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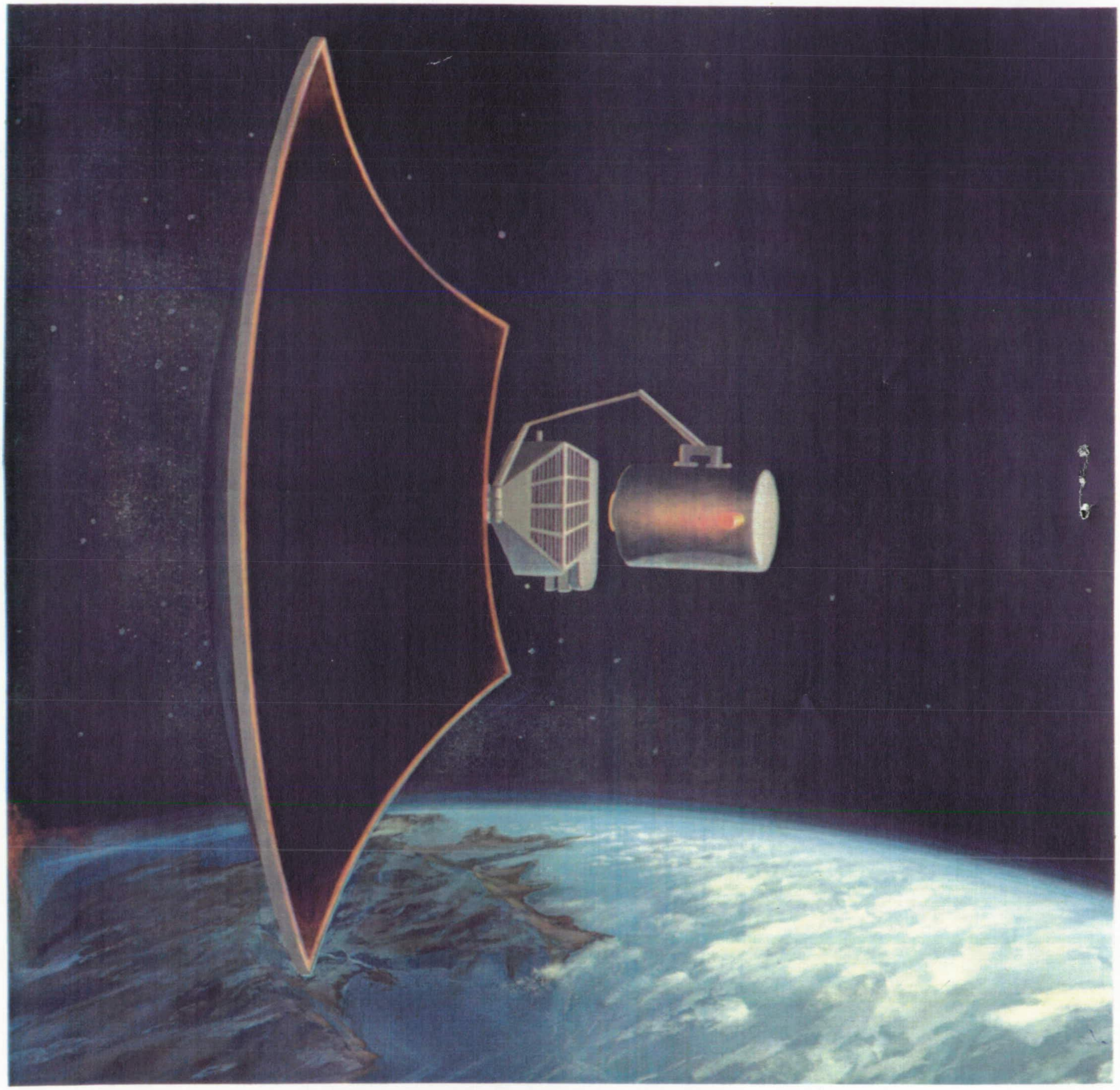
THERMION: VERIFICATION OF A THERMIONIC HEAT PIPE IN MICROGRAVITY



UTAH STATE UNIVERSITY
1990 - 1991

FORWARD

This final report describes the verification of a thermionic heat pipe in microgravity. This project consists of two design scenarios: THERMION-I and THERMION-II. The design was done entirely by a team of Utah State University (USU) students. The project has been completed under the sponsorship of NASA through the Universities Space Research Association (USRA) and in cooperation with the Idaho National Engineering Laboratory (INEL).

We at Utah State University are very pleased with the results of this design effort. We are proud of the product, and we are excited about the achievement of all our learning objectives. The systems design process is one that cannot be taught, it must be experienced. The opportunity to use our maturing engineering and scientific skills in producing the THERMION final design has been both challenging and rewarding. We are proud of the skills we have developed in identifying system requirements, spreading them into specifications, communicating with each other in all kinds of technical environments, conducting parametric and trade-off studies, learning to compose for the good of the system, and selecting actual hardware to be used.

The critical design review (CDR), conducted in May of this year at USU, was presented to INEL, Phillips Laboratory, JPL, and Space Dynamics Laboratory/USU personnel as well as other distinguished members of the engineering community. A revision of the paper will be presented at the AIAA/USU Conference on Small Satellites in August of this year. INEL and Phillips Laboratory, in collaboration with the Space Dynamics Laboratory/USU, are considering further development and testing of the THERMION project.

We would like to express our appreciation and thanks to the following people who have helped us with this design project:

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THERMION: VERIFICATION OF A THERMIONIC HEAT PIPE IN MICROGRAVITY

ABSTRACT

The Idaho National Engineering Laboratory (INEL) is conducting intensive research in the design and development of a small, excore heat-pipe-thermionic space nuclear reactor power system (SEHPTR). Progress in this research effort has identified the need for an in-space flight demonstration of a solar-powered, thermionic heat-pipe element. The proposed demonstration will examine the performance of such a device and verify its operation in microgravity.

The Utah State University space system design project for 1990-1991 focuses on the design of a microsatellite-based technology demonstration experiment to measure the effects of microgravity on the performance of an integrated thermionic-heat-pipe device in low-earth orbit. The specific objectives are to verify the operation of the liquid-metal heat pipe and the cesium reservoir in the space environment. The scope of the project includes the design of the flight test vehicle, the solar collection subsystem, the satellite attitude control subsystem, test instrumentation, data collection/processing and the telemetry subsystem. Interfaces with various launch systems are examined with emphasis on flight as secondary payload on the McDonnell Douglas Delta II.

This report describes two design configurations, THERMION-I and THERMION-II, which meet the requirements specified above. THERMION-I is designed for a long-lifetime (greater than one year) investigation of the operations of the thermionic heat pipe element in low earth orbit. Heat input to the element is furnished by a large mirror which collects solar energy and focuses it into a cavity containing the heat pipe device. THERMION-II is a much simpler design which is utilized for short-term (approximately one day) operation. This experiment remains attached to the Delta II second stage and utilizes energy from 500 lb of alkaline batteries to supply heat energy to the heat pipe device. The cost for fabrication, integration and flight are estimated at almost $3 million.

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1.0 INTRODUCTION

The Idaho National Engineering Laboratory (INEL) is conducting intensive research in the design and development of a small excore heat-pipe-thermionic space nuclear reactor power system (SEHPTR). The SEHPTR spacecraft will be able to supply 40 KW of power in any given orbit. Figure 1-1 shows a conceptual diagram of the SEHPTR spacecraft. The key components in this reactor the thermionic heat pipes. The heat pipes have two major functions: first, to convert heat energy into electrical energy, and second, to radiate the excess heat to space.

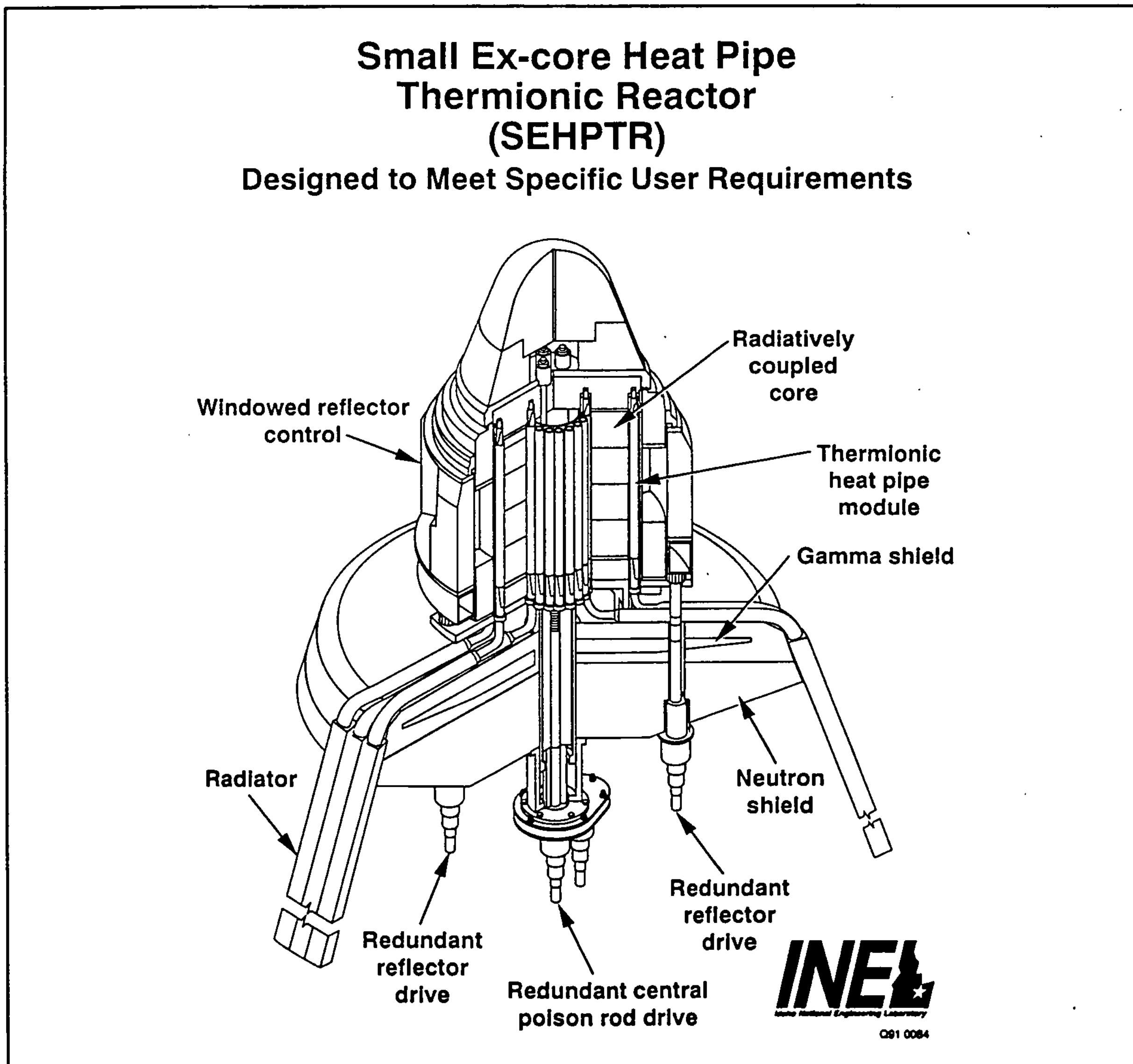

Figure 1-1 Space nuclear reactor power system (SEHPTR).

Thermionic power conversion is the process of converting heat energy into electrical energy with no moving parts. Heat is applied to the cathode surface, as shown in figure 1-2. This heat will boil off electrons that will jump across the gap to the cooler surface of the anode. This will cause a potential difference between the two plates and induce a current through the load.

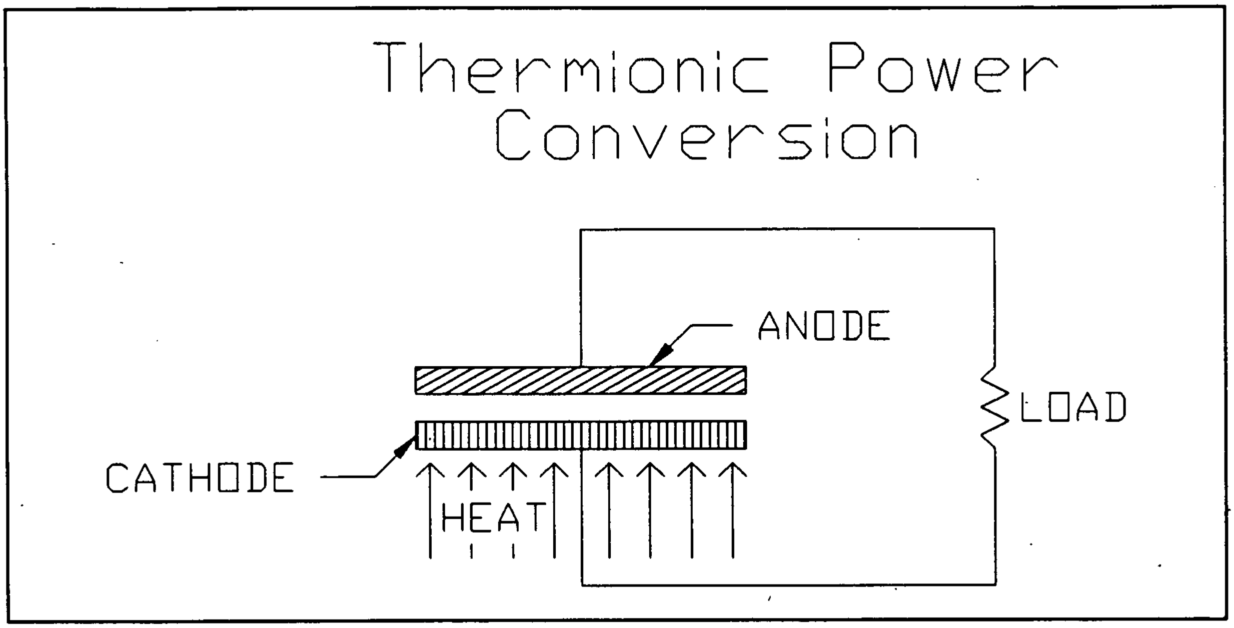

Figure 1-2 Thermionic power conversion converts heat energy into electrical energy.

The heat pipe uses the thermionic power conversion to generate the electrical energy. The heat pipe, which is being developed by Thermacore Inc., is actually two heat pipes. It uses a radial heat pipe, called the emitter, and an axial heat pipe collector, as shown in figure 1-3. The emitter heat pipe will pass the heat from the nuclear core to the cathode surface. The collector heat pipe keeps the anode surface cooler by transferring the heat from the anode surface and radiating it to space.

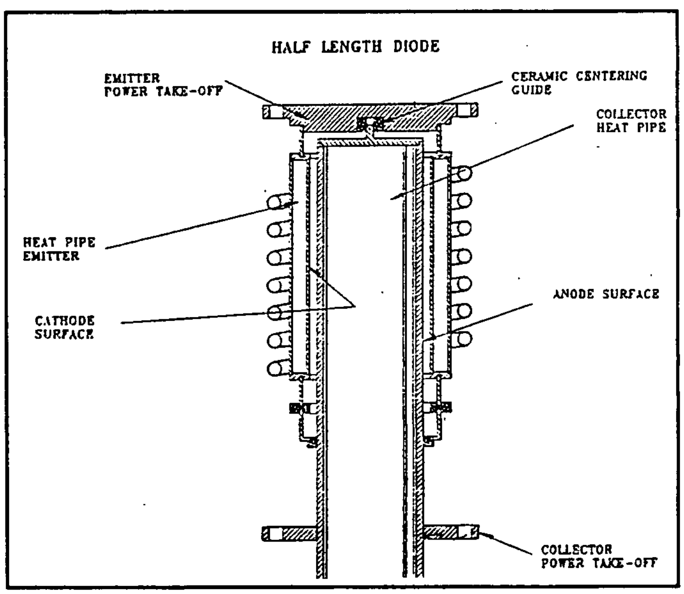

Figure 1-3 Heat pipe design by Thermacore.

1.1 DESIGN PROJECT

This year's design project was in collaboration with the Idaho National Engineering Laboratory to design a space flight-demonstration of a scaled-down thermionic heat pipe. The mission of this satellite would be to demonstrate the performance of an integrated thermionic heat pipe device concept in micro-gravity. The letter of contract can be found in Appendix A. During the course of the design, have developed two design scenarios - THERMION-I and THERMION-II.

1.1.1 THERMION-I

THERMION-I is a small satellite that will have a one year mission. A solar collecting mirror will be used to focus the Sun's energy into a cavity that will transfer the heat into a 6 cm heat pipe. The 6 cm heat pipe is a scaled down version of the 40 cm heat pipe that will be used in SEHPTR. The satellite will be the secondary payload on the Delta II.

1.1.2 THERMION-II

THERMION-II is a more brute force approach of conducting the experiment. Instead of being deployed from the Delta II, it will be permanently mounted to the secondary stage. THERMION-II will use batteries rather than the Sun to heat up the heat pipe. Because of the limited amount of batteries we can launch on the Delta II due to their weight, the experiment will only last a little more than a day.

1.2 DESIGN CONSIDERATIONS

Confirm that the thermionic heat pipe will operate in a micro-gravity environment is the primary performance driver for this THERMION mission. To confirm this, temperature measurements of the heat pipe are necessary. Measurement of the power generated will also be needed to prove that the device works properly.

1.3 DESIGN EVOLUTION OF THERMION-I

THERMION-I had five major design criteria:

- Test a 6 cm thermionic heat pipe module (THPM)
- Use Solar Energy to heat the THPM
- Small satellite
- 1 year mission
- Delta II as launch vehicle

A summary for each subsystem on THERMION-I are as follows:

1.3.1 Heat Pipe Testing System

The heat pipe temperature will be monitored through 12 thermocouples: 6

distributed over the emitter heat pipe surface and 6 distributed over the collector heat pipe surface.

The thermionic power conversion of the heat pipe will be approximately 11%. A 6 cm heat pipe requires 1050 watts of heat energy. It will convert 11% of the 1050 watts into electrical energy (116 watts). The electrical power breaks down into a 0.7 voltage and a current of 165 amps. The power pick-off points on the heat pipe are at 1000 K. This makes measuring the voltage and current with conventional techniques very difficult. We have chosen to measure the power lost through heat (I^2 R losses) and back out what the power is. This technique is discussed in more detail in Chapter 2.

1.3.2 Solar Collection System
A parabolic mirror that is 58.6 x 41.6 in. (148.8 x 105.7 cm.) will be used to collect 2050 watts of the Sun's energy. The mirror will focus the energy into a cavity that resembles Planck's black body box. Once the energy is trapped into the cavity, it will conduct into the heat pipe.

1.3.3 Attitude Control and Determination System
The mirror has a pointing requirement of ± 0.75°. To accomplish this accuracy, the following sensors and actuators are used:

- Sensors
    - Wide Angle Sun Sensor
    - High Accuracy Sun Sensor
    - Horizon Crossing Sensor
    - Photo Diodes

- Actuators
    - Torque Rods (Magnetometer)
    - Momentum Wheel

1.3.4 Satellite Structure and Configuration System
The structure consists of a bus, mirror, mirror support, payload, payload arm, and internal components. Figure 1-4 shows the satellite in its deployed configuration.

1.3.5 Data Management System
Data acquisition is the main task of data management. The data will be collected in a store and forward manner. The data will be dumped to a ground station every 8 hours.

1.3.6 Communications System
Since the satellite's attitude control will be tracking the Sun a near omni-directional antenna is needed. Two transmitting antennas will use S-band to down link data

to a mobile ground station. Two receiving antennas will be used to uplink any information that might be needed.

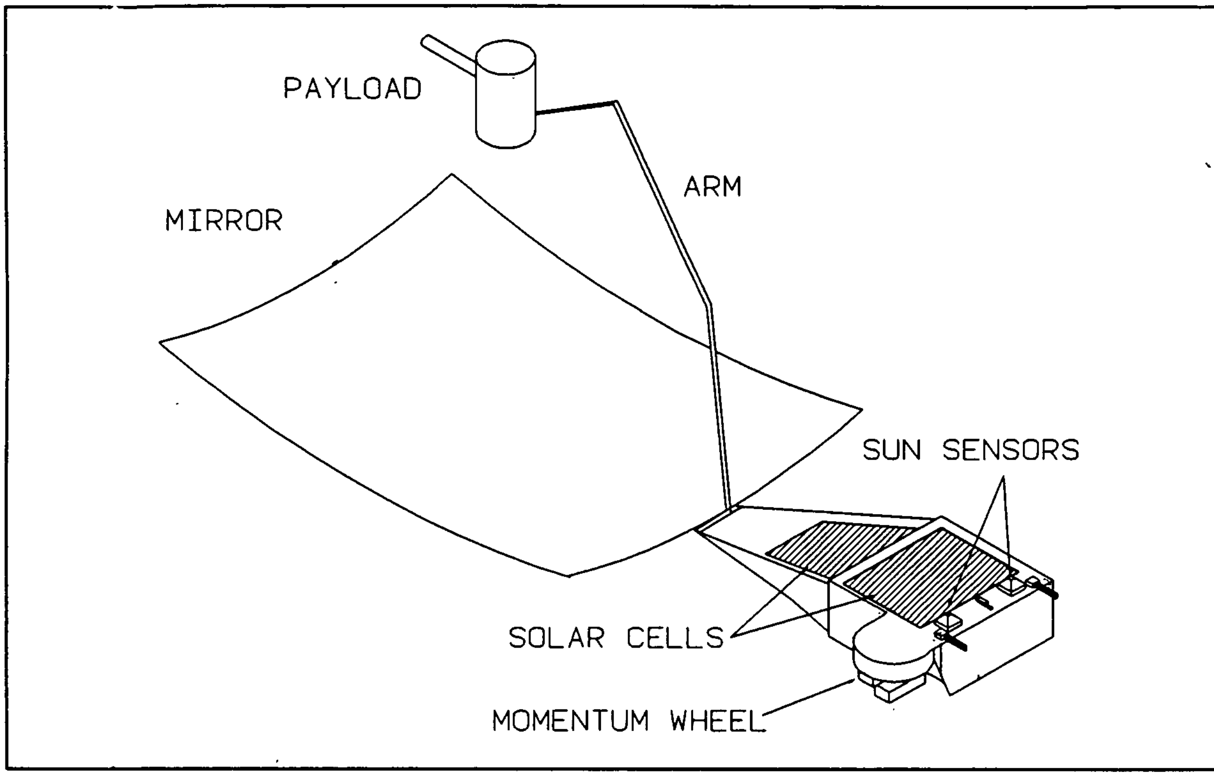

Figure 1-4 THERMION-I satellite in its deployed configuration.

1.3.7 Power System

The power generated by the thermionic heat pipe cannot be turned into useful power for operation of the satellite. Therefore, silicon solar cells were placed on top of the satellite to generate the required operating power. Ni-Cad batteries will be used when the satellite is in the shadow of the Earth. These batteries will then be charged on the Sun side of the orbit.

1.3.8 Launch Vehicle Interface and Deployment System

There will be several Global Positioning Satellite (GPS) missions in the next few years with available space on the Delta II secondary stage to launch a small satellite. THERMION-I was designed to fit in the secondary payload volume of the Delta II. The satellite is mounted to the Delta II via a Marman Clamp. Once the Delta II secondary stage has reached a circular orbit of 375 nmi, explosive bolts will fire and eject THERMION-I from the vehicle into its circular orbit.

1.3.9 Thermal Management System

Temperature extremes of the THERMION-I satellite were determined by considering the hottest and coldest orbits. A 90 minute orbit was assumed with 60 minutes of this orbit being in the Sun. The hottest orbit is during the 60 minutes that the satellite is in the Sun and appropriate instruments are turned on. In this case, all of the on-board components will stay within their operating temperature ranges. The coldest orbit is where the satellite stays in the shadow of the Earth for over 30 minutes. All of the components except the Sun sensors and the solar cells stayed in their temperature limits. This is are not of much concern because they will not be on in the shadow of the Earth. Three watts of power have been incorporated into the power system for heater use, and emissive coatings can be used to bring the instruments into their operating temperature ranges, if deemed necessary.

1.3.10 Test and Evaluation System

The test and evaluation system has been investigating ways of ensuring mission success. Test procedures have been developed to validate instrument operation.

1.3.11 Conclusions

The THERMION-I satellite has a mass of 85.19 lb. (38.64 kg), and will cost less than 1.2 millon the construction of the satellite. The integration costs for mounting to the Delta II will cost approximately 1 millon. The details of the cost estimates can be found in Section 14.0.

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1. Griffin, Michael D. and French, James R.: Space Vehicle Design, AIAA, Washington, DC, 1991.

2. Jaycox, Mike, Thermacore, Inc., Telecom 1-91 to 5-91

3. Thermionic Heat Pipe Module, Final Report, Inc., 1991.

2.0 PAYLOAD

The payload subsystem was responsible for mounting the thermionic heat pipe power generation device, collecting the solar energy necessary to drive the system, and providing the means to verify that the device is operating properly. The hardware consists of: a scaled version of the thermionic heat pipe device designed by Thermacore Incorporated, a parabolic solar collector/concentrator, an insulated cavity to trap heat and funnel it into the thermionic heat pipe device, and temperature recording instruments.

