

ACE - A Canadian Small Science Satellite Mission

Harold Dahl and Wally Eliuk
 Space Systems Group
 Bristol Aerospace Limited
 P.O. Box 874, 660 Berry Street
 Winnipeg, Manitoba, Canada, R3C 2S4
 (204) 775-8331, e-mail: hdahl@bristol.ca weliuk@bristol.ca

Glen Rumbold and Randolph Shelly
 SCISAT Project Manager and Spacecraft Manager
 Canadian Space Agency, Space Science Program
 100 Sussex Drive
 Ottawa, Ontario, Canada, K1A 0R6
 (613) 990-0803, (613) 990-0802, e-mail: glen.rumbold@space.gc.ca, randy.shelly@space.gc.ca

Abstract. The Canadian Space Agency (CSA) has initiated the Small Science Satellite (SCISAT) Program as a part of their ongoing space science program. The Atmospheric Chemistry Experiment (ACE) Mission from the University of Waterloo has been selected as the first SCISAT Mission, and will be launched in Q1, 2002 on board a Pegasus XL vehicle. The ACE spacecraft will be co-manifested on the Pegasus vehicle with another spacecraft which has not yet been selected. The ACE Mission will comprise instrumentation to measure atmospheric chemistry using the solar occultation method. The principal goal of the ACE Mission is to measure and to understand the chemical and dynamical processes that control the distribution of ozone in the upper troposphere and stratosphere. The spacecraft will be designed to operate in a 650 km, 65° inclination orbit for 2 years. The spacecraft will be developed by Canadian Industry, with Bristol Aerospace Limited of Winnipeg, Manitoba being the prime contractor for the bus, and Bomem, Inc. of Quebec City, Quebec the prime contractor for the instrument.

This paper presents an overview of the CSA's SCISAT Program and the ACE Mission. The paper describes the mission concept, the scientific instrument and the concept for the spacecraft bus, highlighting new technology that will be developed in Canada to support this mission.

SCISAT Program Overview

The SCISAT-ACE Mission is being conducted under the auspices of the Canadian Space Agency's (CSA) Space Science Program. The CSA provides opportunities for Canadian scientists to conduct experimentation from space in the fields of Earth Sciences, Space Astronomy, Space Exploration and Life Sciences. Selection of missions is done through an Announcement of Opportunity (AO) process, culminating in competitive selection of science teams through a peer review process. Preparation of the tools necessary to perform the research is accomplished through contracts to

Canadian industry, including the science instrumentation, the spacecraft bus, and the associated ground control and data reception elements.

The SCISAT program is part of a collaboration between the CSA and the US National Aeronautics and Space Administration (NASA), consisting of two missions. Under the terms of the cooperative agreement, each agency will provide a spacecraft and instrumentation, to be co-launched on an expendable vehicle. Mass resources will be shared between the two spacecraft, and missions selected to be reasonably compatible in terms of orbit

requirements and launch schedule. Following deployment of the spacecraft on-orbit, each agency will be responsible for their own flight operations.

The AO for the Canadian elements of the first SCISAT (SCISAT-1) was released in 1997. The ACE Mission was selected for flight in November of 1998. Launch is presently scheduled for the first quarter of 2002. The NASA co-payload has not yet been identified.

SCISAT - ACE Mission Overview

Science Objectives

The principal goal of the ACE mission is to measure and to understand the chemical and dynamical processes that control the distribution of ozone in the upper troposphere and stratosphere. Anthropogenic changes in atmospheric ozone are increasing the amount of ultraviolet radiation received at northern mid-latitudes and in the Arctic, and may affect the climate. A comprehensive set of simultaneous measurements of trace gases, thin clouds, aerosols and temperature will be made by solar occultation from low earth orbit (see Figure 1)

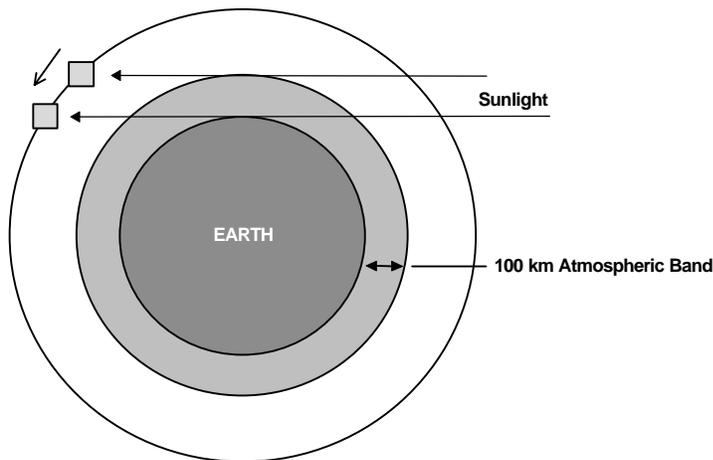


Figure 1 - Mission Operations Scenario

A high resolution infrared Fourier transform spectrometer (FTS) operating from 2 to 14 microns will measure the vertical distribution of trace gases as well as the meteorological variables of pressure and temperature. During sunrise and sunset, the

