MISSION DESIGN AND OPERATIONS: SPARTAN-HALLEY AS A PARADIGM FOR SMALL SATELLITES

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ABSTRACT

The SPARTAN reusable spacecraft opened a new arena for innovative scientific and engineering experimentation in low earth orbit. SPARTAN is a free flying, 3-axis inertially stabilized spacecraft which is deployed and retrieved by the Space Shuttle. The spacecraft is unique in that it performs all scientific observations as well as its guidance, navigation, and control functions completely autonomously.

This paper will describe the innovative approach that was taken in the design of the SPARTAN guidance and navigation system for a 48-hour observation of Halley's Comet near perihelion passage. Autonomous vehicle navigation techniques that utilized solar sensors, a single star tracker, a 3-axis gyro package, and two independent cooperative microprocessors will be discussed. Ephemeris development and structure will be detailed. Creative techniques used to "tune" the navigation and science observations with respect to the actual achieved mission orbit will also be described. Text will also be devoted to describing the vehicle and its sensor and instrument configuration. Lessons learned from SPARTAN will be discussed in the context of small satellite mission operations.

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INTRODUCTION

The SPARTAN-HALLEY vehicle was the only dedicated United States spacecraft designed to observe Halley's Comet. The mission was undertaken to provide unique spectral observations of Halley's Comet near the time of perihelion passage. The purpose of these observations was to determine both the chemical structure of the comet and the distribution of gases throughout the coma and the tail. The specific scientific objectives of the mission were to determine the abundances of water, carbon monoxide, carbon dioxide, and sulfur evolving from the comet.

Perihelion observations of comets are important because the level of heating, and hence cometary activity is greatest at this time (see Figure 6). Making observations of Halley, however, were greatly complicated in 1986 by unfavorable observing geometry.

Unlike most past apparitions, Halley's 1986 return was poorly placed for Earth-based observations. This was because the comet's perihelion occurred when the comet was almost directly behind the sun as seen from Earth (see Figure 1). This observing geometry coupled with the comet's low brightness relative to the sun, made ground-based and conventional earth-orbiting observations difficult or impossible at perihelion. SPARTAN-HALLEY was designed to overcome this dilemma. It's mission presented a demanding challenge to both the guidance and control system engineers and the instrument designers.

The SPARTAN-HALLEY mission team was formed only 18 months prior to launch. This aggressive schedule dictated the utilization of existing designs and hardware whenever possible, and forced the engineering team to compromise between the optimum scientific observation techniques and the technical realities of providing a realizable response. Such a paradigm is likely to be useful for future small satellite programs.

The SPARTAN Program (located at NASA/GSFC) builds low-cost, single string spacecraft that bridge the gap between sounding rockets and long duration satellites. This is accomplished by adapting already-proven designs and integrating them into vehicles that are launched, deployed, and retrieved by the Space Shuttle. Orbiter interfaces are kept to a bare minimum. All free-flying operations are conducted autonomously with no interaction from the ground or the STS. The first SPARTAN spacecraft was successfully flown in June 1985, aboard Shuttle Flight 51-G. SPARTAN-HALLEY was the second SPARTAN spacecraft.

SPACECRAFT DESCRIPTION

The SPARTAN-HALLEY spacecraft is depicted in Figures 2 and 3. The free-flying spacecraft bus is attached to the SPARTAN Flight Support Structure (SFSS) bridge during launch and landing. The SFSS is a SPARTAN-unique version of the STS-Standard MPESS, or Mission Peculiar Equipment Support Structure. During free flight, the detached SPARTAN operates as an autonomous, 3-axis stabilized platform. Shown in the illustrations are the service module, the flat top pneumatics deck, and the instrument optical bench. The service module is the core structure of the spacecraft. It houses the batteries, the attitude control system electronics, and the data handling electronics. The flat top pneumatics deck supports the cold gas (argon) system that is used for attitude control. This deck also provides the structural interface for attaching the instrument optical bench to the service module. SPARTAN weighs ~ 2600 pounds in its free-flight configuration.

The instrument optical bench is mission unique. It is kinematically mounted to the flat top so that structural deformations of the main vehicle do not distort its shape. In the case of SPARTAN-HALLEY, the optical bench supported two ultraviolet (UV) spectrometers, two aspect cameras, the TRIG gyros, a star tracker, a solar optical baffle, and a solar optical bench. The solar optical bench is rigidly attached to the instrument optical bench and supports the intermediate and fine sun sensors at a point where the sunshade is outside of their field of view. The Shuttle grapple fixture is attached to the service module in such a manner that reflected light cannot enter the optical path of the instruments, or reflect on the sunshade. When the comet is near the sun, Halley appears 10^{12} times dimmer than the sun. The sunshade (solar baffle) was designed to provide shadowing of the spectrometers so that the sun is occulted when the instruments are viewing the comet. The innovative, one-sided sunshade was designed to accommodate comet-sun angles of as little as 10 degrees, and provides a factor 10^6 light rejection capability. The leading edge of the sunshade consists of a highly polished, optical quality, nickel plated tube. A departure from knife edge baffling was necessary because of the sharp edge constraints imposed by STS contingency astronaut EVA requirements. Deployable sunshade covers and actuated doors were considered, but rejected because of reliability and complexity concerns. Additional light baffles were placed within the sunshade to trap and attenuate Earth albedo.

The scientific instruments were NASA/JPL/CU Mariner heritage with a proven internal baffling system that provides an additional factor of 10^6 rejection of off-axis light over that provided by the sunshade. The instruments consisted of two ultraviolet spectrometers, each with its own telescope and detector. The optical system of each spectrometer was of Ebert-Fastie folded optics design with a spherical mirror and plane diffraction grating. The spectrometer data rates were set to 6.2 Kbps per spectrometer to match the SPARTAN carrier's telemetry capabilities. This provided for a full spectrum (1200-3200Å) of the comet at 2Å resolution every 1.6 seconds. Spacecraft engineering data and instrument data were stored on tape, since SPARTAN has no uplink or downlink capability.

The SPARTAN attitude control system (ACS) is a hybrid analog/digital derivative of the STRAP V System developed by NASA/GSFC for sounding rocket attitude control. It is capable of stabilized pointing at any target within a deadband of ±10 arc-secs. The ACS utilizes a sequencer to implement the execution of pre-programmed events. The sequencer has a 2304 bit recirculating register CMOS memory. A sequencer program can accommodate up to 61 program steps, and provide up to 57 timed intervals. There are two sets of outputs: Primary and the Secondary. Primary Outputs generate +28 volt signals, and are used mainly for driving the electronics used for gyro support, as well as certain synchronization signals with the computer, sensor interface box, and star tracker. Secondary Outputs generate +5 volt logic signals and drive only the computer, which interprets the logic voltage as bit patterns and executes the resultant instructions.

Although this sequencer has a limited decision-making ability, it can be programmed with loops and conditional jumps out of loops as well as infinite time program steps that are interrupted by external signals. The SPARTAN ACS computer then acts as an executive to the sequencer which stores and loads successive sequencer programs. The order of program execution is usually determined by lookup tables stored (in triplicate) in the computer. The sequencer implements maneuvers and provides long-term timing for the computer. Since the ACS was based around this sequencer, all activities were preprogrammed and of fixed duration. A ruggedized version of the Z80 S+50 standard bus microcomputer manages the ACS and provides the extended memory for the flight sequencer.

Attitude references are sensed by three Teledyne SDG4 two-degree-of-freedom, tuned restraint, inertial gyros (TRIGS); one Ball Brothers CT 201 type star tracker, and coarse, intermediate, and fine analog solar sensors. The trimmed gyro drift rate in orbit is approximately 6 arc-sec/min. Because we chose to use a sun-centered reference frame for comet target-capture, the SPARTAN-HALLEY mission required the addition of a 2-axis, ±32 degree field of view (FOV) digital sun sensor. This sensor had a 1 arc-min accuracy with a resolution of 14.7 arcsecs. This particular (Adcole) unit was chosen because its FOV matched the expected range of comet-sun angles, and because the digital output was amenable to easy offsetting with stored ephemeris data.

The Experiment Microprocessor was used to direct the flight maneuvers. It was based on the National Semiconductor NSC-800 8-bit CMOS processor and contained redundant banks of 16K EPROM for program and ephemeris storage, 512 byte of RAM (4 chips of 128 bytes each), 8 programmable timers, approximately 160 hardware I/O lines,and a watchdog timer with hardware EPROM bank switching and interrupt capability. Of the four RAM chips, one was used to store the microprocessor PCM telemetry buffer. The remaining three chips were used as memory to maintain multiple copies of the program variables. The stack was also stored in the RAMS. Each RAM was divided in such a manner that the bottom 64 positions contained the scratchpad data and the top 64 contained the stack. By definition the stack cannot be stored in triplicate; therefore two of the three scratchpad RAM chips did not utilize the bottom 64 variables; however, on microprocessor powerup (or warm-start), a self-test is run to choose the "healthiest" RAM for stack placement.

The ACS Microcomputer was linked to the Experiment Microprocessor via two unidirectional digital communication busses. This communications arrangement was chosen to minimize modifications to the existing SPARTAN system.

