

# TOMS-EP Spacecraft

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## Abstract

The TOMS-EP spacecraft is a part of NASA's Mission to Plant Earth, sponsored by NASA/GSFC for an ozone mapping/surveillance mission. The mission requires a single instrument in a sun synchronous, noon, low earth orbit.

The spacecraft must provide the mission capabilities and robustness of larger spacecraft within the weight and volume capabilities of the Pegasus LV, and at a relatively low cost, allowing the spacecraft carry a single instrument. Autonomy is provided for both deployment and safe hold modes, using the full functional redundancy provided. Contamination control is designed to protect the payload instrument. The subsystems were optimized to provide the required performance within weight and volume limits.

The spacecraft design has been fully verified both functionally and for the operational environments through unit tests, test bed verification, and protoflight testing.

## Introduction

The NASA/GSFC TOMS-EP Spacecraft shown in Figure 1 consists of the Total Ozone Mapping Spectrometer (TOMS) and the supporting spacecraft bus (Earth Probe, EP). The mission is a follow-on to the NIMBUS mission, (Ref. 1, which also describes the instrument,) designed to provide daily ozone mapping data and to track upper atmosphere gasses and volcanic debris

(Ref. 2.) While other TOMS instruments currently are flying or will be launched on other spacecraft (i.e. Russian METEOR, Japanese ADEOS,) the TOMS-EP is the only one with the same orbit as Nimbus which is dedicated to a single instrument, with a primary mission of mapping ozone. Thus, the orbital characteristics for TOMS-EP are optimized for ozone mapping and the spacecraft bus was both designed to meet these requirements and in addition used these pertinent parameters to simplify and optimize the design. This paper will describe the spacecraft bus design, as driven by the mission and instrument requirements.

## Mission

The TOMS-EP mission is to provide daily ozone mapping data on a continuous basis, via atmosphere reflectance/absorption measurements. The orbit is a high inclination, low altitude orbit which gives data over the entire earth every 24 hours and over both poles every 104 minutes (summarized in Table 1.) The required lifetime is two years ( $P_s > 0.9$ ) with a goal of three years (i.e., no spacecraft consumable or wearable should prevent three year operation.) To make the single instrument concept cost-viable the spacecraft must be launched on NASA's Small Expendable Launch Vehicle (SLEV), the Pegasus.

Altitude: 955 KM  $\pm$  25 Km  
Inclination: 99.3°  
 (Retrograde, sun synchronous)  
Ascending Node: 11:00 to  
 12:00 MLT (i.e., pass through  
 the sub solar point)

Table 1:  
 TOMS Orbital Requirements

Mission command and telemetry operations are via NASA GSTDN ground stations, dictating communications with the spacecraft when it is passing over a ground station (15 min., maximum). The spacecraft design must support communications periods of once per day, but up to 4 contacts per day are planned. In addition launch and early orbital operations are compatible with the USAF SGLS with NASA data protocol.

Science data resolution dictates on-board time knowledge to better than of <100 msec with respect to UTC and a pointing accuracy of < 0.5° in roll/pitch and <1.0° in yaw, and pointing knowledge of <0.25° for all three axis.

Instrument

The TOMS instrument is a scanning ultra violet spectrometer which measures reflected energy in 9 spectral bands. Earlier models were flown on NIMBUS and METEOR, however for TOMS-EP and subsequent missions the instrument has been refined for operational flexibility, stability, and more precise calibration. A description is provided in Reference 1.

The instrument provided requirements for viewing angle, electrical power (both operational power and make-up-heater power), thermal, command and data protocol, contamination, and structural support. These requirements are summarized in Table 2.

Electrical Power(28 VDC):  
 Operational - <25 watts  
 orbit average  
  
 Non-operational - <10 watts  
Command: Serial:  
 50/day, 16 bit word, 150  
 Kbps  
 Discrete: ~7 relay  
 commands  
Data: <750 bps, asynchronous, 150  
 Kbps transfer rate  
FOV: Nadir pointed 50° X 3°  
Thermal: Isolated Interface (<5 watts  
 transfer)  
Mechanical Alignment:  
 <0.030° accuracy  
Pointing Knowledge:  
 0.25° knowledge  
Contamination: <750A at launch

Table 2:  
 Instrument Support Requirements

Launch Vehicle

The Pegasus launch vehicle is an airplane launched (L-1011), three stage solid with an orbit capability of 350 x 950 Km for the TOMS-EP mass. The aircraft launch imposes man-rated safety requirements, and the flight segment between aircraft take off and actual Pegasus launch imposes physical and operational constraints. For example, thermal control and external power must be supplied during this period, prior to drop. Since the launch vehicle cannot achieve direct insertion into the TOMS-EP orbit, the spacecraft must provide integral propulsion to boost itself into the final orbit.