FTS measures infrared absorption spectra that contain information on different atmospheric layers. These spectra will be inverted on the ground to provide vertical profiles (3-4 km resolution) of atmospheric constituents. Temperature and pressure will be derived from the CO₂ lines. Aerosols and clouds will be monitored using the extinction of solar radiation at 1.02 and 0.525 microns as measured by two filtered CCD imagers and in the infrared by the Fourier transform spectrometer.

FTS Instrument

The FTS instrument is an infrared spectrometer with an auxiliary 2-channel visible / near infrared (IR) imager (see Figure 2). The instrument includes a suntracker, which provides the sun radiance to both the infrared spectrometer and the visible / near IR imager during solar occultation of the earth's atmosphere. During sunrises and sunsets, the instrument measures the visible and infrared signals that contain information on different atmospheric layers, which provides the vertical profiles of atmospheric components.

Fourier Transform Spectrometer

The infrared spectrometer is a high-resolution (0.025 cm⁻¹) infrared Fourier-transform spectrometer. It is an adapted version of the classical sweeping Michelson interferometer, using an optimized optical layout. The spectrometer will look at the sun through the atmosphere. From these spectra, the vertical distribution of trace gases and temperature will be extracted.

Double-sided interferograms will be Fourier-transformed on the ground in order to obtain the desired spectra. The IR spectrometer operates from 700 to 4100 cm⁻¹ (2 to 14 μm) over 2 bands.

Indium Antimonide (InSb) and Mercury-Cadmium-Telluride (HgCdTe or MCT) detectors are used in a special configuration. In order to operate under background limited conditions, both detectors must be cooled. However, the InSb has the more

stringent requirements and need to operate below 110 K. Cooling of the detector package will be accomplished using a passive (radiative) cooler.

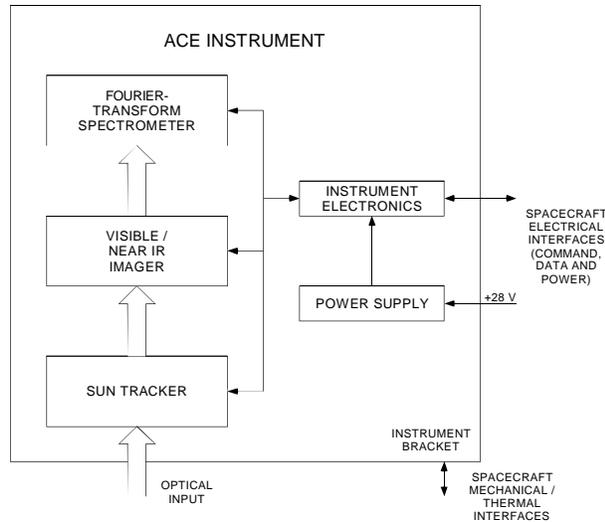


Figure 2 - FTS Instrument

The spectrometer will be designed to achieve a signal-to-noise ratio better than 100 with a field-of-view (FOV) of 1.25 mrad and an aperture diameter of 100 mm, each measurement lasting for 2 seconds. A semiconductor laser will be used as the metrology source.

Visible/Near Infrared Imager

Aerosols will be monitored using the extinction of solar radiation. A 2-channel visible / near IR imager (VNI) provides sun images in two distinct spectral bands: 0.525 and 1.02 μm . A specific interference filter limits each spectral band.

At low altitude, important refractive index changes in the atmosphere deflect the sunrays, thus moving the sun images relative to the instrument axis. To overcome this, each channel consists of a 128 \times 128 effective photodetector array covering 60 mrad with a pixel separation of 0.50 mrad. This instantaneous FOV is 2 times smaller than the FTS FOV. These measurements will be done with a signal-to-noise ratio of 1000 for all sun-illuminated pixels for a 2-second observation time.

