPTN-204 IEH-1 ROMPS-1 GAS BRIDGE
103 DISCOVERY	10 AUG	67	7	ASTRO-2 CMSE-1 OAET-FLYER
104 ATLANTIS	NET 24 NOV*	44	10	DSP IOCM M88-1 AMOS SAM CREAM RME-III UVPI VFT-1 TERRA SCOUT
104 ATLANTIS	[14 MAR]	45	8	ATLAS-1 SSBUV/A GAS(9) IPMP STL-1 VFT-2 RME-III CLOUDS-1A SAREX-II
104 ATLANTIS	[2 JUL]	46	7	TSS-1 EURECA-1L EOIM III/TEMP2A-3 ICBC AMOS CONCAP-II CONCAP-III LDCE-1 PHCF UVPI
104 ATLANTIS	[22 APR]	57	7	SPACEHAB-1 EURECA-1R SHOOT GAS BRIDGE
104 ATLANTIS	3 NOV	61	8	HST REV-1
104 ATLANTIS	5 APR	64	7	LITE-1 SPAS-III PL OPT
105 ENDEAVOUR	[9 APR]	49	7	INTELSAT-6R ASEM CVTE PCG-III-II AMOS
105 ENDEAVOUR	[12 AUG]	47	7	(SL-J) GAS BRIDGE ISAIAH UVPI AMOS
105 ENDEAVOUR	[3 DEC]	54	6	TDRS-F DXS
105 ENDEAVOUR	[23 MAR]	56	9	ATLAS-2 SPTN-201-1 SSBUV/A
105 ENDEAVOUR	1 OCT	60	9	SRL-1
105 ENDEAVOUR	3 MAY	65	9	CRISTA-SPAS ATLAS-3 SSBUV/A

FY92 · FY93 · FY94 — CY91 · CY92 · CY93 · CY94 (quarters 1 2 3 4)

[FDRD DATE] ASSESSMENT DATE* <LM-1> (LM-2) TMU

REFLECTS PROPOSED 8 FLIGHT PER YEAR MANIFEST

OV-102 AND OV-105 NEAR TERM LAUNCH SCHEDULES UNDER REVIEW

11-21-91

U.S. Gov't

program go ahead by SDIO. The cost savings relative to the full price of a medium launch vehicle would be substantial although the orbit selection would obviously be based on the needs of the primary payload.

2.1.3.5 Space Shuttle. The shuttle offers obvious benefits in getting a payload to orbit and successfully deployed. The capability for human intervention is preserved in some situations until after the spacecraft is activated on orbit. Just as obvious is the cost associated with flying on the shuttle and "shuttle qualifying" the spacecraft. This launcher is considered primarily because of the availability of a payload opportunity.

Figure 2.1-1 shows the 21 November 1991 Flight Assessment Working Groups working manifest. STS-64 scheduled for launch in April 1994 currently has SPAS-III and LITE-1 manifested with a payload opportunity available. This mission is a DoD "credit" for the Starlab mission which was canceled in September 1990. This "credit" must be scheduled for a launch not later than 28 September 1994 at which time DoD loses the flight. Having the flight available in the time frame of a LCSE provides the possibility of access to space at a reduced cost. Whether the costs are less than a dedicated launcher must be assessed on a case by case basis evaluating each space experiment.

Figure 2.1-1

A secondary reason for examining the shuttle is the availability of Get Away Specials or "GAS cans" and "Hitchhiker" payloads. The cost of carrying one of these types of payloads on the shuttle is relatively low. However, the associated spacecraft that can be deployed is quite small, on the order of several hundred pounds. Some experiment concepts can be packaged within the constraints making this option worthy of serious consideration.

2.2 Dedicated Spacecraft Buses. The suitability of existing bus designs and hardware was assessed with the emphasis on finding a relatively low cost and available solution. Although there is no such thing as a standard bus, industry efforts such as Fairchild's Multi-Mission Spacecraft and TRW's Space Test Experiments Platform (STEP) are making the commonality between spacecraft much greater. The particular bus hardware and designs explored are described below.

2.2.1 Existing Hardware

2.2.1.1 SPARCS. An ongoing suborbital rocket experiments program out of White Sands Missile Range uses the SPARCS platform. SPARCS, in various configurations and models, has supported NASA and the Air Force Geophysics Lab experiments for over 20 years. SPARCS bus hardware currently exists and is owned by __________. The performance of the attitude control system as well as other routing bus functions such as electrical power, telemetry, and guidance systems appear to be adequate for the proposed applications. Further details on this bus are provided in Appendix 1.

2.2.1.1 P-80. Built for AFP-888, the Teal Ruby Program, the P-80 spacecraft was completed in 1986 and was scheduled to go on the first Vandenberg AFB launch of the shuttle in July 1986 until the Challenger accident occurred in January. P-80 had essentially completed all safety reviews for the Western Test Range (WTR) launch. It was subsequently re-manifested and was scheduled to fly on STS-39 with the Infrared Background Instrument Surveillance System (IBISS) and the Cryogenic Infrared Instrument Survey (CIRRIS). The NASA safety reviews against the new shuttle standards were nearing completion when the Air Force canceled the program in October 1988. The program was canceled for a lack of need for the planned IR air vehicle detection experiments and background data between 2-16 microns Teal Ruby was to collect.

P-80 was assessed in considerable detail for use as the ALTAIR spacecraft bus by engineers at APL, Logicon Control Dynamics, Logicon RDA, and Hughes. The primary reasons it was rejected grew out of the extremely demanding ALTAIR requirements. Specifically, P-80 required repackaging of its subsystems including relocation of the solar array to fit within the 10 foot faring of a Delta II. The electrical power requirements of ALTAIR had approached 5 kW at peak demand which required substantial modifications to the P-80 electrical power system. Quiescent power was being carried at 1 Kw when P-80 was dismissed, exceeding its solar array average orbit output power of 600 W. During the period when P-80 was being evaluated, ALTAIR had a requirement for carrying two 150 Mbps tape recorders which would need to downlink at 25 Mbps in X-band and 5 Mbps and 64 Kbps in S-band. P-80 was only equipped with three 1 Mbps recorders and fully redundant 1 Mbps and 32 Kbps downlinks.

Other concerns during the evaluation were related to P-80's hydrazine reaction control system (RCS). Analysis of an Integrated Pulse Frequency Modulation (IPFM) control system indicated that thruster performance could meet the line of sight accelerations and rates associated with an engagement of a Minuteman II target launched from Vandenberg AFB and still satisfy the jitter specification. Contamination by the hydrazine monopropellant was also investigated and an assessment recently completed by Goddard Space Flight Center uncovered no significant problems (Appendix 2).

Since SDIO has requested LCSE alternatives be examined and flight opportunity exists on the shuttle that P-80 was built to fly on, this spacecraft apparently offers a very viable approach.

2.2.2 Existing Designs

2.2.2.1 STEP. The Air Force currently has a contract with TRW for delivery options of up to 12 Space Test Experiment Platform (STEP) buses. Currently three missions are scheduled with TAOS being the first user. These buses are capable of handling payloads approaching 1000 pounds and communication links can be tailored per mission requirements. The production line features of this bus make it an attractive candidate for some of the experiment concepts.

2.2.2.2 MSX. APL is currently building this spacecraft for SDIO and has done significant work in assessing its design for other experiments such as ALTAIR. The MSX design is essentially two box structures with one containing the normal spacecraft on-orbit functions and the other structure containing most of the payload instrument suite (Figure 2.2-2). A graphite epoxy truss connects the two boxes and also serves as the primary support structure for the primary MSX instrument, the SPIRIT III LWIR telescope. The MSX design could be modified to accept another payload by removing the truss structure and attaching the instrument package to the avionics section. The robust capabilities of the MSX bus such as redundant 25 Mbps X-band links and 25 Mbps tape recorders are strong attributes of this spacecraft but also contribute to a relatively high cost.

2.2.2.3 Other. Data was collected on several other spacecraft buses. McDonnell Douglas provided data on its low end bus "Beta" and mid-size bus "Gamma" (Appendix 3). Lockheed also has a mid-size bus called "F-SAT" that it is marketing and Fairchild has MMS, a well proven design. In general these bus types are in the $45M plus range and given the maturity of the experiment concepts and uncertainty factor associated with any LCSE, no in-depth discussions were held on spacecraft bus designs.

2.3 Secondary Payload Spacecraft Buses. Although there may be experiments that could be packaged as a secondary payload, none were identified by the study panel as viable in dealing with ATP critical issues. It was recognized that volume, mass, and power are available on some experiment platforms such as P-91-1, but the actual real estate available is probably insufficient or the primary mission too incompatible to warrant further investigation at this time. SPAS-III may be an exception in that it is already planning to conduct ATP relevant experiments. However, there is not much area available for adding experiments onto the platform and it is a shuttle sortie mission, as

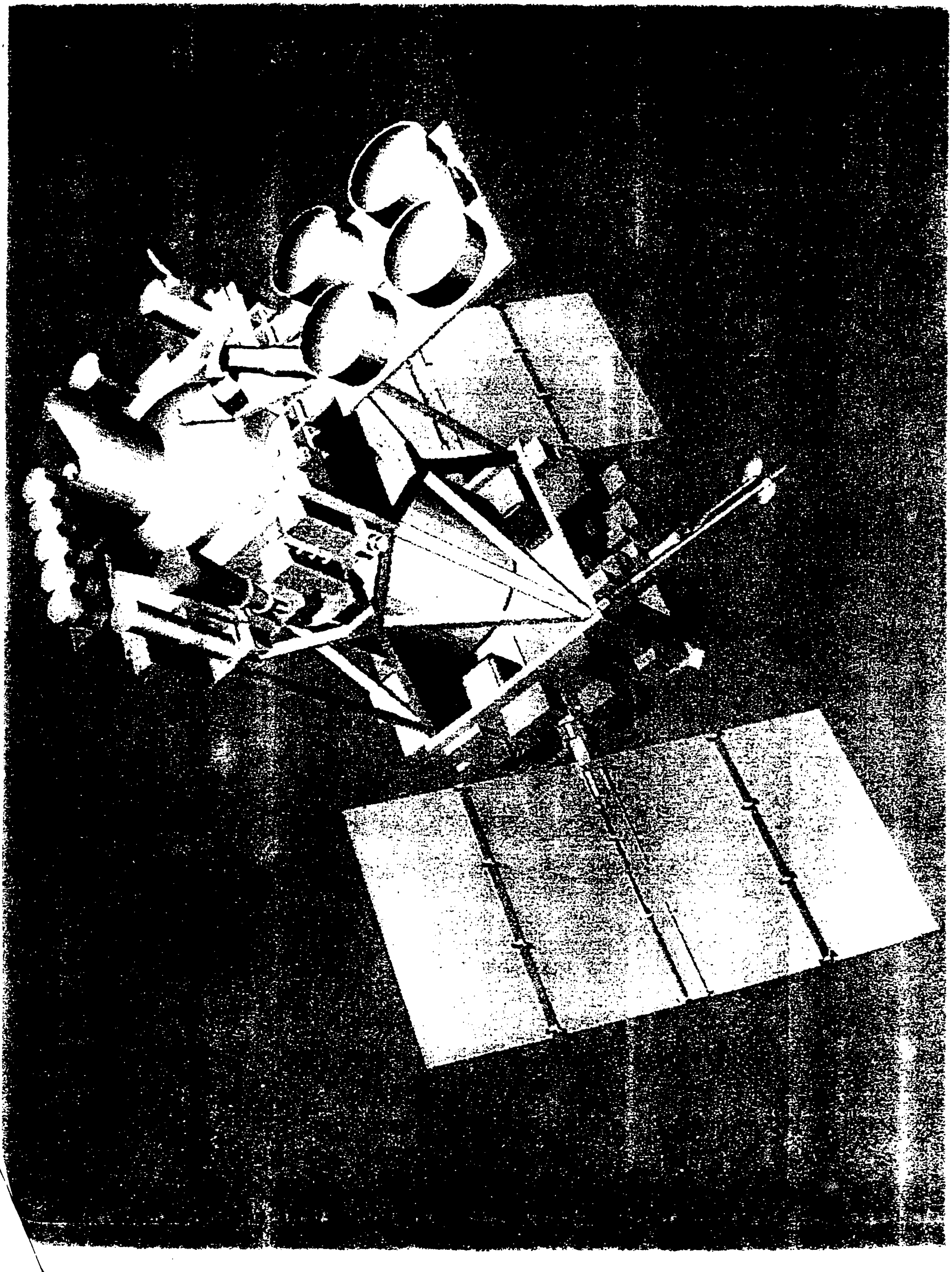

Figure 2.1-2

was Starlab. The difficulties expected to be encountered in conducting an ATP mission would make such experiments from SPAS-III extremely risky. They would better be performed from a free-flyer where time would be available to adjust tracking algorithms. SPAS-III would therefore be best suited for data collection as was done with IBES in April 1991. The liquid plume generator planned to be deployed by the shuttle on this mission and the shuttle itself would be the primary targets and it is assumed at this time that SPAS-III will be instrumented to collect data of interest to SDIO.

3.0 Experiments Concepts. The performance capability, cost, and availability of launch vehicles provided certain bounds on an experiment design. Certain key performance parameters such as the ability to generate electrical power and availability of a spacecraft placed further constraints on any LCSE. As stated earlier, the study approach was to identify certain experiment approaches and then systematically develop them. It was recognized by the panel that the cost/benefit equation was operating on all designs and became an additional constraint. Given an appreciation of preceding factors, the following pages will develop experiment concepts that the study panel has investigated. They are presented in the same sequence as they were identified earlier.

3.1 Plume Reflection Measurement Experiment (Add On). This experiment would gather in-flight plume reflection measurements at the lowest possible cost. It can provide a good estimate of the plume reflectivity while avoiding large optics, precision pointing

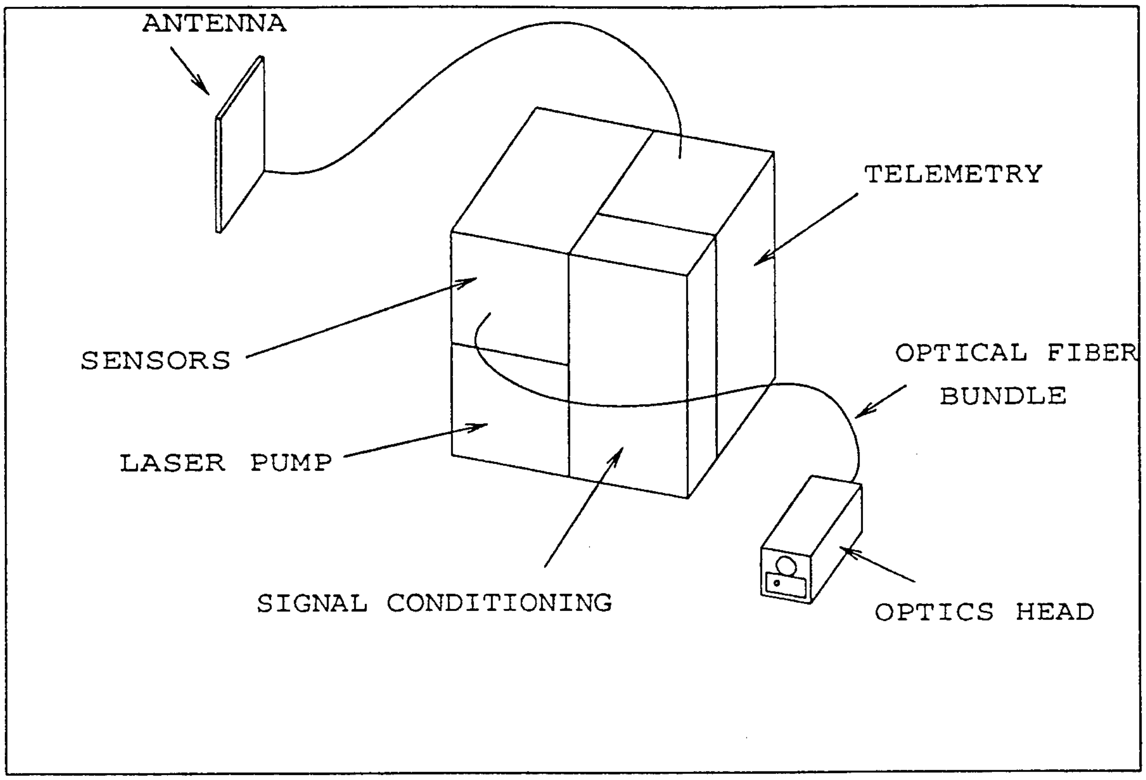

FIGURE 3.1.1. Experiment Conceptual Layout. Note there are two modules, connected by a fiber optic cable, and an antenna.

systems, and high power illuminator lasers. The information will help understand rocket exhaust particulate sizes and composition, and it is vitally important to designing active tracker systems and associated hard body handoff and aimpoint control algorithms. The information is difficult to gather in a rocket test chamber because the environment of the chamber affects the exhaust particulate characteristics and because only a very small region near the nozzle can be tested.

3.1.1 Experiment Objective. The experiment will measure the reflection of laser light from a plume from on-board the rocket itself, using a small probe laser and non-imaging detector with range resolution. The equipment will be inexpensive, small, and light enough that it can be flown on several rockets on a space-available, non-intrusive basis.

3.1.2 Experiment Description.

Figure 3.1-1 Experiment Conceptual Layout. Note there are two modules connected by a fiber optic cable and an antenna.

3.1.2.1 General. Figure 3.1-1 provides a conceptual layout of this experiment. The experiment is divided into two principal sections: a main module and an optical head. The two are connected by a small fiber optics bundle. The main module contains the electronics, sensors, batteries, and laser pump diodes. The optical head contains a very small laser and some optical components. The main module can be located wherever there is some free space, and the optical head can be located to give the desired view of the plume. Optionally, a the optical head or a small mirror could be scanned to provide two dimensional coverage of the plume reflectance. The detector high voltage supply, amplifier, high speed sampler, and FIFO data buffer will be based on readily available designs and off the shelf components.

3.1.2.2 Instruments.

3.1.2.2.1 Sensors. The detector is a non-imaging photomultiplier tube, with a response time of about 4 nanoseconds. It will have a very narrow band filter centered at the laser wavelength to minimize the interference from plume emissions. When a pulse from the laser is emitted, the output of this detector will be sampled at intervals corresponding to its response time, to produce a sequence of numbers corresponding to the light received vs. time at the approximate 4 nanosecond resolution. Since longer times correspond to longer round-trip path lengths, the signal vs. time will give an indication of the plume reflectance vs. distance along the laser path. With the presently postulated laser, light will be emitted at both 532 and 1064 nm, so a detector can be used at both wavelengths to help characterize the particulates.

3.1.2.2.2 Laser. The most immediately promising laser is a two-part frequency double Nd:YAG laser such as that produced by Spectra-Physics. The pump diode output is injected into a fiber optic, which then carries the pump light to a separate head. This head contains the laser rod and frequency doubler. There are no electronic components in the laser head itself, so there is no significant heating. The laser head is quite rugged, and is less than 4 inches long, and less than 2 inches wide and deep. Its output varies from 40 to 150 micro-Joules, depending on the mix of 532 or 1064 nm output. Its typical pulse rate is once per second.

3.1.2.3 Other Experiment Components.

3.1.2.3.1 Fiber Bundles. The fiber bundle is made up of 3 or 4 separate fiber bundles. One (the largest, with a bundle size of about 1/16") is for the laser pump light. There is one fiber for the laser output monitor, and one for the received light for the detector, which is located in the main module. The fourth fiber is for a second wavelength detector, if used.

3.1.2.3.2 Controller. The experiment will be controlled by a very simple sequencer. The system will require only a few seconds of warmup, so it can be powered up by a single command from the host vehicle or by a G-switch to detect launch.

3.1.2.3.3 Power Supply. The power supply will be contained within the main module and will be comprised of lithium or silver-zinc batteries.

3.1.2.3.4 Data Handling and Telemetry. The total data rate from the experiment will be less than 10 Kbps. The data will be telemetered to the ground in real time, with no on-board recorder. A 5-watt S-band transmitter similar to that being planned by the

Phillips Laboratory for their SIE and balloon experiments has been tentatively selected. It has much more capacity than needed, but it is a standard piece of hardware and relatively inexpensive.

3.1.2.4 Power. The largest power consumer will be the telemetry transmitter, at around 20 watts. The controller, laser, and detector will each be under 5 watts. The detector electronics will be between 5 and 10 watts. An conservative total power consumption would be 50 watts for the experiment duration of around 5 minutes. Eagle Pitcher produces a small lithium battery that weighs 7 to 8 pounds with more than enough capacity. It is used in the Shuttle booster recovery package.

3.1.2.5 Mass. The optical head will be less than 0.5 cubic foot and weigh 5 to 10 pounds, depending on the environmental isolation (thermal and vibration) and whether the optional scanning feature is used. The environmental isolation requirements will be determined by the location of the head in the host vehicle. The main module will be 1 to 2 cubic feet and weigh 20 to 40 pounds. The fiber optic bundle weight will depend on the particular installation, but will be a few pounds at the most. Total weight is thus in the 40 pound regime.

3.1.2.6 Options. Several options are available for each mission, depending on the installation and the data desired.

3.1.2.6.1 Multiple Wavelength Detection. The selected laser emits light at both 532 and 1064 nm, so both wavelengths could be detected. This would require a slightly more complex optical head, two detectors, and extra electronics. The size of the system would not be appreciably affected, and the weight would probably be increased less than 10 pounds to a total of perhaps 50.

3.1.2.6.2 Scanned Beam. The simplest system has only a single probe beam in a fixed direction, so the data is limited to reflectivity along that beam. Either the head itself or a small mirror could be slowly panned to allow developing a two dimensional map of the reflectivity. The scanning feature would add about 5 pounds to the head and somewhat increase its physical dimensions.

3.1.2.6.3 Optical Head Location. The optical head has two potential locations: under a payload shroud and at the vehicle aft end, near the rocket nozzle. If the head is located under a shroud, it would need a simple mechanism to deploy a mirror, as shown in Figure 3.1-2, to allow the sensor to view the plume. In this configuration, reflections in the mixing region could be measure (if significant), and the deployed mirror would be slowly scanned in one axis to give a two dimensional characterization of the plume. In this configuration, no data could be obtained until the payload shroud was removed. Also, the plume region closest to the rocket nozzle would likely be hidden by the rocket body. The second optical head location, at the aft end of the vehicle, would provide a better view of the part of the plume nearest the vehicle. The aft location is sketched in Figure

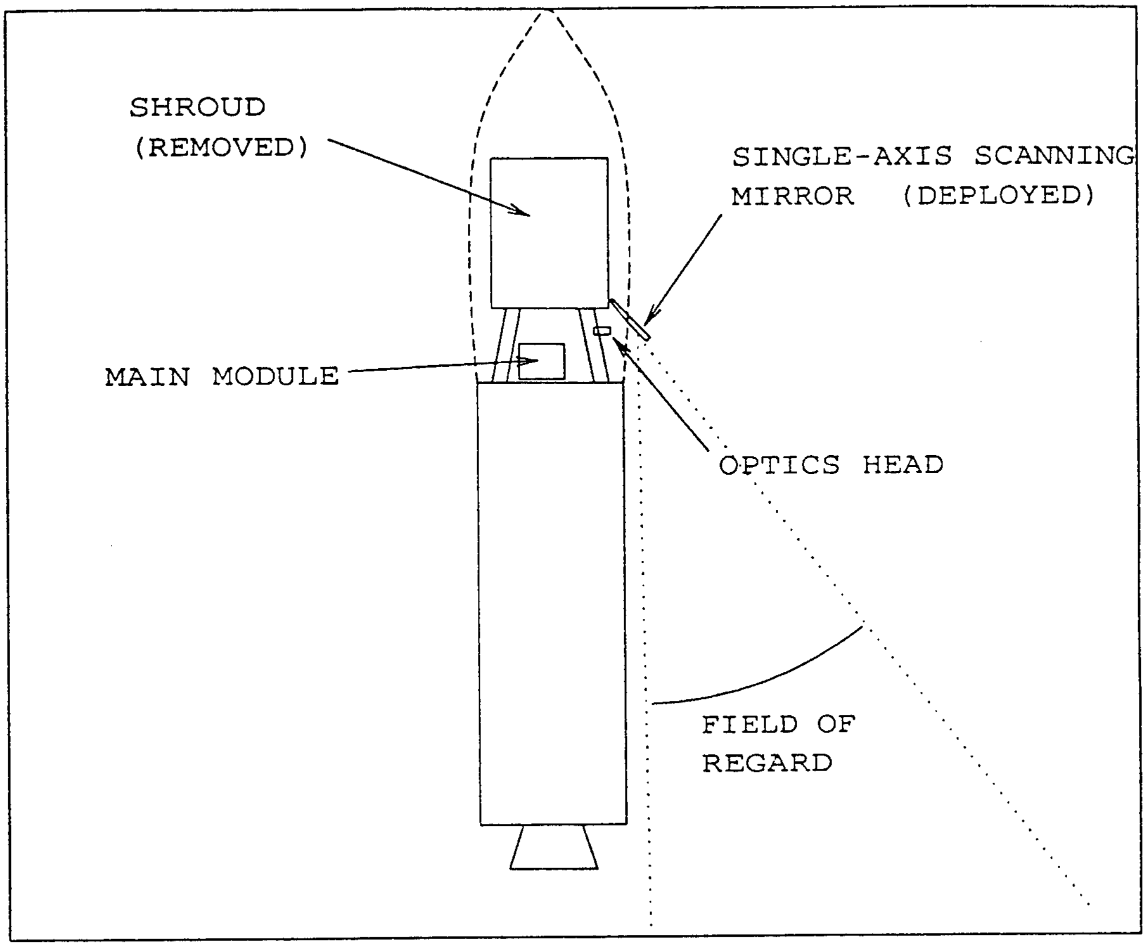

FIGURE 3.1.2. Forward Optical Head Configuration. The laser and sensor line of sight is directed by a mirror that is deployed after payload shroud removal.

Figure 3.1-2 Forward Optical Head Configuration. The laser and sensor line of sight is directed by a mirror that is deployed after payload shroud removal.

3.1-3. The environment in this location is likely to be more hostile and would require better isolation. It would also be difficult to view the mixing region next to the rocket body and ahead of the exhaust.

3.1.3 Experiment Operations Concept.

3.1.3.1 Experiment/Engagement Requirements. This experiment would be flown as an add-on to other missions as a strictly self-contained system. There would be no special requirements for the engagement, and multiple flights would be expected to characterize various plumes under different conditions.

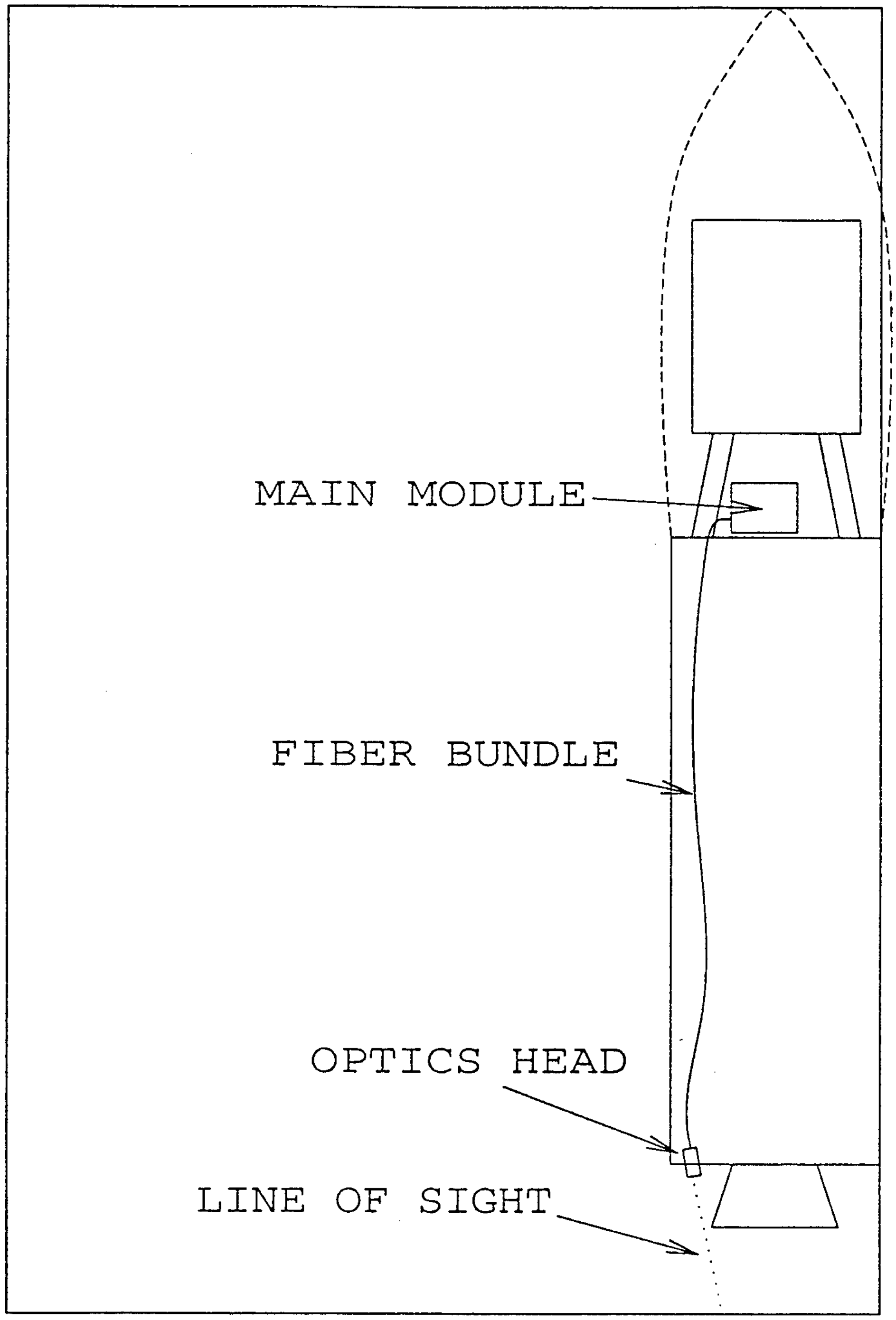

FIGURE 3.1.3. Aft Optical Head Configuration.

Figure 3.1-3 Aft Optical Head Configuration.

3.1.3.2 Experiment/Engagement Events and Sequence. The system would be inert prior to launch. At launch, a timer would be initiated to enable the laser, sensor, optional steering mechanism, and telemetry transmitter.

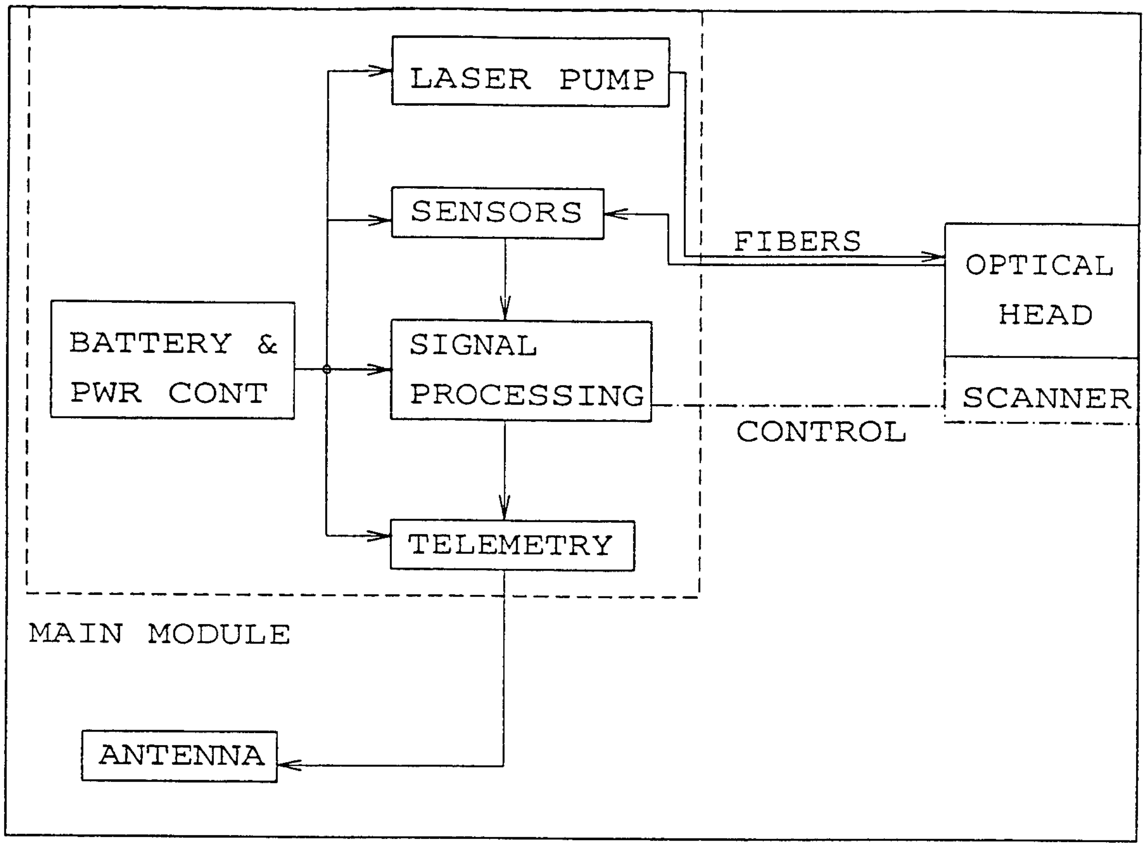

FIGURE 3.1.4. Experiment Block Diagram. The scanner and its control cables are an optional part of the experiment.

3.1.3.3 Operations Interfaces (i.e., CSTC, AFSCN, Ops Center, Targets, etc.). The only requirement would be for an S-band receiver during the boost phase of the rocket and for a data recorder. There would be no interaction with the payload during the experiment. Data would be reduced at the experimenter's home facility.

3.1.4 Experiment Performance Predictions. Readily available electronics limit the data resolution to 8 bits per range bin, so the maximum dynamic range is 128 (assuming the least significant bit is not useful). The 40 micro-Joule pulse should give a signal to background noise ratio of at least 10 with a plume reflectance as low as 0.1%. The detector will be operating well above the photon counting regime.

3.1.5 Technical Risk.

3.1.5.1 Complexity. As shown in Figure 3.1-4, this system is quite simple. It has a small and simple laser, simple detectors, and simple electronics. It is entirely self-contained, though it is in two connected modules.

Figure 3.1-4 Experiment Block Diagram. The scanner and its control cables are an optional part of the experiment.

3.1.5.2 Flexibility. The experiment can be configured in a number of ways, depending on the host vehicle and the data desired. It is small and light enough that it can be fit many different vehicles without affecting their performance significantly.

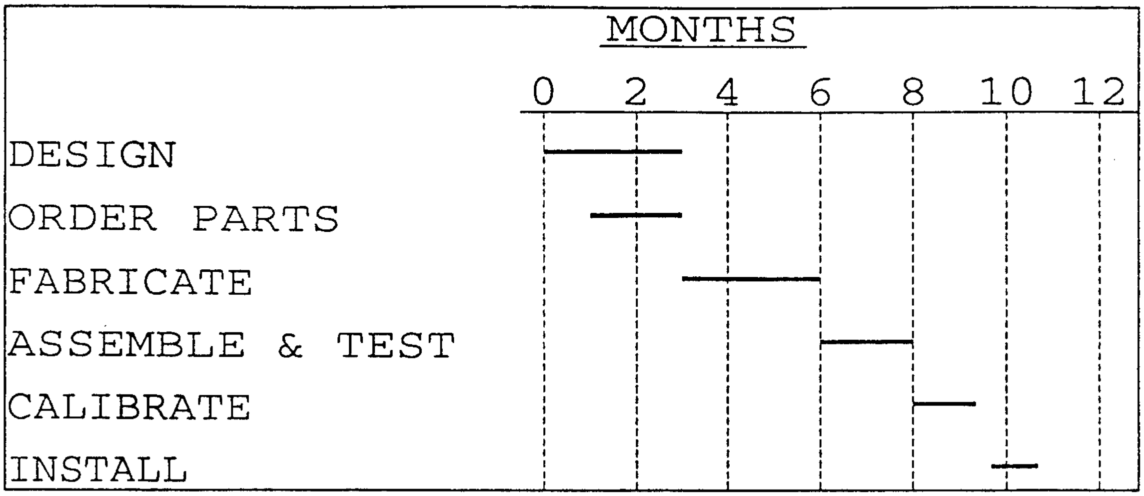

FIGURE 3.1.5. Schedule

3.1.5.3 Equipment Maturity. Most of the components in this experiment are available off the shelf, though most will need to be repackaged for this application. Some electronics will probably need to be custom designed, as will the optics, but they are well within the current state of the art. No new technology is involved.

3.1.6 Schedule (Milestone Chart, etc.). The plume reflection experiment schedule is shown in Figure 3.1-5. The schedule is based on efficient work schedules, and could be compressed slightly (with attendant higher risk and cost) by perhaps 25%.

Figure 3.1-5 Schedule.

3.1.7 Summary. The plume reflection measurement experiment is a low-cost, low risk, and timely approach to obtaining critically needed plume reflection data.

3.2 Suborbital Rocket ATP Experiment. Lockheed primarily developed this experiment concept based on their experience with the SPARCS Sounding Rocket and Beam Aboard Rocket (BEAR) programs. This approach involves launching two suborbital rockets with one carrying the payload and the other acting as a target. Appendix 4 contains Lockheed's concept. The major advantages of this experiment are the payload can be recovered and flown again, the short mission duration of minutes places substantially reduced demands on the support bus, and the per experiment sortie cost is relatively low. One other important aspect is the experiment is conducted in a space environment operating in micrograity and beginning an engagement against earth background, but continuing it through the limb and against deep space.

3.2.1 Experiment Objective. This experiment is intended to address passive acquisition of a target against an earth background, track the plume handover to active track of the rocket hardbody, and precision pointing of a marker laser at the hardbody.

3.2.2 Experiment Description.

3.2.2.1 General. The experiment payload will be mounted on top of an Aries or Black

3.2 Suborbital Rocket ATP Experiment. Lockheed primarily developed this experiment concept based on their experience with the SPARCS Sounding Rocket and Beam Aboard Rocket (BEAR) programs. This approach involves launching two suborbital rockets with one carrying the payload and the other acting as a target. Appendix 4 contains Lockheed's concept. The Major advantages of t his experiment are the payload can be recovered and flow again, the short mission duration of minutes places substantially reduced demands on the support bus, and the per experiment sortie cost is relatively low. One other important aspect is the experiment is conducted in a space environment operating in microgravity and beginning an engagement against earth background, but continuing it through the limb and against deep space.

3.2.1 Experiment Objective. This experiment is intended to address passive acquisition of a target against an earth background, track the plume handover to active track of the rocket hardbody, and precision pointing of a marker laser at the hardbody.

Table I ATP Issues

Comprehensive List of ATP Issues

More Important	**A. Plume -to- Hardbody Handover**
	- EPD-5: Plume -to- Hardbody Handover
	B. Fine Tracking and Illumination
	- EPD-6: Illuminator Point-Ahead / Active Track Handover
	- EPD-7: Hardbody Discrimination / Active Fine Track
	- EPD-8: Precision Point-Ahead / Aimpoint Designation
	- EPD-14: Active Fine Track or M/C Objects
	C. Integrated ATP Performance
	- EPD-10: Autonomous Sequencing
Important	**D. Phenomenology Data Collection**
	- EPD-17: General Plume Phenomenology
	- EPD-18: Phenomenology Data Collection
	E. Space Operability
	F. Rapid Retargeting and Multiple Target Fire Control
	G. High Power and Weapon Interface With ATP

The aim of the experiment is to reolve the important issues as identified in Table I

Table II Resolution of Critucal Issues by Suborbital Experiment

RELEVANCE OF TECHNOLOGY DEVELOPMENT - AND - VALIDIATION TESTING

- o SIGNIFICANT CONTRIBUTION TO TECHNOLOGY DEVELOPMENT
- ◯ PRIMARY DEVELOPMENT OF TECHNOLOGY
- • SUPPLEMENTARY VALIDATION TESTING
- ● PRIMARY PROGRAM FOR VALIDATION TESTING

ATP & RELATED ISSUES — SDIO ISSUES [EPD Equiv]

	PROGRAMS	Multi-Target Acquisition / Track (A1) 2, 18	Plume -to- Booster Handover (A2) 5, 17	Passive Tracking (A3) 3, 4, 18	Active Tracking (A4) 6, 7	Aimpoint Selection & Maint. (A5, A13) 7	Damage Assessment (A6, A14)	ATP for MCID (A7, S4) 16	Boresight, Align., & Stabilization (A8, A16) 8	Multilevel Rapid Retargeting (A9, A18) 1	Pointing, Disturbance Rejection (A10, A17) 9	LOS Retargeting in FOV (B3, A12) 14	High Power Effects on ATP (H2)	Single Target Autonomous Control (S2) 10	Multi-Target Autonomous Control (S3)	Weapon Effectiveness (S4)	DE Sensors in Discrimination (E1) 12, 13, 18	DE Sensors in Surveillance & Tracking (E2) 11, 15	*Requirements Definition & Systems Engineering*
ASSUMED BASELINE	Advanced Tracking / TFE / Illuminator	o	o	◯	◯	o		o						o	o				
	Fire Control / AMDF	◯	◯			◯	o	o						o	◯		o	o	
	Pointing & Control / SPICE / IPSRU								◯	o	◯	o							
	Rapid Retargeting / R2P2								o	◯	o	o			o				
	ADAPT Phase I	o	o	o	o	o	o	o	o	o	o	o	o	o	o				◯
	ADAPT Phases II - III (Ground)	•	•	•	•	•		o•	•	•	•			◯		o		o	●
	High Power Tests								◐				◯						
	Other DE Technology										o•	◯				◐		◯	
	Other Non-DE	o	o	o	o		o	o			o						o	o	
	Balloon Tests (Phenomenology)		o	o				o									o	o	
EAP OPTIONS	*Balloon Program*																		
	Low Cost Satellite Program																		
	ALTAIR (or equivalent) Program																		
	Sub-Orbital Flight Program		●	●	•	•		o	●		●			•					
	DE Space Experiment - ATTD (Zenith Lite)																		

ATP Issues — Beam Control Issue | High Power Issue | DE Systems Issues | Expanded Utility Issues

3.2.2 Experiment Description

3.2.2.1 General.

The engagement selected for the sub-orbital experiment was chosen to give acquisition against an earth background and to track a thrusting target to above the horizon. To analyze this an Aries was loaded down with a heavy payload and a Starbird was launched from an adjacent pad at WSMR. The timing is such that when the shroud is popped off the Aries the Starbird has finished 2nd stage burn and is in the coast mode. The Payload is mounted on an existing SPARCS bus with a recovery capability. This payload is pointing to the area where the third stage ignition will occur in the Starbird. The Payload then acquires the Starbird third stage and tracks the

Starbird through the burnout of the fourth stage, The latter parts of the third stage and the fourth stage are actively tracked and a marker beam is pointed at the target to simulate HEL pointing.The experiment payload will be mounted on top of an Aries launch vehicle equipped with a 44 inch shroud. It would be mated to an existing SPARCS bus that provide normal spacecraft services.

The engagement is shown in figures 3.2.1 a-

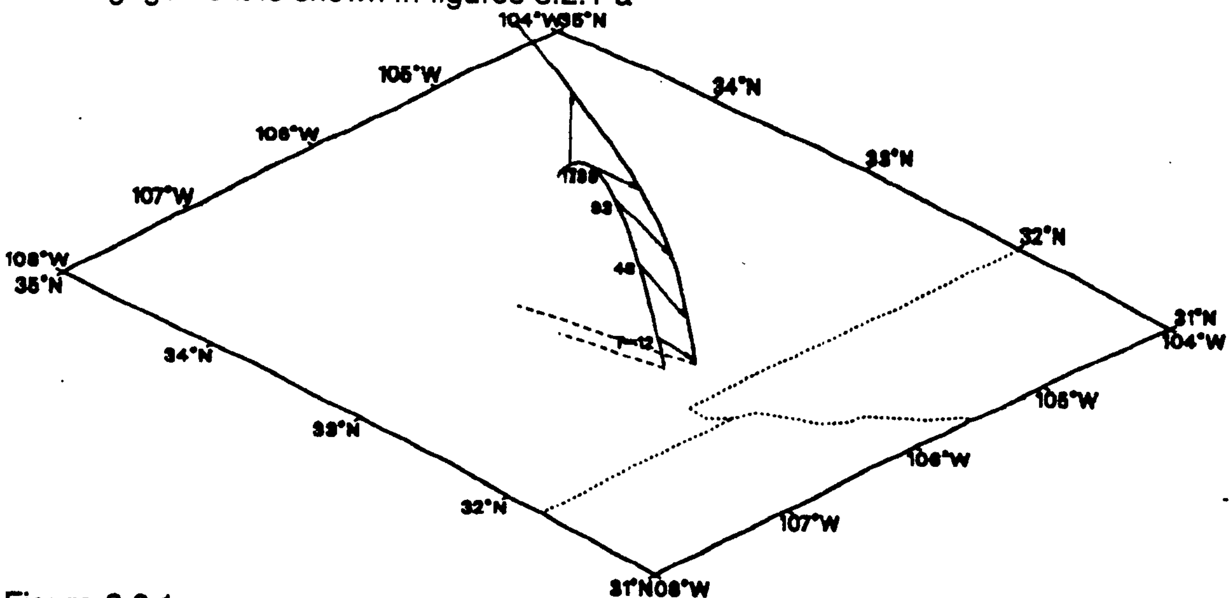

Figure 3.2.1a

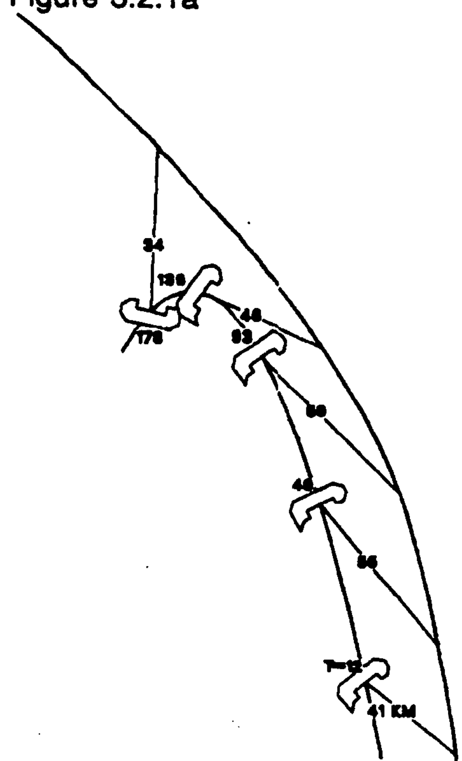

Figure 3.2.1 b

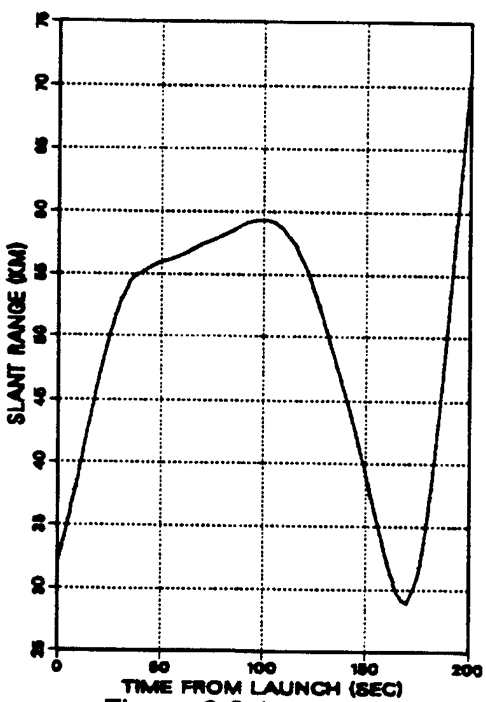

Figure 3.2.1 c

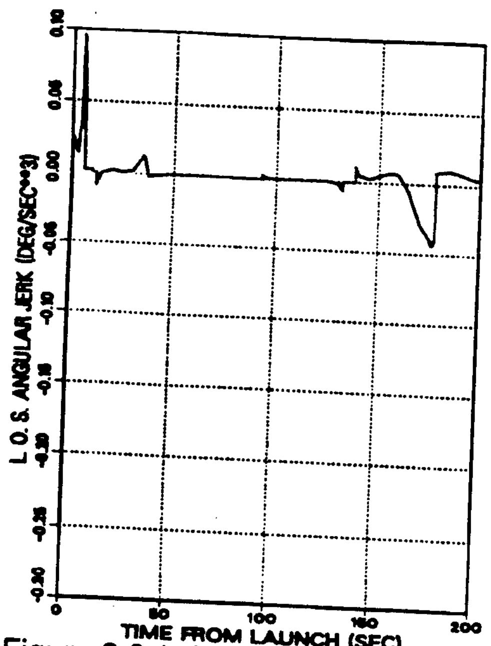

Figure 3.2.1 d

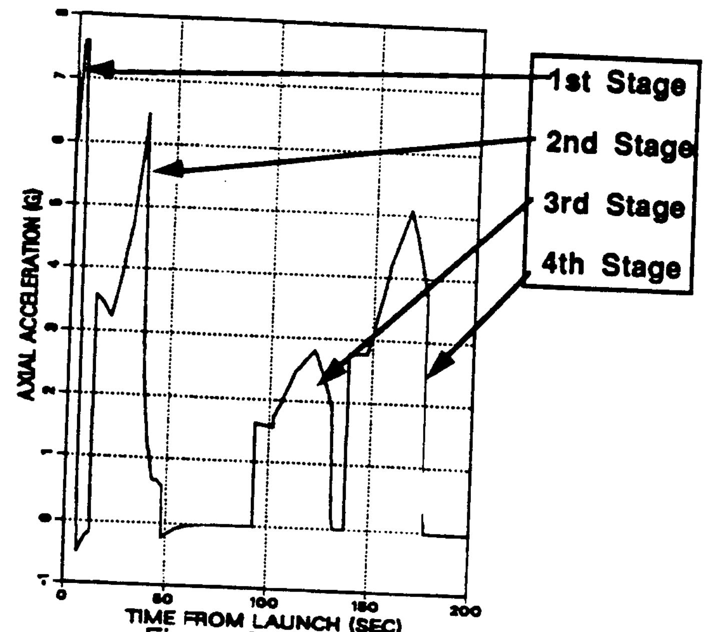

Figure 3.2.1.e

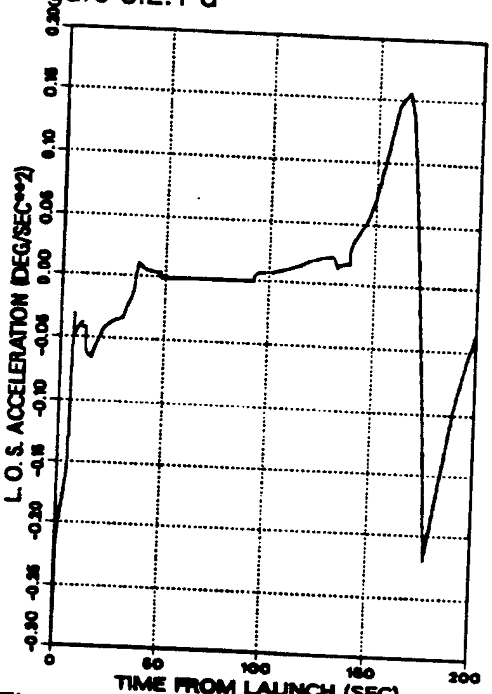

Figure 3.2.1.f

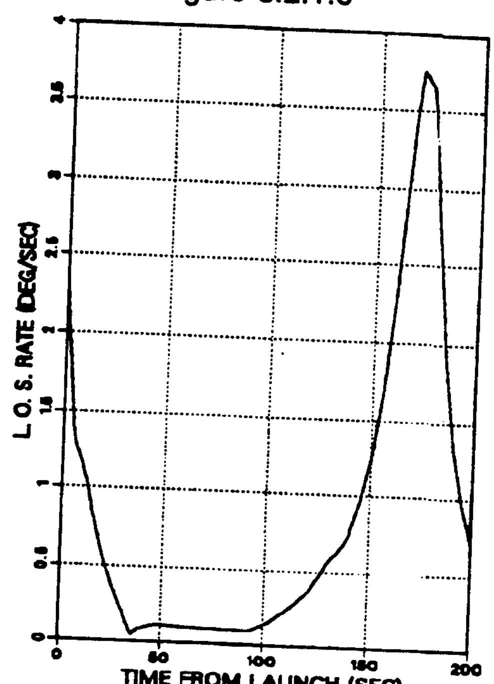

Figure 3.2.1.g

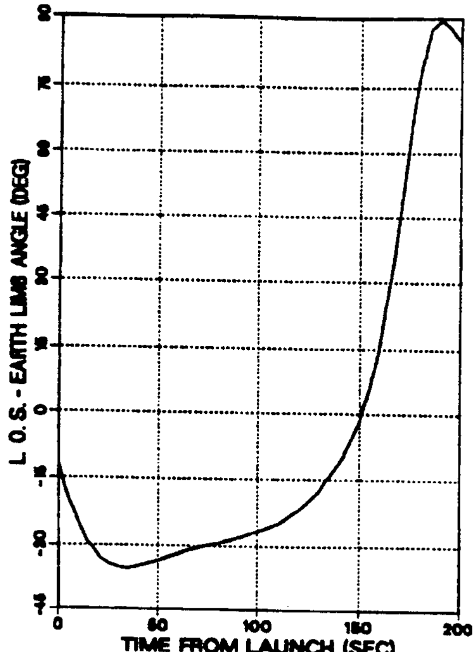

Figure 3.2.1.h

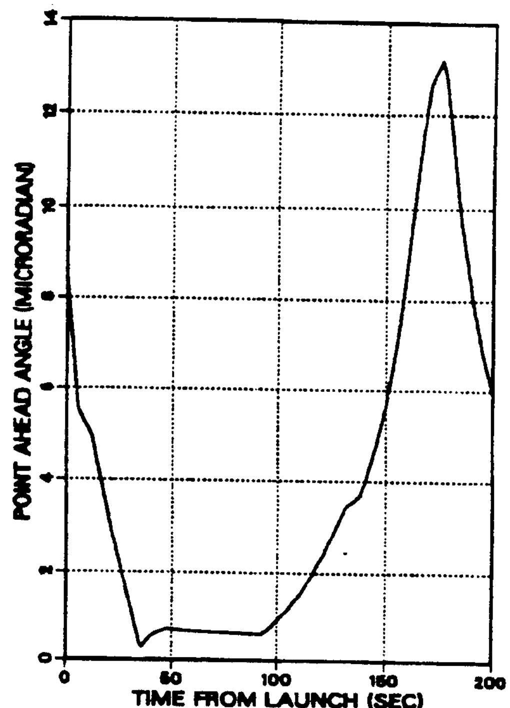

Figure 3.2.1.i

This engagement geomtery will allow pointing at the target with an accuracy of less than 1μradian using a 1 Hz bandwidth tracking and pointing system. There still will be a highbandwidth stabilization system that uses a fast steering mirror and the inertial sensors from Starlab/ALTAIR

To achieve active tracking at these short ranges (30-50Km) a small sensor aperture can be used (20 cm) and a bulidable illuminator (50mJoules at 20 pps) can be used. An isometric of the payload is depicted in Figure 3.2-1. The basic instrument suite consists of an acquisition sensor which is currently planned to be a "Space Maverick" seeker as flew on the DELTA series. This could be replaced by a SWIR sensor or a visible CCD TV camera.. A 20 cm telescope is the primary aperture and it is shared by visible intermediate and fine trackers, an illuminator laser, and steering mirrors and internal alignment sensors. The optical layout is shown in Figure 3.2-2.

