NASA Contractor Report 145252

GE Report 77SDS4246

(NASA-CR-145252) A STUDY OF SELECTED ENVIRONMENTAL QUALITY REMOTE SENSORS FOR FREE FLYER MISSIONS LAUNCHED FROM THE SPACE SHUTTLE (General Electric Co.) 66 p HC A04/MF A01 CSCL 05B G3/43 N77-33585 Unclas 50214

A STUDY OF SELECTED ENVIRONMENTAL QUALITY REMOTE SENSORS FOR FREE FLYER MISSIONS LAUNCHED FROM THE SPACE SHUTTLE

H.W.Goldstein and R.N.Grenda

CONTRACT NAS1-13815
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1.0 INTRODUCTION

This report presents the results of "A Study of Selected Environmental Quality Remote Sensors for Free Flyer Missions Launched from the Space Shuttle" prepared for the National Aeronautics and Space Administration, Langley Research Center under Contract No. NAS1-13815 by the General Electric Co., Space Sciences Laboratory. In the initial phase of the study 15 sensors developed or being developed for the Advanced Applications Flight Experiments (AAFE) program, or available as hardware from other satellite programs, were investigated for their possible adaptability as payload components on a Multimission Modular Spacecraft (MMS) to be launched from the Shuttle.

The sensors were examined for adaptability to Shuttle by reviewing pertinent information regarding sensor characteristics as they related to the Shuttle and Multimission Modular Spacecraft environments. This included physical and electrical characteristics, data output and command requirements, attitude and orientation requirements, thermal and safety requirements, and adaptibility and modifications for space. The relevant sensor data employed in this study were gathered by sending a questionnaire to NASA technical contract monitors involved in the development of the sensors under study. These data were supplemented with available technical reports, and personal contacts with the contract monitors and the manufacturers where necessary to clarify technical details concerning equipment. Appendix A contains a list of the principal investigators and the manufacturers for each sensor considered.

Based on experience gained in designing and building sensors of the type being considered (e.g., CIMATS), and based on experience as an intergrator of many of the sensors on spacecraft (NIMBUS, LANDSAT), the sensor requirements and characteristics were compared with the corresponding Shuttle and Multimission Modular Spacecraft characteristics and capabilities. On this basis the adaptability and necessary modifications for each sensor were determined. A number of the sensors were examined in more detail and estimated cost for the modifications were provided by the manufacturers. These sensors included LACATE, CIMATS, VTPR, THIR, SBUV/TOMS, SAGE, LIMS and the VM cooler for LACATE and LIMS. Detailed task costing were provided for all these sensors, except THIR and SBUV/TOMS where only a total cost for modification were provided.

2.0 CONSTRAINTS FOR SENSOR MODIFICATIONS

For the purpose of this study, the instruments are assumed to be operating under the following constraints:

(1) The equipment must operate on the Multimission Modular Spacecraft (MMS) which is carried into orbit by the Space Shuttle and then released and retrieved at the appropriate times and then returned from orbit.

(2) The mission duration is for a period of 12 months.

(3) There will be no instrument operation until the satellite has been launched from the Shuttle.

(4) The Multimission Modular Spacecraft orbit will have an inclination in the range 25 to 70 degrees and an altitude ranging from 500 km.

(5) The flight instrument hardware will be derived from modification of existing hardware or from hardware to be developed under current program schedules.

(6) The modifications are intended to accomplish the following objectives:

  (a) Enable the instrument to make its design measurements in the automated mode on the Multimission Modular Spacecraft. The modifications are NOT intended to upgrade the experiment science.

  (b) Permit the instruments to survive the Space Shuttle and space environments.

  (c) Insure that the instruments are compatible with Space Shuttle safety requirements.

2.1 Spacecraft Characteristics

The spacecraft selected for the purposes of this study is the Multimission Modular Spacecraft (MMS)(ref. 1). The spacecraft characteristics of interest for compatibility and sensor modification purposes are related to the electrical power system, the data interface, and the telemetry and command systems. The launch and return-from-orbit environments also dictate the modifications necessary to insure survivability. These parameters are summarized in Appendix A. The details of mechanical and thermal interfacing with the spacecraft are not considered since the spacecraft experiment module definition is not firm at this time. If a particular sensor appears to have potential mechanical or thermal interface problems then this has been noted.

The Multimission Modular Spacecraft Bus shown in Figure 1 includes the following subsystems required to provide spacecraft services:

(1) Mechanical System

(2) Thermal System

(3) Power Subsystem

(4) Attitude Control Subsystem (ACS)

(5) Communications and Data Handling Subsystems (C&DH)

(6) Electrical System

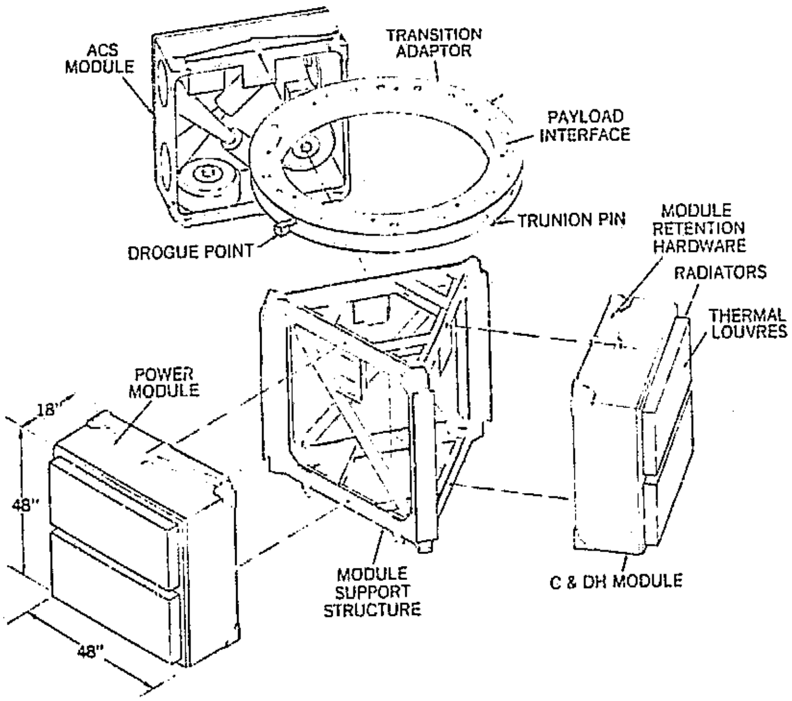

Figure 1. Multimission Modular Spacecraft

The Mechanical System consists of a module support structure which provides a basic framework for mounting the Attitude Control Subsystem, Communications and Data Handling Subsystem, and power modules. A transition adaptor provides the payload attachment points and a capture drogue point which allows the Shuttle to capture and deploy the spacecraft. The module support structure also supports mating connectors and electrical harnessing. All of the mission unique payloads interface with the top of the transition adaptor.

When the payload configuration is not compatible with the standard interfaces, then a mission adaptor provides the solar array restraints and the drive interface.

The thermal control system is intended to maintain the temperature of the primary spacecraft systems within acceptable limits for a variety of missions ranging from low orbits (500 to 1665 km) to geosynchronous. The experiment module is insulated from the Multimission Modular Spacecraft and must have its own thermal control system.

The Power Subsystem provides a source of unregulated power operating at 28 ± 7 VDC negative ground. The primary source of power is a deployable solar panel array mounted to the spacecraft structure. Power needs exceeding the solar array output are provided by up to three 20 ampere-hour to 50 ampere-hour NiCd batteries. The power subsystem can support an average load of 1200 watts in any orbit from 500 to 1665 km and at geosynchronous altitude. 350 watts are required to support spacecraft loads and the remainder is available for payload operation.

The Attitude Control Subsystem orients and stabilizes the spacecraft relative to a desired target, whether it be nadir viewing or a celestrial target. An on-board computer located in the Communications and Data Handling Module processes information derived from a sensor complement which consists of:

(1) An inertial reference unit.

(2) A pair of fixed star trackers.

(3) A three-axis magnetometer.

(4) Two coarse sun sensors located on the solar array.

(5) One precision digital sun sensor located in the Attitude Control Subsystem module.

Based on the sensor information, the computer generates appropriate control signals to operate the reaction control devices. Payload instruments which have fine error sensors can be used to enhance the control performance.

The Attitude Control Subsystem will control the spacecraft to within 0.01 degrees of a specified inertial reference. Short term attitude perturbations will be less than 2 seconds of arc with a maximum long term attitude drift of less than 10^{-6} degrees per second.

The Communications and Data Handling Subsystem provides the means for ground and on-board control of all spacecraft and sensor functions and for relaying spacecraft housekeeping and payload sensor data. Tape recorders are not condidered to be standard equipment but may be flown if considered necessary for a particular mission. An interface has been provided for the NASA Standard Recorders (10^8 or 10^9 bit versions). In addition, six square feet of mounting surface is available for mission dependent equipment. Telemetry equipment has been designed to cover a wide range of mission bit rate requirements, with rates of 1, 2, 4, 8, 16, 32, or 64 Kbps being selectable by command. The higher bit rates required by imaging sensors may be accommodated with specific mission hardware. The basic command rate is 2000 bps with a 48 bit word although rates of 1000 and 125 bps are also provided. A command table may be loaded into the on-board computer in order to permit the spacecraft to operate without ground intervention for 24 to 72 hours.

3.0 SENSORS

The sensors selected for this study include sensors developed for the Advanced Application Flight Experiments (AAFE) program, and hardware remaining from various satellite programs. When NIMBUS sensors are discussed it is assumed that a complete sensor identical to the NIMBUS flight unit is available for modification. Table I lists the sensors and indicates their source.

TABLE I

REMOTE SENSORS EVALUATED FOR A MMS MISSION LAUNCHED FROM SHUTTLE

APPS	Aerosol Physical Properties	AAFE
LACATE	Lower Atmosphere Composition and Temperature Experiment	AAFE
CIMATS	Correlation Interferometry for Measurement of Atmospheric Trace Species	AAFE
VTPR	Vertical Temperature Profile Radiometer	NOAA 2 B/U
THIR	Temperature Humidity Infrared Radiometer	NIMBUS G
SBUV/TOMS	Solar Backscatter UV/Total Ozone Measurement System	NIMBUS G B/U
SAGE	Stratospheric Aerosol and Gas Experiment	AEM - B FLIGHT
IHS	Infrared Heterodyne Spectrometer	AAFE
MAPS	Monitoring of Air Pollution from Satellite	AAFE
MAPS	Monitoring of Air Pollution from Satellite	NIMBUS G B/B
MOCS	Multichannel Ocean Color Sensor	AAFE
CZCS	Coastal Zone Color Sensor	NIMBUS G B/U
ESP	Eclectic Satellite Pyroheliometer	AAFE
MTS	Microwave Temperature Sounder	AAFE
LIMS	Limb Infrared Monitor of the Stratosphere	NIMBUS G B/U

3.1 Sensor Parameters

The sensor parameters of initial interest in interfacing with the spacecraft are size, weight, and power. Other considerations such as sensor data rate, spacecraft pointing and stabilization requirements, command requirements, detector cooling requirements also bear heavily on the adaptation of a sensor for interfacing with a Multimission Modular Spacecraft to be launched from Shuttle.

This spacecraft permits the interfacing of various power, data handling modules and sensor modules into a basic bus. As a result the spacecraft capabilities can be adapted to many sensors or combinations of sensors.

The physical characteristics of the sensors being evaluated are summarized in Table II.

3.2 Sensor Modifications

The Advanced Application Flight Experiments (AAFE) are generally designed for an aircraft or balloon type of flight environment which is considerably less demanding than that to be encountered during a Shuttle mission. Even experiments which were designed with satellite operation as an ultimate objective were probably not intended to survive the high random vibration levels resulting from the acoustic environment inside the Shuttle cargo bay. As a result the adaptation of these experiments requires careful consideration if program costs are to be kept at a minimum. In this study consideration was given to economical approaches to implementation of the experiments on a Multimission Module Spacecraft without incurring undue risks to safety and performance.

The dynamic environment during boost, reentry and landing is particularly severe, relative to the design criteria that was used in the development of the Advanced Application Flight Experiments equipment. For instance, random vibration levels of 0.25 to 0.6 G^2/Hz have been predicted for pallet-mounted payloads of 100 Kg to 25 Kg. Typical levels for conventional equipment is 0.04 G^2/Hz.

Random vibration inside the cargo bay results from the acoustic environment acting on the external surface of the fuselage, and the attenuation characteristics of the payload support structure and its interfaces with the actual equipment. The specified overall acoustic level inside the cargo bay is 145 dB, although it is entirely possible that this level will be exceeded. These uncertainties suggest that the approach to dynamic environment protection in the program include safeguards against unforseen environmental and structural factors that may render the mechanical interface design unsatisfactory.

A dual approach is suggested:

(1) Components will be analyzed for potential susceptibility to damage from acoustically and mechanically induced vibration or shock. The susceptible components will be reinforced or de-tuned as necessary.

(2) The experiment assembly or even the entire payload module will be partially isolated from the environment by using an acoustic enclosure; also, vibration isolators will be used as the mechanical interface between the assembly structure and the mount structure.

TABLE II

PHYSICAL CHARACTERISTICS OF SENSORS

SENSOR		SIZE LENGTH (cm)	LENGTH (in)	WIDTH (cm)	WIDTH (in)	HEIGHT (cm)	HEIGHT (in)	WEIGHT (kg)	WEIGHT (lbs)	VOLTAGE	POWER (watts)
APPS		35.6	14.0	34.6	13.6	55.9	22.0	19.6	43.3	+28 VDC	16.3
LACATE	SENSOR/COOLER	112	44.1	50D	19.7D			45	99.5	+28 to	75
	SCAN ASSEMBLY	56	22.1	30D	11.8D					+35 VDC	90
CIMATS	SENSOR	76.2	30	39.4	15.5	45.7	18	73.8	163		
	FORE-OPTICS	42.5	16.75	15.2D	6D					+28 VDC	280
	ELECTRONICS	48.3	19	28	11	26.6	10.5	6.8	15		
	SUITCASE	48.3	19	48.3	19	22.8	9	22.6	50		
VTPR		46.0	18.1	22.1	8.7	29.5	11.6	9.0	20	-26 VDC	7.5
IHS	SENSOR HEAD	55.9	22	91.5	36	30.5	12	67.9	150		
	ELECT.PROCESSORS	50.8	20	48.2	19	50.8	20	33.9	75		
	LASER P/S	43.2	17	48.2	19	71.1	28	83.2	184		
	LASER CONTROL	20.3	8	48.2	19	16.5	6.5	10.9	24	110 VAC	2400
	MIXER BIAS CONT.	22.8	9	48.2	19	12.7	5	6.8	15		
	LASER CODER	38.1	15	33.0	13	40.6	16	33.9	75		
THIR	SENSOR	40.0	15.7	25.4	10	28.2	11.1	6.4	14.1	-24 VDC	8.5
	ELECTRONICS	16.5	6.5	17.8	7	15.2	6	2.6	5.8		
SBUV/TOMS		47.2	18.6	36	14.2	53	20.9	44.8	99.0	-24.5 VDC	20
SAGE		71.1	28	45.7	18	71.1	28	30.1	66.5	+28 VDC	25
MAPS (AAFE)		125	49.2	75	29.5	75	29.5	125	276	+28 VDC	150
MAPS (NIMBUS)	SENSOR	91	35.8	53	20.9	48	18.9	34	75.2	110 VAC	750
	ELECTRONICS	71	28.0	63	24.8	107	42.1	114	252		
MOCS		48.2	19	15.2	6	19.1	7.5	11.3	25	+28 VDC	8.5
CZCS		86.8	34.2	60.6	23.9	55.1	21.7	39.3	87	-24.5 VDC	46
ESP	SENSOR	33.5	13.1	9.0	3.4	18.8	7.4	9.6	21.2	+28 VDC	15
	ELECTRONICS	26.7	10.5	14.7	5.8	25.0	9.8	3.8	8.4		
MTS	RF ASSEMBLY #1	40.7	16	15.2	6	15.2	6	7.4	16.3		
	RF ASSEMBLY #2	40.7	16	15.2	6	15.2	6	5.2	11.4		
	SCAN ASSEMBLY	40.7	16	15.2	6	10.2	4	7.3	16.1	110 VAC	57
	VIDEO IF	40.7	16	15.2	6	5.1	2	3.7	8.1		
	PROGRAMMER/MUX	40.7	16	15.2	6	10.2	4	8.6	19.1		
	POWER SUPPLY	45.7	18	20.3	8	20.3	8	21.3	47.1		
LIMS	FRAME HOUSING	52	20.5	64	25.2	108	42.5	50.0	112.0		
	MMS MOUNTING	63.5	25.0	78	30.7	117	46.1	9.0	20.2		
	INTERFACE ELECTRONICS UNIT	21	8.3	18.5	7.3	18	7.1	5.0	11.2	-24	26
	FRAME ELECTRONICS UNIT	18.5	7.3	16	6.3	18	7.1	5.5	12.4	-24	
	COOLER CONTROL MODULE	21	8.3	16	6.3	18.5	7.3	5.0	11.2	+28	90
	MMS INTERFACE ELECTRONICS UNIT	21	8.3	16	6.3	18	7.1	7.0	15.5	+28	25

The acoustic enclosure recommended for this application (ref. 2) consists of a box-like or cylindrically shaped shield, sized according to the instrument or payload dimensions and constructed of visco-elastic epoxy. A sandwich-type construction should be used, consisting of a material such as AMRD-100 F90 visco-elastic epoxy strips connecting two metallic sheets. Acoustically absorptive material is placed on the inside wall of the enclosure to prevent noise reverberation. It is estimated that an overall reduction of 20 dB can be attained, with a frequency reduction of 30 dB at the high and low frequency extremes.