2.1 THERMIONIC HEAT PIPE DEVICE

THERMION-I uses a scaled version of the thermionic device designed by Thermacore, Inc. Two liquid-metal heat pipes, one inside the other, operate at different temperatures. A small gap between the two heat pipes is filled with Cesium vapor. The temperature difference drives electrons from the hot surface across the gap to the cold surface. This process is known as thermionic power conversion and results in an electric current. Thermacore's double heat pipe design is represented in figure 2-1.

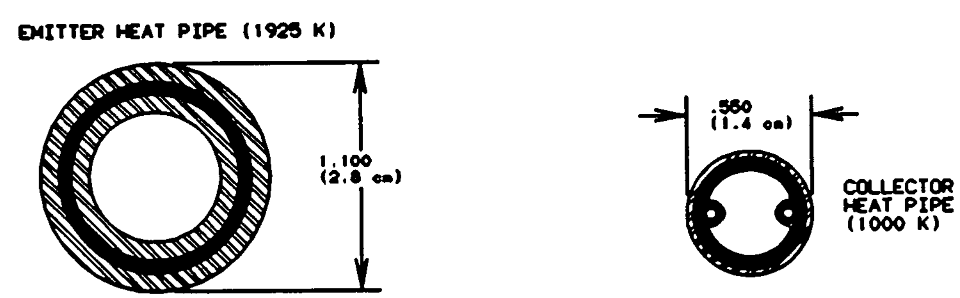

Figure 2-1 Cross sectional view of heat pipe.

The outside, or emitter heat pipe, receives solar energy over thirty percent of its circumference. The operating temperature of this heat pipe is approximately 1925 K. Lithium is the working fluid, and the wick structure is such that heat is transferred circumferentially as well as radially until the entire heat pipe is isothermalized at 1925 K. The inside, or collector heat pipe, receives its input heat from the emitter and operates at 1000 K. To maintain the 1000 K surface temperature, the collector heat pipe uses sodium as the working fluid to transfer excess heat axially through the heat pipe for radiation to deep space.
The emitter and collector heat pipes have outside diameters of 2.8 and 1.4

centimeters respectively which are the same as Thermacore's. The collector length for THERMION-I is 6 centimeters while that of Thermacore's full scale version is 40 centimeters long. The 6 centimeter heat pipe requires 1350 watts to operate. It is the largest heat pipe which can be run with the amount of energy available from the solar collector.

2.2 INSULATING CAVITY

2.2.1 Design Evolution

Initially, an insulating cavity was not thought necessary. The heat pipe would be fixed to a metal conduction plate with high-temperature braze. This tungsten plate would be the target for the mirror and would conduct the focused solar energy to thirty percent of the surface area of the emitter heat pipe. As the design evolved, it became evident that there would be large radiative losses from this plate and the heat pipe if they were not insulated to retain the heat.

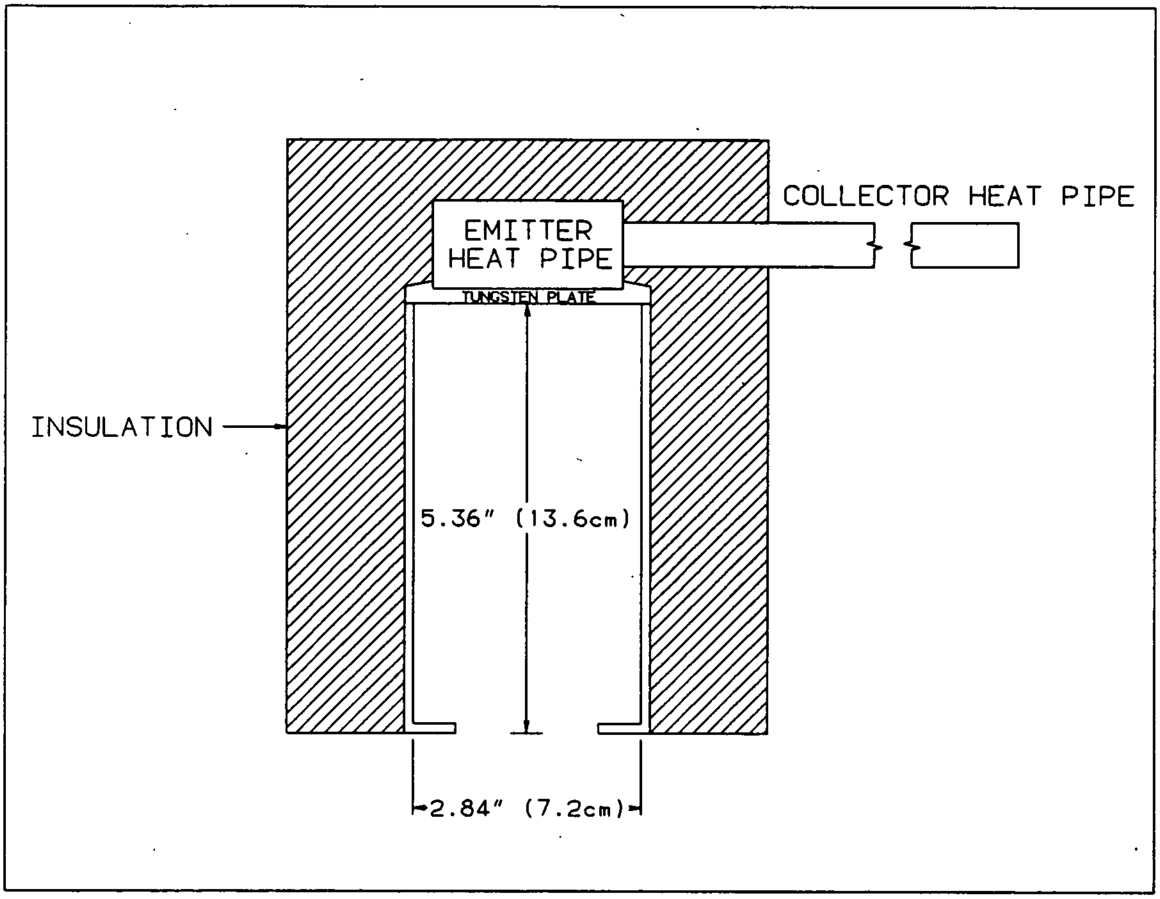

Figure 2-2 Insulating cavity configuration.

2.2.2 *Insulating Cavity*

The configuration for the insulating cavity is shown in figure 2-2. The cavity aperture is sized such that it radiates as a black body with a given energy input from the solar collector of 2050 watts. This energy input, minus the black body radiation back out the aperture and losses through the cavity wall, gives a net of 1350 watts which remain in the cavity and are eventually transferred into the heat pipe (1). A solar concentrator is used to guide light rays into the cavity. This design will be explained in detail in section 2.3.

2.2.3 *Insulation*

The cavity is insulated with multi-layer foil insulation. Tungsten foil was chosen to be used for the first layers where the temperatures will be the highest. As the temperatures become more manageable, a less expensive foil such as aluminum, will be used.

2.3 *SOLAR COLLECTION*

2.3.1 *Solar Design Constraints*

It was determined that the average solar flux available would be approximately 0.1385 W/cm^2. From this, an approximate size for the concentrator was determined to meet the power requirements. Development of the solar collection devices were influenced by several factors. Elements that influenced the solar collector's design include the following:

1. Maximum possible solar collecting area
2. Fit in Delta II secondary payload
3. Highest possible efficiency
4. Concentrate solar energy to a small area
5. Operate in space for one year
6. Withstand acceleration forces during liftoff

Because of these requirements, many methods of collecting and concentrating solar energy were looked at before a final configuration was finalized.

2.3.1.1 Fresnel Lens. Using a Fresnel lens was initially looked at as a method of concentrating solar energy. A Fresnel lens has a series of concentric stepped zones that extend from the center to the outer margins. As light passes through the lens, each concentric line acts as part of a lens to focus the light. The Fresnel lens was ruled out because of the material that the lenses are manufactured from. The material that these lenses are most commonly made from is either a plastic or a glass. The plastic and glass are eroded or clouded by the monatomic oxygen present in the Earth upper atmosphere. Because of the effect that the monatomic oxygen has on the Fresnel lens, alternative designs were researched.

2.3.1.2 Inflatable Collector. Another design consideration was to use an inflatable mirror as the solar concentrator (2). The weight of the inflatable mirrors that were examined ranged from 600-800 pounds. Problems of integrating an inflatable mirror in the design were due to the mass and volume constraints imposed on the Delta II's secondary payload. Also, because of the equipment used for attitude control, the large mass of the mirror would cause problems in controlling the attitude of the satellite.

2.3.1.3 Reflecting Mirror. It was determined as the design of the satellite progressed that a parabolic reflecting mirror could be used as the primary solar collector and concentrator (3). A parabolic shape would be optimized so that the largest possible mirror could be fit into the Delta II's secondary payload area. The mirror would be placed over the top of the bus and then be deployed to its operating position. Using a reflecting mirror for the solar concentrator proved to be the easiest and most effective way of collecting and concentrating the solar energy for our satellite system.

2.3.2 Solar Concentrating Mirror

2.3.2.1 Mirror's Configuration and Operation. To use a reflecting mirror as a solar collector, the geometry of the mirror needed to be designed so it would concentrate the solar energy to a small area. To accomplish this, a parabolic shape mirror was utilized (3). Not only does the parabolic shape concentrate the solar energy to a small area, but it allows the point at which the energy is to be concentrated (the focal point of the mirror) to be set as required. Two of the design constraints placed on solar collection were to have as large of a collecting area as possible yet the lightest possible weight for the structure. It was determined that the mirror could be manufactured from either a light weight composite or 6061-T6 aluminum alloy and still meet the design requirements. The size of the mirror was limited by the nonuniform volume of the Delta II's secondary payload area. The mirror's dimensions were maximized with the shadow area becoming a rectangle measuring 58.6 x 41.6 inches as shown in figure 2-3. The focal length was designed as shallow as possible and was set at 40 inches. This would allow the mirror to fit over the bus in the secondary payload area.

2.3.2.2 Mirror's Reflective Surface. The mirror was designed as a first surface reflector. To accomplish this, an enhanced aluminum alloy will be deposited on the surface of the structure to form the reflective surface (4). It is common practice to produce a reflecting device by depositing a highly reflective metal material on a surface. This will produce a reflective surface with a maximum reflectivity of 95% and will, therefore, have a low absorbtivity. This meets the design constraint of having a high efficiency concentrator/ collector. Warping due to thermal expansion caused by the

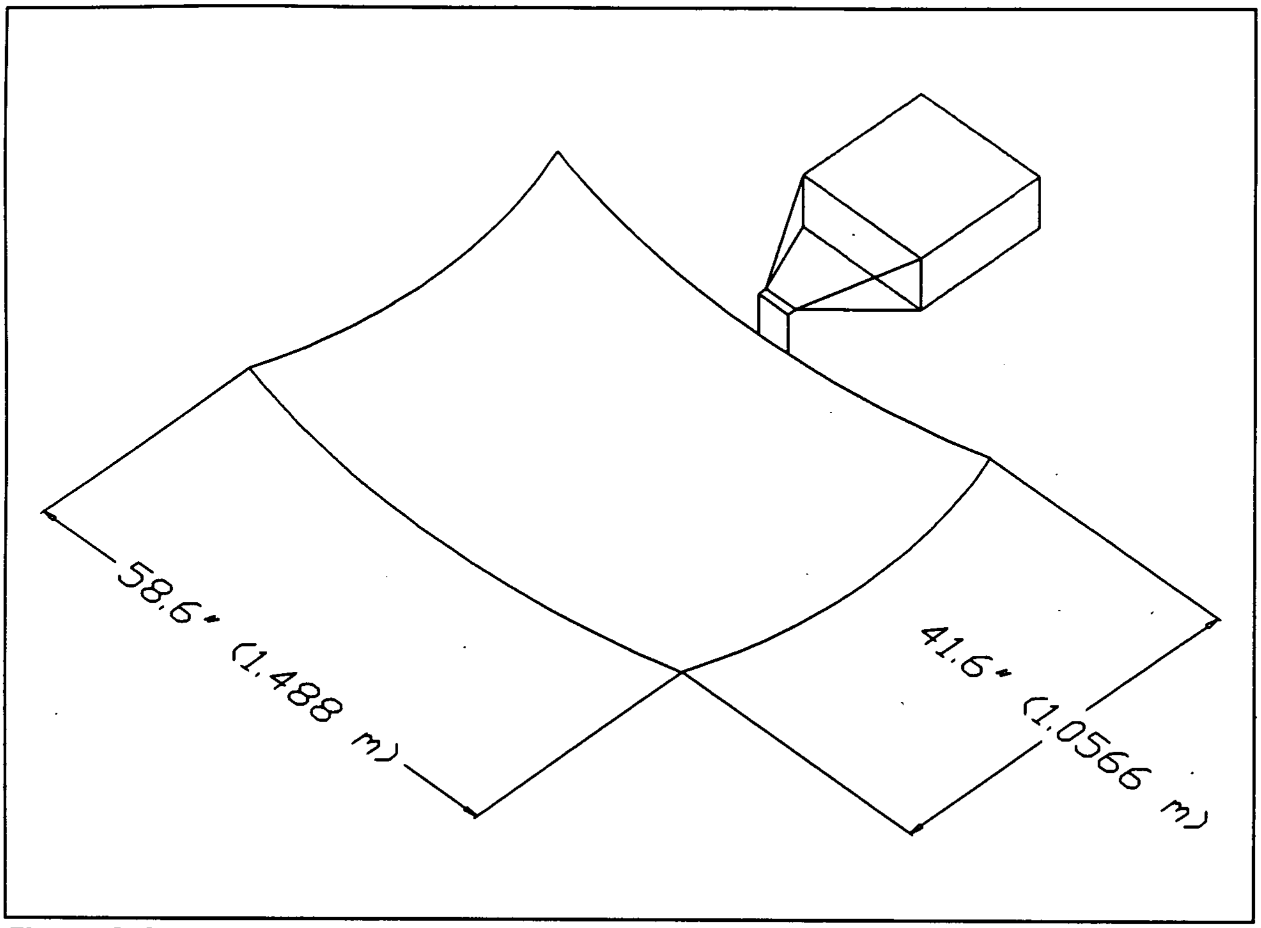

Figure 2-3 Mirror dimensions.

thermal cycling of the mirror will not effect the efficiency of the reflective surface, but there needs to be further consideration of the effect this has on the structure of the mirror. The atomic oxygen environment of space will not effect the aluminum reflective surface of the mirror, but the reflective surface will have to be protected after the mirror is manufactured and during launch from the Earth's atmosphere.

2.3.2.3 Structure and Strength of the Mirror. In the micro-gravity environment of space, the mirror will have only the forces of solar pressure and attitude control acting on it. The real structural support of the mirror needs to be during the launch of the satellite. It was determined that the mirror's structure would need to withstand forces of up to 20 g's. This figure was determined by lumping the worst expected acceleration forces and random vibration into the same figure and adding in a small safety factor. It was agreed upon that as a worst case this figure would be 20 g's.

Using this as a constraint, the mirror was modeled as a cantilever beam and subjected to this 20 g force. This analysis determined the final structure of the mirror.

The use of either composite materials or aluminum alloys was looked at for manufacturing the support system (5). Because of outgasing problems, atomic oxygen erosion in orbit, and the higher costs of manufacturing composites, an aluminum alloy was selected. The aluminum alloy chosen was 6061-T6 aluminum. This alloy has previously been space proven, is light weight, easy to work with, and reasonably priced. The mirror's main structure, upon which the reflecting surface will be deposited, is a 1/16" thick sheet of 6061-T6 aluminum alloy formed to the parabolic shape. A frame, running completely around the perimeter, will then be placed on the back side of the mirror as shown in figure 2-4. This frame will be made from a 1" x 3/8" bar of 6061-T6 aluminum alloy.

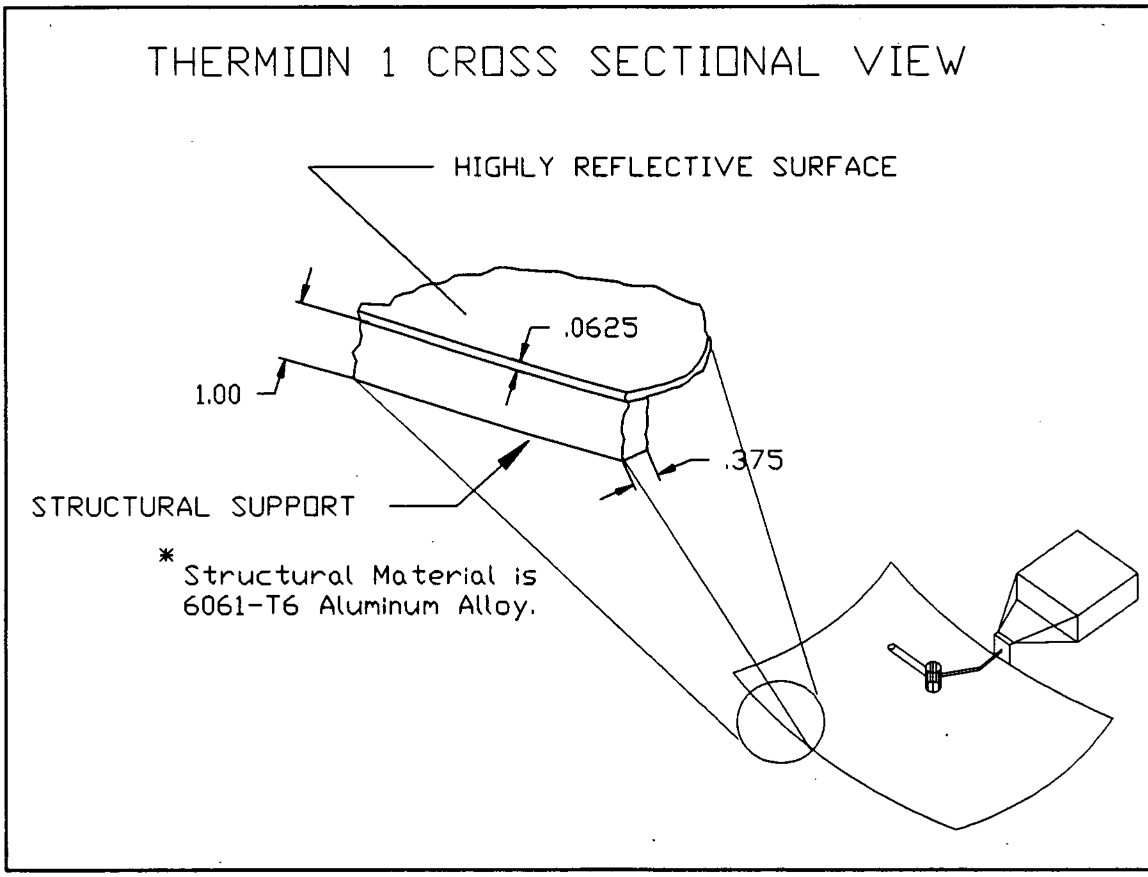

Figure 2-4 Cross sectional view of structural support frame for the mirror.

Through analysis of this configuration, the mirror was theoretically shown to withstand the design criteria of the 20 g load. A testing plan should be incorporated to verify these results. The final mass of the mirror and its support structure is 23.2 lbs or 10.5 kg, this is an acceptable mass and can be controlled by the methods selected by attitude control. Special consideration needs to be given to how the mirror will rest over the bus while in the launch configuration. Soft pads could be placed on the reflective surface for the mirror to be supported against. If this is done there will need to be extra reinforcement on the back side of the mirror where these pads will be placed. This will give the mirror hard points to protect against any damage by the pads that may occur during the launch.

2.3.3 Secondary Concentrator

The satellite has a pointing accuracy of ± 0.75 degrees. Due to this pointing error, the mirror will not focus the energy at a precise area constantly. To control the deviation where the mirror will concentrate the energy, a secondary concentrating device was place at the entrance of the heat pipe cavity as shown in figure 2-5. The concentrating device employed is called a Compound Parabolic Concentrator (CPC) (6,7,8). The CPC not only takes care of the deviation where the mirror focuses the energy, but will also function to minimize energy losses

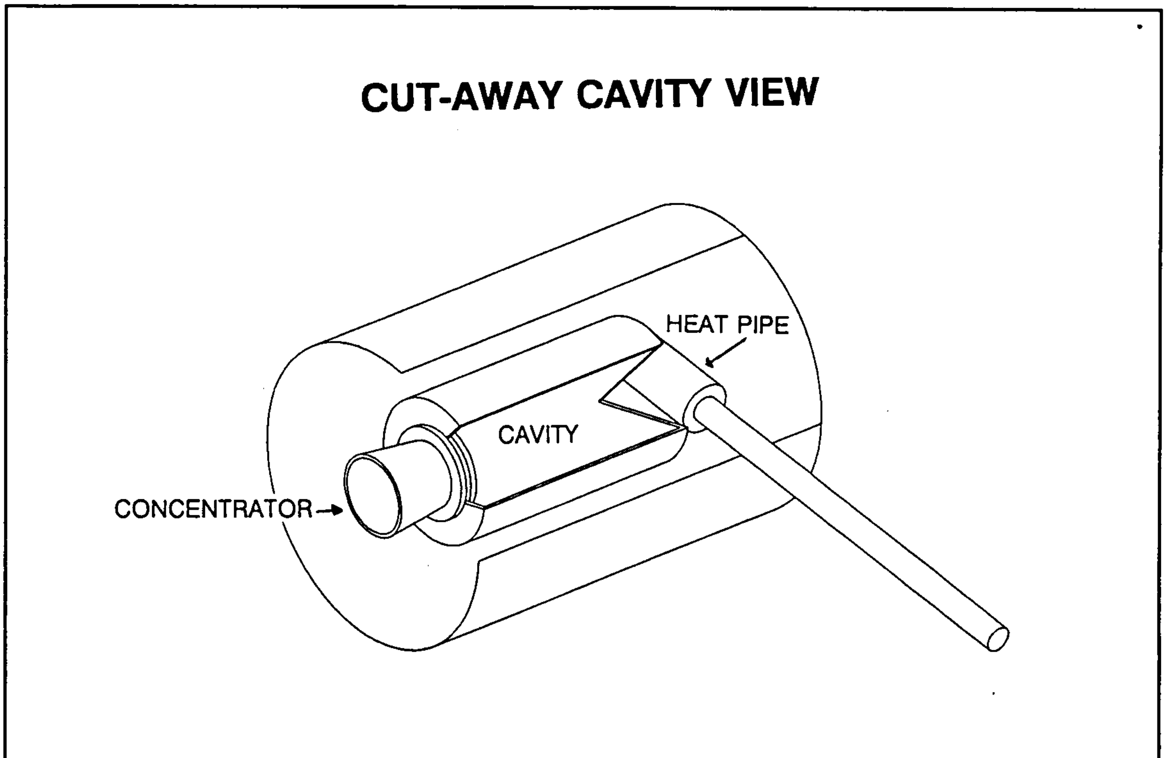

Figure 2-5 Compound Parabolic Concentrator.

from the cavity by reducing the exit aperture of the cavity. To increase the efficiency of the concentrator, it was purposed that the cone be a solid piece of sapphire because of its high index of refraction (8). The high index of refraction enables the concentrator to bend the incoming light from wider angles to the desired angle. This allows the solar energy to be concentrated to supersolar intensities (8). Further information is needed on the sapphire material to ensure that it will operate as intended. If the sapphire cone does not perform as expected, a hollow concentrator with a highly reflective coating on the inside surface will be used. A ray-trace on the hollow concentrator was preformed to ensure that the light will be properly reflected into the cavity as shown in figure 2-6.