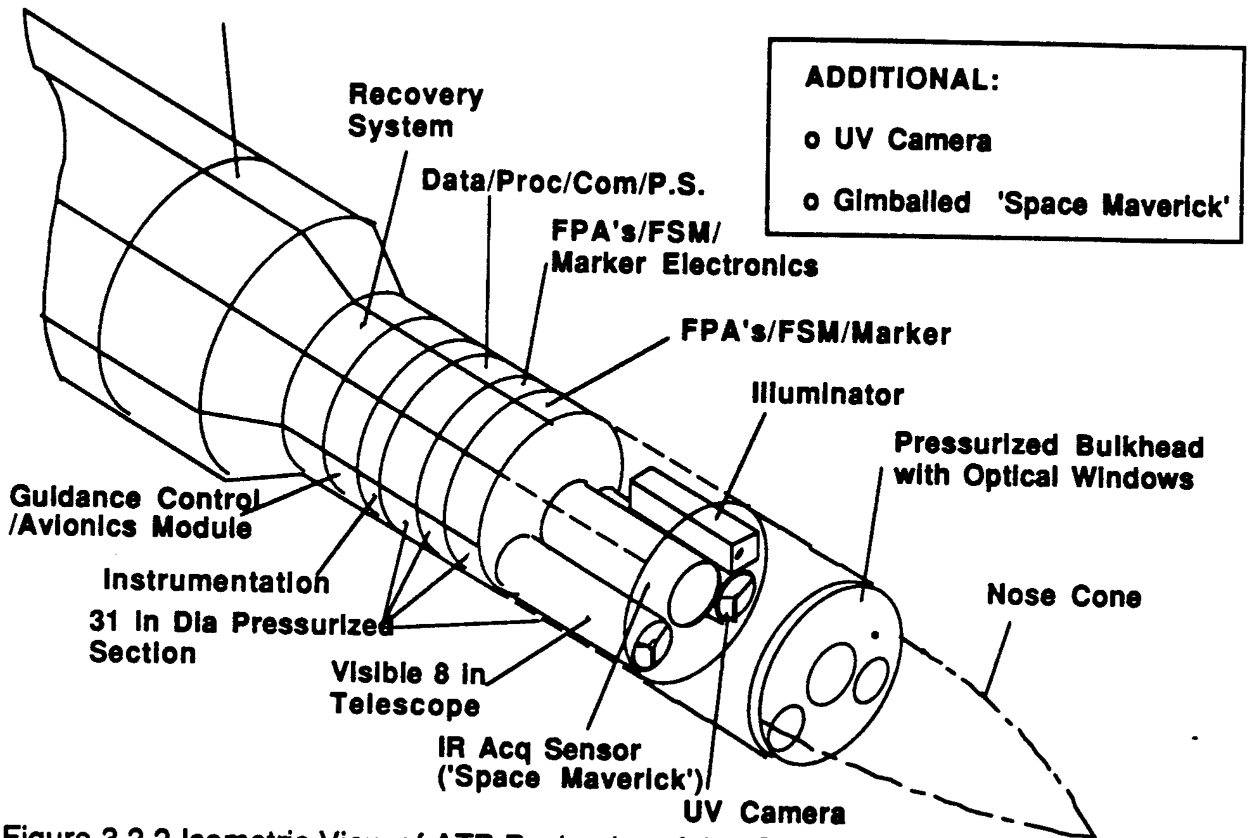

Figure 3.2.2 Isometric View of ATP Payload on Aries Sounding Rocket

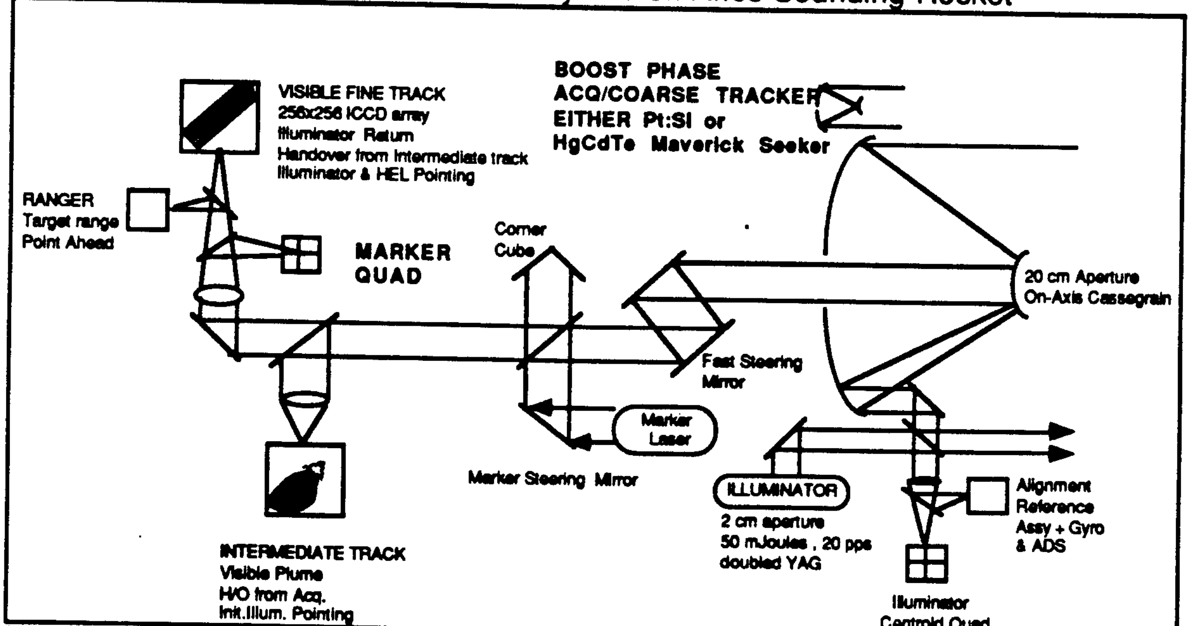

Figure 3.2.3 Optical Layout of ATP Payload

3.2.2.2 Instrument Descriptions.

3.2.2.2.1 Sensors. The sensors for this experiment will be relatively standard, off-the-shelf items. They will not be fully space qualified nor will their design life be excessive considering the nature of the experiments.

Table III Payload Sensor Parameters

FINE TRACKER	
Wavelength	= 0.5321 μm ± 25 Å
Aperture D	= 20 cm
IFOV	= 5 μrad
FOV	= 2 mrad

ACQUISITION (Maverick)	
Wavelength	= 7 - 11 μm
Aperture D	= 7.5 cm
FOV	= 2°x2° -> 1°x1°
Scanned Vol.	= 45°
FPA	= HgCdTe

COARSE TRACKER	
Wavelength	= 0.45 - 0.75 μm
Aperture D	= 20 cm
IFOV	= 20 μrad
FOV	= 8 mrad

ILLUMINATOR	
Wavelength	= 0.5321 μm
Aperture D	= 2 cm
Pulse: Rate	= 20 - 60 pps
Power	= 50 mJ

MARKER LASER	
Wavelength	= 0.83 μm
Aperture D	= 50 mW

3.2.2.2.2 Illuminator Laser. The laser for the sub-orbital ATP experiments will be relatively low power, doubled YAG laser. The closest target encounter geometry of the experiments will be modulated to ensure sufficient laser energy is reflected from the unenhanced hardbody. The type of laser planned is fairly common and has the sam specifications as the laser built by MDAC for the DELTA series. Their laser is probably not suitable but this type of laser is a significantly downgraded SSLRS device.

3.2.2.2.3 Steering Mirrors. The experiment will be equipped with a fast steering mirror estimated to operate at 300 Hz with a maximum throw of 1 mrad. It controls the optical line-of-sight during fine tracking. Illuminator and marker steering mirror are also included for maintaining system alignments and to enable precision pointing.

3.2.2.2.4 Track Processor. In addition to the SPARCS bus controller and processor, two track processors are included in the payload. It will control the line-of-sight through the optical line-of-sight within the control constraints of the fast steering mirror. Off loads are made to the SPARCS control system for coarse attitude control.

There is also a track processor included in the Maverick seeker as well as the two Hugh4es Dual Mode Tracklers (DMT's)

3.2.2.2.5 Strap Down Alignment Reference The inertial reference for the system will be the PISA (Pallett Inertial Sensor Assembly) from Starlab. This is an existing piece of qualified spoace hardware.

3.2.2.3 SPARCS Bus Description.

3.2.2.3.1 Attitude Control, Guidance and Navigation. A GN2 cold gas system provides coarse attitude control of the SPARCS bus. This system can accommodate the line-of-sight rates and acceleration of ?????. It is equipped with an inertial sensor for attitude determination but either GPS (CAN GPS GET ATTITUDE??) or a star tracker may be necessary to update the IMU accuracy sufficiently to acquire the target.

3.2.2.3.2 Command and Data Handling. A Megabit downlink fron the bus supports video control of the payload, a 2-way link exists with the rocket during all stages of the flight. In addition to a set of real-time health and status data, a command destruct link is also maintained. Experiment data will be preserved on-board using a low cost recorder, such as a VHS type system. The power of the laser should eliminate safety concerns for positive control, but a man-in-the-loop is a standard SPARCS capability. and will be maintained for an ATP experiment.

3.2.2.3.3 Electrical Power. With the short mission duration and a low power laser, the existing SPARCS ????? battery and a 28 volt system, with a peak power capability of ???? watts, should be adequate with little or no modifications.

3.2.2.3.4 Parachute recovery system. The SPARCS is equipped with a payload recovery system that has been used many times and thuis bus is re-used often at WSMR.

3.2.2.4 Software. The payload will use a commercially available processor and an array processor board for the high speed digital control.

3.2.2.5 Mass and Power Summary

3.2.3 Experiment Operations concept. This approach accomplishing a low cost space experiment will utilize two sounding rockets launched in near proximity to one another, both spatially and temporarily. The launches will occur at at the White Sands Missile Range (WSMR) where all necessary assets already exist, since WSMR routinely launches sounding rockets, with particular experience with the SPARCS platform and a great deal of laser operations.

3.2.3.1 Engagement Requirements. As stated earlier, the rocket carrying the payload would be an Aries vehicle that can place the experiment package on the desired lofted trajectory. The target rocket will be a Starbird class, multi-stage rocket that has sufficient burn time to permit tracking during boost a relatively close range (<50Km).

The rocket carrying the payload will launch several seconds before the target rocket . After it burns out and its shroud has jettison, it would perform a pitch-over maneuver to passively acquire the thrusting target rocket. Active illumination, plume-to-hardbody handover, and precision pointing will all occur during boost. As this sequence occurs, the target rocket would actually overtake and be at a higher altitude than the payload rocket causing a transition of background from earth to space. After the engagement, the payload would be parachuted to earth.

The target rocket will probably not as equipped with a scoreboard in order to minimize cost. An existing WSMR ground station is necessary and will be utilized to control the flight operations.

3.2.3.2 Engagement Events and sequences. The general characteristics of the engagement was stated in paragraph 3.2.3 From launch to payload recovery, the expected turn-around time will be approximately six minutes. There may be a capability build into the payload to conduct preliminary experiments from a balloon but insufficient effort has been placed into that application to assess its viability.

3.2.3.3 Operations Interfaces. Since this experiment is sub-orbital and conducted with a relatively limited geographic area, over a short period of time, the only real interfaces with the experiment occur through a mission operations center. This center can easily accommodate range control and safety, launch operations, real-time data links, and other necessary operations. Line-of-sight considerations will likely prompt the center from being at the Phillips Laboratory (PL). Although fiber optic or microwave links from WSMR stations to PL might make it possible. Given the existing capabilities and short mission duration, it is more cost effective to conduct operations out of WSMR..

Integration of the payload package with the SPARCS bus and refurbishment activities will take place at PL. After transport to WSMR, a set of baseline tests will be performed using test set hardware that could be packaged in a van and connected to the hardware at WSMR or during integration/refurbishment at PL.

3.2.5 Technical Risk. In assessing technical risk, several factors must be addressed and given the maturity of the concept, these estimates are very qualitative.

3.2.5.1 Complexity. This experiment incorporates nearly all experiment systems found in ALTAIR. Although individually they may be smaller or simpler instruments when the system is integrated it becomes quite complex with all of the assort ed software and control loops, instrument alignments, and numerous other factors. One important factor is the small payload shroud envelope. Another is the launch loads that an Aries vehicle exposes this sensitive payload to. Even though the precision pointing performance being sought is not the same as ALTAIR, it is a significant challenge and any hope to achieve even those reduced goals, demands a complex interactive payload.

3.2.5.2 Flexibility. The sub-orbital ATP Rocket approach provides the opportunity to recover, refurbish, and refly the payload. This is a very important feature and relatively affordable to take advantage. The short duration of the engagement and mission does not accommodate in-flight reprogrammable processors, but they can certainly be reprogrammed between flights. Having a recoverable payload also permits the change-out of instruments, or perhaps changing optical filters as further requirements demand other types of data. This experiment's sensor suite might permit some mid-course experiments since the rockets are inf act flying ballistic trajectories. flexibility is a strength of this concept.

3.2.5.3 Equipment Maturity. The sensor suite, illuminator, and optics are all of low technical risk. Either the hardware exists on-the-shelf or the design exists. The short duration of time the experiment hardware actually is required to operate in space permits significant relaxation of space qualification requirements, further expanding the list of existing hardware.

3.2.6 Schedule.

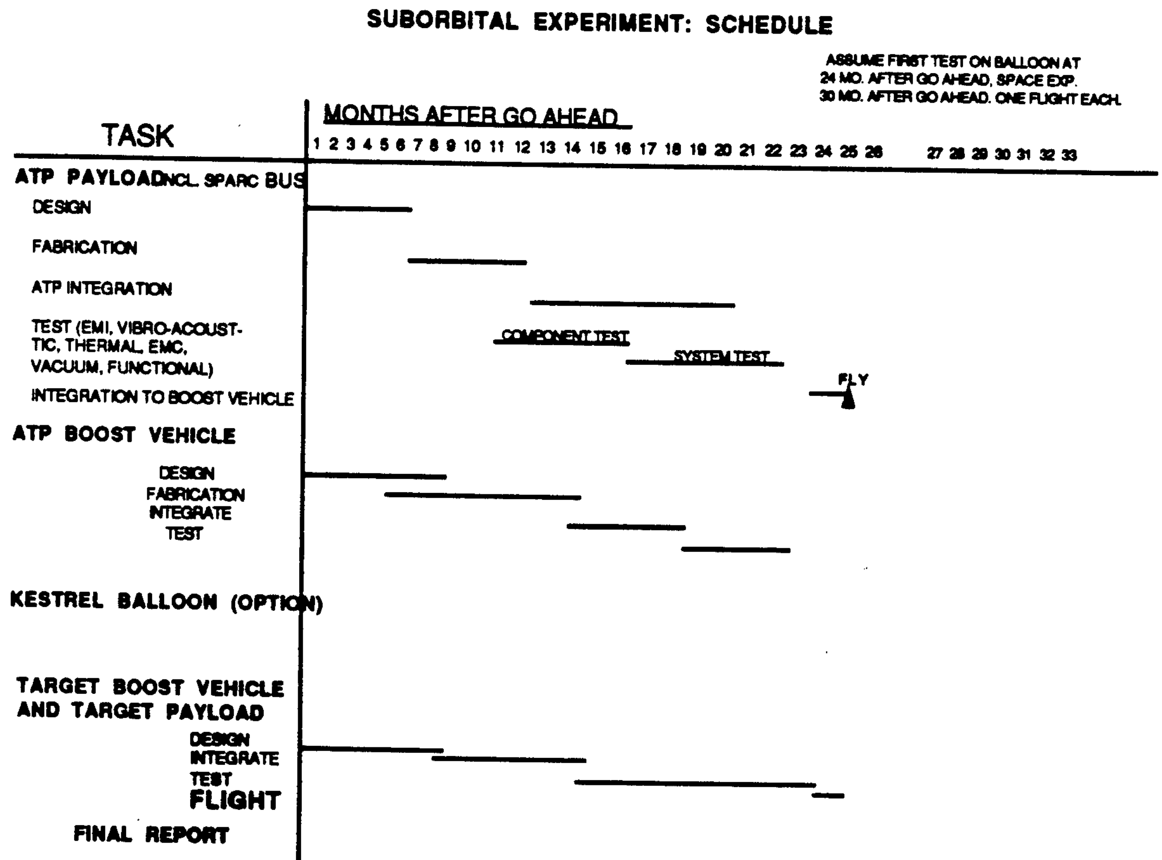

Figure 3.2.4 Suborbital ATP Experiment Schedule

experiment's sensor suite might permit some mid-course experiments since the rockets are in fact flying ballistic trajectories. Flexibility is a strength of this concept.

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3.2.6 Schedule. See Appendix 4.

3.2.7 Summary. A combination of well understood hardware components from the SPARCS program and a great deal of experience in ATP experiment design by Lockheed makes this concept attractive with inherent cost, schedule, and risk reduction benefits of recoverable, sub-orbital rockets. The approach recognizes the need to resolve the critical ATP issues in a cost effective and timely manner. It does so by going to space, where space-based platforms of a DEW system must operate, yet reduces the cost of building a space system by minimizing exposure to that environment. The flexibility of the approach brings the experiment target encounters in close proximity permitting the use of simpler, more mature instruments and hardware to do the essentials of ATP. Finally, from an operations point of view, it is a much simpler task to manage.

3.3 UV Plume and Background Phenomenology Experiment. This, as well as the following experiment, were configured to address a limited but important set of issues generally felt to be resolvable with relatively low cost, small satellite experiments. These experiments also are ones which would be difficult to conduct from rocket or balloon platforms or from the ground. Thus, if the decision were made to accomplish a basic subset of the original ALTAIR objectives with a balloon experiment, for example, then these experiments might also be selectable as affordable means of filling remaining data gaps. The UV Plume and Background Phenomenology Experiment would gather the two types of ultraviolet data which its name implies, from an altitude permitting unattenuated propagation and from a viewing aspect characteristic of space-based laser and kinetic energy platforms.

3.3.1 Experiment Objectives. The purpose of this experiment is twofold: (1) to provide long term solar-blind UV background data covering all latitudes and seasons; and (2) to characterize the UV plume emissions of a set of cooperatively launched (primarily liquid fueled) rockets. With a 40 cm UV telescope aperture, approximately 6-10 μrad pixel resolution from its wideband UV imager and less than 1 nm spectral resolution from its slit spectrograph, the experiment should provide resolution of many of the UV issues identified recently by SDIO (see Appendix 3.3-a). Maximum use was made of existing instrument hardware, existing satellite designs and already planned launcher capability which together should allow a launch approximately 28 months after program initiation, in time for the data to be of significant value for design definitization for kinetic energy as well as directed energy systems.

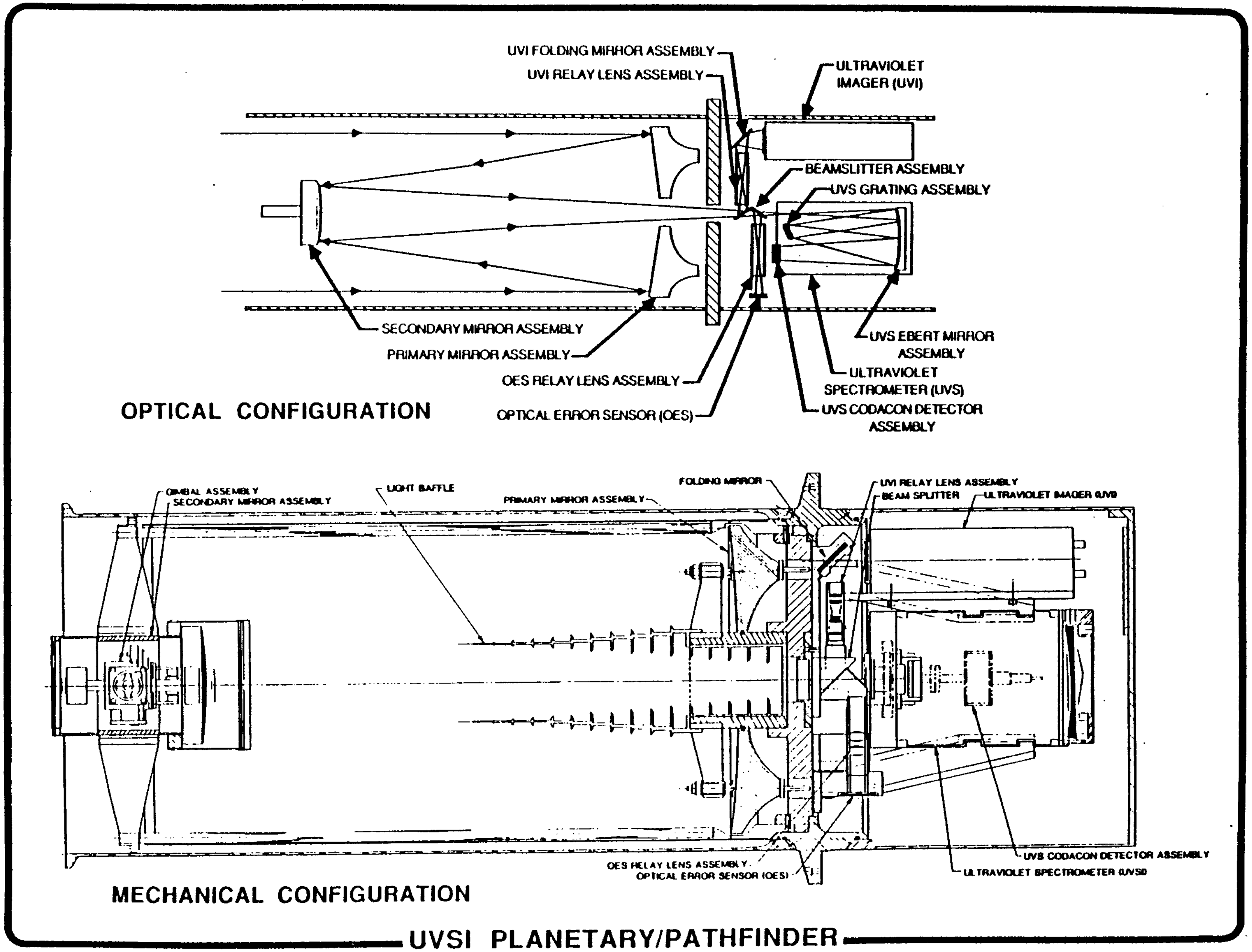

UVI FOLDING MIRROR ASSEMBLY
UVI RELAY LENS ASSEMBLY
ULTRAVIOLET IMAGER (UVI)
BEAMSLITTER ASSEMBLY
UVS GRATING ASSEMBLY
SECONDARY MIRROR ASSEMBLY
PRIMARY MIRROR ASSEMBLY
OES RELAY LENS ASSEMBLY
OPTICAL ERROR SENSOR (OES)
UVS EBERT MIRROR ASSEMBLY
ULTRAVIOLET SPECTROMETER (UVS)
UVS CODACON DETECTOR ASSEMBLY
OPTICAL CONFIGURATION
GIMBAL ASSEMBLY
SECONDARY MIRROR ASSEMBLY
LIGHT BAFFLE
PRIMARY MIRROR ASSEMBLY
FOLDING MIRROR
UVI RELAY LENS ASSEMBLY
BEAM SPLITTER
ULTRAVIOLET IMAGER (UVI)
OES RELAY LENS ASSEMBLY
OPTICAL ERROR SENSOR (OES)
UVS CODACON DETECTOR ASSEMBLY
ULTRAVIOLET SPECTROMETER (UVS)
MECHANICAL CONFIGURATION
UVSI PLANETARY/PATHFINDER

Figure 3.3-1

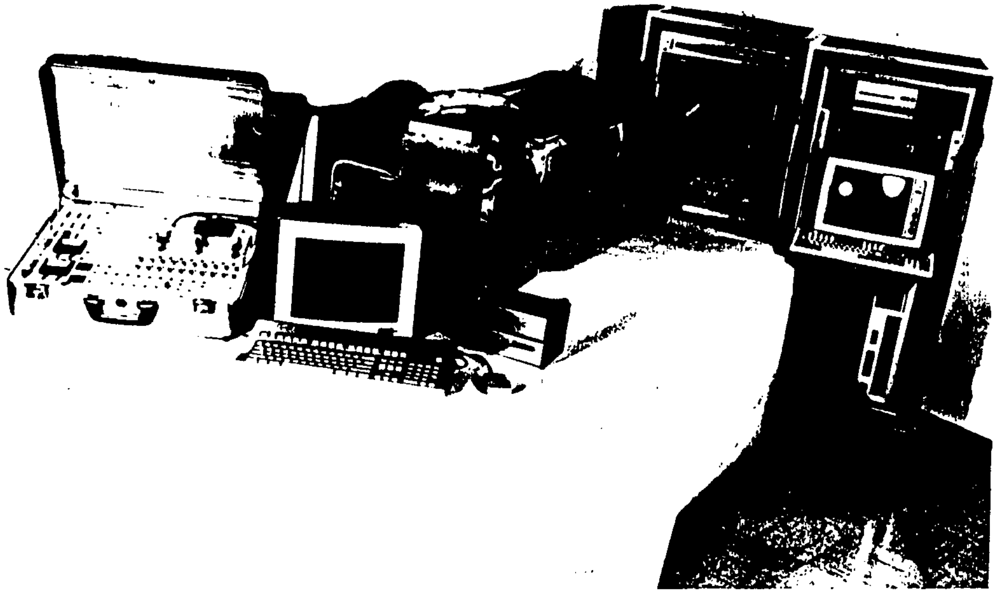
3.3.2 Experiment Description.

3.3.2.1 General. The experimental hardware consists of the 40 cm ungimballed UV telescope with associated internal tracking mirror, UV images and UV spectrograph, a smaller aperture, wider field of view visible acquisition telescope, and a standard modular satellite bus. The launcher is an improved Pegasus or Taurus, and the ground operations would be handled by CSTC. The satellite would fly in an approximately 300 nautical mile circular, near-polar orbit with the telescope nadir pointing except for limited duration limb or plume viewing.

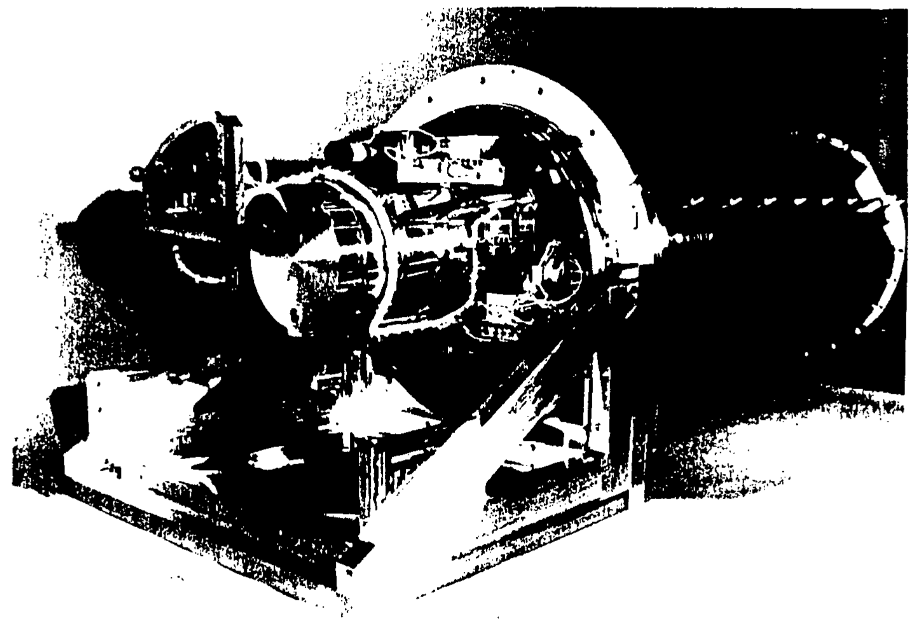
3.3.2.2 Instruments.

3.3.2.2.1 Sensor (UV Telescope with Imager and Spectrograph). The apparatus would either be the existing Ultraviolet Spectrometer and Imager (UVSI) instrument developed by the University of Colorado, Loral, and the Jet Propulsion Laboratory, or a close derivative design incorporating some lightweighting. UVSI and its associated GSE (Ground Support Equipment) was fabricated in the late 1980s for SDIO's PATHFINDER experiment, which was not flown as planned on the shuttle due to funding limitations. A detailed description of the instrument and its hardware heritage from earlier successful space missions is found in Appendix 3.3-B; a short synopsis is given here.

Figure 3.3-1 Telescope Configuration.



UVSI FLIGHT UNIT AND GSE - JANUARY, 1990



UVSI SPECTROMETER

Figure 3.3-2

Figure 3.3-1 shows the telescope configuration, which is a tilted aplanat (hyperbolic primary and hyperbolic secondary mirrors) with a diameter of 40 cm and an effective focal length of 240 cm. The secondary mirror is mounted on a shaft which pivots about a point behind its vortex. By using actuators to move the secondary, targets within a 10 mrad ($\pm$5 mrad) cone about the telescope design allows investigation of localized phenomena within a fairly large field of view to be investigated with the high resolution normally associated

Figure 3.3-2 The Telescope with the Focal Plane Assembly.

with nearly on-axis viewing. Photographs of the telescope with its focal plane assembly and with its GSE are shown in Figure 3.3-2; the overall length of the telescope and focal plan assemblies is 151 cm (59.5 in).

There are basically three focal plan assemblies which share light from the full aperture:

a. An optical error sensor which provides feedback for the centering function just described.
b. The slit spectrograph.
c. The UV imager.

The key parameters, as well as those of the telescope and overall assembly, are summarized in Table 3.3-1. Note that the spectral bandwidth of the UV imager is currently

TELESCOPE

(CONCENTRIC FOLDED TWO MIRROR WITH TILTING SECONDARY FOR IMAGE STABILIZATION)

OPTICAL SPECIFICATION	TILTED APLANT
FOCAL LENGTH	2.4 m
APERTURE	39.6 cm
CENTRAL OBSCURATION	15.0 cm
FOCAL PLANE SCALE	11.5 µm per arc sec
IMAGE STABILIZATION SYSTEM	CLOSED-LOOP OPTICAL FEEDBACK
ACQUISITION FIELD OF VIEW	10 milliradian DIA (35 arc min)
DETECTOR	ITT QUADRANT ANODE MCP
PHOTCATHODE	KCsSb(Bi-alkali)
SPECTRAL RESPONSE	200 nm - 550 nm
BANDWIDTH	30 Hz

SPECTROGRAPH

OPTICAL CONFIGURATION		EBERT FASTIE
FOCAL LENGTH		250 nm
GRATING	(PRESENT CONFIGURATION)	900 g/mm BLAZED AT 250 nm
DISPERSION	(PRESENT CONFIGURATION)	4.21 nm/mm
RESOLUTION	(PRESENT CONFIGURATION)	0.37 nm
COVERAGE	(PRESENT CONFIGURATION)	110 nm - SIMULTANEOUS
WAVELENGTH RANGE	(PRESENT CONFIGURATION)	210 - 320 nm
WAVELENGTH RANGE POSSIBLE		160 - 350 nm
DETECTOR		LASP CODACON MCP
ANODE		CODED ARRAY
NUMBER OF CHANNELS		1024
PIXEL SPACING		0.025 mm
PHOTOCATHODE		CESIUM TELLURIDE

UVS SENSITIVITY

(FLUX REQUIRED TO PRODUCE 1 COUNT/SEC)

2.5×10^{-19} WATTS/cm^2	200 nm
4.0×10^{-19} WATTS/cm^2	250 nm
1.7×10^{-18} WATTS/cm^2	300 nm

IMAGER

(COHU CAMERA)

OPTICAL SPECIFICATION	X2.9 RELAY LENS FOLDING SYSTEM
FORMAT	754 X 488 PIXELS
FIELD OF VIEW	4 X 3 milliradian
PHOTOCATHODE	RUBIDUM TELLURIDE
SENSITIVITY	2×10^{-15} WATTS/PIXEL @270 nm

UVSI CHARACTERISTICS

UVS (INCLUDING TELESCOPE)	
MASS	305 LBS
LENGTH	59.5 INCHES + APERATURE COVER
WIDTH	23.0 INCHES
POWER	27.0 WATTS
UVI	
MASS	14 LBS (+/- 2 LBS)
POWER	8.0 WATTS
INTERFACE BOX	
MASS	35 LBS (+/- 10 LBS)
LENGTH	26.0 INCHES
WIDTH	8.0 INCHES
HEIGHT	TBD
POWER	TBD WATTS

ULTRAVIOLET SPECTROGRAPH-IMAGER SUMMARY

Table 3.3-1

250-300 nm, but is capable of being extended somewhat. Other relatively easy changes could also vary the spectrograph coverage and resolution. The integration time can be varied from approximately 0.01-3 sec and is programmable with mirror modifications.

3.3.2.2.2 Visible Acquisition Telescope. This instrument has not been sized in detail or selected, although no problem is anticipated in finding an available off-the-shelf unit capable of being space qualified or already so. The device is estimated to have an aperture size of approximately < 3 inched. Its primary function is to ensure that plumes initially outside the UVSI field of view of 10 mrad (approximately 0.6 deg) can be acquired by moving the spacecraft (which has altitude knowledge and accuracy to approximately < 0.2 deg. It also furnishes low resolution visible image data.

3.3.2.3 Other Spacecraft and Payload Subsystems.

3.3.2.3.1 Standard Satellite Bus. Several "lightsat" buses have been discussed in recent years as means for reducing the cost and development time associated with experiments of the general class being discussed here. One of these is used for the discussion here, the STEP (Space Test Experiments Program) bus currently being fabricated for multiple flights by TRW for the Air Force Space Division. This modular bus is described in detail in Appendix 3.3-C. The core module of this spacecraft provides general housekeeping, telemetry, power management, mass memory, processing, and altitude control functions as described in the Appendix. Mission unique requirements beyond those achievable with the core module capabilities (e.g., expended solar power, more precise altitude control, increased mass memory, propulsion, etc.) are currently being handled on a mission-by-mission basis with generally modular units, also described in the Appendix. The requirements for this particular mission are generally consistent with those configurations under current fabrication and design. The available power should not be an issue within the general constraints of the Pegasus diameter shroud (approximately 42 in diameter) for which STEP was designed, although the exact panel configuration will be subject to a more detailed design effort than is possible here. Likewise, it is probable that a reasonable extrapolation of the existing core solid state memory up to approximately 100 Mbytes of solid state memory will not greatly impact schedule or cost. The key mission unique trades will be in the area of attitude control (the requirement for attitude and knowledge near or just beyond the existing detailed designs, the need for an IMU for short duration off-nadir pointing, and the need for some controlled slew capability), although these requirements are not expected to greatly change the existing configuration. A rough drawing showing the general size and layout of the satellite and payload is shown in Figure 3.3-3. Note the telescopes are fixed to the core module deck (ungimballed). The solar panels around the telescopes are used to provide thermal shielding for the instruments.

3.3.2.3.2 Memory and Data Communications. To keep costs as low as possible it is proposed that the data collection, storage, and downlink capabilities of the experiment be kept generally consistent with the solid state storage and s-band/CSTC design philosophy of STEP. This would allow the range of 100 frames of on-board storage at programmable

SMALL SATELLITE
UV BACKGROUND/PLUME EXPERIMENT

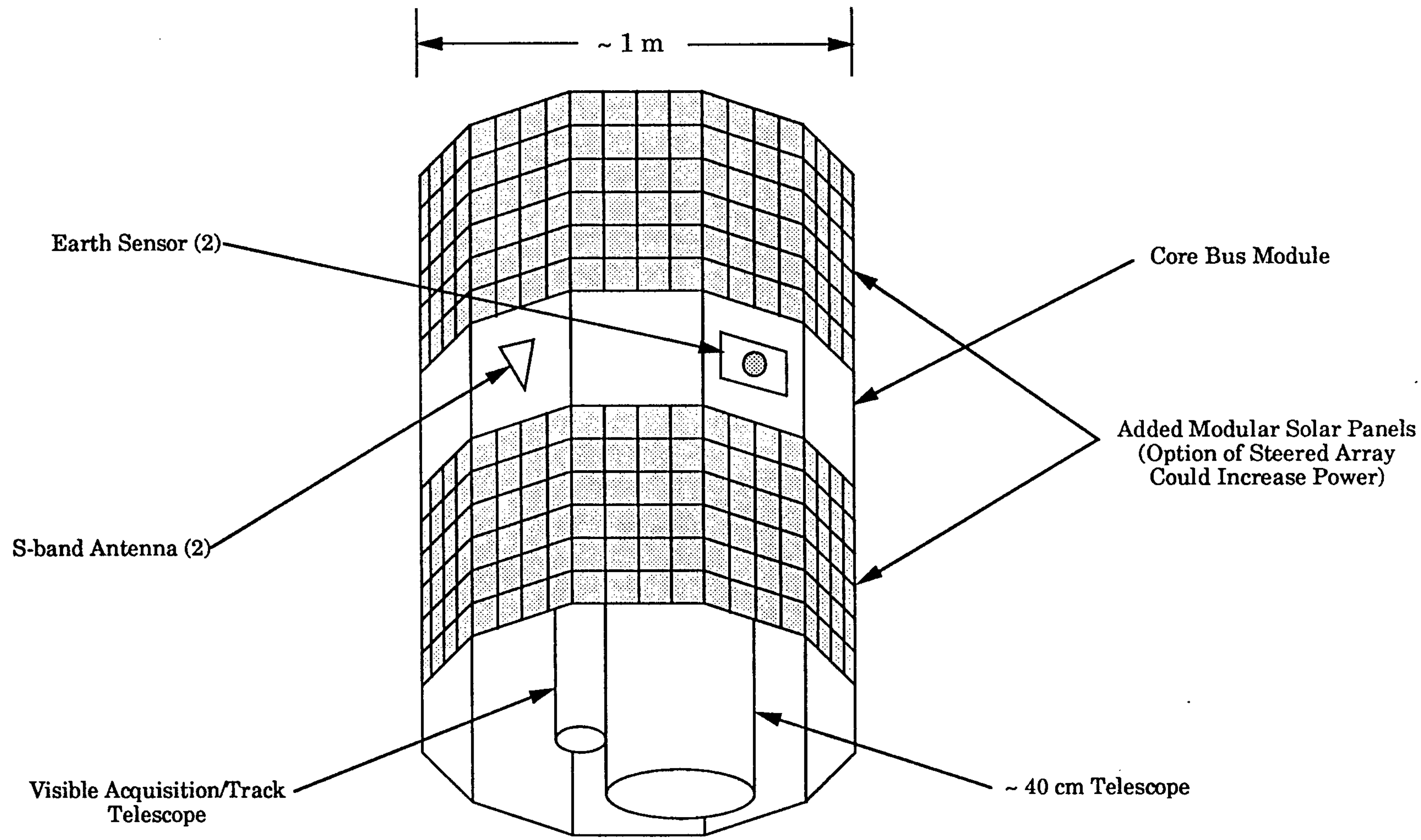


Figure 3.3-3

Approximately to Scale

Figure 3.3-3 Satellite and Payload Configuration.

frame spacing and frame integration time, which should be suitable for plume phenomenology. Downlink at 1 Mbps should also be adequate since pass times over the ground station(s) of a few minutes each will occur approximately twice daily.

3.3.2.3 Software. While the spacecraft attitude control and sensing system could conceivably position and slew the bus adequately to allow plume observation with the UVSI, the visible acquisition telescope, with a field of view of a few degrees increases the design margin. The penalty is that software must be written for spacecraft attitude control based on the acquisition sensor output.

3.3.2.4 Bus Propulsion. No bus propulsion has been included, although STEP does have an available hydrazine monopropellant module which fits on the launcher side of the core module. An analysis would be required to determine if orbit adjustment to accommodate launch timing is feasible or desirable.

3.3.2.5 Launch Vehicle. Without propulsion, the core module with instruments attached could fit in the existing Pegasus shroud. However, the weight of payload and bus (approximately 600 lbs) will not allow use of the existing Pegasus. Modified Pegasus (see launcher section) or Taurus are capable of phasing the payload at polar circular orbit of > 300 nmi altitude to ensure an orbit lifetime of at least a few years.

3.3.2.6 Mass and Power Summary. Table 3.3-2 illustrates the mass and power estimates.

	MASS (lbs)	POWER (watts)	COMMENTS
UVSI Payload	354	45	
Visible Acquisition Payload	36	10	(at negligible duty factor)
Bus		250	

Table 3.3-2 Mass and Power Summary.

3.3.3 Experiment Operations Concept. The experiment would be scheduled from CSTC. For background observation, periodic or programmed interval "snapshots" would be taken and downlinked on command. Limb viewing would require a spacecraft maneuver, somewhat more complicated, but still done as a programmed event without need for ground contact during its execution. Plume viewing could also be conducted in generally the same manner; an option would be to conduct the target launch in a manner to allow some real-time satellite control (e.g., useful in the event of a short launch delay). In this latter mode, an additional UHF satellite control/status link with a portable ground station might be advantageous (as has been designed for other missions).

3.3.4 Experiment Performance Predictions. The experiment should provide ground resolution (for images) of approximately 3-5 m. For plumes at a distance of approximately 200 km, the resolution should be better than 1 m. At these ranges, with a target crossing space of 2 km/sec, the spacecraft slew rate required would be < 1 deg/sec which (subject to analysis) can probably be handled with the existing 3-axis reaction design.

3.3.5 Technical Risk. The following pages assess the risk in the format requested by the Measurement of Effectiveness Panel. In addition, the following comments (and some above) apply.

3.3.5.1 Complexity. The complexity of the experiment is judged to be low to medium because the spacecraft is basically one rigid body (no large optical gimbals or antenna gimbals. The major complexity is one of controlling the attitude to enough precision with the visible acquisition telescope as the sensor.

3.3.5.2 Flexibility. Programmable experiment options (as well as uploadable software and parameter changes) would be provided for. The major drawback comes in the area of plume observations, where target coordination is essential.

3.3.5.3 Equipment Maturity. The instrument maturity is obviously good. By the time the mission would fly, at least 2, and perhaps 3, very similar buses will have flown, as will have several somewhat similar launch vehicles.

3.3.6 Schedule. See Figure 3.3-4.

SCHEDULE FOR UV PLUME AND BACKGROUND PHENOMENOLOGY EXPERIMENT

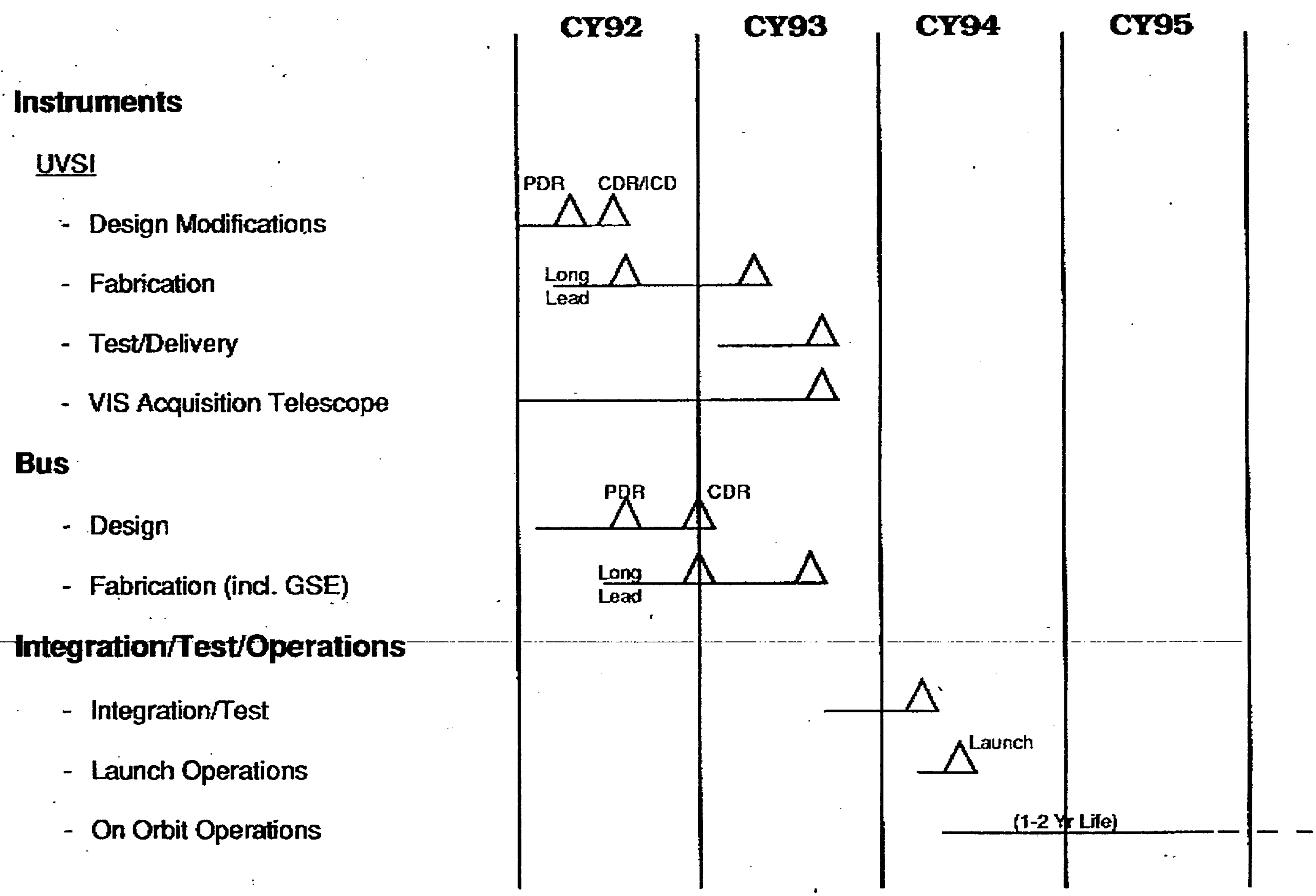


Fig 3.3-4

Figure 3.3-4 Schedule.

3.3.7 Summary. This experiment was conceptualized as an example of what could be done with minimal cost and schedule to address issues requiring (or at least favoring) a space platform for resolution. It should be further pointed out that the incremental complexity, cost, and schedule impact of configuring the experiment to take multispectral visible data would probably not be excessive.

3.4 Pointing & Structural Dynamics Experiment.

3.4.1 Experiment Concepts. One of the approaches undertaken by the LCSE Study Panel was to investigate whether significant elements of ATP might be demonstrated within a low-cost relatively small satellite in Earth orbit. An eye was to be kept open for possible "Value-added" (to ATP) ingredients that might be of interest to SDIO. One result of this investigation was the "Pointing & Structural Dynamics Experiment." This would comprise a small "Hitchhiker" satellite to be ejected from a Shuttle Orbiter. Using sparse elements from a full-scale SBL concept, the experiment would provide a scaleable "large structure" experiment. It would lend itself not only to demonstrating pointing (out of ATP), but also to investigating and demonstrating structural dynamics associated with large deformable "SBL-type" mirrors. It is reasonably well-known that structural dynamics in space can be dramatically different from those exhibited in a non-zero-g and atmospheric background. Cases abound supporting this observation. The successful SKYLAB and the Hubble Space Telescope provide a rich set of examples. The Dahl Solid Friction model

D - Phenomenology Data Collection

OVERALL COLOR EVALUATION

MISSION NAME: UV PLUME & BACKGROUND (UVPB)

NARRATIVE: See Text

ALTAIR EPD ISSUE	SUB-ISSUE FROM DE ISSUE LIST		MOE AND SCALE	EVALUATION RELATIVE TO MOE
17. General Plume Phenomenology	A.02.03	High Resolution Measurements of Plume Phenomenon to Validate Codes		
	A.02.04	Measurement of Plume Reflectivity at Active Track Wavelengths		
	A.03.01	Passive Track Phenomenology Datas for All Types of Targets Against Various Backgrounds		
	———	Active Track Signatures of Hardbodies		
	E.01.01	Enhanced High Resolution Signature (Active and Passive) to Support Bulk Discriminations		
18. General Background Clutter	A.03.01	Passive Phenomenology Data For All Types of Backgrounds		

Kalonak/Eval General 11/22/91 WD11

Table 3.3-2

D - Phenomenology Data Collection

TECHNICAL WORTH: TEST FIDELITY

OVERALL COLOR EVALUATION: Green

MISSION NAME: UV PB

NARRATIVE: See Text

ALTAIR EPD ISSUE	SUB-ISSUE FROM DE ISSUE LIST		MOE AND SCALE	EVALUATION RELATIVE TO MOE
17. General Plume Phenomenology	A.02.03	High Resolution Measurements of Plume Phenomenon to Validate Codes	Resolution <1 m (HI)	Hi
	A.02.04	Measurement of Plume Reflectivity at Active Track Wavelengths	NA	NA
	A.03.01	Passive Track Phenomenology Datas for All Types of Targets Against Various Backgrounds	All target/All Backgrounds (HI) Limited Tgts and Bg's (Med)	Med
	—	Active Track Signatures of Hardbodies		
	E.01.01	Enhanced High Resolution Signature (Active and Passive) to Support Bulk Discriminations	NA (UV)	
18. General Background Clutter	A.03.01	Passive Phenomenology Data For All Types of Backgrounds	Long Durations All Latitudes	Hi

Katonak/Evaluation 11/14/91 WD11

TABLE 3.3-3

D - Phenomenology Data Collection

TECHNICAL WORTH: EQUIPMENT FIDELITY

MISSION NAME: UVPB

OVERALL COLOR EVALUATION GREEN

NARRATIVE:

	SUB-ISSUE FROM DE ISSUE LIST	MOE AND SCALE	EVALUATION RELATIVE TO MOE
17.	A.02.03	• Passive Plume Measurements:	
	A.02.04 A.03.01	• Correct λ Vis → VIS → VIS 1λ → 1 IR(4.3) → Mult IR (2.3, 4.3) Low → Med → High UV Added Value	High, but only UV
	——— E.01.01	Single λ Non-Simult. → Mult λ Simult Meas Low Resolution → High Spatial, Spectral Resolution	Spatially resolved – High
18.	A.03.01	• Temporal Rate 10 fps → 30 fps Low → High	Low rate, but adequate for phenom. data – Med
		• Adequate Sensor Characteristics - Sensitivity - Dynamic Range - LOS Stability	Yes High
		• Active Plume Measurement - λ - 0.53 → 1.06 - 10 pps → 100 pps Low Hi	NA

Katonak/Eval MOE 11/21/91 WD11

TABLE 3.3-4

D - Phenomenology Data Collection

TECHNICAL RISK: COMPLEXITY

MISSION NAME: UV BP

OVERALL COLOR EVALUATION Green

NARRATIVE:

	SUB-ISSUE FROM DE ISSUE LIST	MOE AND SCALE	EVALUATION RELATIVE TO MOE
17.	A.02.03	• Payload Design (Need Schematic Configuration)	Existing Instrument Modular Spacecraft High to Med
	A.02.04	• Experiment Complexity - # of Exp Elements	
	A.03.01	• Data Processing Complexity	
	E.01.01	• Number of Proc. Step Between Data and the results	
18.	A.03.01		

Katonak/Eval MOE 11/21/91 WD11

TABLE 3.3-5

D - Phenomenology Data Collection

TECHNICAL RISK: FLEXIBILITY

MISSION NAME: UVBP

OVERALL COLOR EVALUATION Green

NARRATIVE:

	SUB-ISSUE FROM DE ISSUE LIST	MOE AND SCALE	EVALUATION RELATIVE TO MOE
17.	A.02.03	• Ability to Change Software	Yes
	A.02.04	• Ability to Change Exp. Locations	Yes (if target cooperative)
	A.03.01	• Dependence on Target Parameters	No.
	E.01.01		
18.	A.03.01		

Katonak/Eval MOE 11/21/91 WD11

TABLE 3.3-6

D - Phenomenology Data Collection

TECHNICAL RISK: EQUIPMENT MATURITY

MISSION NAME: UVPB

OVERALL COLOR EVALUATION Yellow

NARRATIVE:

	SUB-ISSUE FROM DE ISSUE LIST	MOE AND SCALE	EVALUATION RELATIVE TO MOE
17.	A.02.03	• Flown vs Lab Hardware (Hardware Pedigree List)	Actual Instrument not Flown, but some components have flown
	A.02.04	• Calibration Capability (Description of Approach) Pedigree	
	A.03.01	• Availability of Spares (List as Known)	Spare Optics exist
	E.01.01		Spacecraft has not flown, but similar models will have flown
18.	A.03.01		

Katonak/Eval MOE 11/21/91 WD11

(experimentally discovered by Phil Dahl and turned into an analytical design tool by Seltzer) provides another example of how a spaceborne reaction wheel can cause deleterious limit cycles in accurate pointing experiments. No known spaceborne experiments have examined the detailed behavior of a flexible segmented mirror. To date, investigation of the LACE program results does not indicate more than a confirmation of structural eigenvalues -- the more important eigenvectors do not appear to have been addressed. Because of their nonlinear characteristics, a more representative portrayal of the structural dynamics of a large deformable mirror is desired.