The proposed vibration isolators would connect the experiment structure with hard-points on the mounting structure. The bracket will be designed to provide controlled spring constants in three dimensions. The overall functions of the brackets will be to provide firm support under all mission conditions, de-tune the experiment assembly structure with respect to structural resonances, and dampen vibration in critical modes.

In general all of the sensors will require the following analyses to determine their adaptability and the extent of the modifications:

(1) A thermal analysis to determine whether the current sensor thermal control system, whether it is active or passive in nature, is adequate to maintain the sensor and its electronics within its allowable operating and nonoperating temperature range.

(2) A mechanical analysis to determine whether the sensor and its components are able to withstand the expected acceleration, shock, and vibrational loads imposed by the spacecraft during launch and re-entry.

(3) A mission study to determine what effect a typical orbit may have on the sensor output.

(4) A modifications design study to examine in detail the problem areas which have been identified for each sensor and to identify less obvious areas of modification.

The design constraints with respect to available power, telemetry, and command interface requirements for the Multimission Modular Spacecraft subsystems, and the thermal and mechanical environments of the Shuttle during launch and re-entry are set forth in Appendix B.

With regard to the sensors which require conversion to operate from positive input voltage, it cannot be determined a priori how one should modify a sensor that has been constructed to operate at a negative voltage to operate at a positive voltage. There are several choices that can be considered: (a) a DC to DC power supply to convert the available positive 28 volts to negative 24 volts; (b) rework of selected electronics so as to be able to operate at positive 28 volts; and (c) simple changes as reversing leads which might be satisfactory for some motors. The main disadvantage to (a) is that a large amount of power must be dissipated in the converter, which requires a heat sink, and in the case of the Multimission Modular Spacecraft limited power is available. Thus, a converter does not appear to be a very satisfactory approach. Some sensors which are being considered in this study may have been originally designed for operation at positive 28 volts and were modified for NIMBUS operation, thus the changes to sensors of this type should be straightforward.

The most power effective and cost effective approach would be to investigate each sensor in a detailed study to determine the approach or combination of approaches to use for modification of a specific sensor for operation at positive 28 volts. In addition, sensors operating at negative 24 volts will also require a change (inversion) of voltage in their digital data system.

By considering the state of sensor development, and its characteristics and operating requirements as related to Shuttle and the Multimission Modular Spacecraft, the modifications and potential problem areas for the sensors in Table I are outlined in the following sections.

3.2.1 Aerosol Physical Properties (APPS).- The APPS instrument is a multispectral photometer which measures the scattered solar radiation from the earth's limb within narrow spectral bands in the near ultraviolet and visible (Figure 2). The sensor is intended to monitor the spatial and temporal variations in aerosol physical properties, ozone, nitrogen dioxide, and neutral density in the stratosphere and lower mesosphere by inverting the photometric data obtained as the sensor scans vertically through the earth's limb. The spectral regions of interest and the constituents measured are:

WAVELENGTH mμ	SPECTRAL BANDWIDTH mμ	CONSTITUENT
310	20	O_3
340	10	O_3, Neutral Density
400	10	NO_2
420	10	Neutral Density
500	10	Aerosol, Neutral Density
600	10	Aerosol
700	10	Aerosol
900	10	Aerosol

The APPS program has been terminated upon completion of the Instrument Design Phase of the contract. Since the development phase of the contract is not scheduled for support at this time, it is not expected that hardware will be available for inclusion in a shuttle payload

3.2.2 Lower Atmosphere Composition and Temperature Experiment (LACATE).- The LACATE sensor (Figure 3) is a multichannel infrared radiometer employing an array of ten HgCdTe detectors cooled by liquid nitrogen (ref. 3). It is designed with a two-axis scan system which permits both an elevation scan of

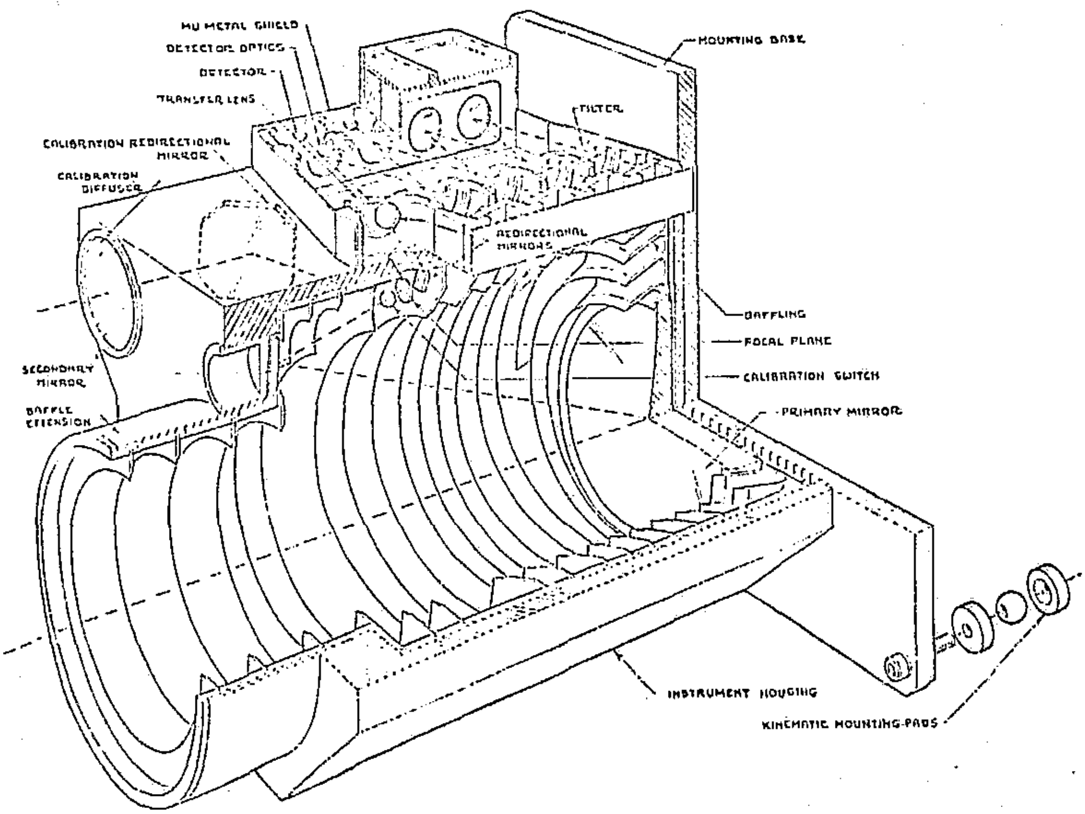

Figure 2. Aerosol Physical Properties Instrument

the Earth's limb, and a 180° azmiuth scan of the horizon. Measurements are made in the following spectral bands:

CHANNEL	SPECTRAL BAND (μm)	FIELD OF VIEW (milliradians)
NO_2	6.076 - 6.440	1 x 2.5
H_2O	6.373 - 7.246	1 x 2.5
CH_4	7.938 - 8.335	1 x 2.5
O_3	8.644 - 10.595	0.5 x 2.5
AEROSOLS	10.326 - 10.806	1 x 2.5
HNO_3 (2)	10.794 - 11.696	0.5 x 2.5
CO_2	13.158 - 17.291	0.25 x 2.5
CO_2	14.770 - 15.870	0.25 x 2.5
N_2O	16.391 - 17.906	0.5 x 2.5

The spectral radiance profiles are operated on by inversion algorithms to determine stratospheric temperature and concentration profiles of NO_2, H_2O, CH_4, O_3, HNO_3, N_2O, and aerosols.

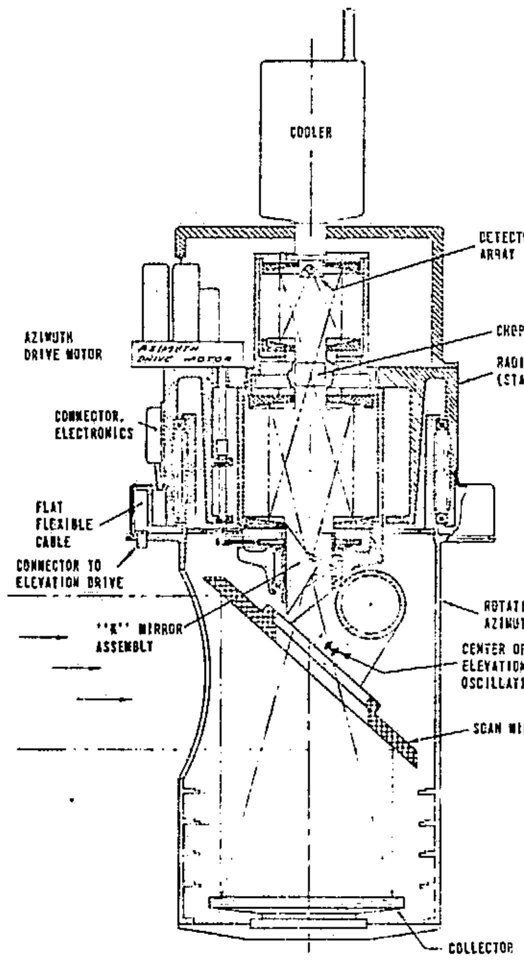

Figure 3. LACATE Instrument

The LACATE engineering model, which has been tested on a high altitude balloon, is currently available for refurbishment and modification. Since some of the sensor systems were designed for the balloon test, a number of modifications are necessary to adapt the sensor to a spacecraft. These modifications include the following:

(1) Design, fabricate, assemble and test a detector capsule assembly which will interface with the LACATE and with the VM cooler (ref. 4) which will be supplied. Design and fabricate a mechanical interface for the cooler to the LACATE. Integrate and test the detector-cooler subsystem.

(2) Develop a VM cooler system for automated spacecraft operation with a 1 year lifetime.

(3) Design and fabricate a VM cooler control unit to power the cooler, and control the temperature produced at the detector. This includes provision for output of housekeeping data, and command functions.

(4) Design a heat sink to dissipate the 70 to 100 watts of power from the VM cooler alone.

(5) Redesign and rework the azimuth scan assembly to eliminate a bearing malfunction; redesign and rebuild the scan mirror servo system using the LIMS design to obtain smoother scan performance.

(6) The existing scan angle range is sufficient for the Multimission Modular Spacecraft mission provided that the center of the scan range is properly boresighted by the spacecraft mounting structure.

(7) The sensor housing was lightened for the balloon flight and will probably require strengthening to withstand the launch environment. The initial design study will determine the extent of the modifications.

(8) The data handling system requires redesign to eliminate the filler bits currently inserted into the data stream. A system based on the LIMS sensor sampling concept is recommended. The command system requires redesign to be compatible with the Multimission Modular Spacecraft.

(9) The LACATE commercial power supply requires replacement with a redesigned power supply based on the LIMS dc-dc converter design. All LACATE electronics circuits will require repackaging, including new board layout and design, package design, board fabrication and checkout, package assembly and functional tests.

(10) A mechanical adaptor plate is required to mechanically interface LACATE to the Multimission Modular Spacecraft and incorporate the boresight pointing requirements dictated by the orbital altitude.

(11) Following its balloon flight, the LACATE sensor requires a general refurbishment, including:

(a) clean optics
(b) recoat K mirror
(c) reposition chopper
(d) off-axis rejection analysis.

The sensor data rate of 8 Kbps is well within the capabilities of the C&DH subsystem. Similarly, the requirements of maximum pointing accuracy of $0.5°$, and stabilization of $0.003\ °/\mathrm{sec}$ are attainable with the ACS subsystem.

3.2.3 Correlation Interferometry for Measurement of Atmospheric Trace Species (CIMATS).- The CIMATS sensor is a type of field-widened Michelson interferometer in which the optical path difference is scanned by means of an oscillating scan plate of refractive material. It operates in two spectral regions; the nonthermal infrared from 2 to 2.4 μm, and the thermal infrared from 4 to 9 μm. A PbS detector operating at 195 °K is used in the 2 to 2.4 μm spectral channel, and a HgCdTe detector operating at 77 °K is installed in the 4 to 9 μm channel. A filter wheel in the optical path of each channel is capable of containing 5 narrowband spectral filters, thus providing the capability of making 10 different spectral measurements. Two measurements, one in each channel, are made simultaneously with a measurement time of approximately 1.3 seconds. This sensor is capable of operating in both the nadir viewing mode with a 7° field of view, and in the solar looking earth limb mode, with appropriate sun tracking. Figure 4 is a plan view of the CIMATS sensor currently being laboratory tested.

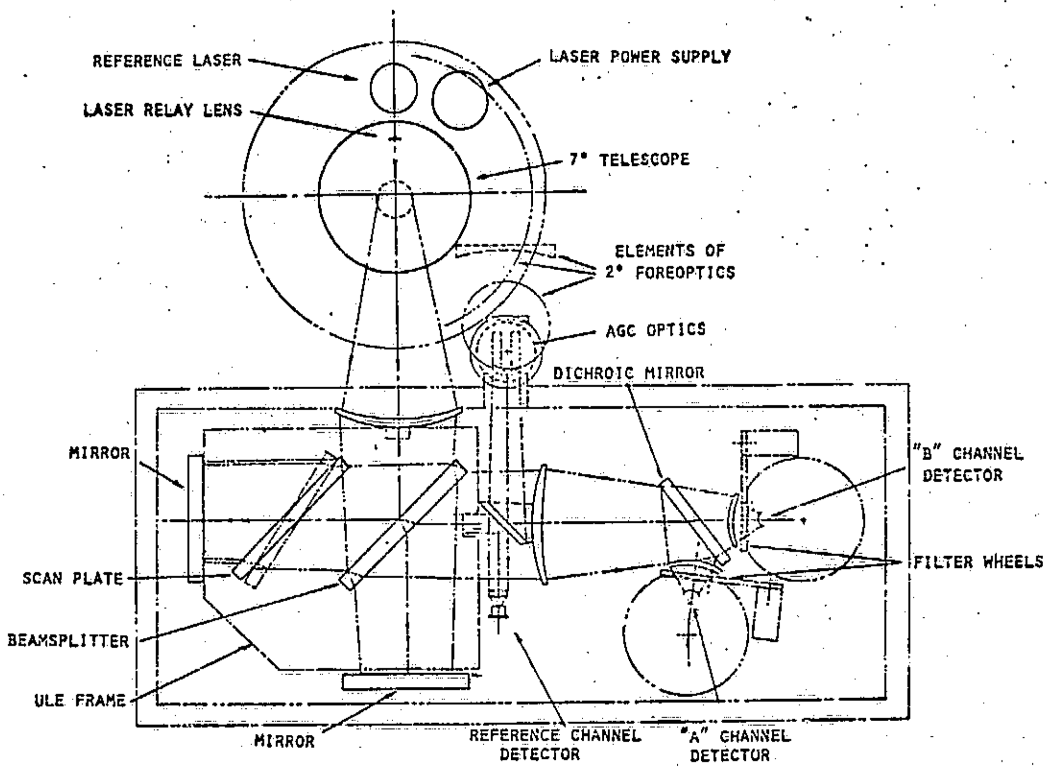

Figure 4. CIMATS Instrument

The CIMATS sensor has been designed for an aircraft or balloon type of environment. The sensor was designed with power supplies and monitoring and display functions built into one electronics rack and the sensor electronics built into a second electronics rack. This enables the minimum amount of electronics which must be flown with the sensor to be readily separated from the total electronics package. The modifications necessary to adapt the sensor to an automated spacecraft are the following:

(1) Redesign of the LN_2 and dry ice cooled detectors for replacement with detectors cooled by other means. The easiest to implement is the incorporation of thermoelectric cooling for the PbS, and, if the long wavelength channel is restricted to 6 μm, the substitution of thermoelectrically cooled PbSe for the HgCdTe detector system. If the 4 to 9 μm capability is necessary, then other possibilities for the HgCdTe detector include the use of a closed cycle VM cooler, or solid cryogen cooler. Any of the systems considered will require additional power and control circuitry to be designed.