Suntracker

An important subsystem of the instrument is the suntracker, which provides the field-of-view tracking of the sun. It will autonomously keep the instrument looking at the sun radiometric center within an accuracy of 1 mrad. The suntracker is a mirror mounted on a gimbal whose elevation and azimuth are controlled by balancing the signals from a quadrant detector.

Orbit Selection

The selection of an appropriate orbit is critical for the ACE mission to achieve its science goals. Several orbit options were analyzed before the baseline mission orbit was chosen. These orbits included a 57° inclined, 65° inclined, and a 10:30 am sun-synchronous orbit. The baseline orbit was chosen to be circular with an inclination of 65° and an altitude of 650km. This orbit provides coverage of tropical, mid-latitude and polar regions. The vertical resolution will be about 3-4 km from the cloud tops (or the boundary layer for clear scenes) up to about 100 km. Because reference spectra of the sun will be recorded outside the earth's atmosphere, ACE will be self-calibrating.

Launch Segment

The ACE spacecraft, together with another TBD co-manifested spacecraft, will be launched on a Pegasus XL launch vehicle from the NASA's Western or Eastern Test Range in Q1, 2002.

Figure 3 shows the ACE spacecraft in the Pegasus fairing and the available envelope for the lower TBD spacecraft.

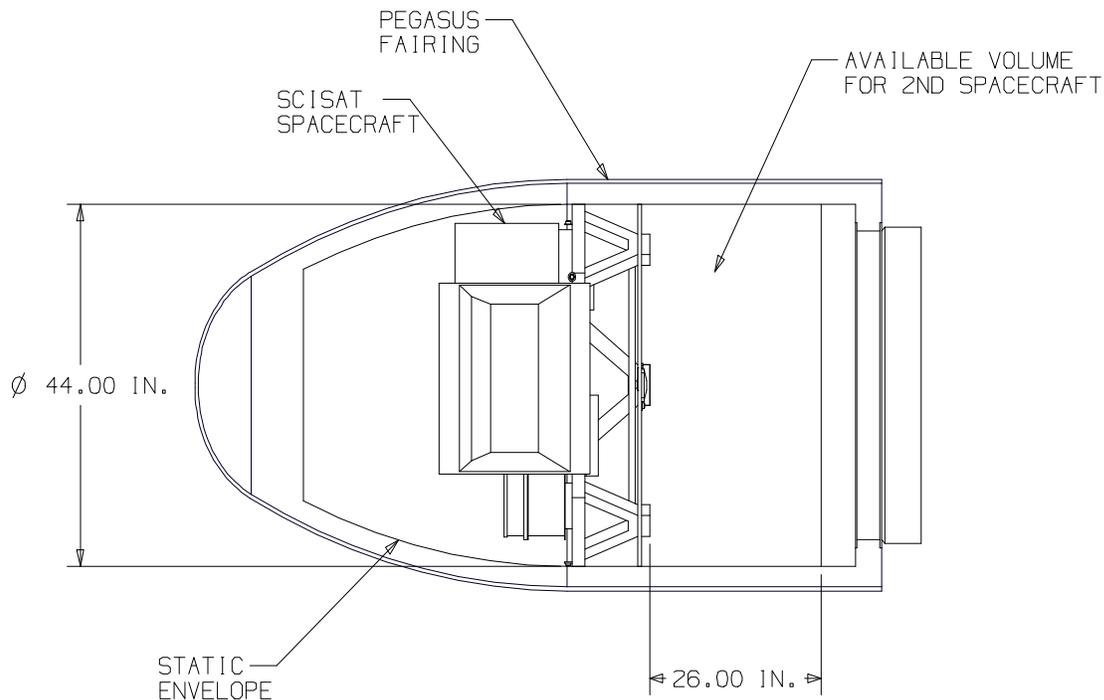


Figure 3 - SCISAT-ACE Pegasus Launch Configuration

Ground Segment

CSA Ground Segment requirements for flight operations, command and data handling and science operations are presently being defined. It is expected that flight operations and primary data reception will be conducted from existing facilities located at the CSA headquarters in St. Hubert, Quebec. Data reception will also be done at new facilities in Belgium. Science operations will be conducted from the University of Waterloo.

The SCISAT - ACE Team

The mission scientist and principal investigator is Dr. Peter Bernath of the University of Waterloo, Waterloo, Ontario. The FTS instrument prime contractor is Bomem, Incorporated of Quebec City, Quebec. Bristol Aerospace Limited of Winnipeg, Manitoba is the prime contractor for the spacecraft bus including bus design, instrument integration,

and spacecraft verification testing, and mission operations. Co-investigators are located at several Canadian universities, at the NASA Langley Research Center, and at research institutes in Belgium, Japan, France and Sweden

System Design Concept

Spacecraft Architecture

The ACE spacecraft will be comprised of the functional blocks as shown in Figure 4. Each block identifies the associated major components. The spacecraft will employ a single string architecture with a reliability goal of 0.80 over the 2 year mission lifetime.