3.4.2 Pointing & Structural Dynamics Experiment.

3.4.3 Experiment Objectives. The primary objectives are two-fold. A third "secondary" objective is stated.

3.4.3.1 First Objective. The first Primary Objective is to demonstrate a scalable LOS Pointing-at-rate, "Solar-inertial" pointing, Earth pointing, Re-pointing, LOS hierarchical control, & Alignment System on a representative orbiting structure (emphasizing deformable mirrors). Mission-representative slew rates and accelerations would be used. The controls architecture would be mission-representative. Two-axis pointing would be tested in a realistic zero-g non-atmospheric atmosphere (impossible on the ground or in the atmosphere, such as a balloon). Hence mission-level LOS stabilization and pointing in space can be investigated and demonstrated. A new stable platform (IPSRU) can be incorporated in the LOS control concept, and IPSRU's performance can be evaluated in a realistic space environment. Performance will be scored against a ground-based laser beacon.

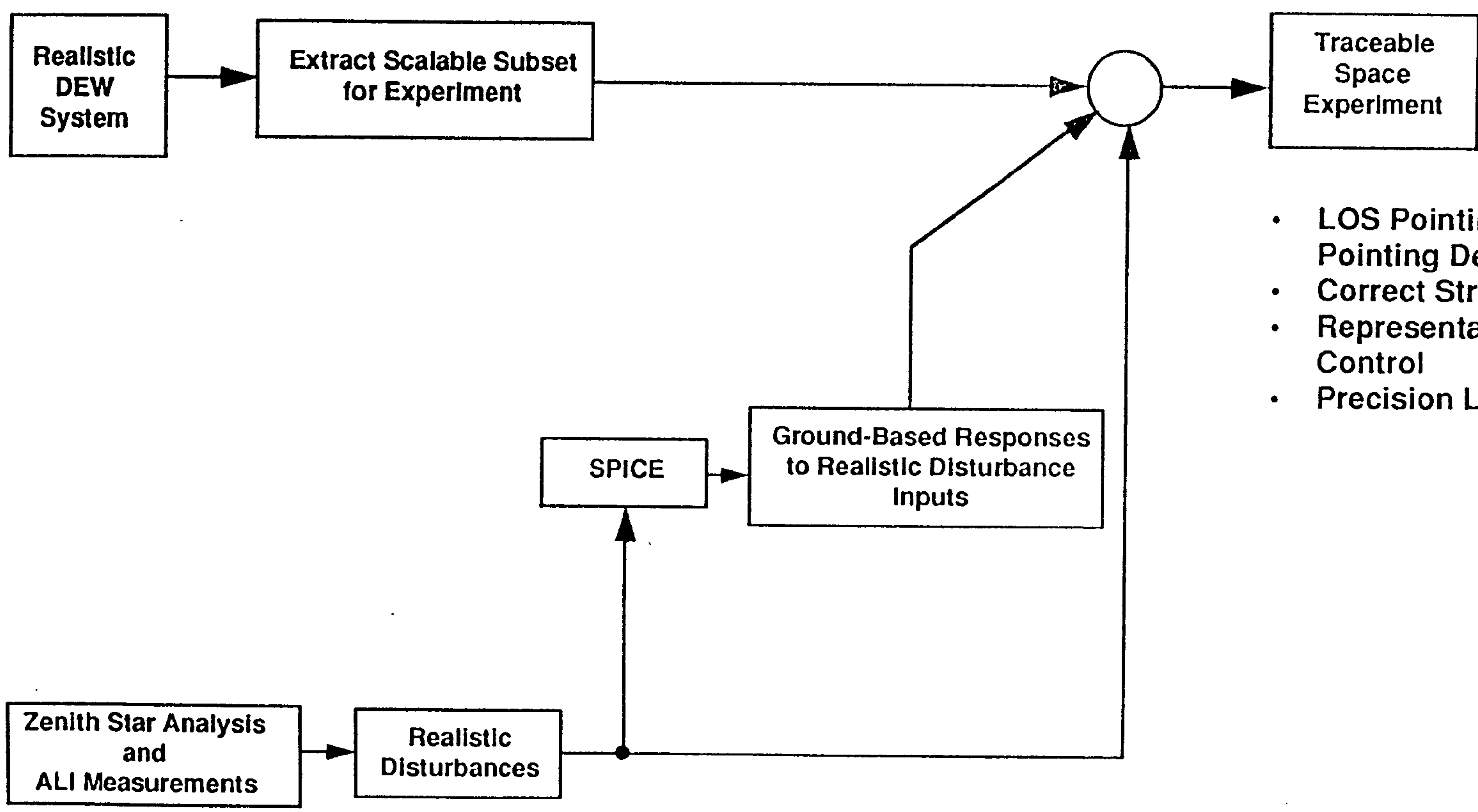
3.4.3.2 Second Objective. The second Primary Objective is to develop and validate combined structural dynamics and optics design tools for future (e.g. STARLITE & ATTD) Programs. These tools currently are outdated or non-existent. As indicated above, it would be possible to demonstrate zero-g and non-atmospheric structural dynamics modeling, testing, and validation in a realistic space environment -- not possible on the ground or in a balloon.

3.4.3.3 Third Objective. A third (Secondary) Objective is to enhance the development of an engineering design and a mission support operations team within the USAF Phillips Laboratory. This proposed experiment is sufficiently simple as to provide a "starter kit" for unseasoned designers of spacecraft. Further, it would provide a long duration space platform that would enable experimentation and on-orbit space design by Phillips Lab engineers and Principal Investigators.

3.4.4 Experiment Description.

3.4.4.1 General. The approach undertaken is portrayed in block diagram form in Figure 3.4-1. First,a realistic operational DEW system is selected (see Figure 3.4-2 for an example of such) to be emulated by the "Pointing & Structural Dynamics Experiment. Because this

Pointing and Structural Dynamics Experiment Approach



- LOS Pointing and Re-Pointing Demo
- Correct Structural Dynamics
- Representative Hierarchical Control
- Precision LOS Scoring

Fig ~~X-1~~ 3.4-1

Pointing and Structural Dynamics Experiment

SBL Fine Tracking Concept

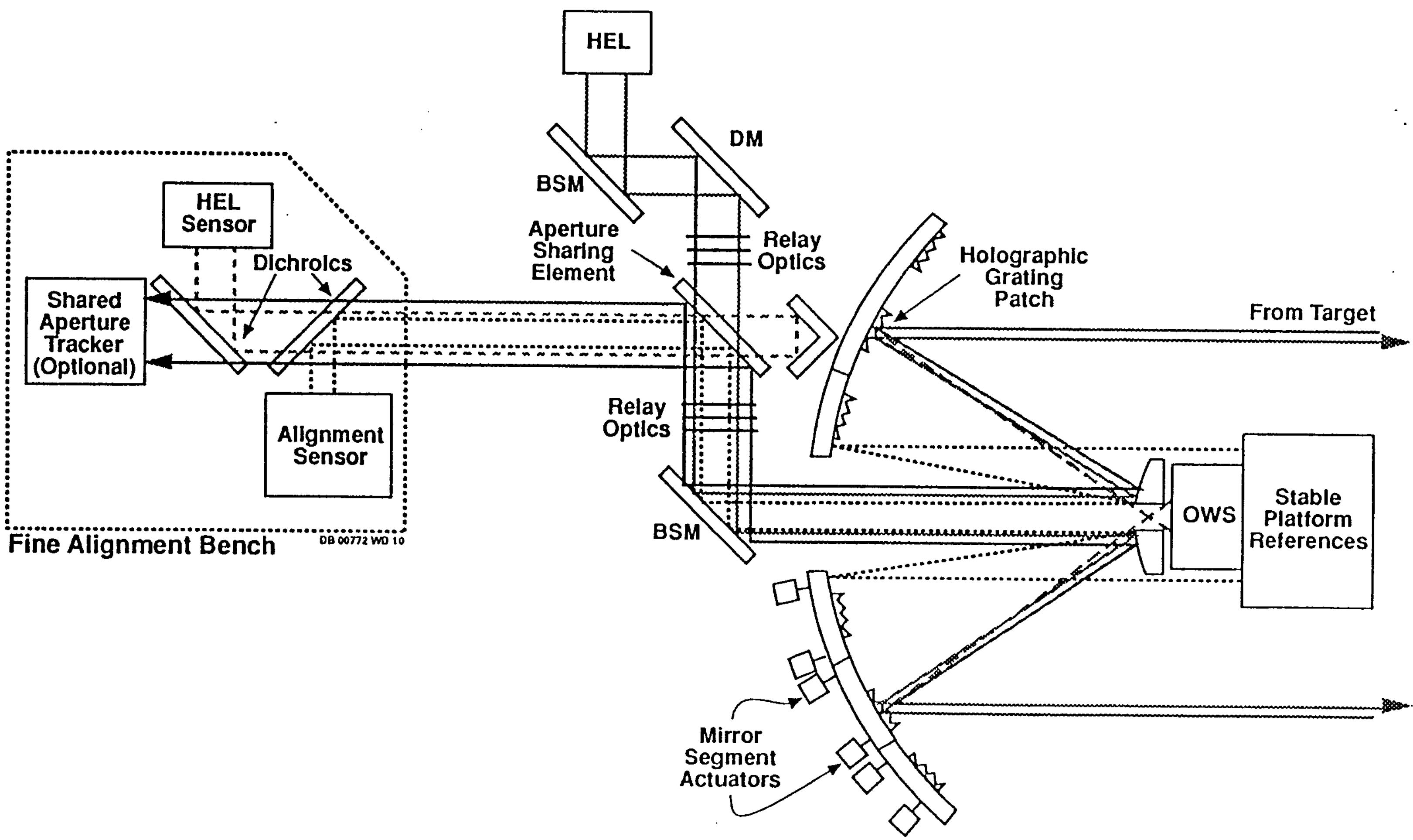


Fig. ~~v~~ 3.4-2

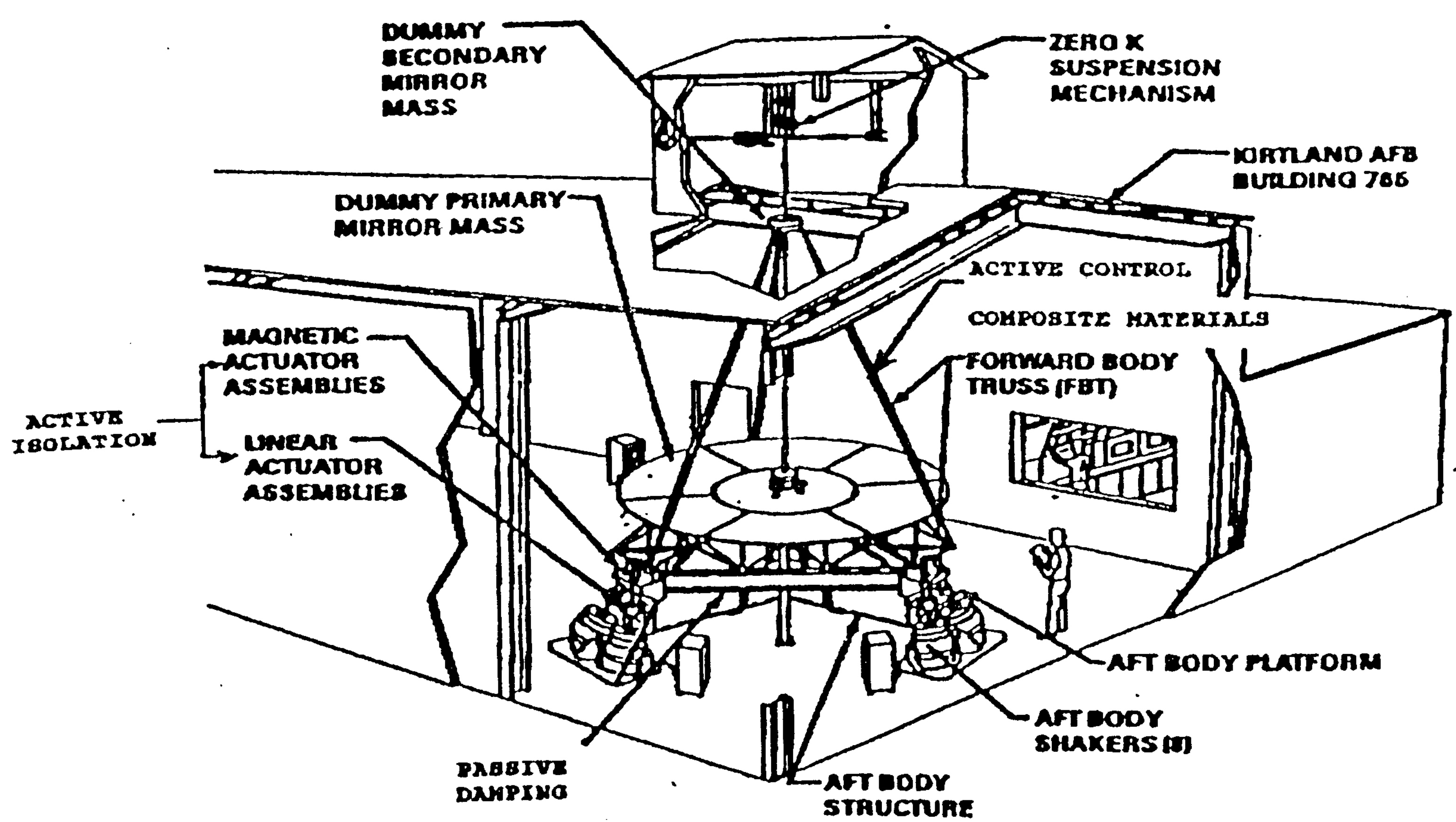
Figure 3.4-1 Pointing and Structural Dynamics Experiment Approach.

Figure 3.4-2 SBL Fine Tracking Concept.

is essentially the same structure emulated in hardware by the SPICE Program (Figure 3.4-3), results from that program could readily be incorporated in the experiment. Structural dynamics of the SPICE could be reproduced in the candidate LCSE experiment. Further, past SBL-type disturbances were characterized (Seltzer) and analytically incorporated in ZENITH STAR investigations. Conversations with Dr. Skolnik have led (hopefully this past

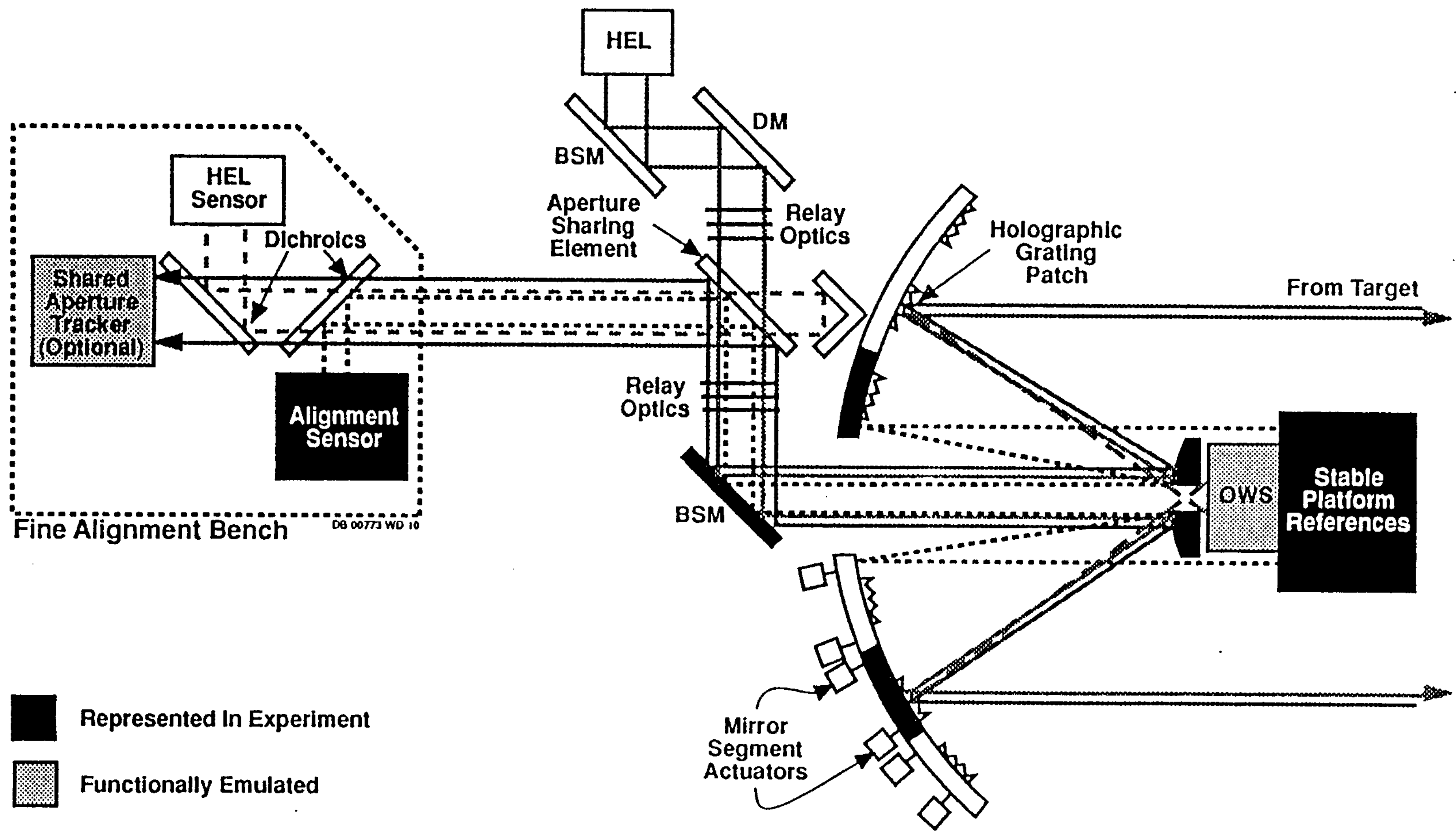
Pointing and Structural Dynamics Experiment
Space Integrated Controls Experiment (SPICE)

SPICE TEST BED



Pointing and Structural Dynamics Experiment

SBL Fine Tracking Concept



Seltzer/LC ATP 11/19/91 WD11

Fig ~~X-4~~ 3.8-4

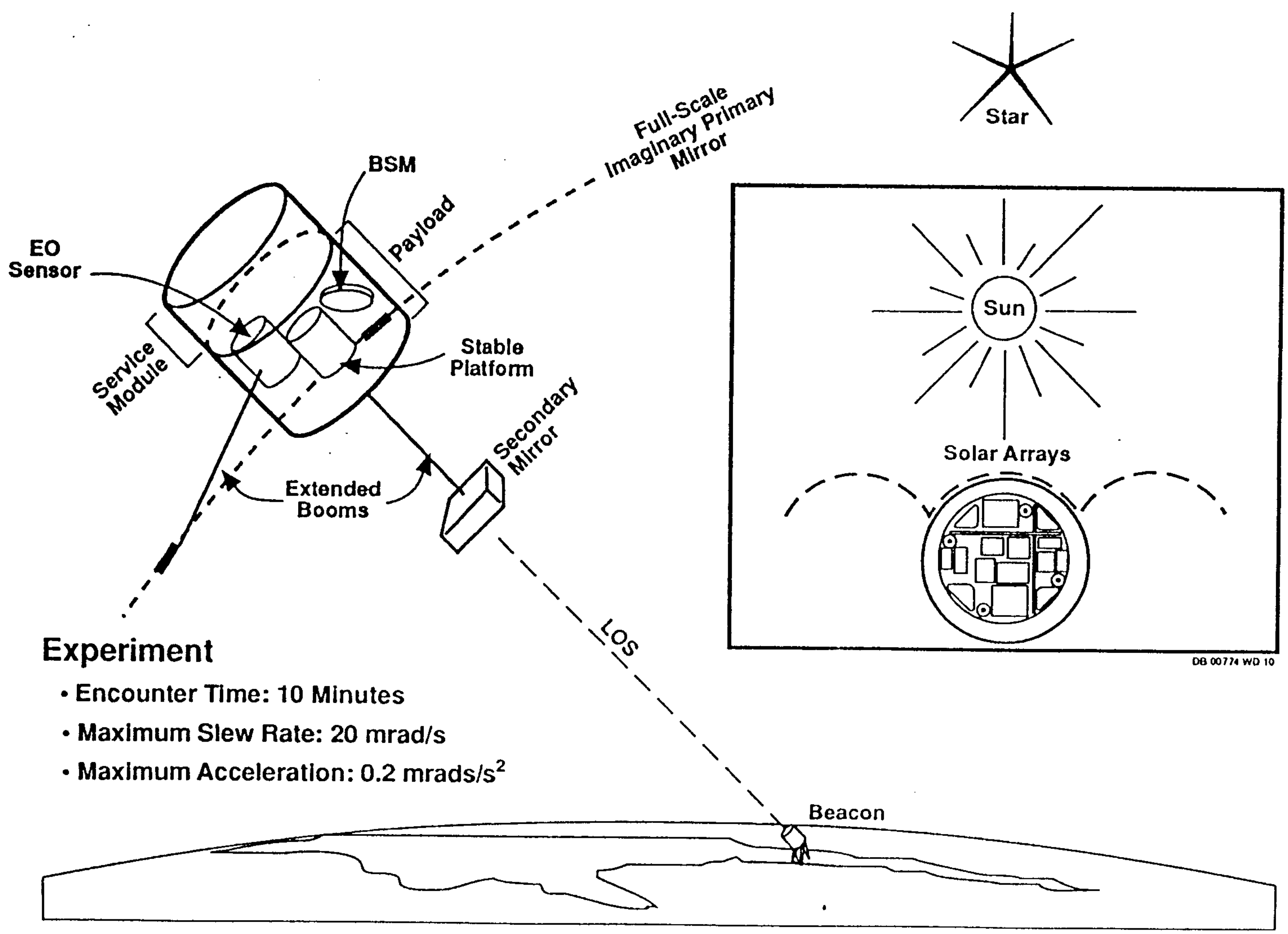
Figure 3.4-3 Space Integrated Controls Experiment (SPICE).

Figure 3.4-4 SBL Fine Tracking Concept.

month) to the measurement of realistic ALI laser firing disturbances. All these disturbances can be incorporated into the SPICE, investigated, and then incorporated into the "Pointing & Structural Dynamics Experiment." Meanwhile, sparsely (to drastically reduce weight) selected portions (shown in black on Figure 3.4-4) of the full-scale operational SBL could be incorporated to provide the experimental satellite.(The "functionally emulated" Shared

Pointing and Structural Dynamics Experiment

Space Experiment Concept



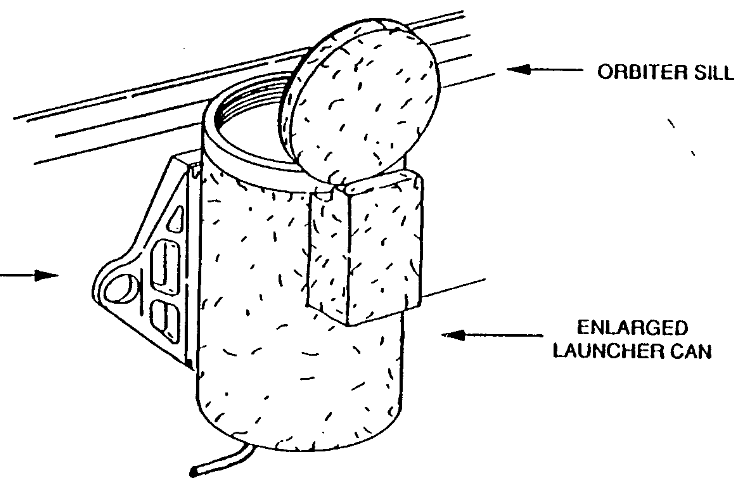
Experiment

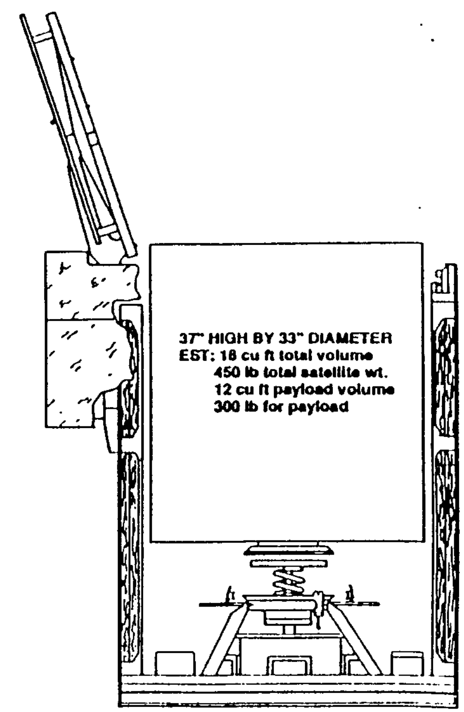
- Encounter Time: 10 Minutes
- Maximum Slew Rate: 20 mrad/s
- Maximum Acceleration: 0.2 mrads/s^2

34-5

Pointing and Structural Dynamics Experiment

Hitchhiker G Option



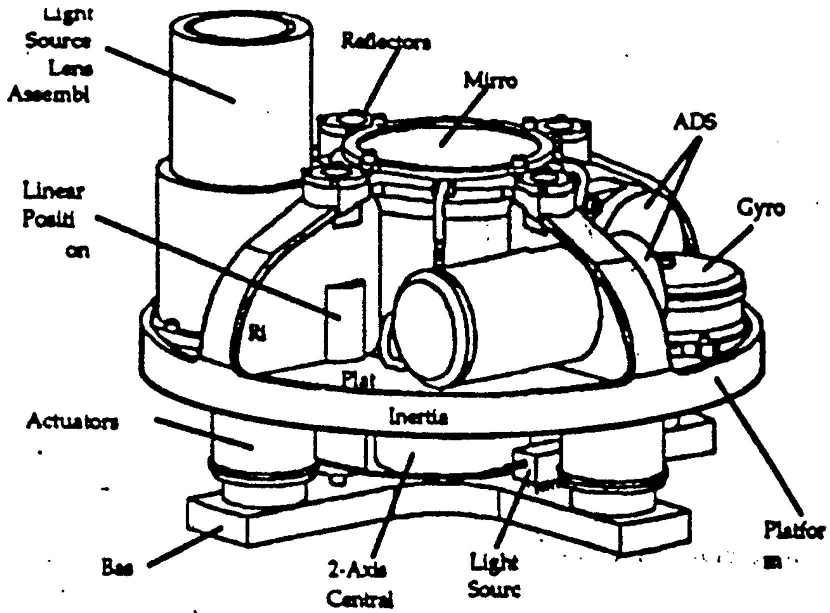


- BUILD LAUNCHER CAN LARGER THAN GAS CAN AND MOUNT ON STANDARD GAS BEAM
- LAUNCHER CAN WOULD BE 30 TO 33 INCHES (INSIDE) DIAMETER CARRYING SATELLITE WEIGHING UP TO 450 LBS.
- POWER, COMMANDING, AND SATELLITE STATUS WOULD BE AVAILABLE BEFORE DEPLOYMENT

Figure 3.4-5 Space Experiment Concept.

Figure 3.4-6 Hitchhiker G Option.

Aperture Tracker -- shown in grey shading -- will be described later.) The resulting satellite experiment is portrayed simply in Figure 3.4-5. It would be cylindrical to fit in a Shuttle Orbiter- borne "Hitchhiker" support structure (Figure 3.4-6). The "operational" portion of the flight meets the first primary objective, although it is of a relatively short duration (five minutes). The major portion of the orbital flight would be used to collect solar energy to



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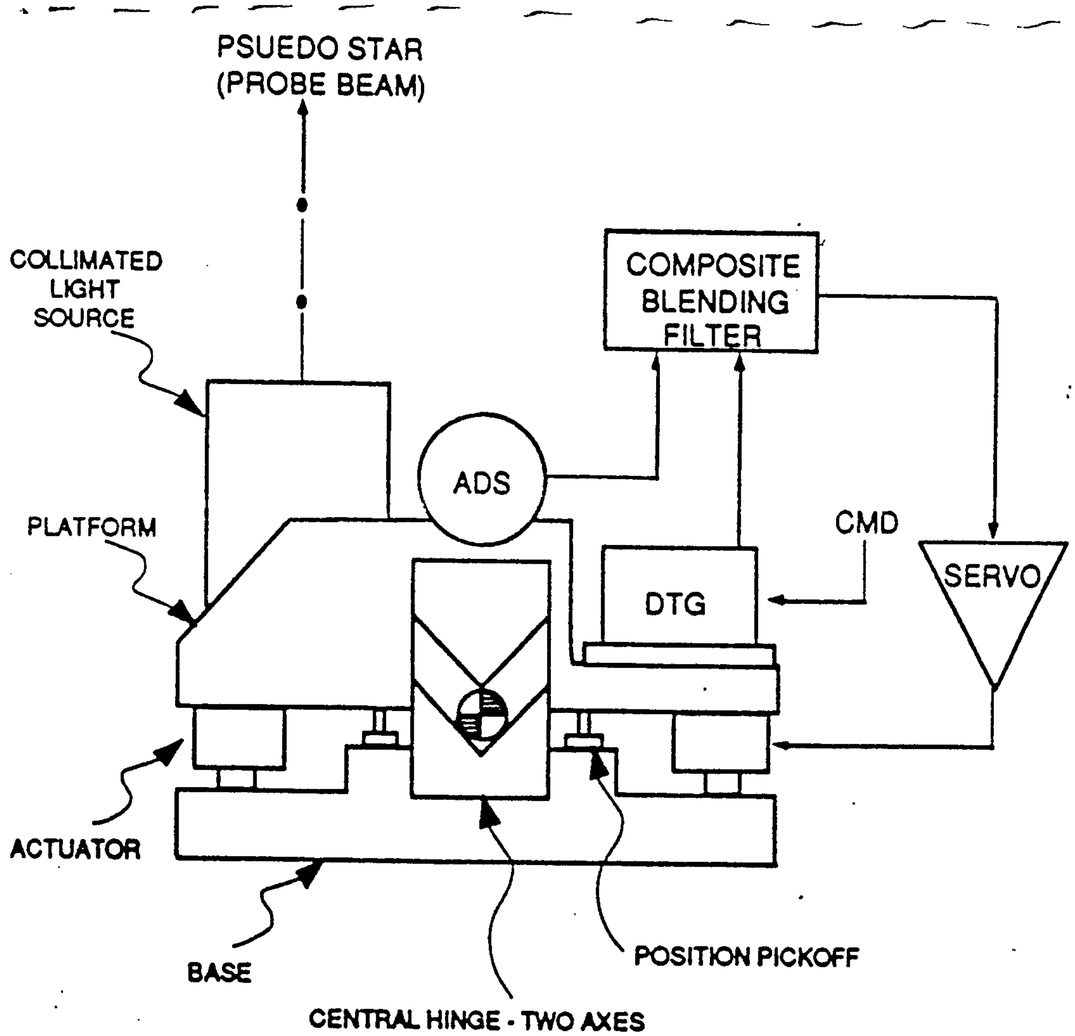
3.4-7a
Figure ~~1-6.~~ Inertial Psuedo-Star Reference Unit (IPSRU)



DTG – DYNAMICALLY TUNED GYRO
ADS – ANGULAR DISPLACEMENT SENSOR

3.4-7b
Figure ~~1-1.~~ IPSRU Mechanical Functional Schematic

FIG. ~~1-7~~ 3.4-7

recharge the batteries and to meet the second primary objective. This "Solar-inertial" attitude can best be maintained with a satellite moment-of-inertia distribution approaching a sphere (such as the "squatty" cylinder shown). Clearly, the satellite can also perform Earth-pointing, re-pointing, and other LOS stabilization experiments.

3.4.4.2 Satellite. It is assumed that the "Satellite" will be comprised of a "Payload" and a "Spacecraft" (sometimes termed a "bus"). Further, it is envisioned that the Satellite will contain on-board devices (e.g. shakers) to induce realistic SBL-type disturbances.

Figure 3.4-7 IPSRU Mechanical Functional Schematic.

3.4.4.3 Payload. The main ingredient of the experiment is the Payload. (Again, refer to Figures 3.4-4 and 3.4-5.) The Primary Mirror will be represented by two 8-inch Off-axis parabolas at representative distances from each other (achieved by an extendable boom). The Secondary Mirror will be portrayed by a pair of 3-inch off-axis parabolas at a representative distance from the Primary Mirror -- again achieved through the use of an extendable boom.Three on-board non-imaging sensors would "functionally emulate" the Shared Aperture Tracker (Figure 3.4-4). Two would sense the incoming laser from the ground beacon. The third would sense the IPSRU (Figure 3.4-7) generated "pseudo-stellar" light beam. The output would be used to drive the IPSRU to the desired attitude so the LOS would be pointed to the beacon. The IPSRU in turn would sense the difference between its attitude and that of the spacecraft, mechanically zeroing out the difference in attitudes. Hence, in essence, the Spacecraft is commanded, to follow the IPSRU. In operation, the beam from the tracker is reflected by the Primary Mirror segments to the Secondary Mirror and thence to the Beam (Fast) Steering Mirror (BSM) and then to the sensors. The BSM is controlled by the light beam from the IPSRU (try to see the dashed

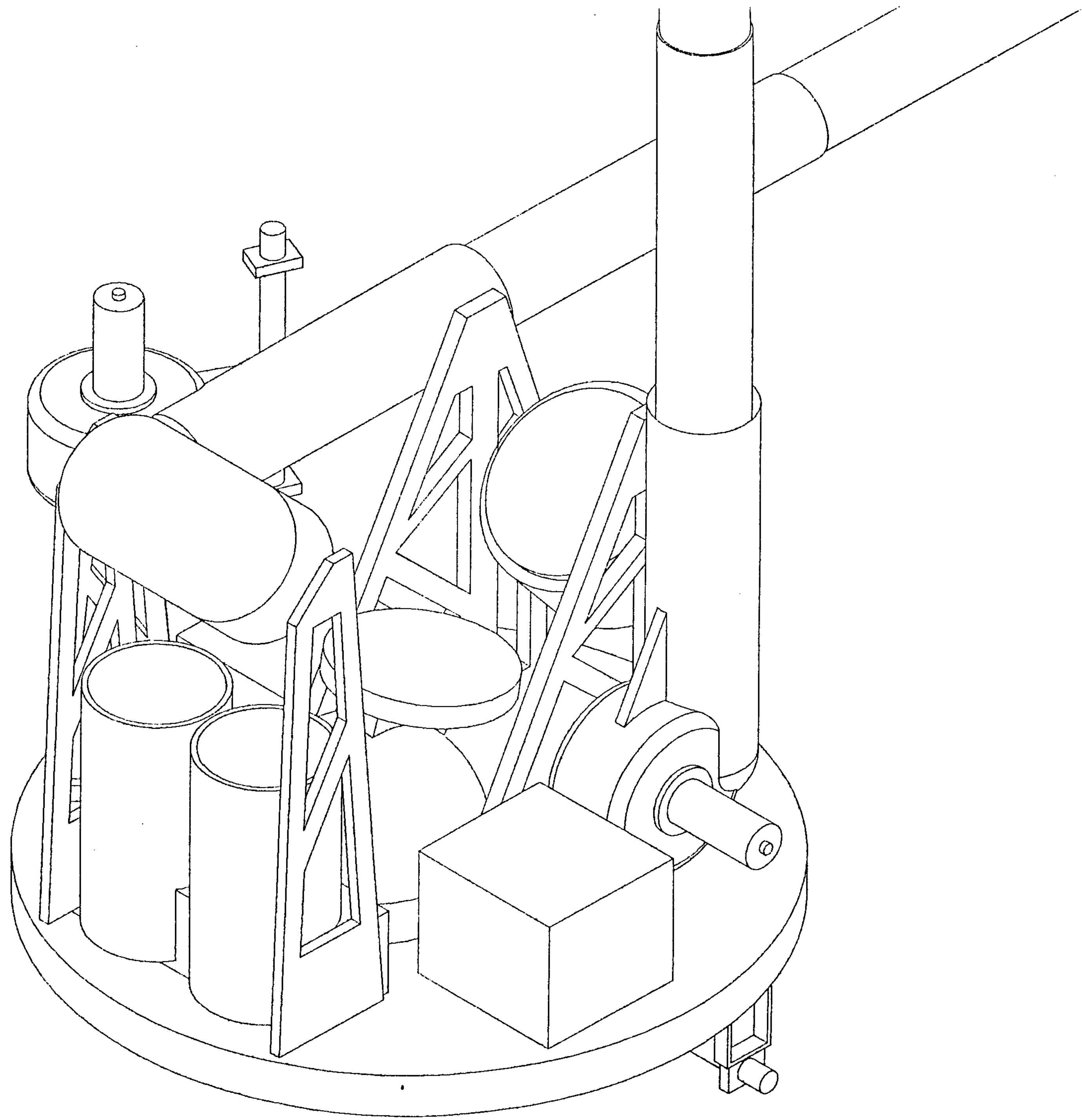


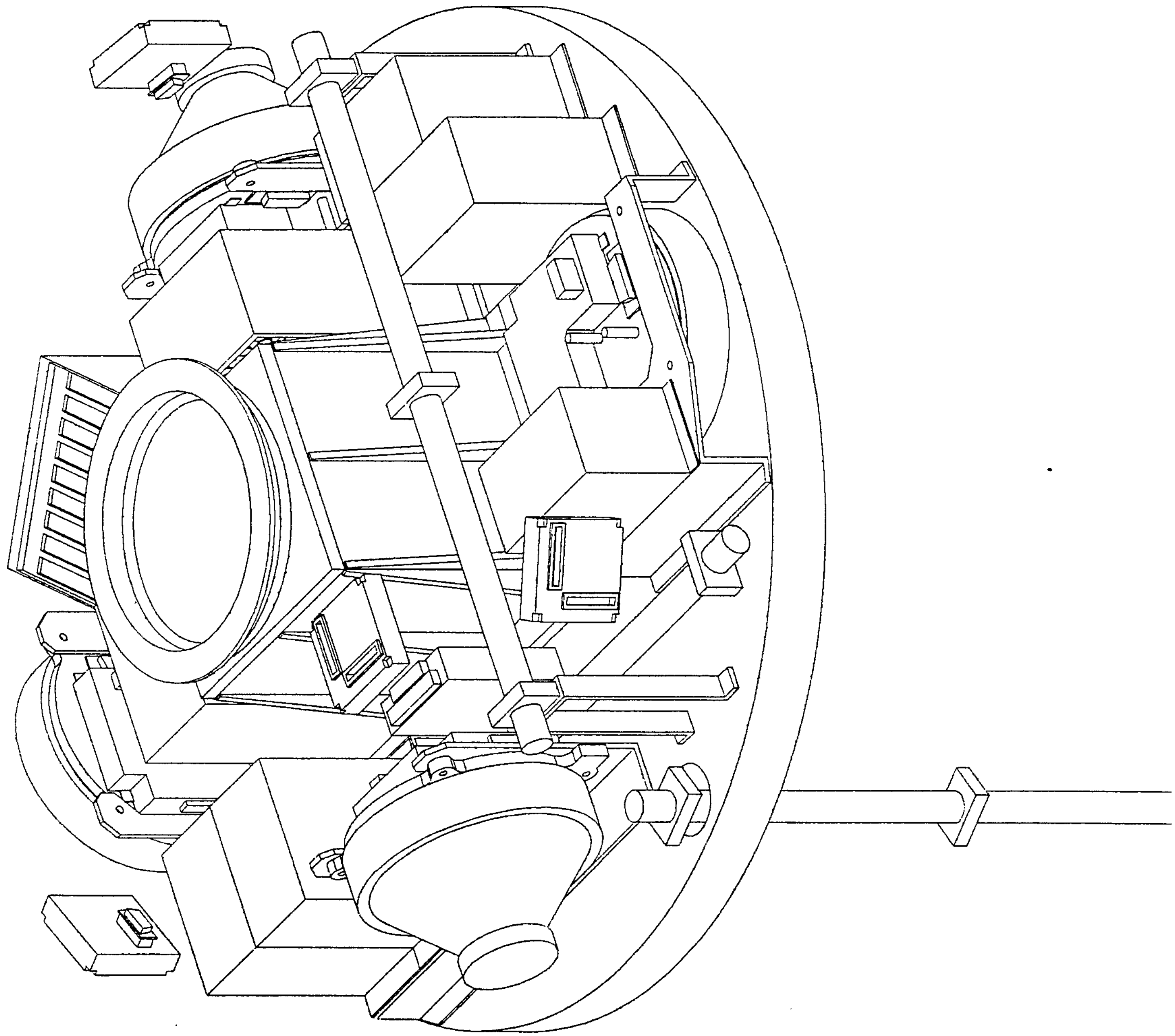
Fig. 2a-8

lines between the Stable Platform and the BSM on Figure 3.4-4). The BSM forces the optics (again, the illusive dashed lines) to remain aligned to the IPSRU. A schematic view of the payload and its components is shown in Figure 3.4-8.

Figure 3.4-8 Payload View 1.

3.4.4.4 Spacecraft. The spacecraft is relatively elementary. It is cylindrical in shape with a diameter of 33" and a height of 37". Its geometric envelope is shown in Figure 3.4-6. It is ejected from the Shuttle Orbiter by a spring. The volume of the spacecraft is approximately 18 cu. ft. with a total mass of 450 lbs. The design goal is to provide most of a 12 cu. ft. dedicated volume to Payload elements which would weigh less than the budgeted 300 lbs. The major elements are shown notionally in Figure 3.4-8. The primary attitude control actuation would be provided by reaction wheels. Their momentum would be exchanged ("de-saturated") through the use of magnetic torque rods interacting with the Earth's magnetic field. Because the IPSRU provides only 2-axis reference, a separate simple IRU is to be used within the Spacecraft. The Attitude Control System (ACS) would also incorporate horizon sensors ... Simple extendable rods would be deployed on-orbit to position the representative Primary Mirror and Secondary Mirror optics. The on-board cross dipole transmitting antenna might have to be deployed on orbit. The clamshell solar cells also would be deployed on orbit. An alternative solar array approach (still under investigation) is shown in Figures 3.4-10 and 3.4-11. All of these components and techniques are simple and have been reliably used before in space. An equipment list is shown in Figure 3.4-1.

3.4.4.5 Control Loops. In essence, the satellite has a hierarchy of three control loops.



3.4.4.5.1 IPSRU. The IPSRU is controlled by the tracker (tracking the "Target" ground beacon). This is done by the on-board tracker which gets its information from the far field target ground beacon (see the solid lines labelled "From Target" on the right hand side of Figure 3.4-4 and the schematics portrayed in Figures 3.4-12 and 3.4-13). The tracker sends commands to the IPSRU to align the IPSRU to the target (either directly or more likely using a filter algorithm to account for delays, speed of light, etc.). The Spacecraft ACS commands the Spacecraft to follow the IPSRU. Meanwhile, their IPSRU eliminates the effects of jitter and transient distortions due to Spacecraft "base motion disturbances" imposed on the optical system. Essentially this sensing is done by two pairs of linear position transducers mounted co-linearly with two pairs of voice- coil actuators for performing attitude control of the IPSRU. In summary, the Spacecraft controls the low frequency attitude and disturbance dynamics. The next two control loops respond to this first control loop.

Figure 3.4-9 Payload View 2.

3.4.4.5.2 Tilt Control. After the IPSRU has been aligned to the target by the tracker, the Spacecraft must be aligned to "follow" the now-tilted (w.r.t. the Spacecraft) IPSRU. Position pickoffs detect this relative tilt, and IPSRU actuators cause the Spacecraft to remove the tilt in a controlled manner.

3.4.4.5.3 BSM. The alignment loop controls the BSM. This is done by the Alignment Sensor which receives its signal from the IPSRU via the BSM (again, refer to the innermost pair of dashed lines on Figure 3.4-4). From this signal, the Alignment Sensor generates command

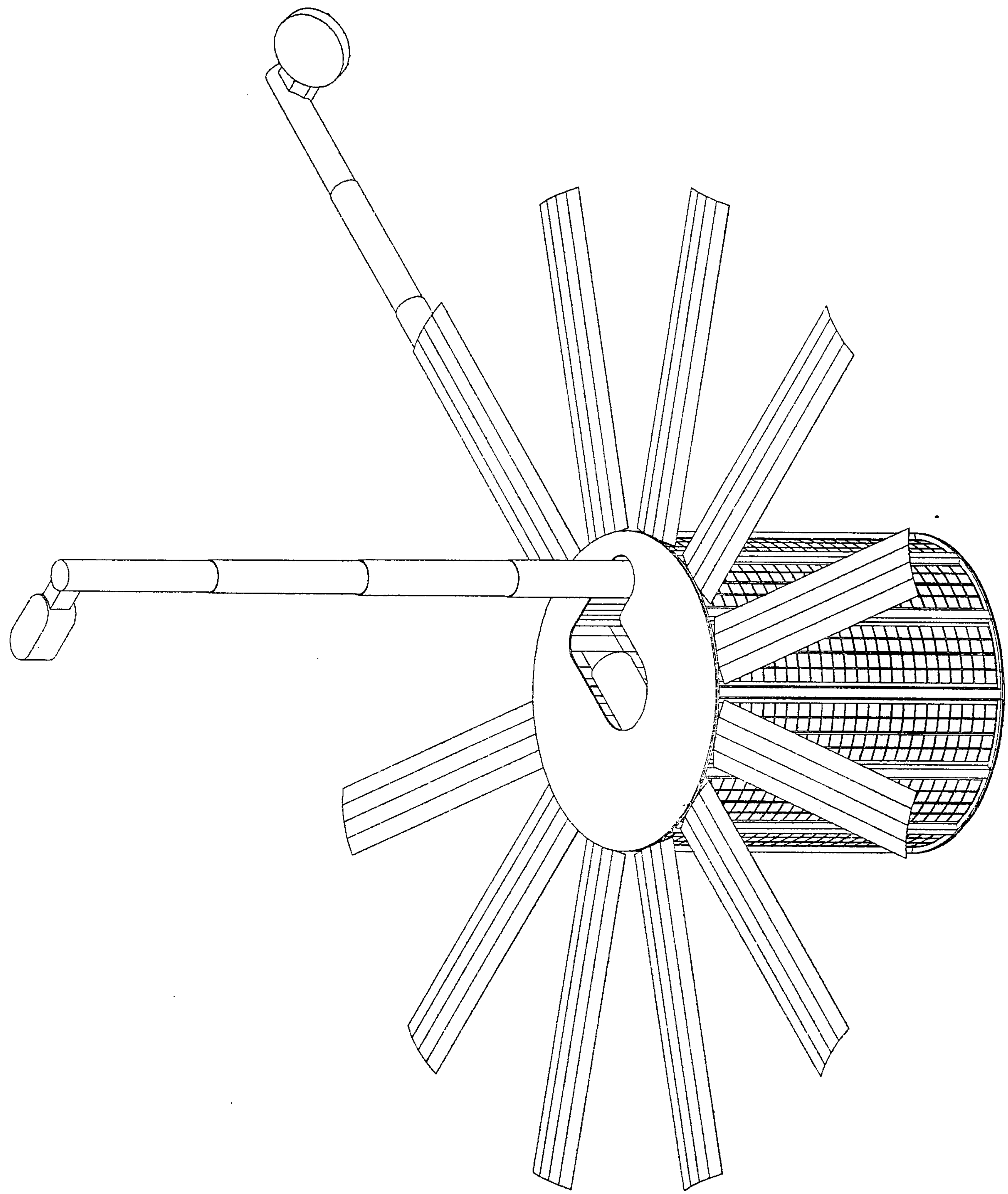


Figure 3.4-10 Solar Arrays Deployed.

information to move the BSM to achieve alignment of the optics.

3.4.4.6 Communications and Data Flow. It is planned to down-link information in real time so that an on-board recorder will not be needed. The predicted data rate is 1 Mbit/s which can be transmitted using S-band (2200-2300MHz) and PSK modulation. A simple cross-dipole antenna, with little or no gain, should suffice. It probably would be spring-loaded for simple deployment. Each cross member would be 5-10 cm in length (corresponding to 1/2 or 1/4 of the 13-14 cm wavelength). The maximum slant range is estimated to be 500 km. An low rate uplink system for software modifications and any other commands is needed. The receiver antenna on the ground would probably be a 3-meter parabolic dish with a bit error rate (BER) of 10^6.

3.4.4.7 Electrical Power. The electrical power will be provided by on-board (probably NiCd) batteries. They will be recharged from deployed body-mounted solar cells similar to the three successful NASA HEAO (High Energy Astronomical Observatory) satellites. Refer to Figure 3.4-5 or Figures 3.4-10 and 3.4-11.

3.4.4.8 Ground Beacon. The ground beacon will have an RME gimballed scoreboard. Scoreboard pointing will be compared to the pre-loaded ephemeris position of the Spacecraft. This latter is provided by the Sunnyvale "Blue Cube" (USAF). On the RME Program there were two ground sites. We'll use the successful RME techniques, but perhaps with only one ground site. The RME beacon on the Spacecraft goes to the ground people

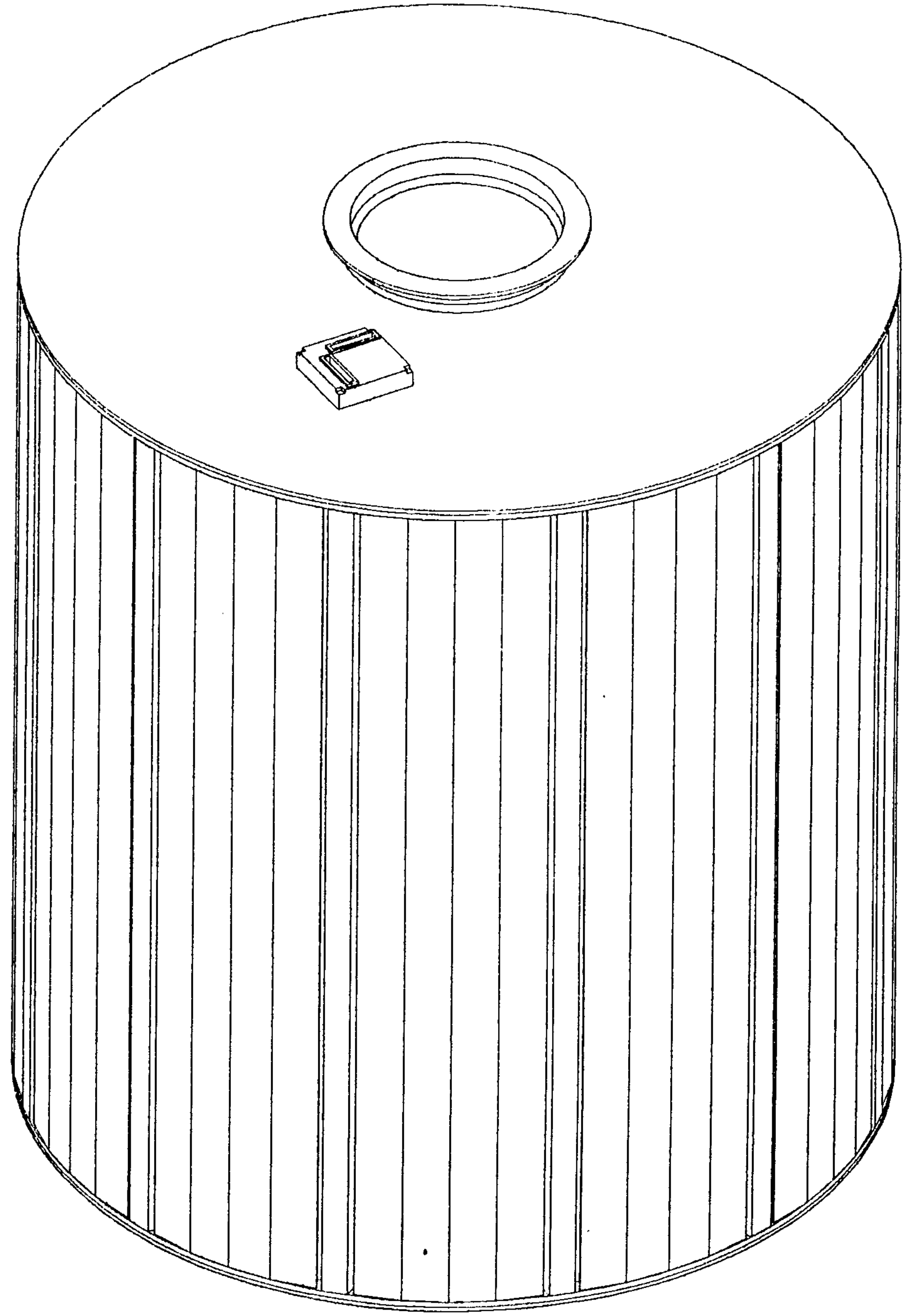


Fig. 24B (cont)

Figure 3.4-10 Solar Arrays Stowed.

who can track it. (It is a fairly broad beacon that need only be pointed in the general direction of the Spacecraft, which tracks it).

3.4.4.9 Scoring. Scoring will be accomplished by considering the full bandwidth signals from the tracking sensors to represent "Truth" (reference signal). This signal contains the full richness of all three control loops. Recall that these same signals had been sent through a low pass filter to more realistically represent signals that would drive the IPSRU. The latter signal is compared to the "Truth" signal to score the pointing that has been achieved.

3.4.4.10 Satellite Power and Mass Estimates. The weight estimates are bounded by the overall capability of the Hitchhiker satellite and ejection mechanism: 450 lbs. It is estimated that the Spacecraft will weigh 250 lbs, the Payload 150 lbs, and the ejection mechanism 50 lbs. This is broken out into a bit more detail in Table 3.4-1. The power required by the Payload is estimated to be 180 watts during the 5-minute operational phase and essentially none during the quiescent phase. In all phases it is assumed the Spacecraft will require 30 watts. This also is shown in Table 3.4-2.

3.4.5 Trade Studies. As soon as possible, trade studies must be made to substantiate the figures arrived at by quick analysis and engineering judgement. The following areas are identified to be studied.