## Spacecraft Bus Design

The following aspects were considered during the spacecraft design phase:

- Functional Requirements
- Reliability and Autonomy
- Mission Operations
- Instrument Support
- Launch Vehicle Integration

Cost, schedule, and risk were strong drivers towards simple, multi-functional designs, using existing or modified components where possible. The Pegasus launch vehicle imposed significant weight and volume constraints. In summary, the goal was to build a cost efficient, reliable small spacecraft with operational flexibility and autonomy.

### Functional Requirements

The functional requirements are grouped into the following subsystem areas:

- Command and Data Handling Subsystem (CADHS): Data processing and transmission
  - Altitude control and Determination (ACDS): Altitude control and determination for orbit changes (deltaV), science pointing and safe modes:
  - Power and Distribution Subsystem (EPDS): Electrical power generation, storage and distribution (including switching)
  - TOMS Flight Software (TFS): Software support for the above subsystems
  - Orbital Adjust Subsystem (OAS): Impulse for deltaV maneuvers and attitude stabilization
  - Structure and Mechanisms Subsystems (SMS): Support structure for components and interfaces, and deployment mechanisms

- Thermal Control Subsystem (TCS): Thermal control for the spacecraft bus

Spacecraft design was interactive. Each subsystem was designed to specific subsystem requirements with an overall system design providing system trades, subsystem interfaces, reliability, autonomy, contamination control.

Figure 2 shows a sectional view of the spacecraft, and Figure 3 a block diagram of the functional elements. In general, the basic spacecraft configuration was defined from the functional requirements, L/V constraints, and instrument requirements. Given this basic architecture, reliability analysis was used to add redundancy, either physical or functional, to provide a reliability of 0.97 for two years and 0.93 for three years. From this reliability analysis the detailed architecture was developed. Although the mission requirement is  $>0.9$  for two years, a further constraint of no credible single point failures was imposed to increase confidence in mission success. An example of this is the decision to provide redundant thrusters. They do not contribute significantly to the numerical reliability calculations since they are only used for a short period during the initial orbit insertion, but thruster loss would mean mission loss for a non-redundant system.

Autonomy was incorporated to safe the spacecraft in case of component failure. Two levels of safeing were provided. The first level is safe hold, provided primarily for loss of attitude control in the normal science mode. The second level is safe power, provided for power anomalies and problems with safe hold. Safe hold offers quick recovery from

minor problems and is implemented using a B-Dot altitude control law utilizing momentum wheels and magnetic torque rods controlled by a magnetometer. Safe power is a spin stabilized mode which activates the OAS thrusters and points the spacecraft at the sun, and is the most robust level of safeing, utilizing only redundant units.

Primary considerations in the design of the safe modes were parameters to be monitored for fault detection and the equipment complement to be used in the implementation. Secondary voltage undervoltage and spacecraft computer health as indicated by a watch-dog-timer (WDT) are maintained in hardware. Other parameters such as events in the attitude control and power subsystems are monitored in software.

No fault isolation is done on-board. In case of an anomaly or failure the spacecraft autonomously reconfigure from primary equipment to redundant. Fault isolations and recovery functions performed on the ground. The safe modes are designed to provide a minimum of 24 hours of safe operations following a failure or anomaly. This allows time for mission operations to detect the failure, and affect a recovery as appropriate. The spacecraft is safe in either mode indefinitely. The only limitation is that the battery charge rate may have to be adjusted within 24 hours, in the case of power subsystem or computer failures.

Contamination control was a key design consideration. The goal was to minimize particles and volatile condensable material coating of sensitive surfaces. A contamination analysis was performed using material outgassing properties, spacecraft physical layout, and thermal models. The effect of molecular contamination

on the various surfaces (instrument calibration diffuser, earth and sun sensors, solar panels, etc.) was determined and a contamination budget established. Vents from internal compartments were designed to outgas away from sensitive surfaces. Access holes, particularly those adjacent to optical surfaces were sealed. Rigid standards were established for spacecraft assembly and test with provisions were made for periodic inspection and cleaning.