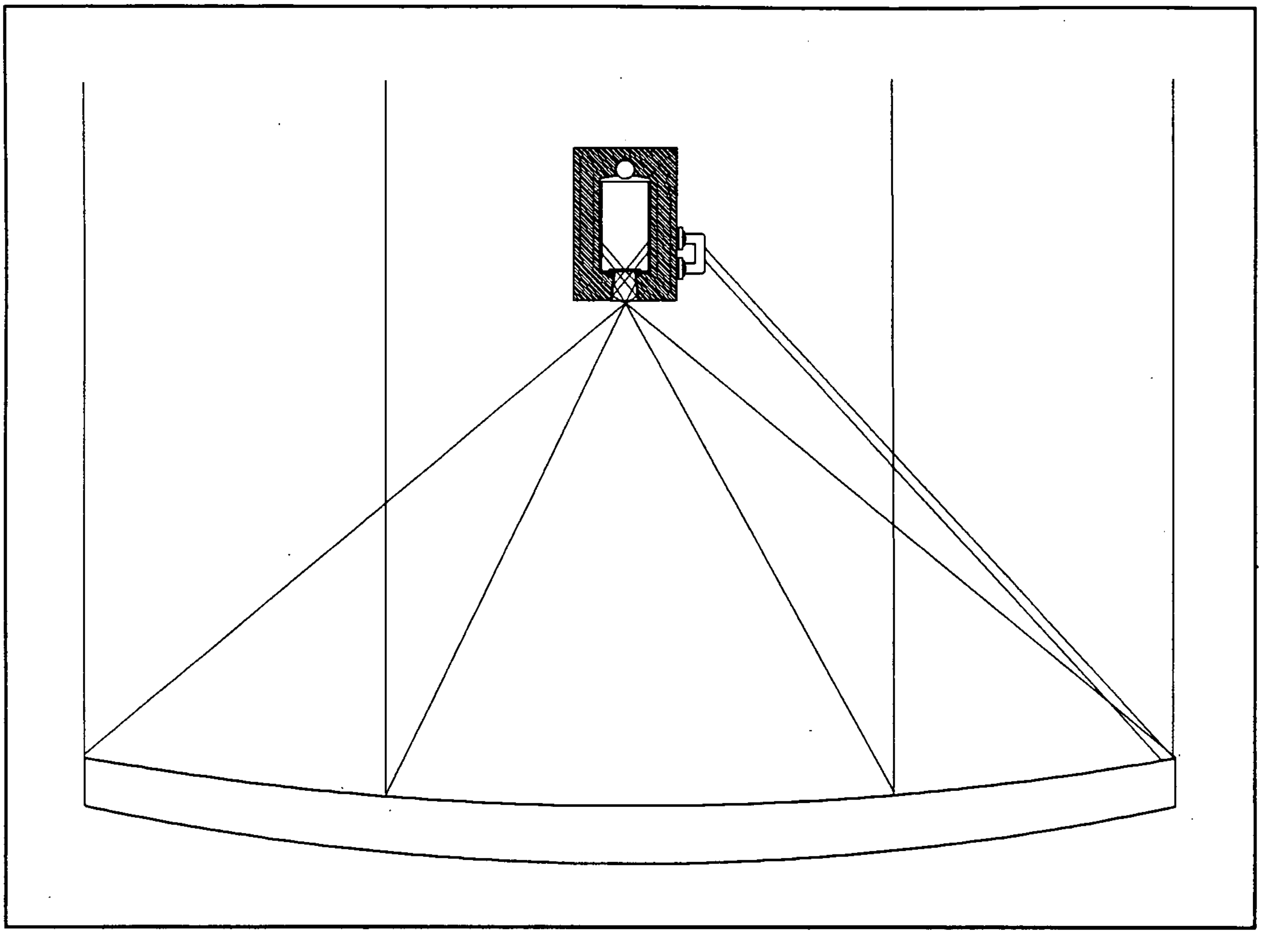

Figure 2-6 Raytrace of secondary concentrating cone.

2.4 POWER MEASURING DEVICE

The power output expected from the scaled version of Thermacore's thermionic generator is 110 watts. In order to verify that this power is generated, a power measuring device is implemented in the design of the satellite. This device allows the power to be deduced from a temperature measurement rather than measuring it directly. This was necessary because conventional power measuring instruments could not contact the hot tungsten wire and remain operational.

2.4.1 Conceptual Operation

The power measuring device is a simple design consisting of an insulated tungsten wire and a high temperature thermocouple as shown in figure 2-7.

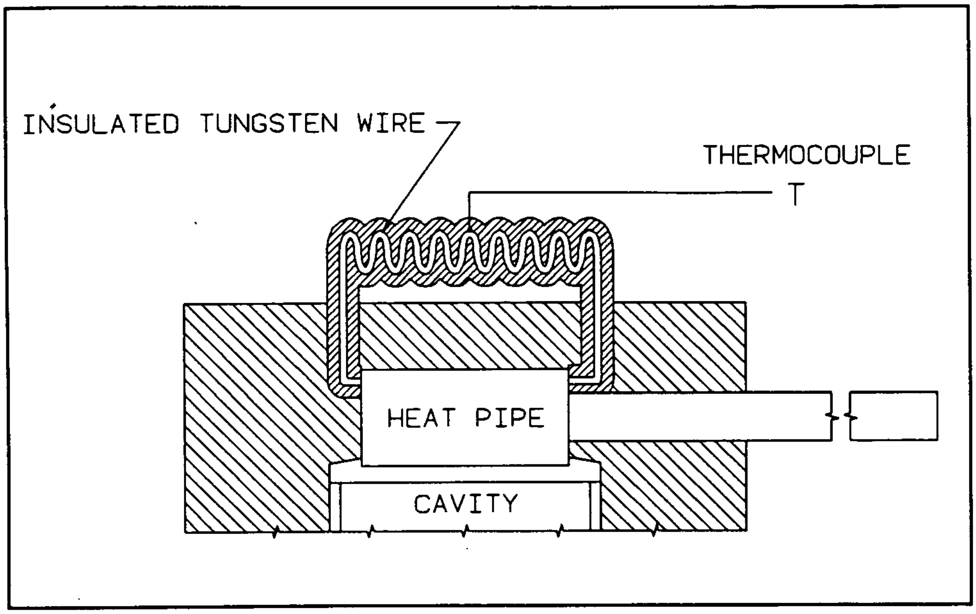

Figure 2-7 Power measuring device.

The tungsten wire is connected to the power leads of the generator and a current of approximately 170 amps passes through it. The wire is well insulated to prevent the loss of electrical power to the atmosphere. The temperature of the wire is approximately 2330 Kelvin. In order to withstand this high temperature, a tungsten-rhenium thermocouple is connected to the center distance of the wire. Based on thermal analysis of the wire, an equation for the power generated as a function of temperature is derived. A detailed analysis of the wire was completed with the aid of Dr. J.C. Batty and is included in Appendix B.

2.4.2 Governing Equations

An energy balance of the wire shows that the heat per unit volume of tungsten is equal to the product of its thermal conductivity and the square of the temperature gradient throughout the wire:

$$-q = k(dT/dx)^2$$

The heat per unit volume in the wire is related to the power drawn from the thermionic device by the following equation:

$$q = I^2R/(2\Pi DL)$$

Combining these equations and integrating, an expression for the power as a function of temperature is obtained:

$$P = I^2R = (T - T_o)4\Pi Dk/L$$

T is the temperature value measured at the wire center length (estimated to be 2330 K); T_0 is the boundary temperature estimated to be 1000 K. The other variables in the equation are associated with material properties and geometry.

The above equation is only an approximation of the power generated by the thermionic heat pipe generator. A constant thermal conductivity and a boundary temperature of 1000 K were assumed. Any error introduced to the system by the simplifying assumptions can be overcome by extensive testing and calibration of the power measuring device before launch.

2.5 TEMPERATURE MEASUREMENT

One of the requirements given to the payload subsystem was to set up temperature measuring devices. This is an important consideration in order to verify that the heat pipes cycle properly in micro-gravity.

2.5.1 Sensor Configuration

The temperature distribution along the length and circumference of the emitter heat pipe and along the length of the collector heat pipe is measured. As shown by figure 2-8, six thermocouples are evenly spaced along the length of the collector heat pipe, and six thermocouples are evenly spaced along the circumference and length of the emitter heat pipe. This configuration will allow isothermalization of the heat pipe to be verified. A complete listing of thermocouple specifications is included in Appendix B.

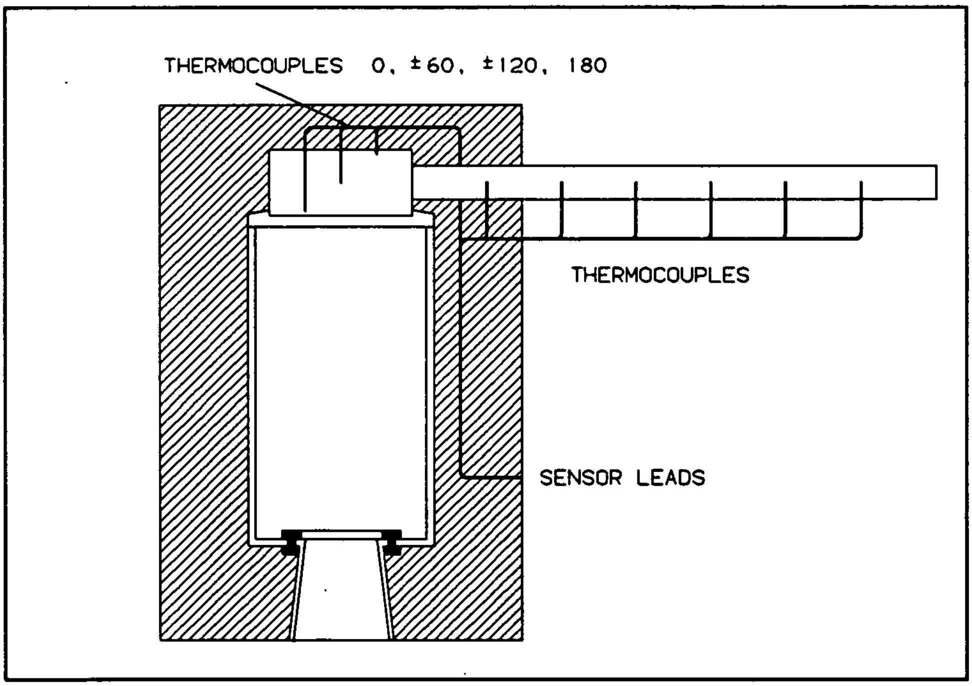

Figure 2-8 Placement of thermocouples.

2.6 CONCLUSIONS

2.6.1 Conceptual Design

The payload design meets the design requirements in concept. The mirror is sized to fit in the secondary payload space of the Delta II launch vehicle. It also delivers 2050 Watts to the insulated cavity, which contains the energy and transfers the required 1350 Watts to the thermionic heat pipe device. Temperature sensors and a power measurement device are implemented to collect data during the operation of the thermionic generator.

2.6.2 Options and Concerns

One alternate design to consider is the use of a disk-shaped heat pipe in place of the conduction plate to enhance the heat transfer from the cavity to the emitter heat pipe. A major concern with the operation of the heat pipe is thermal cycling of the tungsten. A crude heat transfer analysis (see Section 9.5) on the heat pipe indicated that after the first cycle, it would never drop below a temperature of 550 K. This temperature is high enough to avoid damage to the tungsten components. The availability and use of multi-layer tungsten foil needs to be researched more carefully. A more complete thermal analysis can then be done to verify that the heat pipes will thaw and reach the required operating temperatures during the first cycle.

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3.0 ATTITUDE DETERMINATION AND CONTROL

The attitude control system determines the orientation of the satellite in three-dimensional space, points the solar collecting dish at the sun, and maintains this orientation while the sun is not hidden by the earth.

3.1 DESIGN REQUIREMENTS

The final requirements placed on the attitude control system for this project were:

1. point the solar collector at the sun to within a tolerance of ± 0.75 degrees,
2. use flight proven technology and components,
3. use sensors and actuators which fit inside the dynamic envelope of the launch vehicle (see section 8.2) and which in no way would endanger the primary payload, and
4. minimize cost as much as possible.

3.2 DESIGN EVOLUTION

Three axis stabilization was chosen early in the design process due to the pointing requirements and the fact that excessive spin on the satellite would violate the micro-gravity requirements of the payload.

A tradeoff study was conducted early in the design to determine the relative advantages of different types of sensors and actuators. Sun sensors, star sensors, photo diodes, horizon crossing sensors, magnetometers, and accelerometers were considered for attitude determination. Of these, two sun sensors, two photodiodes, and a rotating horizon crossing sensor were chosen for the final design.

Actuators which were considered include gas thrusters (cold gas, monopropellant, and bipropellant), torque rods, momentum wheels, and ion propulsion systems. Initially, cold gas thrusters were chosen for this mission. Hydrazine based mono and bi-propellants endangered the primary payload, ion propulsion required excessive amounts of power, momentum wheels were initially thought too bulky for the limited space, and torque rods were initially thought too weak for this mission.

Uncertainty concerning the amount of propellant required for a one year mission and the threat that the tanks of compressed gas would endanger the primary payload forced the design to change to torque rod propulsion, stabilized with a small momentum wheel.

3.3 *ATTITUDE DETERMINATION SENSORS*

The combination of the sun sensors and the horizon crossing sensor will yield two vectors in space from which the three dimensional attitude of the satellite will be determined. Attitude determination by this method will only be possible while the satellite is out of the Earth's shadow.

The locations of the sensors on the satellite are shown in figure 3-1.

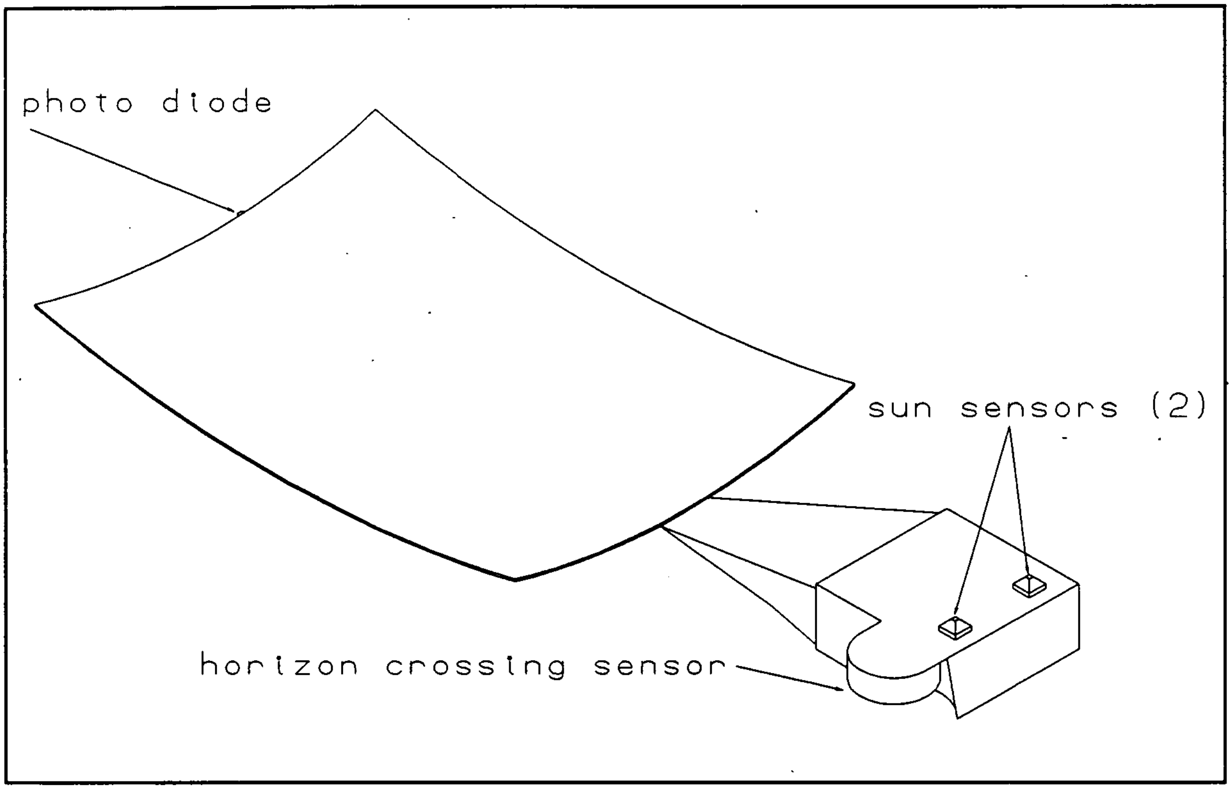

Figure 3-1 Positions of the attitude determination sensors.

Specific information for each sensor is listed in Table 3-1

SENSOR	*COARSE SUN SENSOR*	*FINE SUN SENSOR (2)*	*HORIZON CROSSING SENSOR*	*PHOTO DIODES (2)*
VENDOR	Adcole	Adcole	Ithaco	UDT
FIELD OF VIEW (Degrees)	Total: ± 135 Linear: ± 30	Total: ± 8.0 Linear: ± 2.0	45 Degree Cone	± 90
SENSITIVITY (degrees)	± 0.5	± 0.15	± 0.1	NA
SAMPLE RATE (Hz)	(Analog) 1	(Analog) 1	(Digital)	(Analog) 1
POWER (Watts)	0	0	1.1 Ave	0
MASS lb (kg)	0.5 (0.227)	0.5 (0.227)	3.53 (1.6)	.14 (.06)
VOLUME in^3 (cm^3)	3 (49.2)	3 (49.2)	93 (1524)	.03 (.51)
TEMPERATURE RANGE (Degrees C)	-140 +100	-44 +66	-20 +70	-20 +60

Table 3-1 Technical information for attitude determination sensors. (1,2,3)

3.3.1 Sun Sensors

Two sun sensors were selected for the final design due to the combination of a precise pointing requirement for steady state operation and a large field of view for initial pointing. High accuracy sensors tend to have limited fields of view, while wide angle sensors have lower accuracy. Therefore, a wide angle, low accuracy sensor will be employed for initial pointing and a high accuracy sensor with a narrow field of view will be used during steady state operation. The coarse sensor is a two axis sensor consisting of a pyramid of four detectors. The fine sensor consists of two single axis sensors mounted 90 degrees to each other. Both of these sensors will be mounted so that the null reading points the solar collector directly at the Sun.

The collection plate and radiator will emit light due to their high temperatures. The Sun sensors must be shielded from this light. Otherwise there would be ambiguity as to which light source was the Sun during steady state operation. A photodiode will determine if the Sun is in the area which has been shielded from the sensors. Another photodiode will be placed on the opposite side of the bus from the Sun sensors to detect if the satellite is facing away from the Sun.

3.3.2 Horizon Crossing Sensor

The horizon crossing sensor and the momentum wheel are a single unit. They are mounted in the same housing and are spun by the same motor. Horizon crossings are detected due to differences in the collected radiation. This sensor uses a single cone scanner. Altitude data must therefore be telemetered to the satellite in order to correctly determine the vector from the satellite to the earth.

The position of the Marman clamp mounting plate forced the horizon crossing sensor and momentum wheel to be extended from the satellite bus due to interference with the horizon crossing sensor's field of view.

3.4 ATTITUDE CONTROL ACTUATORS

The attitude control actuators must maintain the position of the solar collecting mirror within the prescribed tolerance of ± 0.75 degrees in the presence of external torques. Figure 3-2 shows the locations of the various actuators and related equipment.

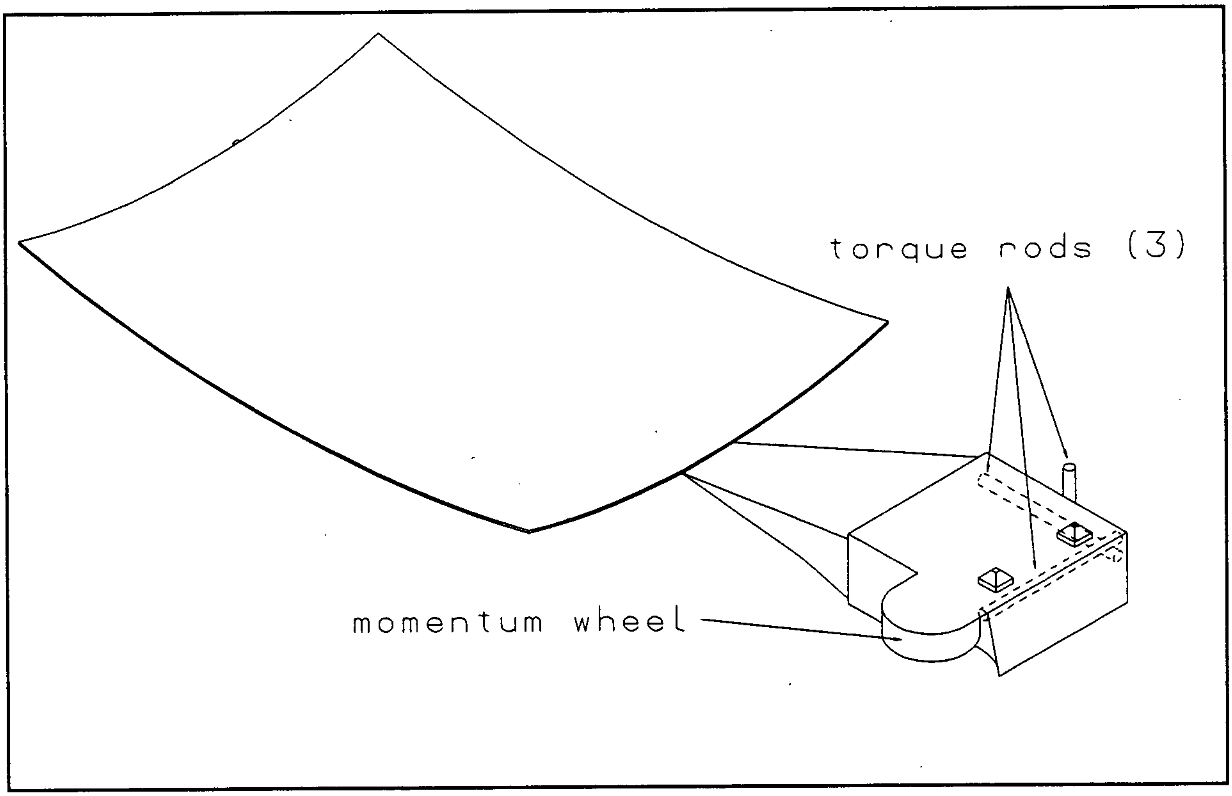

Figure 3-2 Positions of control actuators.

Table 3-2 gives specific information about the actuators.

COMPONENT	*MOMENTUM WHEEL*	*TORQUE RODS (3)*	*MAGNETOMETER*
VENDOR	Ithaco	Ithaco	Schonstedt
AVAILABLE TORQUE (N M)	20×10^{-3}	0.45×10^{-3}	NA
POWER (Watts)	3.1 (Ave)	1.35 at saturation	1.0
MASS lb (kg)	7.9 (3.6)	3.0 (1.36)	1.3 (.595)
VOLUME in^3 (cm^3)	246 (4035)	17.3 (283.8)	55.2 (904.5)
TEMPERATURE RANGE (Degrees C)	-20 +70	-55 +71	-18 +74

Table 3-2 Technical information for control actuators. (1,3,4)

3.4.1 Torque Rods

The attitude of the satellite will be altered primarily by torque rods. Torque rods consist of a ferromagnetic core and wire coils, similar to an electromagnet, which create a magnetic field longitudinally along the rod. Torque is produced on the satellite as the induced magnetic field interacts with and tries to align itself with the magnetic field of the Earth.

Three torque rods will be mounted orthogonally on the satellite bus. Due to space limitations along the shortest dimension of the bus, one of the torque rods will be mounted in the same plane as the other two, and then deployed to an orthogonal position at the same time that the payload and mirror are deployed. A spring and positive locking mechanism will accomplish this deployment. This mechanism will be discussed further in section 4.3.

A three axis magnetometer will be used to provide the magnitude and direction of the Earth's magnetic field. Although this information is detectable only to approximately $\pm$ 1.0 degrees, this will not significantly degrade the pointing accuracy of the control system due to cosine relationship between magnetic fields and the fact that the cosine of a small angle is very close to unity.

3.4.2 Momentum Wheel

The momentum wheel will be placed such that its spin axis is aligned with the vector normal to the solar collector and Sun sensors. This orientation will take maximum advantage of the gyroscopic stability of the momentum wheel. A control torque about this axis, approximately 45 times stronger than that of the torque rods, is available by changing the rotational speed of the momentum wheel. This torque will allow a control algorithm to find the optimum torque rod orientation in the plane of the momentum wheel.

A momentum wheel will eventually saturate as it is used to torque a satellite. To prevent this, the torque rods will be used to perform occasional "momentum dumps", wherein the momentum wheel speed is decreased or increased as required but the satellite position and velocity are not changed.

The momentum wheel will be spun up while the satellite is still connected to the Delta II, but after separation of the second stage from the primary payload. The increase in angular momentum of the satellite will thus be compensated for by the launch vehicle's attitude control system.

3.5 ENVIRONMENTAL TORQUES

In the space environment the torques acting on a small spacecraft are very small compared to torques experienced in daily activity on the Earth. In the absence of larger torques, however, these minuscule torques determine the attitude of the satellite.

The sources of torques considered were solar pressure, gravity gradient, aerodynamic force, radiation pressure, magnetic forces, meteoroid impact forces, and radiation forces from transmitting antennas. It was determined that the two dominant environmental torques were caused by the aerodynamic forces and solar pressure. A comparison of the torques experienced by the satellite from the environment and control actuators is presented in Table 3-3.

SOURCE	MAGNITUDE (N M)
Momentum Wheel	2.0×10^{-2}
Torque Rod	4.5×10^{-4}
Atmospheric Drag	2.0×10^{-5}
Solar Pressure	5.0×10^{-5}

Table 3-3 Environmental and control actuator torque comparison. (5)

As shown in the table above, the torque from the torque rods is over six times larger than the combination of the most significant environmental torques. Hence, for typical operation the attitude can be maintained to the desired tolerance. It should be noted, however, that at peak solar activity the solar pressure increases significantly. The resulting torque may reach a value in excess of the torque rod control torque.

This would cause the solar collector to drift to the point where the experiment would be inoperable. Peak solar activity is rare. If the mission is performed at a time of average or below average solar activity, a very small percentage of the mission would be affected by this problem.