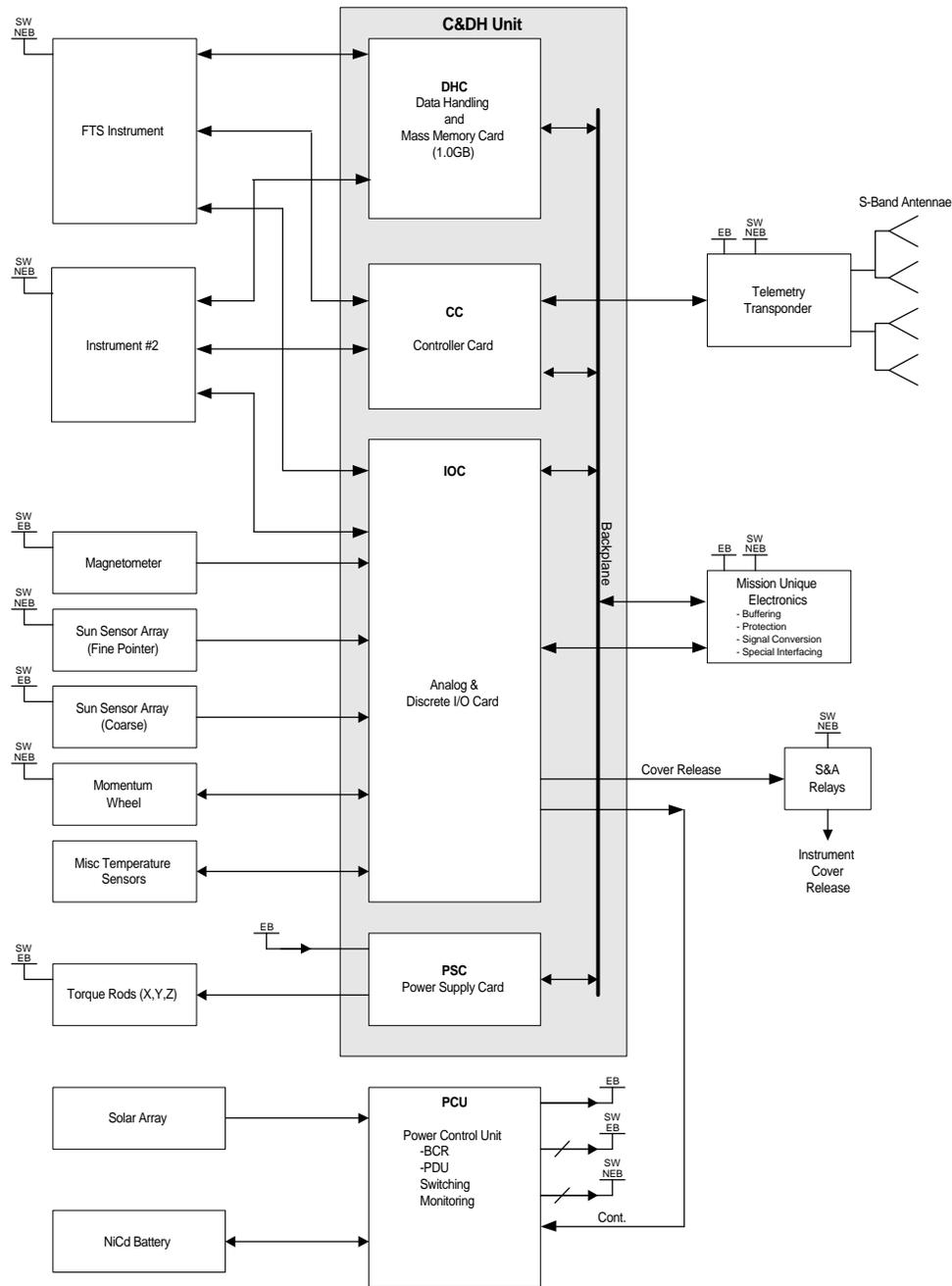


Figure 4- System Block Diagram

Spacecraft Layout

The spacecraft layout is shown in Figure 5. The layout shows the FTS instrument and a simple box for Instrument #2. The FTS instrument is mounted such that its field of view is pointed along the spacecraft's +X-axis. It is assumed at this time that Instrument #2 will have a similar field of view

and pointing requirement. The spacecraft maintains this axis constantly pointed at the sun, and is configured to have the major moment of inertia axis aligned with the sun vector for optimal stability. Cooling of the FTS instrument is accomplished by an instrument-provided radiator operating at cryogenic temperatures (about 100° K). The cryo-radiator is mounted so that it views

space along the spacecraft's +Z-axis. The spacecraft keeps the cryo-radiator away from the earth by rolling about the X-axis.