## Subsystem Design

### Command and Data Handling

The CADH subsystem performs data processing, formatting, storage, telemetry transmission and command receipt. The operation of the CADH subsystem is built around the periodic ground contact available, requiring both store and forward telemetry, and on-board stored commands sequences. For LEO, adequate communication rates (2 Kbps uplink, 200 Kbps downlink) can be maintained with fixed omni antennas and a 3 watt S-band transponder. A complete 24 hour set of spacecraft state-of-health and instrument science data can be returned in 10 minutes with optional lower rates available for 6 hour recording periods. The CADH subsystem contains two computers: the spacecraft processor (SP) to process commands and stored sequences and provide subsystem computing; and the data processor (DP) to manage the solid state mass memory (16 mbytes), control instrument data and format telemetry. The DP interfaces only with the SP and the instrument. The SP provides dedicated interfaces with all other components, including the RF transponder. Command receipt, error checking, and telemetry encoding and output are all performed in hardware.

All other CADH functions are implemented in software, and thus require absolute software reliability for spacecraft operation. The SP monitors internal operation and external communication with the DP, and constantly resets a watch dog timer. In case of a hardware or software failure, the watch dog timer will not be reset, and its timeout will result in automatic fail-over to the redundant processor. The redundant SP downloads code from EEPROM, checks the spacecraft state, including data stored in DP memory, and configures the hardware and software in the appropriate safe mode.

#### Attitude Control and Determination

The Attitude Control and Determination subsystem performs attitude control and maintains on-board attitude and ephemeris references for use in control and science data reduction. The ACDS provides four distinct modes: deltaV, science, safe hold (B-dot) and safe power (sun pointing.) The deltaV and science modes are operational modes and the safe modes are contingency modes. A detailed description of the ACDS is contained in Reference 3.

The deltaV mode is used to go from the SELV injection altitude to the science orbit. This mode allows either an inertial or LVLH orientation, using inertial reference units and thrusters. The initial reference is established by an earth acquisition prior to the deltaV maneuver. This is followed by a command to the attitude required for the maneuver. A timed burn is then commanded from the ground with attitude controlled to the appropriate profile using the inertial reference system. Due to the low thrust (~ 4 lbf, at beginning of life), each maneuver lasts 20 to 30 minutes (~ 1/4 to 1/3 orbit). Changing from the injection orbit to the final mission orbit will take seven

burns spaced over a period of two weeks. Pointing error during thrusting is  $<1.5^\circ/\text{axis}$ . Altitude control is by off modulation, with roll control provided by a  $10^\circ$  thruster cant. Redundancy is supplied through parallel redundant drive electronics and thrusters.

In the science pointing mode, attitude control is via a pitch momentum bias system utilizing scan wheels. Two scan wheels are arranged in a  $\pm 20$  degree V configuration about the pitch axis in the pitch/yaw plane. These wheels combine the earth sensor and momentum wheel function, with the nominal momentum vector along the pitch axis. The gyros provide the primary short term attitude reference to a quaternion state estimator. The estimator is updated periodically by earth sensor and fine sun sensor measurements. Gyro drift is also updated periodically in the estimator. The earth sensors are corrected for earth oblateness and radiance, and their data are not used over the poles. Two fine sun sensors are used to provide sun aspect angles, once per sensor per orbit. This attenuates the effect of known inaccuracies and increases attitude determination, predicted at  $0.18^\circ$  (all axes,  $3\sigma$ .) Momentum unloading is implemented via a magnetometer and torque rods.

Gyro redundancy is provided by three 2-axis IRU's mounted orthogonally. Momentum wheel redundancy is provided by a third wheel mounted with its spin axis along the yaw axis to compensate the unbalance yaw momentum resulting from the loss of one scan wheel. Both of the scan wheel sensors and both the fine sun sensors are used nominally, but the attitude control algorithms can accommodate the loss of one of each. The magnetic momentum unloading system is

functionally redundant by use of the thrusters.