3.6 EQUATIONS OF MOTION AND CONTROL LOGIC

Assuming the satellite behaves as a rigid body, Euler's equations which describe the motion may be written as:

$$M_1 = A\dot{\omega}_1 + (C-B)\,\omega_2\omega_3$$

$$M_2 = B\dot{\omega}_2 + (A-C)\,\omega_1\omega_3$$

$$M_3 = C\dot{\omega}_3 + (B-A)\,\omega_1\omega_2$$

where

- A,B,C = Principle axes of inertia
- $\omega_1, \omega_2, \omega_3$ = Spin rate of the principle axes
- M_1, M_2, M_3 = External Torques on the satellite

The satellite was modeled as a dual spin spacecraft (6). Euler's equations were modified to account for the spinning momentum wheel. Since steady state operation will require that deviations from nominal pointing be less than 0.75 degrees, the equations were simplified by using small angle approximations. Euler angles were applied to transfer the motion from body fixed axes to an inertial coordinate system. After adding a position plus velocity feedback controller for two axes, the resulting equations were:

$$\psi = \frac{-C^R}{A}\Omega^R\phi - (k_{p1}\phi + k_{v1}\phi)$$

$$\dot{\phi} = \frac{C^R}{B}\Omega^R\psi - (k_{p2}\psi + k_{v2}\psi)$$

where

- Ω^R = Spin rate of the momentum wheel
- ψ = Angular position about axis 1
- ϕ = Angular position about axis 2
- $k_{p1,2}$ = Position feedback gains
- $k_{v1,2}$ = Velocity feedback gains

The attitude control system will minimize coupling of axes by applying torque about one principle axis at a time.

Attitude determination sensors will provide the necessary information to an algorithm that will determine the satellite's attitude with respect to the Sun. Information from the magnetometers will provide information about the strength and direction of the Earth's magnetic field. Information from the attitude determination algorithm and magnetometers will then be processed by a second algorithm that will determine which axis to rotate about.

Preference will be given to the two axes not aligned with the momentum wheel spin vector. This will allow control torques from the momentum wheel to optimally align the torque rods with the magnetic field in the given plane. The position plus velocity controller will then be used to position that axis within a dead band of ± 0.2 degrees. Once within that range, the dead band for that axis will be relaxed to ± 0.4 degrees and the process will be repeated for the other axis. Position about the third axis is not important as long as the rotation rate is small so that centripetal accelerations do not violate the micro-gravity requirement.

3.6.1 Control Simulation

The software package TUTSIM was used to simulate the satellite motion and the controller during steady state operation. A simulation of the satellite using the equations of motion along with the controller in operation is given in figure 3-2 (see page 3 - 9).

Rotations of 0.4 degrees are shown about two axes. The two rotations are then reduced to 0.2 degrees one at a time. Both of the axes simulated were perpendicular to the momentum wheel. Significant oscillations can be observed due to interaction between the momentum wheel and the control torques. The oscillations will dampen out as energy is dissipated by structural flexure. If necessary, a nutation damper could be added to increase the energy dissipation rate.

3.7 FUTURE DESIGN WORK

Further investigation into the details of the attitude determination algorithm is still to be done. Separate algorithms will be required for transient and steady state cases since inputs from the photodiodes must be considered for the transient case.

Simulation should be done for transient operation in which small angle assumptions are not used in the model. A closer look should be taken at how much the satellite drifts while in the shadow of the earth. If this is excessive, attitude determination may be obtained from magnetometer data to keep the drift to approximately one degree. This would require the up-linking of maps of the Earth's magnetic field as well as additional computer resources allocated to attitude control.

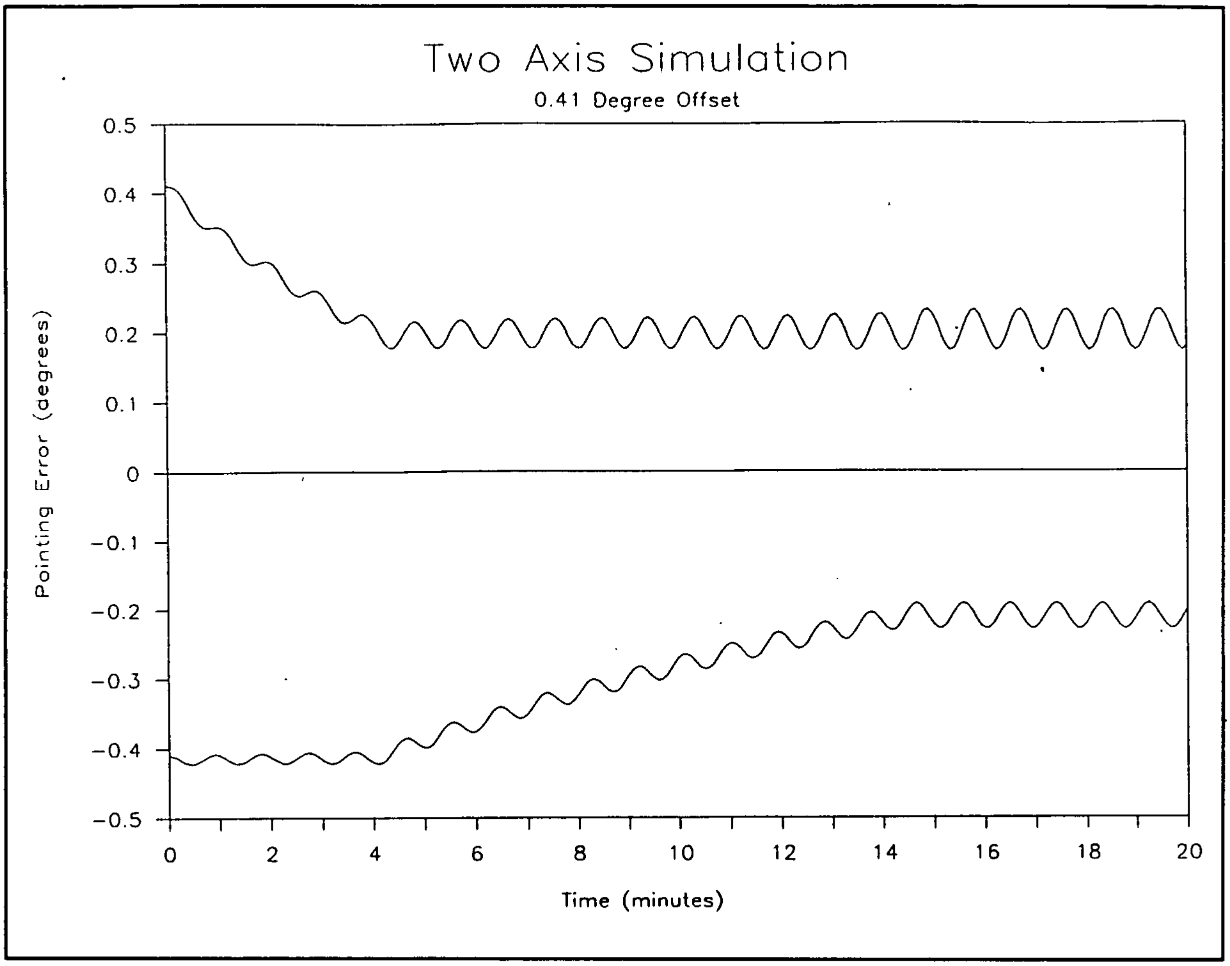

Figure 3-2 Control simulation for two axes.

With the addition of a nutation damper the proportional plus velocity feedback controller implemented here should be adequate. Increased performance is possible by implementing optimal control techniques. The final control algorithm must be coded as a digital controller for implementation in the satellite's computer. Note that the satellite and the computer have up-linking capabilities which will allow the control equations to be modified if needed.

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4.0 STRUCTURES

This section of the report describes the activities of the structures subsystem. This subsystem was responsible for the actual design of the physical structure as well as integrating all of the subsystems together into one workable design.

4.1 SYSTEM REQUIREMENTS

The following design requirements were imposed and met:

1. Satellite required to fit in Delta II secondary payload
2. Utilize off-the-shelf technology
3. One year life
4. Able to withstand 6-G launch loads

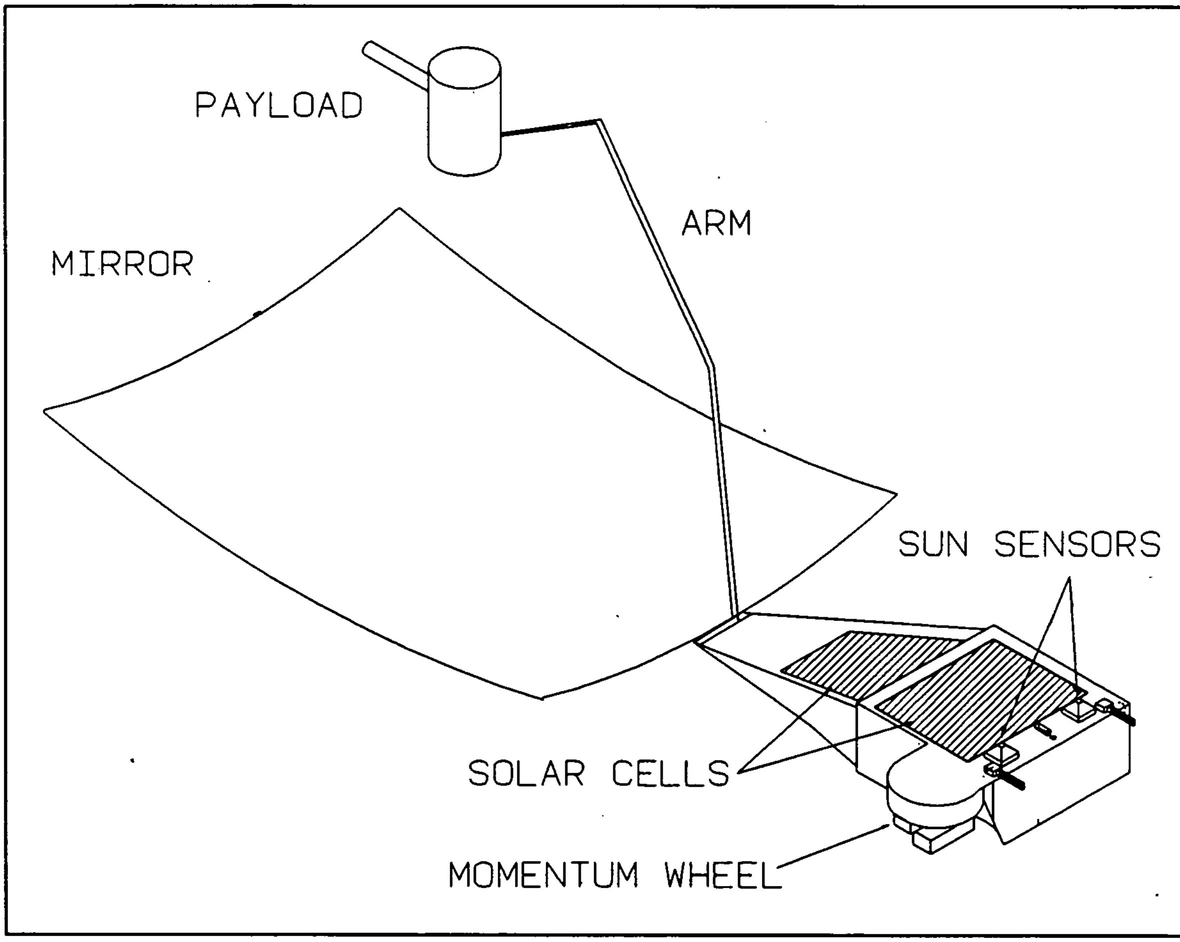


Figure 4-1 Thermion-I Final Design

4.2 STRUCTURE

At the outset of the project, it was determined that the satellite would consist of a relatively large solar collector, the payload, which would sit in the focal point of the collector, and a bus to house the subsystems and equipment.

4.2.1 Design Phases

The mirror was the determining factor in the satellite design. As the mirror configuration was optimized, its shape changed several times in order to maximize power collected while still being able to fit in the launch vehicle payload volume. As the mirror changed so did the rest of the satellite. The initial design consisted of a trough shaped mirror mounted on the top of a rectangular bus. From this point, the design eventually developed into the clam like final design as shown in figure 4-1.

4.2.2 Final Design

The final design consists of the parabolic mirror, the payload, (which is held in place with an aluminum arm), the bus containing the subsystems equipment, and a bracket/hinge configuration to connect everything together. The structure is described in detail below.

4.3 DEPLOYMENT

The mirror and payload are connected to the bus by means of a spring loaded hinge located at the end of the bracket. The hinge allows the mirror and payload to be folded up over the bus into the launch position. The mirror is held rigidly to the bus in the launch position by a bolt that is connected to a bolt cutter mounted on the satellite bus. This launch position allows the satellite to fit in the payload volume and also protects the surface of the mirror and sensors which are mounted on top of the bus. This position also offers more rigidity to withstand the launching forces which the Delta II produces. See figure 4-2. After the satellite has been ejected from the Delta II secondary payload space, it will be deployed into the operating position. Deployment will be accomplished by discharging the bolt cutter. Once the bolt is cut, the torque produced from the compression in the spring loaded hinge will rotate the mirror and payload into their final positions.

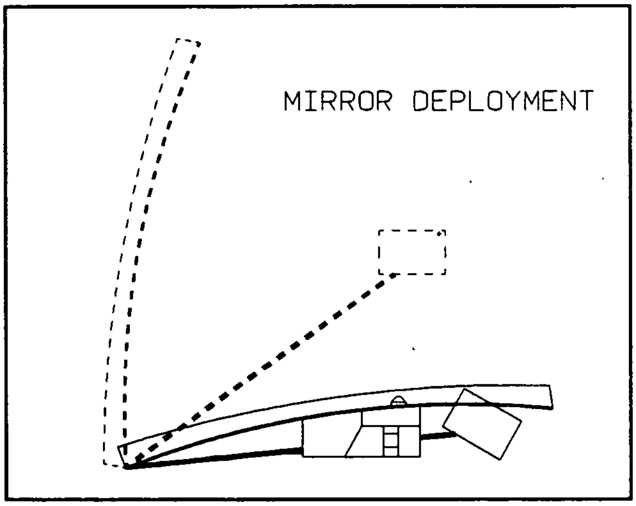


Figure 4-2 Deployment

The hinge is actually two hinges, one for the mirror and one for the arm. The mirror hinge is spring loaded while the arm hinge is not. See figure 4-3 and 4-4. Once the mirror has rotated 53 degrees, a tab on its collar meshes with a tab on the collar of the arm hinge and pulls the arm around into position. The mirror and arm will continue to rotate until they hit rigid stops. Once against the stops, the mirror and hinge will be in their final operating positions. At this point, the spring is still in compression and will hold the mirror and arm in the desired positions for the duration of the experiment.

Supports will help hold the mirror rigidly to the bus in the launch position during the launch. These supports can be seen in figure 4-5. These devices are simple aluminum rods with protective pads on the ends which will contact the delicate mirror surface. The mirror supports are mounted to the top of the satellite by small spring loaded hinges. Once the mirror is deployed and the pressure is removed from these supports, they will rotate down flush to the surface of the satellite top such that they will not interfere with any of the sensors located on top of the bus.

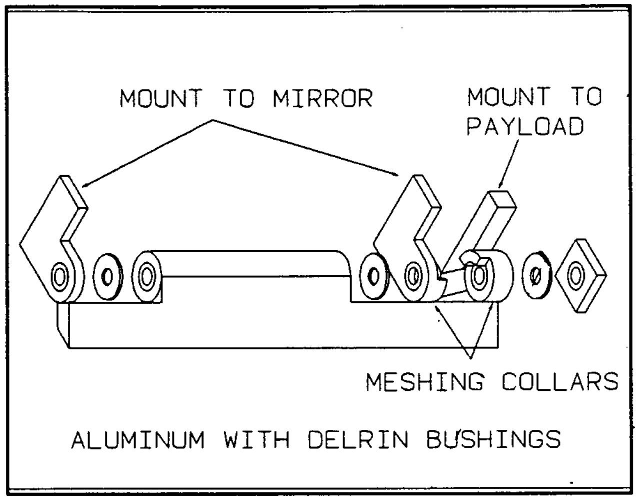


Figure 4-3 Hinge

The exterior torque rod deploys to a position perpendicular to the base plate at the same time as mirror deployment. The torque rod is mounted to the bus with a torsional spring. During launch a bracket mounted to the payload arm holds the torque rod parallel to the base plate. When the mirror and arm deploy the bracket moves away from the torque rod allowing it to rotate into its operating position. A spring loaded pin stops the torque rod in its proper position.

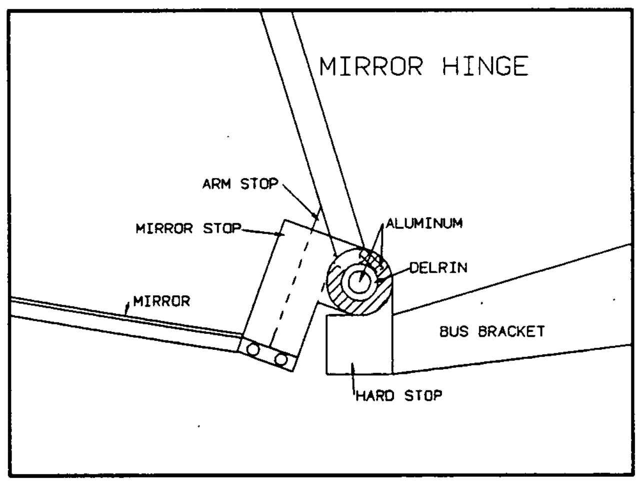


Figure 4-4 Hinge Side View

4.4 MATERIALS

The material selection for the structure of the Thermion satellite was based on three criteria:

1. Previously space proven
2. High strength to weight ratios
3. Ability to withstand maximum accelerations during liftoff and separation.

4.4.1 Subsystem Housing

The subsystem housing or bus is constructed of 6061-T6 aluminum. The bus consists of three major parts:

1. A 0.25 inch thick base plate which all subsystem components are mounted to. The plate is sixteen inches square.

2. The sides consist of a single sheet of 0.04 inch thick aluminum wrapped around the base plate using monocouque type construction. The height of the sides is seven inches.

3. The top is also a 0.04 inch thick sheet connected to the sides using pem screws and nuts.

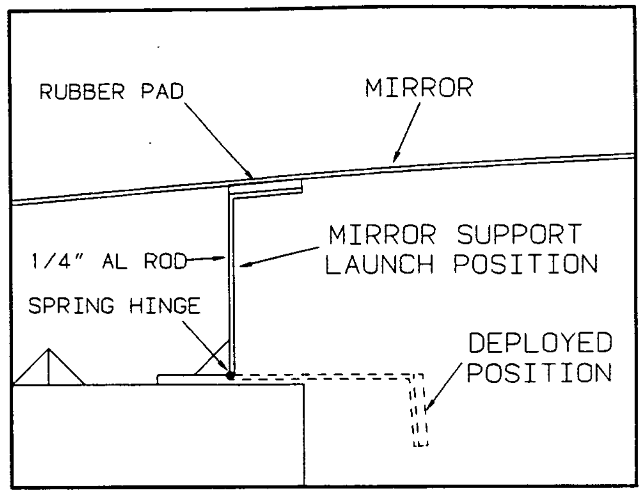


Figure 4-5 Mirror Supports

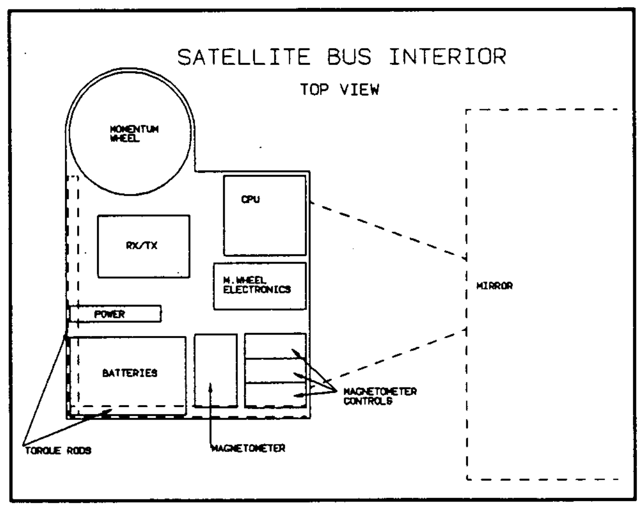


Figure 4-6 Satellite Interior

4.4.2 Attachment Bracket and Hinge

The attachment bracket is also constructed of 6061-T6 aluminum 0.06 inch thick. The rectangular cross section of the bracket tapers from 15.50 x 6.50 inches to 6.00 x 1.00 inches at a distance of 24.06 inches from the bus. The bracket is also a monocouque structure and is bolted to the bus.

The hinge is constructed of 6061-T6 aluminum and Delrin. The pin and sleeve are 6061-T6. The bushings are all constructed of Delrin.

4.4.3 Payload Arm

The arm that supports the payload is a 0.50 inch diameter solid 6061-T6 rod. The rod is bent to allow the payload and arm to fold under the mirror in launch position. The arm also positions the payload in the focal point of the mirror in operating position.

4.4.4 Shielding and Mounts

All radiation and magnetic shielding inside the bus is constructed of 0.25 inch 606 1-T6. All mounts will also be constructed of 0.25 inch 6061-T6.

4.5 SUBSYSTEM LAYOUT

A top view of the interior of the satellite bus is shown in Figure 4-6. The momentum wheel is placed such that the Marman band clamp will not interfere with its function. The torque rods are placed so their magnetic ends will not interfere with the CPU. Sufficient space is allotted between subsystems for mounting and wiring.

Figure 4-7 shows an exterior top view of the satellite. The solar cells that recharge the batteries are mounted on the top of the bus and the bracket using adhesives. The sun sensors, mirror supports and bolt cutters can also be seen in this view.

A third torque rod is mounted to the exterior of the bus. This torque rod needs to be orthogonal to the other two torque rods and will not fit in the interior.

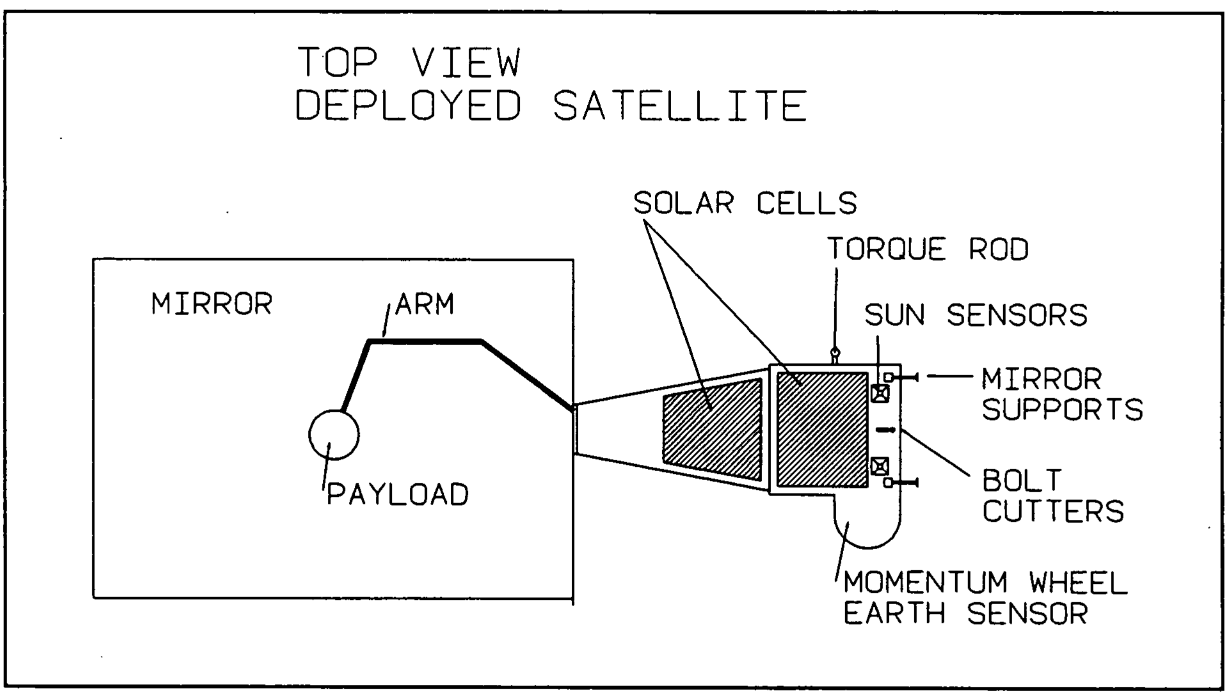


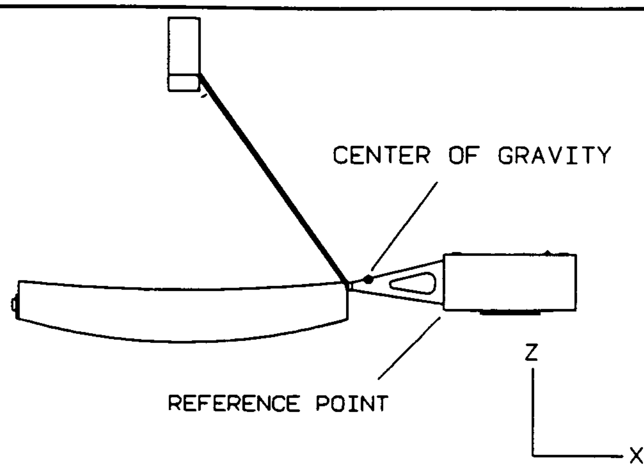
Figure 4-7 Thermion-I Top View

4.6 MASS PROPERTIES

The mass properties of the Thermion satellite are shown in figure 4-8. These properties were determined using the software program "MOMENTS".