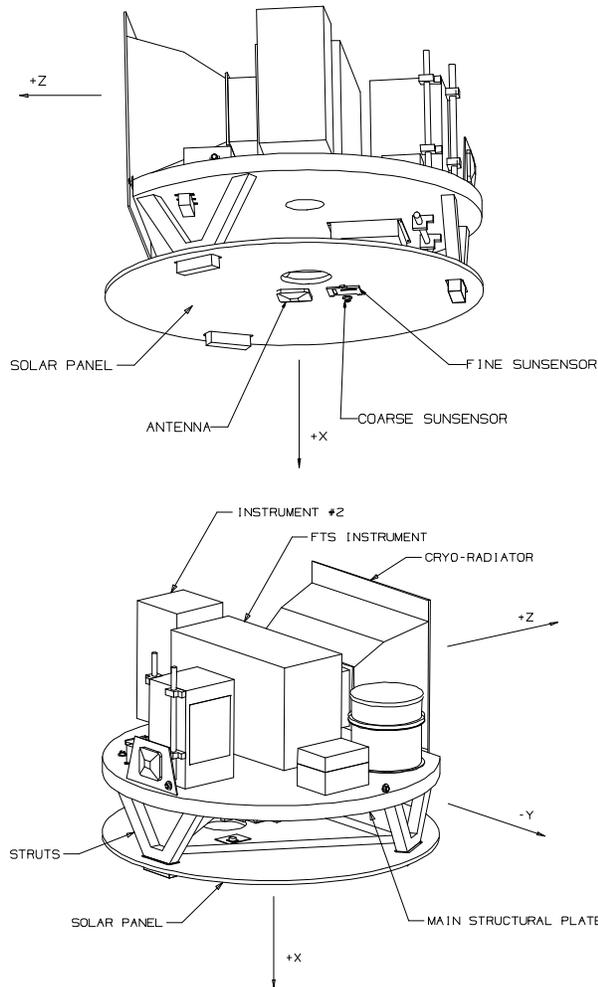


Figure 5 - ACE Spacecraft Layout

The solar array is fixed in position and supported by struts that provide some thermal isolation of the array from the main structural plate. The array is spaced well below the structural plate to allow for sufficient radiation view factor off the back of the array for cooling to ensure manageable array temperatures. The array also serves as a mount for one of the S-band telemetry antennae, a fine sun sensor, and one of the 6 coarse sun sensor elements. The instruments and the balance of the bus components mount to the main structural plate and are covered on the upper surface with multi-

layer insulation (MLI) blanket for thermal isolation from the 4°K temperature of space. The bottom of the plate is also covered with MLI to minimize parasitic heat transfer to the bus from the solar panel. An FTS instrument aperture penetrates through the structural plate and the solar array for sun viewing. It is intended that Instrument # 2 will use the same optical path as the FTS. In the layout shown in Figure 5 the FTS instrument is surrounded by the bus components including the battery, the momentum wheel, the transponder, and the spacecraft Command and Data Handling (C&DH) unit.

System Resource Budgets

The current allocation of mass and orbit average (OA) power to each of the spacecraft subsystems is shown in Table 1. The mass values include contingencies. The OA power values are based on a $\beta=0$ orbit plane (spacecraft in eclipse 36% of the time).

Table 1 - Spacecraft System-Level Budgets

<i>Subsystem</i>	<i>Mass (kg)</i>	<i>OA Power (W)</i>
Structure	48.8	-
Thermal Control	5.5	2.0
Power	16.8	-
Attitude Control	9.2	14.6
C&DH / MUE	5.8	10.0
Communications	4.0	6.3
Wire Harnessing	10.2	-
Payload	38.6	29.5
Power Contingency	-	8.2
Spacecraft Total	138.9	70.6

Subsystem Descriptions

Structure

The current mass budget should allow for a conservative, non optimized structure design that is both low-cost and low-risk. Emphasis will be placed on structure design with conservative safety factors, high stiffness and natural frequencies that will prevent interaction with the launch vehicle or lower spacecraft structural modes. This approach

is intended to reduce risk of major changes to the spacecraft later in the project due to instrument/component mass growth.