Two contingency modes are used during science operations: safe hold and safe power. The safe hold mode provides a minimal configuration change if the science pointing altitude is lost. Wheel speed is maintained to provide yaw stiffness, and the output of the magnetometer controls the torque with a B-dot algorithm for pitch control. This results in spacecraft pitch rotational twice earth's rate, adding a 1 rev/orbit rate w.r.t. the earth. This maintains the pitch axis of the solar arrays perpendicular to the orbit plane, providing adequate power for the spacecraft for an indefinite period. If power or attitude control cannot be maintained in safe hold, the spacecraft will autonomously go to safe power mode.

The safe power mode is used for more severe problems (wheel failure, power problems) for which the ACDS will drive the spacecraft so that the sunline is along the roll axis and perpendicular to the solar array average. This is done with the coarse sun sensors (mounted on the solar arrays), and the redundant thrusters. Once the spacecraft reaches the sunline, a nominal roll rate is established (2-4°/sec). Rate control around the sunline is provided by the redundant (third) gyro. At this point the spacecraft is spin stabilized, and the thrusters are used only for precession control by ground command, providing long term storage and time for the ground crew to diagnose and recover from the fault.

#### Electrical Power and Distribution

The Electrical Power and Distribution subsystem provides for power regulation, conversion and distribution. In addition, the SELV electrical interface (both data and power) is contained in

the EPDS. A single super NiCd battery is connected directly to the fused 28 VDC primary power bus. Redundant DC/DC converters generate and distribute secondary power. All power switching is contained in the power control unit (PCU) and dedicated power lines are provided to individual units as required. Power generated by the solar arrays is regulated and supplied to the main bus, and is used for supporting the loads and charging the battery.

A primary design driver for the EPDS is to provide energy balance for the worst case single orbit. Power generation capability is constrained by SELV weight and volume capabilities, a design requirement to minimize deployables and not use rotating actuators, and the charge characteristics of NiCd batteries.

Maximum use of the allowable length of the SELV fairing provided single array panels with enough area to produce the required power and eliminated the need for a secondary fold in the panels. High efficiency silicon cells, optimized for degradation over a three year period, were used on both sides of the panels. Three panels were used per array, allowing compact stowage around the spacecraft structure for launch. Each array was canted 45° to the spacecraft longitudinal axis, about the pitch axis, to eliminate battery discharge at the sub solar point. The power output thus has three maxima as the spacecraft passes through the sunlit portion of the orbit. The first maxima is greatest as the leading array (the one perpendicular to the sunline first) is cold and generates maximum power when the battery can accept the most charge current. The sum of the two panels provides enough power at the subsolar point to eliminate battery discharge, and the trailing array provides adequate power for battery

taper charge. An orbital average load of ~130 W. can be supported.

Power conversion is done through a pulse width modulated converter, giving converter efficiencies of ~93%. The converter duty cycle is controlled by computer algorithms to provide peak power tracking or direct energy transfer for the initial portion of the charge cycle. NASA standard temperature compensated voltage level (TCVL) battery charging is used for the remainder of the charging cycle after the TCVL limit is reached. Slightly higher charging efficiency can be obtained by using an optional computer controlled taper charge algorithm in place of the TCVL.

The battery is a 9 amp-hr 22 cell super NiCd, which has been qualified by NASA and flown on SAMPEX in a similar LEO orbi. Redundancy is provided by including a extra cell, since an open cell failure is considered non-credible. A maximum depth of discharge of 30%, with a recharge ratio of 108%, was used for design.

A number of failure monitoring points have been provided for the EPDS. The current and voltage sensors are active redundant. If they do not agree, a third value is calculated using Kirkoff's law, and the closest pair are used. The battery state-of-charge is calculated by a computer based integrator, and if full recharge is not reached, the spacecraft is put into safe power mode with concomitant load shedding. While the primary power bus is fused, the secondary busses are not, and a secondary hardware undervoltage detector is provided. If an under/over voltage is detected, the redundant converters and redundant loads are selected.

#### TOMS-EP Flight Software

The Flight Software is contained in two asynchronous computers: the spacecraft processor (SP) and the data processor (DP). Total code size is 326 Kbytes (SP), and 132 Kbytes (DP). The software is written in a combination of C and assembly language executing in an 80C86 running at 5 MHz. The EPDS and ACDS code was translated into C from FORTRAN code used in simulations of these subsystems. The I/O routines are coded in assembly language for efficiency. A task priority operating system (VERTEX) is used in each computer, wherein timed interrupts start specific tasks at fixed intervals, with the higher frequency tasks interrupting lower frequency tasks. If tasks are not completed within the required time, operation is halted and the watchdog timer forces the appropriate failover response.