(2) Redesign of the electronics using low power components and repackaging of the circuits and power supplies.

(3) Replacement of filter wheel drive motors with vacuum qualified units. The present scan plate torque motor is suitable but a space qualified version is available.

(4) Repackaging of the reference laser and power supply to prevent arcing and corona at low pressures. Redesign to include a redundant dual laser system appears advisable to provide long term reliability.

(5) Analysis and modification of the thermal control system and thermal insulation for adequacy on the spacecraft. Provision for heat dissipation from the power supplies and electronics.

(6) Design and fabrication of a command system and data interface to the spacecraft is required to permit automated operation.

(7) Analysis and redesign of the AGC chopper system to adapt to the vibration environment.

(8) Design of a spacecraft mounting interface.

Since the sensor was designed for interfacing to a digital tape recorder and a minicomputer, the data output is all digital with an output rate of less than 3000 bps. A data frame, corresponding to a single interferogram scan, consists of 242, 12 bit words, which includes housekeeping functions. This data rate presents no problems to the Multimission Modular Spacecraft data system.

The sensor pointing requirements are not excessive, with pointing accuracy being 0.5° and the motion rates being 0.1 °/sec. These requirements are easily accomodated by the spacecraft ACS system.

3.2.4 Vertical Temperature Profile Radiometer (VTPR).- The VTPR is an infrared scanning radiometer designed to make highly accurate measurements of the atmospheric infrared in 8 narrow spectral bands between 11 and 19 μm (Figure 5). Each spectral radiance measurement corresponds to radiation coming predominantly from a particular altitude above the earth's surface. From these radiance measurements in the atmospheric CO_2 band and the 18.7 μm H_2O band, the vertical temperature profile is computed.

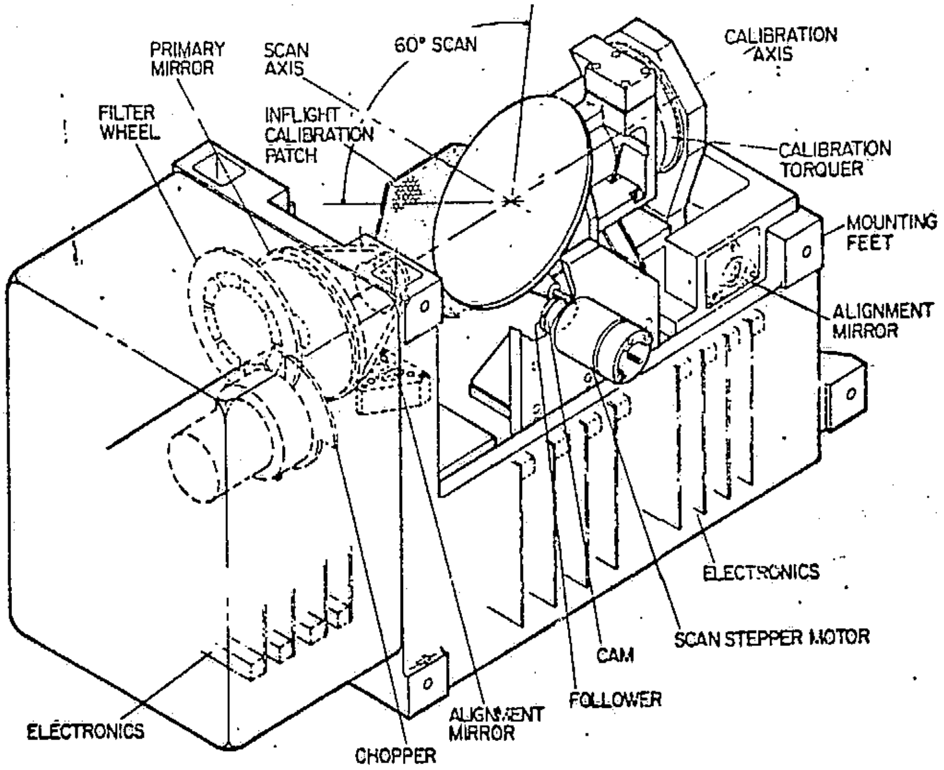

Figure 5. VTPR Instrument

The sensor has an instantaneous field of view of 2.2° x 2.2° and scans across the orbit track, ± 31.7° about the nadir, in 23 equal steps. A complete scan takes 12.5 seconds, one second of which is devoted to retrace of the scan mirror (ref. 5).

VTPR is a sensor which has been previously flown on the ITOS satellite, and 2 units are currently available, lacking only environmental testing. Since the equipment is designed to operate in an automated mode on a spacecraft, the modifications are minimal and primarily electrical in nature. The modifications which have presently been considered to be necessary are the following:

(1) The most cost effective approach to modifying the electronics is to design an interface between the sensor and the spacecraft in-

corporating the following functions:

(a) A switching pre-regulator and DC/DC converter to permit operation from +28 VDC spacecraft power.
(b) Signal conditioning converters for all VTPR analog signals.
(c) A clock interface to generate VTPR clock rates and gate.
(d) A command decoder to interface with VTPR commands.
(e) A data interface to buffer VTPR data and put it into the Multimission Modular Spacecraft format.
(f) A caging circuit battery system to provide power during launch.

(2) In order to compensate for the particular orbital altitude selected, the only economically practical modifications appear to be:

(a) A change in the image compensation focal plane mask, and
(b) an increase in step scan speed (at the expense of reduced sensitivity.

(3) The sensor requires a look at cold space during part of each scan, a factor which will have to be considered during interfacing with the spacecraft, but which does not present fundamental difficulties.

(4) If the mission requires complete ground coverage, then it may be necessary to redesign the crosstrack scan angle range, which would entail substantial redesign.

The digital data rate is only 512 bps, which can be easily accommodated by the Multimission Modular Spacecraft data handling systems. The requirements for pointing accuracy of ± 0.2° and the motion rate of 0.01 degree/sec, which were derived from the corresponding values for the ITOS satellite on which the VTPR was previously flown, also can be met by the Multimission Modular Spacecraft attitude control system.

3.2.5 Temperature Humidity Infrared Radiometer (THIR).- The THIR is a two-channel scanning radiometer which contains a 6.5 to 7.0 μm (water vapor) channel and a 10.5 to 12.5 μm (atmospheric transmission window) channel (Figure 6). The spectral radiance measurements in these two spectral bands provide the following information on a 24 hour basis:

(1) Pictures of cloud cover.
(2) Three-dimensional mapping of cloud cover.
(3) Temperature mapping of clouds, land, and ocean surfaces.
(4) Cirrus cloud content and contamination.
(5) Relative humidity.

The 6.5 μm channel has a 21 milliradian field of view and the 10.5 μm channel has a 7.0 milliradian field of view (ref. 6). The two channels use common primary optics with the incident energy being separated by a dichroic beamsplitter which directs the radiation to separate bolometers. Cross track scanning is provided by a scan mirror which rotates at 48 rpm. The sensor

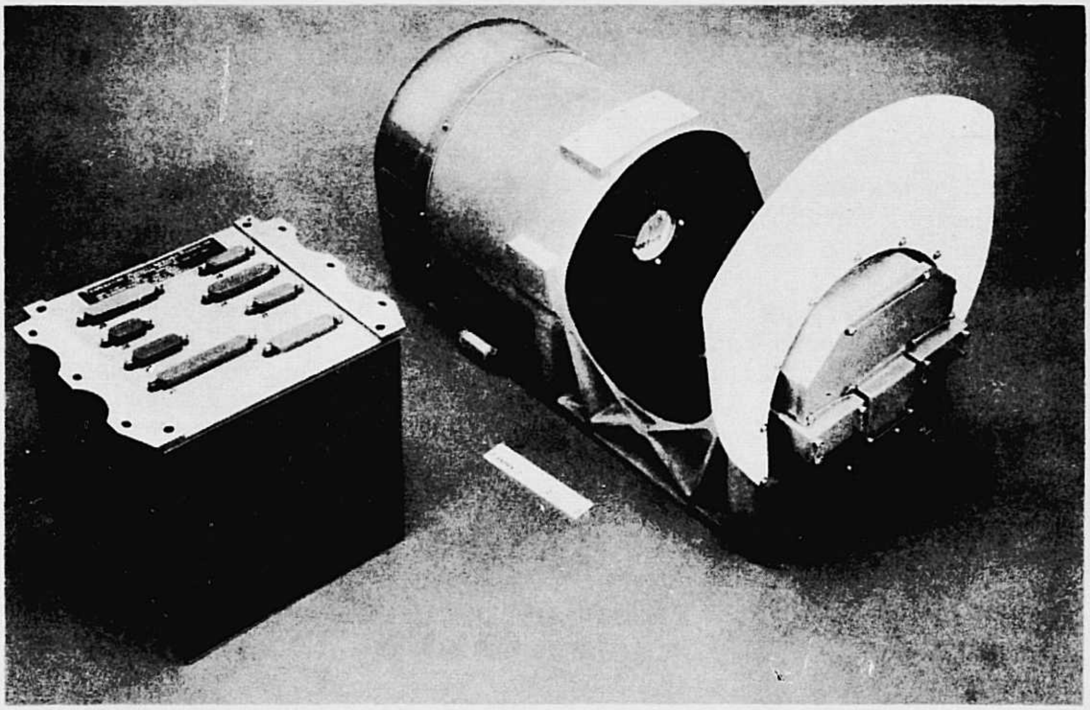

Figure 6. Temperature Humidity Infrared Radiometer (THIR)

has an unobstructed scan opening of ± 75° about the nadir. At an altitude of 740 km the earth subtends an arc of 128° and at 1100 km, it subtends 117°. Thus the sensor is able to look at cold space above the horizon during each scan.

Since the THIR sensor will be flown on NIMBUS G the mechanical changes are expected to be minimal. The principal modifications are:

(1) Redesign of the power supplies to operate from +28 VDC instead of the -24.5 VDC on NIMBUS.

(2) Redesign of the scan motor power supply which requires a 100 HZ clock signal as an input.

(3) Inversion of the analog sensor outputs which are currently in the 0 to -6 volt range.

(4) Redesign of the command and data system interfaces. The Multimission Modular Spacecraft analog data system may not provide sufficient accuracy with only 8 bit digitization. Digitization of the analog data with greater accuracy will require new circuitry and output digital data interfaces.

If the output data is digital, then the data rate is estimated to be 5 Kbps, which is within the Multimission Modular Spacecraft capabilities. Pointing requirements and spacecraft angular motion rates are also within the Multimission Modular Spacecraft attitude control system capabilities.

3.2.6 Solar Backscatter Ultraviolet/Total Ozone Measurement Spectrometer (SBUV/TOMS).- The SBUV/TOMS is a dual sensor which provides incident solar ultraviolet radiation and ultraviolet radiation backscattered from the earth with high photometric stability, in order to allow long term monitoring of the total ozone distribution and the vertical ozone profile (ref. 7).

The SBUV module (Figure 7) is a double grating monochrometer which measures the ultraviolet earth radiance in the nadir direction. The field of view is 11.33 degrees and measurements are made at 12 selected wavelengths from 2500 to 3400Å in a step scan mode, or in a continuous scan from 1600 to 4000Å. At appropriate positions in the orbit, a diffuser plate can be deployed on command for the measurement of solar irradiance.

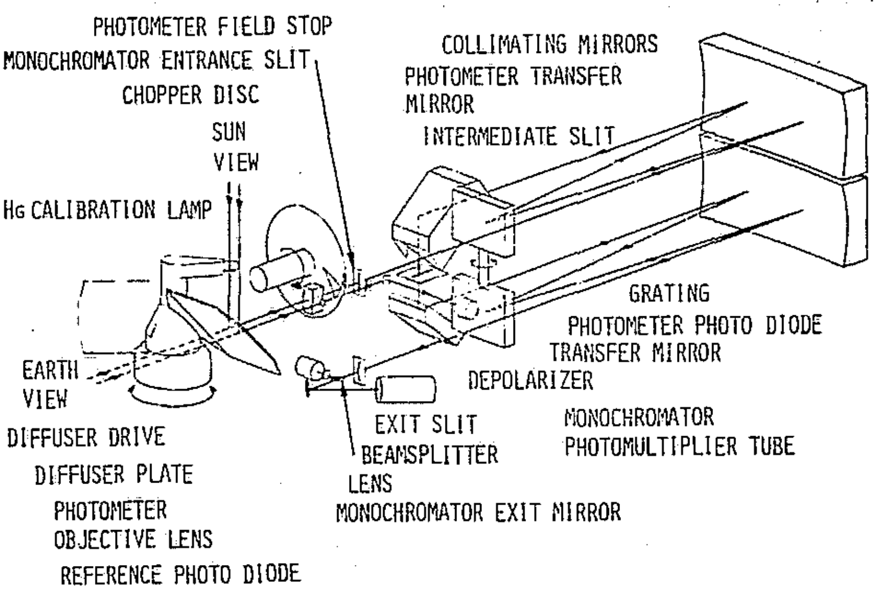

Figure 7. SBUV Optics Diagram

The TOMS module (Figure 8) consists of a single grating spectrometer that scans the earth by means of a stepped scanning mirror whose scan rates and angles are selected to provide spatially continuous observations with the 3 degree sensor field of view. The wavelengths are mechanically step scanned in 12 steps from 2555 to 3398Å or continuously scanned from 1600 to 4000Å.

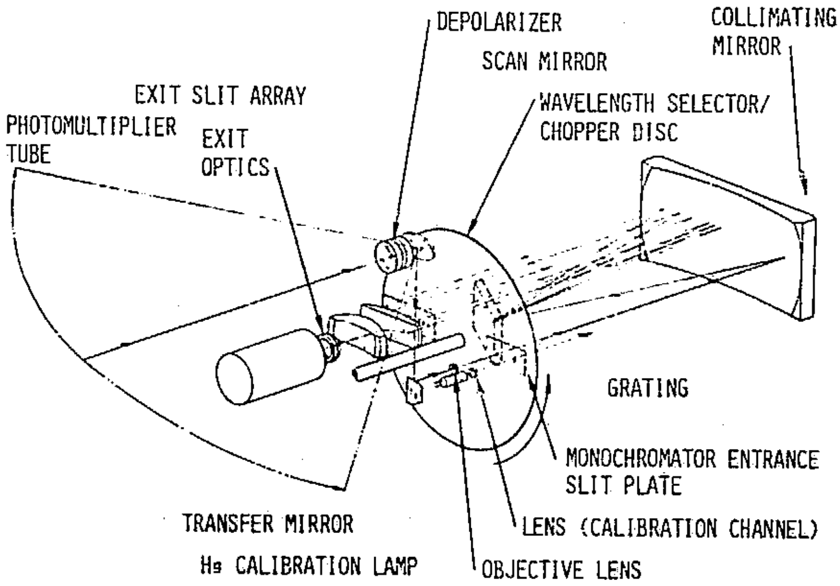

Figure 8. TOMS Optics Diagram

The SBUV/TOMS sensor is scheduled to be flown on the NIMBUS G spacecraft. An engineering model is available for modification. It has commercial type electronics (high reliability in most cases) in its electronics module. Since the instrument has been designed for automated spacecraft operation, the mechanical design modifications do not appear to be significant. The electronic changes appear to be substantial due to the sensor data format, and the large number of electrical interfaces with the NIMBUS. It appears that the most practical approach is to design and build an Interface Module to interface between the existing design and the Multimission Modular Spacecraft. The list of modifications include the following:

(1) Design of a power converter to provide -28 VDC power for the sensor.