The Experiment Microprocessor took the inputs from the digital sun sensor and converted them to the required analog control signals which were then passed to the ACS computer along the comm busses. This arrangement was chosen because these were all mission-unique functions and were not within the capability of the existing SPARTAN system, whereas the Experiment Microprocessor was new and could easily accommodate these functions. The decision to split the microprocessors and their functions worked well because (1) it minimized changes to the basic SPARTAN System; (2) it provided a distinct division of responsibility; and (3) it fully integrated the payload engineering organization (at the University of Colorado) into mission design and planning activities.

Pre-deployment checkout, ephemeris initiation, and entry of pre-deployment data to the Experiment Microprocessor were accomplished by the use of the NASA Get-Away-Special (GAS) Autonomous Payload Controller (APC). The APC is a hand-held controller that an astronaut uses to activate/deactivate and interrogate a series of control relays within the SPARTAN spacecraft. Relay positions are read as bit states to effect information transfer.

NAVIGATION PLAN

A solar-centered azimuth/elevation (i.e., clock and cone) type of reference frame was chosen for this mission. The position of the sun provided the elevation origin; the azimuth origin was defined by the star Canopus. This navigation plan was chosen for several reasons, the most dominant being that comet motion is naturally sun-centered. Canopus was chosen as a guide star because it is quite bright and hence easily recognizable by the use of magnitude discrimination techniques. Also, the celestial geometry was such that Canopus was nearly orthogonal to the sun and visible at night throughout the launch window (see Figure 5). Indeed, stellar acquisition was greatly simplified by the fact that Canopus remained relatively stationary (see Figure 6) throughout January-March mission window.

By fixing the star tracker/solar sensor geometry to the "nominal" sun/Canopus angle, initial acquisition could be achieved by locating the sun, and then rolling about that line until the guide star Canopus was located. Vehicle deployment attitude errors could be compensated for by choosing the initial deployment coordinates such that the star tracker line-of-sight would be offset to one side of the star by slightly more than expected worst case deployment error. Sun centering the reference frame also gave the appearance of slowing down the pre-perihelion angular motion of comet Halley. This allowed the number of ephemeris data points to be reduced because the update step sizes were small enough to allow linear interpolation between points. The sun centered coordinate frame also provide commonality between the geometric and experiment viewing (lighting) references used for this mission.

MISSION STRATEGY

Due to the dimness of the comet and its angular proximity to the sun, we chose to point the instruments toward the comet location based on a stored on-board ephemeris, that is, open-loop. Our ephemeris was built from data compiled by JPL. The ephemeris had to be stored in EPROM memory 5 months prior to launch because of schedule constraints. This was risky because the comet trajectory could change (due to highly asymmetric gas venting) over this period. Therefore, potential dispersions had to be taken into account when assembling the pointing error budget. To compensate for this, we decided to overscan the comet's expected position.

The ephemerides were stored in the azimuth/elevation coordinate frame discussed above. Spacecraft elevation angle changes were accomplished by biasing the output of the digital sun sensor. Azimuth angle changes were accomplished by the execution of combinations of pre-programmed gyro maneuvers. A detailed error analysis was conducted to determine the frequency at which the control system attitude reference frame had to be updated. This error budget included alignment errors, sensor drift errors (primarily the gyros), ephemeris uncertainty, and the rate of change of the comet position (see Figures 7 and 8). Since the navigation plan relied on specific source (sun and Canopus) sightings instead of targets of opportunity, there was significant science down-time associated with each attitude update. Because of this, we had to trade off the benefits of comet overscanning versus the lost viewing time associated with performing the update. Comet elevation changed rapidly. During integration, we optimized our situation by aligning errors in elevation with the narrow axis of the spectrometer slit (see Figure 4). Errors in elevation would quickly place the comet out of the instrument field of view. Therefore, it was decided to overscan in this axis by 30 arc-min and update the gyro null reference once per orbit. Fortunately, the comet tail is always pointed away from the sun and thus the same azimuth angle could be used for both coma and tail observations. Comet azimuth changed more gradually than elevation, except right at perihelion (see Figure 8). Since the comet elevation angles near perihelion were small, they were not significantly aggravated by azimuth positioning errors. Comet motion in azimuth could be compensated for by updating the desired azimuth position, via the ephemeris, at the time of the solar (elevation) update. This, however, would not compensate for gyro drift. Updating azimuth would require rolling around the sunline and using the star tracker to sight the star Canopus. This process was time consuming in flight, but simplified the spacecraft design, served to reduce costs, and contributed to mission success. Since the gyro drift was slow, and azimuth was aligned primarily with the long axis of the spectrometer FOV, we determined that azimuth updates could be done as much as five orbits apart.

At the time that we had to construct the observation sequence, we did not know the STS mission on which we would fly. Consequently, we did not know the exact orbit period, eccentricity, or nodal parameters which we would encounter. Certain science requirements dictated precise knowledge of the orbit sunrise and sunset conditions. We determined that this would be impossible with a fixed, pre-programmed observation sequence stored months before flight; therefore, we chose to place variable holds (waits) at 4 locations within each orbit observation sequence. These variable waits were designed to allow us to "sync up" on the actual flight orbit in real time, and to maintain such synchronization througout the deployed flight. Values for these holds were to be entered during in-flight powerup and microprocessor initialization by the Shuttle crew using the APC described above. The "day-waits" were at orbit mid-morning and orbit mid-afternoon, and the "night-waits" were at pre- and post-midnight (see Figure 12). The day-waits were used to synchronize the observations with respect to the orbit lighting conditions. The night-waits were used to match the observation sequence loop-time to the orbit period.

The optical bench configuration (see Figure 0) was designed so as to minimize the effect of nonoptimal and changing celestial geometry associated with this mission by hardware design rather than by complicating operational procedures. Mounting the spectrometers tilted 15 degrees from the solar sensor line of sight enabled us to extend the maximum elevation angle achievable from the 30 degree FOV of the sun sensor to the full 45 degrees. This allowed us to accommodate the full 60-day launch window with an existing sensor design. The star tracker was offset from the solar sensor line of sight by approximately 105 degrees in order to allow acquisition of Canopus without leaving the sunline. Since the launch window was so long, the guide star would migrate out of the FOV of the star tracker during post-perihelion. This problem was solved by having a per-perihelion setting and a post-perihelion setting for the tracker orientation, which could be changed at the launch pad.

The observation sequence (see Figure 10) was broken into four distinct segments that were apportioned into timeframes that represented one orbit. The Acquisition Segment (see Figure 11) provided for the initial vehicle attitude acquisition and the synchronizing of the observation profile with the orbit lighting conditions via the detection of sunset of the third orbit. The Science Segment (see Figure 12) contained all of the comet observation sequences. Notice that a solar update was performed every orbit near noon. The Stellar Update Segment (see Figure 13) was executed every fifth orbit (i.e., after 4 Science Segments) and consisted of both a solar update and a stellar (azimuth) attitude update. The final segment, Recovery (see Figure 14) served the purpose of preparing and positioning the SPARTAN for retrieval by the STS. The observation sequence segment pie-charts, if followed in a counter-clockwise direction, illustrate the major activities that would occur throughout the mission.

DESIGN IMPLEMENTATION

Ephemeris data were stored in 90 minute intervals and configured in the azimuth and elevation coordinate frame that was defined earlier. The elevation data represented a point in the celestial sphere which we set 30 arc-minutes further away from the sun than the true position of the comet nucleus. This offset had the effect of placing the spectrometer aperture on the coma/tail boundary such that useful data could be taken between scans. Further, all elevation ephemeris data were biased, to compensate for known solar sensor calibration errors. Elevation data was stored with a resolution of 14.7 arc-sec.

Following the nulling of the solar sensor on the sunline at solar acquisition, the ACS microprocessor would request the experiment microprocessor to add the appropriate solar bias (comet elevation) to the output of the digital sun sensor. The experiment processor would respond by (1) extracting the proper value from the ephemeris, (2) sum it with the solar sensor output, and (3) then notify the ACS microprocessor that the solar error signal was ready for tracking. The ACS would then null the summed signal, causing the spacecraft to maneuver to the comet location. When a new solar update was required, the ACS microprocessor would, in a similar fashion, request the experiment microprocessor to remove the solar bias from the sensor error signal. By nulling the resultant error signal, the spacecraft would then maneuver back to the sun-line.

The azimuth data represented the roll angle about the sunline from the guide star Canopus to the comet-sunline. Azimuth data were stored with a 1 arc-min resolution. The ephemeris data were utilized in the following fashion. Azimuth maneuvers were done while under gyro control and were accomplished by the application of a precise voltage over a specific duration of time to the summing junction of the torque rebalance loop of the gyros. The SPARTAN ACS was sequencer-driven, and thus could only execute a finite number of pre-calibrated pre-programmed maneuvers. Therefore, it was necessary to select the appropriate maneuver(s) from a menu that would place the spacecraft in the desired attitude. When azimuth maneuvering to the comet was required, the ACS microprocessor would request the experiment microprocessor to select the appropriate maneuver set based on the value of data stored ephemeris. The experiment microprocessor would select the ephemeris data, place it in the guidance (where we want to be) register, and then select from the menu (see Table 1) that maneuver which would most reduce the magnitude of the difference between the desired (guidance) azimuth angle and the current vehicle attitude. In order to achieve greater positional accuracy above what could be stored in the ephemeris, average values of the ephemeris data were computed and used for guidance if the time-of-update occurred near the mid-point between the current and next exact value stored. The current vehicle attitude was stored in the navigation (where we are) register and consisted of the summed total of all menu maneuvers that had been executed since the last update (via the star tracker and the guide star Canopus) of this axis inertial reference. This process was iterated nine times during each comet acquisition in order to converge on the correct comet azimuth with sufficient accuracy. In order to return to the guide star for re-acquisition and updating of the reference frame, the guidance register was set to zero and the maneuver selection process iterated until the star tracker was pointed towards Canopus. The update sequence was iterated only 6 times because the star tracker FOV was considerably larger than the spectrometer FOV; for this reason, the required open-loop positioning error could be significantly larger on return to Canopus.