3.4.5.1 Representative Structural Dynamics & Boom Characteristics.

Pointing Experiment Concept

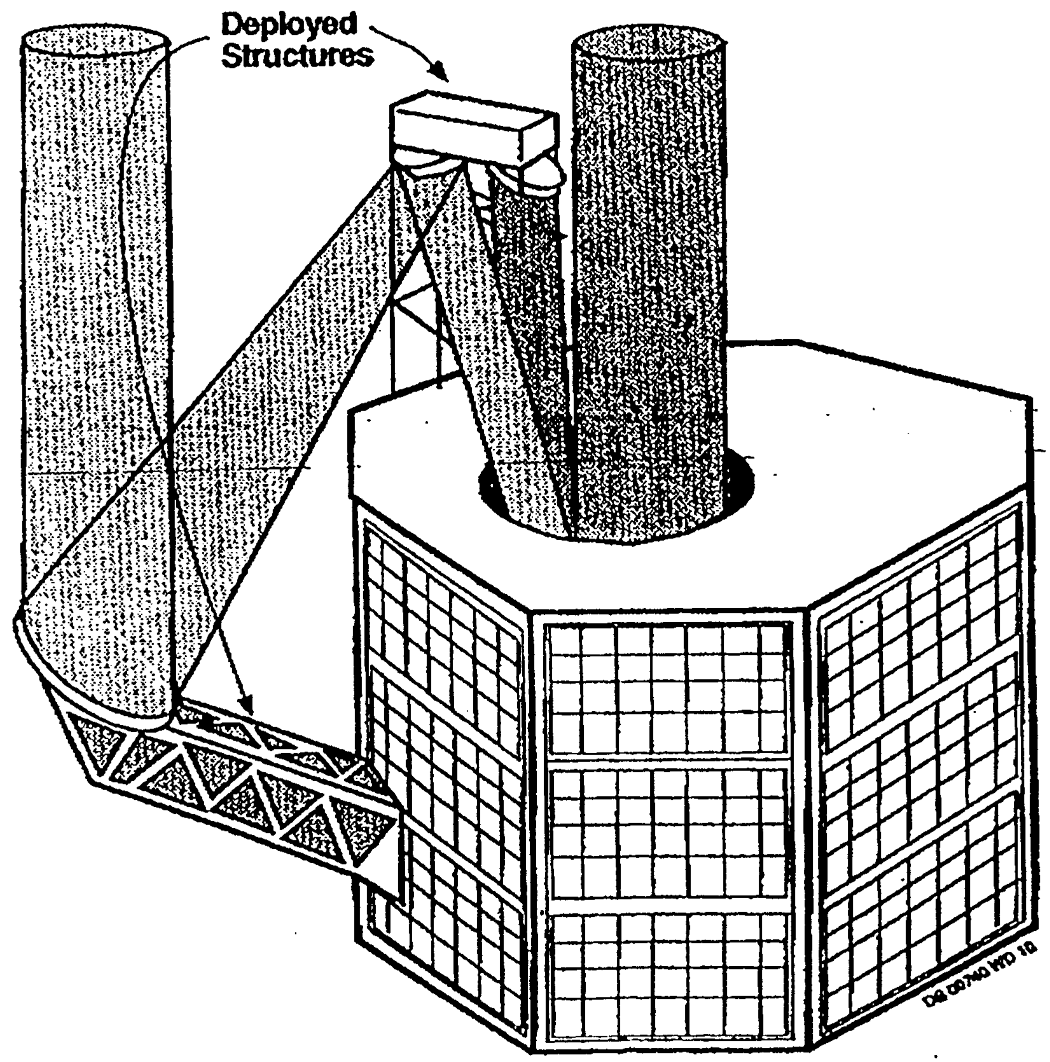


Fig ~~X-0~~ 3.4-17

Figure 3.4-12 Satellite Deployed.

3.4.5.2 Literature & SOTA survey.

3.4.5.3 ACS.

3.4.5.4 Modes of Operation.

3.4.5.5 Thermal Control.

3.4.5.6 Electrical Power.

3.4.5.7 Communications & T/M (include storage).

3.4.5.8 Flexibility of Design. Flexibility of the design & digital updates, including structural dynamics updates.

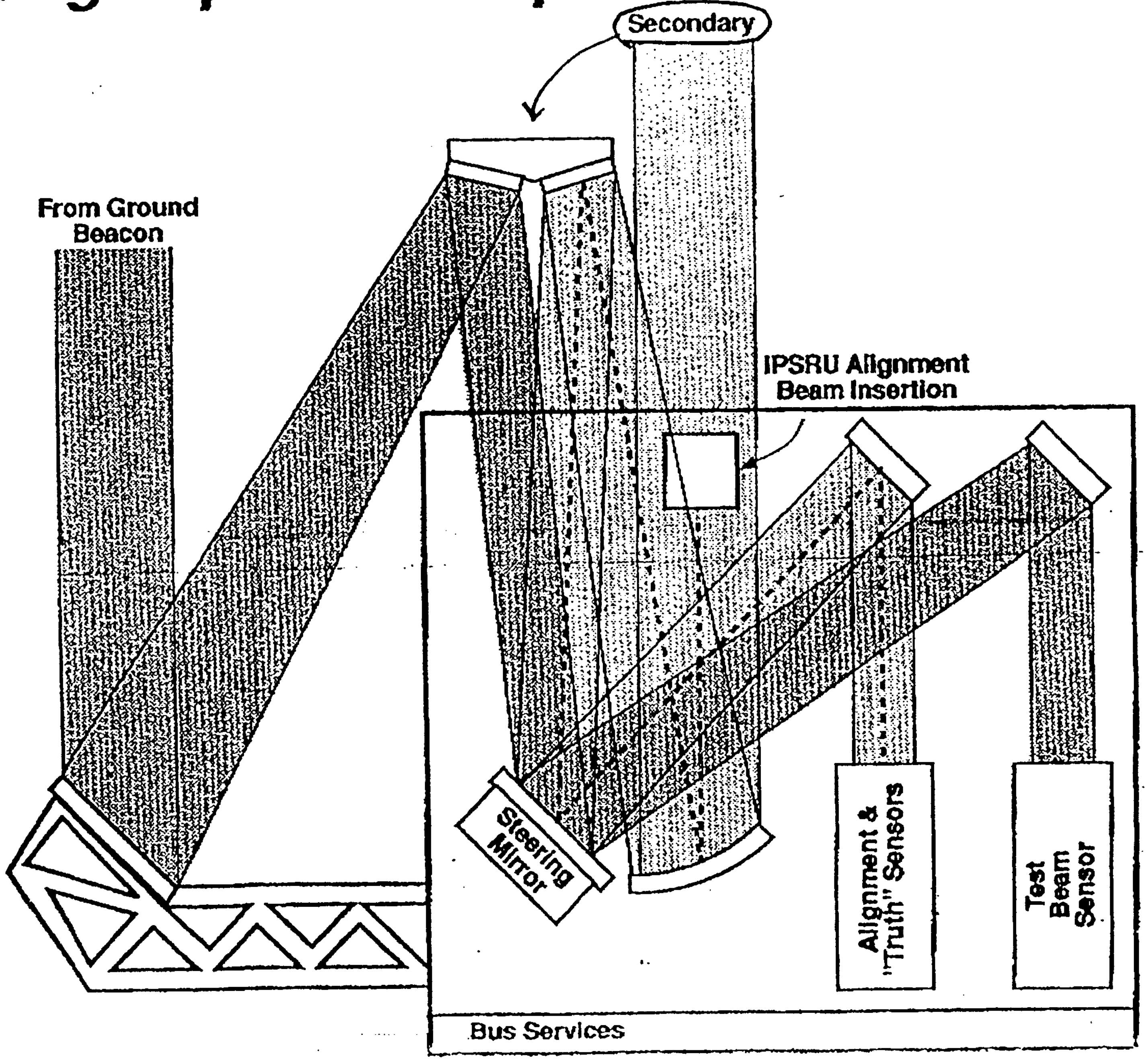
3.4.5.9 Computer.

3.4.5.10 Ground Support.

3.4.5.11 Operations Support Team.

3.4.5.12 Instrumentation.

Pointing Experiment Optical Sketch



3.4-13

Figure 3.4-13 Optical Schematic.

3.4.5.13 Scoring.

3.4.5.14 Disturbance Modeling & Implementation (shakers, etc.).

3.4.6 Experiment Operations Concept.

3.4.6.1 Experiment/Engagement Requirements. Since there will be a high SNR and the aperture is small with many photons from the ground beacon laser impinging on the system, the pointing accuracy should be on the same order as RME (which is roughly three times more accurate than ALTAIR predictions). Hence it appears that LOS jitter should be compatible with the (classified) requirements that might be expected of an early operational SBL system. Unclassified simulations (performed by Logicon/Control Dynamics Co.) showed that an ALTAIR-type P-80 TEAL RUBY system with an "enlightened" control algorithm and bang-bang mass expulsion thrusters could provide Spacecraft pointing on the order of 100 mrad (against a requirement of 600 mrad). When the simulation fidelity was improved (again, by Control Dynamics Co.) to incorporate the fine control afforded by the BSM, pointing accuracies on the order of 50 nanorad were found in the face of thruster disturbances, sensor and gyro noise and quantization, etc.

3.4.6.2 Experiment/Engagement Events & Sequence.

3.4.6.3 Operations Interfaces.

TABLE 3.4-1 SPACECRAFT [illegible]

Now only (not S.E. or S/W)

People in staffing		A	B	C	D	E	F	G
	1	SYSTEM	ITEM	VENDOR	WEIGHT (LBS)	POWER (WATTS)	ROM COST ($K)	NOTES
	2							
9	3	ELEC POWER	SOLAR PANELS	SPECTROLAB	15.00	0.00	425	
	4	ELEC POWER	(2) BATTERY PACKS	GATES	32.00	0.00	195	
	5	ELEC POWER	REGULATOR	CUSTOM	2.00	1.00	33	
	6				49.00	1.00		
	7							
	8	ATT CONTROL	(2) SCANWHEELS	ITHACO TYPE B	30.00	13.20		
	9	ATT CONTROL	(2) REACTION WHEELS	ITHACO	20.00	10.00		
6	10	ATT CONTROL	(3) TORQUE RODS	ITHACO (25")	10.20	0.70		
	11	ATT CONTROL	3 AXIS MAGNETOMETER	ITHACO	1.10	0.70		
	12	ATT CONTROL	ACS CONTROLLER	ITHACO (9 CARD)	15.00	5.00		
	13				76.30	29.60	1625	
	14	ATT CONTROL	SUN SENSORS	ADCOLE 27530	4.20	0.50	228 (full coverage)	
	15	ATT CONTROL	3 AXIS IRU	HONEYWELL (ASAT)	3.78	17.50	520	
	16				7.98	18.00		
	17							
6	18	COMP/DATA	SATELLITE CONTROLLER	HONEYWELL S-5	5.00	25.00	130	5 card unit
	19	COMP/DATA	DATA STORAGE UNIT	RAYMOND 6420	4.50	1.00	65	Hi alt 2 [illegible] unit
	20				9.50	26.00		
	21							
	22	TELEMETRY	TRANSMITTER (5W)	LORAL	1.00	1.00	130	
	23	TELEMETRY	RECEIVER	LORAL	2.00	1.00	98	
4	24	TELEMETRY	ENCODER/DECODER	LORAL	2.00	1.00	780	No S GLS Transponder -- [illegible]
	25	TELEMETRY	DIPLEXER	LORAL	1.00	1.00	13	Cmd/[illegible]
	26	TELEMETRY	ANTENNA		1.00	0.00	20	
	27				7.00	4.00		
	28							
2	29	TEMP CONTROL	HEAT STRIPS/SENSORS	CUSTOM	5.00	0.00	26	
	30							
5 / 27	31	STRUCTURE	MAIN SHELF	CUSTOM	45.00			
	32	STRUCTURE	BOX/FOOT	CUSTOM	10.00			
	33	STRUCTURE	HAT STRINGERS	CUSTOM	12.00			
	34	STRUCTURE	TOP/BOTTOM	CUSTOM	10.00			
	35				77.00		260	incl bracket
	36							
~40 man time	37	OVERHEAD	CABLES/CONNECTORS		15.00		68	
60 man yrs	38							
	39				246.78	78.60		

Doesn't incl { S/W, Sys Engr, $, Doc'n, [illegible]

Σ 4514

Most comps don't require valid [illegible]

Table 3.4-1 Spacecraft Equipment List.

3.4.7 Experiment Performance Predictions. The concept is being sized to meet with the pointing requirements outlined in Section X.1.3.1 (above). Further, during pointing-at-rate, the spacecraft must slew at a rate of 20 mrad/s with an maximum acceleration of 0.2 mrad/s ^ 2.

3.4.8 Technical Risk.

3.4.8.1 Complexity. Weight restrictions on this small satellite has virtually dictated a simple system composed of a minimal amount of hardware. Maximum (nearly exclusive) use is made of existing hardware, emphasizing some government-owned hardware.

3.4.8.2 Flexibility. On-board calibrations will be designed into the system. Updates and modifications of on-board software from the ground will be permitted with careful design by engineers experienced in this area.

3.4.8.3 Equipment Maturity. Nearly all equipment has already been used in space and airborne applications.

3.4.9 Schedule. See Figure 3.4-14 milestone chart that includes long leadtime items.

3.4.10 Summary.

TABLE 3,4-2.

Pointing (and Structural Dynamics) Experiment (Cont'd)

Mass, Power, and T/M

	Operational Power (W)	Quiescent Power (W)	Mass (#)
PAYLOAD			
IPSRU	50	0	40
FSM	50	0	40
Optics	30	0	20
Instrumentation, Signal Conditioning, & Control Electronics	50	0	20
	180	0	120
Contingency (25%)	45	0	30
P/L Total	225	0	150
SPACECRAFT	30	30	130
Contingency (15%)	5	5	20
S/C Total	35	35	150
ENTIRE SATELLITE			300
EJECTION MECHANISM			150
Estimated Hitchhiker Orbital Capability:			450

Telemetry Requirement: Estimate 1 Mbps

Pointing and Structural Dynamics Experiment Schedule

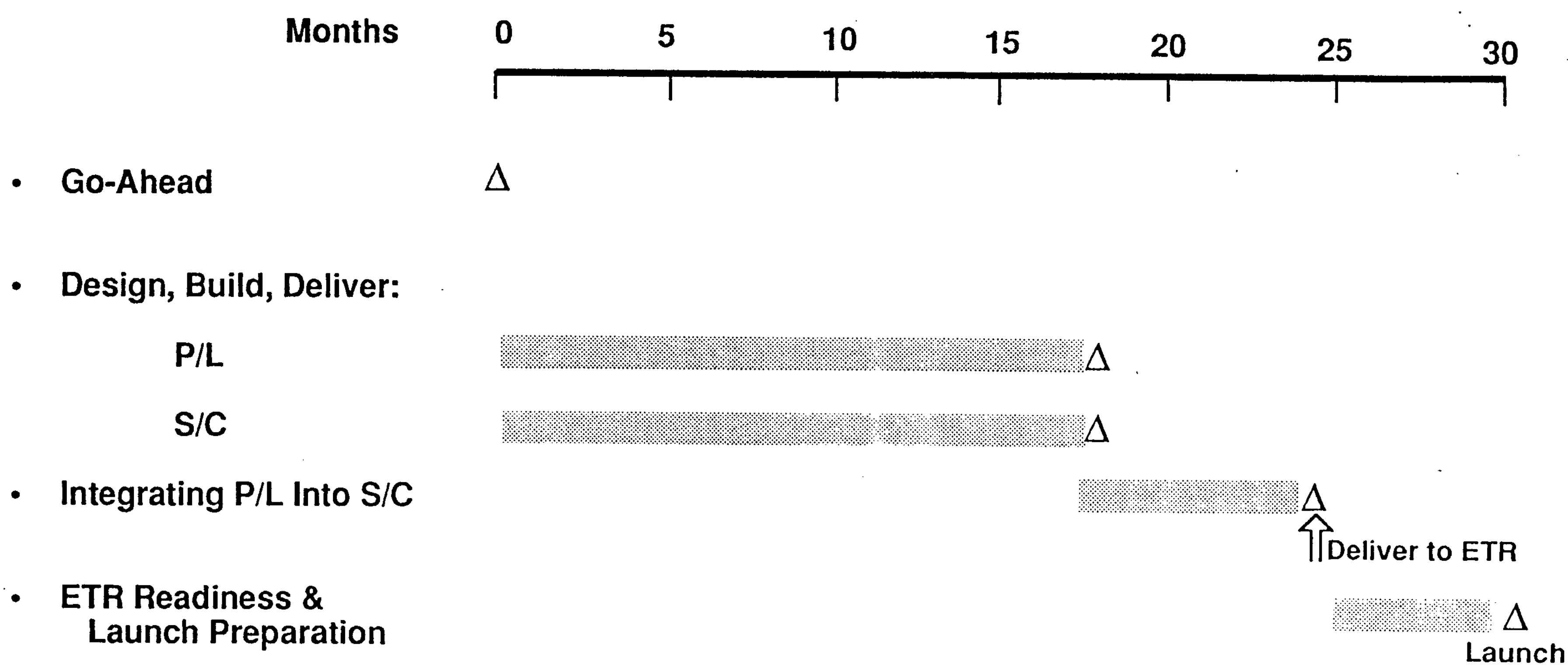


Table 3.4-2 Mass, Power, and T/M.

Figure 3.4-12 Schedule.

3.4.11 Cost.

3.4.12 Acknowledgements. I wish to acknowledge the contributions of Teledyne Brown Engineering (in particular, Mr. Mort Eldridge and Mr. Lee Huffman) for their conceptual development of the Hitchhiker Spacecraft. Further, the efforts of Dr. Dan Herrick and Mr. Denny Boesen of Logicon RDA in helping develop the overall concept but in particular in conceiving the Payload concept. Finally, I appreciate the support from Control Dynamics Co. in providing performance information (Dr. Gene Wells) communications information (Mr. Walt Frost), and helping evolve the electrical power system (Mr. Charlie Graff).

3.5 P-80 Based ATP Technology Demonstrator. The combination of an available shuttle qualified spacecraft, the potential use of a DoD-paid shuttle slot, and a well developed ALTAIR payload design and associated procurement packages lead to a consideration of this concept. The combination of having low cost, highly capable spacecraft and launch resulted in sufficient margin on the $150M program cap to develop a relatively sophisticated payload. It is the payload, however, that makes the schedule most demanding. The shuttle opportunity is currently on 5 Apr 94 and even though it will likely move to Aug 94, this concept must fit within a 30 month development schedule.

The tight development schedule can be met only under the following conditions:

a. SDIO provides full authority to proceed on 6 Jan 92.

b. Full FY 92 funding is provided in early Jan 92 per the program funding requirements identified in the cost supplement to this package.

c. Funding for FY 93 and subsequent program out-years is made available as early in the fiscal year as possible (in consideration of Congressional Budget passage normally being late) and that funding is provided consistent with the program requirements.

d. Phillips Laboratory releases all pending procurement packages by 20 Jan 92 and awards a laser contract in the same time frame.

e. Phillips Laboratory and SDIO execute this program in a manner that insulates it from external perturbations. This includes establishing a tightly integrated program team, limiting external reviewers to a <u>small</u> elite group who stays with the program through its development launch, and on-orbit operations, and maintaining clear and direct channels of communications between SDIO and Phillips Laboratory.

3.5.1 Experiment Objectives. The primary objectives are to demonstrate approaches to plume-to-hardbody handover and to demonstrate active tracking in the presence of a plume. Secondary objectives for this experiment are to collect phenomenology data in the UV and IR on plumes and backgrounds. In addition, a tertiary objective is to provide a platform for testing an ion propulsion system that is currently integrated into the P-80 spacecraft.

3.5.2 Experiment Description.

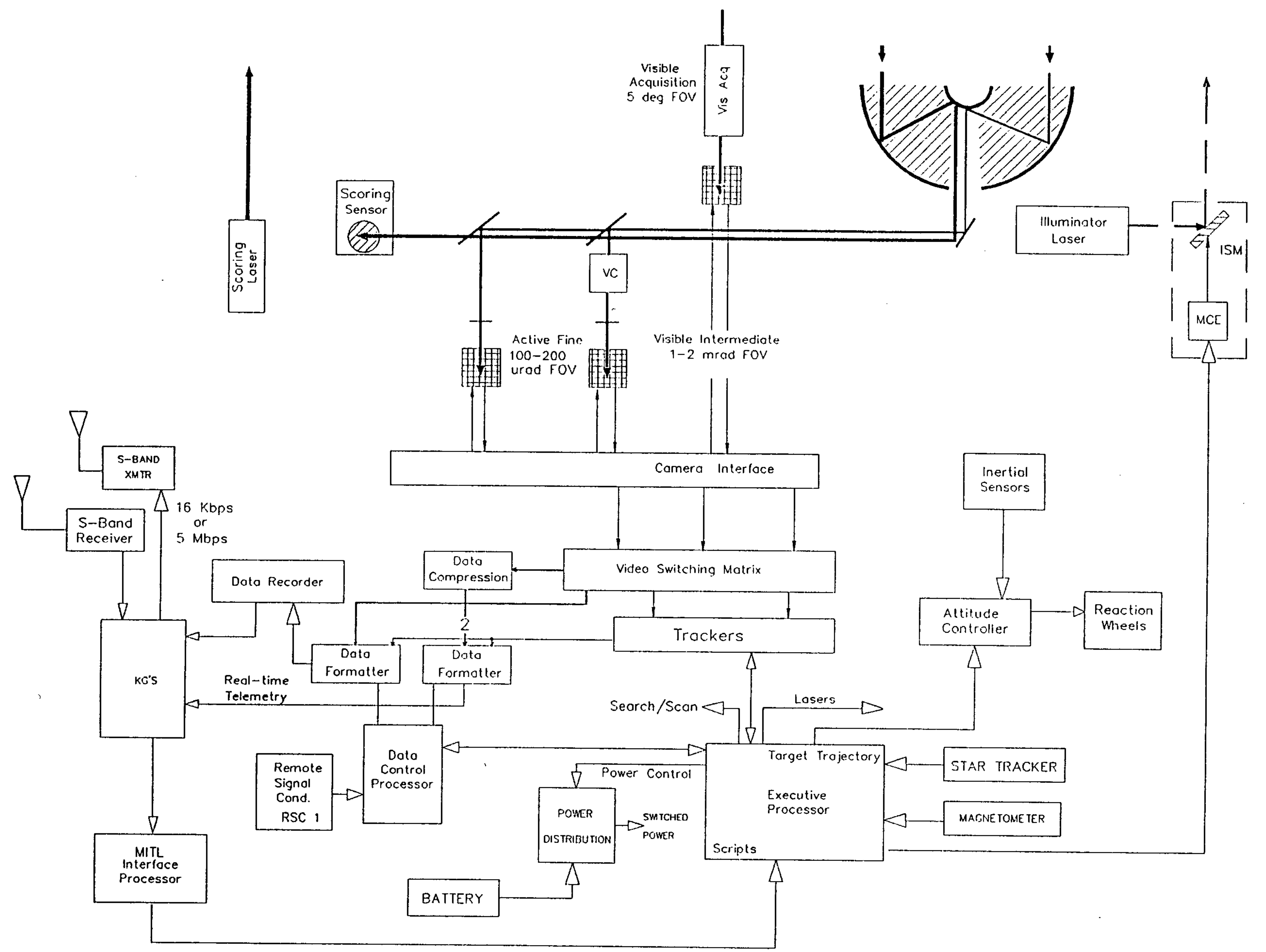
Figure 3.5-1 Conceptual Layout of Instruments.

3.5.2.1 General. The primary components of this experiment include a suite of visible trackers, an illuminator laser of approximately 0.5 Joules for active tracking, IR and UV instruments for plume and background phenomenology data collection, three carry along target rockets, and a spacecraft bus that provides all necessary on-orbit support functions to the payload. Figure 3.5-1 provides a conceptual layout of these instruments and Figure 3.5-2 is a block diagram depicting the relationships of the spacecraft systems.

3.5.2.2 Payload Instruments.

3.5.2.2.1 Sensors. The sensor suite comprises four visible focal planes for the tracking experiments plus UV and IR sensors for phenomenology data gathering. The IR and UV sensors would not be used in the track loop. Multiple filters for collecting data at various IR wavelengths during the boost phase of the target rocket would be included in the IR camera. UV data would be collected using the JPL UV imager/spectrometer. The availability of this instrument, its self contained nature, low power requirements, weight margins in the spacecraft result in minimal impact to the overall experiment. Details on this instrument can be found in Appendix 5. (THIS IS THE SAME AS THE UV EXPERIMENT)

The sensors used in the tracking experiments include a visible acquisition camera, a visible intermediate track camera, a visible active fine track camera, and a scoring sensor. The first three of these sensors would be essentially identical to the Altair visible sensors except that



SATELLITE FUNCTIONAL DESIGN

Figure 3.5-2 Spacecraft Block Diagram.

the FOVs would be increased, consistent with the use of the smaller telescope. The scoring sensor is new, i.e., it was not part of the Altair payload concept. The P-80 concept does not include a marker laser or target boards on the target missiles. The targets would include a corner cube which would give a return at the wavelength of a scoring laser included in the payload. The scoring sensor would detect this return and give a measure of the position of the retro. This position would be compared to the estimate of the corner cube position derived from the active fine tracker or the passive intermediate tracker. The FOV and IFOV of the scoring sensor would be identical to those of the AFT sensor.

An option that would be considered is to use the 40 cm telescope that is part of the UV instrument as the telescope for the tracking sensors. A preliminary evaluation of this approach has been made, and it appears to be difficult to integrate the tracking system with the UV instrument. Therefore, we have chosen to include a separate 30 cm telescope which is already available to the program for the tracking system. The option of using the UV 40 cm telescope would be more thoroughly evaluated if a program were approved, but the current baseline P-80 concept includes the 30 cm telescope.

3.5.2.2.2 Lasers Systems. The illuminator laser for this experiment would be developed under existing ALTAIR procurement activity. The extremely aggressive schedule would increase the likelihood that performance parameters might have to be relaxed in order to stay within the 600 watt power consumption specification. To reduce cost, procurement of a single laser instead of a single laser instead of a redundant system is probable. The marker laser is absent in this design since precision pointing is not being attempted. A scoring system is necessary however to verify point ahead algorithms operate properly. A scoring laser will be utilized for this function. The difference between where the system should point versus where it actually points as scored by the scoring laser return from retros on the hardbody will provide the point-ahead error. The key parameters of these lasers are listed in Table 3.5-1.

Type	Diode Array
Wavelength	0.81 micrometers
Pulse Rate	CW (modulated)
Power	5 watts
BQ	2
Divergence	10 mrad

Table 3.5-1 Laser Parameters.

3.5.2.2.3 Line-of-Sight Stabilization. No LOS stabilization system would be included. This would save cost, power, weight, and complexity. No fast steering mirror would be included in the main beam path.

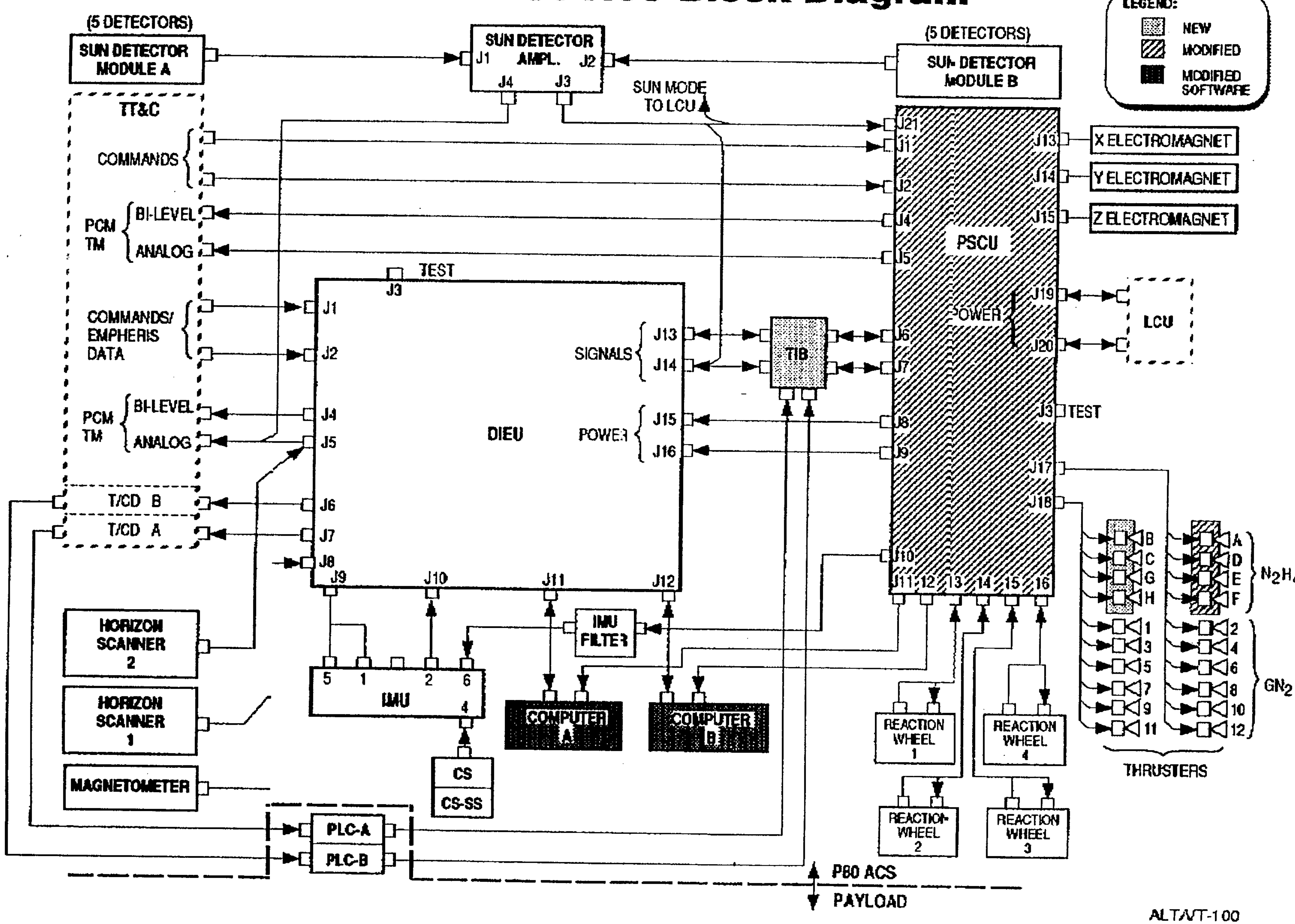
3.5.2.2.4 Payload Processors.

Track Processor. This system will be considerably simpler than ALTAIR and it is desirable that an off-the-shelf Hughes Dual Mode Tracker be used. Although the experiment capability will be reduced, performance will be consistent with the passive and active tracking requirements. The tracker will operate at ???? Hz.

Controller Processor. This processor will control the operation of the payload systems. It will normally operate in a script mode and execute the power up, sequencing, power down, and alternative operations of the payload elements. Its memory can be uploaded with new sequences, trajectories, gain settings, etc. As experience is gained with the hardware on-orbit, the tables can be updated through an upload. The spacecraft computer probably lacks the capability to handle this function since it relies on a pair of system control assemblies (SCAs) operating the Teal Ruby sensor and payload.

Data Handler. The assumption is that a 40 Mb/sec recorder would be available. The video that is being tracked at a given time would be recorded at full fidelity at a 30 Hz rate. The full frame rate of all the cameras would be recorded at a compression of about 10. The rest of the bandwidth would go to non-video data. 5 Mb/sec downlink would still be needed for the MITL. The functional block diagram shows the pieces necessary in the data handler. It includes a simple video switching matrix, a 3-channel data compression capability (each channel could be a small PC card), formatters (again, a couple of cards), a controller card, and the signal conditioning for the non-video data (probably 2-3 cards).

Modified P-80 ACS Block Diagram



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3.5.2.3 P-80 Spacecraft Bus.

Figure 3.5-3 Attitude Control & Determination.

3.5.2.3.1 Attitude Control and Determination System (ACDS). The P-80 bus is equipped with four independent systems for controlling spacecraft attitude. A set of four reaction wheels provide routine three-axis stabilization during quiescent periods. The authority of the wheels is limited (Figure 3.5-3) and probably in sufficient for any type of rapid slew. The configuration of P-80 with a large solar array on one side actually can create aerodynamic drag load sufficiently unsymmetrical at altitudes below ???? Km to exceed the torque capacity of the wheels. It is therefore highly desirable to conduct operations above this altitude. A magnetometer and torquing rod are normally to unload the excess wheel momentum.

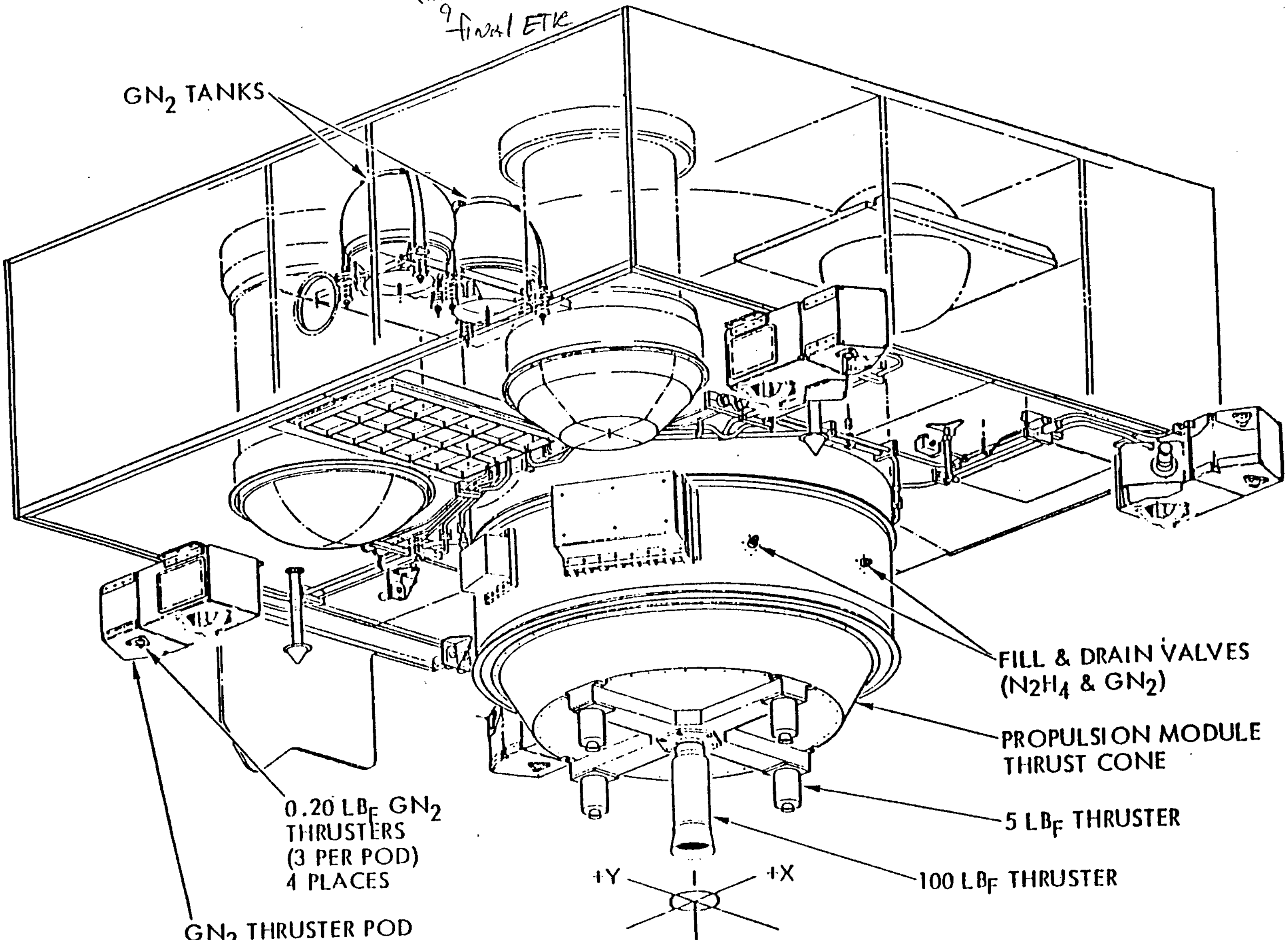
Two independent hydrazide systems are available, the primary system, as the spacecraft is currently configured consists of a TDRSS propulsion tank, four 5 lb thrusters and a single 100 lb thruster. The propellant capacity of this system is 740 lbs (?) when operate in a blow-down mode and within the qualification range of the 5 lb thrusters. A second hydrazide system has a capacity of 140 lbs of hydrazide. P-80 also has a gaseous nitrogen reaction control system. It consists of two tanks containing 6.5 lbs of GN2, each at 2400 psi, propellant with 12 - 0.2 lbs thrusters. The propulsion system is depicted in Figure 3.5-4. The sensors for the ACDS include a pair of horizon sensors, a celestial sensor, a Sun sensor, and an inertial measurement unit (IMU), as well as the magnetometer mentioned earlier. The specifications for these sensors are provided in Figure 3.5-3.

The ACDS is controlled through a redundant digital computer. In an ATP application,

P80-1 PROPULSION MODULE AND GN_2 REACTION CONTROL SYSTEM PERSPECTIVE

(WESTERN TEST RANGE CONFIGURATION)

final ETK



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Rockwell International

MJ-SSDB4226

(3-5-4)

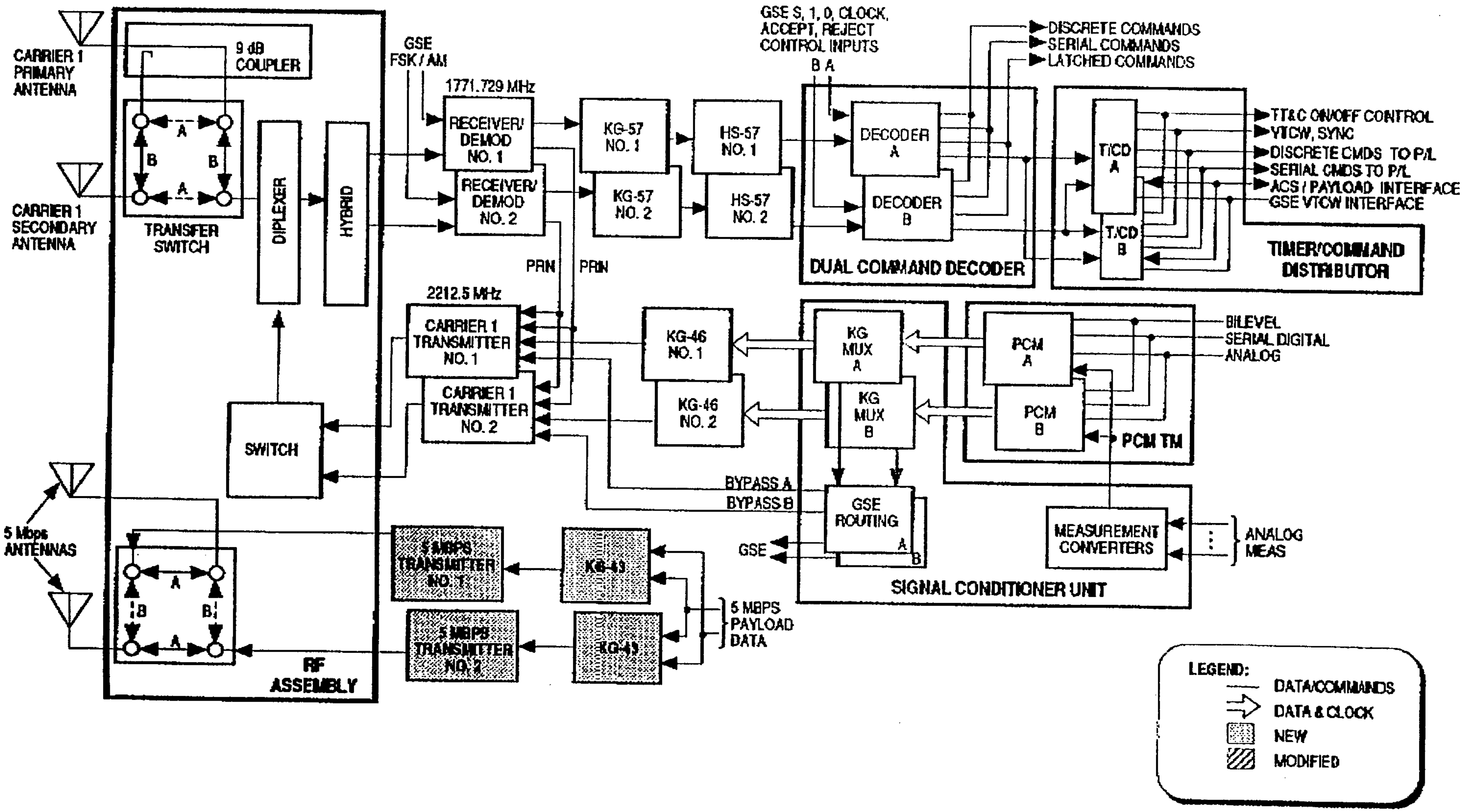
Figure 3.5-4 Propulsion System.

an interface with the payload processor to ACDS control to the payload track processor during an engagement. This interface would occur through a thruster interface box and is necessary because the P-80 processor lacks the processing capability to conduct the tracking experiments. The spacecraft software has a Sun safe mode which will use the GN2 system to ensure the solar panel is illuminated and the vehicle is in a safe mode, even when a major control anomaly occurs.

3.5.2.3.2 Telemetry, Tracking, and Control System (TT&C). The TT&C system on the P-80 is made up of redundant 1.024 Mbps S-band transmitters for downlink of science data, redundant 32 Kbps S-band transmitter for house keeping data, and a 2 Kbps uplink. The S-band system utilizes KG-46 encryptors that are standard government GFE items. The 2 Kbps link uses a KG-57/HS-57 decryptor, also a GFE item. The spacecraft was built to be equipped with three Odetics DDS 5500 recorders. These recorders are currently in storage at Odetics and continue to be serviced and tested on a routine basis. Although these recorders are of very high quality and the same model as those on the Hubble Space Telescope, their record rates are 1.024 Mbps and 32 Kbps, inadequate for the data rates produced by an ATP experiment. The TT&C system is depicted in Figure 3.5-5.

3.5.2.3.3 Electrical Power System (EPS). The EPS consists of a 131 sq ft solar array producing an orbit average power of 570 watts (Figures 3.5-4). P-80 was configured with a pair of 35 Ahr NiCd batteries and a pair of 85 Ahr AgZn for the propulsion module. The NiCd's will be replaced with 50 Ahr batteries which will require additional copper on the bus. This change will permit the EPS to handle peak power draws of approximately 2500 watts during the expected engagements. Excess electrical power from the solar array is

Modified P-80 TT&C Block Diagram



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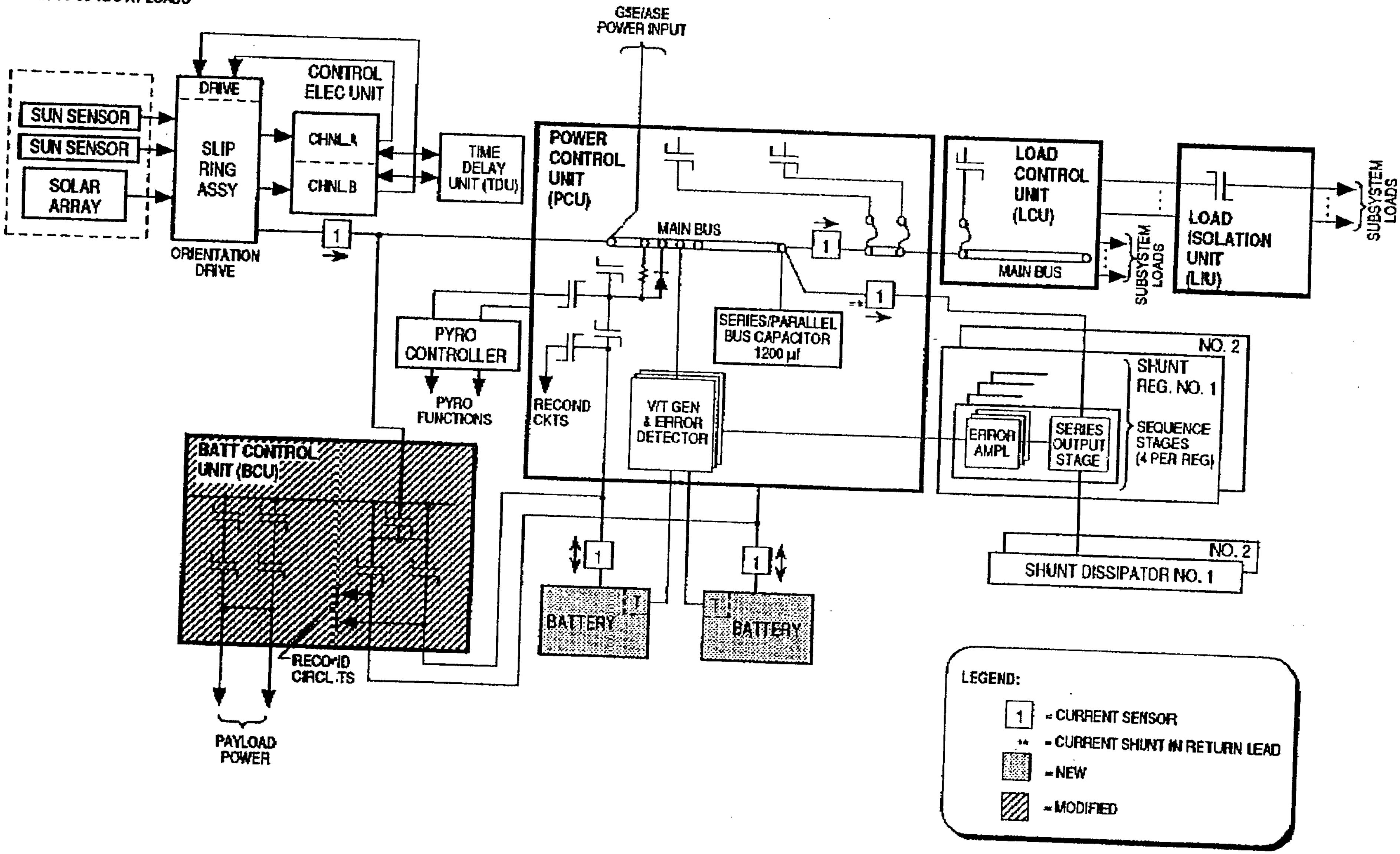
Modified P-80 EPS Block Diagram

STEADYSTATE VOLTAGE REQUIREMENTS
- 24 TO 33 VDC AT LOADS



ALT/VT-101

Figure 3.5-5 TT&C System.

Figure 3.5-6 EPS System.

dissipated through a pair of shunts.

3.6 Hardbody Identification & Plume Dynamics.

3.6.1 Introduction. The principal job of the acquisition, tracking, and pointing/fire control (ATP/FC) system of a space-based weapon platform is to detect the target and then to estimate the target's position, velocity, acceleration, rotation, and aspect with sufficient detail for a weapon to engage and destroy the target. The attributes of the target are collectively referred to as the state vector of the target. The purpose of an ATP/FC system is to estimate the target's state vector well enough to engage and destroy it with a weapon. For a directed energy system, this means pointing a beam at a vulnerable location on the target. For a kinetic energy weapon, this means pointing the interceptor so that it can strike the hard-body. The ALTAIR program was to address several of the functions required to accomplish ATP for DEW weapons.

3.6.2 Background. On 22-23 October, a conceptual design review of ALTAIR was held at the Phillips Laboratory, Albuquerque, New Mexico. Two items of major significance emerged from this review. First, from a technical standpoint, no plume to hard-body handover algorithm has yet been shown to work. Simulation results presented using highly idealized and symmetric plumes were still not achieving hard-body handover. Accurate simulation of the entire fire control process is needed and must be refined to include actual plume data. Second, from a fiscal perspective, creeping cost growth in the face of diminishing funding appeared ominous.

On 4 November 1991, a meeting was convened at Phillips Lab where representatives from SDI made clear the fiscal constraints. Any program must be carried out within three (3) years at a total cost not to exceed $150M. Funds would be provided at a rate of $50M per year.

Three panels were formed to study and evaluate various options. The low-cost space experiment panel was charged with the task: "Investigate the feasibility of resolving critical issues through conducting inexpensive orbital and/or suborbital space experiments. Develop technical approach, cost and schedule data for variable tests."

3.6.3 Experiment in Hardbody Identification and Plume Dynamics. The Applied Physics Laboratory was asked to serve on the low-cost space experimental panel. In addition to providing inputs to the panel, the Laboratory felt that an experiment to validate the plume to hard-body handover problem was an essential ingredient to all SDI programs. An experiment was conceived with the following constraints in mind:

a. It must address the fundamental issues common to all SDI intercept missions, i.e., the plume to hard-body handover and the associated algorithms.

b. Design to cost: Specified at $150M total at a rate of $50M per year for three years.

c. It must be responsive to as many of the critical technical issues as the cost constraint permits.

3.6.3.1 Experiment Implementation. To effect the economies of a design-to-cost program, extensive use of existing hardware, designs and facilities is mandated. Experience gained from Deltas 180, 181 & 183, all implemented by the Laboratory and its associated contractors, provides a base for technical possibilities and cost estimates. Current experience with the Mid-course Space Experiment spacecraft indicates its suitability as a platform for pertinent instruments. An existing ground station network will complement the space segment and a mission operations team in place for MSX would support this mission. The TITAN II launch vehicle provides ample performance capability, is available and included in the total cost. In summary, the Laboratory concludes that useful, necessary experiments can be performed within the $150M constraint.

3.6.3.2 Experiment Objectives. The objectives of this experiment are as follows:

a. Conduct a controlled experiment to collect data which addresses the critical problems of plume to hard-body hand-off, illuminator point ahead and hard-body illumination in the presence of a plume.

b. Examine the viability of current plume to hard-body hand-off algorithms and assess new algorithms in orbit.

c. Demonstrate, in the space environment, plume to hard-body hand-off and hard-body illumination in the presence of a plume.

3.6.3.3 System Approach. The experiment will be conducted from a free-flying spacecraft in low earth orbit. This spacecraft will carry on board instrumented target rockets. These rockets will be the means of validating the algorithms key to all acquisition, pointing and tracking scenarios. In addition, instrumented ground-launched targets will provide realistic test for further validation.

The sensor suite will consist of a visible acquisition camera whose primary purpose will be for wide field imaging and
ground-launched target acquisition. The primary science sensor will be the IR imager collecting high-resolution plume data at 30 frames/sec in the 4.3 μm band. Another critical science instrument will be the narrow field of view intensified visible imager with a narrow band filter at .532 μm. Crucial to the measurement of hard-body and plume reflectance is the laser Radar/Illuminator for tracking and target illumination. A quad cell sensor will also

be
incorporated into the optical path for tracking the optical beacon included on the target rockets.

The following illustrates the sequence and technique that will be used for an experiment using an on-board target.

First, a spinning target rocket with a narrow band optical beacon and corner reflector will be released from the spacecraft. Tracking on the optical beacon with a low bandwidth track loop closed around the pointing flat and a quad cell will be initiated immediately after release. This will establish initial hard-body location with the beacon. Hard-body location will also be established with the laser radar and corner reflector. When the pre-established range has been reached, the target motor will be ignited. The track loop will continue to be closed around the beacon and quad cell. Backup tracking will be performed by the laser radar. The high bandwidth line of sight stabilization loop will be closed around the quad cell and the steering mirror. IR data of the plume in the region which encompasses the hard-body will be collected by the IR imager. Hard-body location is established by the beacon and quad-cell. This technique will unambiguously establish the plume and beacon (hard-body) relationship.

Using algorithms validated with this experiment, a second experiment will be performed that will predict the hard-body location and the required beam "Point Ahead." The prediction will then be scored on the beacon and corner reflector. Hard-body and plume illumination will then be accomplished using the illuminator beam from the laser and data will be collected by the narrow field visible camera for analysis on the ground.

The experiments described will provide for autonomous acquisition, will collect the relevant imaging data critical to the development of plume to hard-body hand-off algorithms and will examine the question of active plume and hard-body reflectivity.

3.6.4 Summary. An assessment was made as to how many of the critical technical issues to be addressed by ALTAIR would be addressed by the low-cost space experiment. Please note in the final page of the presentation material, a comparison of those issues concerned. The MSX spacecraft, scheduled for launch in the last quarter of calendar year 1993, will complement PBV and mid-course issues not addressed by ALTAIR or the LCSE. Cost and schedule information is presented in the attachment to this summary.

Section 3.8

SHUTTLE PALLET SATELLITE (SPAS) III ACQUISITION, TRACKING, POINTING AND FIRE CONTROL (ATP-FC) EXPERIMENT

OVERVIEW

This experiment will be a functional demonstration to prove the feasibility of ATP/FC for use in advanced Directed Energy system concepts. The experiment will use the LPG or other shuttle released objects as the targets, and also ground launched targets for an extended mission. A short duration experiment (1-2 days) and a long duration experiment (12 months) are required. The short and extended experiment rationale is to complement each other to meet the overall ATP/FC objectives.

OBJECTIVE

The objective of this experiment will be to perform an end-to-end acquisition, tracking and pointing (ATP) functionality experiment. ATP requirements for this experiment include:

- Boresight and alignment
- Pointing control and disturbance rejection
- Target acquisition and track at ranges of (20 - 500km)
- Plume to hardbody handover
- Passive/active track
- Aimpoint selection and maintenance
- Plume phenomenology
- Integrated ground and space operations

The short duration experiment will measure the LPG or other shuttle launched targets. The extended experiment will use these on ground launched targets, and will collect long term background data and be able to observe targets of opportunity in a self contained experiment. Target boards are highly desired on any one of the target options.

SENSOR REQUIREMENTS

Sensor requirements support the ATP functions and other plume measurements, and are generally described in the Experiment Planning section below. The sensor suite will provide long term seasonal background measurements.

TARGET REQUIREMENTS

The targets will be the LPG at ranges of 5-150 km for the short duration experiment and 150-500 km for the extended experiment. Shuttle launched, get-away targets can also be used if they meet the ATP/FC experiment requirements. The long duration experiment should use the LPG or ground launched targets. It is highly desired that the targets include a laser scoring target board. The target measurements will be done in conjunction with other phenomenology measurements to maximize data return and synergism.

EXPERIMENT PLANNING

There are three deployment options being considered:

Option 1. An ATP experiment is performed as an integral part of the SPAS III baseline mission (6-8 days).

Option 2. An ATP experiment package is integrated with the SPAS III Liquid Plume Generator (LPG) to provide an integrated satellite for a long duration (12 months) ATP experiment. The long duration experiment will occur after the conclusion of the SPAS III baseline mission.

Option 3. This is a variation of Option 2. A small ATP satellite is integrated with the LPG such that the LPG provides an orbit transfer, the two objects separate at mission orbit, and the LPG is used as a thrusting target for a long duration (12 months) ATP experiment. Planned targets include Minuteman.

This experiment will require significant down-scaling of both the number and size of the instruments originally projected for the Altair Experiment. Emphasis is on functional demonstration of end-to-end DEW ATP-FC, including active track of hardbodies, hand over from the passive plume, and precursor estimate of aim point designation performance.

The ground rules are as follows:

- For Option 1, ATP instruments and augmentation hardware will be deployed on the SPAS III pallet, or will be integrated with the LPG. The ATP unique system for Options 2 and 3 will comply with the SPAS host platform's (pallet or LPG) ballast form, fit, and function guidelines.
- Option 2 and 3 ATP experiments will be defined to be as transparent (minimum impact) as possible for LPG/Shuttle integration and SPAS mission activities.
- The minimum propellant budget required for Options 2 and 3 will be determined to assess the impact and constraints placed on the SPAS III baseline mission.