The majority of the bus components are mounted on the main structural instrument/component mounting plate which is a machined aluminum component rather than a honeycomb panel structure. While this approach may not be as mass-efficient as a honeycomb and face-sheet structure, the design, manufacture, inspection, testing and qualification of the machined structure will cost significantly less. Once the component placements are finalized, the structure will be evolved to employ attachment interfaces specific to the requirements for each component and each instrument, and then it will be trimmed to minimal size and mass.

The struts to support the solar array will provide for thermal isolation of the array from the rest of the spacecraft. This structure will employ materials with low thermal conductivity (eg. titanium alloys) for the struts themselves and will also employ isolation spacers (eg. G-10) and washers to further tailor the thermal transfer.

The panel substrate for the solar array is comprised of aluminum face sheets with vented aluminum honeycomb core. This provides a high structural natural frequency at a modest weight and cost. It also provides good heat transfer in the plane of the panel to help moderate temperature variations across the panel and support heat rejection to space.

Thermal Control

The spacecraft will utilize a passive thermal design, supplemented with heaters. Thermal control is achieved through the use of thermal paints, tapes, and MLI blankets. Passive thermal control is accomplished using appropriately sized and oriented radiator panels for the bus components and instruments. Spacecraft radiator surfaces are baselined as silver teflon flexible second surface mirrors or white paint, applied to the spacecraft structure and/or bus components. If possible, and depending upon the final component envelopes,

thermal, power and placement requirements, most of the bus components shall be conductively sunk to the common spacecraft baseplate to permit waste heat rejection through radiators looking in the +/- Y directions.

The ACE thermal control system will utilize two heater circuits; an operational circuit which will supplement the passive control during cold operational periods, if required, while a survival circuit will provide needed heat to the spacecraft during launch, safehold, and contingency situations, as required, to keep components above their survival/turn on limits. Heaters will be selected and sized based on location and bus voltage requirements. The heaters will likely be of the Kapton/FEP foil-type.

Due to its tight temperature limits, the battery will be thermally isolated from the rest of the spacecraft, cold-biased with its own radiator (looking +/-Y direction), and will use a heater system to precisely control its temperature. A low conductivity mounting structure and MLI blankets will serve to isolate the battery from the rest of the spacecraft.

The spacecraft C&DH Unit will interface with a set of thermistors to monitor spacecraft temperatures. Each subsystem component will have internal thermistors mounted in key locations. Other thermistors will be located throughout the spacecraft to monitor key structural and interface locations.

The body-mounted solar panel will be cooled passively with a radiator surface coating on the back of the array; likely a high emittance white paint. The solar panel will be thermally isolated from the structural support struts.

Power

The satellite will be sun pointing and power will be generated by a single body mounted solar panel. Power will be stored in a single NiCd battery. The power system will be controlled by the Power Control Unit (PCU) which consists of a Battery

Charge Regulator (BCR) and a Power Distribution Unit (PDU). The system uses an unregulated direct energy transfer architecture (i.e. the bus runs at the battery voltage during eclipses, and is shunt regulated during sunlight).

The maximum solar array size is limited to 44 inches diameter by the Pegasus fairing. The available area is reduced by a 0.25 inch keep out zone around the periphery, a cutout for the FTS instrument aperture, and a cutout to provide an isolated mount for a fine sun sensor, coarse sun sensor, and an antenna. The current baseline is to have Instrument # 2 use the same optical path as the FTS, and so a second instrument aperture cutout on the solar panel is not required.

From a detailed power and mission timeline analysis, it was determined that high efficiency Cascade cells would be required to meet the spacecraft power budget. Specifically, the solar panel will be populated with at least 14 strings of Cascade cells (21 cells per string). These cells are rated for continuous operation at 100°C. Preliminary thermal analysis indicates that peak cell temperatures should remain at or below 85°C throughout the mission.

Attitude Control

The ADCS configuration is based on a bias momentum stabilization approach. The subsystem consists of a momentum wheel, torque rods along all three body fixed axes, one fine sun sensor, a magnetometer and a set of six coarse sun sensors. All sensors and actuators are off-the-shelf components with flight heritage. A single string design is baselined to minimize cost, and still meet mission reliability requirements.