4.7 CONCLUSIONS

At this point there are still a few items which require more attention. Analysis has been done, and it is felt that the structure can withstand all expected loads. However, more analysis and documentation needs to be performed. Another item to be addressed includes further analysis to insure that the mirror and arm/payload are held rigidly enough during launching. The mirror and payload are delicate items and crucial to the experiment.



* TOTAL MASS: 85.19 LB
* CENTER OF GRAVITY

X: -9.127 IN

Y: 8.054 IN

Z: 3.839 IN

Ixx: 1.92E04	Ixy: 5.84E02
Iyy: 7.09E04	Iyz: -2.61E03
Izz: 7.19E04	Ixz: 3.83E03

(LBM IN**2)

Figure 4-8 Mass Properties

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5.0 DATA MANAGEMENT

The data management system provides control and processing for the following: data acquisition, attitude determination and control, communications control, charging batteries, and various housekeeping tasks.

This chapter will describe the data management hardware components. Thought was given for software design, but no software has been written to handle any of the data management tasks.

5.1 DATA MANAGEMENT REQUIREMENTS

Data management's major task is data acquisition. Fourteen temperature sensors (thermocouples) line the heat pipe. The data will be collected for 8 orbits then down-linked to a ground station. The data management system was required to select a system that has low power, minimal mass, and a small volume.

5.2 SYSTEM COMPONENTS

The system components will include a CPU, expanded memory card, two data acquisition cards, prototyping card, and storage container. Detailed information on these components can be found in Appendix E.

5.2.1 CPU

The CPU is based on the NEC V20 CPU (8088 microprocessor). It is the Q88 board manufactured by the QSI Corporation. The Q88 CPU board can accept a DC voltage from about 4.8 to 18 volts and only consumes 270 milliwatts. The on-board memory is capable of 128 Kbytes EPROM, 256 Kbytes RAM, and 128 Kbytes EEPROM. The Q88 also has a UART that operates at baud rates from 150 to 38,000 baud.

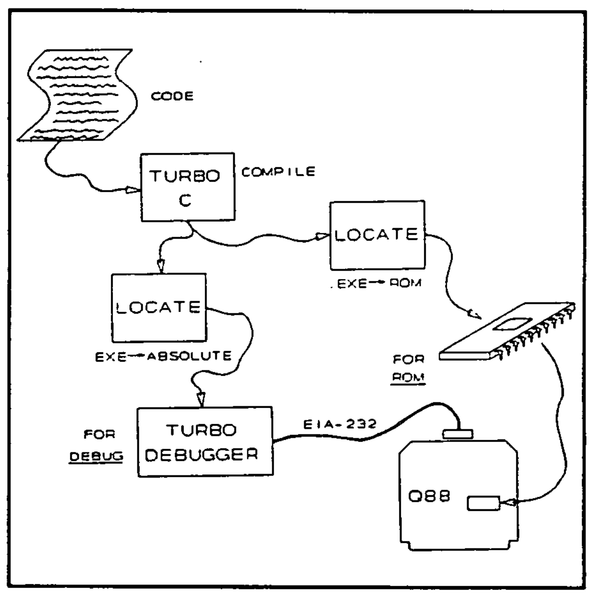

Figure 5-1 Flow diagram of TurboPak.

The main advantage of using the QSI CPU is that it allows Turbo C code to be used in ROM-based system. The process, called TurboPak, uses Turbo C on the Q88 to make ROM algorithms is as follows:

1. Write a C program with an editor.
2. Compile and link program, using the Turbo C.
3. Use Paradigm's Locate program to convert from the .EXE format to the required debug format.

4. Debug the program with it actually running on the Q88 using the Turbo Debugged and Paradigm's Turbo Debugged Remote (TDREM) program.
5. Once the program is bug free, the locate program to make a ROMable version, then EPROM is programmed.

5.2.2 Expanded Memory

The expanded memory card will be used to store the data acquisition and up linking new EEPROM algorithms. The QSI memory card (QMEM-2) can be independently configured for 256 Kbytes to 4 Mbytes EPROMs, RAMs, or EEPROMs. For the mission needs, the card will contain 256 Kbytes of EEPROM and 3 Mbytes of RAM. The EEPROM memory will be used if for some reason new algorithms are required. This can be used to change the data rates or change attitude control algorithms. The RAM memory will be used for storing the test data. The data will be down loaded to a ground station every 8 orbits.

5.2.3 Data Acquisition

The data will be acquired through the QSI data acquisition card (QADC-3). This QADC-3 gives 14-bit resolution, and has a 18 μs conversion time. It is equipped with an instrumentation amplifier. There are 14 temperature sensors, and the card only allows 16 single-ended inputs or 8 differential inputs. Therefore, two QADC-3 cards will be used. The data acquisition cards can be set to individual addresses so that there will be no confusion on which card is being accessed.

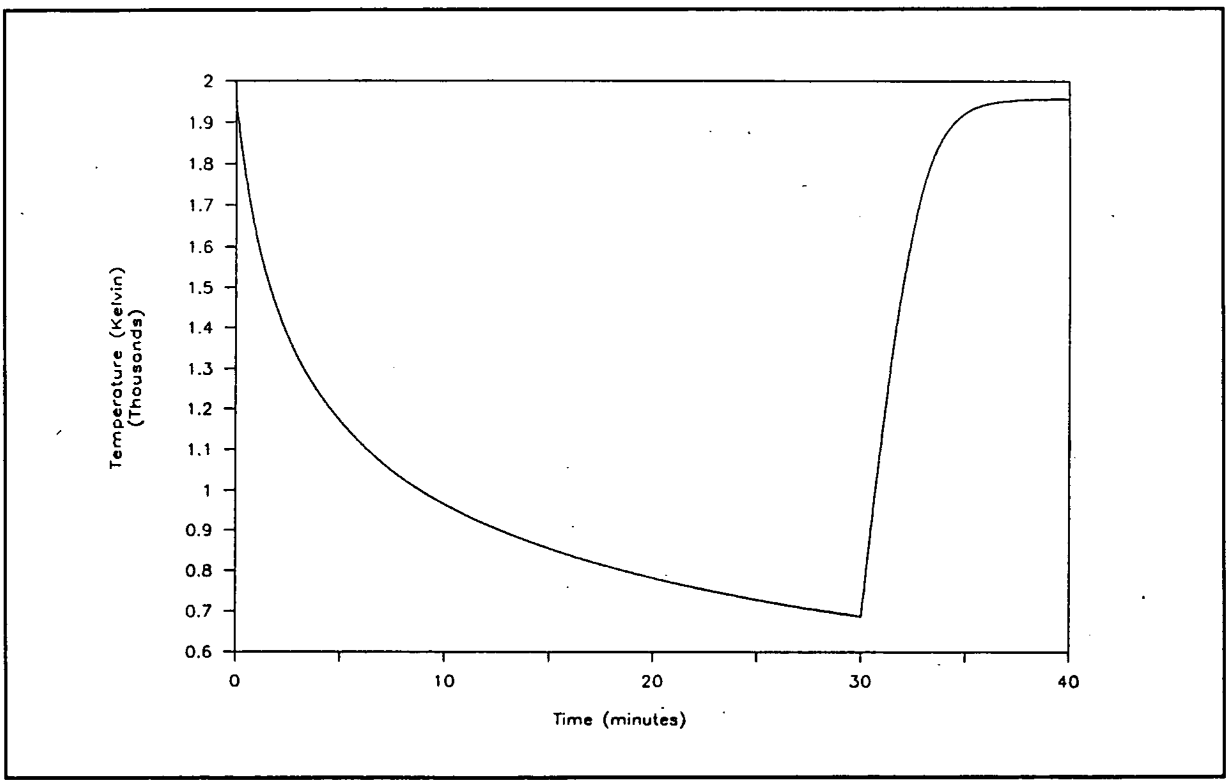

Figure 5-2 Cavity temperature vs. time curve.

5.2.3.1 Cavity Temperature. The cavity temperature vs. time is showed in figure 5-2 (see page 5 - 2). This analysis was a lumped heat capacity model using one node. The program can be found in Appendix I.

The curve starts out with the cavity temperature at about 1950 K, and the satellite orbit is coming into the "dark side." As the satellite orbits in the shadow of the Earth, the cavity temperature falls off, as shown in the figure 5-2. At the end of 30 minutes, the cavity is about 650 K when it enters into the "Sun side." In the next 6 minutes the cavity will reach 95% of its max temperature (1950 K).

5.2.3.2 Data Rates. The data acquisition rates were picked to follow the curve in figure 5-2. Table 5-1 shows how the different data rates break down on the "dark side" of the orbit. These numbers include the 14 temperature sensors. So a 98 Hz sampling frequency is really 7 Hz per sensor.

Dark Side	***Time Interval***	***Frequency***	***Number of data Points***
	Entire 30 min	***7 Hz***	***12,600***

Table 5-1 Data rate break down in shadow of the Earth.

Table 5-2 shows how the different data rates break down on the "Sun side" of the orbit. These numbers also include the 14 sensors.

Sun Side	Time Interval	Frequency	Number of data Points
	First 2 min (0 to 2 min)	98 Hz	11,760
	Next 2 min (2 to 4 min)	42 Hz	5,040
	Next 2 min (4 to 6 min)	9 Hz	1,080
	Next 54 min (6 to 60 min)	3 Hz	9,720

Table 5-2 Data rate break down Sun side of orbit.

A total of 102,857 data points (11.5 Mbytes) can be down loaded every 8 orbits. The total data points taken in 8 orbits is 40,200. This leaves a margin of 62,657 data points. The can be used up by adding additional sensors or changing the data rates.

5.2.4 Housekeeping
All housekeeping electronics will be mounted to a prototyping card that is manufactured by QSI. The interfacing electronics to the computer bus are available through this card.

5.2.5 Storage Container
All the cards will slide in a compact container that is 5.38 x 5.05 x 5.20 in (13.67 x 12.83 x 13.21 cm). The card cage's 6 slots will be completely full with the cards selected for this mission. There are larger card cages available if more components are required.

5.3 RADIATION DAMAGE PREVENTION
To prevent radiation damage to the computer components, which are not RAD hard, 3/16" aluminum plate shielding will be used. This should limit the amount of radiation that enters into the computer area.

5.4 CONCLUSIONS
The computer system that was selected has flown several space shuttle missions, and has sat on the bottom of the ocean for two-year periods. This system offers low power, a small volume and weight, and is inexpensive (see chapter 14 for cost). Software needs to be developed in future work.

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6.0 COMMUNICATIONS

6.1 REQUIREMENTS

The functional requirements of this system are to:

1. Transmit acquired data at least every eight orbits.
2. Provide command up-link to the satellite.

6.2 COMMUNICATION SYSTEM

The communications system for THERMION-I includes both the on board and ground station equipment. An overview of the entire system is presented below.

6.2.1 THERMION-I Satellite Antenna

A functional requirement to the communication system is to be able to transfer stored data from the satellite to the ground from a relatively unknown attitude. This is because the satellite only has to point at the Sun and there is no pointing requirement for the satellite with regards to the antenna. A four antenna system

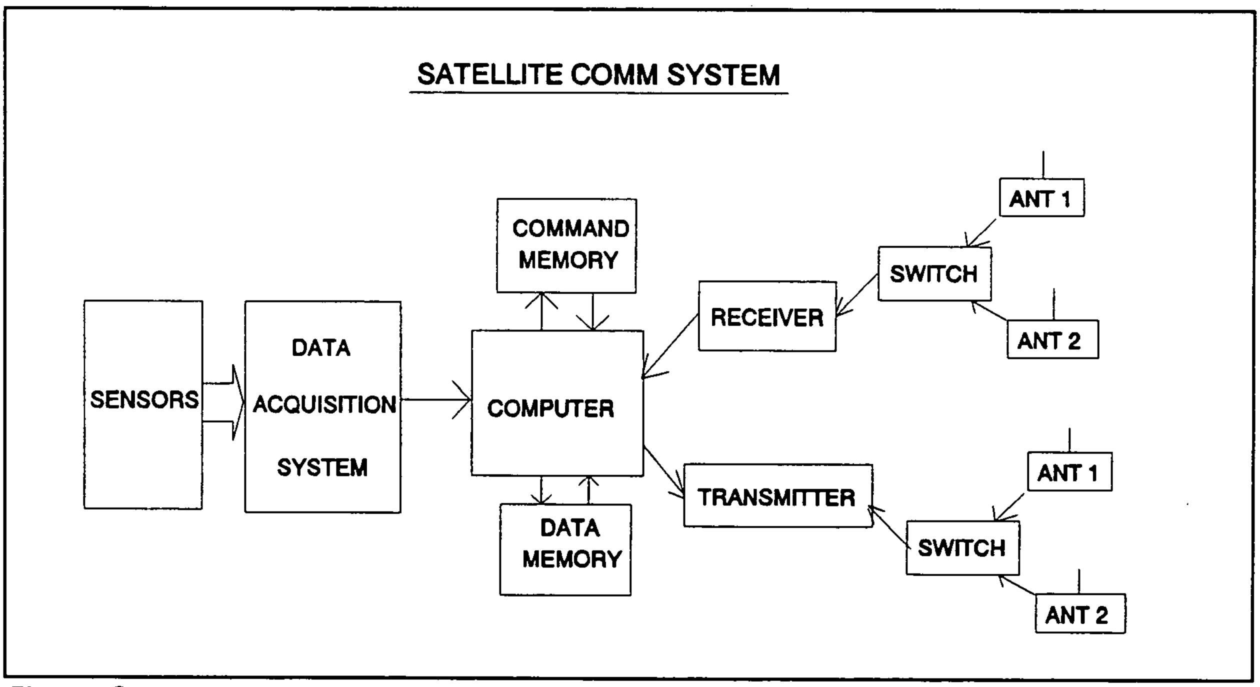

Figure 6-1 Block diagram of communication system for Thermion I.

(two for transmit and two for receive) was chosen in order to maximize the communication links. Figure 6-1 illustrates the antenna connection to the communication system. A switch matrix will be used in conjunction with a software gain maximization algorithm to choose the antenna with the highest gain. The transmit antennas are S band half-wave dipole, while the receiver antennas are helix UHF band. The antennas will be positioned at orthogonal angles with respect to each other and located at the farthest corner from the heat pipe. Final placement will be determined through testing in an anechoic chamber. This testing should provide the optimal placement for maximum gain based on the satellite geometry.

6.2.2 Ground Station Antenna
Two types of antennas will be used on the ground station system. A high gain (38.2 dB) 16 ft. parabolic antenna is used to receive the S band link, and a helix antenna is used for transmitting in the UHF band. THERMION-I, because of its low power communication system, requires a high gain receiver and high output transmitter at the ground station. Thus the two antenna configuration was chosen so that the high output of the transmitter would not desensitize the high gain receiver. The UHF helix antenna will be mounted on the end of the parabolic antenna stem. The parabolic/helix antenna combination is mounted on a tracking pedestal which will be used to track the satellite as it passes over the ground station. A PC type computer is used as the controller for the tracker. The tracking antenna system and controller software is provided by Aydin Corporation(1). Initial observations of the proposed satellite orbit indicate that a minimum of two passes over the ground station will occur daily. The down loading of stored data and the transmitting of various commands from the ground station will occur at this time.

6.2.3 Spacecraft Transmitter and Receiver
The on board transmitter is comprised of a PCM encoder (Aydin model MMP-900) and the 2.2 GHz transmitter (Aydin model T-105SE). The encoder takes the data acquisition output data and puts the data into frames and adds frame sync code for standard IRIG transmission through the 5 watt PSK transmitter. The command receiver (model RCC-101) operates in the 400-500 MHz range. It has eight command codes which can be used to instruct the on-board computer.

6.2.4 Off-the-Shelf Technology
The communication package is available from Aydin Corporation and is comprised of proven satellite communication components.

6.3 THERMION-I DATA LINK

The following sections describe the communication link budget calculations and data transmission scheme.

6.3.1 RF Downlink
The transfer of information from one point in space to another using electromagnetic wave propagation requires a transfer of energy. Each operation in

the data link contributes a gain or a loss to the energy transfer. These operations can be expressed as an equation that will detail the data link budget (2).

In quantifying the link budget, the following equation was used that yields a margin of safety or signal margin (M):

$$M = \frac{P_t G_t G_r}{(E_b/N_o)\,kTBL_s L_a L_o}$$

This equation is often expressed in Db:

$$\begin{aligned} M = {} & P_t(dBW) + G_t(dB) + G_r - (E_b/N_o)_{req}(dB) \\ & - B(dBb/s) + 228.6\,dB - T(dB^{\circ}\,K) \\ & - L_s(dB) - L_a(dB) - L_o(dB) \end{aligned}$$

where,

P_t = power of transmitter
G_t = gain of transmitter
G_r = gain of receiver
$(E_b/N_o)_{req}$ = required bit error rate
B = bandwidth
k = Boltzmann's constant (1.38×10^{-23})
T = receiving noise temperature (degrees kelvin)
L_s = free space loss
L_a = atmospheric attenuation of the wave
L_o = non-ideal system components loss

Using the transmitter and receiver stated above and phase shift keying (PSK), the signal margin for the THERMION-I data link is 29 dB at 2.2 GHz. Refer to appendix F for actual calculations.

6.3.2 Transmission

At the beginning of a transmission, a training sequence is necessary to verify that the data link has been established. During this sequence the identity of the individual satellite is relayed.

Precise transmission of digital information via an RF link requires the consideration of the probability of bit error. At frequencies in the S-band region, a bit error rate of 12.5 dB is necessary for the probability of bit error (P_e) to be 10^{-4}. This P_e is sufficient if the data is validated, using handshaking, every 16 Kbits. Handshaking is accomplished using parity and/or overflow checking. The following is a breakdown of total transmission time.

Transmission of 8 MBytes Stored Data

Bandwidth = 1 MHz
Delay = 8 m sec each way
Training Sequence = 10 sec
Handshaking every 16 K bits
(Total 500 handshakes)
Check Time = 8 sec
(500 x 2 x 8 msec)

Transmission Time

Training sequence =	10 sec
Data transfer =	64 sec
Handshaking =	8 sec
Total =	82 sec

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7.0 POWER SYSTEM

7.1 REQUIREMENTS

The power system must supply 6.6 W orbit average power to the satellite. Initially, it must supply power to all systems while the satellite is orienting itself towards the Sun. It is desired to use inexpensive, commercially available parts when appropriate.

Subsystem	***Power (watts)***	***Duty Cycle (percent)***
Computer	1.0	Continuous
Communications	50.0	0.2
Attitude Control		
Sun Sensors	0.0	Continuous
Horizon Sensor	4.3	Continuous
Magnetometer	1.0	10.0
Torque Rods	5.25	10.0
Heater (optional)	3.0	20.0

Table 7-1 Power Budget

The total orbit average power required is 6.77 W. The orbit average power available is 10.0 W. This leaves a large power margin.

7.2 DESCRIPTION

As shown in figure 7-1 (see page 7 - 2), the power system is composed of three basic subdivisions.

The satellite is powered up once it receives an enable signal from the launch vehicle. The computer then completes a checkout of systems and spins up the horizon sensor/momentum wheel assembly. The computer is also simultaneously signaling the launch vehicle with a constant frequency signal. The computer then powers the bolt cutters, freeing THERMION-I from the launch vehicle. At separation, the Delta II has lost signal, and the Delta II may in turn signal to ground that THERMION-I has separated. After ten minutes, the attitude control system is enabled and commanded to point the satellite towards the Sun. Once pointed at the Sun, the computer will control charge from the solar cells to the batteries and measure Thermionic Heat Pipe Module (THPM) performance.

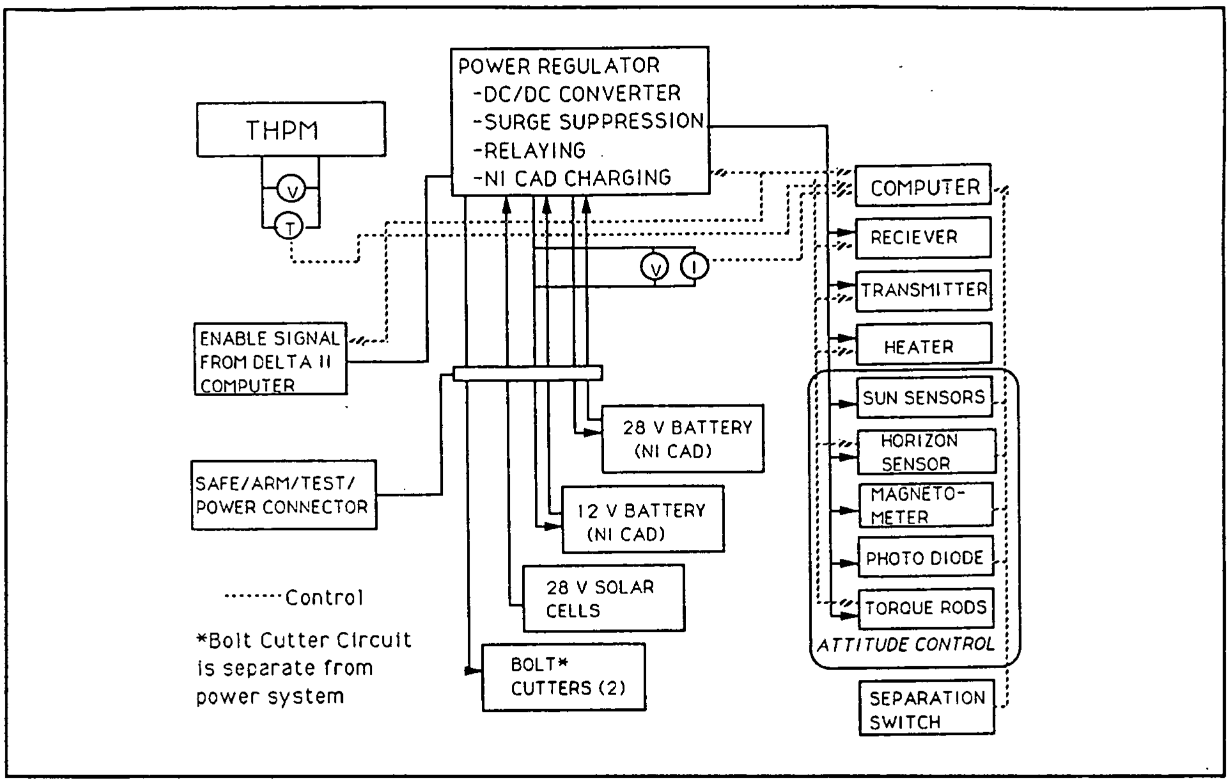

Figure 7-1 Power Block Diagram

An energy balance model of the power system was created to facilitate the design of the power system. Variables such as battery capacity, packing efficiency, solar cell efficiency, and Sun angle were among the 20 variables in the model. The model was made conservative by operating THERMION-I in a combination of large Sun angle and end of life conditions. Nominal condition calculation using the model is included in Appendix G.

The power regulator board will most likely have to be designed and fabricated custom for THERMION-I.

7.3 POWER STORAGE

THERMION-I will use a battery of 36 D-size Nickel-Cadmium cells. A battery is defined as a group of cells. Thirty six cells will make up the battery (the power pack). This high number is required to power the subsystems for the interim between separation from Delta II and final lock of the attitude control on the Sun. The cells are broken up into three groups:

Group 1: 10 cells, 12 V
Group 2: 20 cells, 24 V
Group 3: 6 cells, 7.2 V

The first two groups provide a 12 V and 24 V bus, respectively. These voltages are typical for the subsystems. The third group is a swing shift group to be relayed in when a sub group of five cells has to be taken off-line for reconditioning or if a cell in that subgroup has failed. Group 2 is actually two group one's connected in series. So the whole battery of 36 cells may also be viewed as 7 groups of 5 cells at 6 V each and one cell at 1.2 V. This approach allows the battery to provide 24 V and 12 V even if 3 groups of 7 have in some way failed.

Sanyo HIGH CAPACITY CADNICA batteries (E-Series) 'D' cells are selected as the power storage. They are commercial nickel-cadmium cells and are rechargeable. The critical limitation on all Ni-Cad cell chemistries is at temperatures below 0° C at which they will not accept charging. When in the normal temperature range (0-45° C), the computer (by reading voltage and current sensors on the groups of batteries) will switch on charging. Potentiometers also controlled by the computer will limit current rates to avoid overcharging.