To offset the low-cost hardware philosophy used on SPARTAN, the ephemeris processing software was designed (whenever possible) to compensate for potential hardware failures. Every EPROM (2K bytes) was compared with a checksum during initialization, and if an error was found, the microprocessor switched to its redundant. Upon microprocessor initialization, RAM was tested, and should any errors be found in one of the chips, the stack was moved to a different chip that had verified successfully. Run time variables were stored in triplicate (in three different chips) and each variable was voted on (and dynamically refreshed) whenever retrieved. The consensus value was nominally utilized. If all values were in disagreement, that value stored in the same RAM as the stack was chosen by default.

The ephemeris was stored in groups of eight bytes, with seven bytes containing data, and the eighth being a record checksum. If the record checksum did not match when a record was retrieved, then it was not used. Successive records were stored in an interleaved fashion between the four EPROMs that the ephemeris was stored in. Therefore, should a failure occur in one EPROM, only one out of four records would be lost, rather than losing all records in a contiguous subset (2 weeks) of the mission window. Finally, a watchdog timer was used to switch EPROM banks, interrupt the microprocessor, and vector to a recovery routine should a Single Event Upset or other anomaly cause the experiment microprocessor to crash. Each of these software design implementations contributed to our desire for "graceful degradation" in the event of an in-flight anomaly.

The Autonomous Payload Controller (APC) was used to enter the orbital element data (i.e., day and night waits), position the ephemeris pointer, and start the ephemeris clock. Unfortunately, no system for making data entries was included in the SPARTAN spacecraft's original design. Because of the tight schedule of this project, it was decided that there was not time to add a communications system to SPARTAN. By utilizing the APC, we were able to accomplish this requirement without the need of an rf system, however, because of the APC's primitive nature, we encountered several problems which a more sophisticated interface would have solved.

MAGNETIC CONTROL SYSTEM

Because of the nature of a SPARTAN mission, development costs needed to remain small, and the designs were not allowed to become redundant except where failures might cause a loss of the spacecraft. Hence, the SPARTAN spacecraft, including the ACS, is single string and provides little in the way of operational redundancy. Failure of electronics, pneumatic system components, or deficiencies in the mission software could result in loss of control of the spacecraft. Rapid tumbling of the vehicle would preclude recovery by the STS. In order to ensure recovery of the SPARTAN spacecraft, a rate-nulling Magnetic Control System (MCS) built by Ithaco was added to the spacecraft. It consisted of a 3-axis magnetometer, control electronics module, 3 orthogonal electromagnetic torquers, a separate wiring harness, and an independent battery. The control logic senses the rate at which the Earth's magnetic field vector is rotating with respect to the magnetometer axis. The resulting rate information is used to generate an error signal that regulates the current driven through the torquers. The resulting magnetic dipole interacts with the Earth's dipole and creates a stabilizing torque. This system utilized 90 Kpole-cm torquers and was capable of despinning the spacecraft from rates as high as 20 degrees per second.

The magnetic control system was completely independent of a primary vehicle systems for operation (and contained its own power supply), but relied on sensors within the primary subsystems for activation. It was enabled if (1) primary ACS pneumatic system reservoir pressure was low, (2) the primary spacecraft battery voltage was low, or (3) if the spacecraft body rates exceeded expected rates for more than 10 seconds. The steady-state spacecraft attitude that results from control by the MCS is with the vehicle rotating about its axis of maximum inertia and that axis aligned parallel to the orbit normal vector. Aerodynamic and gravity gradient torques will modulate this attitude to an extent governed by the spacecraft geometry. The spacecraft inertial rate is never nulled, but rather averages twice orbital rate due to the vehicle's orbital motion around the Earth's dipole. Twice orbital rate is sufficiently slow that STS retrieval by the RMS is possible. The use of a simple system such as this economically insured the recovery of the SPARTAN without adding the complexity of primary system redundancy.

LESSONS LEARNED

1) Multiple cooperative microprocessors can provide an exceptionally versatile system while minimizing cost and complexity. The mix of a single state processor and an asynchronous processor provides an exceptionally stable and suitable arrangement for missions of this type. A clearly defined authority chain must, however, still be established.

2) The use of too simple a data input device (such as the APC) is cumbersome. User-friendly, microprocessor based devices could be more easily utilized, offer less chance for operator error, and provide increased checkout and command capability.

3) Operational procedures, software code, and database accessibility by GSE during testing must be given high priority. The inclusion of these features from the start greatly aided testing of the SPARTAN-HALLEY spacecraft and ensured our ability to "make" the 18-month spacecraft design, fabrication and test schedule imposed on us.

4) Always allow for more computer memory space than you feel will be adequate. Code will grow to occupy all existing memory no matter what you do, so double the requirements set down by careful analysis. It is much easier to debug and correct flight software when the software team doesn't have to optimize code for every last byte of space.

5) Judicious minimization of STS interfaces greatly reduces the time and workload associated with JSC/KSC integration. The time and resources saved can be more effectively utilized on spacecraft and payload development.

6) Launch dates slip no matter how stringent the requirements! The decision to expand our launch window capability with more flexible software contributed significantly to SPARTAN-HALLEY becoming manifested by the STS.

7) Short development times, while risky, promote the development of innovative SYSTEM solutions as well as nurturing enthusiasm among the troops. Tight schedules also ameliorate the tendency toward excessively complex state-of-the-art devices when existing technology is sufficient. The small group, systems approach reduces program costs and subsystem testing time.

8) The use of the same technical team for mission conceptualization, spacecraft design and development, launch integration, and mission operations is tremendously efficient and effective. The technical cohesiveness that develops helps to cross-train engineers in differing skills and promote the development of well versed systems managers.

CONCLUSIONS

The SPARTAN-HALLEY effort was a phenomenal exercise in personal cooperations, initiative, communication, and technical innovation.

These factors were key to overcoming the enormous schedule pressures imposed by the mission. The resultant spacecraft would have demonstrated that useful space science can be accomplished without the need for extravagant budgets and long development programs. By taking advantage of existing systems we can accomplish much. Though the spacecraft was destroyed aboard Challenger, we feel that the design approach taken represents a clever and unique utilization of the SPARTAN spacecraft.

We have illustrated that very complex missions can be executed by a "back-to-the-basics" utilization of simple, inexpensive spacecraft, and that these missions can be flown within a timeframe that reflects adaptive mission priorities.

We pay tribute to our friends in the final crew of the Challenger whose dedication and personal efforts contributed greatly to the preparation of the mission. We also wish to acknowledge and thank the entire Halley team of the GSFC Special Payloads Division and the University of Colorado's Laboratory for Atmospheric and Space Physics for their untiring efforts, without which the spacecraft could never have been built, and the schedule never maintained. Ad astra per aspera!

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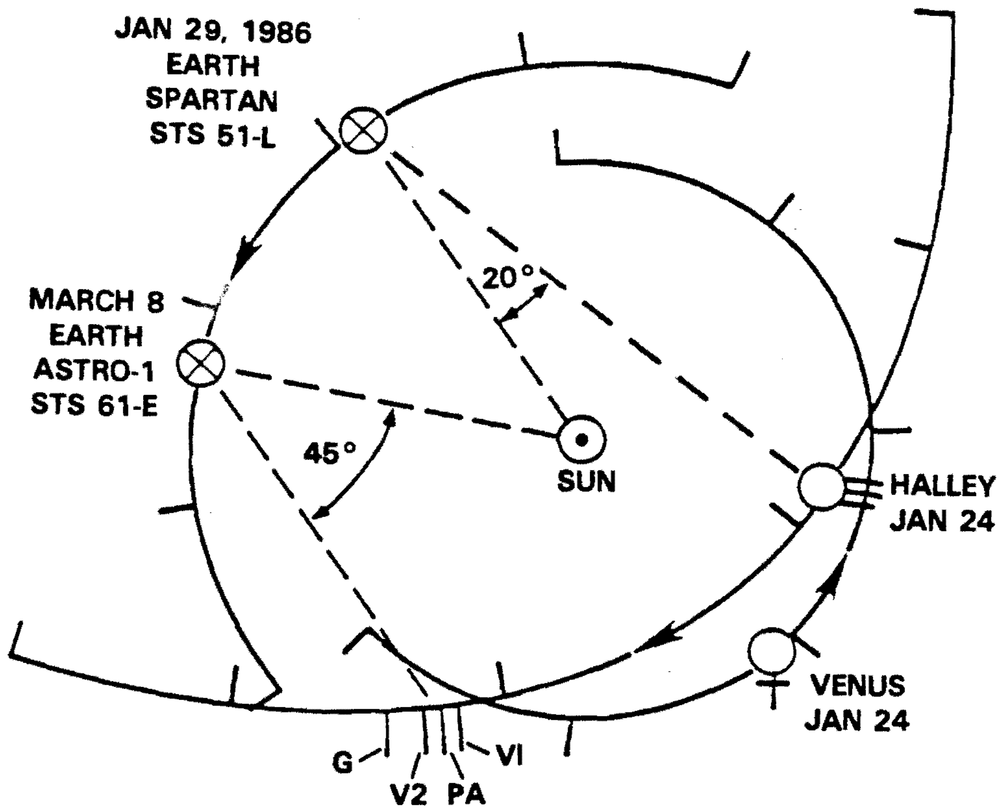
OBSERVATIONS OF COMET HALLEY