(2) Design of command translators to stretch the command pulses to 40 ms duration, and convert them to the proper levels required for SBUV/TOMS.

(3) Design of a 400 kHz clock circuit with the waveform locked to the 1.024 MHz clock from the Multimission Modular Spacecraft.

(4) Design of translators for the analog, digital B, and thermistor telemetry to provide proper voltage output levels.

(5) Design of a translator for the digital A telemetry output to provide timing and reformat the data for the Multimission Modular Spacecraft.

(6) The sensor is susceptible to contamination of the optical surfaces and may require a removable cover for the optics if the shuttle bay environment is not acceptable.

(7) The TOMS field of view is intended to provide global coverage from NIMBUS. The lower inclined orbit for the Multimission Modular Spacecraft mission may not provide adequate coverage unless the scan angle is changed. A mission study will determine whether this is necessary.

(8) While the SBUV is nadir looking, a calibrated diffuser must be periodically solar illuminated, a factor which must be considered during integration into the spacecraft.

After re-formatting, the digital data rate for the SBUV/TOMS is only 1000 bps, which is within the Multimission Modular Spacecraft data system capabilities. In addition, the orientation and stabilization requirements of ± 0.25° and 0.028°/sec, respectively, are also fulfilled by the spacecraft attitude control system.

3.2.7 Stratospheric Aerosol and Gas Experiment (SAGE).- SAGE is a radiometer that measures stratospheric aerosols, ozone, and nitrogen dioxide as a function of altitude, and latitude and longitude. The instrument makes measurements of the attenuation of the sun's radiation as it passes through the earth's atmospheric limb in four spectral intervals: namely, 1.0, 0.6, 0.45, and 0.38 μm. The sensor scans the sun during sunrise and sunset with a circular field of view of 0.47 milliradians, which is capable of providing a 1 km vertical resolution in the atmosphere for these measurements. Figure 9 illustrates the details of the SAGE instrument.

The SAGE sensor is being constructed for flight on the AEM-II satellite with the engineering model being delivered in October 1977 and the protoflight model in July 1978. Since the sensor has been designed for the launch environment associated with the Scout vehicle, major mechanical modifications are not anticipated. The following electrical modifications may be required:

(1) The SAGE input power requirement is +28 ± 4 VDC while the MMS output is +28 ± 7 VDC which means that the SAGE requires some regulation of input power to reduce the variations to an acceptable value.

(2) The command system requires some redesign. The SAGE 32 bit serial word may not be acceptable to the Multimission Modular Spacecraft system. Also, the Multimission Modular Spacecraft command signal for a logical "1" has a range of +2.4 to +5.0 while the SAGE system minimum value is +3.5 V. Some changes to set the minimum at +2.4 V will be necessary. In addition the switch closure signals for Multimission Modular Spacecraft are inverted from those required for the SAGE system. Circuitry changes to provide a pull-up resistor with a diode when driving solid state logic are necessary.

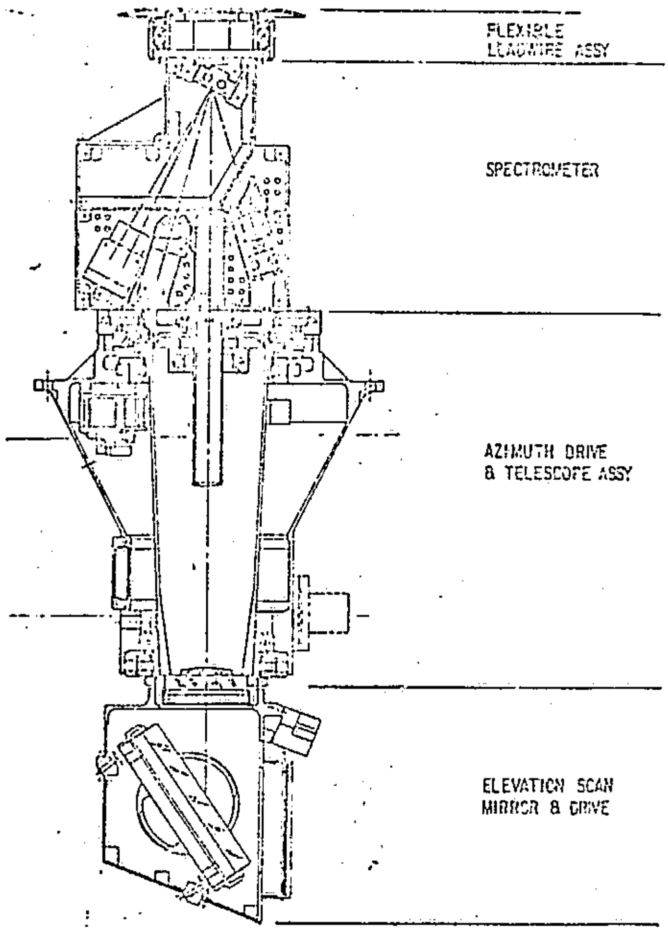

Figure 9. SAGE Sensor Assembly

(3) The Multimission Modular Spacecraft clock frequency is not the same as required for the SAGE. Circuit changes to count down the Multimission Modular Spacecraft clock, or to provide an independent clock will be necessary.

(4) Since the SAGE only outputs data during sunrise and sunset, data may be lost unless transmission coverage is continuous. A recorder may be considered necessary if continuous data coverage is not possible.

(5) The optical components are sensitive to contamination and will require a remotely removable cover.

(6) The SAGE optics and electronics are meant to be flown on a specially designed mounting structure. If this mount is not used, then analysis will be required to determine whether structural changes to the sensor are necessary.

(7) If the sensor package is thermally isolated from the spacecraft, then no changes are required in the thermal control system.

(8) The elevation scan range for SAGE is designed for an orbital range of 600 ± 200 km. If the Multimission Modular Spacecraft orbit is significantly outside of this range then the elevation scan range will require changing.

The pointing accuracies of ± 1° for pitch and roll, and ± 2° for yaw are well within the pointing capabilities of the Multimission Modular Spacecraft attitude control system, as is the stabilization of 0.001 °/sec. The sensor is capable of a 360° azimuth scan of the horizon which may present problems when the sensor is interfaced in the payload with other sensors, but presents no fundamental integration problems. The data rate of 8 Kbps is also within the Multimission Modular Spacecraft data system capabilities.

3.2.8 Infrared Heterodyne Spectrometer (IHS).- The technique of infrared heterodyne detection provides a means for determining the atmospheric spectral radiance and solar absorption with spectral resolution capable of scanning individual atmospheric spectral lines. This permits the concentration profiles of minor atmospheric species to be derived. The IHS is a 4 channel heterodyne spectrometer with a 2 GHz bandwidth designed for ground based and airborne measurements of atmospheric NH_3 and O_3 in both the atmospheric emission (nadir viewing) and solar radiance viewing modes. The instrument, shown in Figure 10, uses two HgCdTe photomixers cooled with LN_2, two grating tunable $C^{13}O_2^{16}$ laser local oscillators, four IF networks with RF filters which spectrally channelize the incident irradiance, two black-body sources for measurement reference and absolute calibration, and four radiometer processing channels (ref. 8). For NH_3 measurements in both viewing modes the laser transitions at 927.300 cm^{-1} and 920.219 cm^{-1} are used as the pollutant and reference local oscillator transitions. For O_3, the pollutant and reference local oscillator transitions selected are 1034.318 cm^{-1} and 1043.473 cm^{-1}, respectively, for the nadir viewing mode, and 1048.866 cm^{-1} and 1043.163 cm^{-1} for the solar looking mode.

The IHS was designed for flight test on the CV990. Flight testing occurred May 1977 and both solar and nadir viewing experiments were conducted. The sensor presently operates from 120 VAC and draws substantial power (2400 W). As a result a satellite version of the sensor will require electrical redesign to reduce the power requirements significantly. The sensor modifications include the following:

(1) Redesign the electronics to reduce power and permit operation from +28 VDC.

(2) Design solid cryogen cooler for the optical photomixers. A one year mission may require a VM cooler.

(3) A conductive cooling system for the lasers and power supply must be designed. This may require heat pipes and a radiative cooler if the vehicle structure cannot absorb the added heat load.

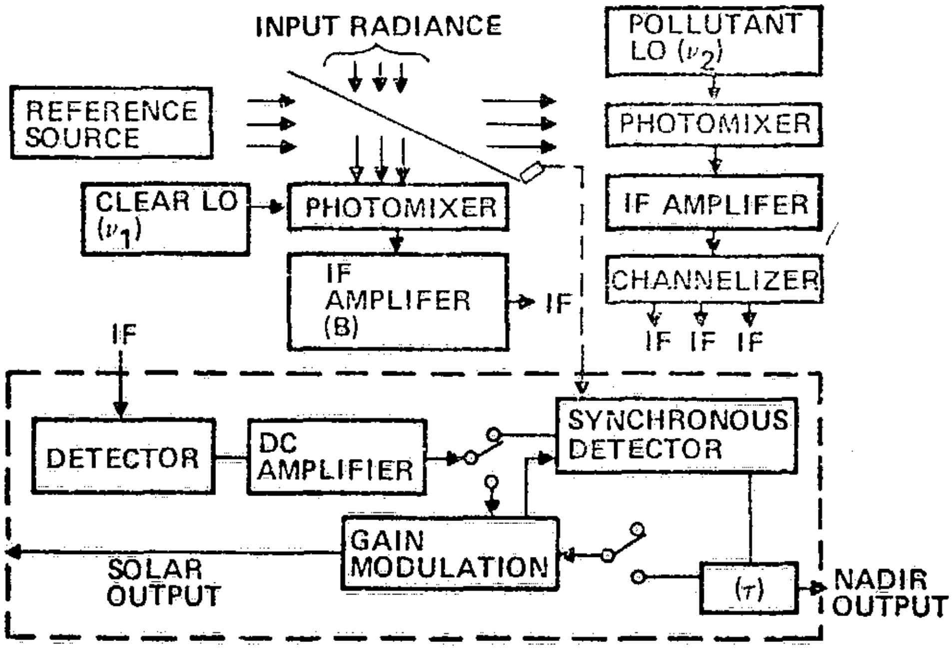

Figure 10. Simplified Block Diagram of IHS

(4) The sensor will operate to an altitude of 10,000 feet. The high voltage components must be insulated to prevent arcing at low pressure.

(5) The IHS requires a pointing system to maintain the solar disc within the field of view. The pointing accuracy required is ± 0.05 degrees with a rate change of 2 arc seconds/second.

(6) The laser and infrared sensor head may require some mechanical redesign to prevent misalignment during the shuttle launch.

(7) The sensor calibration procedure must be automated.

(8) Laser tuning and optimization electronics must be incorporated.

(9) A command and data interface is required. The analog data requires accuracy better than that present on the Multimission Modular Spacecraft. The analog data may require sampling, digitizing, and multiplexing circuitry.

The spacecraft pointing and stabilization are adequate to maintain the proper sensor attitude even if it does not have its own pointing system. However other considerations may dictate that the sensor include a dedicated pointing system. The data rate, including possible housekeeping functions, will be of the order of 200 bps, and does not significantly load the spacecraft data handling system.

3.2.9 Monitoring of Air Pollution from Satellites (MAPS-AAFE).- The MAPS instrument uses the non-dispersive infrared technique of gas filter correlation to selectively absorb the radiation emitted by the pollutant component of the atmospheric gas mixture (ref. 9). A reference cell containing the specified pollutant to be measured, is used to modulate the incident radiation in order to separate out the pollutant component. The sensor is designed to be used in a nadir viewing mode to provide radiance data, from which species concentration profiles may be derived (Figure 11). The MAPS sensor measures CO using the spectral band at 4.6 µm and has an instantaneous field of view of 7°.

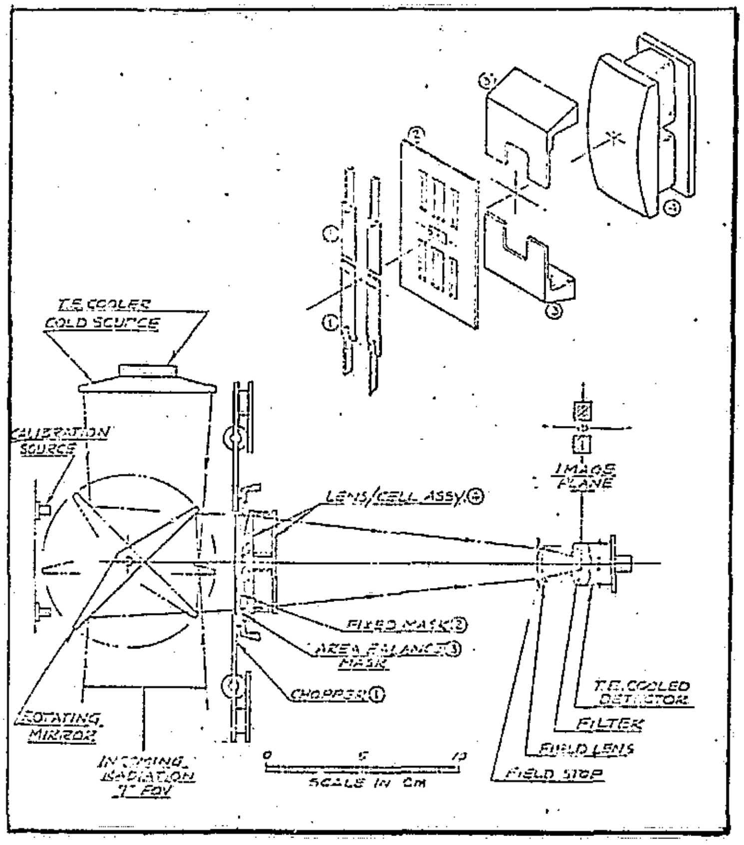

Figure 11. Schematic of the Gas Filter Correlation Instrument

The MAPS sensor developed under the AAFE program has been aircraft tested and was designed for operation from +28 VDC power. The modifications necessary to adapt it to a spacecraft are the following:

(1) Design and fabrication of a command system (approximately 10 commands) and interface.

(2) Design and fabrication of a data system to perform A/D conversion of the radiometric signals (26) and housekeeping functions (30), and format the data output for input to the spacecraft data system.

(3) The sensor must be modified for mechanical and thermal integration with the spacecraft.

(4) A stable temperature sink (approximately 300 °K) must be designed to permit the thermoelectric coolers to dissipate 30 watts of power.

The sensor data rate after digitizing will be of the order of 1000 bps and hence well within the spacecraft capabilities. The pointing requirements of 5° from nadir and stabilization of 1 °/min also present no problems of implementation.

3.2.10 Monitoring Air Pollution from Satellites (MAPS-NIMBUS G).- The MAPS sensor is an infrared radiometer which employs the principle of gas filter correlation to extract the target gas radiation from the background. The NIMBUS G brassboard was designed as a single channel instrument, capable of being configured for either CO or NH_3 measurements at 4.66 μm and 11.1 μm, respectively (ref. 10). Used in a nadir viewing mode, the sensor provides radiance data which permits species concentration profiles to be derived. The detector system are comprised of thermoelectrically cooled PbSe for the CO measurement and TGS for the NH_3 measurement. The sensor has a field of view of 4.33°. The sensor optical head is shown in Figure 12.

This version of the MAPS sensor was constructed as a NIMBUS G brassboard and has been tested on an aircraft. The present electronics operate from 110 volt, 60 Hz AC power and require redesign. The following modifications are necessary:

(1) Redesign the electronics to operate from + 28 VDC power.

(2) Redesign electrically and mechanically to permit sensor operation in vacuum.

(3) The sensor optical head alignment may be affected by the shuttle launch environment and the optical head may have to be modified to prevent misalignment from occurring.

(4) The present data system is all analog with up to 10 bit digitizing requirement, which exceeds the spacecraft analog capabilities. For this data an A/D converter system and data formatter will be required. The lesser accuracy analog data may still use the spacecraft system if desired.

(5) A command control system is required to permit automated sensor operation.

(6) An automated balancing and calibration system is also required.