A conditioning procedure for the Ni-Cads must be completed prior to flight. This is necessary to match all the individual cells in terms of charging rates and capacity. Further all the cells must undergo a thermal vacuum and vibration cycling to weed out any susceptible cells. This procedure will assure a reliable battery of cells. D-size cells are selected because they are the most widely manufactured and hence will have the least variation in performance from cell to cell. They provide for a high energy density when packaging is taken into consideration.

7.4 SOLAR CELLS

Silicon based solar cells are selected because they are comparatively inexpensive, the allowable surface area for the solar cells is more than required, and the 11.5% efficiency easily satisfies requirements. Solar cells are laminated to the top surface of the main bus and also along the large section connecting the bus to the mirror (the"Arm"). Because THERMION-I is always pointing at the Sun only the exposed surfaces are covered. The required total area of solar cells is 0.178 m^2 but 0.232 m^2 will be covered with solar cells. This adds little cost to the satellite but adds significant power margin to the design. When in the Sun the solar cells will generate in excess of 30 W with 12.8 W going to battery charge and 6.4 W devoted to normal satellite operation. Diodes are in series from the solar cells to the batteries to prevent reverse current.

Initially, THERMION-I was to be powered by the 105 W generated by the thermionic process of the payload however this energy comes at 0.7 V and 110 Amperes! The high current is not so much a problem as finding or designing a DC/DC converter that could step up the voltage to 12 or 24 V. Including resistances, it was determined that 0.7 V is below the threshold of any known DC/DC converter. Solar cells are much easier to implement than a custom designed DC/DC converter. Appendix G contains the power model used to size the solar cell area.

7.5 RESULTS

The power system will always provide more power than is required. There is a 3.4 W orbit average power margin built in, plus there are more solar cells and Ni-Cads than needed. This will add negligible cost in materials but insures high reliability and versatility. Cost is kept low by using commercial parts, and reliability is kept high by using redundancy in batteries and solar cells. Additional subsystems, if required in the future, will not require expansion or redesign of the power system.

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8.0 LAUNCH VEHICLE INTERFACE AND DEPLOYMENT

8.1 LAUNCH VEHICLE SELECTION

In selecting the launch vehicle for THERMION-I, the primary goals were to find the least expensive access into space while achieving an orbit which would yield a minimum one year mission, plus have enough payload volume to allow for the required solar collector area. Due to the relatively small size of THERMION-I and a limited budget, the focus of our selection was on secondary payload opportunities and small dedicated launch vehicles.

Most of the launch vehicles in use today offer a secondary payload opportunity. We researched four launch vehicles in determining the one best suited for our needs. The vehicles we researched included the General Dynamics Atlas, the Martin Marietta Titan II, the Orbital Sciences Pegasus, and the McDonnell Douglas Delta II (1-7).

The Atlas and Titan II both have secondary payload volumes which would have been very suitable for our needs. The major drawback on each of these launch vehicles is the orbit in which the secondary payloads are jettisoned. The Atlas offers two orbits for a secondary payload, one in Low-Earth Orbit, and the other in Geosynchronous Orbit (1). Geosynchronous altitude is far too high for our purposes and the Low-Earth orbit at 80 X 269 nmi, is not at a sufficient altitude to sustain a one year mission life. Similarly, the Titan II ejects the secondary spacecraft in a ballistic trajectory, and therefore, both vehicles would require the use of a secondary booster (3). The added expense and complexity of a secondary motor convinced us to seek an alternate solution to the orbit problem.

The Pegasus launch vehicle is small enough that we considered it as a dedicated launch vehicle for THERMION-I. Using Pegasus would allow us to design a much larger and heavier satellite. The Pegasus vehicle would also allow us to place THERMION-I in a sun-synchronous (4,5), and thereby eliminate the frequent thermal cycling of the payload. The critical drawback for using Pegasus is the approximately 8

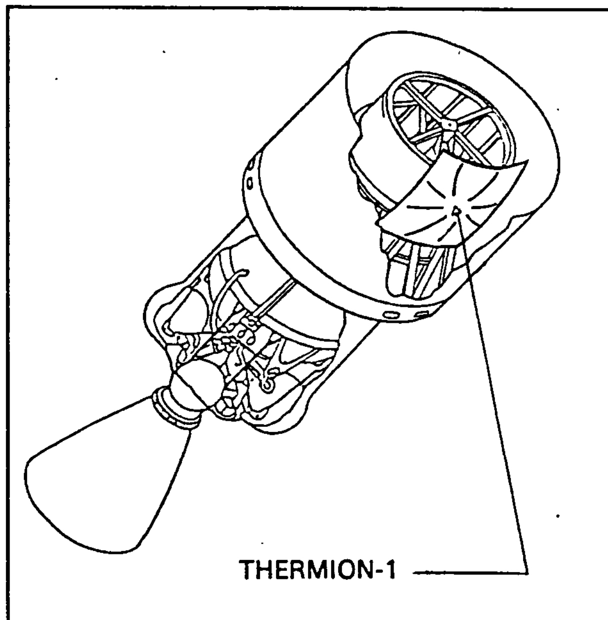

Figure 8-1 Artist's conception of the THERMION-I satellite mounted to the 2nd stage.

million dollar fee for launch (8). However, enough payload volume exists on the Pegasus that we could launch as a companion payload and share the launch costs with another mission. If a suitable companion to sun-synchronous orbit were found, Pegasus would be reconsidered.

The Delta II offers the best compromise of secondary payload volume, orbit, and launch costs for the THERMION-I project. The Delta II is the dedicated launch vehicle of the U. S. Air Force Global Positioning System (GPS) satellite. Because this experiment would be funded by the Air Force, deployment via the GPS mission would reduce overall launch costs. Furthermore, this mission offers excess fuel on the second stage which will eliminate the need for a secondary booster (9).

The secondary payload envelope for the Delta II is an annular region between the payload fairing, and the guidance section of the thrust cone of the second stage. Figure 8-1 shows an artists conception of the THERMION-1 spacecraft mounted to the second stage. While this envelope has a somewhat limited and complex geometry, it proved large enough to test a scaled length version of the heat pipe.

8.2 DELTA II LAUNCH CONFIGURATION

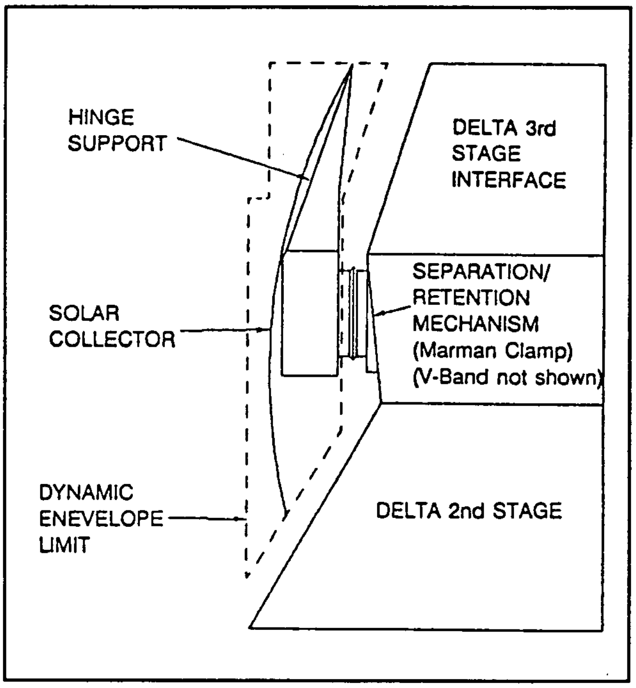

Figure 8-2 Side view of launch configuration.

The restrictive payload envelope presented a significant challenge in the design of the spacecraft and the overall launch configuration. In order to optimize this configuration the payload envelope was reproduced on a CAD (Computer Aided Design) system so that as the design progressed the dimensions could be constantly checked. Working in conjunction with Solar Collection and Payload the final dimensions of the collector were determined to be 58.6 X 41.6 inches (148.8 X 105.7 cm). Based on these dimensions and the radius of curvature (40 inches, 101.6 cm) the final launch configuration was chosen. The side and top views of this configuration are shown in figures 8-2 and 8-3 respectively.

8.2.1 Delta II Plume Induced Environments

In order to achieve the required area for the solar collector, it was necessary to eliminate the standard secondary payload plume shield as shown in figure 8-4. Possibly dangerous environments result from the spin rockets which fire less than 1 ft. above the top of the payload, and from the Star-48B 3rd stage which will fire approximately 42 ft. in front of the payload and 2nd stage. Approximate spatial relationships are shown in figure 8-5. The resulting environments are:

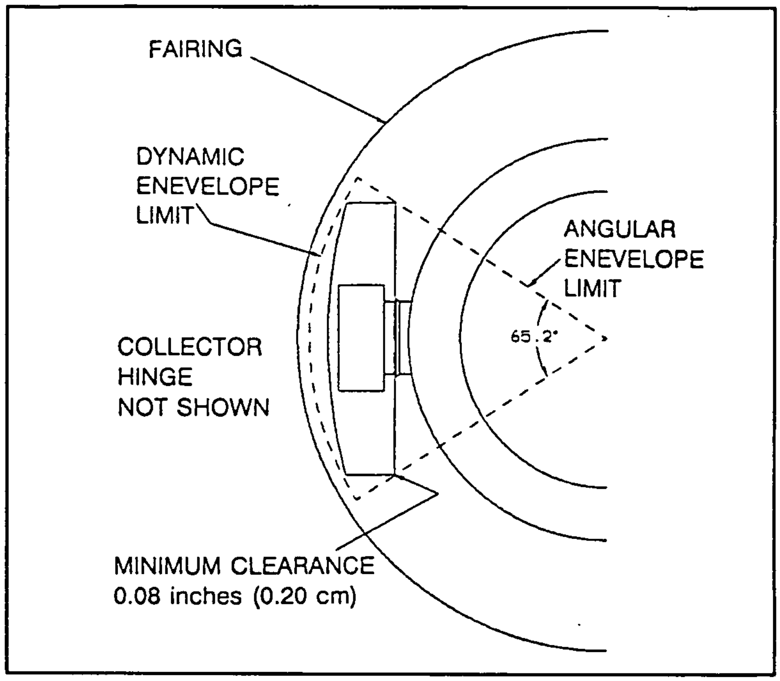

Figure 8-3 Top view of launch configuration.

- Aluminum Oxide particle impingement heating, and abrasion from the 3rd stage
- Convective heating from the 3rd stage and spin rockets
- Shock amplification heating
- Contamination from the 3rd stage and spin rockets

These concerns have been the source of extensive analysis for the SEDS (Small Expendable Deployer System) mission which is another secondary payload scheduled for a Delta II/GPS launch (10,11). While such an analysis has not yet been performed for THERMION-I, none of these environments are considered to pose any particular danger at this time (12).

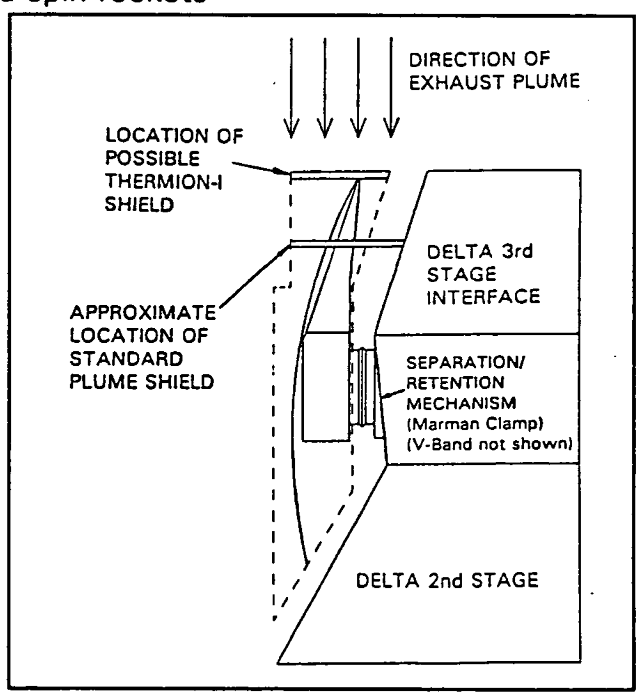

Figure 8-4 Possible plume shielding.

The critical surfaces for THERMION-I are the inside surface of the solar collector, sun sensors, horizon crossing sensor, and the solar arrays. With respect to impingement and abrasion from the third stage exhaust, these surfaces are protected by the huddled launch configuration. Note the

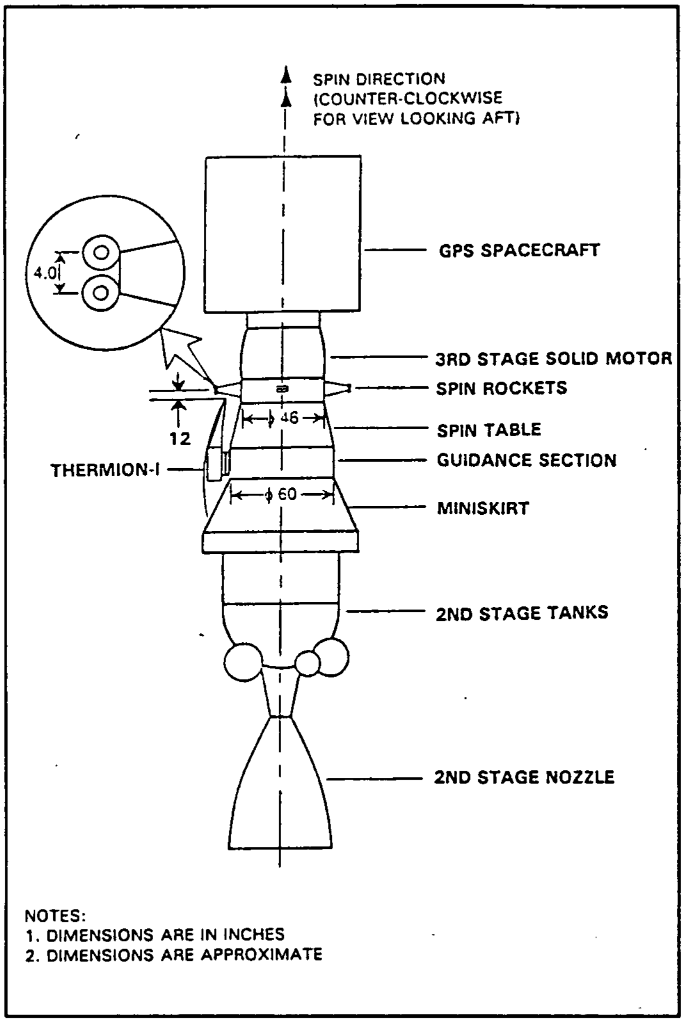

Figure 8-5 2nd/3rd stage launch configuration.

direction of exhaust plume in figure 8-4. While all these surfaces could be subject to contamination, normal condensation of the gas-phase species originating from either the third stage or the spin rockets occurs at cryogenic temperatures, and therefore only very low levels of deposition would be possible. In addition to being well above the condensation temperatures, the solar collector would be relatively insensitive to low levels of deposition, and such levels would be quickly eliminated by evaporation and atomic oxygen (12).

It should be noted that if after further analysis it was determined that the above environments posed a significant danger, the Delta II would not be eliminated. Because THERMION-I is well under the weight limits, protection could be provided by a shield mounted to the collector hinge which would stay with the spacecraft after deployment as shown in figure 8-4.

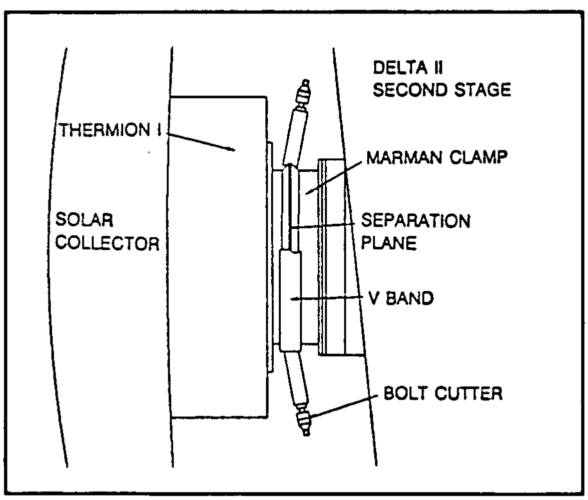

Figure 8-6 The Marman clamp.

8.2.2 Interface/Deployment

THERMION-I will be mounted to the guidance section of the Delta II second stage via a standard V-band Marman clamp. The Marman clamp, shown in figure 8-6, consist of one plate mounted to the base of THERMION-I, and the another plate to mounting rails attached to the second stage. The spring loaded V-band will clamp over a lip on each plate and hold THERMION-I to the launch vehicle.

When the Delta II second stage has circularized the parking orbit, two pyrotechnic bolts will be set off, releasing the spring loaded V-band. Four axially compressed springs, mounted internal to the Marman clamp and perpendicular to the bottom plate of THERMION-I, will then push the spacecraft away from the Delta II second stage, deploying the satellite into orbit.

Deployment spin must be minimized because of the time required for the attitude control system to correct. The asymmetric mounting of the satellite about the Marman clamp will not allow the force of the ejection springs to be directed through the center of mass. To avoid causing the satellite to spin or tumble on deployment, the four ejection springs can be shimmed during integration to give the proper ejection alignment. In addition, THERMION-I has no separation speed requirements, so separation springs of relatively low force can be used to further reduce deployment spin to a tolerable level (9).

8.3 LAUNCH PROFILE

A nominal mission profile for the launch and deployment of THERMION-I is shown in figure 8-7. The profile is based on the SEDS trajectory and should be very similar for THERMION-I. The second stage will achieve a parking orbit of approximately 100 X 375 nmi at 37° inclination (13). The primary payload and 3rd stage separate at 1264 sec.(9), leaving the 2nd stage and THERMION-I in the parking orbit for 1 more period. At the next apogee crossing, the 2nd stage will be refired circularizing the orbit at 375 nmi. This maneuver will expend approximately

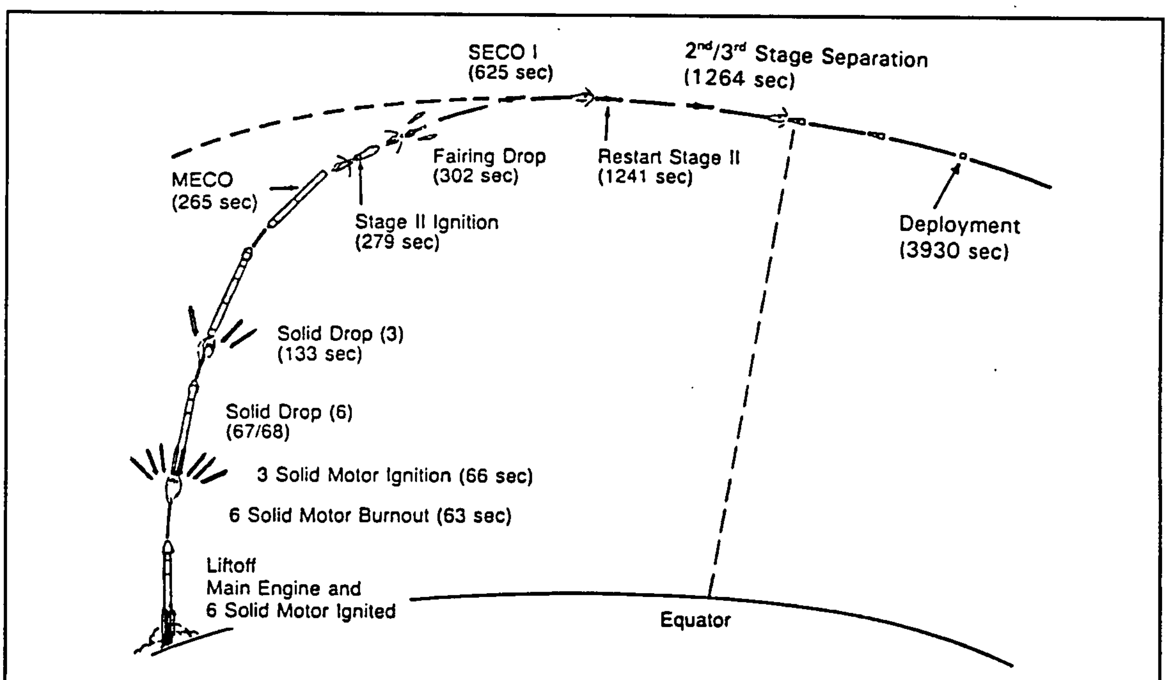

Figure 8-7 Nominal mission profile.

29 % of the excess 1600 lb. of fuel on the 2nd stage. Shortly after circularization, THERMION-I will be deployed at approximately 3930 seconds. After this, a depletion burn will be performed on the 2nd stage. In this process the orbit of the 2nd stage will be modified slightly to avoid any interference with the THERMION-I spacecraft (9).

8.4 CONCLUSIONS

While the main objectives of Launch Vehicle Interface and Deployment have been met, we have identified some particular areas in which future work will be necessary. Probably the next step will be an exhaustive analysis concerning the plume induced environments that were discussed in section 8.2.1. The analysis, performed by McDonnell Douglas, will determine whether or not additional plume shielding is required. Following this, a structural analysis will determine if the structure can withstand the predicted flight environments. And although THERMION-I is well within the Delta II secondary payload weight limit, a mass model analysis will most likely have to be performed due to the awkward launch configuration. Once the structural integrity and dynamic effect on the launch vehicle have been predicted, the standard series of launch qualification tests will be required. These tests are discussed in detail in section 10, Test and Evaluation.

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9.0 THERMAL MANAGEMENT

The THERMION-I small satellite is a thermally dynamic system. Although the concentration of solar energy provides an acceptable operating environment for the heat pipe payload, it also causes much higher component temperatures than small satellites generally have. Due to the complexity of the entire satellite, the systems thermal analysis was completed in two parts. The satellite bus was modeled with traditional thermal analysis techniques and the heat pipe payload configuration was modeled with a more general lumped capacitance method.

9.1 OBJECTIVES

Thermal Management primary objectives are:

- Verify that component operating temperatures are within specified limits.
- Develop possible thermal control techniques to maintain allowable temperature range.
- Coordinate systems thermal analysis.

9.2 THERMAL REQUIREMENTS

In general, each subsystem of THERMION-I specified maximum and minimum allowable component temperatures. The Payload subsystem considered energy input and operating temperature requirements in the design of the solar collection device and heat pipe payload configuration. The major components and temperature limits of each subsystem are listed below:

TEMPERATURE LIMITS (Degree K)

SUBSYSTEM	ALLOWABLE LIMITS
• PAYLOAD	1925 ± 25
• STRUCTURE	< 900
• POWER DISTRIBUTION	
• Power conditioning	250 to 330
• Batteries	270 to 320
• Solar cells	300 to 333
• ATTITUDE CONTROL	
• Sun sensor	250 to 330
• Magnetometers	255 to 350
• Torque rods	218 to 344
• Momentum wheel	250 to 330
• COMMUNICATIONS	250 to 330
• DATA MANAGEMENT	233 to 358

9.3 ANALYSIS EVOLUTION

Initially, radiation heat transfer principles were examined to determine the heat flux inputs into the satellite system. In the space environment, THERMION-I is exposed to inputs from the sun, the earth, space and internal heat generation. Q_{sun} is the average direct solar flux and was assumed to be 1380 W/m. $Q_{reflected}$ is the amount of solar flux that is reflected off the earth onto the satellite. Q_{earth} is the radiation exchange with earth, which is assumed to be a black-body object. Q_{space} is the energy radiated to space. $Q_{internal}$ is the internal heat generation by components inside the satellite. The following equation defines the energy balance of THERMION-I.

$$\dot{Q}_{SUN}+\dot{Q}_{REFLECTED}+\dot{Q}_{EARTH}+\dot{Q}_{INTERNAL}=\dot{Q}_{SPACE}$$

where:

$$\dot{Q}_{SUN}=\alpha_s A_{\perp} I_s$$
$$\dot{Q}_{REFLECTED}=a\alpha_s F_{s,se} A_s I_s$$
$$\dot{Q}_{EARTH}=\sigma\epsilon_s\epsilon_e F_{s,e} A_s (T_s^4-T_e^4)$$
$$\dot{Q}_{SPACE}=\sigma\epsilon_s F_{s,sp} A_s (T_s^4-T_{sp}^4)$$

where:

$A_{\perp}$ = the orbit-average spacecraft projected area perpendicular to the sun.
A_s = the surface area of the satellite.
a = the albedo of the earth.
α_s = the absorptivity of the satellite.
ϵ_s = the emissivity of the satellite.
ϵ_e = the emissivity of the earth (for a black-body $\epsilon = 1.0$).
$F_{s,se}$ = the radiation exchange view factor of spacecraft to sunlit earth.
$F_{s,e}$ = the view factor of spacecraft to earth.
$F_{s,sp}$ = the view factor of spacecraft to space ($F_{s,e} + F_{s,sp} = 1$).
I_s = the solar flux at earth.
σ = the Stefan-Boltzman constant.