March 1986

SPACECRAFT/HALLEY ENCOUNTERS

V1 = VEGA 1 (RUSSIA)
PA = PLANET A (JAPAN)
V2 = VEGA 2 (RUSSIA)
G = GIOTTO (ESA)

Figure 1

SPARTAN PHYSICAL OVERVIEW

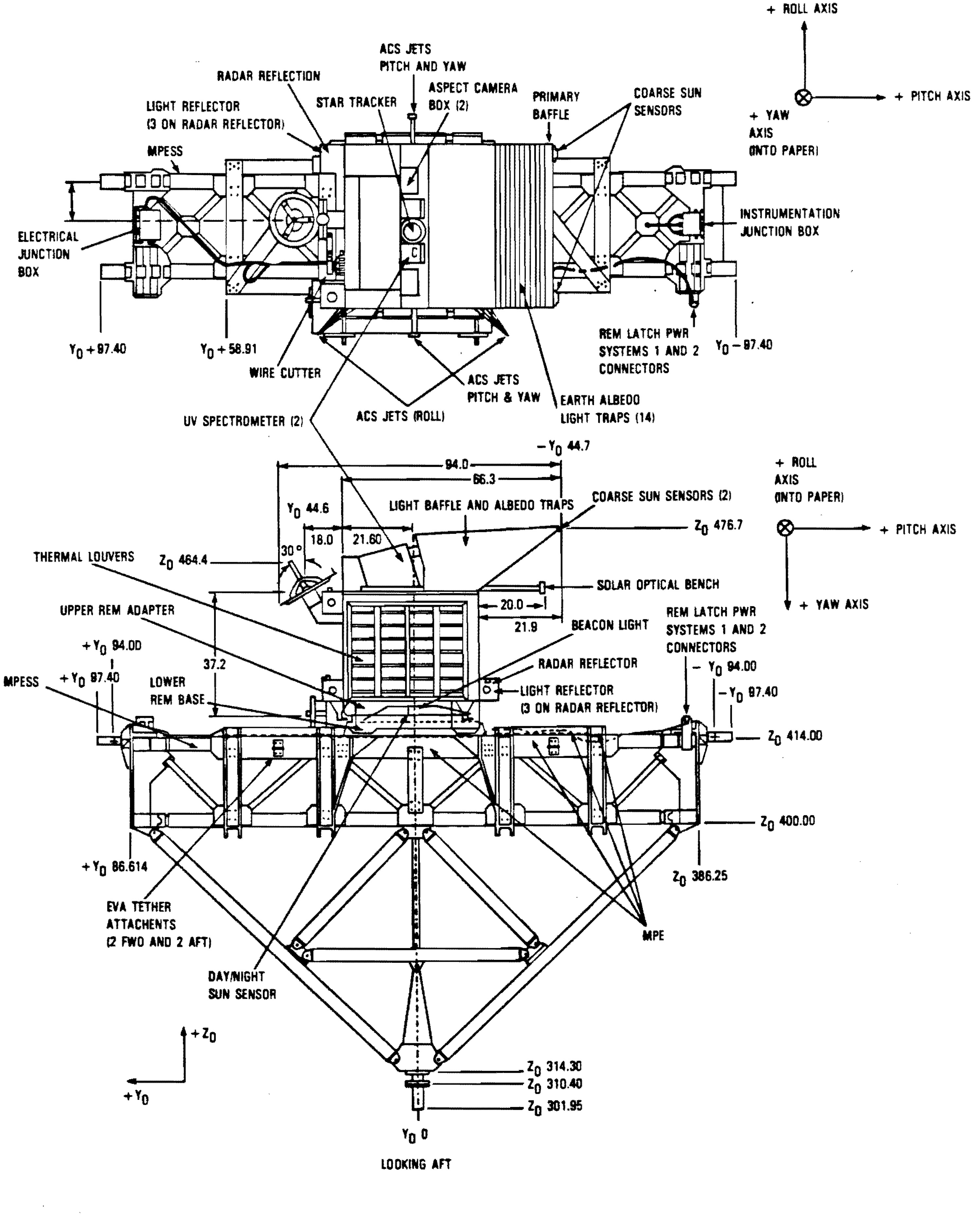


Figure 2

EXPANDED VIEW OF SPARTAN-HALLEY SPACECRAFT