For Option 1, an active imaging system will be deployed on the SPAS pallet to measure passive visible and laser radiation from the LPG during the planned SPAS baseline mission. The laser will be a diode pumped Nd:YAG with approximately one watt power at 1.06 µm. A visible focal plane receiver will provide spatial resolution at one square meter.

For Options 2 and 3, a small satellite will be designed which meets the SPAS form, fit, and function requirements for the SPAS III baseline mission. The LPG would provide orbit transfer for the smallsat to approximately 450 kilometers and 40 degrees inclination, assuming the SPAS mission is performed at 40 degrees. The instrument suite is projected to include a 40 centimeter telescope, a 60 watt Nd: YAG laser illuminator, and two visible focal plane instruments. Functions to be performed include acquisition, coarse intermediate tracking, and active fine tracking.

EXPERIMENT IMPLEMENTATION

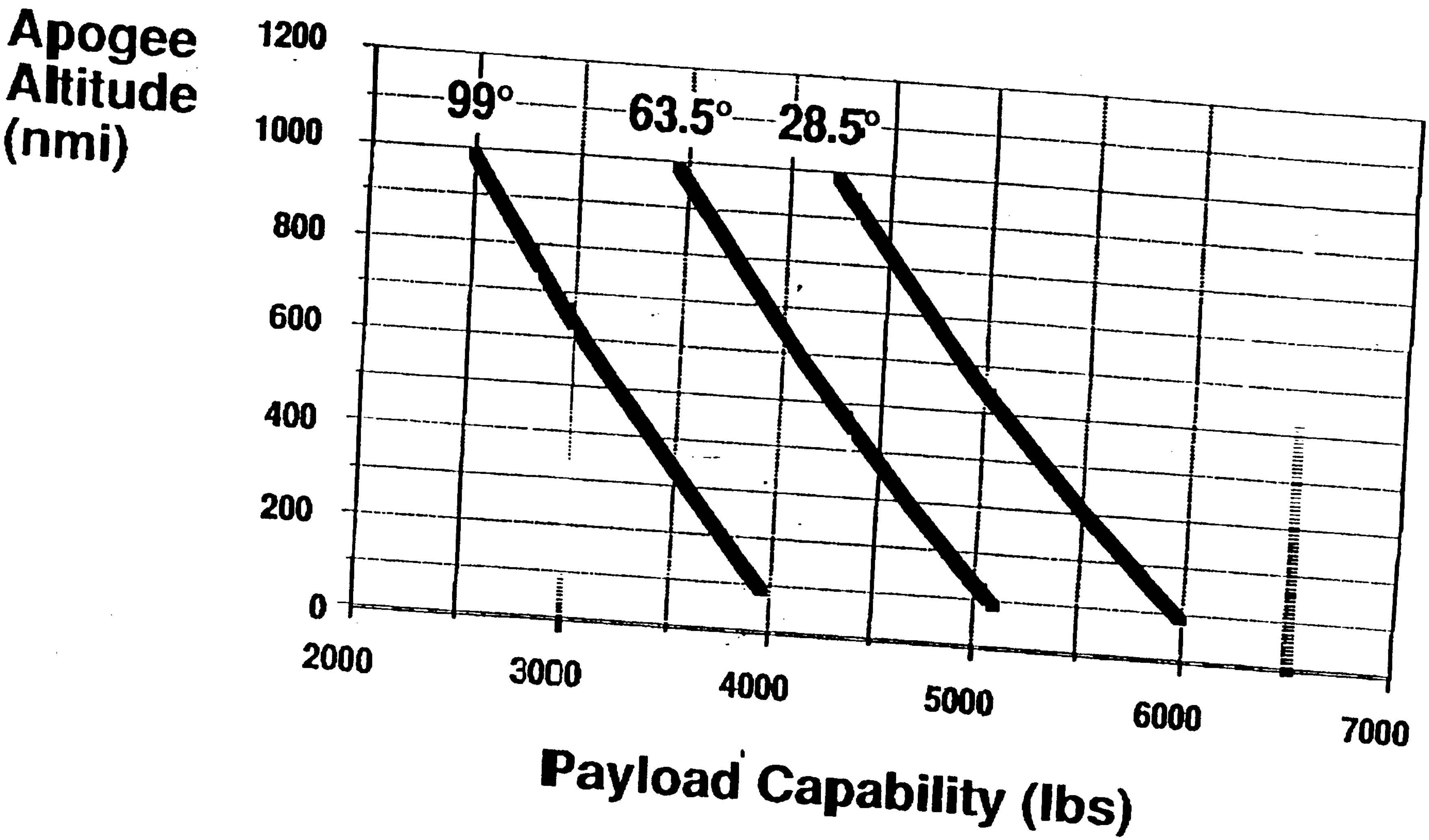
Fairchild Space & Defense Corporation, SPAS III System Integration Contractor, is currently conducting a study to assess the technical, cost and schedule feasibility of the three options outlined above. A preliminary report will be provided to SDIO on 18 December 1991. A second, more detailed report will be submitted on 15 January 1992.

RIGEL CONCEPT

WHY CONSIDER RIGEL?

- LOW RISK APPROACH TO DEMONSTRATE AUTONOMOUS END-TO-END ATP-FC
- UTILIZE EXISTING "OFF-THE-SHELF" TECHNOLOGY
- COMPATIBLE WITH EVOLVING SMALL SATELLITE TECHNOLOGY

Titan II G Elliptical Performance



Perigee Altitude: 100 nmi

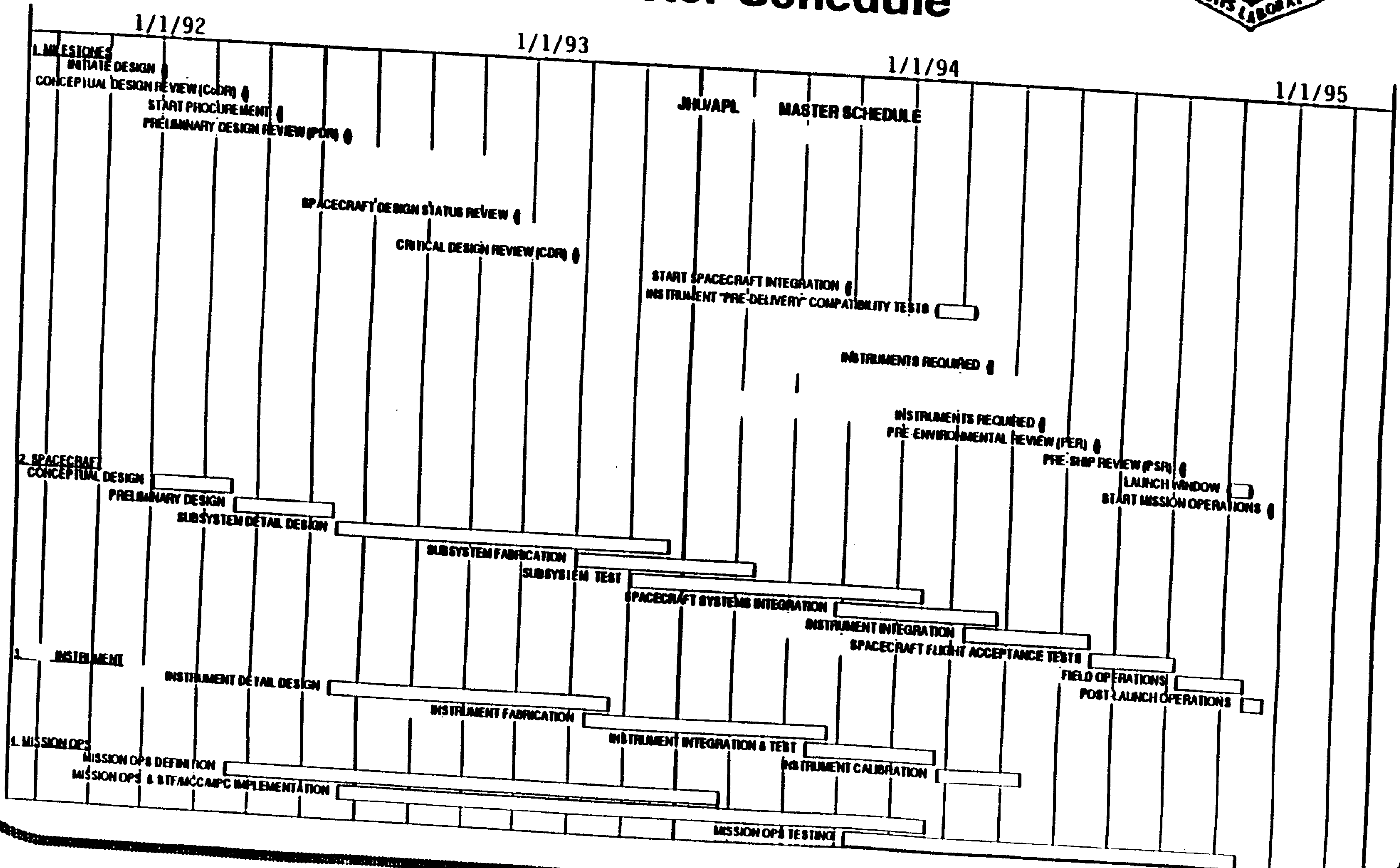
LOW COST SPACE EXPERIMENT
PRELIMINARY COST ESTIMATE SUMMARY

	Total (K dollars)
Management, Administration	4232
Spacecraft	66725
Instrument	20563
Launch Vehicle	20000
Mission Ops	3133
Data Reduction	10000
	124,653
Targets	
STAR-13 (4)	1200
S-13 Refurb (3)	1500
MM II (3)	6000
	133353
*Management Reserve	18033
	151386

* 15% x (M&A, SC, INST, LV, TGTS)

Hardbody Identification & Plume Dynamics

Master Schedule



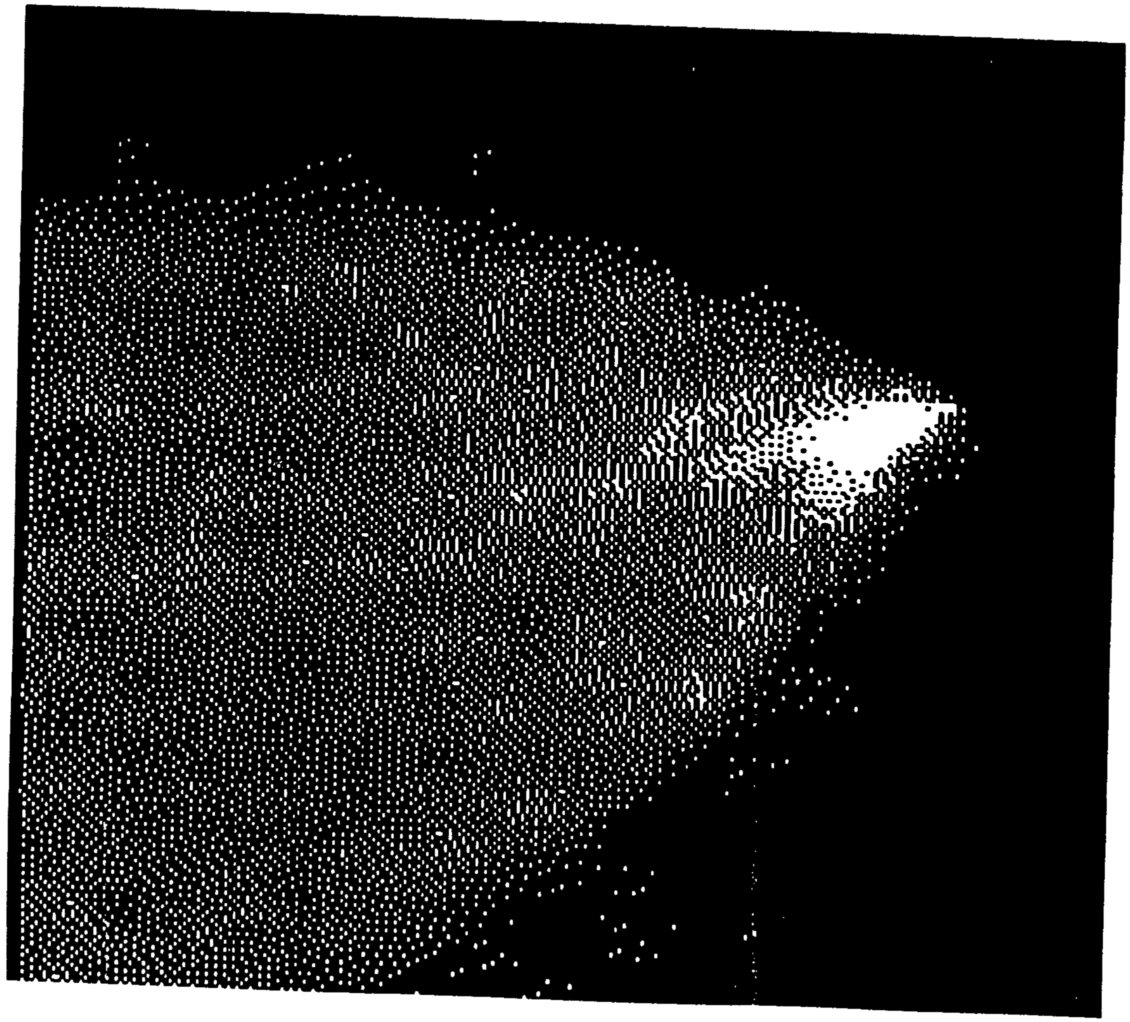
Basis For Concept

- Design to Cost < $150 M
- Design to Schedule < 3 yrs
- Design Objectives that Are Achieveable Given Above Listed Constraints

OBJECTIVES

- Conduct a Controlled Experiment to Collect Data which Addresses the Critical Problems of Plume to Hard-body Hand-Off, Illuminator Point Ahead and Hard-body Illumination In the Presence of a Plume.
- Examine the Viability of Current Plume to Hard-body Hand-Off Algorithms and Assess New Algorithms in Orbit
- Demonstrate, in the Space Environment, Plume to Hard-body Hand-Off and Hard-body Illumination In the Presence of a Plume

Hardbody Identification & Plume Dynamics



Hardbody Identification & Plume Dynamics

SYSTEM APPROACH

- Free Flying Spacecraft in L.E.O.
- On Board Instrumented Target Rockets
- Instrumented Ground Launched Targets
- Autonomous Sequencing

Hardbody Identification & Plume Dynamics

Phased Experimental Approach

Experiment 1.

a. Release Spinning Target Rocket with Beacon & Corner Reflector
b. Establish Hard-Body Location with a Beacon & Corner Reflector
c. IgniteTarget Motor and Track on Plume
d. Establish Plume and Beacon (Hard-Body) Relationship

Experiment 2.

a. Repeat a , b & c Above
b. Using Algorithims Validated with Experiment 1, Predict Hard-body Location & Point Ahead
c. Score Prediction on Beacon and Corner Reflector
d. Illuminate Hard-body & Corner Reflector with Laser Radar

Hardbody Identification & Plume Dynamics

Phased Experimental Approach

Experiment 3.

a. Repeat a , b & c Above

c. Using Algorithims Validated with Exp. 1 & 2
Predict Hard-body Location & Point Ahead

d. Illuminate Hard-body Based on c.

e. Score on Beacon & Corner Reflector

Experiments 4,5,6 or 2,4,6

Repeat 1,2,3 With Ground Launched Representative Target

Hardbody Identification & Plume Dynamics
Instrument Requirements

- Spatial & Temporal Resolution
  - a. IR Tracker Must Localize Hard-body & Therefore Plume to ~ 1m to insure Plume to Hard-body Hand-off
  - b. Vis. Imager Must Define Hard-body to ~ 0.1m to Adequately Image Laser Return
  - c. Sample Rate must be Commensurate with Plume Dynamics & Control Loop

- Spacecraft Must be <u>Small & Low Cost -- Use A Shared 20 cm Aperture</u>

- Diffraction Limit @ 532nm ~ 3 μr, Determines Range to Target

  For ~0.1 m Resolution Range < ~40 km

- Instrument FOVs Assume 200 x 200 Pixel CCD
  - a. L IR ~ 0.035°
  - b. IR ~ 0.35°

Instrument Suite

- VISIBLE ACQUISITION CAMERA
  - SONY INDUSTRIAL CAMERA
  - 3 - 5° FOV; DIGITAL OUTPUT
- INFRARED SENSOR
  - GE ASTRO INFRARED CAMERA
  - PtSi; 4.3 μm FIXED FILTER; DIGITAL OUTPUT
- VISIBLE SENSOR
  - INTENSIFIED VISIBLE CAMERA
  - TWO FILTERS (OPEN, 532 nm)
- LASER RADAR
  - McDONNELL DOUGLAS DIODE-PUMPED DOUBLED YAG
  - 532 nm; 200 mJ PER PULSE; 10 PULSES PER SECOND
- QUAD CELL
  - LOS STABILIZATION; CENTROID TRACKING

Hardbody Identification & Plume Dynamics

Instrument Summary

SENSOR	VISIBLE ACQUISITION	INFRARED SENSOR	VISIBLE SENSOR
MANUFACTURER	SONY	GE ASTRO	APL
HERITAGE	DELTA STAR	DELTA STAR	DELTA STAR
FIELD OF VIEW	5°	0.35°	0.10°
VERTICAL RESOLUTION	256	160	600
HORIZONTAL RESOLUTION	256	244	600
BITS PER PIXEL	8	8	8
FRAMES PER SECOND	5	30	3
OUTPUT BIT RATE (Mbps)	2.50	8.94	8.24
SCIENCE DATA	√	√	√
TRACK DATA	√		

Hardbody Identification & Plume Dynamics

Instrument Weight & Power

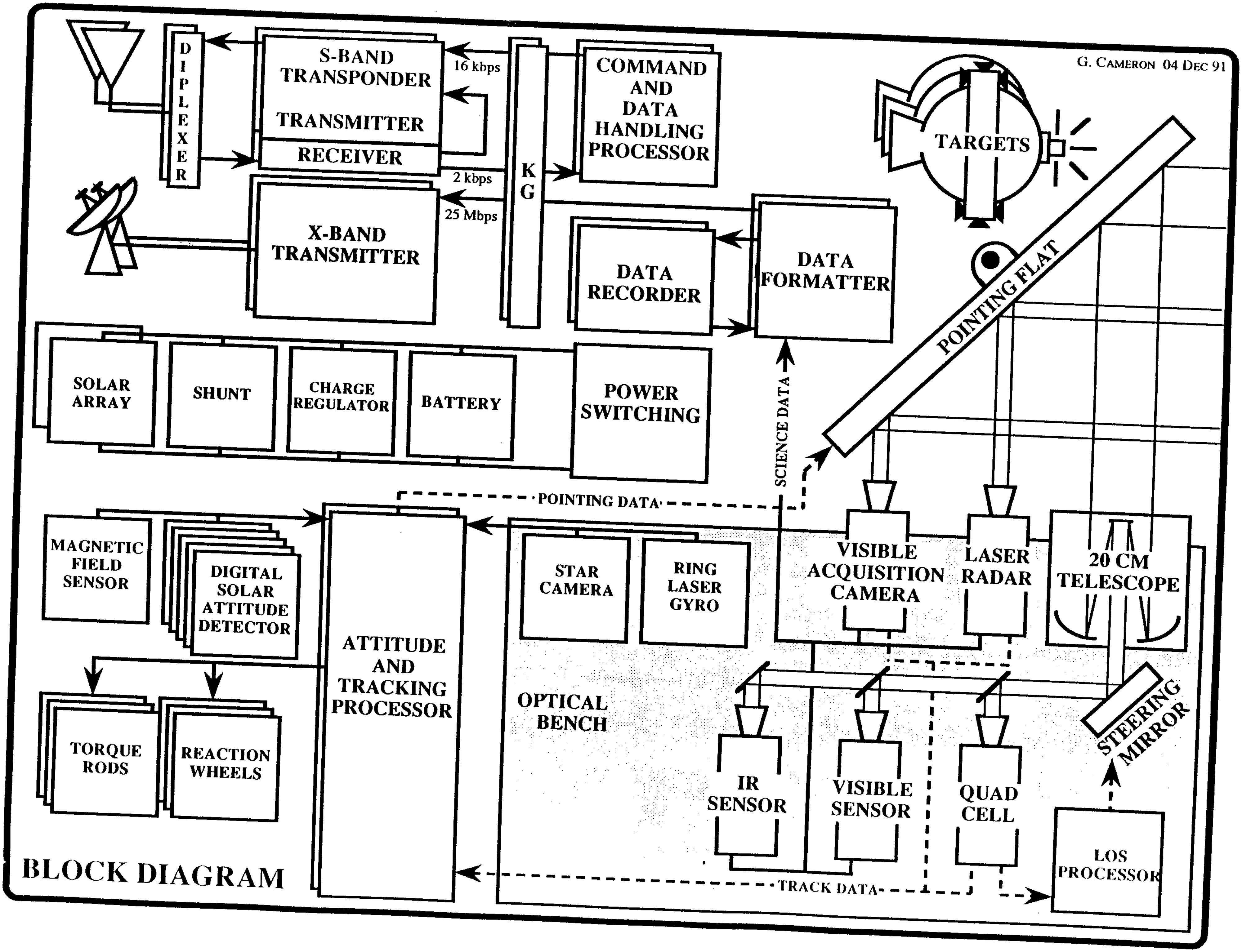
Heritage

Instrument	Power	Weight	Heritage
Telescope	NA	40 lbs	Delta 183
Acquisition	25 W	20 lbs	Delta 183
IR Track	75 W	35 lbs	Delta 183
Vis Imager	32 W	41 lbs	Delta 183
Laser Radar	100 W	50 lbs	B.P. & M.O.
Image Processor	30 W	20 lbs	DMT
---------	---------	----------	
Total	262 W	206 lbs	

Hardbody Identification & Plume Dynamics

Spacecraft Concept

SYSTEM	Heritage
• Three Axis Reaction Wheel Stabilzation	MSX
• Two Axis Gimbaled Optical Path	Delta 181
• 25 Mb/sec Realtime Downlink	MSX
• 5 -10 Mb/sec Onboard Recording 1 G bit Capacity	NASA



BLOCK DIAGRAM

Attitude Subsystem

- ATTITUDE PROCESSOR (2)
  - HONEYWELL GVSC (3): 9 MIPS; 1 Mb RAM; RAD-HARD
- STAR CAMERA
  - BALL (MSX): 10 μrad; 6th MAG STARS
- DIGITAL SOLAR ATTITUDE DETECTORS (5)
  - ADCOLE (MSX): 0.5° QUANTIZATION; 0.25 ° ACCURACY
- MAGNETIC FIELD SENSOR (1)
  - SCHONSTEDT (MSX): 3-AXIS ANALOG; 1% ACCURACY
- RING LASER GYROS (2)
  - HONEYWELL (MSX): 0.01°/hr^(1/2); 5 μrad NOISE
- REACTION WHEELS (4)
  - HONEYWELL (MSX): 1 ft-lb TORQUE; 100 ft-lb-s MOMENTUM
- TORQUE RODS (3)
  - APL: 100 A-m^2

C & DH Subsystem

- COMMAND AND TELEMETRY PROCESSOR (2)
  - GULTON: UTMC MIL-STD-1750A @ 6 MHz; MIL-STD-1553 BUS
- KG/TEMPEST UNIT
  - CUBIC (MSX): RICEBIRD CHIPSET (KG-A, KG-B)
- DATA FORMATTER
  - APL (MSX): FIXED FORMAT; 25-Mbps AND 10-Mbps OUTPUTS
- DATA RECORDER
  - ODETICS: SOLID STATE; 1.8 Gbit CAPACITY; 10 Mbps RATE
- POWER SWITCHING UNIT
  - APL (MSX): (24) 25-AMP; (40) 10-AMP; (32) 5-AMP; (128) 2-AMP

RF Communications Subsystem

- S-BAND TRANSPONDERS (2)
  - MOTOROLA (MSX): 2 kbps SGLS UPLINK; 16 kbps DOWNLINK
- DIPLEXER
  - MOTOROLA (MSX): SGLS COMPATIBLE
- S-BAND ANTENNAS (2)
  - APL (MSX): DUAL HEMISPHERICAL QUADRIFILAR HELICES
- X-BAND TRANSMITTERS (2)
  - APL (MSX): DIFFERENTIAL QPSK; 5W OUTPUT POWER
- X-BAND ANTENNAS (2)
  - APL (MSX): 8-IN ALUMINUM DISH; BACKFIRE HELIX FEED

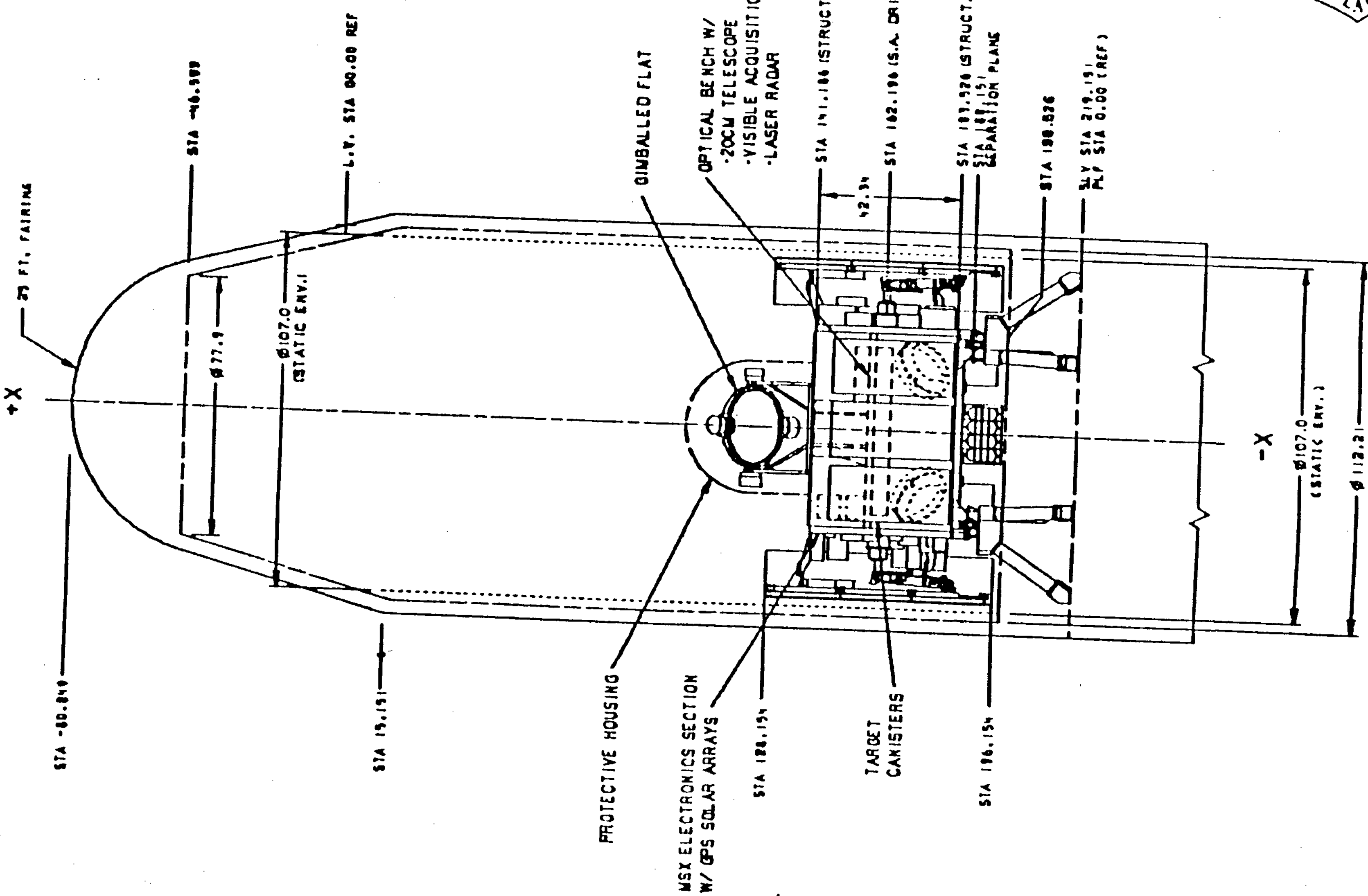
Electrical Power Subsystem

- SOLAR PANELS (2)
  - ROCKWELL (MSX OR GPS): ROTATABLE 150 SQ. FT. ARRAY
- SHUNT
  - APL (MSX): NON-DISSIPATIVE DIGITAL SHUNT
- CHARGE REGULATOR
  - APL: FOUR V/T LIMITS FOR CHARGE CONTROL; 2 CURVES
- BATTERY
  - SINGLE 22-CELL 50-A-HR NICKEL CADMIUM BATTERY

Other Hardware

- TELESCOPE
  - LORAL (LACE, PATHFINDER): 20 cm APERTURE
- STEERING MIRROR
  - BALL AEROSPACE
- POINTING FLAT
  - HONEYWELL: 20 ft-lb; 2-AXIS; 70° FOR; LOW NOISE
- OPTICAL BENCH
  - COMPOSITE OPTICS
- TARGETS
  - APL: STAR-13 MOTOR; OPTICAL BEACON; CORNER CUBE
- STRUCTURE
  - APL (MSX): HONEYCOMB PANELS; ALUMINUM FRAME

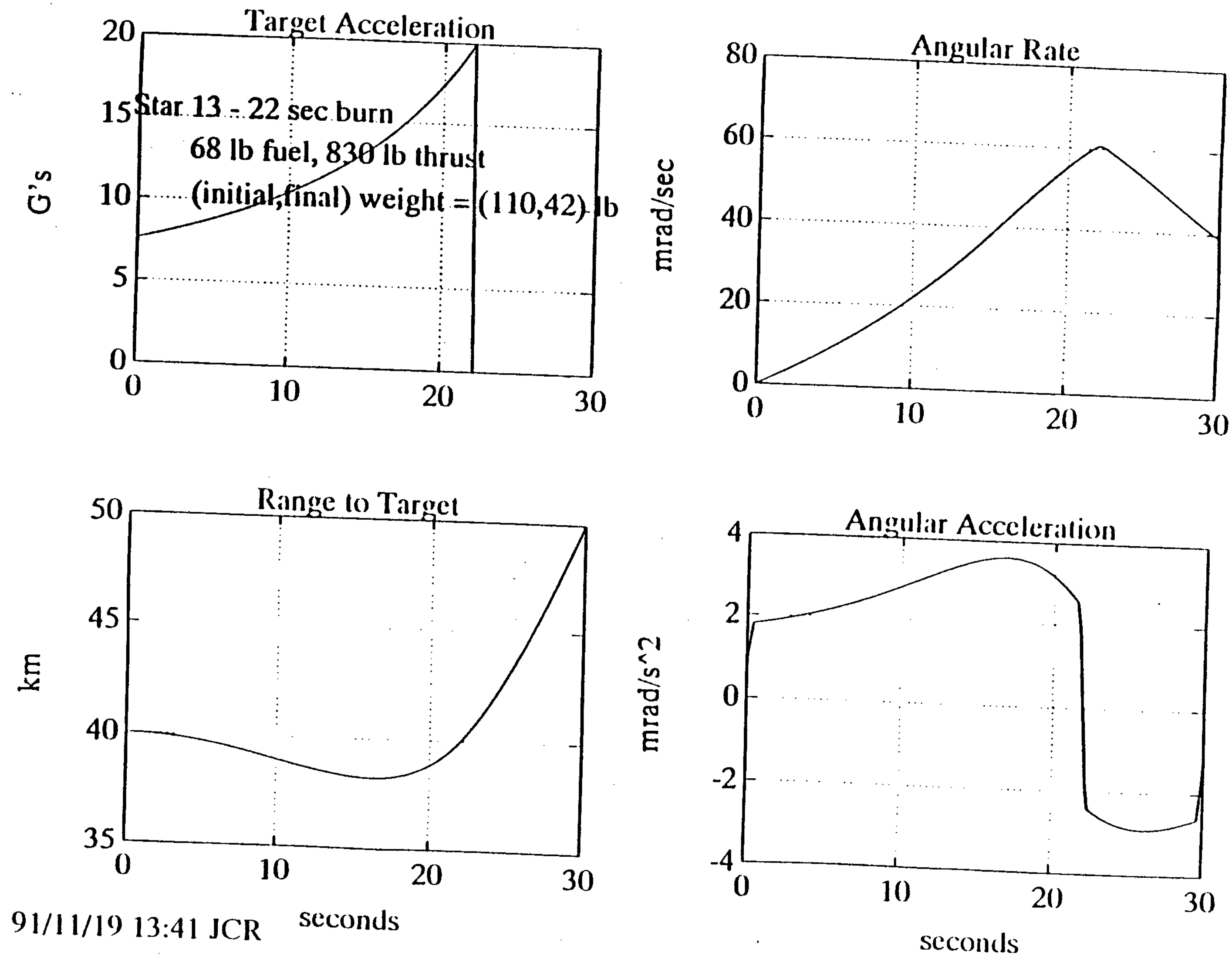
Hardbody Identification & Plume Dynamics



Data Collection

- Onboard Target Events
  - Realtime Downlink, X Band @ 25Mb/sec
  - 5-10 Mb/sec Recorded Compressed
- Ground Launched Targets
  - Wallops Island
    - Realtime @25Mb/sec
    - 5-10 Mb/sec Record/Playback
  - WTR
    - 5-10 Mb/sec Record/Playback

Hardbody Identification & Plume Dynamics



Hardbody Identification & Plume Dynamics

Major Issues Addressed

- Acquisition
- Plume to Hardbody Hand-Off
- Plume & Hardbody Reflectivity

Hardbody Identification & Plume Dynamics

CRITICAL TECHNICAL ISSUES TO BE ADDRESSED

	TECHNICAL ISSUE	BOOSTER	PBV	MIDCOURSE
I.	COARSE POINTING/TARGET ACQ	100%	MSX	MSX
II.	TARGET TRACK/TARGET ID	100%/0	MSX	MSX
III.	PASSIVE TRACK HANDOVER	100%		
IV.	PASSIVE INTERMEDIATE TRACK	100%		
V.	PLUME-TO-HARDBODY HANDOVER	100%		
VI.	ILLUMINATOR POINT AHEAD/ ACTIVE TRACK HANDOVER	100/30%		
VII.	HARDBODY DISCRIMINATION/ ACTIVE FINE TRACK/ AIMPOINT SELECTION	100% 0 0		
VIII.	PRECISION POINT AHEAD/ AIMPOINT DESIGNATION	0		
IX.	PRECISION BEAM POINTING AT RATE	20%		
X.	AUTONOMOUS SEQUENCING	100%	MSX	MSX

Hardbody Identification & Plume Dynamics

CRITICAL TECHNICAL ISSUES TO BE ADDRESSED (CONTINUED)

XI.	PBV BUS TRACKING--TTP, IPP, HANDBACK DATA	50%	MSX	MSX (NOΔV)
XII.	PBV BUS WATCHING--ΔV	100%	MSX	MSX
XIII.	PBV BUS WATCHING--OBSERVABLES	100%	MSX	MSX
XIV.	ACTIVE FINE TRACK OF MIDCOURSE OBJECTS	0	MSX	MSX
XV.	MIDCOURSE OBJECT--TTP, IPP, HANDBACK DATA	0	MSX	MSX
XVI.	MIDCOURSE OBJECT-- V	0	MSX	MSX
XVII.	GENERAL PLUME PHENOMENOLOGY	0	MSX	MSX
XVIII.	GENERAL BACKGROUND CLUTTER	0	MSX	MSX

CRITICAL TECHNICAL ISSUES TO BE ADDRESSED BY ALTAIR

	TECHNICAL ISSUE	BOOSTER	PBV	MIDCOURSE
I.	COARSE POINTING/TARGET ACQ	√ 100%	√ MSX	Note 1 MSX
II.	TARGET TRACK/TARGET ID	√ 100/0%	√ MSX	Note 1 MSX
III.	PASSIVE TRACK HANDOVER	√ 100%	√	Note 1
IV.	PASSIVE INTERMEDIATE TRACK	√ 100%	√	Note 1
V.	PLUME-TO-HARDBODY HANDOVER	√ 100%	√	Note 2
VI.	ILLUMINATOR POINT AHEAD/ ACTIVE TRACK HANDOVER	√ 100/30%	√	√
VII.	HARDBODY DISCRIMINATION/ ACTIVE FINE TRACK/ AIMPOINT SELECTION	√ 100% 0 0	√	Note 3
VIII.	PRECISION POINT AHEAD/ AIMPOINT DESIGNATION	√ 0	√	√
IX.	PRECISION BEAM POINTING AT RATE	√ 20%	√	√
X.	AUTONOMOUS SEQUENCING	√ 100%	√ MSX	Note 1 MSX
XI.	PBV BUS TRACKING—TTP, IPP, HANDBACK DATA	— 50%	√ MSX	— MSX (NO ΔV
XII.	PBV BUS WATCHING—ΔV	— 100%	√ MSX	√ MSX
XIII.	PBV BUS WATCHING—OBSERVABLES	— 100%	√ MSX	√ MSX
XIV.	ACTIVE FINE TRACK OF MIDCOURSE OBJECTS	— 0	— MSX	√ MSX
XV.	MIDCOURSE OBJECT—TTP, IPP, HANDBACK DATA	— 0	— MSX	√ MSX
XVI.	MIDCOURSE OBJECT—ΔV	— 0	— MSX	√ MSX
XVII.	GENERAL PLUME PHENOMENOLOGY	√ 0	√ MSX	Note 2 MSX
XVIII.	GENERAL BACKGROUND CLUTTER	√ 0	√ MSX	Note 1 MSX.

Note 1: Since the ALTAIR experiment does not incorporate a LWIR sensor, ALTAIR will not have a traceable function for mid-course object acquisition and passive track. For midcourse targets, the ALTAIR experiment will focus on issues regarding active track and precision beam pointing.

Note 2: This issue does not apply to midcourse targets since midcourse objects have no plume.

Note 3: This issue does not apply to midcourse since it involves the illumination and active track of an *extended target* under *thruster acceleration* in the *presence of a plume*. Midcourse active tracking is considered in Issue XIV.

COMPONENT	HERITAGE	MASS (LBS)	SIZE (INCHES)	BUDGET (W) PARK MODE AVG.	PARK MODE PEAK	ATP MODE AVG.	ATP MODE PEAK	DUMP DATA AVG.	DUMP DATA PEAK	LAST UPDATE
PAYLOAD										
VISIBLE ACQUISITION CAMERA		20.00	12" X 12" X 6"	2.00	2.00	25.00	25.00			
VISIBLE SENSOR		41.00	12" X 12" X 6"	2.00	2.00	32.00	32.00			
INFRARED SENSOR		35.00	12" X 12" X 10"	5.00	5.00	55.00	75.00			
LASER RADAR		50.00		10.00	10.00	100.00	100.00			
QUAD CELL		5.00	2" X 2" X 4"	2.00	2.00	5.00	5.00			
LOS PROCESSOR		10.00		2.00	2.00	10.00	10.00			
20 cm TELESCOPE		50.00	10" X 10" X 25"	5.00	5.00	5.00	5.00			
POINTING FLAT		90.00	20" X 20" X 3"	10.00	10.00	10.00	10.00			
STEERING MIRROR		5.00	2" X 2" X 4"	5.00	5.00	5.00	5.00			
TRASFER OPTICS		50.00		10.00	10.00	10.00	10.00			
OPTICAL BENCH		50.00		10.00	10.00	10.00	10.00			
SUBTOTAL		406.00		63.00	63.00	267.00	287.00	0.00	0.00	
ATTITUDE CONTROL										
RWA #1	MSX	60.00	18" Dia. X 9"	50.00	50.00	50.00	50.00	50.00	50.00	
RWA #2	MSX	60.00	18" Dia. X 9"	50.00	50.00	50.00	50.00	50.00	50.00	
RWA #3	MSX	60.00	18" Dia. X 9"	50.00	50.00	50.00	50.00	50.00	50.00	
STAR CAMERA	MSX	39.60	7" Dia.,18" L	2.00	20.00	2.00	20.00	2.00	20.00	
RING LASER GYRO	MSX	15.35	see below	19.20	19.20	19.20	19.20	19.20	19.20	
SENSOR	MSX	-	5",5",5.5"	-	-	-	-	-	-	
ELECTRONICS	MSX	-	8",4.5",8"	-	-	-	-	-	-	
MAGNETOMETER AND ELECTRONICS	MSX	4.20	see below	1.50	1.50	1.50	1.50	1.50	1.50	
SENSOR (X,Y,Z)	MSX	-	6.4",6.7",5.7"	-	-	-	-	-	-	
ELECTRONICS	MSX	-	6.5",5.6",2.1"	-	-	-	-	-	-	
TORQUER #1	MSX	5.50	1.3" Dia.,33.3" L	0.10	15.00	0.00	0.00	0.00	0.00	
TORQUER #2	MSX	5.50	1.3" Dia.,33.3" L	0.10	15.00	0.00	0.00	0.00	0.00	
TORQUER #3	MSX	5.50	1.3" Dia.,33.3" L	0.10	15.00	0.00	0.00	0.00	0.00	
DSAD SYSTEM	MSX	-	see below	1.00	1.00	1.00	1.00	1.00	1.00	
SENSOR #1	MSX	0.60	3.2",3.2",0.8"	-	-	-	-	-	-	
SENSOR #2	MSX	0.60	3.2",3.2",0.8"	-	-	-	-	-	-	
SENSOR #3	MSX	0.60	3.2",3.2",0.8"	-	-	-	-	-	-	
SENSOR #4	MSX	0.60	3.2",3.2",0.8"	-	-	-	-	-	-	
SENSOR #5	MSX	0.60	3.2",3.2",0.8"	-	-	-	-	-	-	
ELECTRONICS	MSX	1.00	5.2",4.5",2.2"	-	-	-	-	-	-	
ATTITUDE & TRACKING PROCESSOR #1	NEW	30.00		75.00	75.00	75.00	75.00	75.00	75.00	
ATTITUDE & TRACKING PROCESSOR #2	NEW	30.00		75.00	75.00	75.00	75.00	75.00	75.00	
SUBTOTAL		319.65		324.00	386.70	323.70	341.70	323.70	341.70	
COMMAND & DATA HANDLING										
1553 CMD/TLM PROCESSORS #1,#2	NEW	16.00	9",10",8"	15.00	15.00	15.00	15.00	15.00	15.00	
DATA RECORDER #1										
DATA RECORDER #2										
KG-TEMPEST #1	MSX	12.00	4.3",10.5",8.3"	0.03	0.03	8.00	8.00	8.00	8.00	
KG-TEMPEST #2	MSX	12.00	4.3",10.5",8.3"	0.03	0.03	8.00	8.00	8.00	8.00	
DATA FORMATTER #1	NEW									
DATA FORMATTER #2	NEW									
SPACECRAFT POWER SWITCHING #1	MSX	16.00	7",7",8"	0.50	0.50	0.50	0.50	0.50	0.50	
SPACECRAFT POWER SWITCHING #2	MSX	16.00	7",7",8"	0.50	0.50	0.50	0.50	0.50	0.50	
SUBTOTAL		72.00		16.06	16.06	32.00	32.00	32.00	32.00	

12/4/91

COMPONENT	HERITAGE	MASS (LBS)	SIZE (INCHES)	BUDGET (W)						LAST UPDATE
				PARK MODE		ATP MODE		DUMP DATA		
				AVG.	PEAK	AVG.	PEAK	AVG.	PEAK	
POWER										
PLUS Y SOLAR ARRAY										
MINUS Y SOLAR ARRAY										
SHUNT ELECTRONICS/SHUNTS	MSX	32.00		10.00	10.00	10.00	10.00	10.00	10.00	
BUS PROTECTION BOX	NEW	5.00	9", 9", 6"	0.00	0.00	0.00	0.00	0.00	0.00	
BATTERY CHARGE REGULATOR	MSX	26.00	20",10",7"	11.00	11.00	11.00	11.00	11.00	11.00	
BATTERY	NEW	166.00	32",24",12"	0.00	0.00	0.00	0.00	0.00	0.00	
SUBTOTAL		37.00		21.00	21.00	21.00	21.00	21.00	21.00	
RF COMMUNICATIONS										
TRANSPONDER #1 -CMD/HOUSEKEEPING	MSX	8.00	5.8",5.2",5.5"	-	-	-	-	-	-	
NB TRANSMITTER #1	MSX	-	-	6.50	26.00	6.50	26.00	6.50	26.00	
CMD RECEIVER #1	MSX	-	-	4.80	4.80	4.80	4.80	4.80	4.80	
DIPLEXER #1	MSX	0.90	7.8",2.0",2.6"	0.00	0.00	0.00	0.00	0.00	0.00	
TRANSPONDER #2 -CMD/HOUSEKEEPING	MSX	8.00	5.8",5.2",5.5"	-	-	-	-	-	-	
NB TRANSMITTER #2	MSX	-	-	0.00	0.00	0.00	0.00	0.00	0.00	
CMD RECEIVER #2	MSX	-	-	4.80	4.80	4.80	4.80	4.80	4.80	
DIPLEXER #2	MSX	0.90	7.8",2.0",2.6"	0.00	0.00	0.00	0.00	0.00	0.00	
SCIENCE (XB) TRANSMITTER #1	MSX	5.00	7.5",4.4",3.0"	0.00	0.00	0.00	0.00	3.80	37.50	
SCIENCE (XB) TRANSMITTER #2	MSX	5.00	7.5",4.4",3.0"	0.00	0.00	0.00	0.00	0.00	0.00	
ANTENNA, S/B TX/RX (6)	MSX	8.00	0.5" Dia.,4" L	0.00	0.00	0.00	0.00	0.00	0.00	
ANTENNA, XB TX #1	MSX	2.50	8" Dia.	0.00	0.00	0.00	0.00	0.00	0.00	
ANTENNA, XB TX #2	MSX	2.50	8" Dia.	0.00	0.00	0.00	0.00	0.00	0.00	
ULTRA STABLE OSCILLATOR / DISTR. AMP #1	NEW	5.00	6.5",4.0",6.0"	4.00	4.00	4.00	4.00	4.00	4.00	
ULTRA STABLE OSCILLATOR / DISTR. AMP #2	NEW	5.00	6.5",4.0",6.0"	0.00	0.00	0.00	0.00	0.00	0.00	
COAX CABLES	NEW	8.00	-							
SUBTOTAL		58.80		20.10	39.60	20.10	39.60	23.90	77.10	
THERMAL										
OPERATIONAL HEATERS	NEW									
SURVIVAL HEATERS	NEW									
BLANKETS	NEW									
HEAT PIPES	NEW									
SURFACE COATINGS	NEW									
SUBTOTAL		0.00		0.00	0.00	0.00	0.00	0.00	0.00	
STRUCTURE										
STRUCTURE	MSX									
SPACECRAFT HARNESS	NEW									
FASTENERS	NEW									
SUBTOTAL		0.00		0.00	0.00	0.00	0.00	0.00	0.00	
SUBTOTAL		893.45		444.16	526.36	663.80	721.30	400.60	471.80	
CONTINGENCY (15%)		134.0175		66.624	78.954	99.57	108.195	60.09	70.77	
TOTAL WITH 15% CONTINGENCY		1027.47		510.78	605.31	763.37	829.50	460.69	542.57	

12/4/91

APPENDIX 7

BALL ATP Space Experiment

Ball Aerospace Systems Group

ATP Space Experiment
and
Balloon Options

Briefing for the
AFPL
by Ball Aerospace
December 6, 1991

Briefing Outline

Introduction

Mission Requirements

ATP Experiment System Overview

Payload Element

Satellite Configuration

Balloon Option Overview

Mission Operations Summary

Issues/Approaches

Summary

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Mission Requirements

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Key SDIO Issues Are Addressed

I.	Coarse Pointing/Target ACQ	√
II.	Target Track/Target ID	√
III.	Passive Track Handover	√
IV.	Passive Intermediate Track	√
V.	Plume-to-Hardbody Handover	√
VI.	Illuminator Point-Ahead/Active Track Handover	√
VII.	Hardbody Discrimination/Active Fine Track/Aimpoint Selection	√
VIII.	Precision Point Ahead/Aimpoint Designation	√
IX.	Precision Beam Pointing at Rate	√
X.	Autonomous Sequencing	√
XVII.	General Plume Phenomenology	√
XVIII.	General Background Clutter	√

System Performance Requirements

Acquire Boosting Targets Against Earth Background

Pasively Track Boosting Targets Against Earth Background

Handover to Actively Illuminated Precision Track with Minimal Target Enhancement

Lay Marker Beam on Target With Total Pointing Accuracy

-Bias	_____	Meters
-Drift (< 3Hz)	_____	RE , RMS (1 Sigma)
-Jitter (> 3Hz	_____	rad, RMS (1 Sigma)

Target Range @ Acquisition:	> 1,000 km
Target Angle rate, max:	< 35 mrad/sec
Target LOS rate, Max	< 1 mrad/sec^2
Target Position Knowledge	< 1,000 m/axis, 1 Sigma
Satellite Position Knowledge	< 1,000 m/axis, 1 Sigma
Mission Life, Months	7; 12 Goal
Mission Orbit	450 km @ 40 Deg

ALTAIR Values Used for Key Parameters

- Plume irradiance
- Earth backgrounds
- Engagement scenarios

ATP Experiment System Overview

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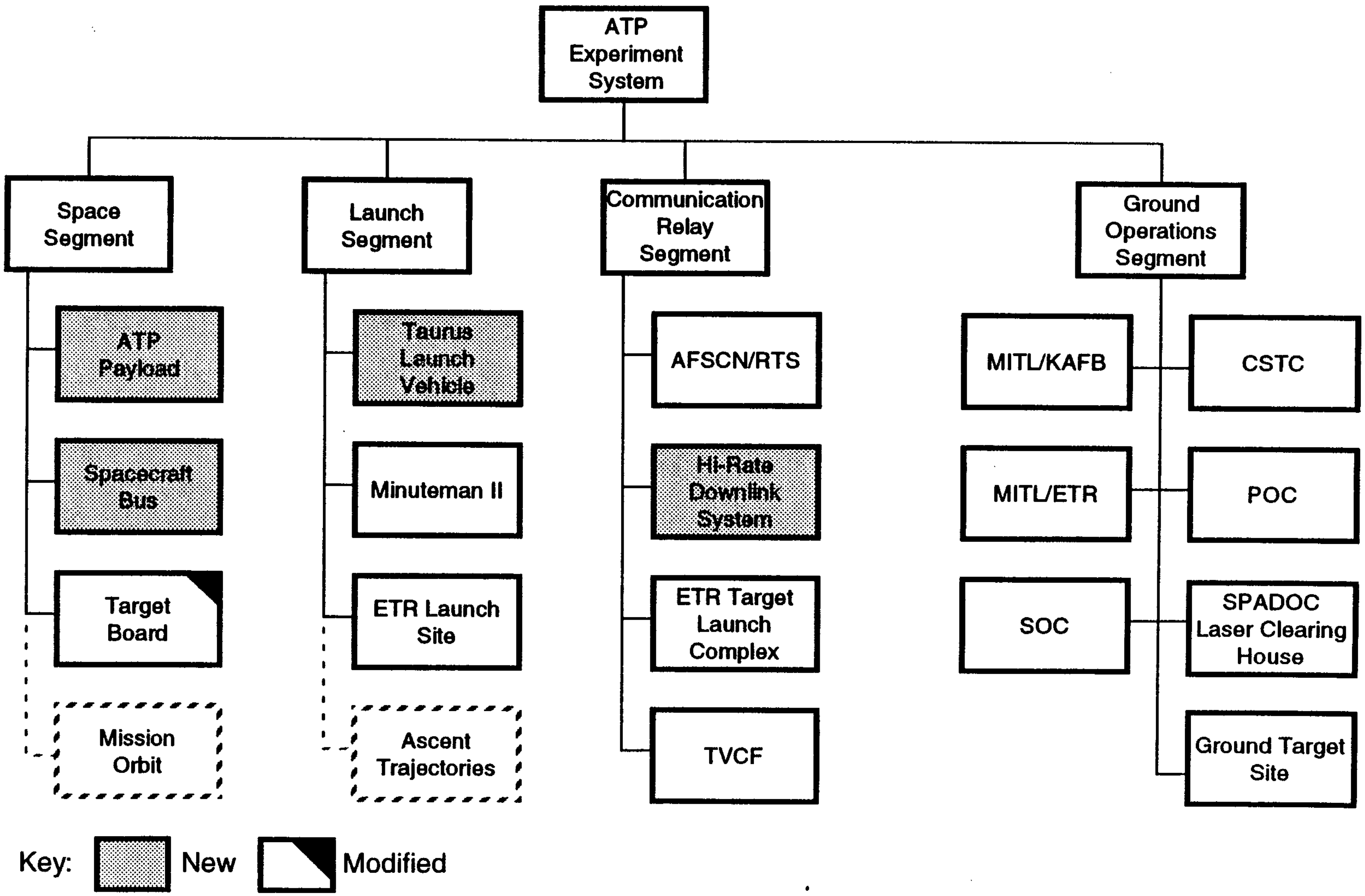
System Hierarchy Diagram Identifies ATP Elements