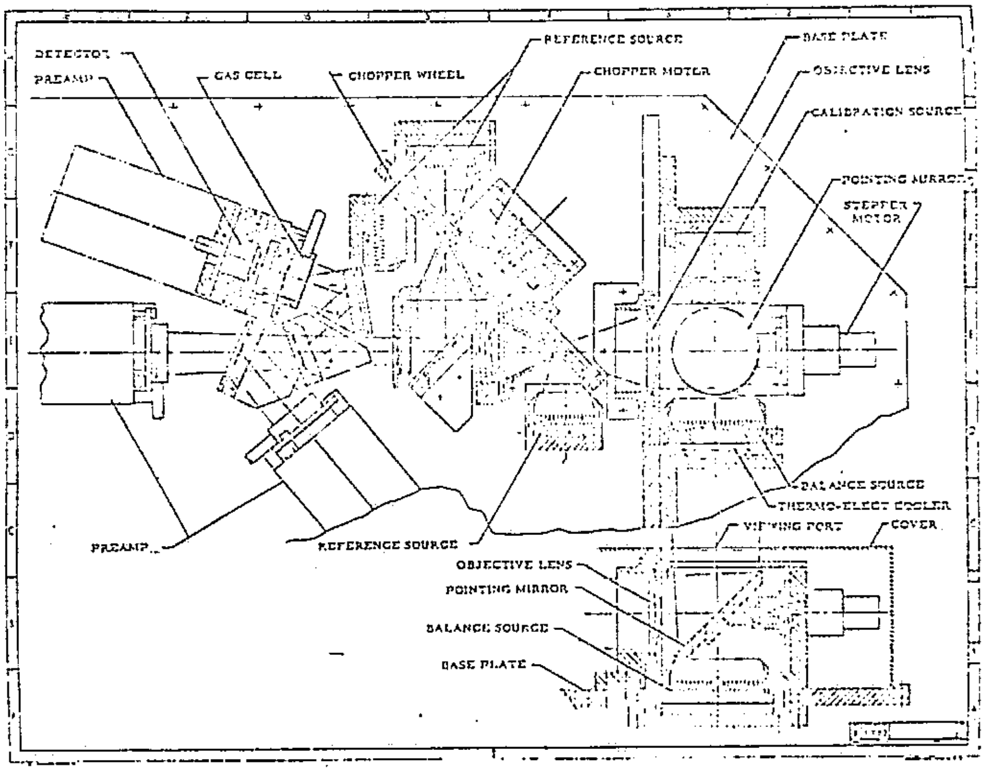

Figure 12. MAPS-NIMBUS G Electric-Optical Head Layout

The sensor data rate is about 500 bps which presents no problem to the spacecraft system. The pointing requirement of 5° from the nadir and stabilization of 1°/min are also easily accommodated.

3.2.11 Multichannel Ocean Color Sensor (MOCS).- The MOCS is a spectroradiometer designed for measuring small differences in ocean color from space in the spectral range 0.40 to 0.70 μm (ref. 11). It has two operating modes which require some manual disassembly to implement. In one mode, the electronically scanned MOCS records spectral radiance in 20 spectral bands at each of 150 spatial sites within the instantaneous field of view of 17.1 degrees. In the second mode, 60 spectral bands at each of 50 spatial sites are measured. Figure 13 indicates the MOCS optical configuration.

The MOCS sensor developed for the AAFE program is currently being used in an aircraft flight test program. Modifications to this sensor are not considered to be major with only minor electronic interfacing considered necessary.

(1) A problem in interfacing with the standard spacecraft data system concerns the current high data rate. The sensor output is 114 Kbps, which exceeds the spacecraft capabilities. This can be accommodated by flying a non-standard data system with sufficient capacity, or redesigning the sensor output to limit the data rate

to 64 Kbps. In either case, some redesign or interface circuitry will be necessary.

(2) The sensor requires a command control system for command functions.

The attitude requirements of 0.01° for pointing accuracy, and 2 x 10^{-5} °/sec for stabilization are near the limits of the spacecraft ACS subsystem.

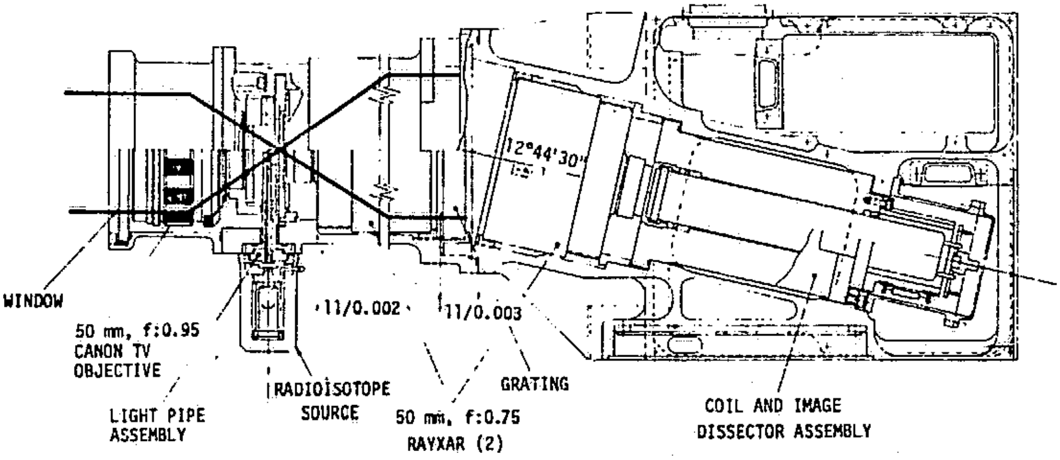

Figure 13. MOCS Optical Configuration

3.2.12 Coastal Zone Color Sensor (CZCS).- The CZCS experiment will measure ocean and coastal zone water color at five spectral bands centered at 0.443, 0.520, 0.550, 0.670 and 0.750 μm and water surface temperature at 11.5 μm (ref. 12). The instantaneous field of view for each of the spectral channels is 0.865 milliradians. Cross track scanning of the orbital path is provided by a scan mirror rotating at 485 rpm, which provides an unobscured scan range of ±40 degrees from nadir. The scan field can also be tilted, in 2 degree increments; a total of ± 20 degrees along the orbital track. Spectral dispersion is accomplished with a Wadsworth-type grating spectrometer which directs the radiation to 5 separate silicon photodiode detectors for the short wavelength channels. The energy in the 11.5 μm channel is separated in the foreoptics by means of a beamsplitter which directs the radiation to a radiatively cooled HgCdTe detector. Figure 14 depicts the CZCS sensor.

The CZCS has been developed for flight on NIMBUS G, with the backup sensor being available for conversion to a Shuttle free-flyer mission. Since the sensor was specifically designed for spacecraft, mechanical modifications are believed to be minimal. The electrical system, however, requires the following modifications:

(1) Redesign of the electronics to operate from +28 VDC.

(2) A new data and command system interface will probably be necessary to interface with a higher capacity data system. Depending on the characteristics of the data system, the modifications may involve the analog data circuits, the Digital B outputs, and circuits utilizing the NIMBUS clock frequencies.

(3) The HgCdTe detector employs a radiative cooler which may present a problem in interfacing since the orbital plane is not sun synchronous, as is the NIMBUS orbit.

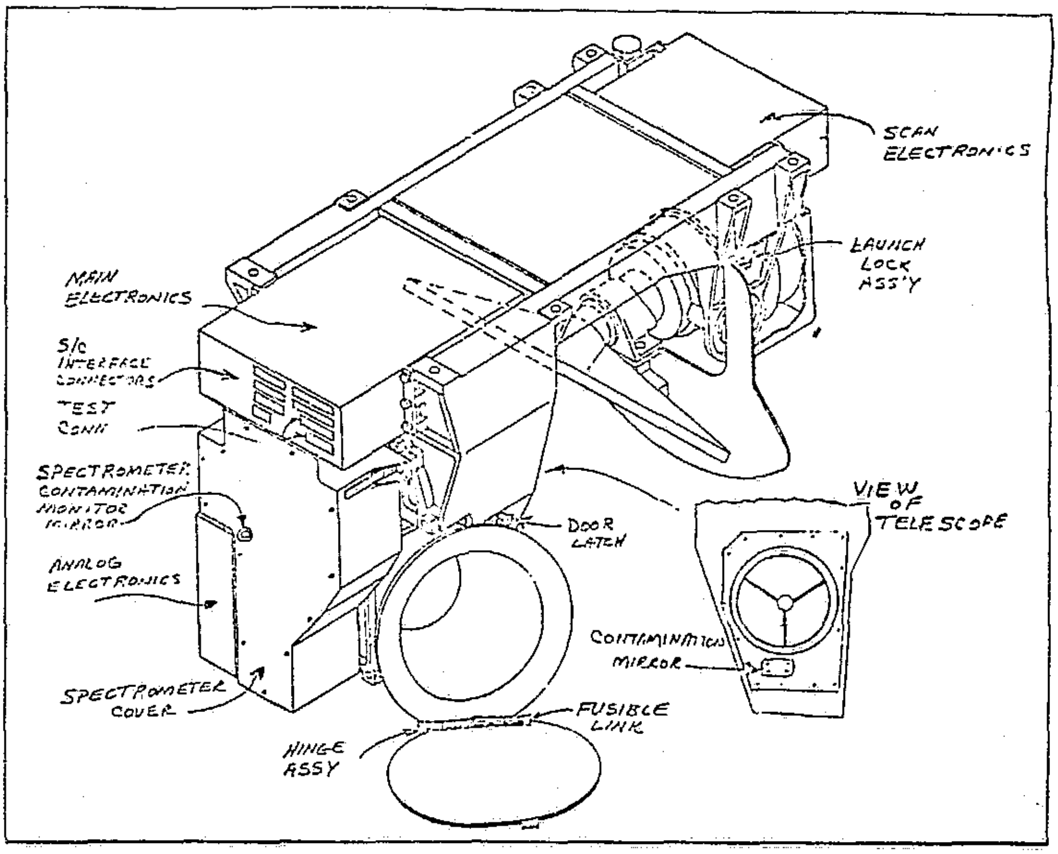

Figure 14. CZCS Sensor

The data output from the sensor is at a maximum rate of 3.49 Mbps with an average rate of 800 Kbps. This is too high to permit the standard Multimission Modular Spacecraft data system to be employed, with the result that a special data handling system would have to be flown. The attitude requirements are about 1° in pointing accuracy, with motion rates of $0.01\ ^{\circ}/sec$ are within the capabilities of the Multimission Modular Spacecraft attitude control subsystem.

3.2.13 Eclectic Satellite Pyroheliometer (ESP).- The ESP is a solar radiometer intended to make long term measurements of

(1) the total solar irradiance to an accuracy of 0.5% and variability with a precision of 0.01%,

(2) the solar spectral irradiance in the broad spectral bands from 0.180 to 3.8 μm, 0.526 to 2.8 μm, and 0.698 to 2.8 μm to an accuracy of 1%,

(3) the solar spectral irradiance in the near ultraviolet bands from 0.20 to 0.25 μm, 0.25 to 0.30 μm, 0.30 to 0.35 μm, and 0.35 to 0.40 μm to an accuracy of 2.0%.

The instrument is a three channel self-calibrating cavity radiometer employing thermopile sensors with a field of view of 1.8° (ref. 13). The sensor units are designed to view the sun with an alignment of better than 0.5 degrees. Figure 15 illustrates the ESP sensor.

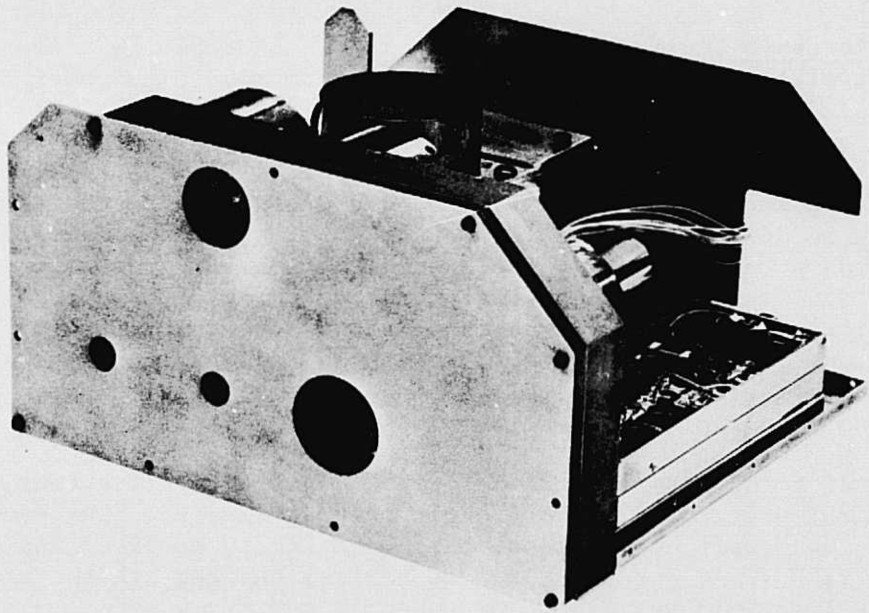

Figure 15. ESP Sensor Assembly

The ESP was constructed under the AAFE program and is now awaiting funding for design verification tests. The sensor has been designed to vibration and acceleration specifications which should permit satellite operation with little mechanical modification. The changes envisioned at this time include the following:

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(1) Design of an output interface. The present electronics has adequate space, power and a connector available.

(2) The power supply input sections may require additional design if the power turn off transients exceed 10 milliseconds duration.

(3) An automatic command sequencer internal to the instrument may be necessary to realize the wide range of sensor modes.

(4) The sensor requires a sun tracker or pointing system to aim the sensor to within ± 0.5 degrees of the sun's direction while measurements are being made.

(5) The command bit rate may be up to 512 bps and has a special format which will require a new command interface.

The sensor data rate is only 160 bps and the pointing accuracy is only 0.5° (spacecraft or separate pointing system), which are both easily satisfied by the Multimission Modular Spacecraft provided that the spacecraft pointing at the sun does not conflict with other experiment requirements. In this case a dedicated pointing system will be required to aim the sensor at the sun.

3.2.14 Microwave Temperature Sounder (MTS).- The MTS is a microwave radiometer with twelve data channels designed to determine atmospheric temperature to an altitude of 80 km (ref. 14). The radiometer measures the microwave emissions from atmospheric oxygen molecules at particular frequencies which predominate at different altitudes in the atmosphere. By the application of suitable weighting functions to the radiometric measurements, the atmospheric temperature can be determined. The sensor has an instantaneous field of view of 7.5° and scans perpendicular to the ground track ± 43° from nadir. The radiometer has a dual antenna system which scans by rotating hyperboloid reflectors about the axes of the feed horns, similar to the NIMBUS G SCAMS radiometer. The instrument uses heated wedge targets as radiometric temperature references. The rotating antenna reflector scans both of these targets during each 26.5 second scan cycle. The sensor is shown in Figure 16.

The MTS radiometer was fabricated for the Advanced Applications Flight Experiments (AAFE) program and is currently being evaluated. The sensor was designed for a spacecraft application, being similar to previous instruments for NIMBUS and TIROS. As a result, the mechanical changes are not great. The modifications are as follows:

(1) The engineering model must be refurbished with strengthening of the internal structure and conformal coating.

(2) The present sensor power is from commercial 110 VAC power supplies. These must be replaced with space qualified +28 VDC power supplies.

(3) A data interface to handle the digital and analog data may be required.

(4) A command and control system requires design and fabrication.

The data rate of 16 bps is extremely low, and the pointing accuracy of 0.1° and motion rate of 0.1°/sec are also within the spacecraft capabilities.