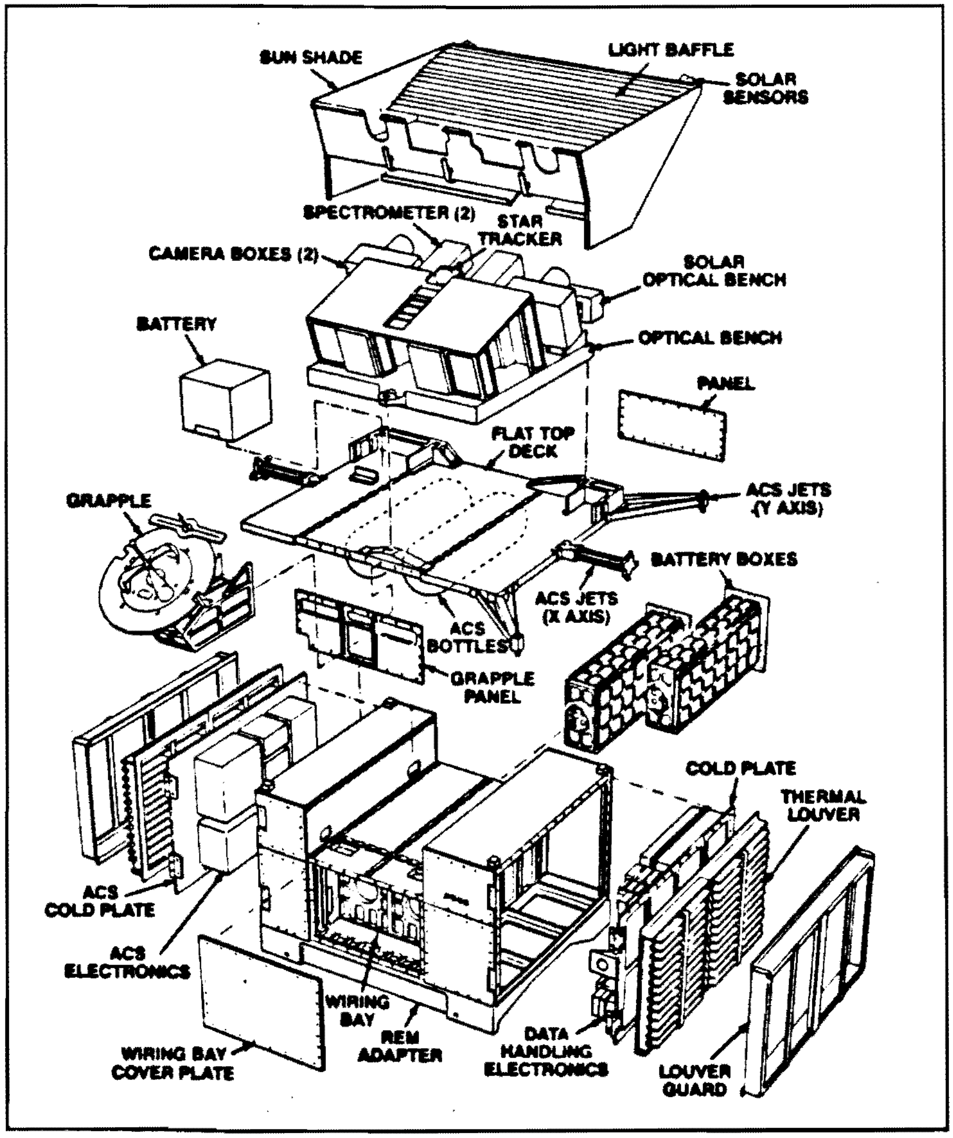


Figure 3

SPECTROMETER SLIT FIELD-OF-VIEW

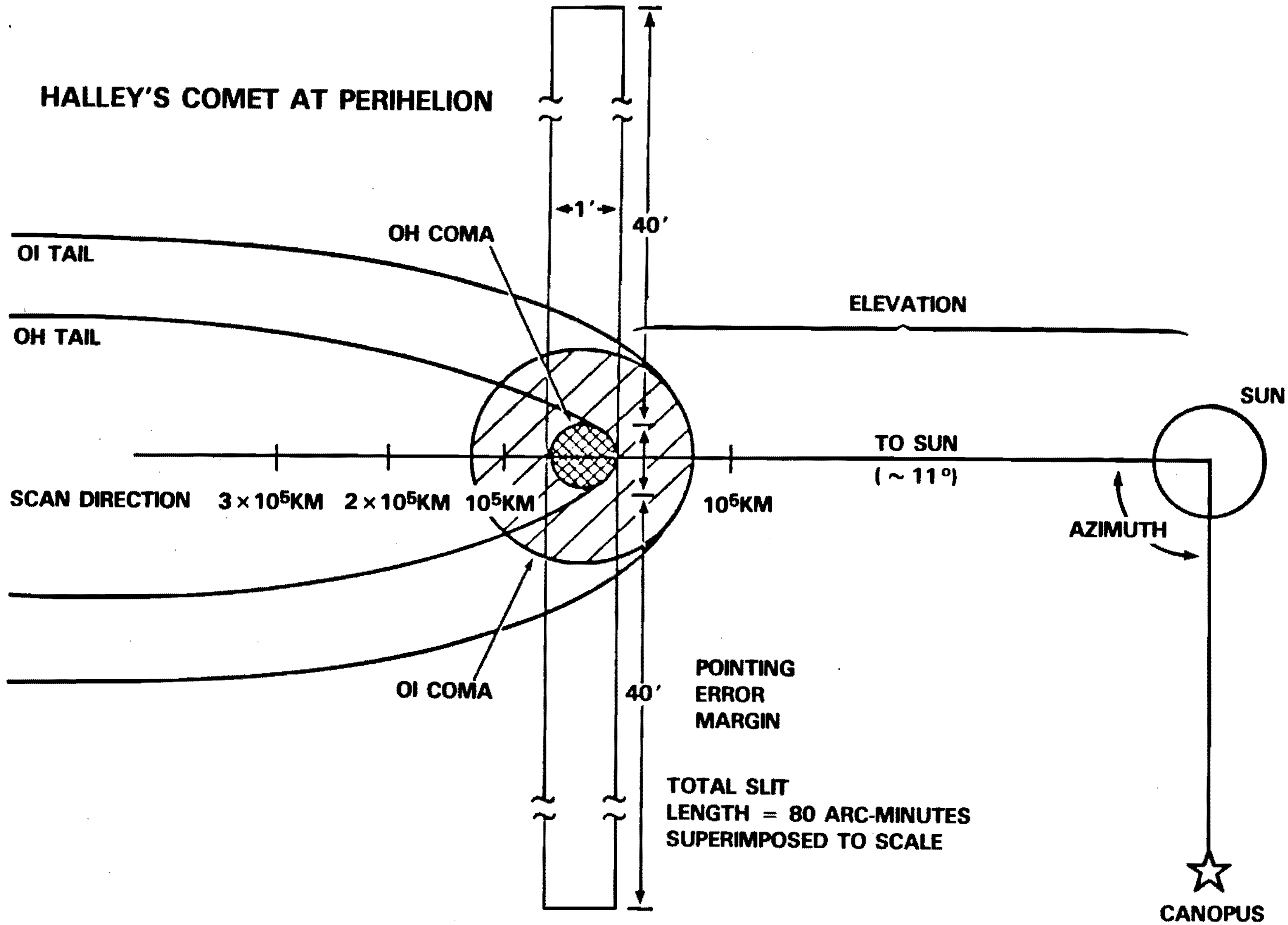


Figure 4

SPARTAN HALLEY LAUNCH PERIOD

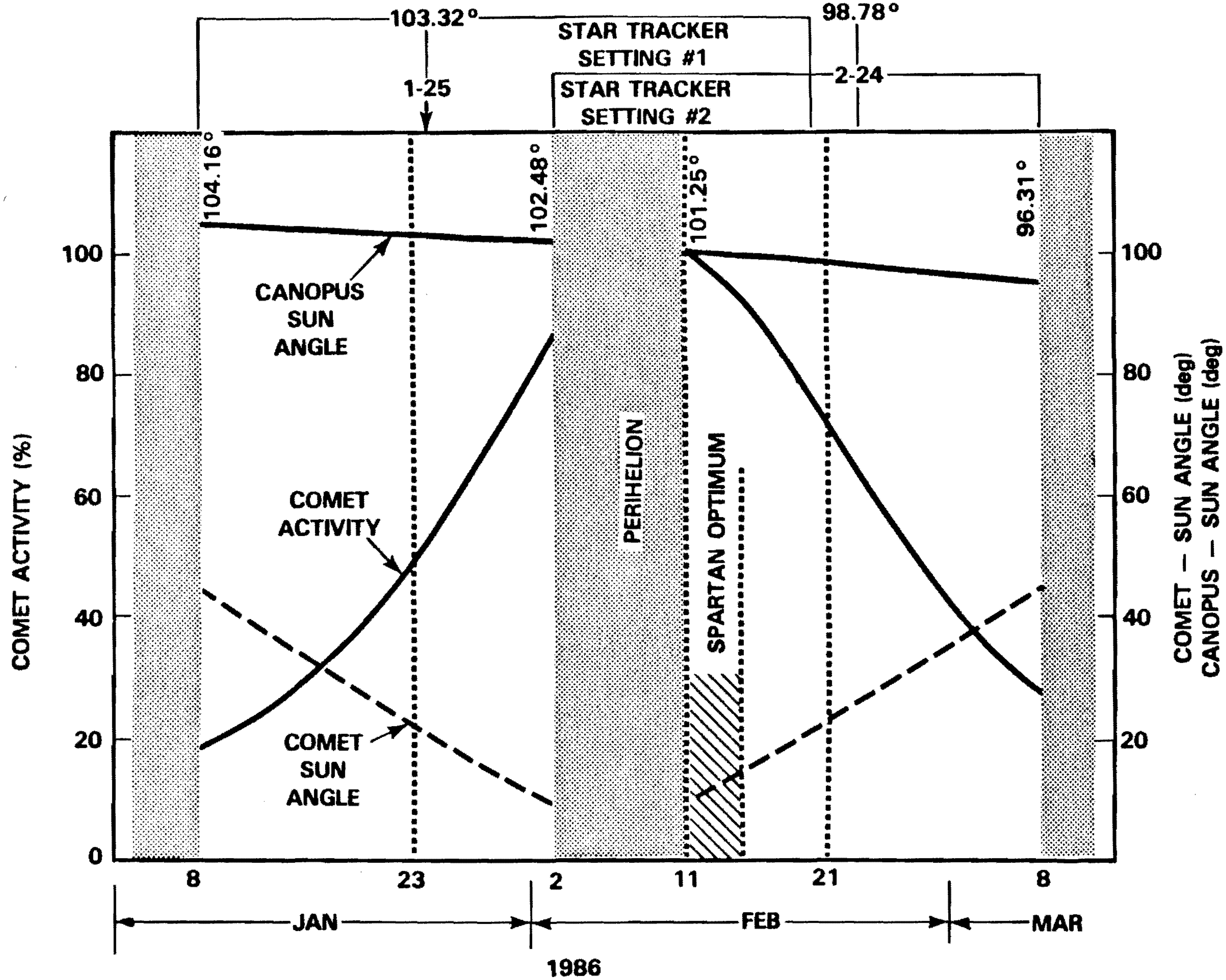
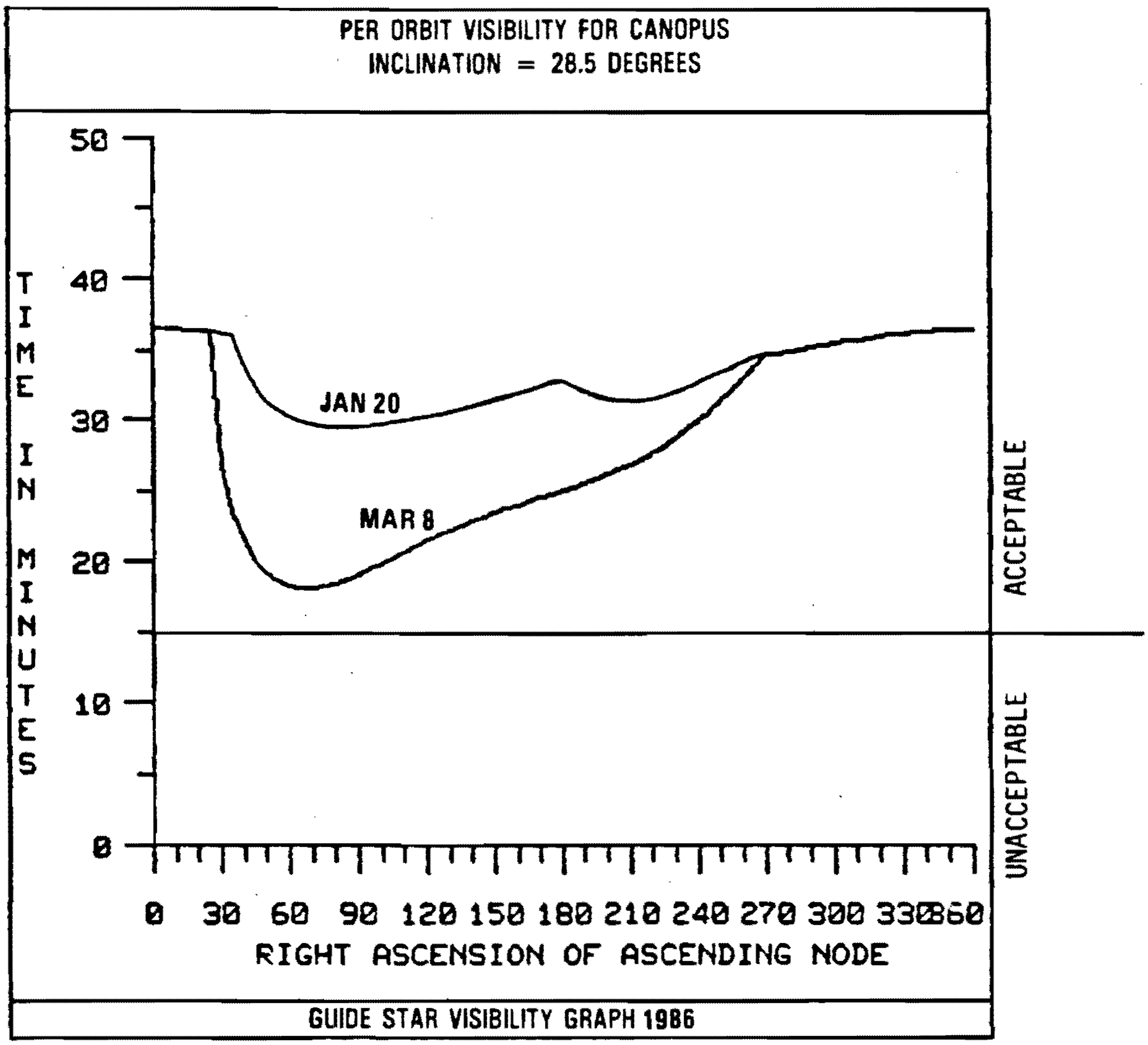