Upon bootup, the program is initialized from EEPROM, with the real time program and data being downloaded into RAM. Data such as parametric values and telemetry formats are stored in a data table and may be updated via ground command. However, if the computer is reset, the data table value will revert to that stored in PROM. Hardline PROM download capability is provided during assembly and test allowing software updates up to final integration with the launch vehicle.

Stored command sequencing capability is provided in software, for both absolute and relative time command sequences. These are used for spacecraft initialization, array deployment and communication upon initial spacecraft injection, and initialization in the safe modes. These mode initialization sequences are stored in PROM, and the SP checks the spacecraft avionics configuration at boot-up to select the proper spacecraft

mode and sequence. Other sequences may be loaded via ground command to perform repetitive operational tasks such as data playback.

The flight software was verified by three methods, all using engineering model (EM) or flight hardware. First, for internal software modules (i.e., those not communicating outside the processors) an EM SP and DP test station was used. Second, for all other modules which did not require hardware redundancy testing, the EM test bed, which provided all I/O signals, was used. After initial test bed software verification, CADH I/O, ACDS and EPDS closed loop operation was verified using either real or simulated (e.g. vehicle dynamics) inputs. In-circuit emulators were used on all test bed verification. At this point the software was accepted for use on the flight spacecraft. Third, the flight spacecraft was used to verify redundancy management and stored sequences. End-to-end testing with the ground system at NASA/GSFC also was used to verify all operational commands and telemetry playback modes. This sequence of verification was very effective, with only minimal problems encountered after test bed sell-off. In addition testing stored sequences late in the spacecraft verification cycle allowed updating these sequences as operational plans matured. During spacecraft system test, software changes, as usual, accommodated hardware problems.

#### Propulsion

The Orbit Adjust subsystem provides the impulse for deltaV maneuvers and altitude stabilization during initial ascent and safe power modes. Therefore, it must operate in two modes: steady state (Isp ~ 210 sec) and pulsed with a minimum pulse of 60 msec (Isp > 40 sec.) For the sake of simplicity, a

monopropellant hydrazine blowdown system using standard components was provided. The system pressure is 350 psia at beginning of life, and 60 psia at end of life.

The OAS schematic is shown in Figure 4, and consists of a 22" titanium tank (160 lbm propellant), two isolation valves to provide redundancy and four each redundant dual seat thruster modules, providing 1 lbf thrust at BOL. The ACDS provides the electrical pulses to all valves via drive electronics. Dedicated prime and redundant heater circuits are used for the tank and the propellant lines.

#### Structure and Mechanisms

The Structure and Mechanism subsystem provides the support structure, the solar panel substrate, the solar panel deployment hinges, and the solar panel release mechanisms.

The support structure is entirely of aluminum, for ease of fabrication and minimum cost. The launch vehicle loads are carried through the tank support cone to the central equipment platform. Extruded, then machined longerons are used to transfer the load to the nadir platform. Longerons are also used to support the zenith equipment platform, where the solar arrays, the OAS thrusters and supporting avionics are attached. Stiffness is supplied by the use of shear panels, which completely enclose the structure. The shear panels and equipment platforms are made of aluminum honeycomb and facesheets, with bonded inserts for attaching equipment. The launch vehicle attach and separation hardware (V-band, etc.) and all separation ordinance is supplied by the SELV.

The design loads were specified by the SELV-to-Spacecraft ICD, and the design and test margins by the

NASA/GSFC General Environmental Verification Specification (GEVS). The critical load was the lateral load imparted when the Pegasus is released from the carrier aircraft. This consists of a lateral acceleration (~ 3.8 g) and a rotational component input at the spacecraft base. Extensive analysis was done using NASTRAN, and testing was done on both a structural test model and the integrated spacecraft to verify that the structure could withstand the applied load. Verification of the strength was by sine burst tests, and random vibration was done to GEVS qualification levels.