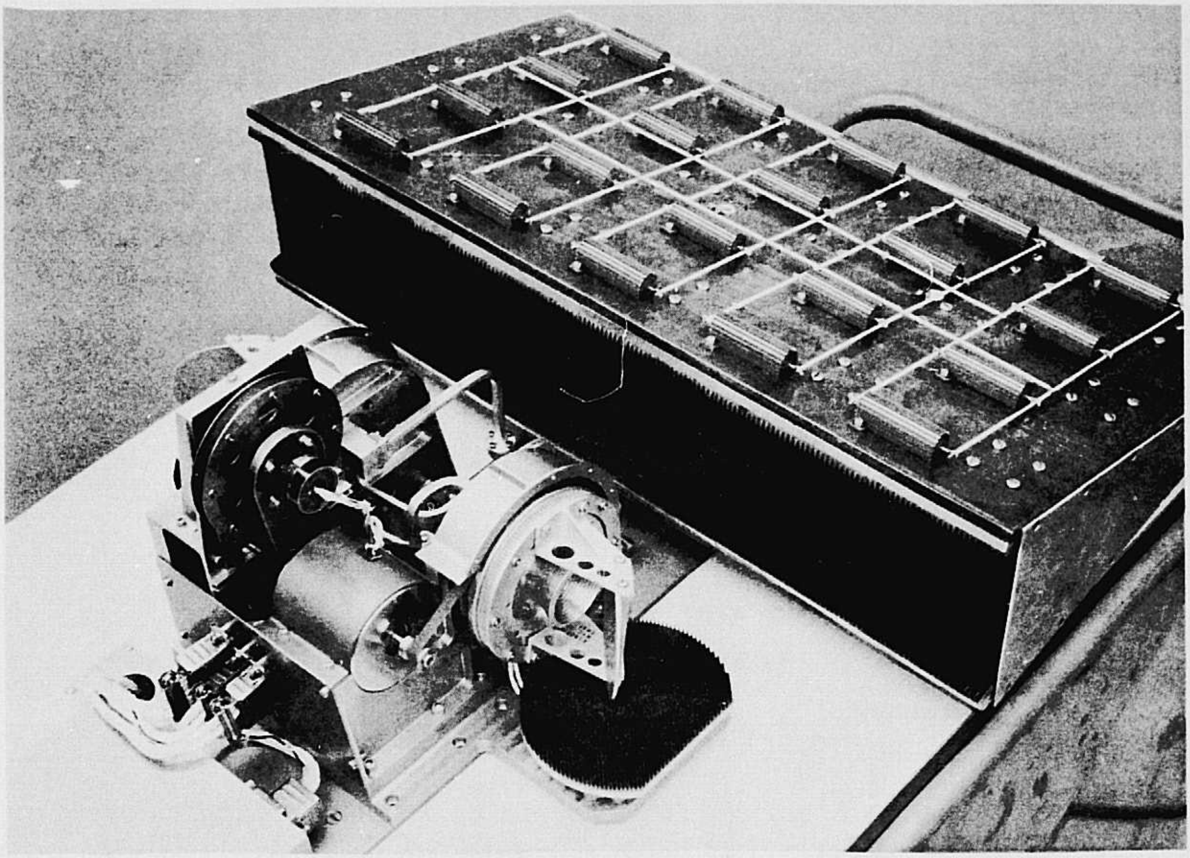


Figure 16. MTS sensor

3.2.15 Limb Infrared Monitoring of the Stratosphere (LIMS).- The LIMS radiometer, to be flown on NIMBUS G, is a version of the LRIR flown on NIMBUS 6, with 6 spectral channels instead of 4. The sensor is intended to determine the vertical distribution of temperature, ozone, water vapor, nitric acid, and nitrogen dioxide in the altitude range from 15 to 60 km. Measurements are made in six infrared spectral regions:

Nitrogen Dioxide	6.1 to 6.4 μm
Water Vapor	6.3 to 7.2 μm
Ozone	8.6 to 10.7 μm
Nitric Acid	10.9 to 11.8 μm
Carbon Dioxide	13.2 to 17.2 μm
Carbon Dioxide	14.7 to 15.8 μm

A programmed scanning mirror in the radiometer causes the fields of view of the six detectors to make a vertical scan across the earth's horizon. The detector array consists of 4 detectors with fields of view 0.5 x 5 milliradians, and 2 detectors with fields 1.0 x 8 milliradians. These yield vertical altitude resolutions of approximately 1 and 2 km, respectively. The limb radiance profiles in the carbon dioxide bands are operated on by inversion algorithms to yield the vertical temperature distributions. The radiance profiles in the other spectral bands in conjunction with the temperature profiles are used to determine the vertical distribution of the other constituents. Figure 17 illustrates the LIMS configuration to be flown on NIMBUS G (ref. 15).

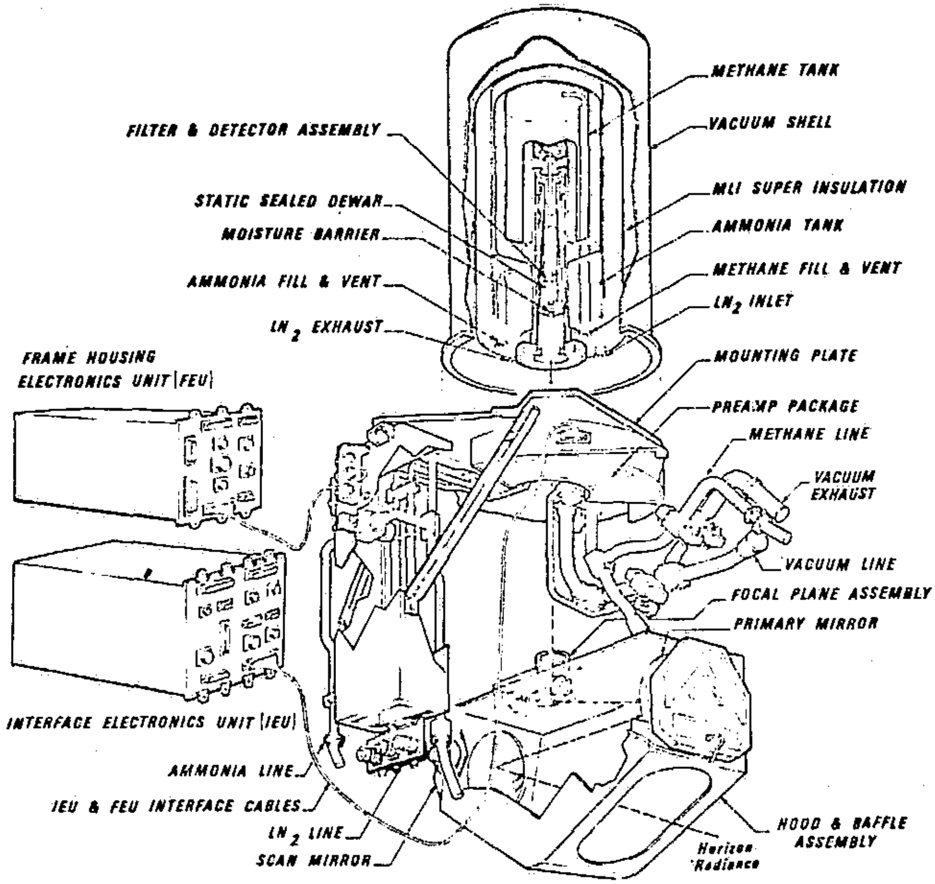


Figure 17. LIMS Sensor

The LIMS requires a number of modifications to be adapted to a free-flyer satellite:

(1) The current LIMS solid cryogen detector cooler requires replacement by a refrigerator type Vuilleumier (VM) cooler with a 1 year lifetime. For the existing LIMS flight back up detector capsule assembly (DCA) this requires:

(a) modify mechanical and thermal interfaces,
(b) design and construct interfaces to the VM cooler,
(c) design and fabricate a VM cooler mounting and vacuum interface.

It will also be necessary to redesign and modify handling fixtures, vibration fixture, and chamber feedthroughs to accommodate the VM cooler and its interfaces.

(2) A VM cooler for this application requires development since presently available cooler designs are not suitable.

(3) A VM cooler control unit will be required to power the cooler, and monitor and control the temperature produced at the detector.

(4) The VM cooler requires a heat sink capable of dissipating about 70 to 100 watts of power. This depends on the detector heat load.

(5) An electrical interface will have to be designed and fabricated to convert the Multimission Modular Spacecraft power, command, and data interfaces to LIMS compatible interfaces, and to multiplex the VM cooler data. The LIMS bench check unit will require modification to accommodate the redesigned interfaces.

(6) The LIMS elevation scan range will require modification to accommodate the Multimission Modular Spacecraft orbital altitude. The modifications will be made to existing circuit boards in the Frame Electronics Unit. Boresight direction changes required will be designed into the sensor mounting frame.

(7) A mechanical adapter to mechanically interface the existing LIMS spacecraft mount to the Multimission Modular Spacecraft structure must be designed and fabricated.

(8) The LIMS requires attitude rates to be measured to ± 0.002 degrees/second at a 10 Hz rate. If these data are not readily available from the spacecraft attitude control system, then additional instrumentation will be necessary.

(9) The sensor should not view the sun directly or it may be damaged. The orientation with respect to the sun must be examined to determine the frequency and duration with which this might occur.

(10) A LIMS flight back-up system does not exist in its entirety to be used for modification as a Multimission Modular Spacecraft instrument. Effort is required to integrate existing flight spare assemblies and engineering model hardware as follows:

(a) Detector Capsule Assembly - a flight spare exists.
(b) Interface Electronics Unit - integrate flight spare boards with the engineering model chassis.
(c) Frame Electronics Unit - integrate flight spare boards with the engineering model chassis.
(d) Preamp Package - integrate flight spare boards with the mounting plate.
(e) Optical-Mechanical Package - integrate flight spare scan assembly with the engineering model optics and structure.

The sensor digital data rate is 4300 bps and 9 analog lines are required with 8 bit accuracy. These requirements are easily satisfied by the spacecraft data handling system. The LIMS pointing accuracy requirement is only 2° which presents no problem.

4.0 ADAPTATION COST ANALYSIS

The process of adapting existing hardware to a Shuttle launched satellite involved consideration of a number of tasks in addition to the actual hardware modifications. A work breakdown of the principal tasks was comprised of the following components:

(1) Modifications Design Study

This task includes a detailed examination of the sensor and its interfaces with the spacecraft mechanical and electrical subsystems. The output of this study is a system definition which details the modifications which must be made to the sensor to enable it to withstand the Shuttle and Multimission Modular Spacecraft environments and permit automated operation when interfaced with the communications and data handling system. Until the spacecraft mechanical interfaces are more precisely defined, the details of the sensor mounting and the thermal interface with the experiment module can only be defined in general terms, outlining potential problem areas. The initial design study is intended to provide specifications for the entire system and subsystems, but not to provide detailed engineering drawings.

(2) Modifications

Includes rework, reinforcement of structure, wiring, as well as the substitution of components or addition of new parts. Generally there are four reasons for such modifications: (a) to permit the equipment to survive the Shuttle and the Multimission Modular Spacecraft environments, and interface with the spacecraft services; (b) to render the equipment operation safe during the manned mission on-board the Shuttle; (c) to adjust operational characteristics such as field of view, mission duration, and others, to the Multimission Modular Spacecraft mission parameters; and (d) to provide compatibility between the experiment and the supporting subsystem(s) in the Multimission Modular Spacecraft. Science upgrading, the enhancement of the experiment through larger rate of data acquisition or improved quality of the data is considered a desirable goal, but is not emphasized in the analysis, since the low cost philosophy dictates that maximum scientific and applications benefit be derived from the available experiment hardware in a configuration that, through economical means, makes it compatible with the Shuttle and its mission without attempting to maximize performance. The distinction is exemplified by modifications in the optics of a remote sensing experiment: necessary changes in the optics to accomodate to the new altitude and field of view in space is mandatory; modifications in the optics merely to gain an extra measure of resolution - where that desirable incremental performance is not essential to the basic experimental goals - is not considered to be compatible with the low cost approach.

(3) Functional Tests

(a) System tests at the PI or manufacturers' facility following assembly of the modified instrument and integration with its electronics.

(b) Acceptance tests to demonstrate performance to specifications prior to delivery.

(c) Calibration of the sensor to provide proper interpretation and accuracy to the sensor data.

(4) Environmental Tests

Include thermal-vacuum test, shock, vibration, and acoustic noise, as necessary.

(5) Ground Support Equipment

Dedicated GSE or facilities peculiar to the given experiment and not generally accepted as integration facility support equipment.

(6) Interface Equipment

Special support equipment to protect the equipment from the environment or provide interface with the Multimission Modular Spacecraft including dedicated mounting structure.

(7) Product Assurance

Quality control and quality assurance efforts including inspection, tests, and analyses. This is a difficult task to cost without knowing the specific requirements which will be imposed by NASA. The sensor modifications have been costed assuming some moderate level of quality assurance requirements. However, it should be recognized that the costs given may change significantly when the exact requirements are known.

(8) Program Management

The cost figures presented include overhead and G&A. In all cases the costs are based on 1977 dollars. The cost estimates for the sensors considered in the following sections were derived from discussions between General Electric representatives and the manufacturers of the original sensors. The cooperation and assistance of the following individuals and organizations outside of the General Electric Company is gratefully acknowledged:

(1) LACATE, LIMS	J. Thomas Honeywell, Inc. Radiation Center 2 Forbes Road Lexington, MA 02173	VM Cooler R. Hall Hughes Aircraft Corp. Culver City, CA 90230

(2) VTPR G. Falbel
Barnes Engineering Co.
30 Commerce Road
Stamford, CT 06904

(3) THIR J. G. Carlin
Santa Barbara Research Center
75 Coromar Drive
Goleta, CA 93017

(4) SBUV/TOMS D. J. Ahern
Beckman Instruments, Inc.
Advanced Technology Operations
1630 South State College Blvd.
Anaheim, CA 92806

(5) SAGE C. A. Jensen
Ball Brothers Research Corp.
Aerospace Division
Boulder, CO 80302

4.1 Lower Atmosphere Composition and Temperature Experiment (LACATE)

Preparing the LACATE for a mission on the Multimission Modular Spacecraft requires the efforts of two different vendors; the sensor manufacturer and the VM cooler manufacturer. While the tasks are generally separable there are areas of overlapping responsibility which require specification before costs can be properly identified. An item in question is the heat sink required for the VM cooler. This could logically be provided by both vendors. However until the spacecraft thermal and mechanical interfaces are defined, the degree of complexity required in the heat sink design is unknown and costing must be deferred until the results of the design study are available.

The modifications design study will investigate the effects of spacecraft orbit, environment, and flight duration on the following:

(a) scan elevation requirements
(b) detector and cooling system
(c) sensor structural integrity,
(d) sensor and electronics thermal control.

The LACATE modifications have been described in Section 3.2.2 and will not be repeated here. The costs associated with these modifications have been listed in Table III with the exception of the costs attributable to the VM cooler, which have been described in Section 4.2. The LACATE modifications program is expected to require 12 months to complete. Upon its completion there would exist the following:

(1) A VM cooler with a new focal plane, integrated with the sensor

optics assembly and ready for functional tests and acceptance tests including environmental.

(2) An electronics package(s) containing flight electronics for power, commands, and data handling functionally tested, at box level, and ready for system integration and acceptance tests.

(3) A sensor mounting frame ready for system integration and acceptance tests.

Tasks remaining include generation of system integration, system test, and acceptance test plans and procedures, conducting the integration and test program; design and fabrication of GSE fixtures (e.g., vibration and T/V). Costs do not include a formal Product Assurance program.

TABLE III

LACATE COST SUMMARY

1.		Modification Design Study		100.0 K$
2.		Modification		
	2.1	Detector Capsule Assembly and VM Cooler Interface		105.0
	2.2	VM Cooler Development (See Section 4.2)		806.0
	2.3	VM Cooler Electronics		
	2.4	VM Cooler Heat Sink	TBD	
	2.5	Azimuth Scan and Servo Redesign		50.0
	2.6	Elevation Scan Angle Range (Included in 2.10)		
	2.7	Sensor Housing	TBD	
	2.8	Redesign Data and Command System		70.0
	2.9	Redesign Power Supply and Repackage Electronics		305.0
	2.10	Sensor Mounting Frame		40.0
	2.11	General Resurbishment		40.0
3.		Functional Test	TBD	
4.		Environmental Test	TBD	
5.		Ground Support Equipment	TBD	
6.		Interface Equipment (See 2.4, 2.10 above)		
7.		Quality Assurance	TBD	
8.		Program Management and Reports		144.0
				1660.0 K$ Plus TBD

4.2 Vuilleumier (VM) Cycle Cryogenic Refrigerator

The costs for this detector cooling system have been itemized separately from the LACATE and LIMS sensor costs since the cooler represents a substantial new development effort. However, the development would benefit future cooler designs for other sensors with long life (1 year or more) and low input power (90 watts) cooler requirements. The costs in Table IV have been estimated on the basis of the design in Reference 4 with the program having an elapsed time of about 24 months. An engineering model and a flight model would be designed, fabricated, and tested along with a test console. The engineering model would be suitable for use with the sensor while it was being tested in order to preserve the life of the flight unit.