Figure 5



PER ORBIT VISIBILITY FOR CANOPUS
INCLINATION = 28.5 DEGREES
50
40
30
20
10
0
TIME IN MINUTES
JAN 20
MAR 8
ACCEPTABLE
UNACCEPTABLE
0 30 60 90 120 150 180 210 240 270 300 330 360
RIGHT ASCENSION OF ASCENDING NODE
GUIDE STAR VISIBILITY GRAPH 1986

Figure 6

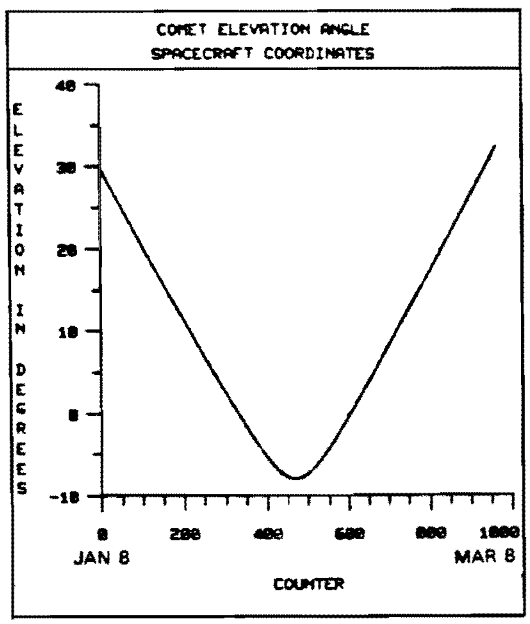

COMET ELEVATION ANGLE
SPACECRAFT COORDINATES
ELEVATION IN DEGREES
40
30
20
10
0
-10
0
200
400
600
800
1000
JAN 8
MAR 8
COUNTER

Figure 7

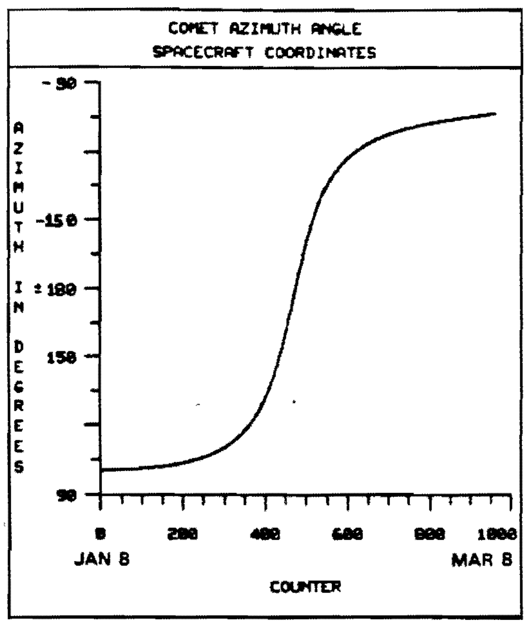
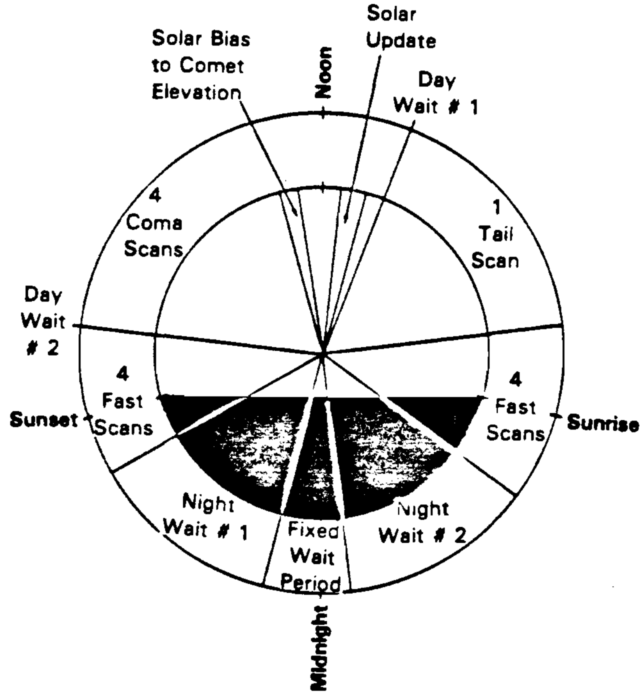

COMET AZIMUTH ANGLE
SPACECRAFT COORDINATES
-90
-150
±180
150
90
AZIMUTH IN DEGREES
0
200
400
600
800
1000
JAN 8
MAR 8
COUNTER