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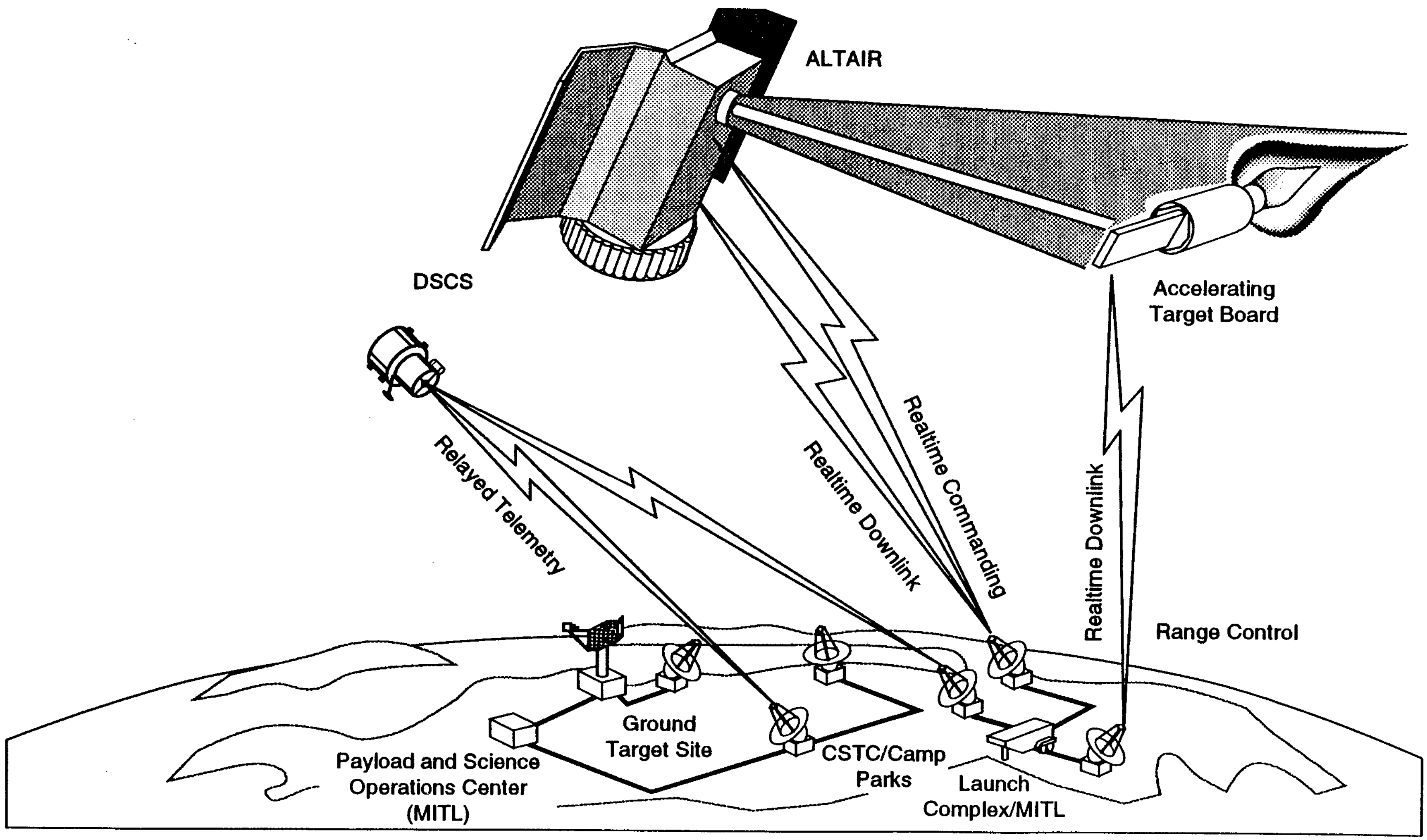
System Schematic Shows ATP Use of Existing USAF Assets



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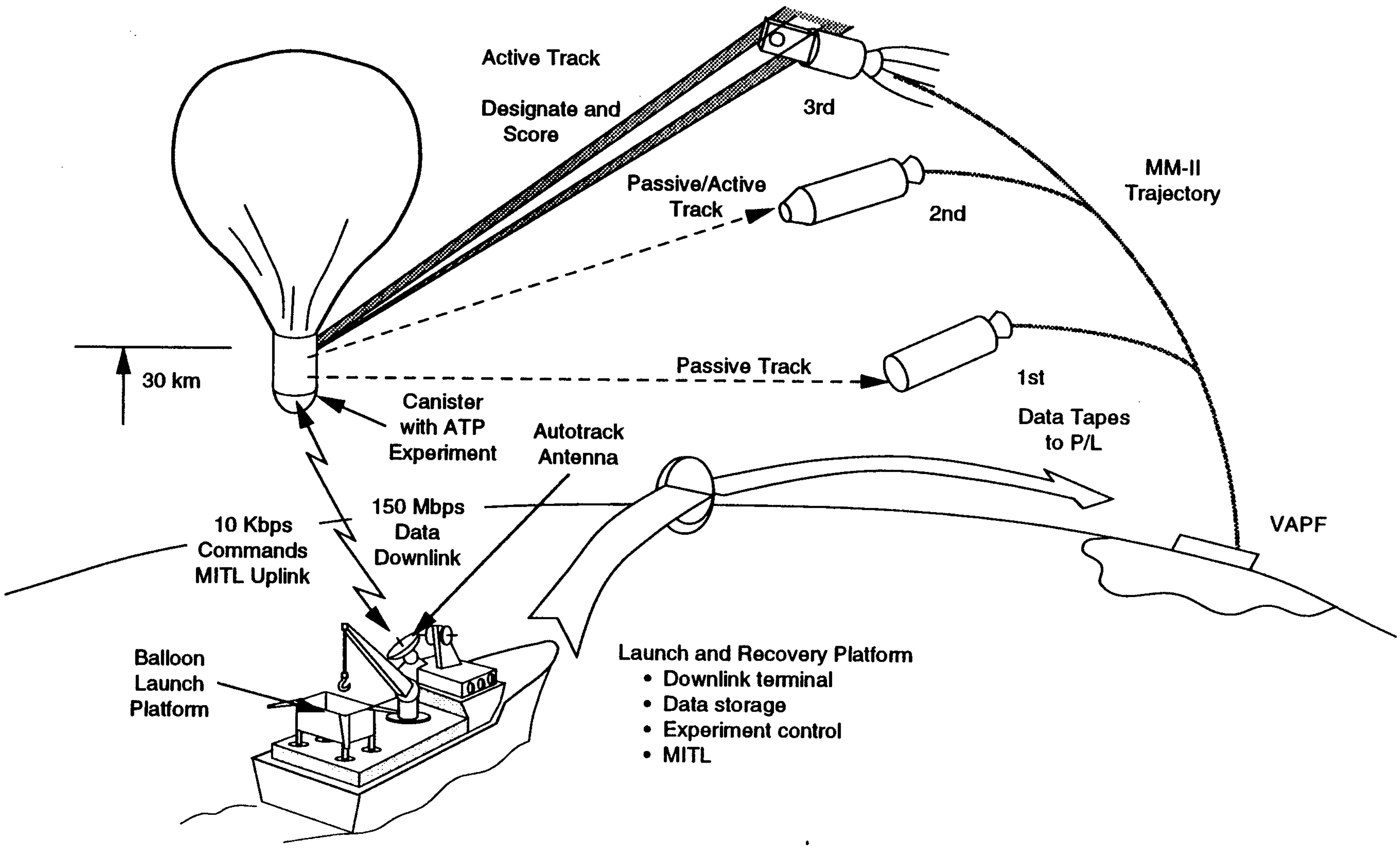
Balloon Experiment Implementation Is Simple, Flexible



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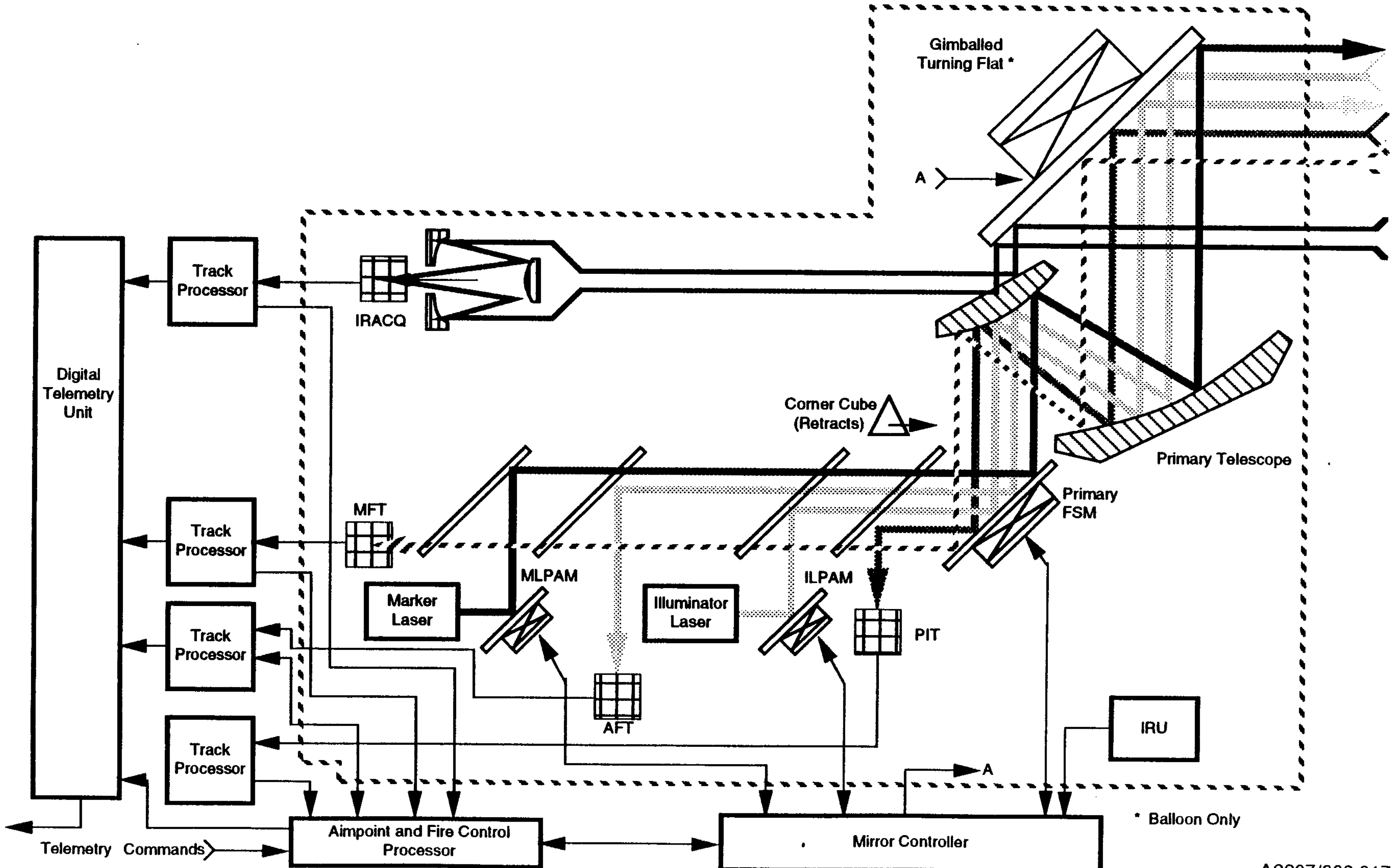
Functional Diagram Shows Use of Existing Designs, Reduced Complexity



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X-Band Communication Links Are Undemanding

LINK PARAMETER	UNITS	BALLOON EXPERIMENT (30 km altitude)	SPACE EXPERIMENT (450 km altitude)	COMMENTS
Transmitter Power	dBm	30.0	30.0	1.0 watt RF output
Misc. Passive Loss	dB	-1.5	-1.5	Coax, filters, etc
S/C Antenna Gain	dBi	8.0	25.7	Helix/0.3 m square microstrip array
Free Space Loss	dB	-147.5	-178.7	Calculated at max range
Pol/Pointing Loss	dB	-1.0	-1.0	Typical
Atmospheric Loss	dB	-0.2	-0.2	Typical
G/S Antenna Gain	dBi	42.3	55.4	2 m/9 m dish
Received Sig.Power	dBm	-69.8	-70.3	Sum all of the above
System Noise Temp.	K	897.0	897.0	Rx NF=5 dB, clear sky temp=270 K
	dB-K	29.5	29.5	10 *log(System Noise Temp)
Boltzmann constant	dBm/K/Hz	-198.6	-198.6	Constant
Received Power/kT	dBm-Hz	99.2	98.7	Rx. Power - Temp - Boltzmann
Data Rate	Mbps	150.0	150.0	
	dB-Hz	81.8	81.8	
Signal/Noise	dB	17.5	17.0	Rx. Power/kT - R
Required S/N	dB	10.8	10.8	Bit error rate = 10e-6
Demod. Impl. Loss	dB	-2.0	-2.0	Typical
Margin	dB	4.7	6.2	

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Various Payload Options Are Possible

Growth →

Payload Components	High Altitude Balloon Experiment		Low Cost Space Experiment
	Commercial Derived	Space Qualifiable	Space Qualified
Lasers	X	X	X
Detectors	X	X	X
Optics	X	X	X
Telescope	X	X	X
Steering flat	X	X	X
FSMs	X	X	X
Power supplies	X	X	X
C&DH hardware	X	X	X
	Performance and functions are essentially equivalent		

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Payload Element

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ATP Experiment Payload Definition and Analysis Objectives

- Develop laser link budgets - active and marker
- Develop passive, active, and marker sensor models
- Use commercial hardware assuming balloon payload
- Achieve high SNRs with realistic design

All Objectives Achieved

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Payload Performance Requirements

Acquire boosting targets against earth background

- $P_{DET} > 99\%$, $P_{FA} < 1\%$, t_{DET} , 1 sec, Range

Passively track boosting targets

- NEA < 10 μrad RMS, sample rate ≥ 30 fps

Handover to active illumination with minimal enhancement

- $t_{HO} < 10$ pulses, with per pulse $P_{HO} > 90\%$, $P_{FA} < 10\%$

Maximize pointing accuracy

- Bias ______ meters
- Drift (≤ 3 Hz) ______ RE, RMS (1 Sigma)
- Jitter (> 3 Hz) ______ rad, RMS (1 Sigma)

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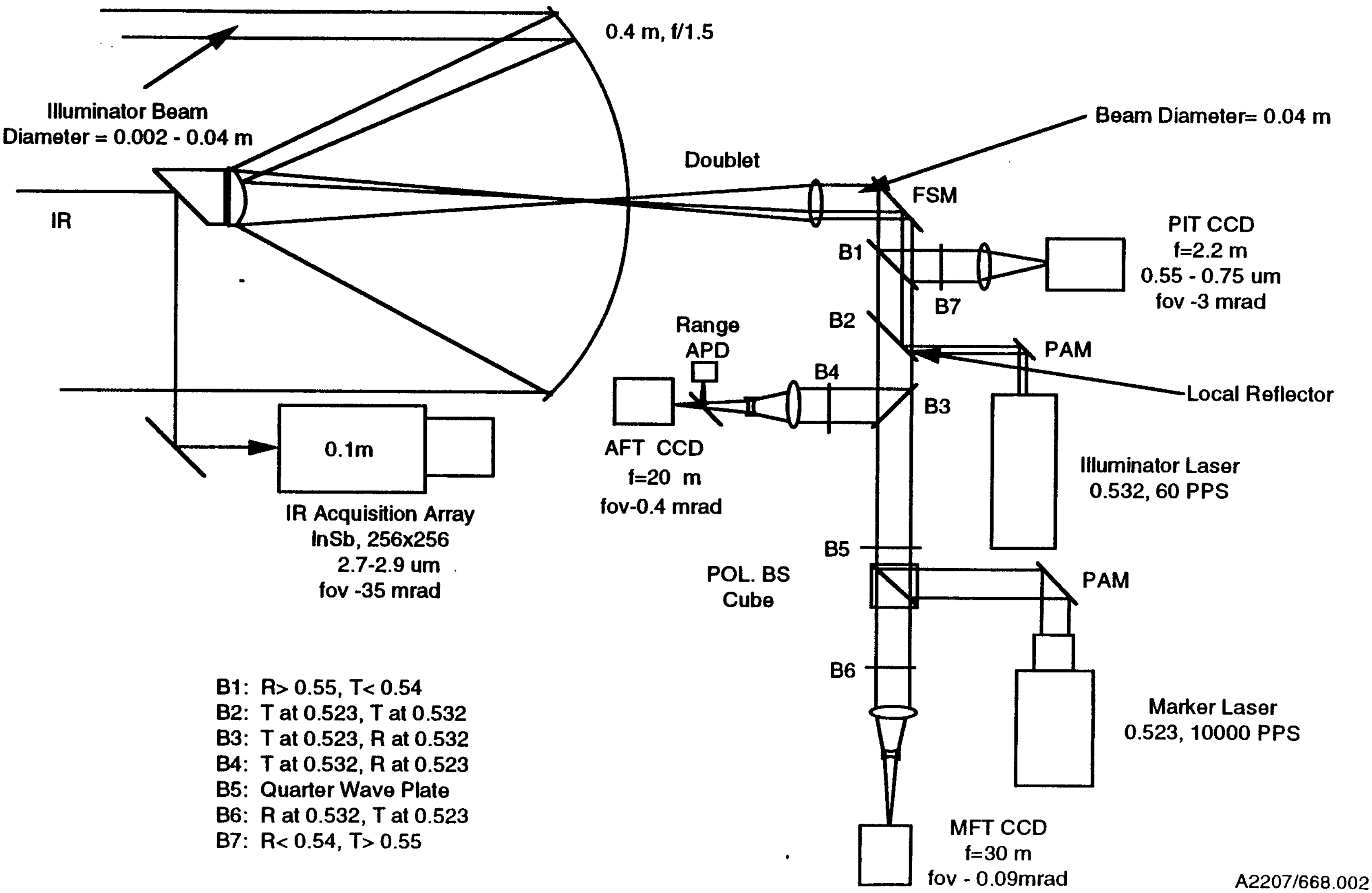
ALTAIR Optical Layout



B1: R> 0.55, T< 0.54
B2: T at 0.523, T at 0.532
B3: T at 0.523, R at 0.532
B4: T at 0.532, R at 0.523
B5: Quarter Wave Plate
B6: R at 0.532, T at 0.523
B7: R< 0.54, T> 0.55

A2207/668.002

Laser Link Budget Developed Using Mathcad

Assumptions

- Telescope
  - 0.4 m aperture
  - t ~0.7 for laser out
  - t ~0.5 to detector

- Target board
  - 10% - 20% reflectivity (10% illuminator, 20% marker)
  - Corner cube effective diameter ~0.5 mm
  - Board length - 1.5 m
  - Board width - 0.9 m

- Illuminator laser
  - Diode pumped, Q switched, frequency doubled Nd:YAG laser from McDonnell Douglas
  - 120 mJ/pulse, 60 Hz rep rate ~10 nsec pulse width
  - 50 μrad divergence

- Marker laser
  - Diode pumped, Q switched frequency doubled Nd:YLF laser from Lightwave Inc.
  - 80 mW average power, 10 KHz rep rate, ~5 nsec pulse width
  - 2.2 μrad divergence

Illuminator Laser Is An Existing Diode Pumped Design

- Supplier is McDonnell Douglas
- Design is similar to the MDESC breadboard
- Spaceflight qualified
- Frequency doubled Nd:YAG system

Specifications:

- 120 mJ/Pulse at 532 nm
- 9.5 ns pulse width Q-switched
- 30 Hz PRF - 60 Hz possible
- 55 amp, 310 μs diode pump pulse
- Laser head and power supply weight - 20 kg
- Power consumption:
  - ~100 watts at 30 Hz
  - ~200 watts at 60 Hz
- Laser -head dimensions (l x w x h)
  7 in. x 3 in. x 1 in.
- Beam quality - 95 percent of TEM_{00} Gaussian

Marker Laser Is An Existing Diode Pumped Design

- Supplier is Lightwave Inc.,
- Design is similar to a spaceflight-qualified system supplied to NASA
- Diode pumped frequency doubled Nd:YLF

Specifications:

- 80 mW average power at 523 nm
- 10 KHz PRF Q-switched
- 5 ns pulse width
- 35 W power consumption
- Laser head dimensions (l x w x h)
  4 in. x 4 in. x 4 in.
- Power supply board - 10 in. square
- TEM_{00} Gaussian beam quality

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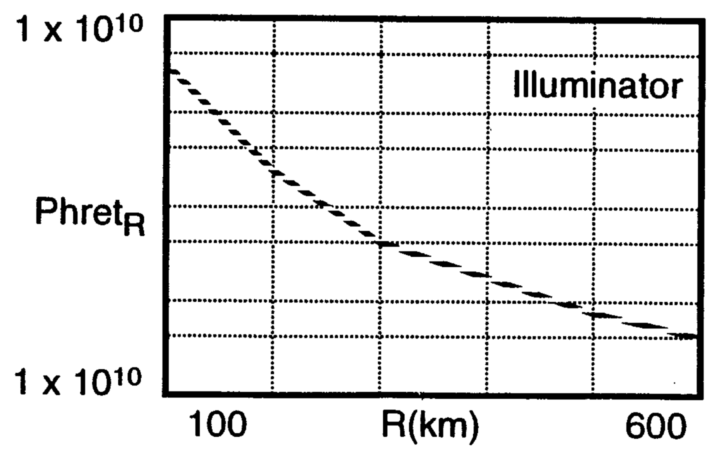
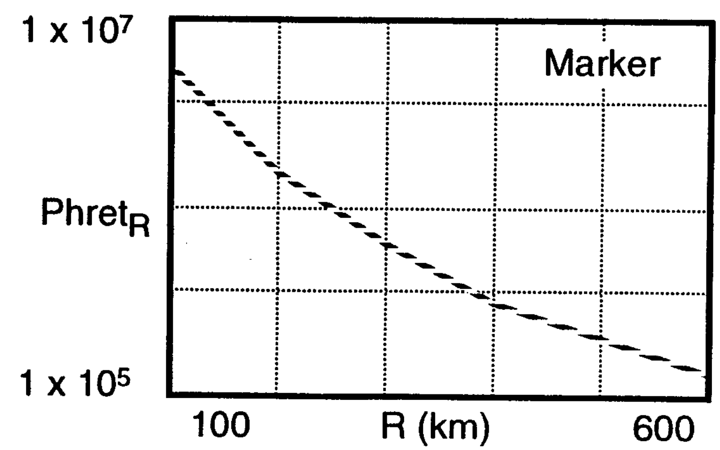
Laser Link Budget Summary
R = 300 km

	Illuminator	Marker
Beam size at target board	15 m	0.66 m
Power at target board	2×10^5 W/pulse	1×10^3 W/pulse
Power returned into transmit aperture	4×10^{-3} W/pulse or 5×10^7 photons/pulse	4×10^{-5} W/pulse or 6.7×10^5 photons/pulse
	Rb = 0.1	Rb = 0.2





A2207/668.005

Sensor Performance Determined Using Existing CCD Model

- Passive intermediate track
  Active fine track
  Marker fine track

  All sensors are frame transfer CCD using IMAGE/MEMORY zones

- CCD selected because
  - Small IFOV provides high SNR
  - Large FOV for acquisition with smaller block for track
  - Video and acquisition/track position information available
  - Centroid and correlation track algorithms exist and proven
  - Leaking signal from plume on AFT and MFT separable from laser signal
  - Commercially available, nothing special
  - Experienced personnel at BASG

- Background is LOWTRAN 7, mid-latitude summer, noon

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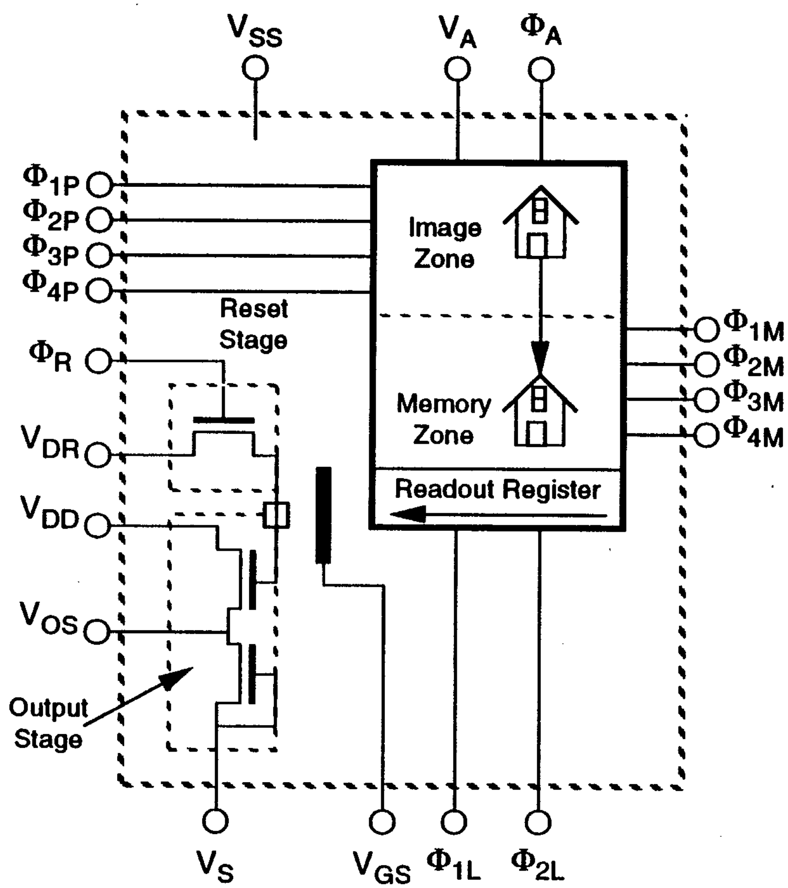
TH 7866 Area Array CCD Image Sensor



- 488 x 500 pixels with antiblooming
- Compatible with EIA RS170 TV standard
- 2/3 in. optics compatible image format (11 mm diagonal)
- Frame transfer organization (dynamic range: 4000/1)
- Optimized resolution and responsivity in the 400-1100 nm spectrum (visible and near infrared)
- Antiblooming efficiency: E_{SAT} x 1000
- Sensitivity: 30 dB S/N at 40 milli-lux
- High contrast: 80% modulation at 412 TV lines
- Low aliasing (moiré)
- No negative bias needed
- Exposure time reduction capability (iris effect): 10 times minimum
- Minimum field readout time: 10 ms

A2207/668.009

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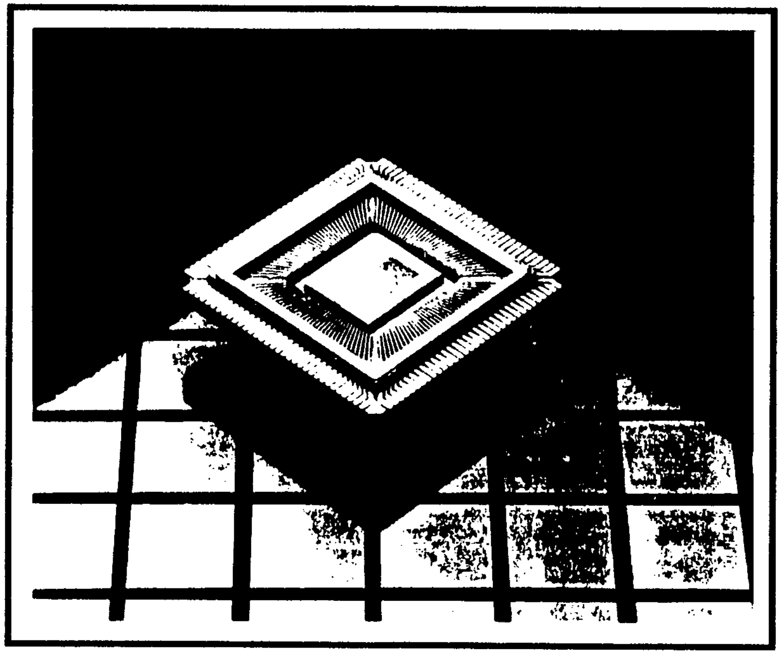
TK064PF CCD Image Sensor



- 64 x 64 pixels
- Frame transfer organization
- Parallel readout
- >5000 frame/sec readout rate
- Dynamic range > 2,000:1

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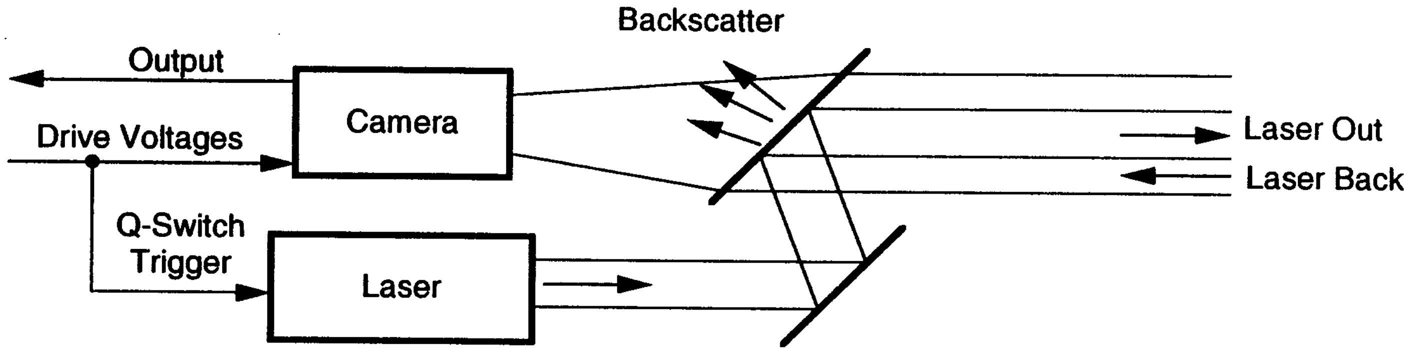
Laser/Sensor Mode of Operation Prevents Backscatter Influence



- Active fine track
  - TH7863 is 50 Hz device (optional* TH7866 is 60 Hz). Both have 0.5 msec image to memory zone transfer "dead" time
  - Laser pulse (~ 10 nsec width) is positioned near beginning of transfer period. To maintain video mode, laser pulse should be triggered by CCD timing
  - Soonest reflected return pulse from T. board occurs 2R/C or ~0.6 msec at 100 km

- Marker laser fine track
  - Tektronix TK064PF has > 5,000 frames/sec readout and > 10,000 frames/sec backclearing (i.e. 100 μsec)
  - Similar to above the marker laser (10,000 pps with 5 nsec width) can be triggered to pulse during backclearing interval. The array also has a pulse controlled reset gate

* Antiblooming capability of the TH7866 allows for controllable reset gate i.e. triggering to laser pulse if necessary

Sensor Performance Summary

	PIT	AFT	MFT
Detector	Thompson TH7866	Thompson TH7863*	Tektronix TK064PF
λc (µm)	0.65	0.532	0.523
$\Delta\lambda$ (µm)	0.2	0.002	0.002
Focal length (mm)	2200	20000	30000
Pixels W *	550	384	64
Pixels H*	244	288	64
IFOV W (µrad)	5.45	1.15	1.4
IFOV H (µrad)	12.27	1.15	1.4
FOV W (µrad)	3000	441	89
FOV H (µrad)	2994	331	89
Image Size (pixels)	50	5	3
Pixel interpolation	0.2	0.2	0.2
Acquisition integration time (sec)	0.0161	0.195	0.005
Acquisition update time	0.0166	0.20	0.0053
Acquisition SNR	389	2610	1190
Track integration time (sec)	0.0161	0.0195	0.002
Track update time	0.0166	0.020	0.0023
Track SNR	389	1430	590

* TH7866 may be used here since it has antiblooming and electronic shutter capability

Sensor Performance Summary Conclusions

- High SNRs provide design options/trades and room for "what abouts", for example:
  - Illuminator laser divergence (50 μrad) may be increased
  - Target board reflectivity, aspect ratio, size may be reduced
  - Illuminator laser divergence may be increased
  - Non TEM_{00} mode operation can be factored in
  - Real world electronics degradation can be considered
  - Processing can be considered

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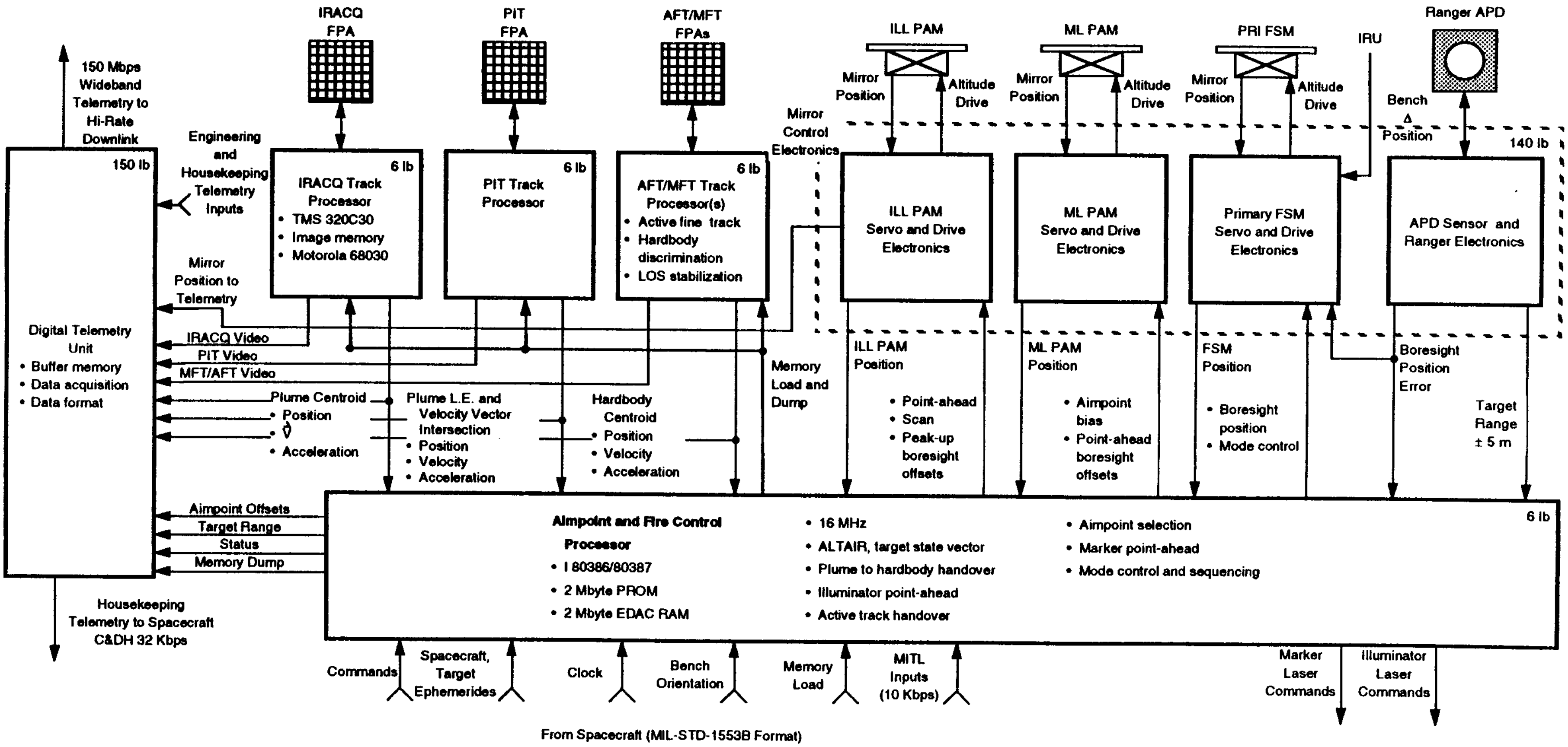
Control Approach Is Based on RME and Ultraseek Designs



A2207/668.003

System/Sensor Performance Conclusions

- Design margin exists, allowing for trade space

- Need information on
  - Background
  - Plume size
  - Plume characteristics at λ

- Mission goals appear very achievable with a cost controllable approach

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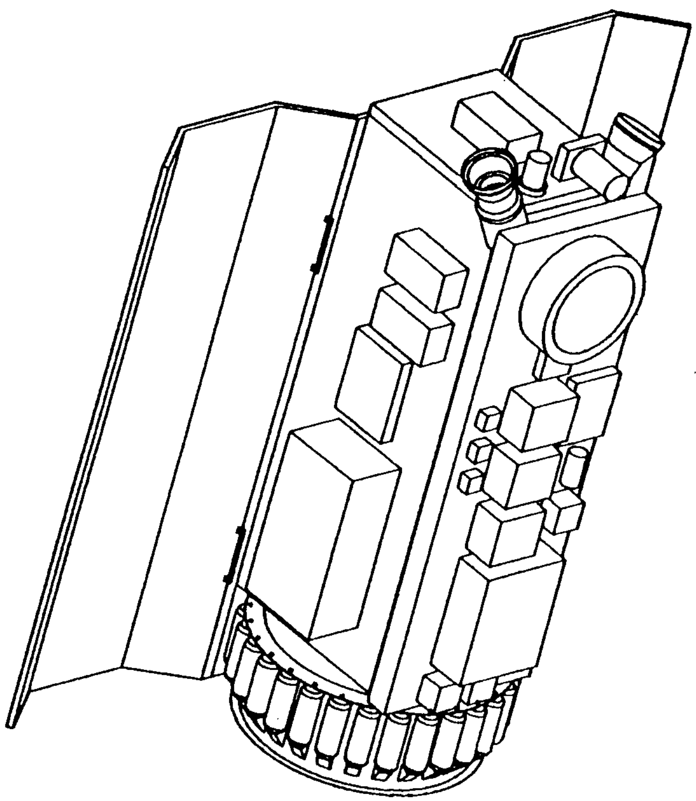
Satellite Configuration

Functional Requirements - Spacecraft



REQUIREMENTS

- Provide data uplink/downlink
- Support data storage and playback
- Furnish power to all satellite systems
- Provide structural support for spacecraft and payload systems
- Establish and maintain safe hardware thermal environment
- Accurately determine attitude and point precisely at rate
- Man in the loop

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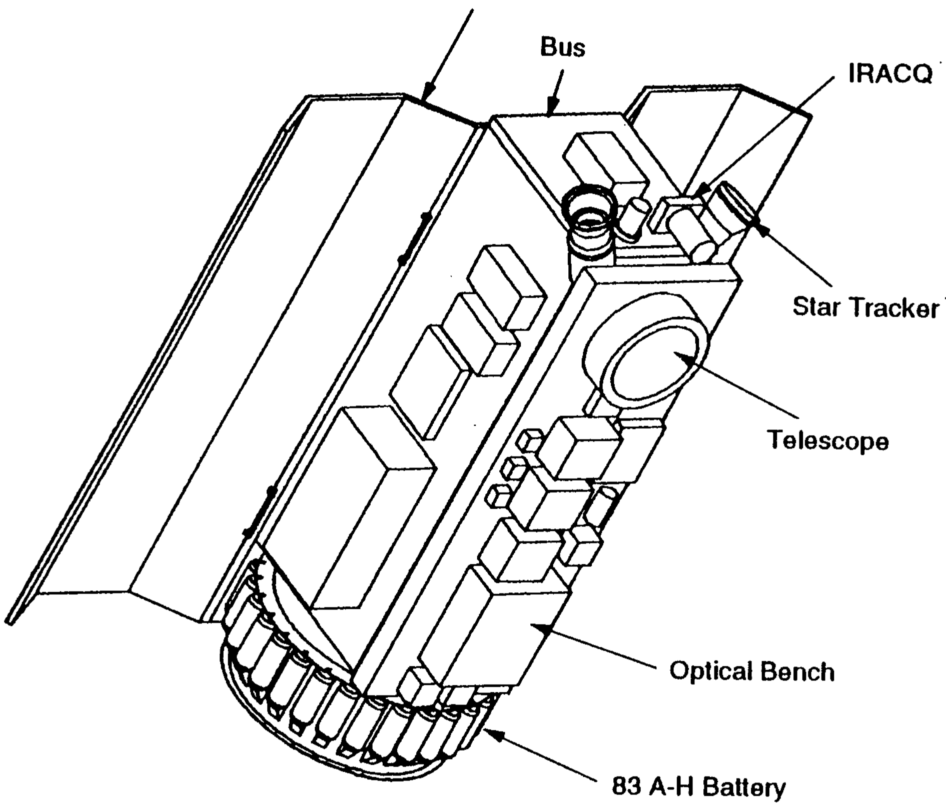
General Arrangement Shows Tight Payload/Spacecraft Integration



- 110 (l) x 100 (w) x 45 (h) in.
- 2087 lbm at separation

- Configuration fits Taurus shroud , adapter
  - Maximum access to payload, components
  - Accommodates ALTAIR-II payload (approximately 2.0 m^3)
- Design life 12 months; selected redundant subsystems
- 3-axis attitude control
  - Internally pointed
  - 1 mrad accuracy/knowledge (3 sigma)
- 28 Vdc electrical power
  - 1,000 W-hr/engagement available to payload
  - 6 hour re-cycle
- S-band SGLS compatible communications
  - 256 Kbps telemetry downlink
  - 2 Kbps command uplink
- X-band 150 Mbps downlink for payload
- Passive thermal control

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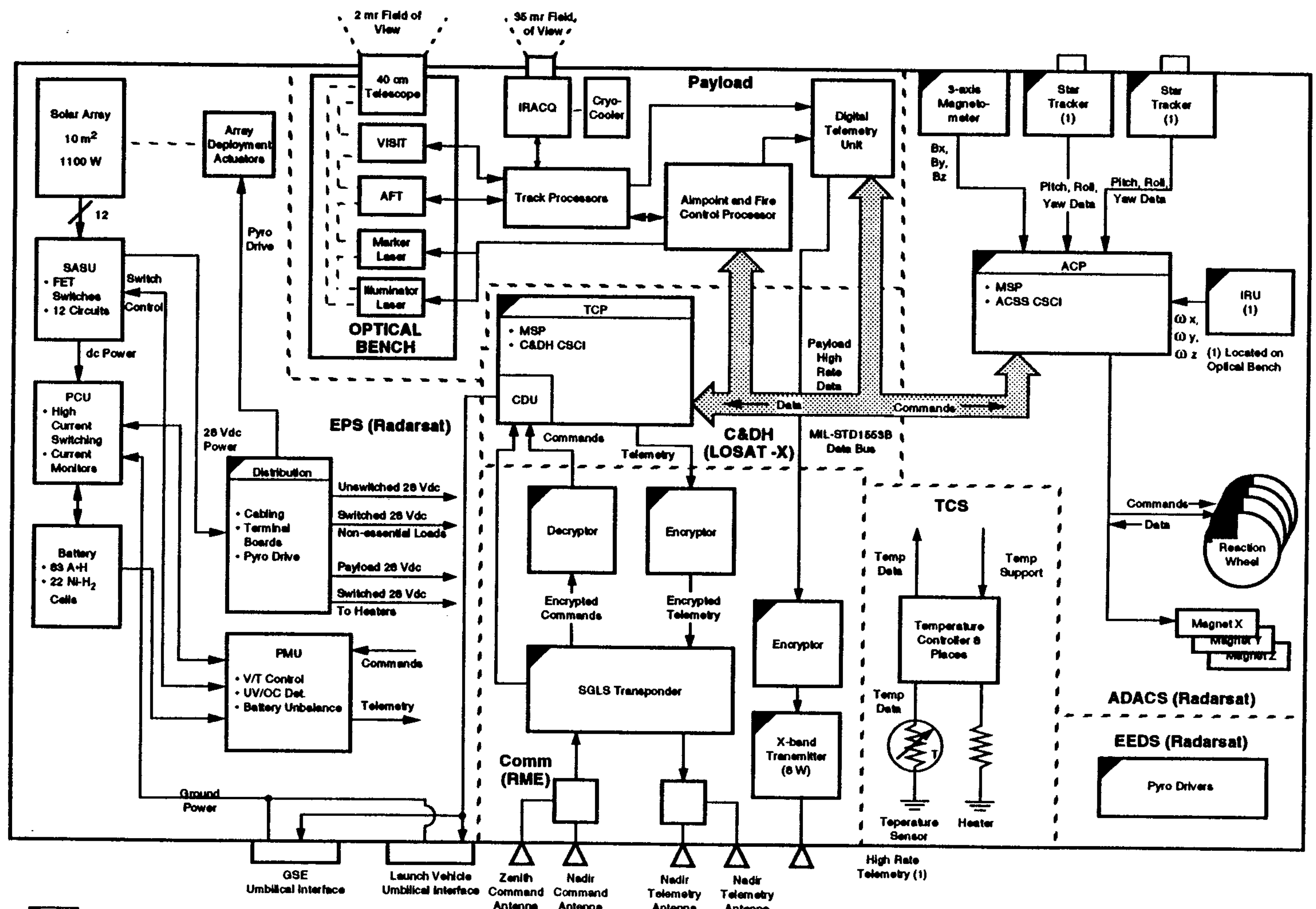
Preliminary Satellite Functional Shows Heritage of Key Subsystems



Note: [box] Denotes Redundant Unit

A2207/668.032

Mass Properties Are Compatible with Taurus Throw Weight to Mission Orbit

SUBSYSTEM/ITEM	ESTIMATED MASS (LBM)	HERITAGE/DERIVATION
STRUCTURE/ MECHANICAL	293	RME; LIME
ELECTRICAL POWER & DISTRIBUTION	323	RME; RADARSAT
COMMAND, CONTROL & DATA HANDLING	22	LOWSAT-X AND IR&D
ATTITUDE DETERMINATION AND CONTROL	265	RADARSAT; MSX
COMMUNICATIONS	66	RME
THERMAL CONTROL	27	RADARSAT; VARIOUS
CONTINGENCY @ 10%	97	
TOTAL, ALTAIR-II S/C BUS	1093	
ALTAIR-II PAYLOAD	994	INCLUDES 20% CONTINGECY
TOTAL, ALTAIR-II SATELLITE	2087	
TAURUS PAYLOAD ATTACH FITTING (PAF)	25	
TOTAL,ALTAIR-II MISSION MASS ON LV	2112	
TAURUS CAPABILITY TO MISSION ORBIT	2500	
AVAILABLE MARGIN ON LV CAPABILITY	388	18 PERCENT

Summary and Conclusion

- A traceable, scaleable ATP Experiment can be supported By a Taurus Launched S/C or Balloon
- Balloon Basing allows phased approach to realization of ATP Experiment Goals
- The ATP Payload is applicable to either approach with minimum modifications
- This commonality provides a straightforward path for migration of the balloon Payload to a spacecraft
- Balloon based experiments can be conducted over a Wide cost range

Appendix 3.3-A

DRAFT, 7 Oct 1991
Photon Research Associates, Inc.

MEMORANDUM FOR DISTRIBUTION

SUBJECT: TNS UV/Visible Program Plan

1. Introduction/Background

There has been considerable speculation in the SDI community concerning the utility of UV and visible sensors for missile defense applications. Much of the interest is based on limited data from SDI programs such as Delta 180, Malabar observations, Probe, Delta Star and UVPI which indicate that plume signatures are generally brighter than anticipated and more compact than their IR counterpart. In addition, compared with IR sensors, UV and visible sensors offer the advantages of mature technology, small size, and low weight and power.

Attempts to credibly explore UV/VIS concepts have been frustrated by the lack of adequate and consistent signatures data to assess the range and robustness of signatures expected from targets of interest. In particular, plume signature data has been tantalizing, but not fully understood. For this reason, a review of the status of UV and visible phenomenology and sensor technology was held at ANSER (Arlington, VA) on 10 September 1991 with a subsequent session on the 11th to outline a plan of action. This memorandum summarizes the state-of-knowledge and outlines the recommended plan developed on the 11th.

1.1 Potential Applications

UV and visible sensors have been postulated as potential adjuncts to existing IR system concepts for functions ranging from second source confirmation of launch, plume-hardbody handover for interceptors, and midcourse discrimination. The Appendix describes some of these potential applications in greater detail.

1.2 Methodology

The working group assessed the current state of knowledge for plumes, backgrounds, hardbodies, and sensor technologies. Some areas are in better shape than others - average background levels are well known, but unknown clutter levels and large variations in liquid plume signatures make meaningful system-level calculations impossible. After some discussion concerning the way to proceed, Col Mill recommended a four-step process to efficiently develop UV/VIS concepts. This process was accepted and consists of:

1. Identification of potential applications of UV/VIS sensors for missile defense functions in both strategic and theater areas.

2. Demonstration of the basic feasibility of UV/VIS sensors, including cost and technology, to perform specific applications.

3. Foster interest and support from the system elements in the development of specific applications.

4. Execute detailed studies or measurements to validate these applications.

Suggestions of potential applications are an ongoing process, but an initial attempt has been developed and is contained in the Appendix. Of present concern to the working session was identification grouping the effort needed to demonstrate the basic feasibility of the proposed applications. A factor of two to five uncertainty in signature prediction was considered reasonable for meaningful calculation and trade-off analyses. Our understanding of some phenomena are much better than others. Therefore, the working session focused on the actions required in each phenomenology/technology area to reduce uncertainties to acceptable levels. The efforts needed to move beyond demonstration of basic feasibility will likely be driven by the maturity of our understanding in the specific phenomenology/technology areas unique to the application.

2. State of Knowledge

2.1 Preliminaries

The UV and visible spectral area can be subdivided into four adjacent spectral regions. The nominal boundaries are the visible (0.4 - 0.7 um), near UV (0.3 - 0.4 um), solar blind UV (0.2 - 0.3 um) and far UV (0.1 - 0.2 um). The generic visible region is from 0.4 - 0.7 um, but is often extended outside this region (e.g. 0.3 - 0.9 um) depending on the application.

The SBUV and FUV regions are influenced by atmospheric absorption features associated with ozone and molecular oxygen, respectively. The UV region offering the highest potential is the SBUV which exhibits low background radiance levels to spacecraft sensors due to absorption of solar ultraviolet radiation by the earth's ozone layer. In addition to lower average radiance levels, the SBUV is believed to possess small clutter values. The sacrifice for the lower background and clutter levels in the SBUV is that target signature transmission losses are severe below about 30 km altitude and target intensity values are typically less.

In the far UV, the small solar irradiance values coupled with oxygen absorption dominant the background. The background radiance level drops another 3 orders of magnitude over the SBUV, but now target detection is not possible below about 90 km making this region almost useless for satellite surveillance operations. In addition, signatures from plumes and hardbodies are considerably lower in the far UV than in the visible or SBUV.

2.2 Plume Phenomenology

Since its inception, SDI has focused data collection and modeling efforts on the IR signatures of high-altitude metallized-solid and liquid hydrazine propellants which represent the bulk of the strategic threat. Most of the database consists of IR data on aluminized solids and amines. Plume signatures in the UV and visible are generally poorly understood due to the lack of data and the large number of parameters that seem to affect plume signatures, such as propellant type, engine design, altitude, velocity, and angle of attack.

Aluminum-based solid rocket plumes are the best understood propellant systems. Blackbody emission from the hot exhaust particles dominates the visible and remains important in the SBUV. When sunlit, solar scattering from the particles is also an important contributor. The mechanisms and behavior of plumes from aluminized solid propellants in the UV/VIS wavelengths at low and high altitudes are sufficiently well understood that we estimate our level of confidence to be within a factor of 2-5 and the signature to be dependable (robust) even though uncertainties about particle size, non-spherical shapes, and optical properties exist.

Non-aluminized solids, which include composites and double-base compositions represent a small fraction of potential threat propellant types and are not known to be used in upper stages. Our understanding of the signatures of these propellants is poor.

More than half of the Soviet and Chinese tactical and strategic missiles utilize liquid amine propellants. The limited data on amine plumes exhibits wide variations in radiant intensity in both the visible and UV region. Postulated mechanisms for UV/VIS emission include chemiluminescence and collisional excitation with ambient atmospheric species (atomic oxygen), but the specific reaction paths have yet to be identified. The estimated uncertainty in intensity predictions is x100 for high altitude environment and x10 for low altitudes. The robustness of the high altitude plume signature is unknown, but the low altitude signature where afterburning is important is considered rather robust. Most molecular emissions are found in the visible and SBUV with some in the far UV. The large proportion of threat missiles and the lack of understanding of UV/VIS signatures from amines make these propellant systems a high priority.

Hydrocarbon propellants are not used in strategic missiles, but are found in several theater missiles, most notably the SCUD and its variants. Missiles using these propellants represent a significant threat. Hydrocarbon fuels such as kerosene have exhibited UV/visible emission from hot carbon particulate (soot) emission and solar scattering. These emissions may be valuable for tracking and plume-hardbody handover functions. Characteristics for carbon formation are not well understood so the impact of engine design excursions on the signature is difficult to assess. The estimated uncertainty in predicting hydrocarbon plume signatures was x10 for the low altitude regime. Afterburning should produce a robust signature.

Table 1 summarizes the estimated uncertainty in our ability to predict plume intensities in the UV/vis and an indication of the robustness or dependability of the signature. Table 2 indicates the range of signatures expected from solid and liquid propellants based on the existing datasets.

Table 1. Estimated uncertainty (factor) in UV/VIS plume signatures

	Low Alt (<30 km)	High Alt (>30 km)
Aluminized Solid	2-5 (Robust)	2
Non-Al Solid	Unknown	Unknown
Amines Liquid	10 (Robust)	100
Hydrocarbon Liquid	10 (Robust)	Unknown

Table 2. Representative plume signatures (W/sr)

Event		Night SBUV	Night VIS	Day SBUV	Day VIS
Low Altitude	Solid (100 klb)	--	5k-20k	--	10k-40k
	Liquid (100 klb)	--	10-200	--	10-200
High Altitude	Solid (20 klb)	5-20	2k-8k	30-120	4k-16k
	Liquid (20 klb)	5-20	1-60	5-25	1-100
PBV	Liquid	.01-.1	.1-1	.4-1	.1-2

2.3 Backgrounds Phenomenology

Background radiance levels in the visible and SBUV are considered well known due largely to data collected from spacecraft missions. Visible sunlit backgrounds are dominated by scattering and reflection of solar radiation. The solar spectrum peaks near 0.5 um and remains strong throughout the visible, but drops off sharply beginning in the near UV and continuing through the SBUV and FUV regions. The earth atmosphere is transparent and terrestrial material reflectivities are high through the visible to 0.32 um. Significant solar radiation is reflected back into space from the hard earth and cloud formations presenting a bright and cluttered background in the visible and near UV that makes target detection in the daylight impossible for all but the brightest targets.

Associated with the dropping solar spectrum in the UV region are atmospheric absorbers which effect target detection in two ways: 1) solar radiation is absorbed on the way down and back up through the atmosphere, and 2) targets located below or within the absorbing specie(s) yield reduced signatures. Sunlit SBUV background level is about 3 orders of magnitude below the visible levels and the clutter levels are expected to be much smaller than the visible, but a definitive value has not been measured.

Night backgrounds emissions from the airglow and other atmospheric emissions are dim and do not impact target detection, but attenuation of target signatures remains an issue. Table 3 provides nominal background radiance values for various illumination conditions and viewing geometries.

Table 3. Representative Background Radiance ($W/cm^2/sr$)

	SBUV	VIS
Celestial (zodiacal 30-180)	$(2\text{-}20)x10^{13}$	$(2\text{-}20)x10^{12}$
Limb - Day (100 km TH)	$(2\text{-}10)x10^{9}$	$(1\text{-}4)x10^{8}$
Limb - Night (100 km TH)	$(3\text{-}12)x10^{11}$	$(.5\text{-}2)x10^{10}$
Nadir - Day	$(1\text{-}6)x10^{7}$	$(1\text{-}6)x10^{3}$
Nadir - Night	$(3\text{-}12)x10^{11}$	$(4\text{-}8)x10^{11}$

2.4 Hardbody Phenomenology

Hardbody signatures of exoatmospheric targets are directly related to the level of the solar radiation and the target reflectivity in the spectral band of interest. Most space material have high reflectivities in the visible which drop continuously and sharply through the UV region. This combined with the solar spectrum means that most solar illuminated target signatures are brightest in the visible, lower in the near UV, lower still in the SBUV, and lowest in the far UV. For this reason the visible region is generally preferred for hardbody functions. However, hardbodies in exoatmospheric flight are only detectable under illuminated conditions. Estimates of reflective hardbody signatures should be accurate within a factor of 2, which is adequate for initial calculations. Nominal exoatmospheric (reflective) hardbody signatures are presented in Table 4.