In order to provide added confidence in the cooler design an additional unit to be used for life tests was recommended. This would also require the fabrication of another test console to be devoted to the life tests.

The design does not consider the heat sink which must be provided to dissipate the heat produced by the cooler. The heat sink design will be largely determined by the spacecraft thermal interfaces and its ability to absorb heat from the equipment. As a result, costing has not been provided for the heat sink design.

The provision for integration support is intended to provide the sensor manufacturer with training and assistance when the VM cooler is delivered for integration with the sensor.

TABLE IV

VM COOLER COST SUMMARY

1.	Modification Design Study	50.0 K$
2.	Modification	
	2.1 Detailed Mechanical and Electrical Design	140.0
	2.2 Engineering Verification & Development Tests	50.0
	2.3 Fabricate & Check Engineering Model	80.0
	2.4 Fabricate & Check Flight Model	85.0
3.	Functional Tests	
	3.1 Acceptance, Launch Qualification Tests	90.0
4.	Environmental Tests (Included in 3)	
5.	Ground Support Equipment	50.0
6.	Interface Equipment (See LACATE and/or LIMS)	
7.	Quality Assurance	40.0
8.	Program Management & Data Reporting	75.0
9.	Integration Support	15.0
	Sub-total	675.0 K$
Fabricate and Check Extra Unit for Life Test		85.0
Additional Test Console for Life Test		30.0
Tear Down/Inspect Unit During Life Test (4 Times)		16.0
		806.0 K$

4.3 Correlation Interferometry for the Measurement of Atmospheric Trace Species (CIMATS)

The present CIMATS instrument is designed for operation in ground and aircraft environments. As a result there are a number of components which must be be replaced or strengthened to permit the sensor to survive the launch environment and operate in space. These have been outlined in Section 3.2.3. The major modifications involve the following:

(1) Replacement of the cryogen cooled detectors with thermoelectrically cooled detectors as specified by the design study.

(2) Redesign and repackaging of the electronics using low power components and printed circuit boards. This also includes new data and command circuitry and interfaces, as well as repackaging of the power supplies

(3) Redesign of the sensor housing incorporating new heaters and insulation. The thermal control problem involves heat dissipation by the sensor, the power supplies, the electronics and the thermoelectric coolers. Depending on the spacecraft thermal interface the redesign may be substantial.

(4) Design and fabrication of a Ground Support Unit involves building a facility which will contain diagnostic displays, command and data gathering functions, test equipment, test gases and gas cells, infrared radiance sources, and spacecraft electrical functions.

The functional tests specified are intended to serve two purposes:

(1) The system tests proposed are intended to verify that there has been no degradation of sensor performance due to the modifications performed. They include measurements using test gases in cells for a range of source radiance levels which cover the anticipated values.

(2) The sensor calibration involves a complete definition of the sensor optical and electrical parameters. This will permit the determination of the instrument transfer function which will allow the sensor output to be computed for a given sensor spectral irradiance. A short set of outdoor ground tests is recommended to verify the calibration.

Table V contains an estimated cost summary for the modifications program, which is estimated to require 30 months.

TABLE V

CIMATS COST SUMMARY

1.	Modification Design Study	67.0 K$
2.	Modifications	
	2.1 Sensor Housing Redesign	40.0
	2.2 Scan Plate/Sensor Arm Protection System	22.6
	2.3 Sensor and Electronics Thermal Control System	38.8
	2.4 Motor Replacement	7.2
	2.5 Reference Blackbody	5.6
	2.6 Reference Laser	50.0
	2.7 Detectors (TE Cooled)	115.0
	2.8 Automatic Gain Control Chopper and Driver	15.3
	2.9 Sensor Wiring and Harness	9.0
	2.10 Sensor Electronics	355.0
	2.11 Component Test	35.0
3.	Functional Test	
	3.1 System Tests	70.0
	3.2 Calibration	160.0
4.	Environmental Test	75.0
5.	Ground Support Equipment	150.0
6.	Interface Equipment	42.0
7.	Quality Assurance	150.0
8.	Program Management and Reports	300.0
		1707.5 K$

4.4 Vertical Temperature Profile Radiometer (VTPR)

In order to incorporate an existing VTPR unit into the Multimission Modular Spacecraft, the most practical solution appears to be the design and fabrication of an electronic interface box which will provide the required power and VTPR data and command interfaces. These have been specified in Section 3.2.4. The only modifications to the VTPR that appear economically practical are:

(1) a change in the image compensation focal plane mask to accomodate a lower orbit altitude, and
(2) an increase in the scan step speed, which will mean some reduced sensitivity.

The estimated costs for providing the modifications described are summarized in Table VI. The modifications program is estimated to require 12 months.

TABLE VI

VTPR COST SUMMARY

1.	Modifications Design Study		
	1.1 Electronic Interface Box Design		50.9 K$
	1.2 General Interface Design		57.0
2.	Modifications		
	2.1 Develop & Fabricate Interface Box		98.8
	2.2 Modify VTPR to Revised Configuration		16.0
3.	Functional Tests		
	3.1 Interface Box & VTPR Integration		47.0
	3.2 VTPR - Interface Acceptance Tests		88.0
4.	Environmental Tests (Included in 3.)		
5.	Ground Support Equipment	TBD	
6.	Interface Equipment	TBD	
7.	Product Assurance		21.3
8.	Program Management and Reports		34.0
			413.0 K$ plus TBD

4.5 Temperature Humidity Infrared Radiometer (THIR)

There is no backup flight hardware available for modification for the Multimission Modular Spacecraft mission. For the purposes of this study it is assumed that a new THIR and electronics module will be built according to the original specifications and configuration, and then modified as necessary for the Multimission Modular Spacecraft. With the present THIR configuration it is impractical to provide thermal control. As a result it is assumed that approximately 12.0 watts will be transferred from the THIR to the spacecraft.

Another consideration which may impact the cost concerns the bolometers used in the THIR. If bolometers similar to those originally used are not available at comparable cost, then procurement of the bolometers will cause a significant increase in the cost. For test purposes it is assumed that the THIR test consoles and fixtures would be provided in working order and calibrated.

Based on the preceding assumptions it is estimated that a new THIR instrument would cost $450 K, and an additional cost of $250 K would be incurred in incorporating the modifications which were outlined in Section 3.2.5. A period of approximately 20 months is required to fabricate the new THIR and make the necessary changes.

4.6 Solar Backscatter Ultraviolet/Total Ozone Measurement Spectrometer (SBUV/TOMS)

In adapting the SBUV/TOMS sensor to the Multimission Modular Spacecraft the most practical approach appeared to be modification of the existing engineering model, and design and fabrication of an Interface Module to interface between the existing design and the Multimission Modular Spacecraft and its services. A brief description of the Interface Module is included in Appendix C. The engineering model has commercial type electronics (high reliability in most cases) in its electronic module.

Based on the Multimission Modular Spacecraft data provided, and assuming that the present electronics are suitable, Beckman Instruments provided an estimate of $2.5 million to modify the existing engineering module and design and build the Interface Module. A design study to investigate the necessary modifications would be in the $100 K to $200 K cost range. For scheduling purposes the program is estimated to require 18 months.

4.7 Stratospheric Aerosol and Gas Experiment (SAGE)

Since the SAGE is a space qualified instrument, the primary modifications involve adapting the sensor electronics to the Multimission Modular Spacecraft electrical, and data and command subsystems. These modifications have been described in Section 3.2.7. Provided that the sensor is flown on its thermally isolated mounting frame the mechanical changes would not be major. If the satellite orbital altitude is in the 400 to 800 km range then the elevation scan mechanism will not require modification. The estimated costs for this fifteen months program are based on this assumption. It is further assumed that the SAGE GSE will be available and that a new unit will not have to be fabricated.

TABLE VII

SAGE COST SUMMARY

1.	Modifications Design Study		
	1.1 Systems Engineering		27.2 K$
	1.2 Electrical, Mechanical, Optical Engineering Support		32.7
	1.3 Program Management		16.4
2.	Modifications		
	2.1 Systems Engineering		93.0
	2.2 Studies and Analyses		109.0
	2.3 Instrument and GSE Design		327.0
	2.4 Fabrication, Assembly and Test		458.0
3.	Functional Tests (Included in 2.)		
4.	Environmental Tests (Included in 2.)		
5.	Ground Support Equipment (As noted above)		
6.	Interface Equipment	TBD	
7.	Quality Assurance		163.0
8.	Program Management and Reports		226.0
9.	Launch Support		65.0
			1517.3 K$ plus TBD

4.8 Monitoring Air Pollution from Satellites (MAPS/NIMBUS G)

Although the major modifications to the MAPS-NIMBUS G brassboard instrument have been identified in Section 3.2.10, detailed costing of the changes was not available during the period of this study. At this time NASA and TRW were in the process of negotiating a contract for a MAPS sensor to be flown as a Shuttle experiment, with essentially the modifications previously noted. It was felt by both parties that providing costing information might compromise the negotiations.

4.9 Limb Infrared Monitoring of the Stratosphere (LIMS)

Adaptation of the LIMS for a mission on the Multimission Modular Spacecraft requires the efforts of two different vendors; the sensor manufacturer and the VM cooler manufacturer. While their tasks are generally separable, there are areas of overlapping responsibility which require specification before the costs can be properly identified. The heat sink for the VM cooler is one item which could be provided by either the sensor or VM cooler manufacturer. However, without detailed information regarding the spacecraft mechanical and thermal interfaces, the complexity of the heat sink design is not known, and the costing must await the results of the design study.

The modifications required for the LIMS sensor have been described in Section 3.2.15, and the costs associated with these modifications are listed in Table VIII. Not included in these costs are those attributable to the VM cooler since these have been described in Section 4.2. In addition the costs associated with the requirements for attitude rate data, and the solar viewing study have been assumed to be tasks which the spacecraft integrator will carry out.

It is expected that the LIMS modification program will require a period of 12 months to complete. At the end of that time there would exist the following:

(1) A Spacecraft Electrical Interface Unit functionally tested, ready for system integration and acceptance tests.

(2) A modified Bench Check Unit, checked out, ready for system integration.

(3) Modified Frame and Interface Electronics Units, box level tested, ready for system integration and any re-qualification.

(4) An integrated mounting plate, Detector Capsule Assembly, cooler, functionally tested and ready for system integration and acceptance tests.

(5) A set of modified handling fixtures and mechanical GSE.

(6) A set of VM Cooler Control Electronics and Bench Check Equipment, provided by the cooler manufacturer.

(7) An upgraded LIMS Engineering Model Optical Mechanical Package, aligned and tested, ready for system integration.

Remaining tasks would be to re-write LIMS Environmental Specifications and Test Plans and Procedures controlling system integration, functional testing and environmental testing; and to perform the integration and test.

Cost estimates do not include costs associated with a formal product assurance program, but do include normal in process inspections and tests.

TABLE VIII

LIMS COST SUMMARY

1.		Modifications Design Study		50.0 K$
2.		Modification		
	2.1	VM Cooler Interface and Integration		300.0
	2.2	VM Cooler Development (See Section		806.0
	2.3	VM Cooler Electronics 4.2)		
	2.4	VM Cooler Heat Sink	TBD	
	2.5	Spacecraft Electrical Interface and Bench Check Unit		140.0
	2.6	Elevation Scan Range		20.0
	2.7	Sensor Mounting Frame		40.0
	2.8	Spacecraft Attitude Rate Spacecraft Inte-		
	2.9	Sun Damage Study grator Tasks		
	2.10	LIMS Subsystem Completion		50.0
3.		Functional Tests	TBD	
4.		Environmental Test	TBD	
5.		Ground Support Equipment	TBD	
6.		Interface Equipment (See 2.4, 2.7, above)		
7.		Quality Assurance	TBD	
8.		Program Management and Reports		85.0
				1491.0 K$ Plus TBD

APPENDIX A

SENSOR PRINCIPAL INVESTIGATORS AND MANUFACTURERS

1. Aerosol Physical Properties (APPS)

Principal Investigator:

Carlton R. Gray
C. S. Draper Laboratory
555 Technology Square
Cambridge, MA 02139
(617) 258-1549

Instrument Design and Fabrication:

Carlton R. Gray
C. S. Draper Laboratory
555 Technology Square
Cambridge, MA 02139
(617) 258-1549

2. Lower Atmosphere Composition and Temperature Experiment (LACATE)

Principal Investigator:

James M. Russell, III
NASA Langley Research Center
MS-271
Hampton, VA 23665
(804) 827-2576

Instrument Design and Fabrication:

Jack Thomas
Honeywell Radiation Center
2 Forbes Road
Lexington, MA 02173
(617) 862-6222

3. Correlation Interferometry for the Measurement of Atmospheric Trace Species (CIMATS)

Principal Investigator:

M. H. Bortner
General Electric Company
Space Sciences Laboratory
P. O. Box 8555
Philadelphia, PA 19101
(215) 962-3282

Instrument Design and Fabrication:

Harold W. Goldstein
General Electric Company
Space Sciences Laboratory
P. O. Box 8555
Philadelphia, PA 19101
(215) 962-3793

4. Vertical Temperature Profile Radiometer (VTPR)

Principal Investigator:

R. Pinamonti
NASA/Goddard Space Flight Center
Greenbelt, MD 20771
(301) 982-6291

Instrument Design and Fabrication:

G. Falbel
Barnes Engineering Company
Space Instruments Dept.
30 Commerce Road
Stamford, CT 06904
(203) 348-5381

5. Temperature Humidity Infrared Radiomenter (THIR)

Principal Investigator:	Instrument Design and Fabrication:
A. W. McCulloch NASA/Goddard Space Flight Center Greenbelt, MD 20771 (301) 982-6951	J. G. Carlin Santa Barbara Research Center 75 Coromar Drive Goleta, CA 93017 (805) 968-3511

6. Solar Backscatter UV/Total Ozone Mapping Spectrometer (SBUV/TOMS)

Principal Investigator:	Instrument Design and Fabrication:
D. F. Heath NASA/Goddard Space Flight Center Greenbelt, MD 20771 (301) 982-6421	Craig Elliot Advanced Technology Center Beckman Instruments, Inc. 1630 South State College Blvd. Anaheim, CA 92808 (714) 634-4343

7. Stratospheric Aerosol and Gas Experiment (SAGE)

Principal Investigator:	Instrument Design and Fabrication:
Michael P. McCormick NASA Langley Research Center MS-475 Hampton, VA 23665 (804) 827-3426	Clarence A. Jensen Ball Brothers Research Corporation Aerospace Division Boulder, CO 80302 (303) 441-4602

8. Infrared Heterodyne Spectrometer (IHS)

Principal Investigator:	Instrument Design and Fabrication:
Frank Allario NASA Langley Research Center MS-283 Hampton, VA 23665 (804) 827-2986	B. J. Payton AIL, Inc. Melville, NY 11746 (516) 595-4442

9. Monitoring of Air Pollution from Satellite (MAPS-AAFE)

Principal Investigator:	Instrument Design and Fabrication:
Henry G. Reichle, Jr. NASA Langley Research Center MS-401A Hampton, VA 23665 (804) 827-2576	Roy Bartle Science Applications, Inc. 1200 Prospect Street P. O. Box 2351 LaJolla, CA 92037 (714) 459-0211

10. Monitoring of Air Pollution from Satellite (MAPS-NIMBUS G)

Principal Investigator:	Instrument Design and Fabrication:
Henry G. Reichle, Jr. NASA Langley Research Center MS-401A Hampton, VA 23665 (804) 827-2576	P. B. Hutchings TRW Systems Group Bldg. 82, Rm. 2067 1 Space Park Redondo Beach, CA 90278 (213) 536-2052

11. Multichannel Ocean Color Sensor (MOCS)

Principal Investigator:	Instrument Design and Fabrication:
Gary W. Grew NASA Langley Research Center MS-473 Hampton, VA 23665 (804) 827-3661	P. G. White TRW Systems Group 1 Space Park Redondo Beach, CA 90278 (213) 535-2036