The solar panel substrate is aluminum honeycomb covered by carbon fiber (P-1000), with hinge attach points bonded in. A life test panel with substrate and a complete solar cell stack (i.e., cell, coverglass, soldered joints, etc.) was cycled in air between -130°F to +180°F for 15,000 cycles to qualify the panel for the 3 year life. In addition to receiving full protoqual sine burst and vibration tests, the flight panels were acoustically tested on the structural test model

The solar array hinge is based on the TDRSS design for both the main array and panel to panel hinges. Extensive testing was done on the STM and in a temperature chamber to characterize the torque/angle characteristics of these hinges, to guarantee >100% worst case margin, to qualify the panels and hinges for the loads. Release and first motion tests were done on the flight spacecraft.

The solar array release mechanisms are nitinol actuated frangible bolts. Nitinol is a titanium/nickel alloy that may be compressed while cold and will retain its compressed shape until heated beyond a critical temperature, causing an abrupt expansion to its original length. This expansion is used to break the frangible bolt. The devices

are actuated by redundant heaters bonded to the outside of the nitinol slug. Such release devices have several advantages: they are safe and easy to handle, they are relatively inexpensive, and the shock level is lower than ordinance. The device has been qualified for actuation at temperatures as low as -30°F with a maximum heater pulse of 50 sec. During on-orbit panel deployment, both prime and redundant heaters are energized sequentially.

#### Thermal Control

The Thermal Control subsystem is primarily passive, with heaters used as back-up for worst case conditions. The primary design drivers are the battery, for which a relatively cold temperature was specified (> 32° F), the earth sensors, for which an ambient temperature was specified, (~ 40°F) and the propellant which must be > 40°F. Since the spacecraft has limited power, the passive design must accommodate normal science operations for both hot and cold cases. The design philosophy is to conduct the heat through the mounting plate to the external panel or radiate heat from the avionics unit to the external panel, and then re-radiating the heat from the external panels. This technique is adequate for the limited  $\beta$  angle experienced (0° to 18°) in the noon sun-synchronous orbit. The amount of heat radiated is controlled by paint patterns and multi-layer insulation and (MLI). In the OAS module, where little heat is generated, MLI is used both internally and externally to limit radiation. The instrument interface is thermally isolated, with the instrument providing separate heaters. The spacecraft heaters, controlled by redundant thermostats, will activate for cold case non-science modes.

The thermal design has been verified by thermal-vacuum tests with the flight spacecraft. Thermal balance to predicted temperatures was demonstrated for both hot and cold cases.

#### **Spacecraft Test and Launch**

Prior to the start of spacecraft assembly, subsystem (ACDS, EPDS, CADH) verification had been performed on an engineering model, and thus, the integrated spacecraft performance testing was for functional performance and spacecraft level interfaces. This was done through a series of comprehensive performance tests, repeated periodically during the environmental tests to ensure that performance met specification. The environmental tests were in full compliance with the NASA/GSFC GEVS requirements: vibration, shock, thermal cycling and EMI/EMC. Tailored versions of MIL-Std-461, and 462 were used for EMC levels.

Upon completion of the final tests, the spacecraft was shipped to Vandenberg

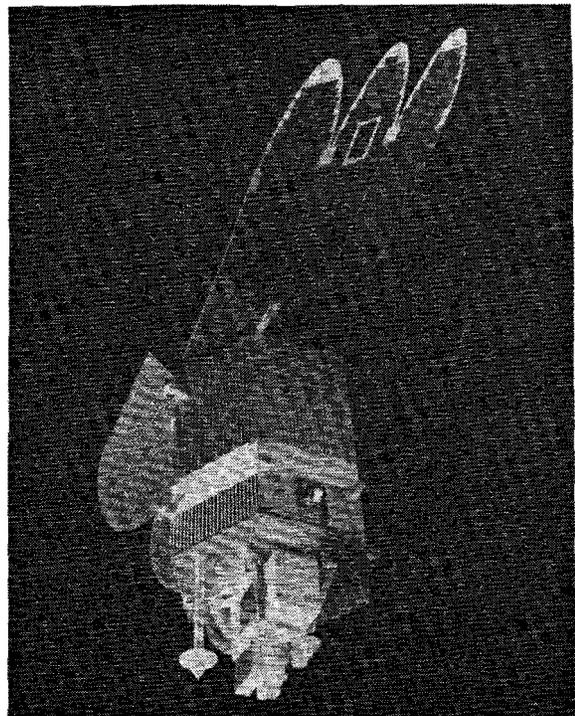
AFB for launch with the L-1011/Pegasus launch vehicle. Final preparation for launch, including fueling and spacecraft/launch vehicle integration and test, is planned for four weeks.