Figure 8

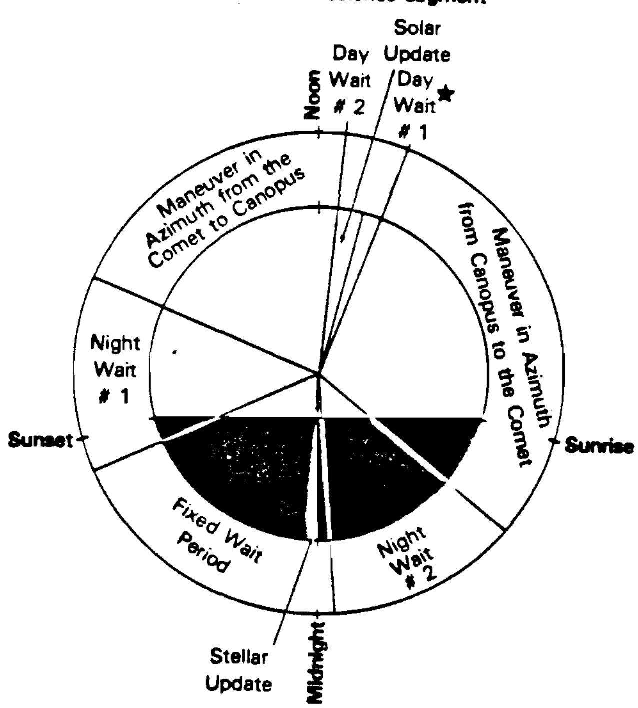
SCIENCE SEGMENT

REPEATS FOR TOTAL OF 4 ORBITS



3-AXIS NAV UPDATE SEGMENT

For 1 orbit after science segment



LAST ORBIT

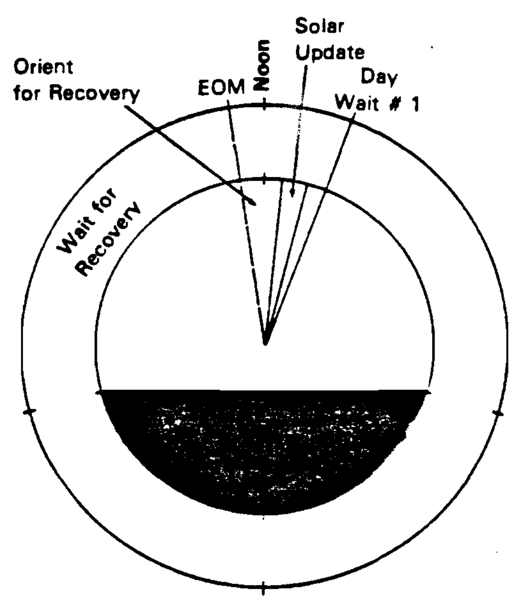
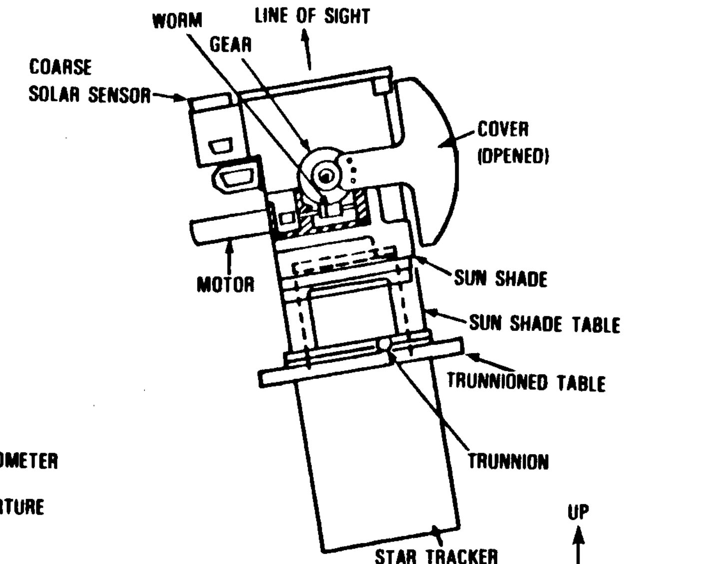
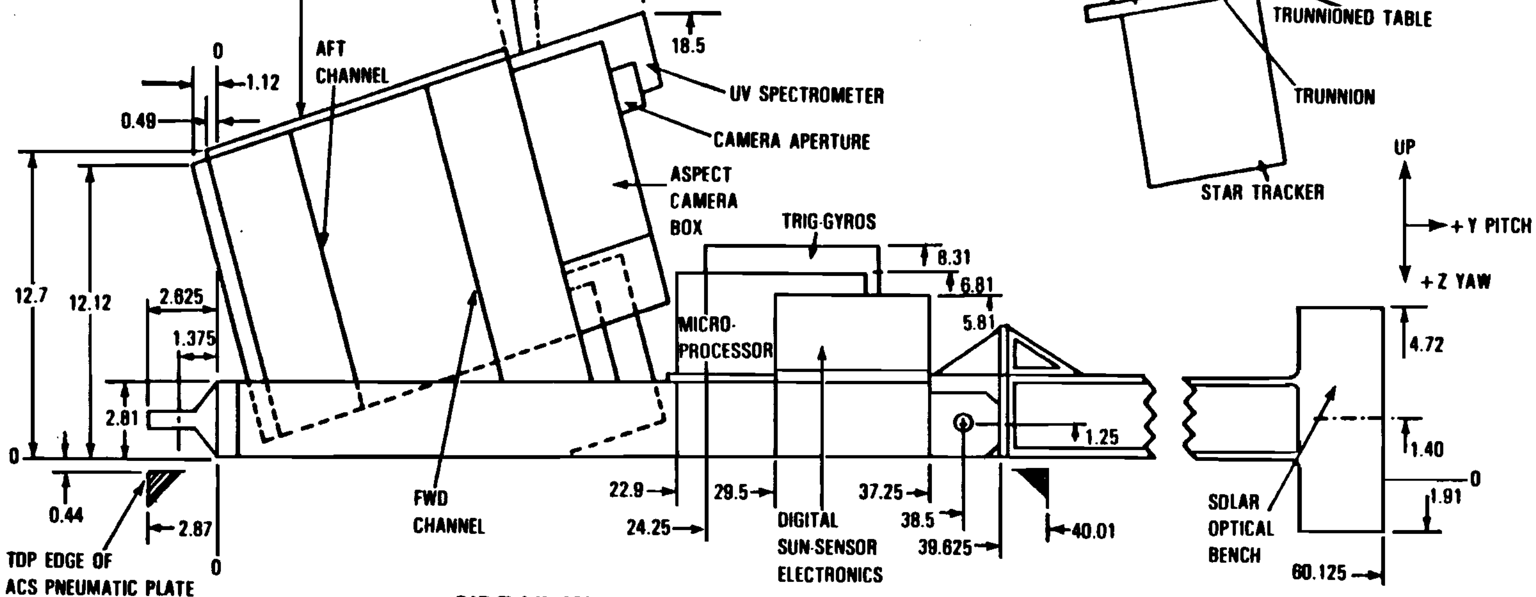