The ADCS operation consists of three main phases: capture, science and safe-hold. The capture phase starts after separation of the spacecraft from the launch vehicle. In this phase, the ADCS will align the spacecraft to a sun-pointing configuration using the torque rods, magnetometer and coarse sun sensors. During the science mode, the ADCS will maintain the sun-pointing orientation and satisfy the science pointing and stability requirements. A

bias momentum approach was selected for stabilization in this mode. The torque rods will be used to perform fine sun-pointing control as well as momentum dumping. The fine sun-pointing control will be performed only during the orbit day period.

In addition to the sun-pointing requirement, it is necessary to point the FTS instrument radiator away from the earth and sun. The sun avoidance is achieved by appropriate positioning of the radiator, i.e. positioning the radiator axis orthogonal to the sun-pointing axis. The earth avoidance is accomplished by rotating the spacecraft about the sun-pointing axis. This roll maneuver is performed by changing the wheel speed, with the roll angle being estimated using magnetometer data.

During emergency situations, the spacecraft will be placed in a safe-hold mode. The safe-hold mode can be entered from other ADCS modes by an autonomous on-board algorithm. All other ADCS mode changes will be performed by ground command.

C&DH Unit

The C&DH Unit is comprised of a controller card, a data handling card, an Input/Output Card, and a power supply card, all housed in a stacking aluminum frame. The C&DH Unit is being developed by Bristol for the SCISAT program. An evaluation model has been produced to verify design concepts and the engineering/flight models are currently in development. The EEE parts have been selected for function, low-power consumption, and radiation characteristics.

Controller Card

The controller card uses the Harris rad-hard RTX-2010 processor operating at 10 MHz and features 16 kbytes of boot ROM, 256 kbytes of SRAM, and 256 kbytes of EEPROM which can be reprogrammed in flight via the command link. It has an interface for the S-band STDN transponder and four general purpose RS-422 full duplex serial ports which can be configured to support synchronous or asynchronous communications. The card uses two Actel RT54SX32 FPGAs. One

implements support logic functions (i.e. address decoding, interrupt expansion, watch-dog timer, and the hardware based critical command reset decoder). The other implements Reed-Solomon encoding to achieve a 10^{-9} bit error rate on the downlink.

The card runs the application software for attitude, instrument, thermal, and telemetry control, power management, and higher levels of data management. Software development is facilitated by a new optimizing ANSI C cross compiler running under Windows 95 or NT.

Data Handling Card

The data handling card collects, stores and formats science and housekeeping data for transmission to the ground station. It has a high-speed parallel port and two RS-422 serial ports for data collection. It can provide up to 1 Gbyte of data storage. The memory is partitioned into separate areas for each science instrument and housekeeping data. Each area functions as a large First-In-First-Out (FIFO) memory. The card performs data time tagging and CCSDS data formatting with a maximum telemetry data rate of 5 Mbps.

The data is stored as 16 bit words with 6 parity bits to provide a single error correct, double error detect (SECDED) capability. The memory uses 32M x 8 bit SDRAMs from Samsung in eight high stacks. These SDRAMs tolerate up to 60 krad(Si) total dose. The shielding provided by the controller box keeps the total dose well below this for LEO missions. The memory is organized with redundant buses so that a failure which affects one of the buses does not cause the whole memory to fail. In addition, areas of 128 kbytes can be bypassed if failures are detected within them. Data handling is performed by two Actel RT54SX32 FPGAs. They handle DRAM refresh, EDAC, and memory scrubbing to correct single bit errors. They also generate the bit rate clock, format the data for transmission, and add optional convolutional coding. The data retention power for the 1.0 Gbyte mass memory is estimated at 2 W.

Input / Output Card

The I/O card provides analog and discrete I/O interfaces to the spacecraft ACS and diagnostic sensors to support spacecraft control and health monitoring. It provides 64 analog inputs with 12 bit resolution, four buffered analog outputs with 12 bit resolution, 16 discrete outputs, and 16 discrete inputs (both 5 V CMOS levels). The card accommodates differential and single ended analog signals of ± 5.0 V and ± 0.5 V and has a configuration area for termination resistors for 10k thermistors and resistive voltage attenuators.

Power Supply Card

The power supply operates from an unregulated 28 V bus (20 to 40 V) and provides +3.3 V, +5 V, and ± 15 volt for the C&DH Unit cards. The power estimate is 10 W orbit average and 15 W peak with 1 Gbyte of memory.

Development Status

The evaluation model cards for the controller card and data handling card are shown in Figure 6 and in Figure 7 respectively. Functional testing has been completed as well as a limited amount of environmental testing. It is envisaged that the flight unit will maintain the same form factor as the evaluation model.