9.3.1 SYSTEMS ANALYSIS

A complete systems analysis was attempted but not preformed due to the complicated heat pipe analysis and the complex radiation exchange between the heat pipe payload and the satellite bus. The systems thermal analysis was completed in two parts. First, The bus was modeled using the Systems Improved Numerical Differencing Analyzer (SINDA), developed by Network Analysis Associates, Inc (2). SINDA is a software system designed for general analysis of lumped capacitance type thermal models. Then, the heat pipe payload configuration was analyzed with a lumped capacitance method to predict the total energy balance for the heat pipe cavity.

9.4 SINDA MODELING OF SATELLITE BUS

A conductor-capacitor network representation of the THERMION-I bus was defined and integrated into SINDA input files. The following basic assumptions were made in modeling THERMION-I.

- Aluminum-Aluminum contact resistance between the structure and all electronic components.
- Unity radiation exchange view factors between the outer structure and space.
- 50 W conducted through cavity insulation (worst case).
- Appropriate instrumentation heat generation.

Also, the following were neglected.

- Radiation between electronic components inside the bus.
- Radiation between satellite bus and heat pipe payload.

Major sections of the structure were divided into parts and modeled as lumped mass thermal nodes. Electronic components were also modeled as lumped mass nodes. The thermal nodes were numbered as shown in figure 9-1 and 9-2.

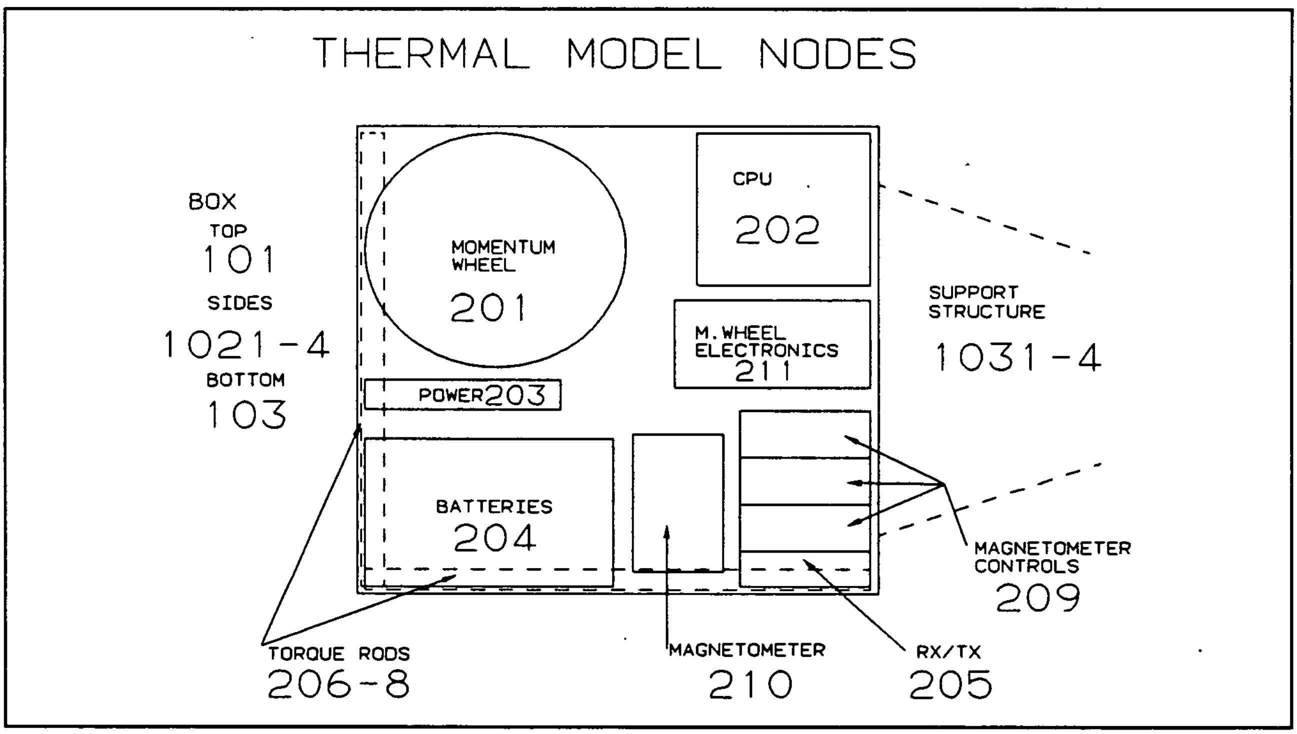

Figure 9-1. Node numbers of electronic components and structure.

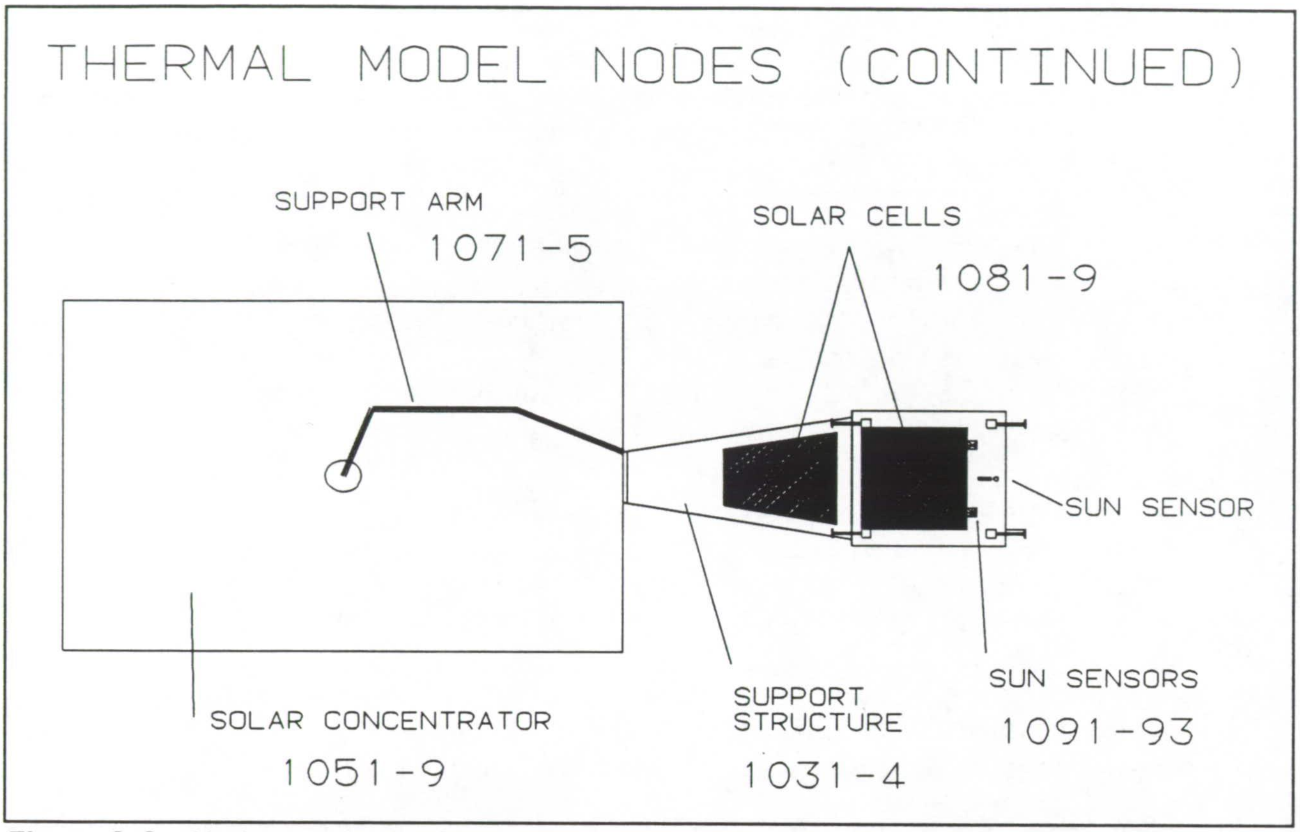

Figure 9-2. Node numbers of external structure and components.

9.4.1 SINDA models

SINDA has the capability of both steady state and transient type analysis. The satellite bus analysis required both. A steady state cold case was developed and analyzed. The cold case development provided experience using SINDA. However, the results were not considered in depth since the cold case steady state would never occur in the life of THERMION-I.

A steady state analysis was developed to determine the maximum expected bus temperatures. Sun side heat output values were determined for the electronic components inside the bus. The radiation heat inputs were also determined using the radiation fundamentals presented above. The sun side case was considered worst case and steady state was approximated as sun synchronous. Appendix I shows the input file for the sun side case.

In order the predict the minimum expected bus temperatures, a transient cold case was developed. The component temperatures were started at the steady state sun side values. The transient was run for 30 minutes which is the expected orbit time out of the sun. Appendix I shows the input file for the cold case transient.

9.4.2 SINDA results

The sun side steady state and the cold case transient results were examined and the expected temperatures are shown below.

	Expected Temperatures (Degrees K)	
	Maximum	*Minimum*
Top	252	251
Sides	244	238
	245	241
	243	239
	246	237
Bottom	278	270
Momentum Wheel	280	278
M.W. Electronics	279	271
CPU	278	278
Power Conditioner	278	272
Batteries	278	270
TX/RX	280	278
Torque Rods Internal	272	255
	243	242
External	228	227
Magnetometers	282	281
Solar Cells	252	251
	250	231
Support Arm	376	376
Support Structure	250	230
Mirror	124	123
Hinge	214	207

9.4.3 SINDA modeling conclusions

The expected temperatures are within the operating range for all of the components except the solar cells and sun sensors. The expected battery temperatures are close to the minimum allowable temperature and some type of thermal control should be used since batteries are a critical component.

With the addition of radiation from the heat pipe payload, it is expected that the solar cell temperatures will increase to the operating range. The radiation from the heat pipe payload was neglected due to the small exchange factor (~ 0.002) and the dependance of the solar cell absorptivity on wavelength. However, the heat pipe payload is operating at high temperatures (from 1000 K to 2000 K) and the radiation exchange between the two may not be negligible. More analysis needs to be completed to verify this assumption in modeling.

The expected temperatures for the mirror and hinge are unreasonably cold and indicate inaccuracy in the SINDA modeling. A more refined nodal network will predict more accurate values. The low structural temperatures are not a problem since only a upper temperature limit was specified. The influence of temperature on mirror distortion and efficiency should be analyzed in more depth.

9.5 HEAT PIPE PAYLOAD ANALYSIS

A lumped capacitance model was developed to estimate the transient response of the heat pipe payload. The model analyzes a single node using basic radiation principles. Figure 9-3 shows the transient response of the configuration. The predicted time to heat the cavity to operating conditions (~ 5 to 7 minutes) is acceptable for a 60 minute sun side orbit. The model also estimates the minimum expected temperature of the cavity for the 30 minute cold side transient. A temperature of 700 K is allowable since it is above the critical thermal cycling temperature of 250 K for tungsten. The FORTRAN source code for the model can be found in Appendix I.

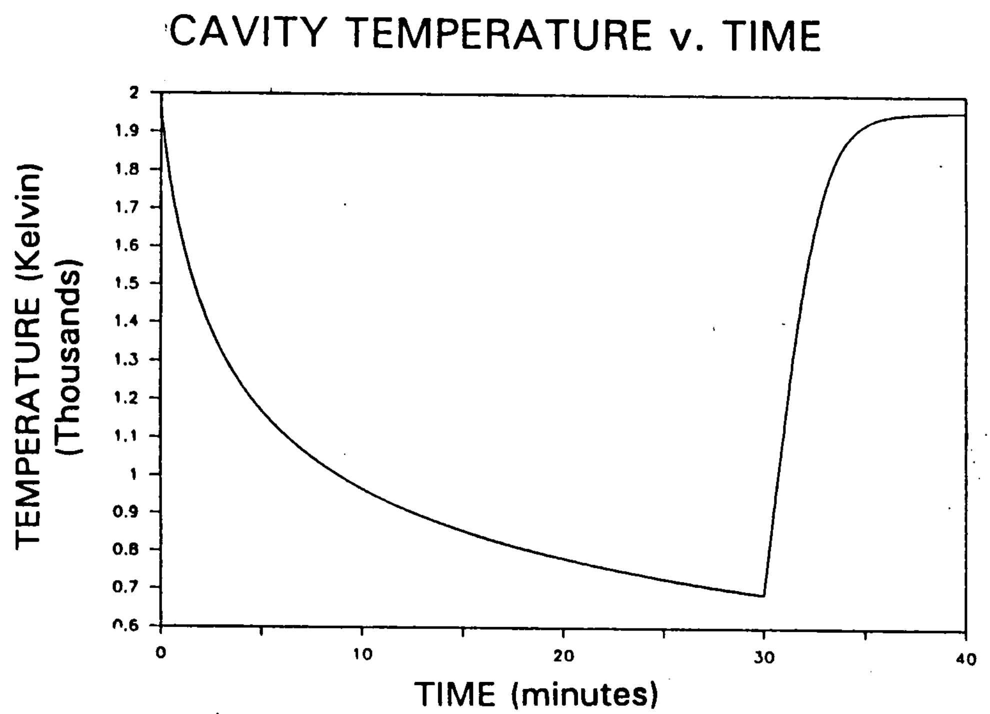

Figure 3. Transient response of cavity temperature.

9.6 THERMAL CONTROL

Since the predicted temperatures of all components were within allowable limits, the thermal control system will be designed according to traditional passive and active thermal control techniques. Coating selection is a passive method of thermal control which will help maintain acceptable operating temperatures by varying the emissive and absorptive properties of each surface. The internal and external surfaces of THERMION-I will be coated after thermal testing is complete.

Heaters are a form of active thermal control. The three watt power budget allows for three one-Watt heaters attached to such components as the batteries to maintain the required operating range. Each heater will be equipped with self-contained control switch. Both methods of thermal control will be used to guarantee allowable temperature limits.

9.7 THERMAL CONCLUSIONS

The sun side steady state case results show that for the given emissivity and absorptivity values all components will remain below the maximum temperature limit. The cold case transient results show most components above the minimum operating limit. The sun sensors and the solar cells are below the temperature limits but are not in operation during the cold case. The batteries are close to the minimum temperature and thermal control will be required. THERMION-I will operate within acceptable limits with proper thermal control.

9.8 FUTURE CONCERNS

Radiation between components and radiation between the heat pipe payload and the satellite bus have a large impact on the complexity of modeling and on the accuracy of the model results. Therefore, the a more complex model is needed to include these factors. The NEVADA and RENO software developed by Network Analysis Associates, Inc. (2) could be used to provide a more comprehensive analysis of radiation exchange factors.

The issue of thermal cycling and initial thawing of the liquid metal heat pipe need to be considered in more depth. Also, the thermal dependance of the solar collection mirror requires more analysis.

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2. SINDA, 1987/ANSI. Network Analysis Associates, Inc. P.O. Box 8007, Fountain Valley, CA 92708, TEL: (714)557-2080.

10.0 TEST AND EVALUATION

10.1 REQUIREMENTS

The Test and Evaluation (T&E) functional requirements are to:

1. Provide validation of the THERMION-I subsystem designs.
2. Provide validation of survivability due to launch parameters.
3. Provide verification of total system function and integration.

10.2 TEST ENVIRONMENTS

In order to meet the functional requirements of test and evaluation the testing programs were broken into three sections. The three sections are Subsystems, launch, and In-orbit tests. Each section describes the general testing requirements needed to be addressed in order to assure proper operation of the THERMION-I satellite.

10.2.1 Subsystem Design Validation

Various new design ideas have been proposed by some of the subsystem groups in order to meet the heat pipe validation requirements. These designs need to be proven as viable before full production of THERMION-I begins. Some of these ideas for which preliminary testing is needed include: the heat output from the collection mirror, mirror deployment system, proper radiation from the heat pipe radiator, and the measurement scheme used to determine voltage and current output of the thermionic conversion. These are the components that if found to be non-functional would require major modifications to the satellite design. This list may change as more detailed design is completed.

10.2.2 Launch

It is anticipated that THERMION-I will be launched as a secondary payload on a Delta II rocket. Therefore, rigorous testing will be required to insure the safety of the primary payload. The actual required testing parameters will need to be obtained from the Delta II launch management for the specific launch.

The launch environment is the most harsh condition that the satellite will be exposed to. The satellite will need to be validated for exposure to vibration, acceleration, acoustic energy, shock, and pressure change from ambient to vacuum. These tests will be performed to a safety factor level, specified by the launch vehicle company, above the actual expected level the satellite will experience during launch. It will need to be determined whether a test satellite will be used in possible destructive tests or risk the actual satellite. Cost will most likely be the determining factor in this decision. The launch environment tests require special facilities, i.e. acoustic chamber, vacuum chamber, etc.. Arrangements to use such facilities need to be made well in advance of the expected test dates.

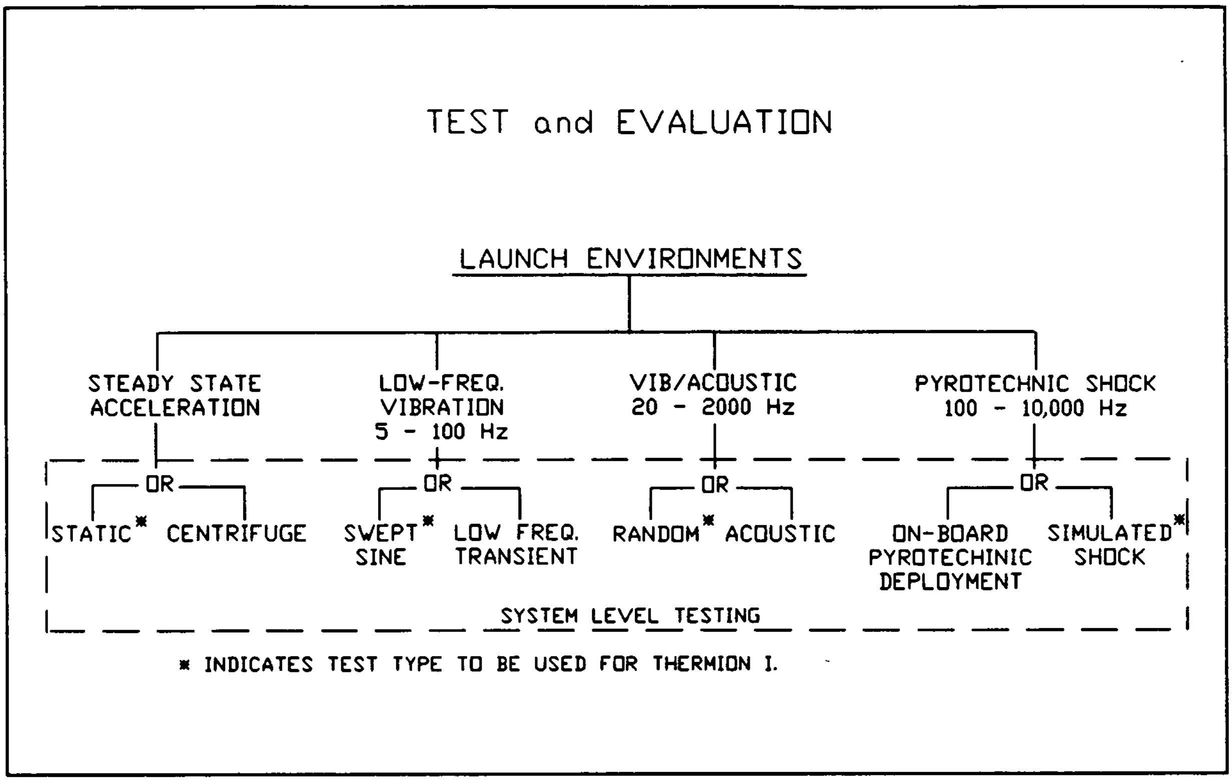

Figure 10-1 Typical launch environment tests.

10.2.3 In-orbit System Checks

Once the satellite has successfully achieved its proper orbit, several on-board tests will be performed to ensure that the data received is real data. Several system checks will be executed. The checks will determine proper release from the Delta II second stage, proper deployment of the solar collector and heat pipe assembly, and functionality of the satellite systems. These checks will be defined more specifically as the production of the spacecraft progresses.

10.3 THERMION-I TESTS

It should be noted that all components to be flown on board THERMION-I will be flight qualified by the specific manufacturer or will be qualified through proper means. A list containing all the now anticipated tests can be found in appendix J, This list is considered to be fluid, in that it will be modified as required by the customer, designers, and/or Delta II launch services.

11.0 THERMION-II

11.1 INTRODUCTION

Halfway through the design of the satellite THERMION-I, it was realized that the Thermionic Heat Pipe Module (THPM) need only be tested in a space environment. A free-flying satellite is just one of several solutions. Flight on NASA's KC-135 parabolic test platform, or on a sounding rocket such as the STARFIRE or JOUST series were considered. They provide 30 seconds, 5 minutes, and 15 minutes of microgravity respectively. We finally settled on flying, as THERMION-I does, on the Delta II launch vehicle. The Delta II can provide months of microgravity and is capable of carrying up to 340 kg as a secondary payload. Unlike THERMION-I, THERMION-II will merely be an appendage to the Delta II; it will not separate. Instead of using a solar collector, THERMION-II will use 253 kg of alkaline cells to power an electric tungsten grid heater on the THPM. Flying on the Delta II gives the best combination of low cost and performance. The experiment can run 21 hours continuously using the alkaline cells or 63 hours using lithium cells. THERMION-II will complete an identical test of the THPM as THERMION-I will but for a much shorter time. THERMION-II's THPM is 65% larger than the THPM presently proposed on THERMION-I and much closer to the full scale THPM proposed for the Small Excore Heat-Pipe-Thermionic Space Nuclear Power System (SEHPTR).

The U.S. Air Force and Idaho National Engineering Laboratory (INEL) would like to conduct a test of THPM because it is a critical section of SEPHTR. SEHPTR is a space nuclear reactor nominally capable of generating 40 kW of power. Due to its extremely toxic nature and high cost, it is desirable to conduct realistic tests of critical new components. THERMION-I and THERMION-II will simulate the reactor heat load on the THPM using a solar concentrator and electric heater respectively.

See Appendix K for more detail on the THPM and SEHPTR.

11.1.1 Description

THERMION-II is composed of five elements: four battery pallets of 480 alkaline cells each and one payload module. They are fixed to the secondary payload volume of the Delta II launch vehicle. Two pallets plus the payload module are on one side, while the remaining two battery pallets are 180 degrees around on the other side of the Delta II second stage.

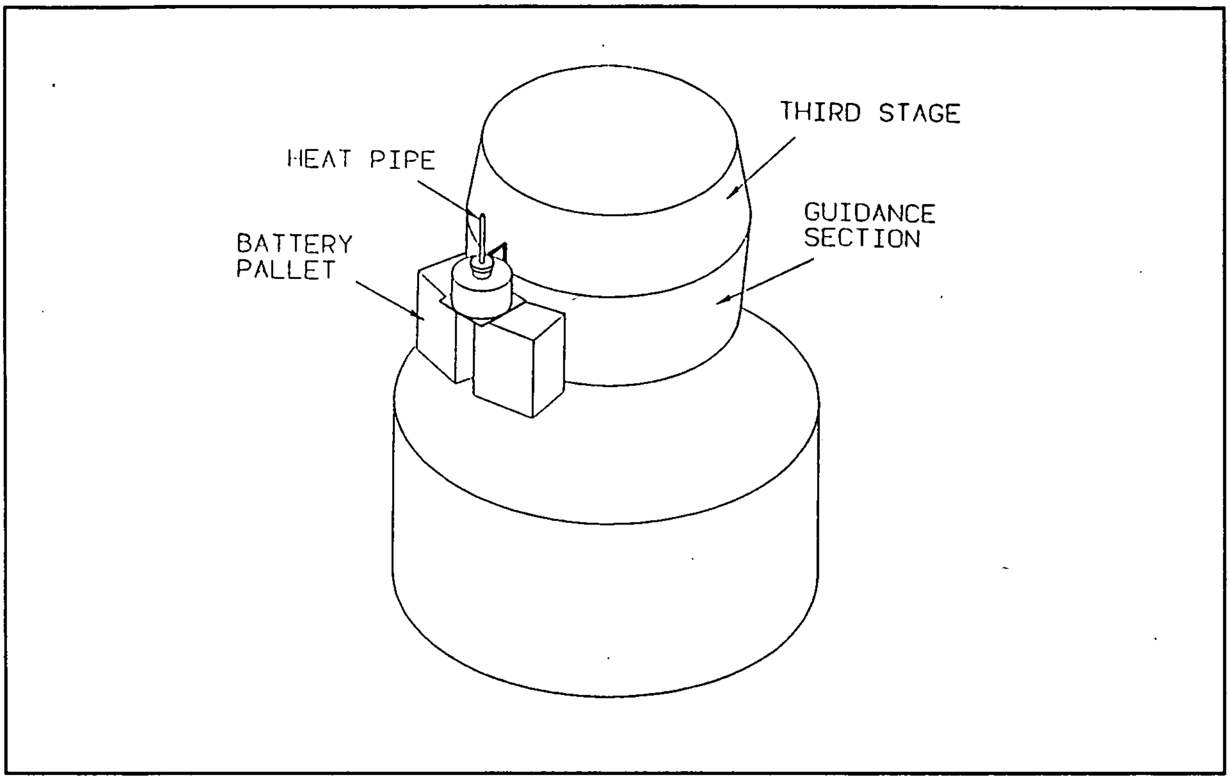

Figure 11-1 THERMION-II in Delta II secondary volume.