Figure 9

MECHANICAL AND ELECTRICAL SYSTEMS

SPARTAN STARTRACKER ASSEMBLY





SIDE VIEW
OPTICAL BENCH

Figure 10

ACS PROGRAM OUTLINE

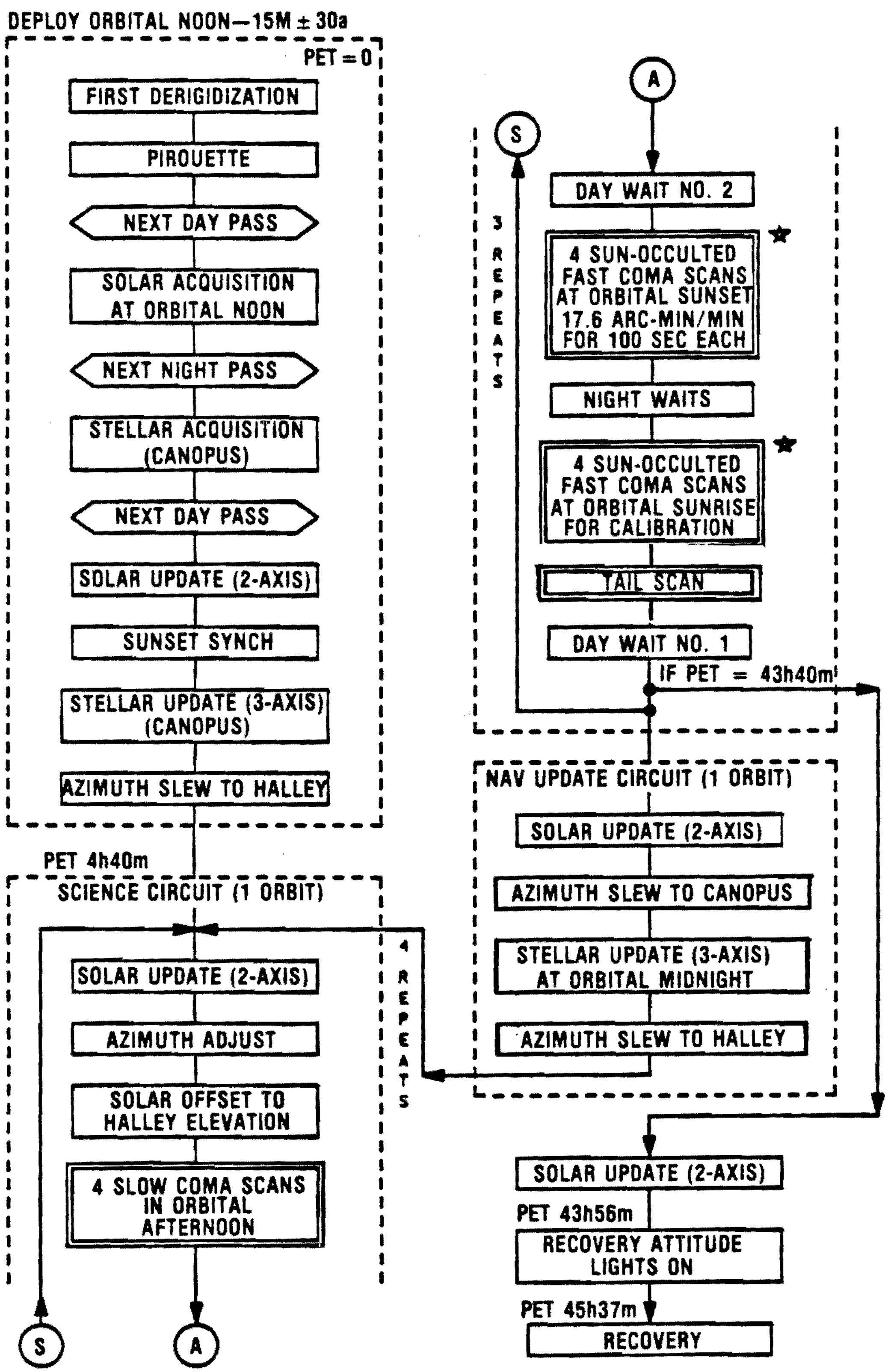


Figure 11

INITIAL ACQUISITION SEGMENT

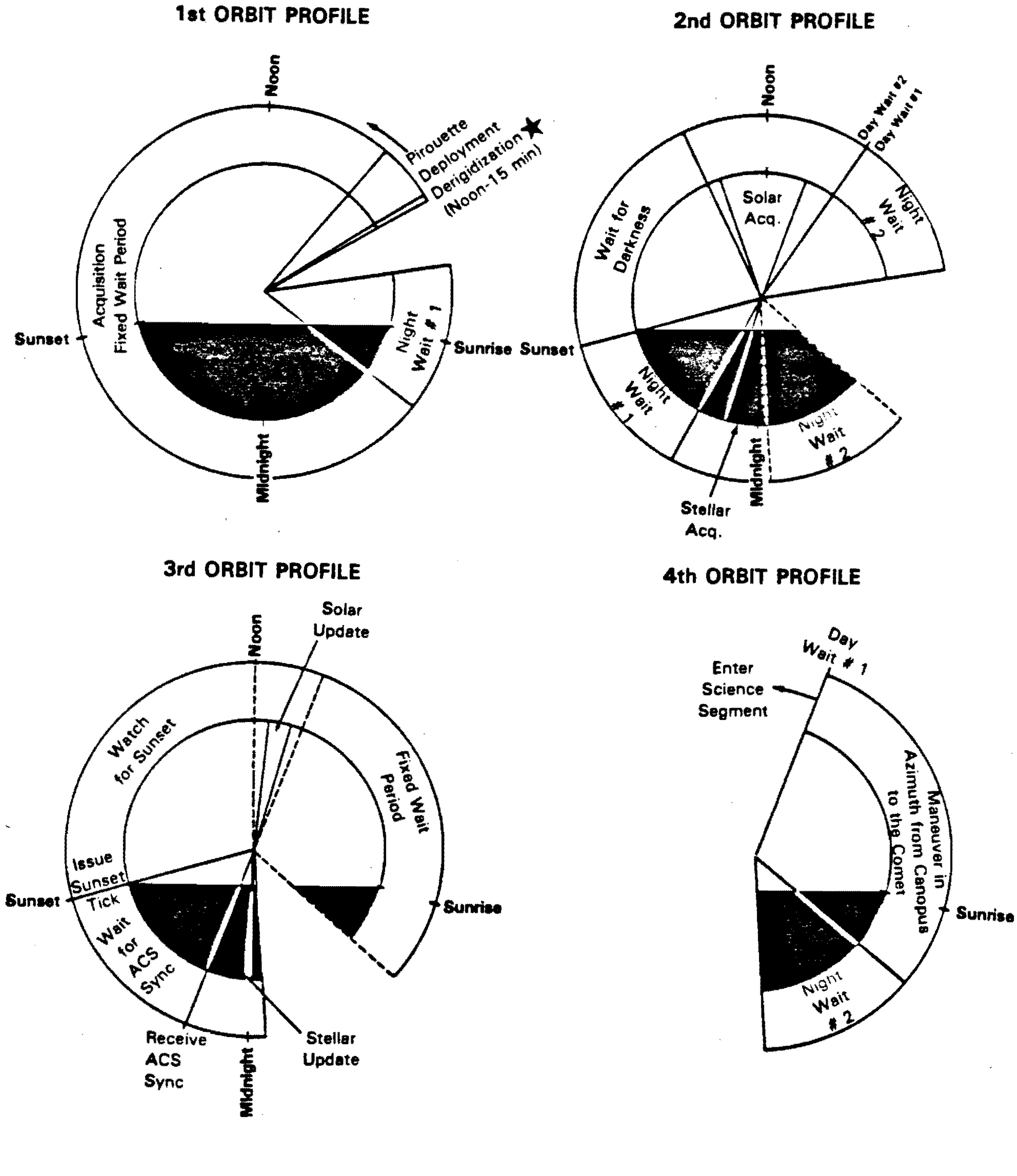


Figure 12

TABLE 1

MANEUVER MENU	
± 60 degree	0 arc-min
± 50 degree	0 arc-min
± 40 degree	0 arc-min
± 30 degree	0 arc-min
± 20 degree	0 arc-min
± 10 degree	0 arc-min
± 5 degree	0 arc-min
± 2 degree	0 arc min
± 1 degree	0 arc-min
± 0 degree	50 arc-min
± 0 degree	20 arc min
± 0 degree	10 arc-min
± 0 degree	5 arc-min
± 0 degree	2 arc-min
± 0 degree	1 arc-min
± 0 degree	0 arc-min

TABLE 2

- DEPLOYMENT ACTIVITIES TABLE

T = 0 SEC I RMS DERIGIDIZATION (15 MINS ± 30 SEC PRIOR TO ORBIT NOON)
T = 0 + SEC I RMS RERIGIDIZATION
T = 30 SEC I RMS GYRO REFERENCE ESTABLISHED AND PYRO VALVE FIRED
T = 33 SEC I RMS ACS SOLENOID VALVES ENABLED
T = 35 SEC I RMS DERIGIDIZATION POSSIBLE
T = APPROX. 40 SEC I DEPLOYMENT
T = 150 SEC I BEGIN PIROUETTE (PCW, PCCW)
T = 300 SEC I PIROUETTE COMPLETE

TABLE 3

EPHEMERIS UTILIZATION

- INSTRUMENT MICROPROCESSOR WOULD PROVIDE COMET EPHEMERIS INFORMATION ONLY WHEN REQUESTED BY ACS MICROPROCESSOR
- AZIMUTH
  - STORE IN 90 MINUTE INTERVALS
  - 1 ARC-MINUTE ACCURACY
  - USE EITHER EXACT OR AVERAGED VALUES
  - UTILIZE 2 REGISTAR SYSTEM
    - GUIDANCE (WHERE WE WANT TO BE)
    - NAVIGATION (WHERE WE ARE)
  - UPDATE THE GUIDANCE REGISTAR WHENEVER AN AZIMUTH MANEUVER WAS REQUESTED
  - SELECT PROPER MANEUVER FROM MENU THAT MINIMIZED THE DIFFERENCE BETWEEN THE GUIDANCE REGISTAR VALUE AND THE NAVIGATION REGISTAR VALUE
  - ITERATE TO ACHIEVE DESIRED ACCURACY
    - 9 TIMES TO COMET
    - 6 TIMES TO CANOPUS
  - UPDATE GYRO AZIMUTH REFERENCE (CANOPUS) EVERY 5TH ORBIT
  - IMPROVE AZIMUTH ATTITUDE 1 TIME PER ORBIT
- ELEVATION
  - STORED IN 90 MINUTE INTERVALS
  - 1 DFSS LSB (14.7 ARC-SEC) ACCURACY
  - OFFSET DFSS OUTPUT WITH EXACT EPHEMERIS VALUE
  - NULL SUMMED SIGNAL TO ACQUIRE COMET POSITION
  - UPDATE GYRO ELEVATION REFERENCE 1 TIME PER ORBIT

TABLE 4

MISSION STRATEGY

- OPEN-LOOP POINT TOWARDS COMET POSITION
  - ON-BOARD COMET EPHEMERIS
  - SPECIFIC NAVIGATION SOURCE (SUN, CANOPUS) SIGHTINGS
- OFFSET POINT IN ELEVATION VIA SOLAR SENSOR BIASING
- CONTROL AZIMUTH BY EXECUTION OF DISCRETE GYRO MANEUVERS
- SCAN COMET UNDER GYRO CONTROL
- OVERSCAN COMET
  - EPHEMERIS ERRORS
  - SENSOR ERRORS
  - UPDATE FREQUENCY
- TUNE OBSERVATION SEQUENCE TO MATCH ORBIT PERIOD AND LIGHTING CONDITIONS
  - VARIABLE DURATION PROGRAM HOLDS
    - DAY WAITS
    - NIGHT WAITS
- REPETITIVE OBSERVATION PLAN
- REQUIRE CIRCULAR ORBIT
- UTILIZE HARDWARE CONFIGURATION WHEREVER POSSIBLE TO SIMPLIFY OPERATIONS
  - SOLAR SENSOR SELECTION
  - CANT SPECTROMETERS
  - TWO POSITION, NON-ORTHOGONAL STAR TRACKER MOUNT

TABLE 5

EPHEMERIS PROCESSING SOFTWARE

- DESIGNED TO COMPENSATE FOR HARDWARE FAILURES
  - REDUNDANT EPROM BANKS
  - RUN TIME VARIABLES
    - STORED IN TRIPLICATE ON SEPARATE CHIPS
    - VOTED ON BEFORE USE
    - CONSENSUS VALUE USED
    - EACH VARIABLE DYNAMICALLY REFRESHED
  - EPHEMERIS STORED IN GROUPS OF 8 BYTES
    - 7 BYTES DATA
    - 1 BYTE CHECKSUM
    - RECORD CHECKSUM VALIDATED BEFORE DATA USED
  - SUCCESSIVE EPHEMERIS RECORDS WERE INTERLEAVED AMONG 4 EPROMS
  - WATCHDOG TIMER WOULD SWITCH EPROM BANKS IN THE EVENT OF A SINGLE EVENT UPSET

TABLE 6

CONCLUSIONS

- BACK-TO-BASICS APPROACH
  - EXECUTE COMPLEX MISSIONS
  - PROVIDE QUICK REACTION TO SCIENTIFIC NEEDS
  - KEEP PROGRAM COSTS LOW
- MULTIPLE COOPERATIVE MICROPROCESSORS
  - EXCEPTIONAL VERSATILITY
  - MINIMIZE SUBSYSTEM COMPLEXITY
  - REDUCE COSTS
- THE USE OF THE APC AS A DATA ENTRY DEVICE
  - CLEVER AND INNOVATIVE
  - COST EFFECTIVE
  - CUMBERSOME
  - NOT USER FRIENDLY
- MINIMIZE STS INTERFACES
  - REDUCES JSC/KSC INTEGRATION TIME AND WORKLOAD
  - IMPROVES MANIFESTING OPPORTUNITIES
  - SIMPLIFIES OPERATIONS
- MINIMIZE LAUNCH WINDOW IMPACT
  - IMPROVES MANIFESTING OPPORTUNITIES
  - PREVENTS SLIP-INDUCED PROBLEMS
- SHORT DEVELOPMENT TIMES
  - PROMOTE INNOVATION
  - NURTURE ENTHUSIASM
  - PREVENT EXCESSIVE COMPLEXITY IN DESIGN SOLUTIONS
- SINGLE UNIFIED TECHNICAL TEAM APPROACH
  - EFFICIENT
  - COST EFFECTIVE
  - PROVIDES TREMENDOUS CONTINUITY
  - DEVELOPS TALENTED SYSTEMS-ORIENTED PERSONNEL