Table 4. Representative SW Hardbody Signatures. Radiant intensity given as E(solar)pA/π: where E(solar) is the in-band solar irradiance, p is the reflectivity, A is the projected area. Actual target signatures depend on sun angle, target shape, material properties, and earthshine component.

Target	Projected Area (m^2)	Reflectivity SBUV	Reflectivity VIS	Radiance Intensity (W/sr) SBUV	Radiance Intensity (W/sr) VIS
RV	0.1 - 0.9	0.1	0.2	.06 - 0.5	3.2 - 30
PBV	3 - 12	0.2	0.5	3 - 12	250 - 1000
2nd Stage	10 - 30	0.2	0.5	10 - 30	800 - 2400
Satellite	3 - 40	0.5	0.5	8 - 100	250 - 3300

Atmospheric heating and surface heating of reentry bodies will produce visible and UV radiation. The magnitude, altitude dependence, and robustness of the signatures will be dependent on reentry velocity and angle. By deep reentry, ICBM-class signatures are extremely bright (megawatts in the visible). Due to absorption by ozone, the SBUV is likely a poor region for distant observations.

2.5 Sensor Technology

Current UV/VIS technology which includes optics, filters provide an adequate capability. Development of CCDs is progressing rapidly and some applications may require specific improvements such as enhanced radiation hardening, but these are not of immediate concern.

3. Recommended Program Plan

3.1 Plumes

The inability to predict liquid plume signatures is, by far, the primary phenomenology hampering our ability to make credible systems-level calculations or assess signature robustness. Three areas were identified for priority efforts, based on our degree of uncertainty in prediction and the prevalence of propellant system.

1) Characterize UV/VIS signatures from amine fueled missiles in both the low and high altitude regimes.

2) Characterize hydrocarbon fueled missiles at low altitude to include carbon formation.

3) Characterize the signatures of non-Al solid propellants at low altitudes.

A detailed measurements plan to accomplish this was not developed during the working session. A program plan that identifies specific measurements and priorities should be developed and coordinated between SDIO/TNS and PL/RPL. Capt Tilton (TNS) is the action officer for this effort. Even though aluminized solids were not identified as priority measurements, SDI should not pass on opportunities to collect UV or visible signatures on these propellants to advance our understanding.

3.2 Backgrounds

SBUV clutter is the outstanding issue for backgrounds. It is thought to be low (a few percent), but no data exists to substantiate that claim. It was recommended that existing datasets be examined to identify an upper limit on clutter values. These data sets might include UVPI, Delta Star, and NASA TOMS. In addition, prompt evaluation of the capability of UVPI to characterize SBUV clutter was strongly recommended and if promising, additional measurements performed. Maj Imker (TNS), with assistance from Capt Tilton, is the action officer for this effort.

Existing datasets do not provide measurements at sensitivities or spatial dimensions of proposed SDI sensors. Therefore, it is likely that an experiment may be required in the future to measure clutter at scale lengths on the order of 100 m in the limb.

3.3 Hardbodies

The signature prediction capability of hardbodies is considered to be good. The capability in this area can be substantially increased by including UV/visible reflectance data in the signature codes. It was recommended to extend on-going BRDF chamber measurements of space materials into the UV region to establish a materials database in the UV/vis region. This can be accomplished at a small additional cost to the current program and the data will be immediately useful in hardbody signature codes such as OSC. Mr. Erwin Myrick (TNS) is the action officer for this effort.

Although terminal signature issues were not discussed during these meetings, it seems prudent to review existing datasets and to characterize reentry signatures. As an example, the extensive data collected under the TRAPS programs cover the near UV to near-IR optical region.

3.4 Sensor Technology

Sensor technology was considered mature enough for current SDI applications. Although no technology action is required, it was noted that the IS&T Office funds promising technologies that may impact UV/VIS sensor development. The SDI community should maintain contact and awareness of these activities. LtCol Swenson (TNS) is the appropriate action office for this coordination.

4. Summary

Our understanding of average background radiance levels and exoatmospheric target signatures is adequate for initial estimates of SBUV and visible sensor utility. Although our understanding of aluminized solid plumes is fairly good, that of other propellants is poor. What is needed quickly is a plume signature database from which correlated relationships can be inferred and interpolations or bounding estimates performed. This can serve as an early predictive capability - signature models and codes can follow.

It is imperative that a sustained and focused effort be applied to the development of UV/visible potential. Only then can we move past the realm of postulated application into demonstrated capability and provide concrete support for SDI system elements.

Appendix
Potential Applications of UV and Visible Sensors for Missile Defense Applications

This appendix briefly discusses potential UV/VIS utility for missile defense functions treating strategic or long range missile systems differently from theater (shorter range) systems. Long range systems are distinguished here as possessing multiple stages and occupying some time outside the sensible atmosphere (150 km). These are the nominal ICBM or SLBM systems. Shorter-range systems exhibit short burn times relative to long-range systems and are contained largely within the atmosphere. The distinction between short and long ranges is somewhat arbitrary, but is useful in the following discussions.

Strategic (Long-Range) Applications

1. Booster plume detection.

Booster plume detection and track with visible sensors offer no unique advantages over current IR systems. IR sensors are filtered to atmospheric absorption bands which attenuate some of the cluttered background radiance from the hard earth scene. IR sensors provide day/night capability, but the downlooking cluttered background under solar illumination is significant. BTH visible detection of booster plumes under sunlit conditions is difficult due to the very bright and cluttered background. Detection and tracking of upper stages in the sunlit upper limb seems possible. BTH visible detection under night conditions is possible, but then the system provides only night detection capability.

Plume detection using the SBUV is attractive if the clutter levels are low enough and the SBUV intensity from plumes if found to be bright and robust. The SBUV can provide a day or night capability at the expense of launch detection near 30 km altitude. In addition, the apparent sensitivity of plume signatures to engine and fuel parameters may prove useful for booster typing in conjunction with IR measurements.

Visible or SBUV sensor data may be useful for launch confirmation. UV and visible plume emission mechanisms are different from IR mechanisms permitting essentially independent measurements.

2. Plume-hardbody transition

The plume-to hardbody handover function for an interceptor should be easy in the visible or SBUV if the hardbody is illuminated by either the sun or a bright plume. In addition, there may be some body/atmospheric interactions that produce radiation in the SBUV or visible regions and indicate hardbody location (e.g. bowshock phenomenon).

Current SDI handover concepts utilize multiple IR bands - one for plume tracking and another for hardbody location.

Homing and intercept on thrusting targets relies on a detectable plume signature with understood spatial dimensions or behavior to allow plume-to-hardbody handover. SBUV plumes have been observed to be more compact than their IR counterparts and different emission mechanisms (and spectral regimes) can be associated with core radiation and others to atmospheric interactions. These features may be useful for interceptor operations.

3. Midcourse tracking/discrimination

Midcourse surveillance and tracking during and after PBV dispensing is possible if the target complex is solar illuminated or the PBV plume is bright enough to provide a local source of radiation. Tracking of sunlit targets can be accomplished in the visible against the upper limb or space background. Current programs such as VIP may show that there are sufficient UV/visible emissions from hardbody/atmosphere interactions to make nighttime detection possible.

Discrimination of midcourse targets during deployment from the PBV is possible with visible and SBUV, sensors if PBV plume signature variations can be associated with deployment sequences or the PBV plume illuminates the targets providing a reflected signature that can be exploited for discrimination.

Solar illuminated midcourse targets may potentially be discriminated based on metrics or temporal signatures. Because of the requirement for some illumination source for hardbody detection/discrimination, visible instruments would supplement other (perhaps IR) sensors. The finer angular resolution of UV/vis sensors could also be used to augment IR sensor data (e.g. unravel the CSO problem for LWIR sensors).

4. Track/homing/discrimination in reentry

High velocity reentry objects emit copious visible radiation. Target acquisition and tracking in the visible during deep reentry has been demonstrated and a large database on these signatures exists (e.g. the TRAPS program). However, the emissions coming from reentry bodies is not limited to the body itself but extends to the surrounding plasma which may complicate homing/interception algorithms.

The total intensity, derived metric information, and temporal features of reentering objects may be useful for target discrimination. These observables could augment radar systems and be particularly useful in the ECM environment.

5. Kill Assessment

Interception of fuel-ladened boosters or PBVs should produce bright SBUV and visible signatures that can be used for kill assessment. However it is not clear what advantages are offered over proposed IR systems although the combined IR/visible or IR/SBUV capability may increase confidence in kill assessment capability.

Visible sensors may prove more sensitive than IR for kill assessment determination of PBV/RV debris in a sunlit environment. Visible signatures show strong shape and geometry dependence and may be valuable for fragment/debris filtering. Visible sensor data may be useful in assessing slow-down or signature modulation for kill assessment in the upper reentry environment to complement radar techniques.

<u>Theater (Short Range) Applications</u>

1. Launch Warning

Detection of launch preparations by opposition forces may be aided with UV/vis techniques from either spacecraft or aircraft-borne sensors in one of two ways; standard photointerpretation techniques using broadband visible sensors, or narrowband-filtered visible sensors to detect specific chemical emission features associated with missile operations. This is a very difficult job and will likely require a multispectral approach (visible and IR) to be at all successful.

2. Boost plume detection

Current IR sensors/concepts achieve launch detection at some nominal time after lift-off (typically 10-20 seconds). TMD effectiveness should increase if detection can be made earlier. Visible sensors can see down to the ground providing instant launch detection, but the high daytime visible background limits their use to night time conditions. Narrow-band visible sensor concepts may provide sufficient signal to noise for detection in the day if, for example, they are filtered to minima in the background radiance such as Fraunhofer lines in the solar spectrum.

Broadband visible sensors can provide early launch detection at night for theater operations. This may be significant if opposition forces are limited to hiding missiles during daylight when surveillance sensors (spacecraft, drones, and aircraft) are overhead as was observed during the Gulf War. Night launches are then most likely.

Aircraft and balloon-based visible instruments may be able to provide launch detection in the theater area. Existing data suggests that detection of solid plume emissions at ranges up to 70 km are possible in daylight conditions from aircraft-based sensors.

Tracking of theater boosters during daylight can be done from space platforms in the SBUV if the burn-out altitude is not too high (below about 30km). After plume burn-out theater tracking can be done by IR, visible, radar, and perhaps SBUV sensors. The visible and SBUV signature comes from reflected sun and skyshine.

SBUV sensors situated under the ozone layer (e.g. an A/C platform) could possibly provide early launch detection for theater operations.

3. Discrimination

Theater concepts for discrimination presently rely on radar systems which provide day/night capability. It is unlikely that ground or A/C based visible or SBUV sensors can provide any advantages since they cannot operate at night (no solar illumination) and the day background is too bright and cluttered for discrimination of these small objects.

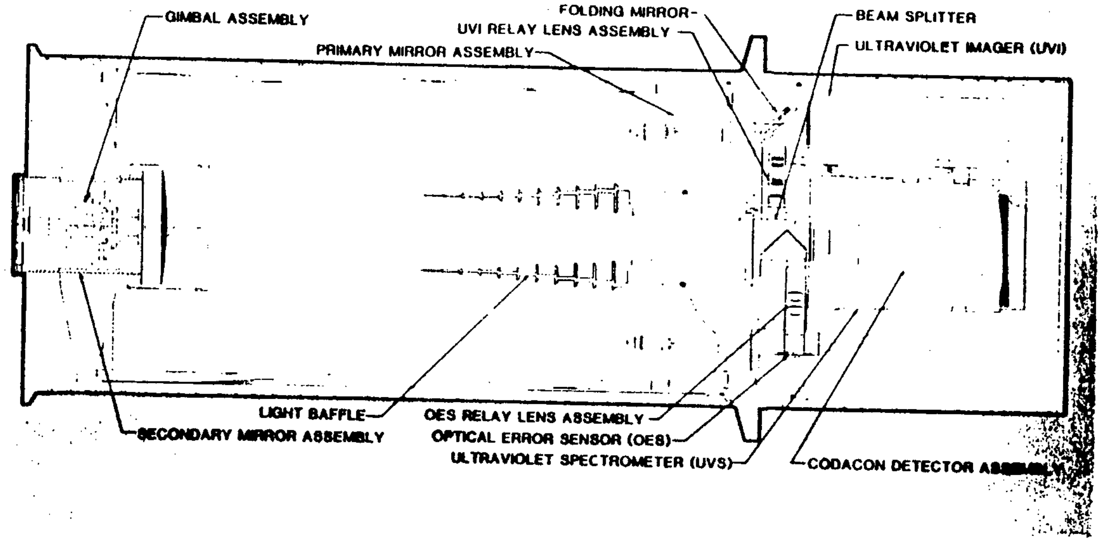
High velocity reentry events may induce visible emissions that can be seen in the night and be useful for metric and temporal discrimination. Low to medium velocity reentry events are unlikely to produce an adequate visible signature.

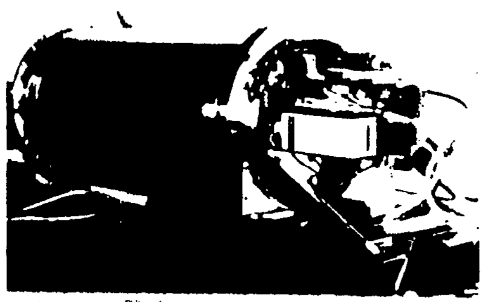
4. Kill Assessment

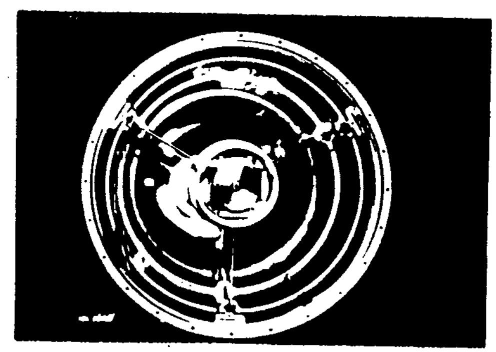
Visible and UV flashes are seen from high-energy intercepts but may not be bright from the lower speed intercepts expected from theater operations. However, since the visible signature of space objects is strongly dependent on object orientation, these sensors may be useful under solar illuminated conditions for KA determination after interception.

Appendix 3.3-B

JPL
UVSI
LASP
LORAL
EOS

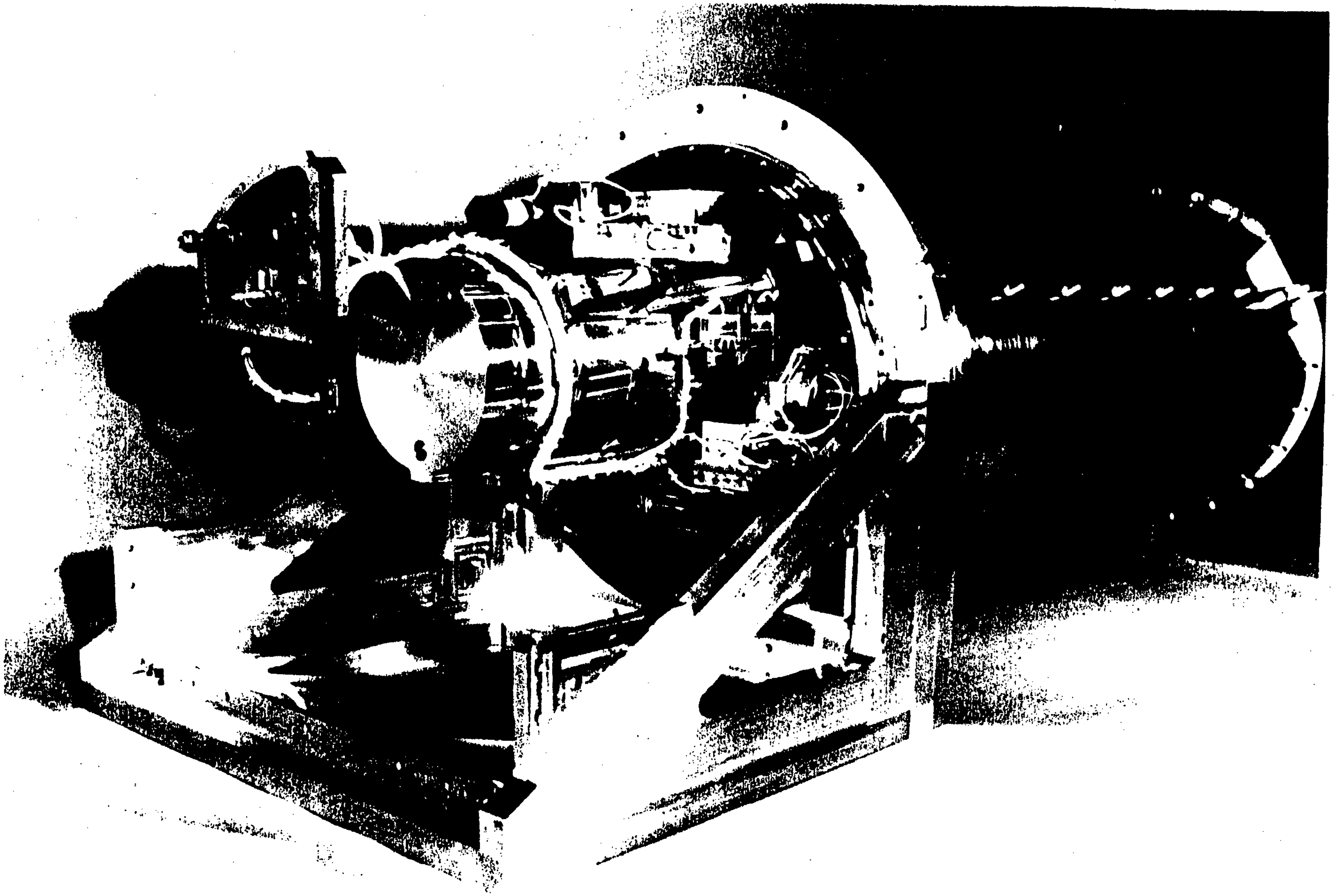






FUNCTIONAL DEMONSTRATION
October 1987

UPDATE APRIL 90



UVSI SPECTROMETER

UV MEASUREMENTS AND AVAILABLE DATA BASE

AFTER DELTA-180 AND PATHFINDER

MEASUREMENT REQUIREMENTS STILL VALID

* UV RADIOMETRIC MEASUREMENTS AT SOURCE (LIQUIDS & SOLIDS)
* < 3 METER RESOLUTION ELEMENTS (STRONGLY PREFER < 1 METER)
* NEED SPECTRAL AND SPATIAL DISTRIBUTIONS IN UV
* NEED MEASUREMENT OF TRANSIENTS AND FLUCTUATIONS
    TEMPORAL SCALES OF 1 TO 100 ms
* NEED BACKGROUND SOURCE MEASUREMENTS IN UV
    SMALL SCALE CLUTTER
    WITH SAME INSTRUMENT PROVIDING THE DATA BASE

HARDWARE INHERITANCE

- INSTRUMENT CONCEPT BASED UPON A SUCCESSFULLY FLOWN PLANETARY ROCKET INSTRUMENT WITH ELEMENTS OF: 0.4 m UV TELESCOPE, ACTIVE TRACKER, 1/4 m SPECTROMETER

- OTHER ELEMENTS COME FROM:

  LASP SPARTAN HALLEY PROGRAM:

  DATA AND CONTROL COMPUTER, 1-D CODACON DETECTOR, SHUTTLE ENVIRONMENT & SAFETY KNOWLEDGE

  LORAL EOS UV IMAGER PROGRAM FOR NAVY & ELECTRONICS & CAMERAS FOR JPL DEEP SPACE PLANETARY MISSIONS:

  INTENSIFIED CID & CCD CAMERAS WITH RbTe CATHODES, PLANETARY MISSION CAMERAS (VOYAGER), SPACECRAFT INTERFACE SYSTEMS, SPACECRAFT INSTRUMENT GROUND SUPPORT SYSTEMS, IMAGING PROCESSING SOFTWARE & IMAGE MANIPULATION

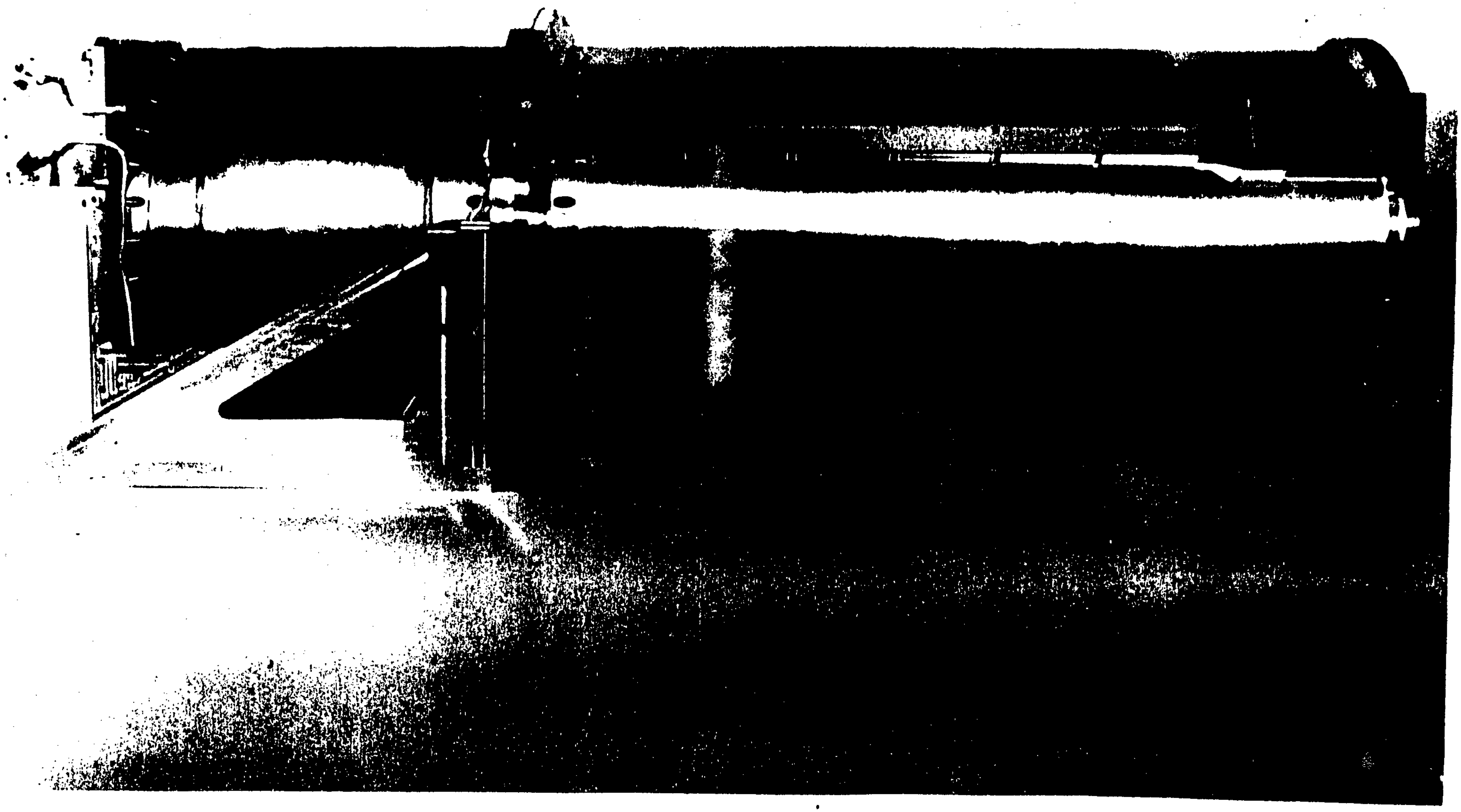
  LORAL LACE/UVPI CAMERA HARDWARE EXPERIENCE DELTA STAR & LACE UVPI

  JPL ENGINEERING MECHANICS SUPPORT FOR STRUCTURES, DOOR, THERMAL ANALYSIS, RELIABILITY & QUALITY ASSURANCE ANALYSIS

  JPL INTEGRATED SCIENCE AND PROGRAM MANAGEMENT TEAM

HARDWARE MODIFICATIONS

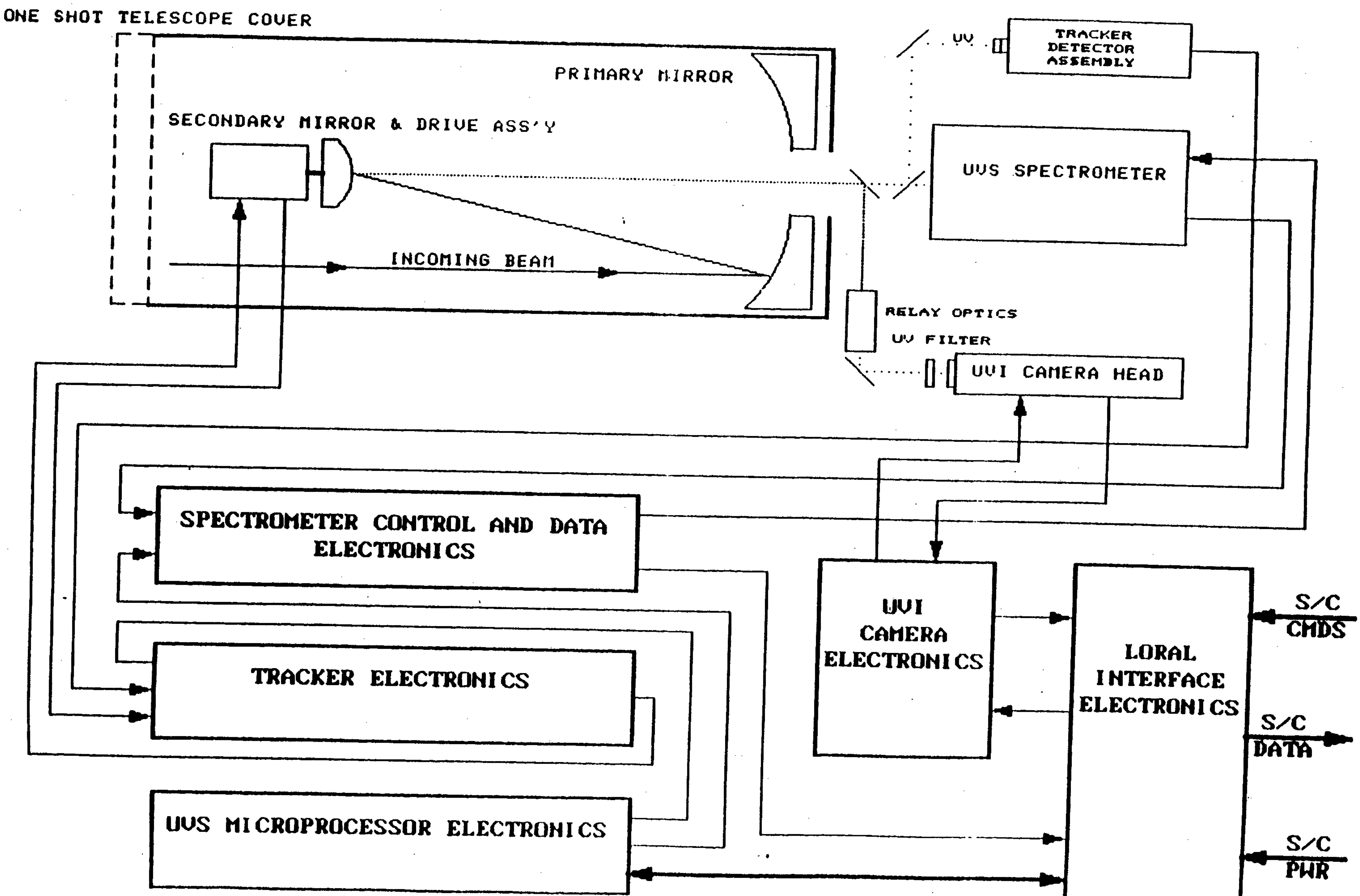
* MODEST CHANGE TO TELESCOPE MIRROR FIGURE TO IMPROVE IMAGING
* ADD 30 Hz ICCD UV SOLAR BLIND IMAGER
* ENHANCE QUALITY OF BEAM SPLITTERS
* USE OF TWO POSITION ENTRANCE SLIT FOR SPECTROMETER
* PROVIDE FOR SPECTRAL FILTERING OF TRACKER LIGHT INPUT
* PROVIDE POSITION FOR POTENTIAL SECOND IMAGER SENSOR
* ADD INTERFACE BOXES TO HANDLE IMAGER
* ADD TRACKER ERROR SIGNALS TO ENGINEERING DATA STREAM
* ADD NEW DOOR TO PROTECT OPTICS DURING LAUNCH



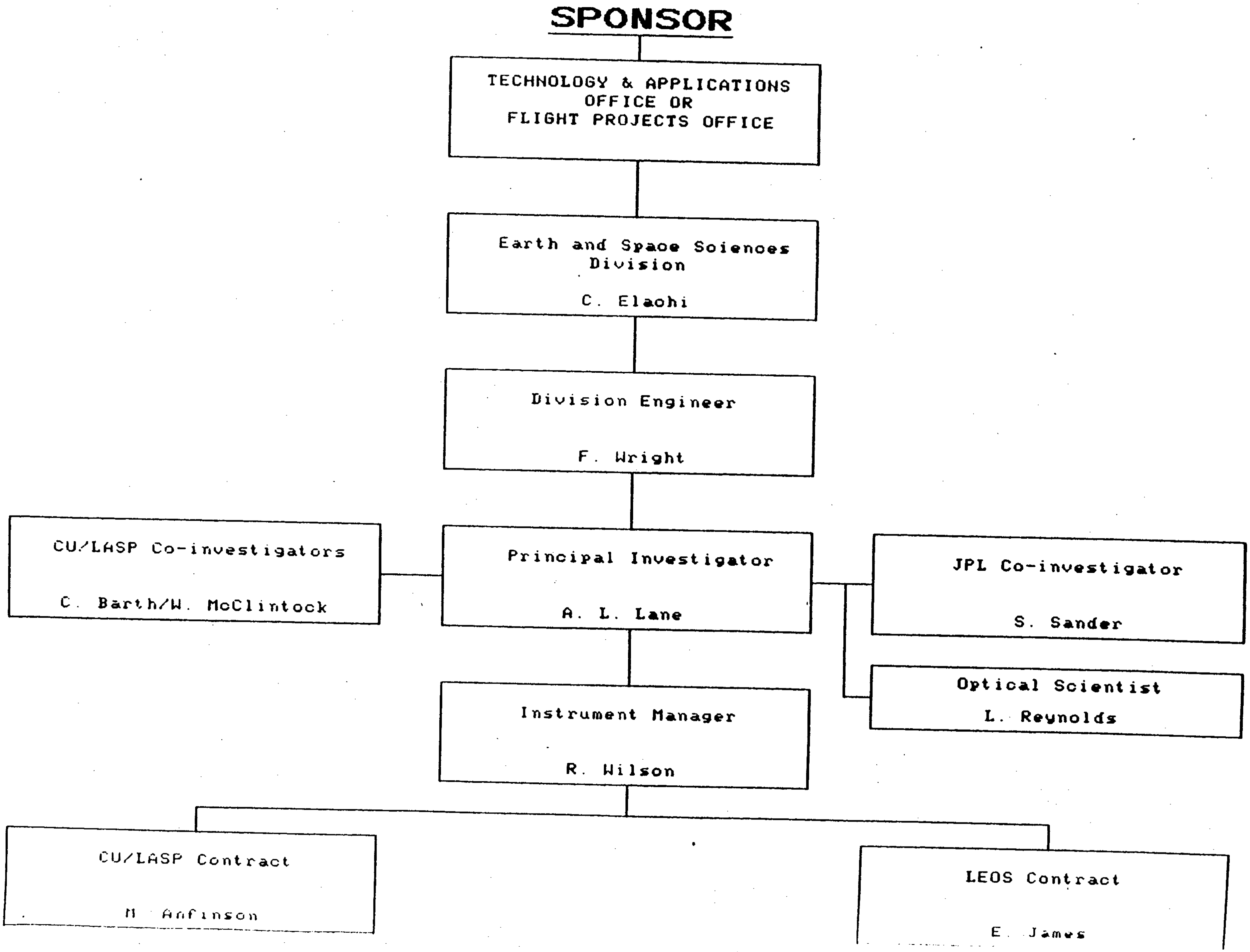
UVSI FLIGHT INSTRUMENT

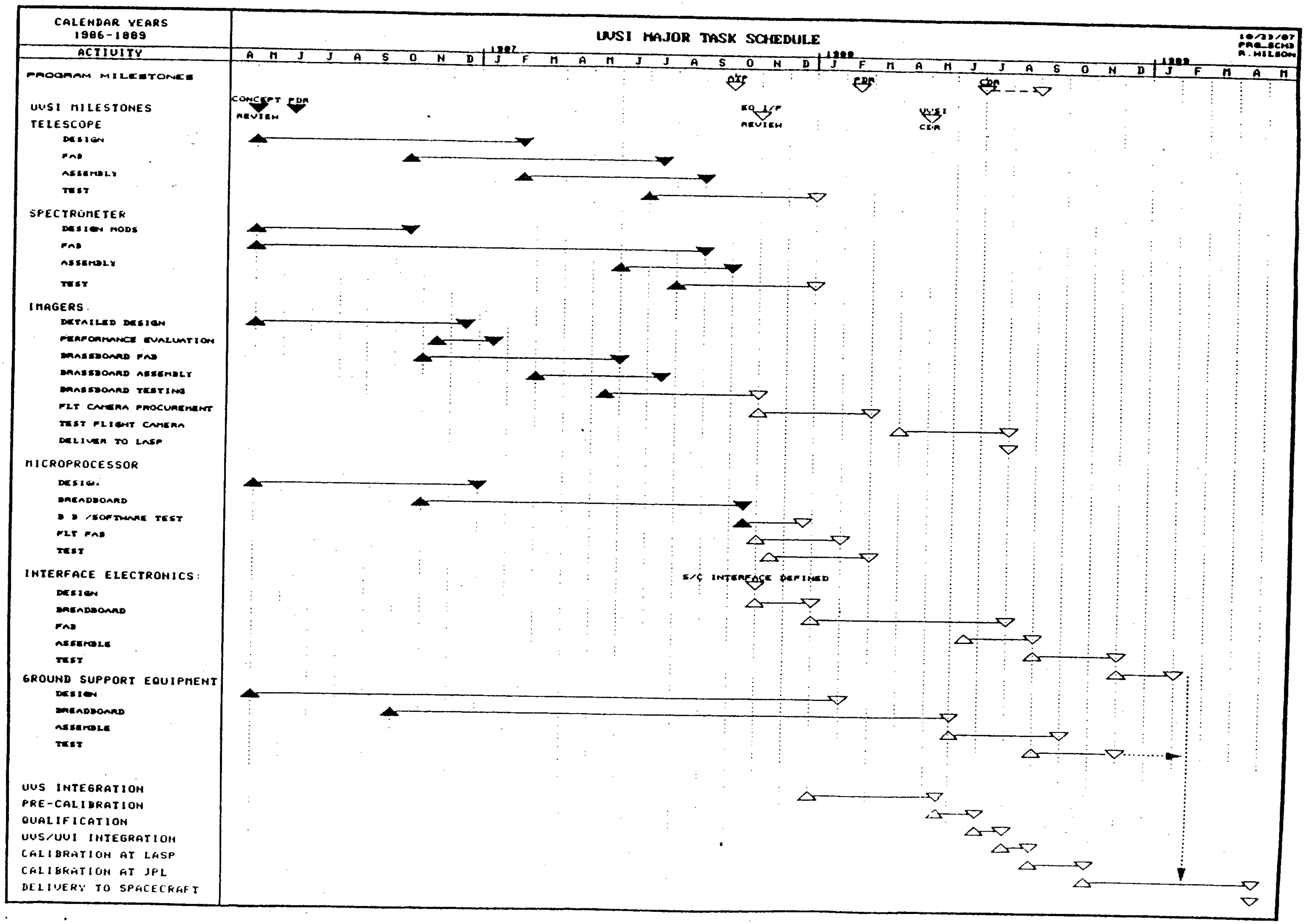
UVSI FUNCTIONAL BLOCK DIAGRAM

R. J. WILSON
10/23/87
UVSISMBK



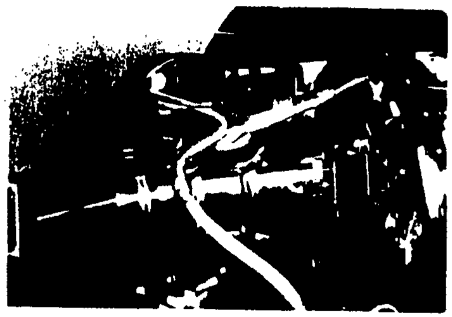
UVSI PROGRAM ORGANIZATION




UVSI MAJOR TASK SCHEDULE
10/23/87
PRG_SCHD
R. WILSON
CALENDAR YEARS
1986-1989
ACTIVITY
A M J J A S O N D
1987 J F M A M J J A S O N D
1988 J F M A M J J A S O N D
1989 J F M A M
PROGRAM MILESTONES
ATP
PDR
CDR
UVSI MILESTONES
CONCEPT REVIEW
PDR
EQ I/F REVIEW
UVSI CDR
TELESCOPE
DESIGN
FAB
ASSEMBLY
TEST
SPECTROMETER
DESIGN MODS
FAB
ASSEMBLY
TEST
IMAGERS
DETAILED DESIGN
PERFORMANCE EVALUATION
BRASSBOARD FAB
BRASSBOARD ASSEMBLY
BRASSBOARD TESTING
FLT CAMERA PROCUREMENT
TEST FLIGHT CAMERA
DELIVER TO LASP
MICROPROCESSOR
DESIGN
BREADBOARD
B B /SOFTWARE TEST
FLT FAB
TEST
INTERFACE ELECTRONICS:
S/C INTERFACE DEFINED
DESIGN
BREADBOARD
FAB
ASSEMBLE
TEST
GROUND SUPPORT EQUIPMENT
DESIGN
BREADBOARD
ASSEMBLE
TEST
UVS INTEGRATION
PRE-CALIBRATION
QUALIFICATION
UVS/UVI INTEGRATION
CALIBRATION AT LASP
CALIBRATION AT JPL
DELIVERY TO SPACECRAFT

TELESCOPE-TRACKER-SPECTROMETER

ULTRAVIOLET SPECTROMETER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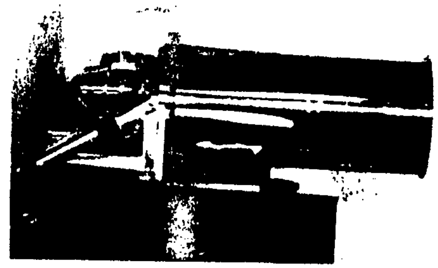
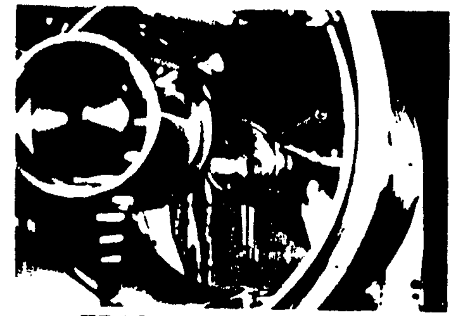

UV TELESCOPE/SPECTROMETER ASSEMBLY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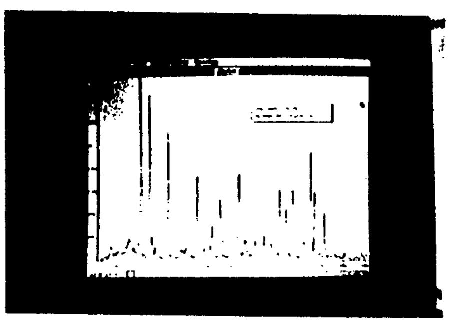
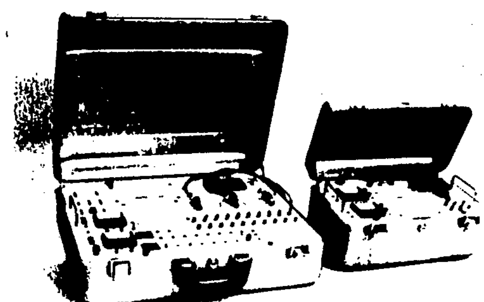

UV PLATINUM SOURCE SPECTRUM


SECONDARY MIRROR ASSEMBLY

LASP


UV SPECTROMETER GROUND SUPPORT EQUIPMENT

UVSI FLIGHT UNIT AND GSE - JANUARY, 1990

INSTRUMENT DESCRIPTION

A. General Description

The UVSI (See Figure 1 and Table 1) consists of a Cassegrain type telescope simultaneously feeding an ultraviolet spectrograph (UVS), an ultraviolet imager (UVI), and a tracking system.

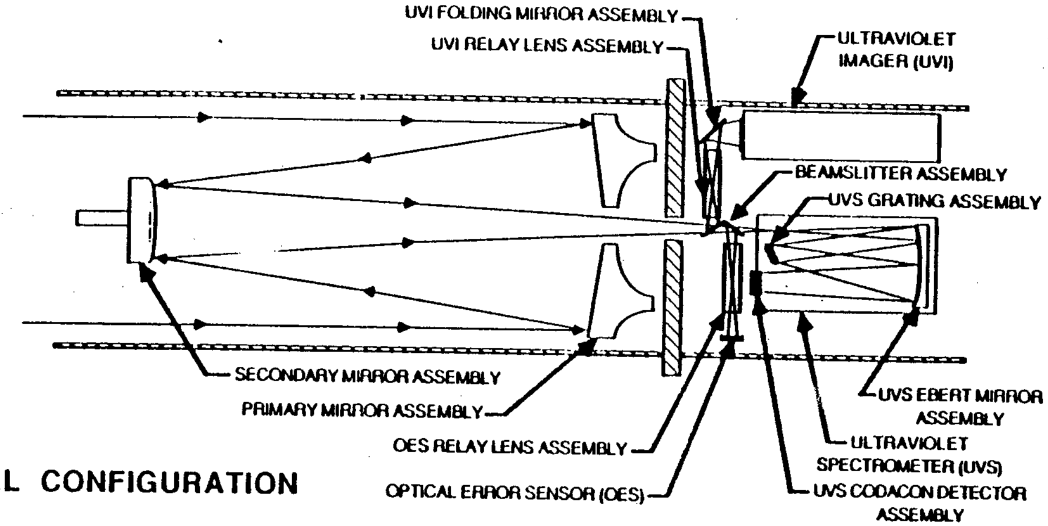
B. Telescope

The telescope is a tilted aplanat (hyberbolic primary mirror - hyberbolic secondary mirror) with a diameter of 40 cm and an effective focal length of 240 cm, which provides a focal plane scale of 0.012 mm per arc second. In order to maintain thermal stability the mirrors are made from low expansion Zero-Dur glass ceramic and their axial separation is maintained by Super-Invar rods and links. Thermal blankets will be employed in order to minimize radiative coupling of the telescope to its environment.

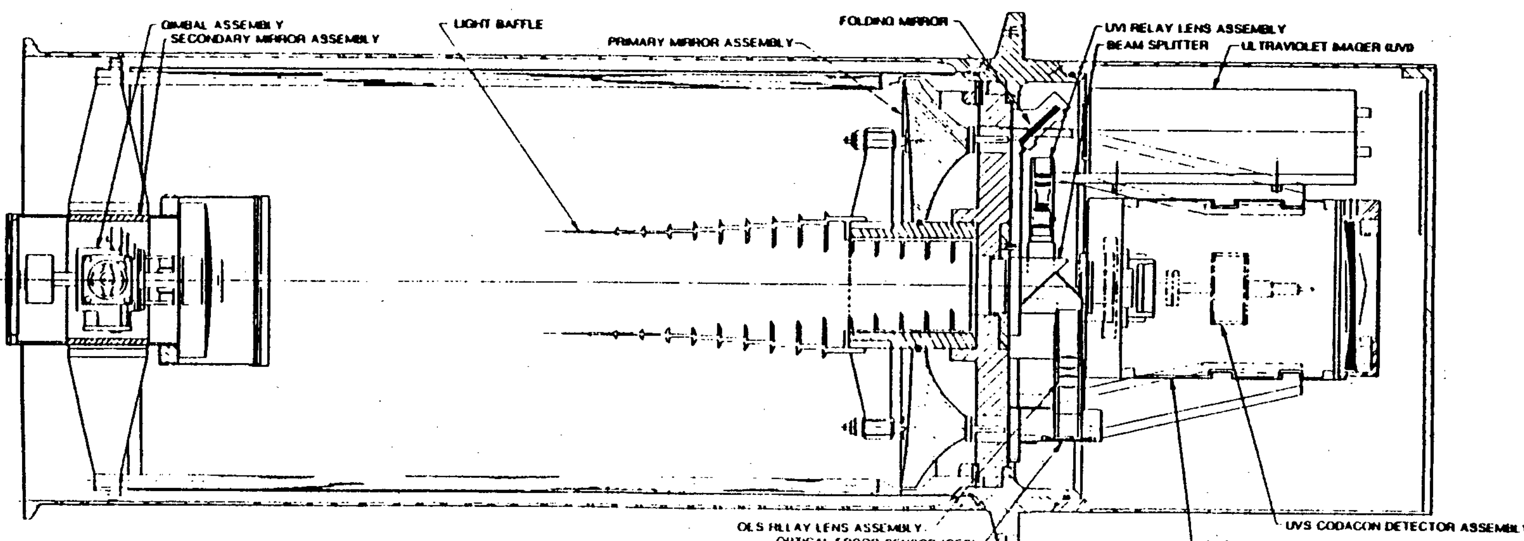
The secondary mirror is mounted on the shaft of a balanced gimbal assembly which allows it to pivot about a point behind its vertex. The optical design of the telescope system is such that targets located at off axis field positions in the telescope focal plane can be repositioned to the center of the focal plane, without degrading the image quality, by pivoting the secondary mirror.

The gimbal system is controlled by feedback from an optical sensor located near the focal plane of the telescope. Approximately 10% of the target signal is directed to a quadrant anode microchannel plate detector by a beam splitter assembly located near the telescope focal plane. Error signals from the sensor are used to drive brushless torque motors to position the secondary mirror axis.

The error sensor is an ITT 4149 microchannel plate multiplier phototube with a bi-alkali photocathode and a 25 mm diameter active area resulting in an acquisition field of view of 10 milli-radians. Because wide dynamic range is required for the UVSI, this device is used in a constant current mode by providing automatic gain control for the microchannel plate high voltage power supply.



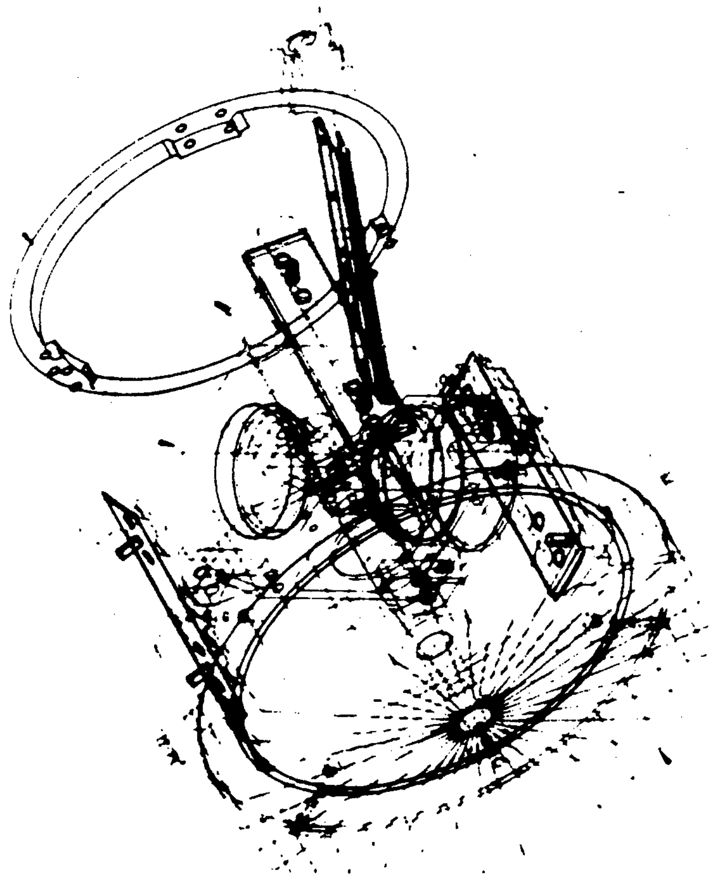
OPTICAL CONFIGURATION



MECHANICAL CONFIGURATION

UVSI PLANETARY/PATHFINDER

FIGURE 1



TELESCOPE

(CONCENTRIC FOLDED TWO MIRROR WITH TILTING SECONDARY FOR IMAGE STABILIZATION)

OPTICAL SPECIFICATION	TILTED APLANT
FOCAL LENGTH	2.4 m
APERTURE	39.6 cm
CENTRAL OBSCURATION	15.0 cm
FOCAL PLANE SCALE	11.5 µm per arc sec
IMAGE STABILIZATION SYSTEM	CLOSED-LOOP OPTICAL FEEDBACK
ACQUISITION FIELD OF VIEW	10 milliradian DIA (35 arc min)
DETECTOR	ITT QUADRANT ANODE MCP
PHOTOCATHODE	KCsSb(Bi-alkali)
SPECTRAL RESPONSE	200 nm - 550 nm
BANDWIDTH	30 Hz

SPECTROGRAPH

OPTICAL CONFIGURATION	EBERT FASTIE
FOCAL LENGTH	250 mm
GRATING (PRESENT CONFIGURATION)	900 g/mm BLAZED AT 250 nm
DISPERSION (PRESENT CONFIGURATION)	4.21 nm/mm
RESOLUTION (PRESENT CONFIGURATION)	0.37 nm
COVERAGE (PRESENT CONFIGURATION)	110 nm - SIMULTANEOUS
WAVELENGTH RANGE (PRESENT CONFIGURATION)	210 - 320 nm
WAVELENGTH RANGE POSSIBLE	160 - 350 nm
DETECTOR	LASP CODACON MCP
ANODE	CODED ARRAY
NUMBER OF CHANNELS	1024
PIXEL SPACING	0.025 mm
PHOTOCATHODE	CESIUM TELLURIDE

UVS SENSITIVITY

(FLUX REQUIRED TO PRODUCE 1 COUNT/SEC)

2.5×10^{-19} WATTS/cm^2	200 nm
4.0×10^{-19} WATTS/cm^2	250 nm
1.7×10^{-18} WATTS/cm^2	300 nm

IMAGER

(COHU CAMERA)

OPTICAL SPECIFICATION	X2.9 RELAY LENS FOLDING SYSTEM
FORMAT	754 X 488 PIXELS
FIELD OF VIEW	4 X 3 milliradian
PHOTOCATHODE	RUBIDUM TELLURIDE
SENSITIVITY	2×10^{-15} WATTS/PIXEL @270 nm

UVSI CHARACTERISTICS

UVS (INCLUDING TELESCOPE)	
MASS	305 LBS
LENGTH	59.5 INCHES + APERATURE COVER
WIDTH	23.0 INCHES
POWER	27.0 WATTS
UVI	
MASS	14 LBS (+/- 2 LBS)
POWER	8.0 WATTS
INTERFACE BOX	
MASS	35 LBS (+/- 10 LBS)
LENGTH	26.0 INCHES
WIDTH	8.0 INCHES
HEIGHT	TBD
POWER	TBD WATTS

ULTRAVIOLET SPECTROGRAPH-IMAGER SUMMARY

TABLE 1

C. Spectrograph

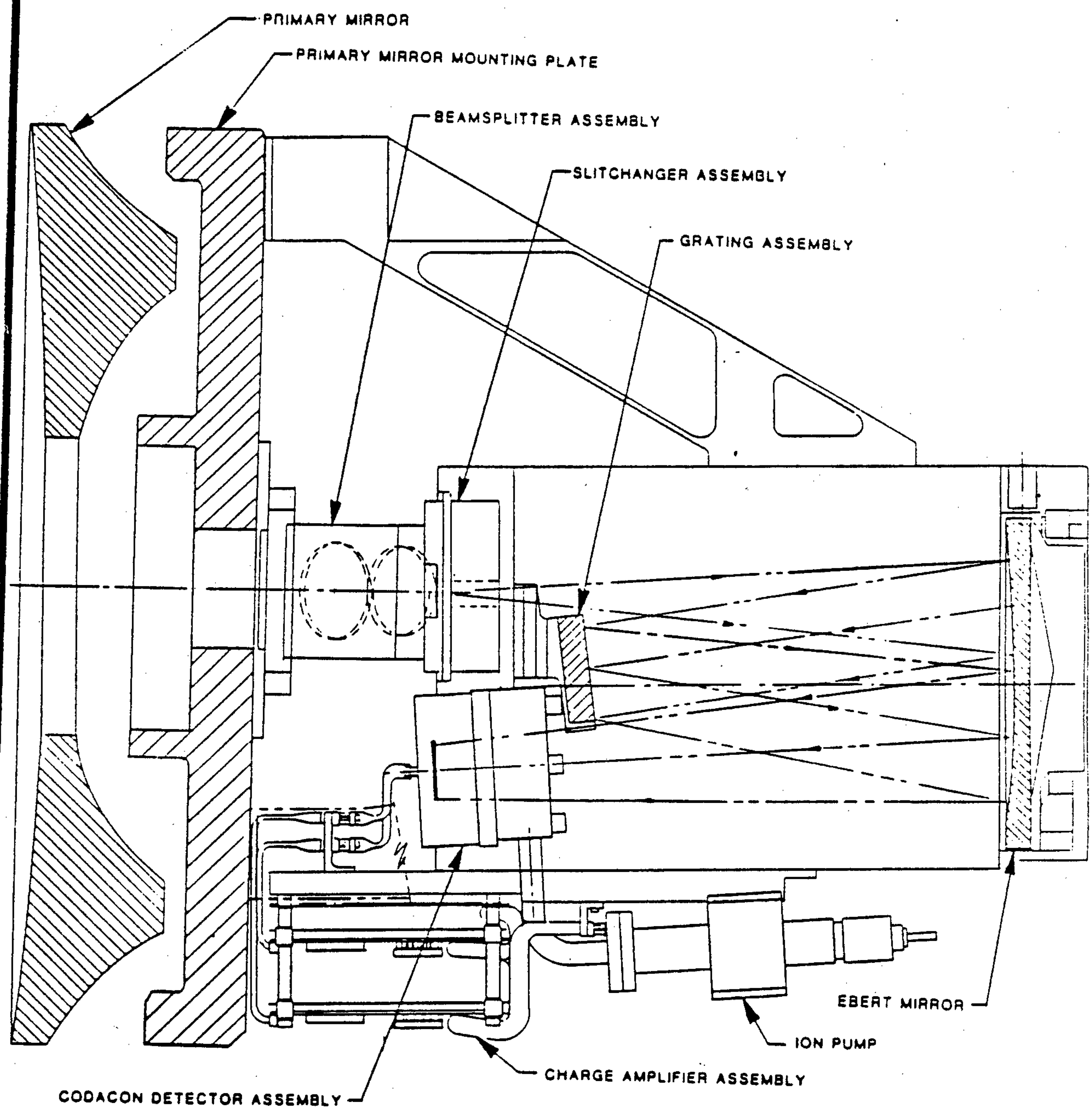
The spectrograph (see Figure 2) is a modified Ebert-Fastie monochromator with a 250 mm focal length. The optical design is similar to the spectrometer used in the Mariner 6, 7 and 9 Ultraviolet Experiments. The exit slits and photomultiplier tubes were removed and the grating was moved 25 mm toward the Ebert mirror in order to install the MCP-CODACON detector and its housing. The spectral coverage is determined by the 26 mm size of the CODACON and the choice of grating. For example, a grating with a ruling density of 900 g/mm and a first order blaze wavelength of 250 nm provides a total coverage of 110 nm with a fixed grating position. The dispersion is 4.21 nm/mm and the detector pixel spacing is 0.107 nm.