12. Coastal Zone Color Sensor (CZCS)

Principal Investigator:	Instrument Design and Fabrication:
Warren A. Hovis NOAA/NESS FOB #4, Rm. 0135 Washington, DC 02033 (301) 763-1847	W. Wallschlaeger Ball Brothers Research Corp. Boulder, CO 80302 (303) 441-4000

13. Eclectic Satellite Pyroheliometer (ESP)

Principal Investigator:	Instrument Design and Fabrication:
John R. Hickey The Eppley Laboratory, Inc. 12 Sheffield Avenue Newport, RI 02840 (401) 847-1020	S. Arthur Cone Gulton Industries, Inc. Data System Division 15000 Central Avenue, S.E. Albuquerque, NM 87123 (505) 299-7601, Ext. 216

14. Microwave Temperature Sounder (MTS)

Principal Inestigator:	Instrument Design and Fabrication:
D. H. Staelin Massachusetts Institute of Technology Cambridge, MA 02139 (617) 835-3711	E. J. Johnston Jet Propulsion Laboratory 4800 Oak Grove Drive Pasadena, CA 91103 (213) 354-2645

15. Limb Infrared Monitoring of the Stratosphere (LIMS)

Principal Investigator:	Instrument Design and Fabrication:
James M. Russell, III NASA Langley Research Center MS-271 Hampton, VA 23665 (804) 827-2576	Jack Thomas Honeywell radiation Center 2 Forbes Road Lexington, MA 02173 (617) 862-6222

16. Vuilleumier Cycle Cryogenic Regrigerator (VM Cooler)

NASA Coordinator:	Instrument Design and Fabrication:
Robert Averill NASA Langley Rearch Center MS-433 Hampton, VA 23665 (804) 827-4261	R. Rall Hughes Aircraft Corp. Bldg. 5, Mail Stn. B144 Culver City, CA 90230 (213) 391-0711

APPENDIX B

SUMMARY OF SPACECRAFT CHARACTERISTICS DATA

1. Electrical Power

(1) Voltage

+28 to +35 VDC negative ground to structure

(2) Transient (Normal): ± 5V (> 1 ms), 3 V (0 to 10 μs)
Transient (Abnormal): 0 to 40 VDC (> 0.5 sec)
Transient (Turn-off): -20 to +55 VDC

2. Experimental Signal Interface Characteristics

(1) The Remote Interface Unit (RIU) will accept bilevel signals, 0 to 5.12 VDC analog signals, 0 to 5 KΩ resistance.

(2) For resistance type sensors, the RIU will provide 1.0 ma to the sensor during the sampling period.

3. Telemetry and Command Systems

All telemetry, command, and clock interfaces are through Remote Interface Units located at the experimental module.

(1) Telemetry

The telemetry data rates are 1, 2, 4, 8, 16, 32, and 64 Kbps. Bit rates higher than 64 Kbps will require a mission dependent wide band telemetry system.

Each RIU multiplexer will have 64 inputs for analog, bilevel, or serial digital signals.

Experiment analog signals are A/D converted during a 62.5 μsec window.

- Analog signals will be 0 to +5.12 VDC, 5 kilo-ohm impedance.
- A/D converter resolution is 8 bits, accuracy is ± 25 mv.

Bilevel digital input signals with the following characteristics are accepted.

Logical 1:	+3.5 to +15 VDC
Logical 0:	-1 to +1.5 VDC
Impedance:	5 kilo-ohms

Serial digital input signal sample length is 8 bits with the following characteristics:

Logical 1:	+2.4 to +15 VDC
Logical 0:	-1 to +1.25 VDC
Impedance:	500 Ohms

Clock signal - frequency is 1.024 MHz (square wave).

Telemetry synchronization -

Signals will be available for experiment synchronization. Signals are single ended switch closure to ground.

(2) Command

Serial digital command reception and execution rate: 1000/sec. max.

- Command Length: 16 bits (∿ 72 microseconds).
- Experiment command-line interface circuit may include a matching Nat. Semiconductor DM7820 receiver.
- The command signals will have the following characteristics:

Signals to be Transmitted	Driver Terminal Voltage with Respect to Signal Ground
Logical "1"	Positive voltage appears on "AND" output terminal
	V_{OH} = +5.0 V maximum
	V_{OH} = +2.4 V minimum
	Low voltage appears on "NAND" output terminal.
	V_{OL} = +0.4 V maximum
	V_{OL} = 0.0 V minimum
Logical "0"	Low voltage appears on "AND" output terminal.
	V_{OL} = +0.4 V maximum
	V_{OL} = 0.0 V minimum
	Positive voltage appears on "NANS" output terminal.
	V_{OH} = +5.0 V maximum
	V_{OH} = +2.4 V minimum
Output Impedance	Z_{OH} = 50 ohms
	Z_{OL} = 12.5 ohms

Discrete (pulse) command rate: 62.5 per second maximum.

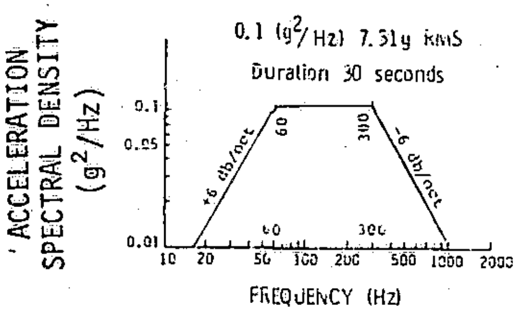
- Commands to drive relay coils or solid state logic will be ≥ 6 ms in duration.
- Gated output line to experiment is a single-ended switch closure to signal ground (active state).
- Experiment must provide a pull-up resistor with a diode, when driving a solid state logic.
- Discrete commands may be used as relay driver signals.
- Discrete command signals will have the following characteristics:

<u>Switch Closure Signal</u>

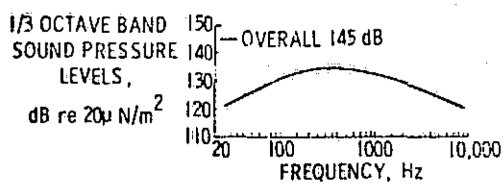
VOH (inactive state)	Floating, or Local (User) VCC (+30 volts, maximum)
ZOH (inactive state)	1 Megohm
VOL (active state)	0.5 volts maximum at 20 ma
I MAX	100 ma
Approximate Closure Time	6.5 to 7.0 milliseconds

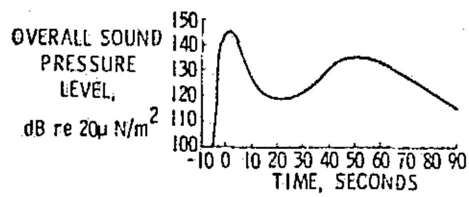
4. Launch Environment

4.1 VIBRATION



4.4 ACOUSTICS





4.2 ACCELERATION

STEADY STATE: X AXIS, 3.3 g MAXIMUM

TRANSIENT: FREQUENCY	AMPLITUDE
5-10 Hz	1.0 inch D. A.
10-21 Hz	5 g
21-35 Hz	1.0 g

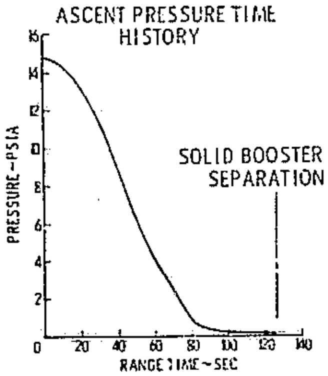
EQUIVALENT DURATION: ONE OCTAVE/MIN. LOGARITHMIC SWEEP

4.3 TEMPERATURE

SLOW STEADY RISE FROM 70°F (21°C) TO ≈ 80°F (27°C) WITHIN SHUTTLE PAYLOAD BAY

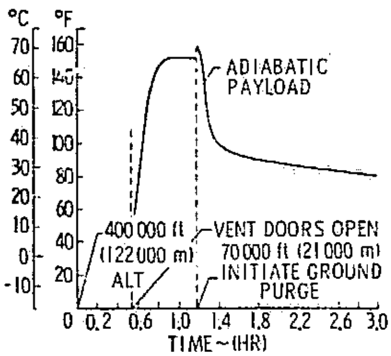
4.5 VENTING

THE PAYLOAD BAY IS VENTED DURING THE ASCENT PHASE, AND OPERATES UNPRESSURIZED DURING THE ORBITAL PHASE OF THE MISSION.



5.0 Return Environment

5.1 BAY AIR TEMPERATURE



5.2 STEADY STATE ACCELERATION

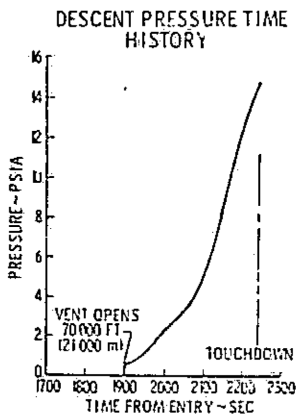
2.8 g MAXIMUM

VIBRATION: NEGLIGIBLE

ACOUSTICS: NEGLIGIBLE

5.3 VENTING

THE PAYLOAD BAY VENTS ARE CLOSED DURING DESCENT AERODYNAMIC HEATING



APPENDIX C

DESCRIPTION OF THE SBUV/TOMS INTERFACE MODULE

Figure C-1 is a simplified block diagram of the Interface Module. As shown in the figure, the box contains many input/output connectors. It also contains a well-filtered +28 VDC to -28 VDC power supply which generates -28 V primary power for the SBUV/TOMS. The translators for the analog, digital B and thermistor telemetry are primarily analog level changers using differential input stages to provide ground isolation. The command translators contain logic elements to stretch the command pulses to 40 ms duration, and level changers to convert them to the MA and MB format required by the SBUV/TOMS. The clock circuit is a phase-locked loop which produces a 400 kHz clock waveform which is locked to the 1.024 MHz clock from the Multimission Modular Spacecraft. This circuit is shown in block diagram form in Figure C-2.

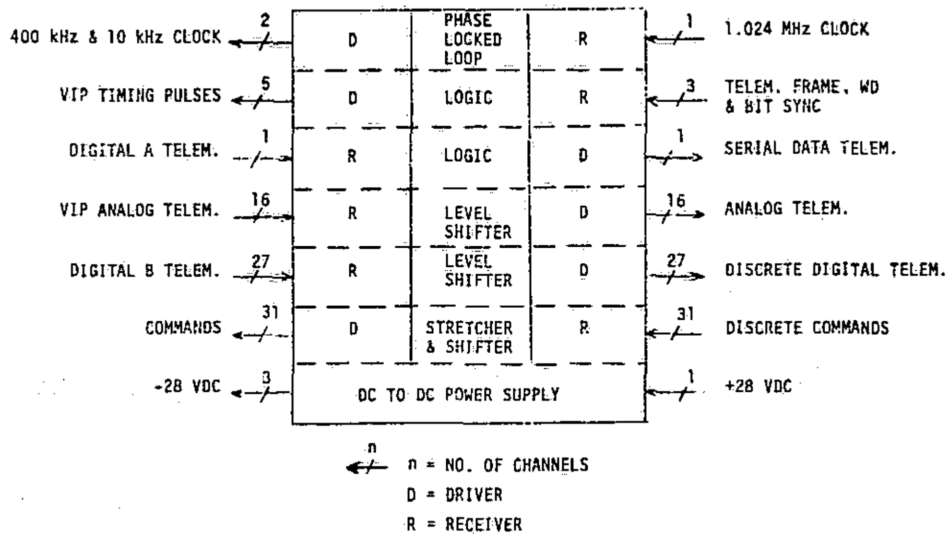


Figure C-1. Block Diagram, Interface Module

The translator for the digital A telemetry is shown in block diagram form in Figure C-3. It contains two shift registers which are loaded in ping pong fashion from the SBUV/TOMS. While Reg. A is loading, Reg. B is being unloaded by the Multimussion Modular Spacecraft serial telemetry interface. The registers inject minor frame ID bits so each minor frame which enters the Multimission Modular Spacecraft is properly identified. Major frame numbers are included in the existing SBUV/TOMS format. The load and unload select gates contol the loading and unloading of the registers at the proper times determined by the load and unload controllers. The load controller interfaces with the SBUV/TOMS while the unload controller interfaces with the MMS. The VIP Simulator derives all of the required VIP timing pulses from the clock and count-down chain and transmits them to the load controller and to SBUV/TOMS. This translator can be implemented with CMOS logic. It can be packaged very compactly using packaging techniques developed for the SBUV/TOMS. The Interface Module requires about a 6-inch cube.

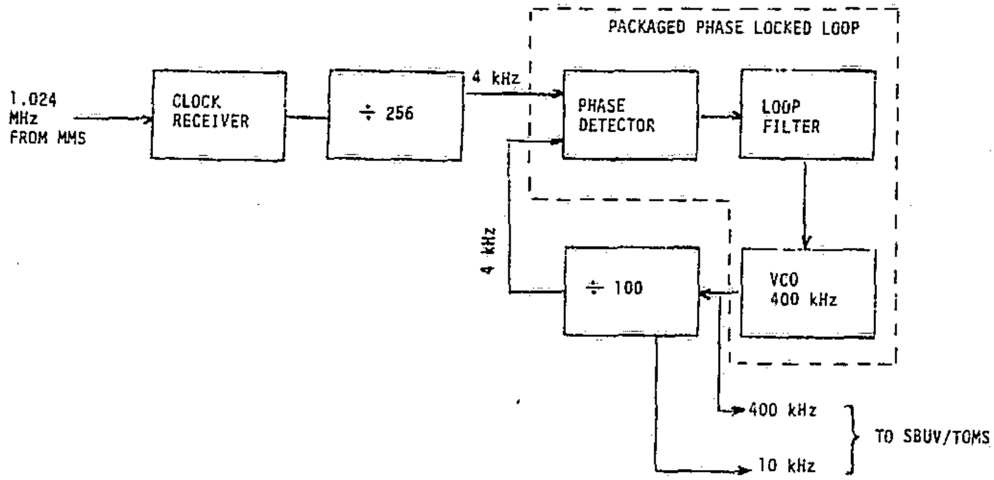


Figure C-2. Phase-Locked Clock

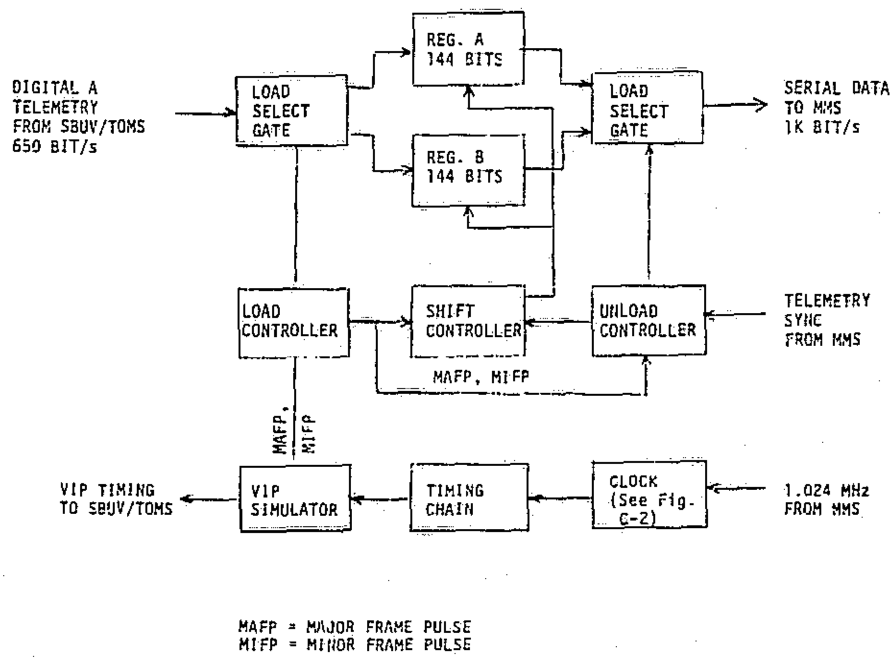


Figure C-3. Block Diagram, Digital Telemetry Translation Logic

All interface signals between the Interface Module and the SBUV/TOMS on the Multimission Modular Spacecraft are buffered with suitable interface drivers and receivers.

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