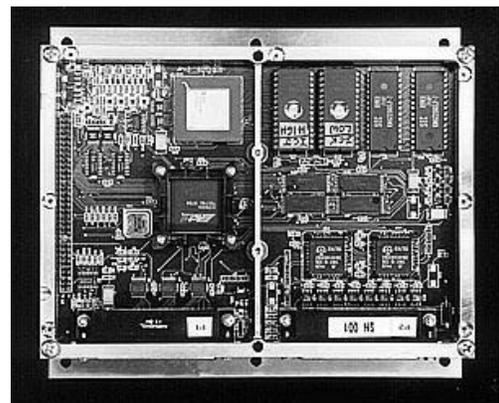


Figure 6 - C&DH Evaluation Unit Controller Card

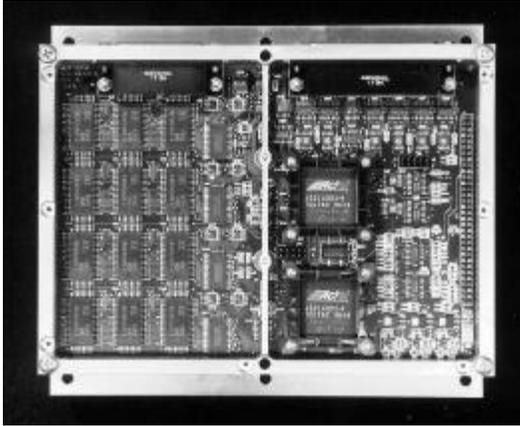


Figure 7 - C&DH Evaluation Unit Data Handling Card

Software

In keeping with the low cost approach for the satellite development, the flight software will implement minimal decision making - deferring control to the ground operations. The software will be developed in ANSI C, with low level routines written in assembler. With the low complexity concept for the software design a formal real time operating system is not required. Instead the flight software will implement a single threaded approach that will operate based on interrupt driven, preemptive, priority based operations and polled routines. The software will provide the executive operations necessary to support real time operations such as time line event management, communications, health and status monitoring, autonomous operations, and emergency operations.

Communication is achieved using the CCSDS recommendations for telecommand and packet telemetry (implemented in hardware for the telemetry down link) with the software being responsible for command uplink decoding, validation and distribution. During normal operations the flight software will be responsible for the autonomous control of the satellite when out of view of a ground station. This includes time line control, where the software schedules and initiates stored actions based on the spacecraft time.

The software will perform health and safety monitoring and will enter one of several predefined safe-hold modes in the event of anomalous

situations. This includes monitoring power system related modes, and allows the software to selectively shed loads in the event the power system diagnostics and sensor readings indicate power out of tolerance conditions.

The software provides the FTS instrument control interface and provides wake up signals to enable the instrument warm up period prior to measurements, and control commands for instrument configuration and status monitoring. The software can also command the FTS instrument to a sleep mode to conserve power. Data transfer out of the instrument is handled via the hardware.

Communications and Tracking

Communications and Tracking will be implemented through use of a NASA STDN (Satellite Tracking and Data Network) compatible transponder. The Communications and Tracking subsystem accepts a downlink bit-stream from the C&DH Unit and baseband modulates the S-band downlink carrier. It receives the S-band uplink, demodulates it and provides uplink data and clock to the C&DH Unit. The Communications and Tracking subsystem interfaces to uplink and downlink antennas directly at the transponder RF connector inputs.

The Communications function is accomplished via a single S-band uplink and a single S-band downlink. The downlink is a CCSDS-compatible digital bit stream comprising all spacecraft data. It is used to baseband modulate (PM or BPSK) the transponder. The uplink is a CCSDS-compatible digital bit stream modulating a STDN subcarrier on the uplink.

The Tracking function is accomplished through ground station Doppler measurements with the transponder operated in coherent mode. Ranging tone operation is not supported. Modulation scheme(s) compatible with both Communications and Tracking needs to be implemented. Tracking performance is dependent on both transponder and Ground Segment performance.

The Antenna Subsystem provides for uplink and downlink S-Band RF communications. It interfaces

to the Communications and Tracking Subsystem at the transponder RF connectors.

Conclusions

A new program has been initiated by the CSA to conduct innovative space science research with the use of small satellite platforms. The first mission of this program is the ACE Mission, which will use a spacecraft bus developed by Canadian industry. This program will lead to enhanced Canadian industrial capability in the development of low cost space systems.