D. Detector

The CODACON is a photon-counting multi-channel detector consisting of a curved-channel microchannel plate which is mounted in proximity focus with a coded anode array. A schematic diagram of the CODACON system is shown in Figure 3. A cesium telluride photocathode deposited on the front surface of the MCP converts incident ultraviolet photons into electrons. A voltage applied across the MCP causes these photoelectrons to be accelerated through tubes (microchannels), producing secondary emission of electrons from the channel walls. With an applied potential of 1600 v, the production of a single photoelectron at the top surface of the MCP results in a localized pulse of 10^6 electrons exiting from the back of the plate. The MCP used in the CODACON was fabricated by Galileo Electron-Optics. It has an active area approximately 27 mm in diameter, and 25 μ diameter microchannels on 32 μ centers.

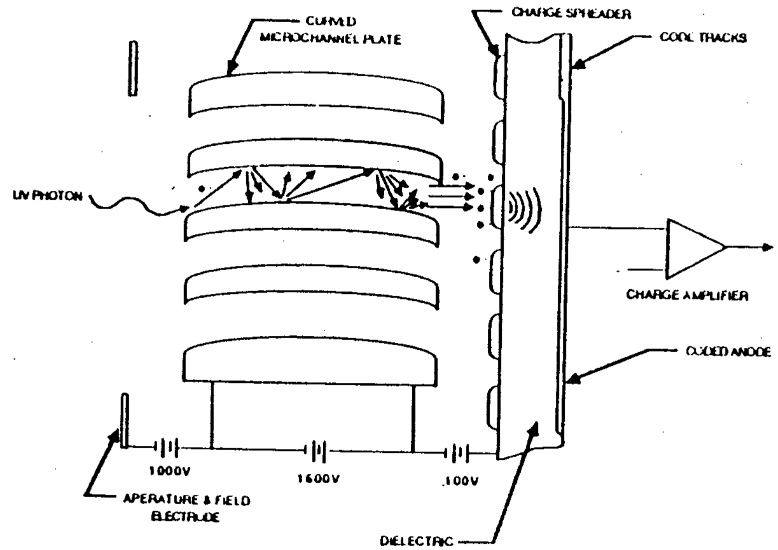
In order to use an MCP as the detector in spectrograph, it is necessary to provide an anode at the back of the plate which is capable of locating electron pulses in one dimension. This is accomplished in the CODACON by having a three-layer code plate serve as the output anode of the MCP. The top layer of the code plate consists of 1024 charge spreaders which are 13 mm long and 15 μm wide. Center-to-center spacing of the spreaders is 25.4 μm; thus, there are 40 spreaders per mm, lying directly below the MCP and running perpendicular to the spectrograph dispersion direction. The spacing between the anode array and the MCP is 50 μ.

The middle layer consists of a dielectric material approximately one hundred thousand angstroms thick. The bottom layer consists of ten pairs of binary code tracks which are used to determine which of the 1024 charge spreaders was struck by any 10^6 electron pulse leaving the back of the MCP.

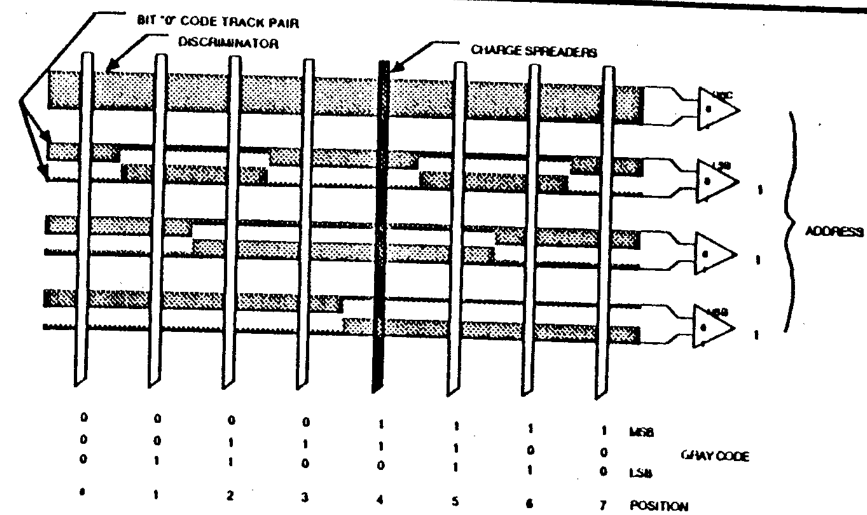


ULTRA-VIOLET SPECTROMETER ASSEMBLY

FIGURE 2



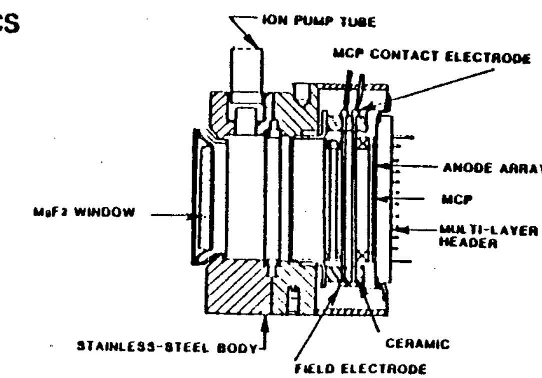
(SIDE VIEW)
ULTRAVIOLET PHOTONS INCIDENT ON THE FRONT SURFACE OF THE MCP RESULT IN A BURST OF 10^6 ELECTRONS EXITING FROM THE BACK SURFACE. THE POSITION IN ONE DIMENSION OF THE PULSE IS DETERMINED BY A CODED ANODE ARRAY. A 10 BIT ADDRESS IS PRODUCED FOR EACH ELECTRON PULSE CORRESPONDING TO WHICH ONE OF 1024 CHARGE SPEADER WIRES WAS STRUCK BY THE PULSE.



EXAMPLE OF AN EIGHT-CHANNEL CODACON
IT CONSISTS OF EIGHT CHARGE SPREADER WIRES, 3 BIT TRACKS, AND A DISCRIMINATOR. A CHARGE PULSE ON A CHARGE SPREADER CAUSES CHARGE TO BE INDUCED IN THE CODE TRACKS. DIFFERENTIAL AMPLIFIERS CONNECTED TO EACH BIT TRACK PAIR DETERMINE WHICH SIDE OF A PAIR HAS THE MOST CHARGE BASED ON HOW WIDE THE CODE TRACK DIRECTLY BELOW THE CHARGE SPREADER. AS AN EXAMPLE, CHARGE INDUCED AT POSITION 4 WOULD PRODUCE A 110 ADDRESS. NOTE THAT THE CODING OF CHARGE SPREADER POSITION IS A GRAY CODE RATHER THAN A BINARY. GRAY CODES FOR EACH OF THE EIGHT CHANNELS ARE SHOWN BY EACH CHARGE SPREADER IN THE ORDER MOST SIGNIFICANT BIT TO LEAST SIGNIFICANT BIT.

DETECTOR SYSTEM SCHEMATICS

DETECTOR HOUSING CONSISTING OF CERAMIC AND KOVAR-RINGS WELDED TO A STAINLESS STEEL FLANGE AND WINDOW.



DETECTOR MECHANICAL CONFIGURATION

MCP-CODACON DETECTOR

FIGURE 3

The detector tube housing for the CODACON is fabricated at EMR Photoelectric. The main features of this device are: a fully bakeable stainless steel housing with copper gasket seals, and an environment free from organic compounds. This housing also allows a photocathode to be applied to the front surface of the MCP without exposure to atmospheric contamination after deposition. A small ion pump is used to maintain a vacuum of less than 10^{-8} torr in the detector.

The detector output is directed to a one of two dedicated 8 bit by 1024 location memories, where the 1024 row corresponds to the 1024 element of the CODACON detector. Spectra are accumulated as photon counts at the corresponding memory locations. When the data (CODACON location) word appears at the detector output, the contents of the memory location with that address are removed, incremented by one, and stored back into that same memory address, up to the 255 count level.

Data are read from this accumulating memory at an externally controlled telemetry rate. After the end of the integration cycle (~400 msec), the new measurement data are directed to the alternate memory and the first memory is clocked out into the telemetry stream.

Laboratory testing is accomplished with ground support equipment (GSE) which simulates the telemetry system. The GSE is interfaced to a Macintosh II computer which is used for real time storage of data and subsequent analyses.

E. Slit Changer

The entrance aperture of the spectrograph is equipped with a bistable slit changer which activates a narrow slit for viewing extended sources and a wide slit for viewing point sources. The system is actuated by a brushless DC torque motor driven by a saturated H bridge amplifier, and is commanded from the on-board microprocessor.

F. Microprocessor

The microprocessor subsystem is used to receive and interpret UVS commands, and to sample and format all housekeeping telemetry.

The design includes use of the proven and tested Spartan Halley microprocessor, a National Semiconductor Corporation's NSC800. The system uses EPROM for the majority of program memory, with RAM being used for variable buffer memory and for uploadable, volatile, program memory. The Input/Output functions are accomplished using the NSC810 and NSC831 programmable I/O devices. A real time clock, as well as various timers, are also included in the design. Finally, the hardware includes a 48 channel, 12 bit resolution, Analog-to-Digital converter for sampling temperatures, voltages, and other analog signals.

G. UVS Performance

1. Telescope Image Quality

The functional goal for the Pathfinder UVSI telescope is an image quality of 5 μ radians over a 2 milli-radian field of view in the presence of boresight errors up to 2.5 milli-radians. This goal can only be realized in a two mirror system using a tilted aplanat design.

The unusual mirror figure of the tilted aplanat caused the vendor (Muffoletto Optical Co. in Baltimore, MD) some difficulty with the original Pathfinder delivery schedule. We therefore opted to refurbish existing optics to achieve a resolution of 2 arc seconds. These "engineering optics" which could be used for early subsystem testing will be replaced by 1 arc second quality "flight optics" before delivery to JPL for system integration. At the present time (mid October, 1987) the engineering optics have been installed in the flight hardware. Our estimates of telescope image quality in this configuration is 1.5 arc sec (7.5 μ radian) which exceeds the specification.

We have also taken delivery (October 12,1987) of the first of two sets of flight optics. The resolution of this system has been measured at delivery to be 0.75 arc seconds.

2. Image stability and tracking sensitivity.

The telescope image motion compensation system (IMCS) was incorporated in the UVSI to overcome installation boresight errors between the telescope, the Pathfinder tracking platform, and potential mispointings caused by spectrally selective trackers on the main platform. In addition this system compensates for any residual jitter in the pointing of the platform toward the target.

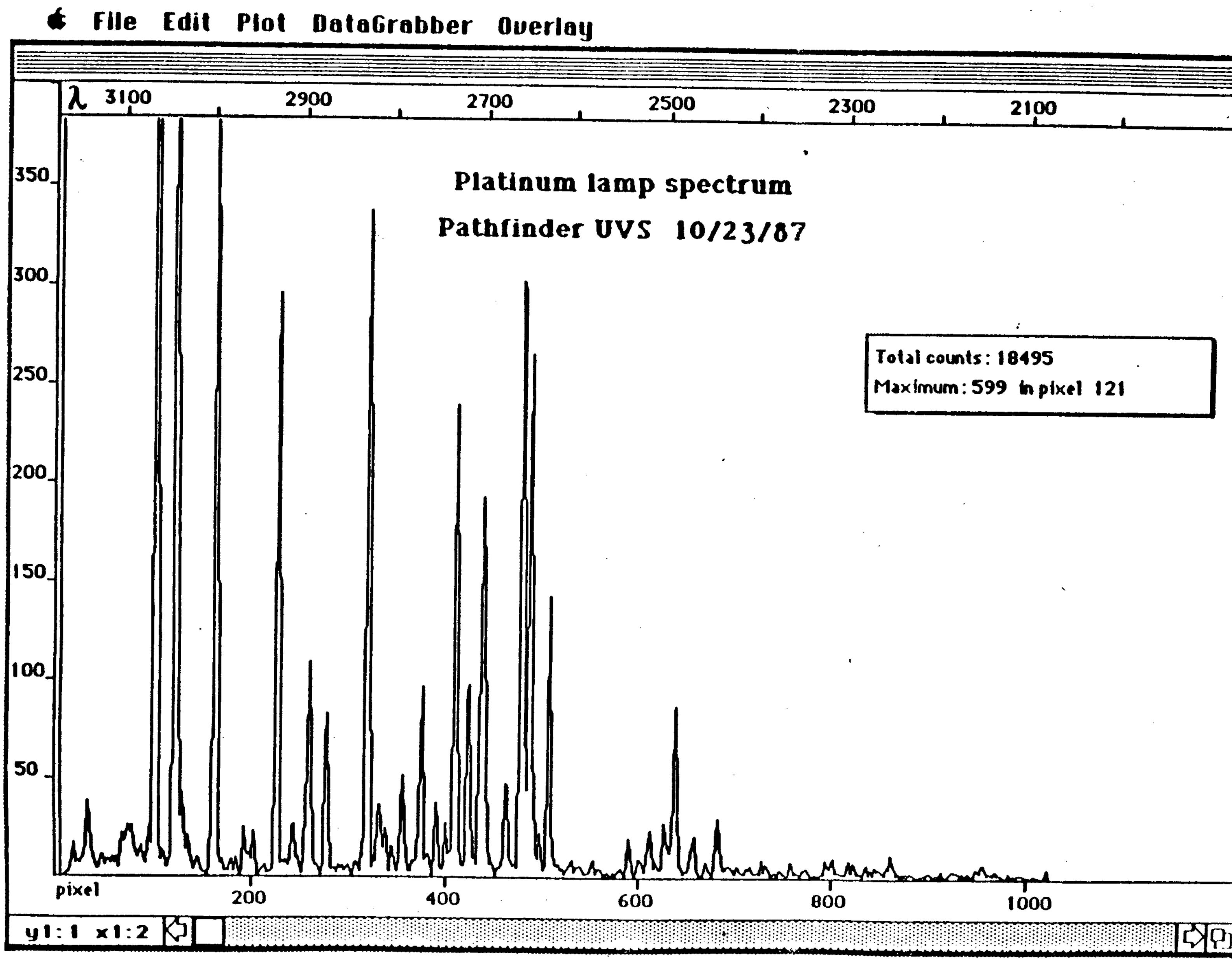
The present configuration of the system can correct for bore sight errors as large as 5 milli-radians. Using the Pathfinder project baseline target intensity of a 2350^{o} K black body normalized to 5 watts/str/micron at 300 nm, the error sensor is sensitive enough to provide 2.5 μ radian image stability with a system band width of 30 Hz. This translates to an error of 2.5 μ radian for target-telescope velocity of 65 arc seconds per second (60 meters per second at 200 KM).

3. Spectrograph resolution and sensitivity

Figure 4 shows a spectrum of a hollow cathode platinum lamp taken through the telescope-spectrograph. The lower abscissa is given in detector pixel number and the upper abscissa is wave length in Angstroms (1 nm = 10A). Images of monochromatic lines have a full width half maximum value of 3.5 pixels which corresponds to a wavelength resolution of 0.37 nm.

No measurements of the UVS integrated sensitivity have been performed. Nonetheless it is possible to make reasonable estimates based on the measured sensitivity of the detector and the optical efficiencies provided by the various vendors for the telescope mirrors, Ebert mirror, grating, and beam splitters.

These values indicate that the absolute quantum efficiency of the telescope-spectrograph varies from 0.4% at 200 nm to 0.04% at 300 nm. This means that a monochromatic point source with a signal of 2.5×10^{-19} watts/cm^2 will produce a single count per detector pixel at 200 nm. At 300 nm a monochromatic source of 1.6×10^{-18} watts/cm^2 will also produce one detector count. The detector has a dark count (produces false counts in the absence of any light) of about 5×10^{-3} counts/pixel/sec so the signal-to-noise of any observation is limited by the statistical fluctuation of photon arrival at the telescope rather than detector dark count, (i.e., the square root of the number of observed counts). To obtain a signal to noise of 5 at 200 nm in one second requires a source intensity of $25 \times 2.5 \times 10^{-19} = 6.25 \times 10^{-18}$ watts/cm^2.

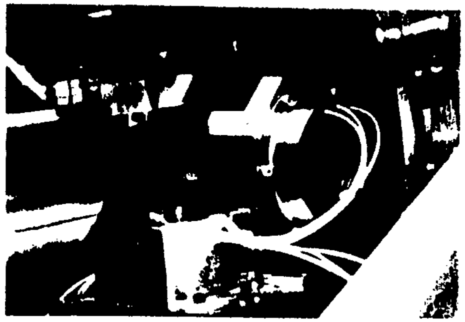


File Edit Plot DataGrabber Overlay
λ 3100 2900 2700 2500 2300 2100
Platinum lamp spectrum
Pathfinder UVS 10/23/87
Total counts: 18495
Maximum: 599 in pixel 121
350 300 250 200 150 100 50
pixel 200 400 600 800 1000
y1:1 x1:2

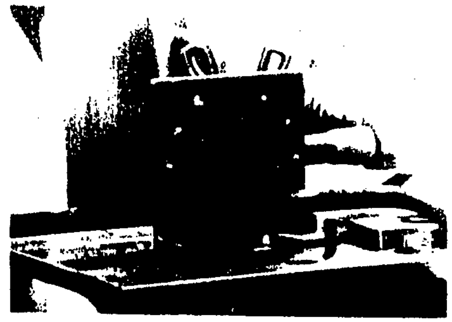
FIGURE 4

ULTRA-VIOLET IMAGER

ULTRAVIOLET IMAGER

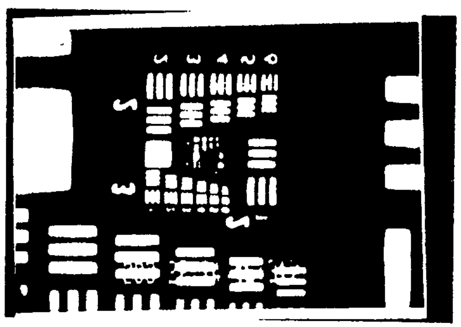


TRACKING SENSOR

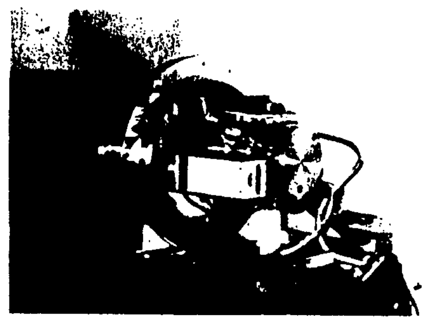


TEST TARGET

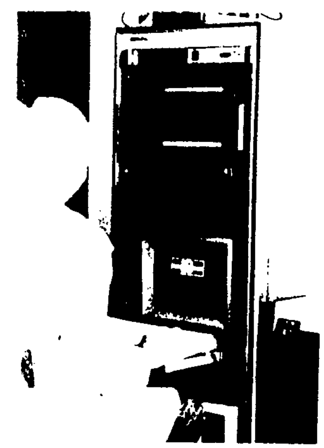
UV IMAGE



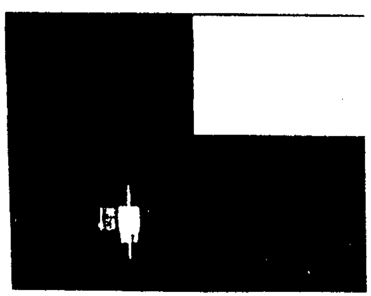
"LIVE" UV IMAGE
AIR FORCE RESOLUTION TARGET



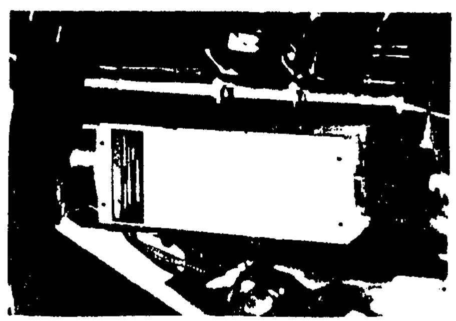
INSTALLED CAMERA



UV IMAGER
GROUND SUPPORT EQUIPMENT



"ZOOMED" POINT SOURCE

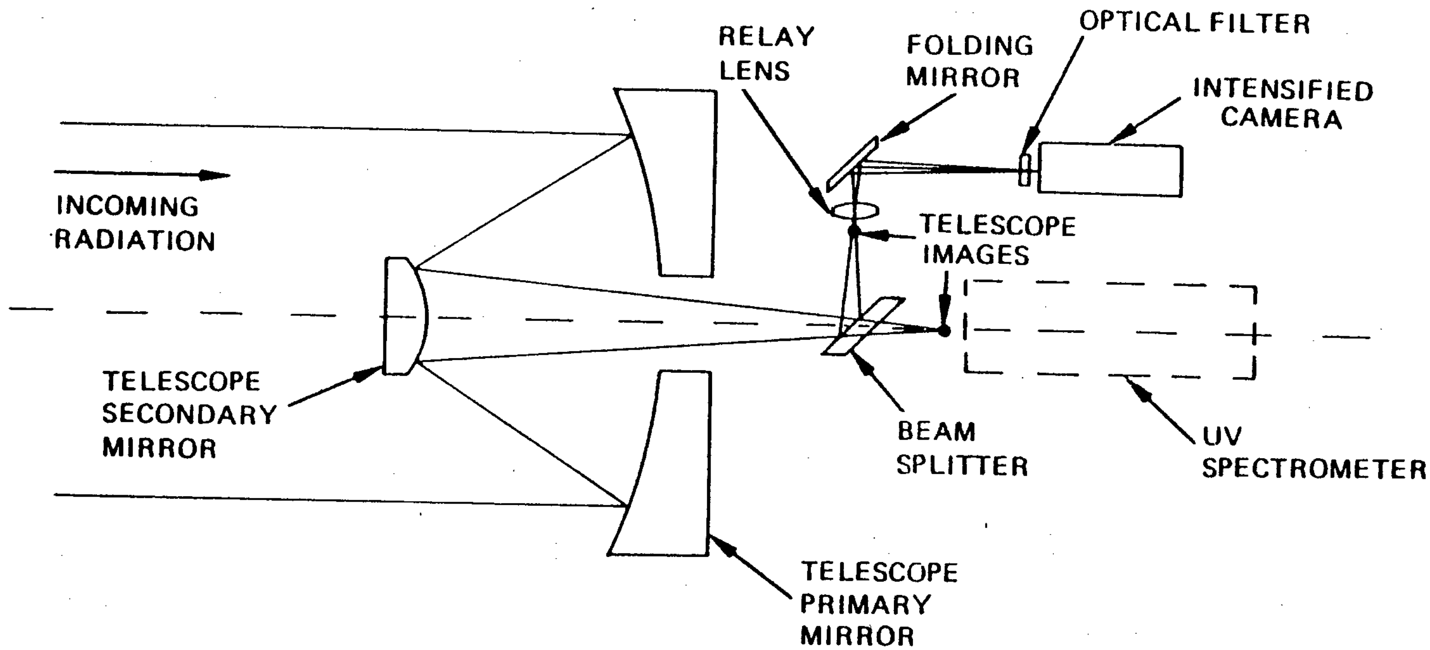


CAMERA CLOSE-UP

LORAL EOS

89256

OPTICAL SCHEMATIC FOR UV IMAGER



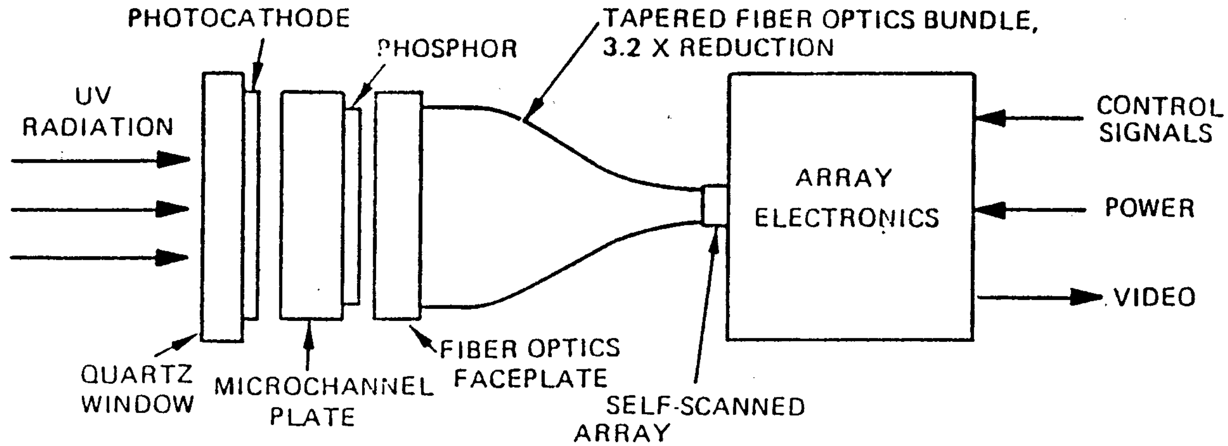
INSTRUMENT DESCRIPTION UV IMAGER

- **FUNCTION** GENERATE VIDEO IMAGERY OF DESIRED TARGETS IN THE 250-300 NANOMETER SPECTRAL REGION

- **HARDWARE**
  - TELESCOPE . . .
    - 40 CM APERTURE, 2.4 METER FOCAL LENGTH
    - TWO-MIRROR DESIGN WITH 1.0 ARC SECOND RESOLUTION
    - SECONDARY MIRROR TILTS FOR TARGET TRACKING PURPOSES
  - BEAM SPLITTER . . .
    - SPLITS THE ENERGY 50/50 BETWEEN THE UV IMAGER AND THE UV SPECTROMETER (DIFFERENT RATIOS AVAILABLE)
  - RELAY LENS . . .
    - MAGNIFIES IMAGE BY 2.9X, TO INCREASE SYSTEM RESOLUTION
  - SOLAR BLIND FILTER . . .
    - ATTENUATES OUT-OF-BAND ENERGY BY 10^{-9}; 40% PEAK IN-BAND TRANSMISSION
  - SENSOR . . .
    - INTENSIFIED CCD ARRAY
    - TWO STAGE MICROCHANNEL PLATE (MCP)
    - 40 mm DIAMETER PHOTOCATHODE, RUBIDIUM TELLURIDE
    - 30 FRAMES/SECOND, RS-170 VIDEO OUTPUT
    - CCD HAS 754 X 488 PIXELS, EACH 11.5 X 13.5 MICRONS
    - SENSOR IS 3.5 X 3.5 X 10 INCHES

SCHEMATIC OF UVI INTENSIFIED CAMERA

91807



PHOTOCATHODE
- 40 mm DIAMETER
- RUBIDIUM TELLURIDE
- 10% QUANTUM EFFICIENCY

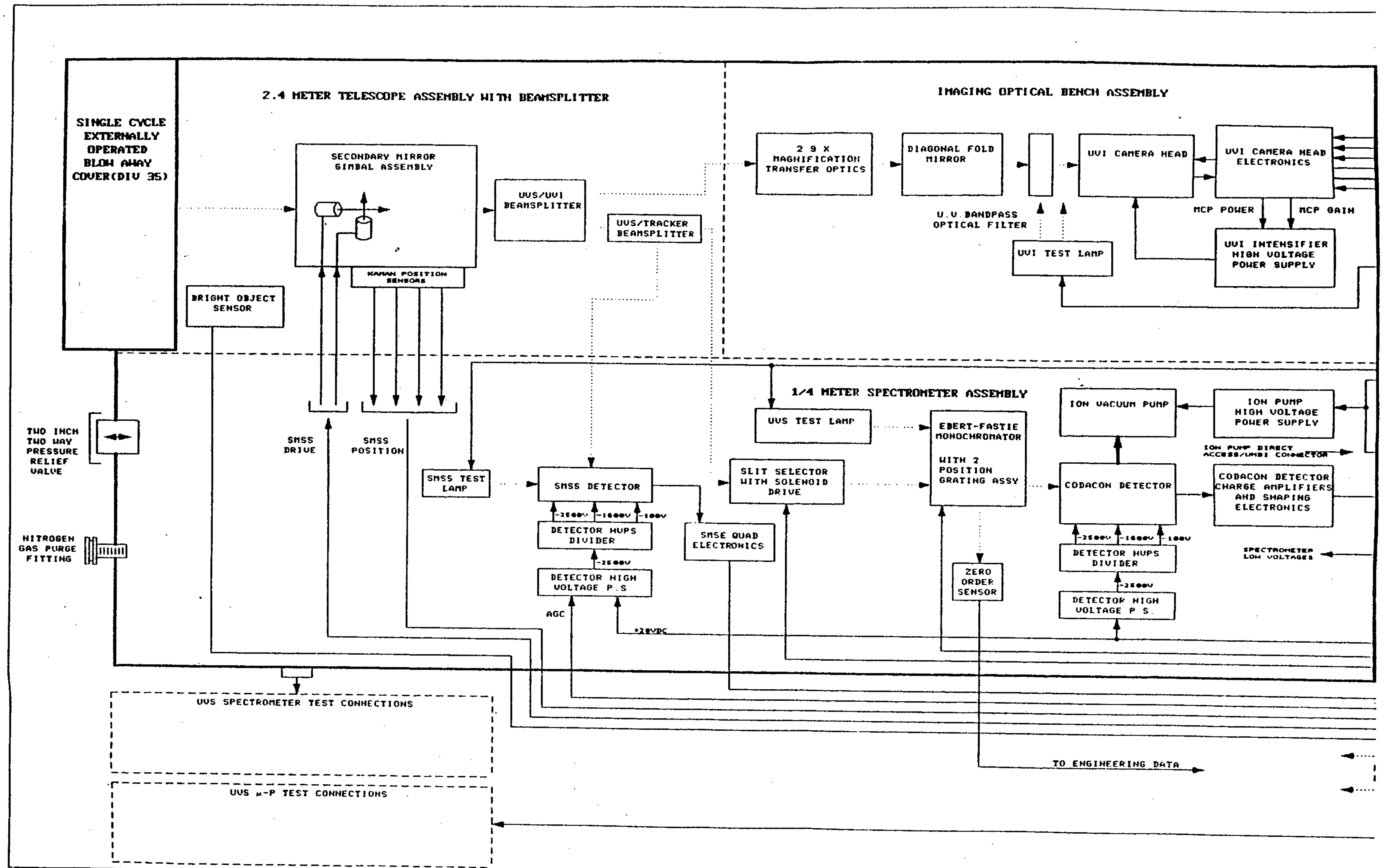
IMAGE INTENSIFIER
- TWO-STAGE MICROCHANNEL PLATE
- 10^5 ELECTRON GAIN
- P-20 PHOSPHOR OUTPUT
- $\sim 10^6$ INTENSITY GAIN ON CCD ARRAY

SELF-SCANNED ARRAY
- TI 241 CCD
- 754 X 488 PIXELS
- ARRAY DIAGONAL IS 11 mm
- 30 FRAMES/SECOND

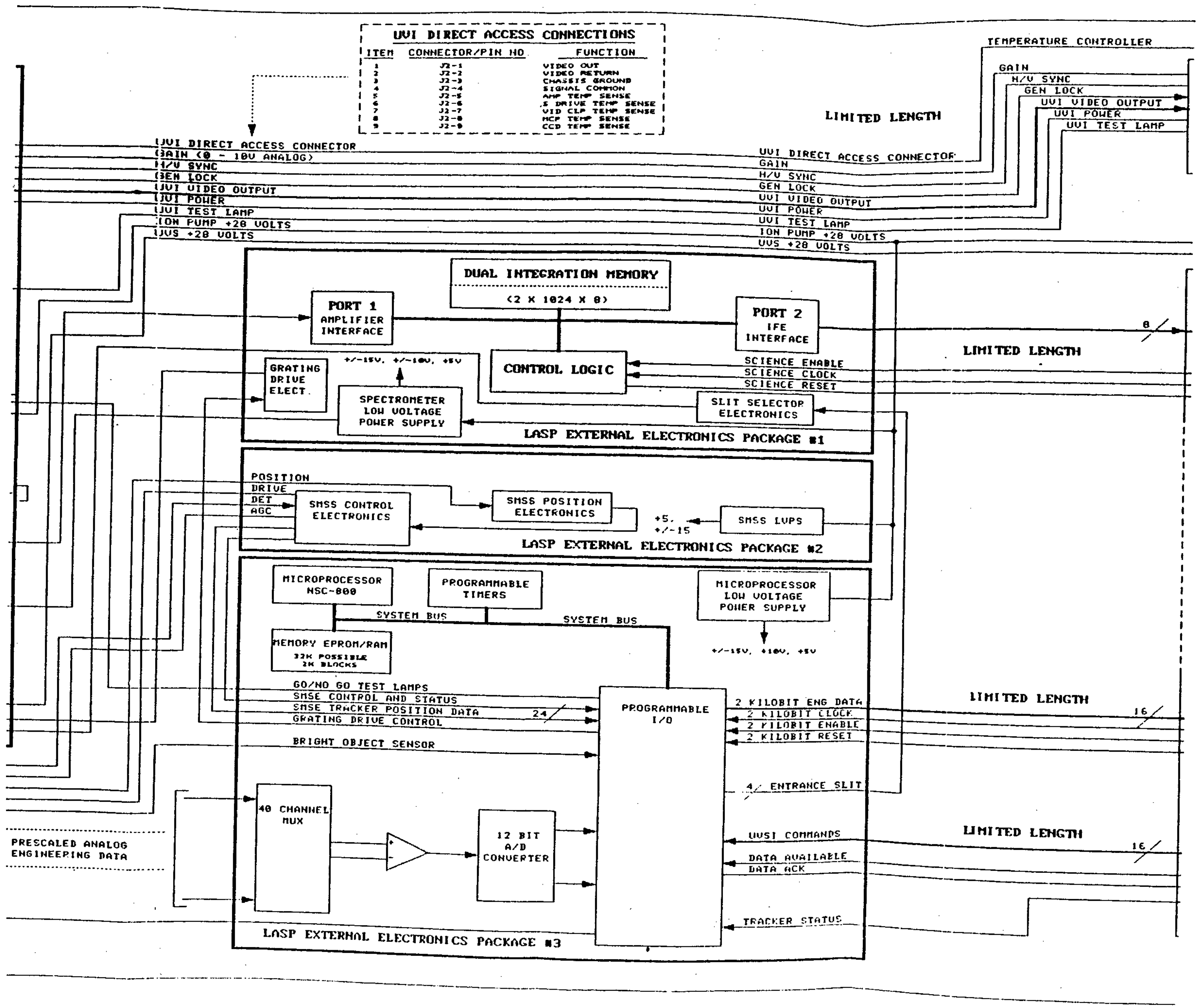
UVI PERFORMANCE

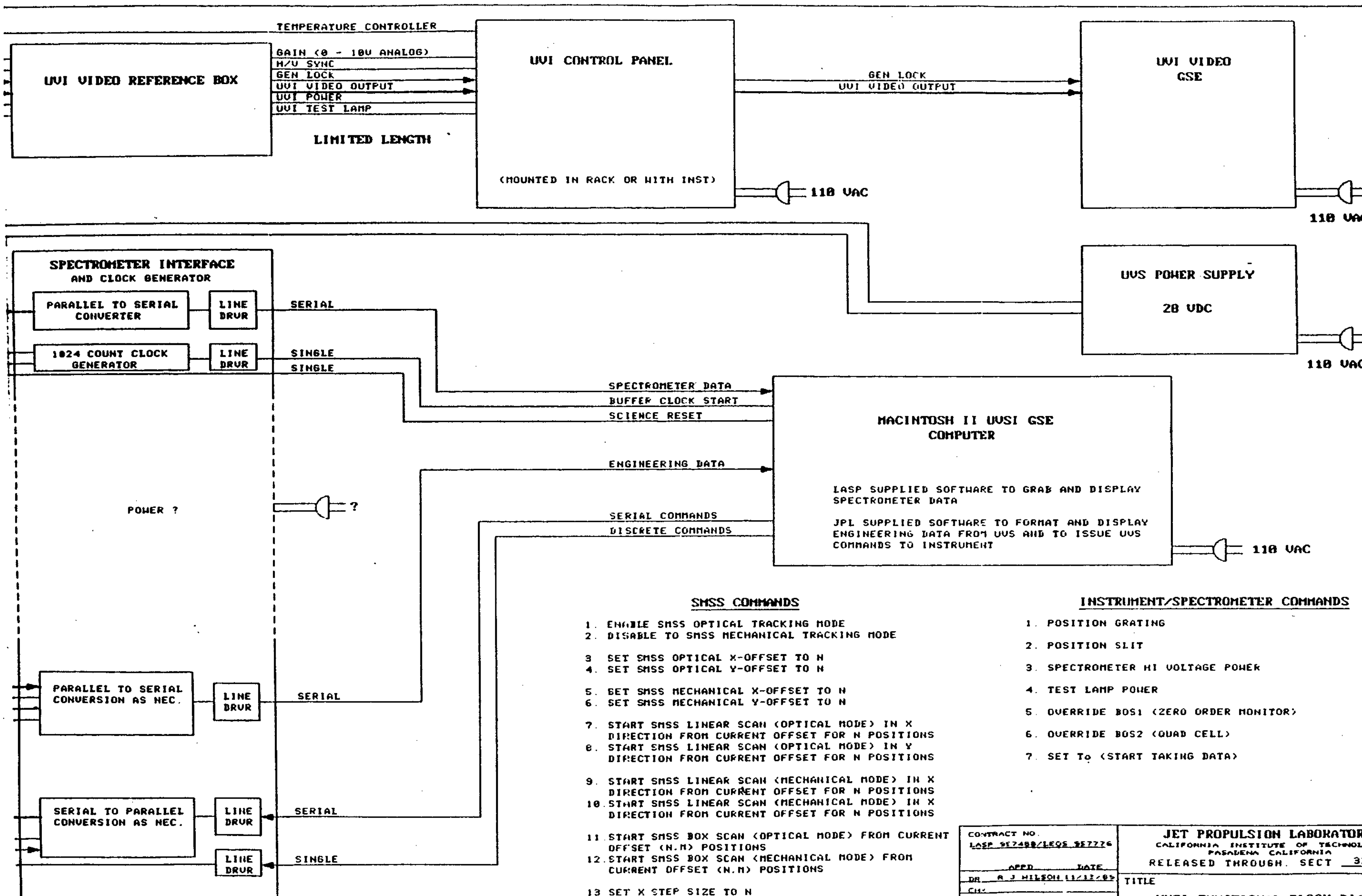
NO. OF PIXELS	754 (HORIZONTAL) X 488 (VERTICAL); ARRAY IS 8.8 X 6.6 mm
FIELD OF VIEW	4 X 3 MILLIRADIANS
PIXEL GEOMETRICAL FOOTPRINT	1.0 ARC SECOND, EQUIVALENT TO 1 METER AT 200 KM
SPECTRAL REGION	PRESENTLY 250-300 NANOMETERS; CAN BE EXPANDED
SYSTEM RESOLUTION	1.3 ARC SECOND (FULL WIDTH HALF MAX OF POINT SOURCE IMAGE); $>$ 20 CYCLES/MM WITH 3-BAR RESOLUTION TARGET
UVI OPTICAL TRANSMISSION	7%
SENSITIVITY	NOISE EQUIVALENT INPUT OF: $\sim 3 \times 10^{-17}$ WATT/SQ CM AT THE TELESCOPE APERTURE $\sim 3 \times 10^{-15}$ WATTS IN A FOCUSED POINT SOURCE IMAGE

TECHNICAL DESCRIPTION



2.4 METER TELESCOPE ASSEMBLY WITH BEAMSPLITTER
IMAGING OPTICAL BENCH ASSEMBLY
1/4 METER SPECTROMETER ASSEMBLY
SINGLE CYCLE EXTERNALLY OPERATED BLOW AWAY COVER(DIV 35)
SECONDARY MIRROR GIMBAL ASSEMBLY
KAMAN POSITION SENSORS
UVS/UVI BEAMSPLITTER
UVS/TRACKER BEAMSPLITTER
BRIGHT OBJECT SENSOR
2.9 X MAGNIFICATION TRANSFER OPTICS
DIAGONAL FOLD MIRROR
U.V. BANDPASS OPTICAL FILTER
UVI TEST LAMP
UVI CAMERA HEAD
UVI CAMERA HEAD ELECTRONICS
MCP POWER
MCP GAIN
UVI INTENSIFIER HIGH VOLTAGE POWER SUPPLY
TWO INCH TWO WAY PRESSURE RELIEF VALVE
NITROGEN GAS PURGE FITTING
SMSS DRIVE
SMSS POSITION
SMSS TEST LAMP
SMSS DETECTOR
-2500V
-1800V
-100V
DETECTOR HVPS DIVIDER
DETECTOR HIGH VOLTAGE P.S
AGC
+28VDC
SMSE QUAD ELECTRONICS
UVS TEST LAMP
SLIT SELECTOR WITH SOLENOID DRIVE
EBERT-FASTIE MONOCHROMATOR WITH 2 POSITION GRATING ASSY
ZERO ORDER SENSOR
ION VACUUM PUMP
ION PUMP HIGH VOLTAGE POWER SUPPLY
ION PUMP DIRECT ACCESS/UMBI CONNECTOR
CODACON DETECTOR
CODACON DETECTOR CHARGE AMPLIFIERS AND SHAPING ELECTRONICS
DETECTOR HIGH VOLTAGE P.S.
SPECTROMETER LOW VOLTAGES
TO ENGINEERING DATA
UVS SPECTROMETER TEST CONNECTIONS
UVS μ-P TEST CONNECTIONS


UVI DIRECT ACCESS CONNECTIONS
ITEM CONNECTOR/PIN NO FUNCTION
1 J2-1 VIDEO OUT
2 J2-2 VIDEO RETURN
3 J2-3 CHASSIS GROUND
4 J2-4 SIGNAL COMMON
5 J2-5 AMP TEMP SENSE
6 J2-6 S DRIVE TEMP SENSE
7 J2-7 VID CLP TEMP SENSE
8 J2-8 MCP TEMP SENSE
9 J2-9 CCD TEMP SENSE
UVI DIRECT ACCESS CONNECTOR
GAIN (0 - 10V ANALOG)
H/V SYNC
GEN LOCK
UVI VIDEO OUTPUT
UVI POWER
UVI TEST LAMP
ION PUMP +28 VOLTS
UVS +28 VOLTS
LIMITED LENGTH
TEMPERATURE CONTROLLER
GAIN
H/V SYNC
GEN LOCK
UVI VIDEO OUTPUT
UVI POWER
UVI TEST LAMP
UVI DIRECT ACCESS CONNECTOR
DUAL INTEGRATION MEMORY
(2 X 1024 X 8)
PORT 1
AMPLIFIER INTERFACE
PORT 2
IFE INTERFACE
8
LIMITED LENGTH
CONTROL LOGIC
SCIENCE ENABLE
SCIENCE CLOCK
SCIENCE RESET
GRATING DRIVE ELECT.
+/-15V, +/-10V, +5V
SPECTROMETER LOW VOLTAGE POWER SUPPLY
SLIT SELECTOR ELECTRONICS
LASP EXTERNAL ELECTRONICS PACKAGE #1
POSITION
DRIVE
DET
AGC
SMSS CONTROL ELECTRONICS
SMSS POSITION ELECTRONICS
+5, +/-15
SMSS LVPS
LASP EXTERNAL ELECTRONICS PACKAGE #2
MICROPROCESSOR NSC-800
PROGRAMMABLE TIMERS
MICROPROCESSOR LOW VOLTAGE POWER SUPPLY
SYSTEM BUS
SYSTEM BUS
MEMORY EPROM/RAM
32K POSSIBLE 2K BLOCKS
+/-15V, +10V, +5V
GO/NO GO TEST LAMPS
SMSE CONTROL AND STATUS
SMSE TRACKER POSITION DATA
24
GRATING DRIVE CONTROL
BRIGHT OBJECT SENSOR
PROGRAMMABLE I/O
2 KILOBIT ENG DATA
2 KILOBIT CLOCK
2 KILOBIT ENABLE
2 KILOBIT RESET
LIMITED LENGTH
16
4 ENTRANCE SLIT
40 CHANNEL MUX
12 BIT A/D CONVERTER
PRESCALED ANALOG ENGINEERING DATA
UVSI COMMANDS
DATA AVAILABLE
DATA ACK
LIMITED LENGTH
16
TRACKER STATUS
LASP EXTERNAL ELECTRONICS PACKAGE #3



SMSS COMMANDS

1. ENABLE SMSS OPTICAL TRACKING MODE
2. DISABLE TO SMSS MECHANICAL TRACKING MODE

3. SET SMSS OPTICAL X-OFFSET TO N
4. SET SMSS OPTICAL Y-OFFSET TO N

5. SET SMSS MECHANICAL X-OFFSET TO N
6. SET SMSS MECHANICAL Y-OFFSET TO N

7. START SMSS LINEAR SCAN (OPTICAL MODE) IN X DIRECTION FROM CURRENT OFFSET FOR N POSITIONS
8. START SMSS LINEAR SCAN (OPTICAL MODE) IN Y DIRECTION FROM CURRENT OFFSET FOR N POSITIONS

9. START SMSS LINEAR SCAN (MECHANICAL MODE) IN X DIRECTION FROM CURRENT OFFSET FOR N POSITIONS
10. START SMSS LINEAR SCAN (MECHANICAL MODE) IN X DIRECTION FROM CURRENT OFFSET FOR N POSITIONS

11. START SMSS BOX SCAN (OPTICAL MODE) FROM CURRENT OFFSET (N,M) POSITIONS
12. START SMSS BOX SCAN (MECHANICAL MODE) FROM CURRENT OFFSET (N,M) POSITIONS

13. SET X STEP SIZE TO N
14. SET Y STEP SIZE TO N

15. SET INTEGRATION PERIOD BETWEEN STEPS TO N
16. OVERRIDE TRACKER ACQUISITION SIGNAL

INSTRUMENT/SPECTROMETER COMMANDS

1. POSITION GRATING
2. POSITION SLIT
3. SPECTROMETER HI VOLTAGE POWER
4. TEST LAMP POWER
5. OVERRIDE BOS1 (ZERO ORDER MONITOR)
6. OVERRIDE BOS2 (QUAD CELL)
7. SET To (START TAKING DATA)

CONTRACT NO. LASP 957400/LEQS 957776	JET PROPULSION LABORATORY CALIFORNIA INSTITUTE OF TECHNOLOGY PASADENA CALIFORNIA RELEASED THROUGH SECT 32E0
APPD DATE	TITLE
DR A J WILSON 11/12/85	UVSI FUNCTIONAL BLOCK DIAGRAM
CHK	SIZE
STAN	CODE IDENT TEMPOPBK
MATL	0001
TC	REV A
PMA	SCALE NONE
ENGR	SHEET 1 OF 1
DSGN A J WILSON	

UVSI POWER (WATTS)
(PRELIMINARY VALUES ONLY)

	CRUISE	W/UP	OPERATE
UVI	0	8	8/10
INTERFACE	0	TBD	TBD
UVI TOTAL	0	TBD	TBD
UVS ION PUMP	5/25	5/25	5
UVS SPECTROMETER	0	0	5
UVS SECONDARY	0	0	15
UVS u-PROCESSOR	0	0	2
UVS TOTAL	5/25	5/25	27
UVSI TOTAL	5/25	TBD	TBD

UVSI WEIGHT (POUNDS)
(PRELIMINARY VALUES ONLY)

UVS 305 lbs. (138.3 Kgs)

DOOR 5 lbs. (2.3 Kgs)
Blow Away Cover

IMAGER & URB 14 lbs. +/- 2.0 lbs (6.4 Kgs)

INTERFACE BOX 26 lbs. +/- 10 lbs (11.8 Kgs)

UVSI TOTAL - 350 lbs (159.1 Kgs)

SIZE

UVSI LENGTH - 59.5 INCHES + DOOR
WIDTH - 23 INCHES

INTERFACE BOX
LENGTH - 20 INCHES
MAX WIDTH - 8 INCHES
HEIGHT - TBD INCHES

UVSI COMMANDS AND FORMAT
(PRELIMINARY VALUES ONLY)

CDH
DATA RATE - TBD
CLK RATE - TBD
INTERFACE - TBD

UVS	UVI
OPMODE 1	PWR ON
OPMODE 2	PWR OFF
NADIR	HV ON
LIMB	HV OFF
PWR ON	TESTLAMP ON
PWR OFF	TESTLAMP OFF
HV ON-S	GAINMODE 1
HV OFF S	GAINMODE 2
HV ON SMSE	GAINMODE 3
HV OFF SMSE	GAINMODE 4
TESTLAMP ON	SPARE
TESTLAMP OFF	SPARE
SPARE	SPARE
SPARE	SPARE
SPARE	SPARE
SPARE	SPARE

PWR SW Unit (PSU)

UVS	UVI
IONPUMP ON	RELAY HEATER ON
IONPUMP OFF	RELAY HEATER OFF
HEATERS ON	
HEATERS OFF	

UVSI DATA AND FORMAT
(PRELIMINARY VALUES ONLY)

DMC
DATA RATE - TBD
CLK RATE - TBD
INTERFACE - TBD

UVS (ENGINEERING DATA)	UVI
VOLTAGE(7)	VOLTAGE(2)
TEMP(14)	TEMP(6)
PRESSURE(2)	GAIN STATE
SMSE(11)	VIDEO LEVEL

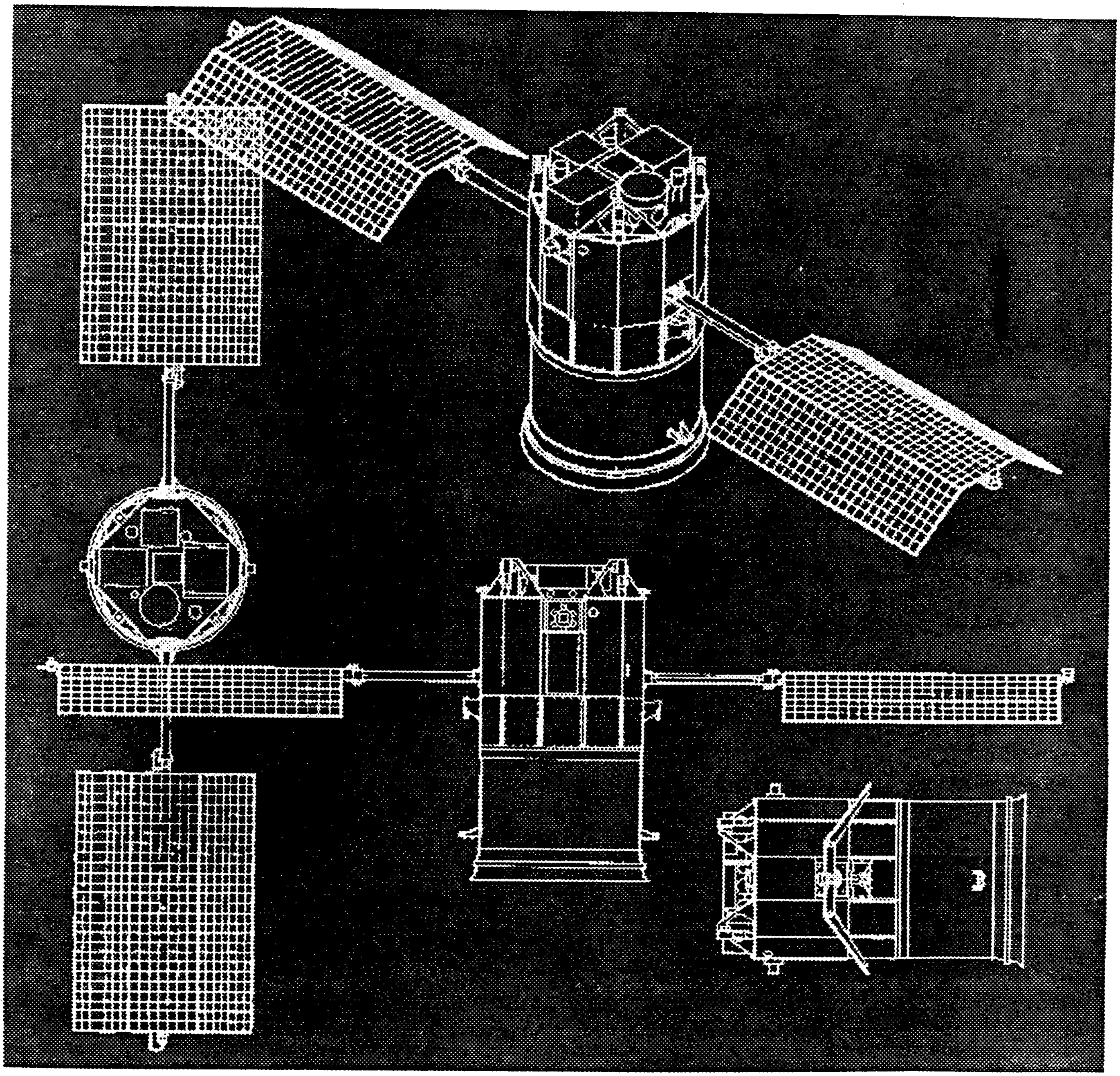
Appendix 3.3-C

Complete Document
Available 9 Dec 91

TRW Space & Technology Group

A User's Guide

E A G L E   C L A S S



L I G H T S A T S