Design development, qualification and preparation for launch represents a span of less than three years from contract award for a complex spacecraft, fully qualified to NASA/GSFC standards, ready to make it's contribution to NASA's Mission to Planet Earth.

#### **References:**

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2. Sighting of El Chicon Sulfur Dioxide Clouds with the Nimbus 7 Total Ozone Mapping Spectrometer; Krueger, A. J.; *Science*, 24 June, 1983
3. Attitude Control System for a Small Earth Probe Satellite; Mendenhall, T., Schurr, H., Schmeichel, H.; 17th Annual AAS Guidance and Control Conference; 2 Feb., 1994

**Figure 1.**  
TOMS-EP Spacecraft



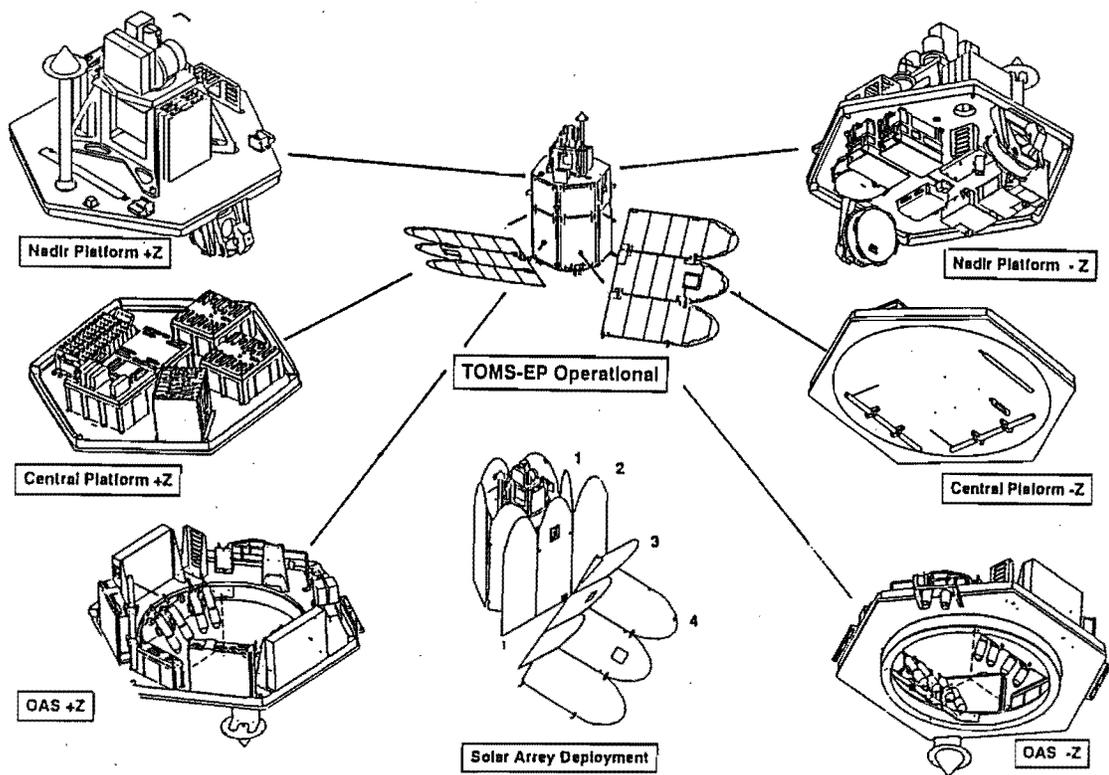


Figure 2. TOMS-EP Spacecraft View

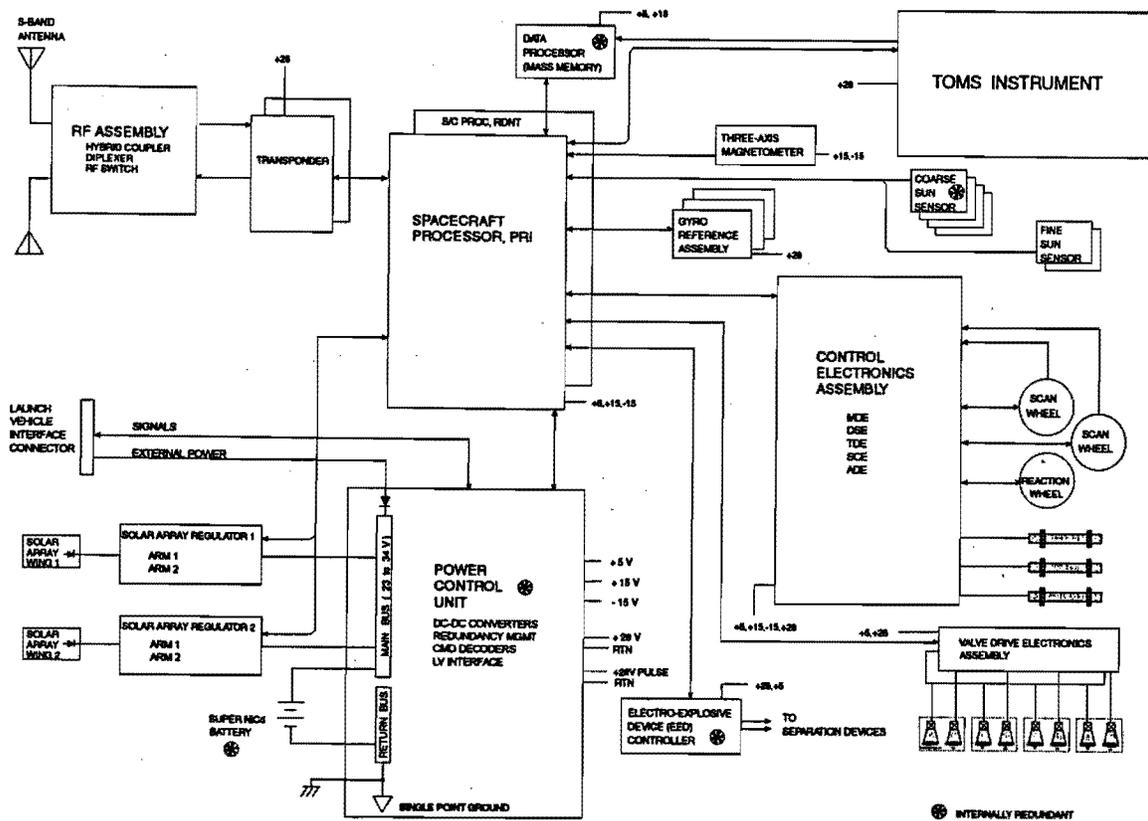


Figure 3. TOMS-EP Block Diagram

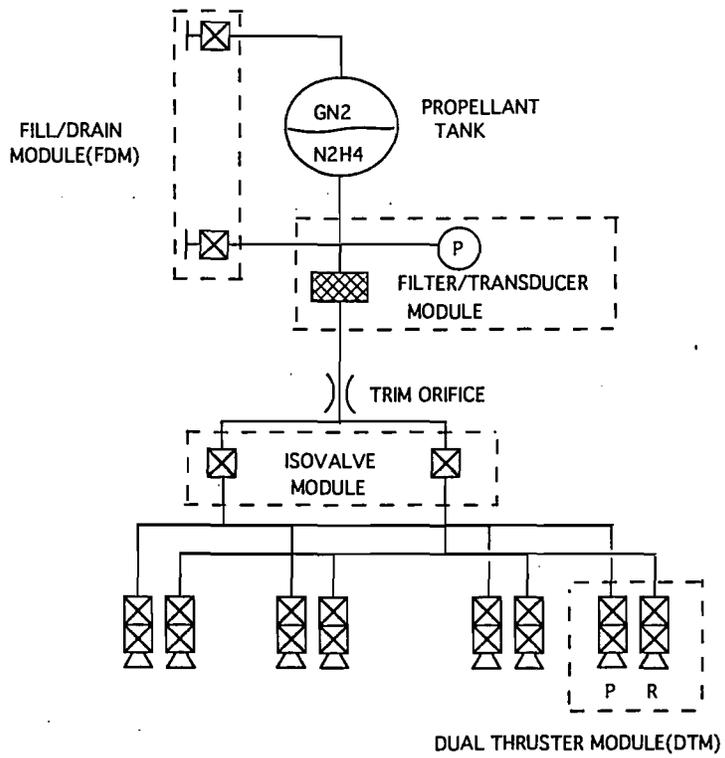


Figure 4. OAS Schematic