11.1.2 Payload Description

The THPM is actually two heat pipes in one. The outer one is shorter. It is called the emitter. The inner and longer heat pipe is called the radiator. It radiates the heat from the nuclear reactor core. The emitter converts 11% of the heat transferred through it into electric power. The other 89% of the heat energy is carried away by the radiator heat pipe. Nominally, the heat input surface of the emitter is at 1925 K The radiator's surface is at 1000 K. So the THPM has two functions:

1) Generate electricity and
2) Transport heat away from the reactor core.

Figure 11-2, which is on page 11 - 3, shows the SEPHTR reactor design.

11.1.3 Nominal Payload for THERMION-II

THERMION-II can test any scale THPM, but as the heat pipe gets larger, the required input power gets larger. The time of the experiment is reduced because of the finite amount of energy available from the batteries. As a baseline, THERMION-II will test a 1/4 length heat pipe. This is equivalent to a heat pipe with a 10 cm emitter length and an approximate radiator length of 70

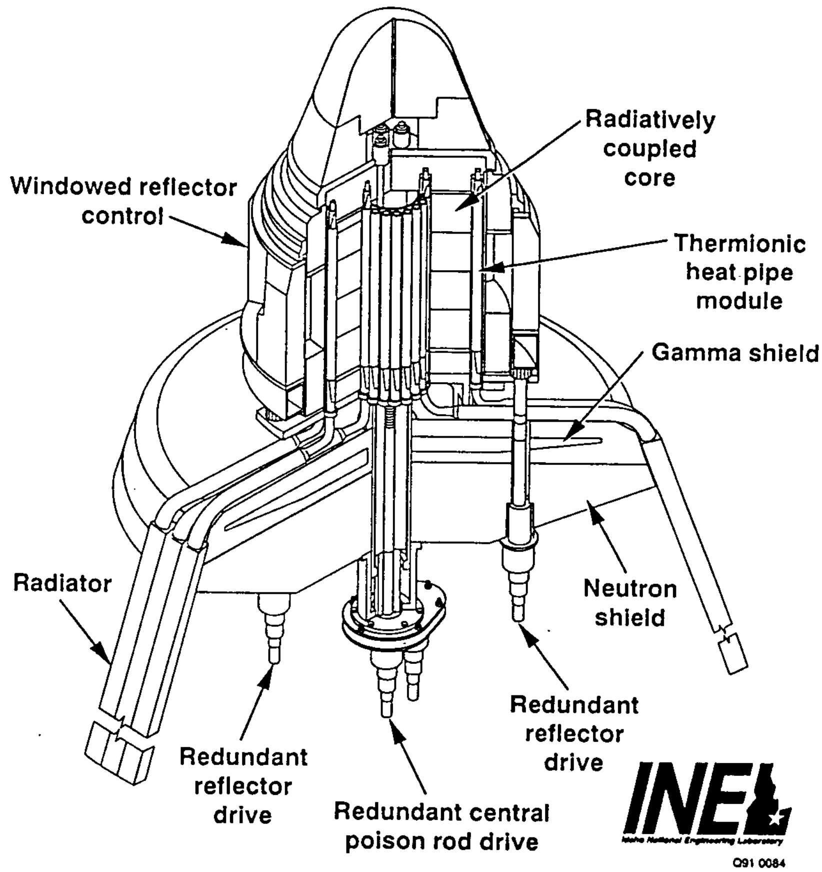

Figure 11-2 The THPM inside the SEPHTR reactor.

cm. The final dimensions on the test THPM are not in because the scale of heat pipe has not been decided on, and the final performance of THPM has not yet been determined. However, Utah State University has been working closely with Thermacore, Inc., the likely manufacturer of the THPM, and we are in the ball park.

11.2 SYSTEM LEVEL REQUIREMENTS OF THERMION-II

The requirements of THERMION-II are to simulate the thermal environment of SEHPTR's nuclear reactor core on a scaled version of THPM on orbit, measure the performance, and telemeter data to the ground.

- Provide a structure to fly in the Delta II secondary payload volume that supports the required batteries and THPM.
- Design a low cost alternative to THERMION-I.
- Provide a safe and reliable design.

11.3 DESIGN PHILOSOPHY

The design philosophy of THERMION-II is that the simplest path between any two points is a straight line.

- Takes a simple straight-forward approach to the THPM space test. Avoids unproven systems and items of questionable safety.
- Offer a low cost option. If it is simple to design and understand, it will be easy and inexpensive to implement. For example, a nail is simpler to conceive, detail and build than a screw.

11.4 MISSION DESCRIPTION

The THPM test requires a microgravity environment for some length of time. The THPM should be exposed to an environment that closely matches that of its intended use, inside a reactor core. To achieve this, the THPM payload will be mounted to the Delta II secondary payload volume along with batteries, power control system computer, and interface to the Delta II telemetry system. Once the orbit has been achieved, the primary payload will separate from the Delta II and the Delta II will zero the rotation rates of the second stage and activate the experiment. From this time on, the Delta II will only provide telemetry capability. No active attitude control system will remain, and some power will be diverted from the batteries to power the Delta II telemetry system. Detailed integration into the Delta II has not been seriously addressed, and it is anticipated the use of its telemetry system will be involved.

Using a 1/4 length THPM, the experiment may be operated continuously for 21 hours or longer if variable heating rates are applied to THPM. For example, the high amount of control of an electric based heater allows for very slow and very fast heating of the THPM.

11.5 CONCLUSIONS

THERMION-II is not a satellite and, consequently, is free of the cost, complexity, and development time that satellite is heir to. Cost estimates are provided in Appendix K. THERMION-II capitalizes on existing systems on board the Delta II (Attitude control and Telemetry) and does not depend on external power sources. Because it has no moving parts, there is no question of mechanical reliability. THERMION-II can also be developed and flown in much less time than any satellite.

THERMION-II is versatile since it has excellent control over heating rates of the electric heater. This is a closed loop system. Any transients can be simulated and excess power input may also be simulated. While THERMION-I may run for a year, it has the draw back that it cycles the THPM every orbit due to the loss of the Sun when it is in shadow.

THERMION-II should be pursued if a 21 hour test or a series of 21 hour tests (several THERMION-II's) will satisfy the reliability and functional concerns of the Thermionic Heat Pipe Module in a space environment. Finally, THERMION-II can be optimized using lithium cells instead of alkaline cells. This will triple the experiment time. Lithium cells have had an explosive history. If the cells are not internally fused they may overheat and explode when shorted. Though modern lithium cells are much safer, their legacy remains. The cost of qualifying them for flight may exceed their benefit.

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12.0 POWER SYSTEM

12.1 REQUIREMENTS

The power system must nominally supply 1800 W to the electric heater continuously for 21 hours using the energy stored in batteries. It must also power the telemetry systems on the Delta II. The power system must be able to vary power input to tungsten heater grid from 0 to 7000 W.

Subsystem	*Power (watts)*	*Duty Cycle (percent)*
Computer	1.0	Continuous
Communications (Delta II)	50.0	0.2
Tungsten Grid Heater	1800.0	Continuous

Table 12-1 Nominal Power Budget

The total orbit average power required will be 1851 W.

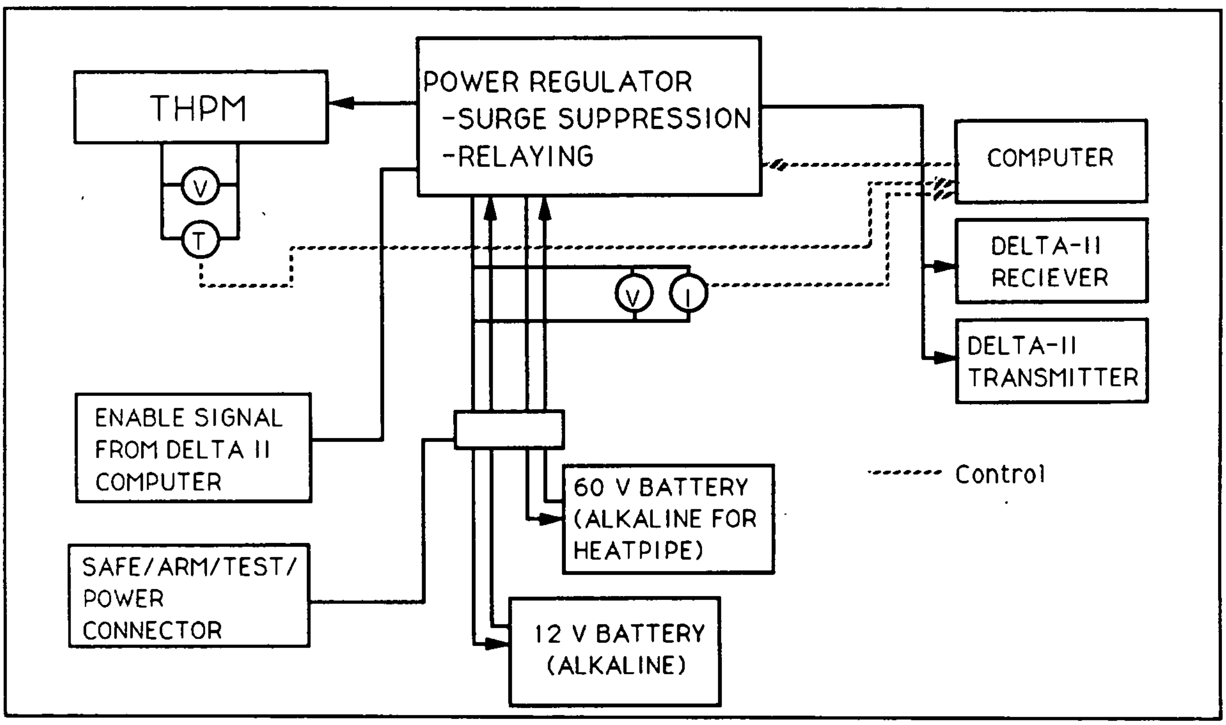

Figure 12-1 Power Block Diagram

12.2 DESCRIPTION

Once the primary payload has separated from the Delta II, the Delta II zeros the rotation rates of the second stage and gives THERMION-II an enable command. At this point the computer in THERMION-II will begin testing of the THPM according to programmed heating rates. The computer reads temperature and performance data from the THPM and relays this to the Delta II telemetry system. Presently, it is unknown if the Delta II uplink is available. As in THERMION-I, the computer controls the power regulator which switches power to the subsystems and heater imbedded in the payload.

12.2.1 Tungsten Wire Heater Grid

Central to the power system is a tungsten wire heater grid. It is very close to the emitter section of THPM. It is a resistive heater that provides the heat input for the THPM. The individual wires in the grid are approximately 5 thousandths of an inch in diameter. When 29 A is passed through the grid, it will heat up to 2600 K which is close to the actual SEHPTR reactor core temperature.

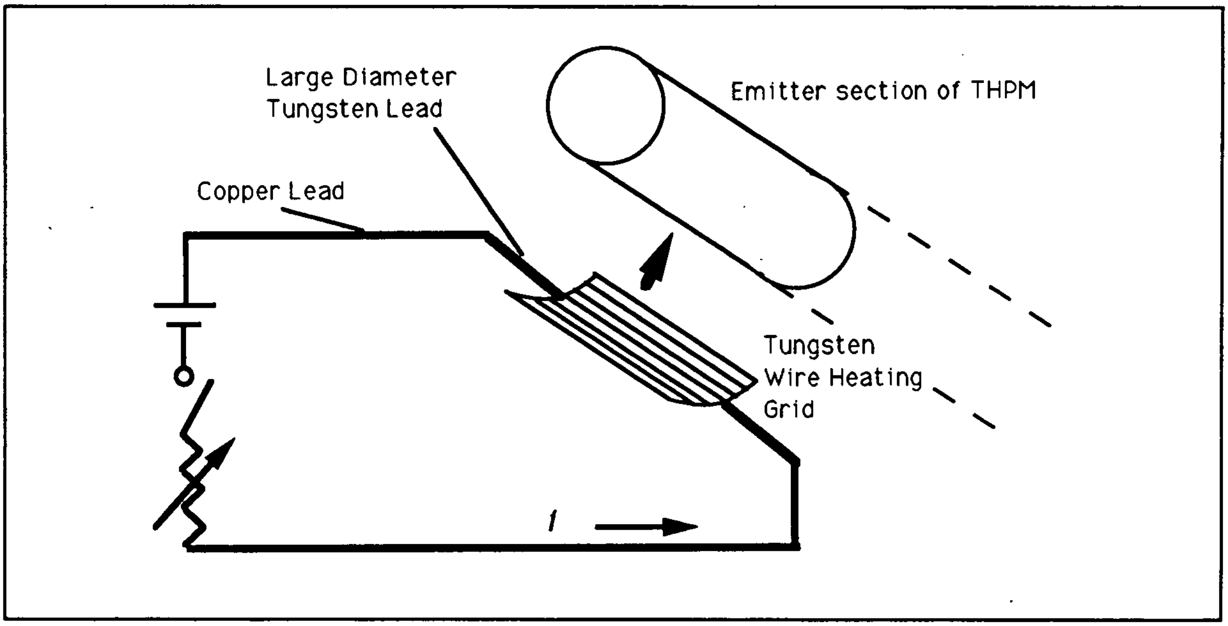

Figure 12-2 Heater Schematic

A coupled thermal/electrical model was created of the heater subsystem to aid in design. It accepts geometry, resistivity, emissivity, current, thermal losses etc. as inputs, and outputs the voltage across the grid and heat lost by conduction through lead wires attached to the grid. It is possible to use one single strand of tungsten wire looped many times to provide the heating grid; however, this will drive up the voltage across the wire to more than 400 V which may cause arcing in the vacuum of space. Because of this, the grid was made into many strands looped to form a grid. This brought the voltage requirement down to 60 V and current up to 29 A. Copper wire is used between the power regulator and the lead wires. The lead wires which must tolerate temperatures in excess of 2500 K are made of 6 mm diameter tungsten. The trade-off is that the larger the diameter of the lead wire, the lower the resistance, but the higher the heat lost by conduction.

The grid itself is electrically insulated from the THPM emitter surface and also from the surrounding insulation by ceramic spacers. All heat transfer to the emitter is by radiation. In actual use, the temperature of the tungsten heater grid could be monitored by monitoring both the voltage and current and then calculating the grid resistance. Tungsten temperature is a strong function of resistivity. This relationship can be determined in calibration tests.

12.2.2 Battery Assemblies and Battery Pallets

Alkaline cells were selected because they are inexpensive, available, have a high energy density, and most importantly, very safe. Lithium chemistries have the highest energy density of commercially available cells but suffer from a dangerous history. Without internal fusing, the cells will explode violently when they are short circuited. Though modern lithium cells have internal fusing and power regulators can be designed to eliminate short circuiting, the legacy still remains. So lithium cells, though flyable on the Delta II, will require involved and expensive safety verification. Therefore, alkaline cells, though less powerful, represent the easiest path to qualification for flight and the lowest cost. Lithium cells remain an option.

The alkaline D-cells are packaged in groups of 120. Four of these battery assemblies fit in one battery pallet, and 4 battery pallets are carried on the Delta II. Each battery assembly and battery pallet is modular and identical. Each battery assembly will have a MIL-SPEC keyed connector on it. The connector carries the power from the battery assembly to a board mounted on the inside lid of the battery pallet. The board serves several functions:

1) Provide for interconnection of the four battery assemblies
2) Fusing of battery assemblies.
3) Safe/Arm of battery pallet
4) Interconnection between battery pallets and power regulator.

On the outer surface of the battery pallet lid are displayed (flange mounted) the interconnection connectors, battery fuses, and safe arm connectors. This will allow testing of the batteries and final assembly without disassembly or access to the batteries themselves. Using keyed connectors eliminates the possibility of incorrect and unsafe connecting.

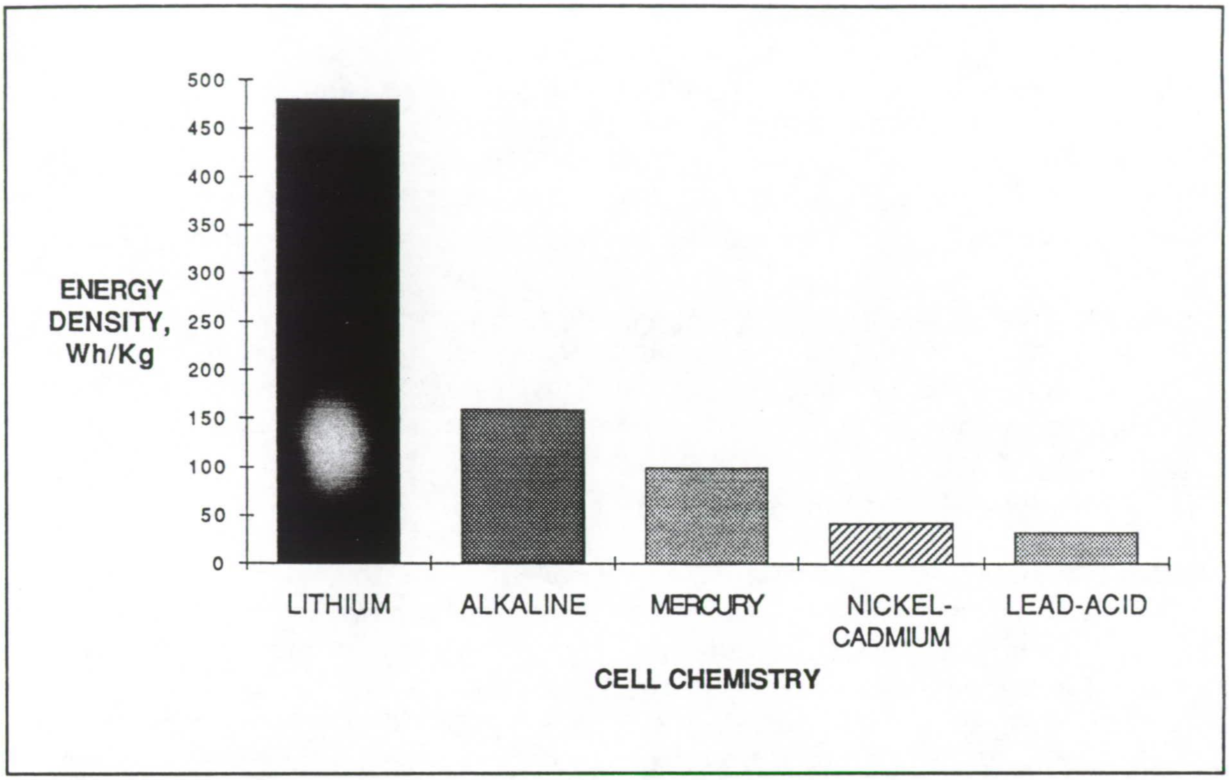

Figure 12-3 Energy Density of Several Available Cell Chemistries.

The battery assemblies are made of the alkaline cells sandwiched between two milled DELRIN™ face sheets 0.25'' thick. DELRIN™ is a non-outgasing plastic that has a good history in satellite applications. The battery assemblies are detailed further in chapter 13.

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13.0 LAUNCH VEHICLE INTERFACE, PAYLOAD, AND STRUCTURE

13.1 LAUNCH VEHICLE SELECTION

In the next four years, there will be approximately nine Air Force Global Positioning Satellites (GPS) launched by the Delta II rocket (1). Each of these missions will have a large enough payload mass margin to place a 750 pound secondary payload into a SECO-II orbit, which is the normal second stage parking orbit for the GPS mission (399 x 100 nautical miles, 33.96° inclination). This orbit has a lifetime of about one month which is ample time for the THERMION-II experiment. Among other reasons, THERMION-II was designed to be launched as a secondary payload aboard the Delta II on a GPS launch because of the large number of launch opportunities and the large payload mass margin.

13.2 LAUNCH VEHICLE INTERFACE

There are two areas on the Delta II second stage where a secondary payload could be mounted (1):

- To the guidance section wall, or
- To the miniskirt support truss.

THERMION-II was designed to be mounted to the guidance section, as shown in figure 13-1, mainly so that the mounting hardware that has already been designed and approved by McDonnell Douglas for secondary payloads could be utilized (2). (See Appendix M for information on the mounting hardware). This should reduce interface costs and facilitate a quick launch (1).

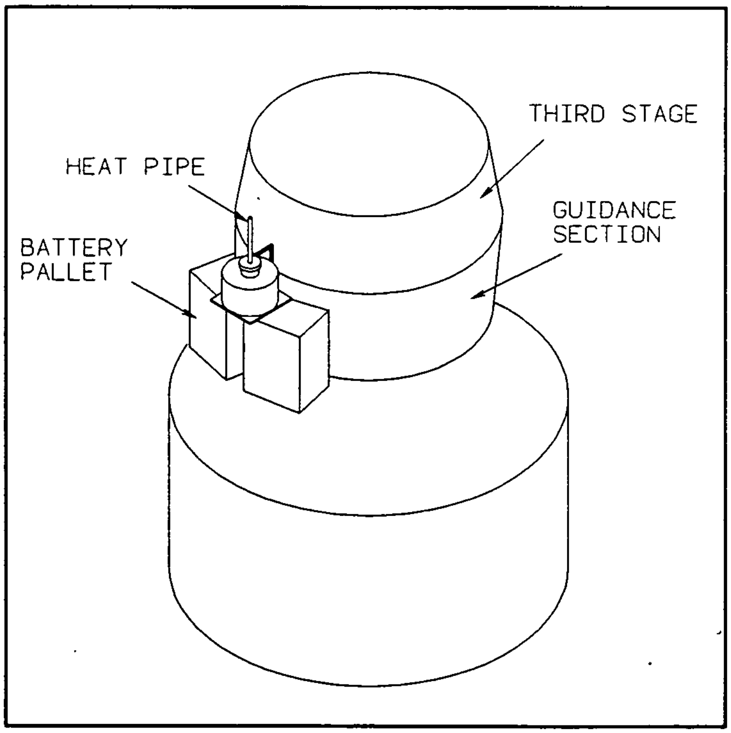

Figure 13-1 Delta II with payload attached.

13.3 PAYLOAD

The payload consists of the battery pallets, the computer box, the heat pipe mounting bracket and the heat pipe insulation. These topics are covered in the following sections along with the payload interface.

13.3.1 Battery Pallet

The purpose of the battery pallet is to hold, in an orderly manner, the maximum amount of batteries in the smallest volume. Each battery pallet is designed to contain four battery packs. Each battery pack contains 120 D-size batteries. The battery pallet and battery pack design will now be explained in detail.

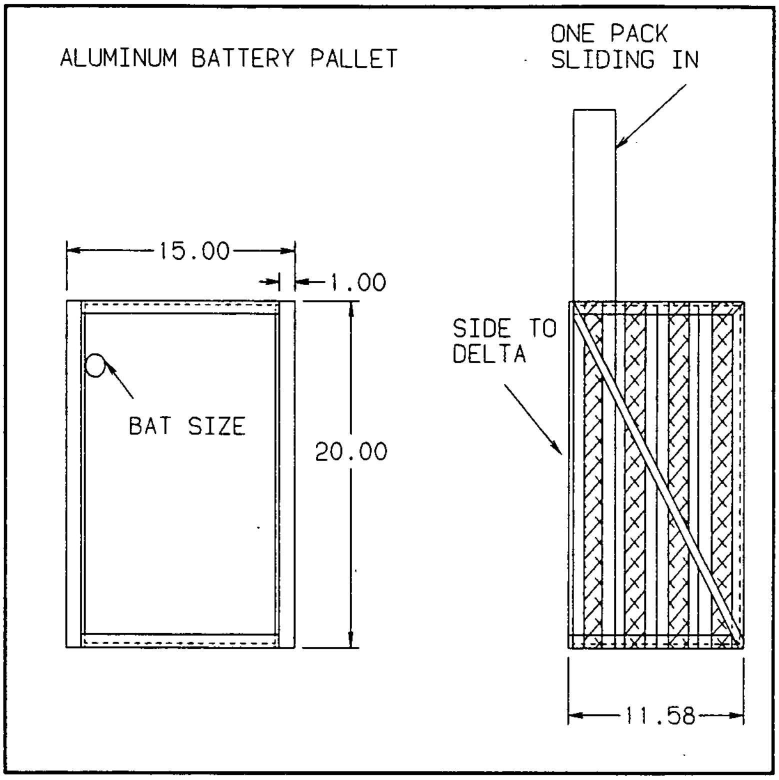

Figure 13-2 Battery pallet assembly.

The battery pallet consists of a frame made from aluminum angle irons. These angle irons form a 11.5 X 15 X 20 inch rectangle. The angle irons dimensions are 2 X 2 X 0.25 inches each. A diagonally placed aluminum bar 0.5 X 0.25 inches carries most of the 160 lb load. The box was designed to withstand this 160 lb load as if it were a point load acting on the end of the box. The box was designed to withstand 10 g's. However, there is an immense safety factor because an aluminum skin of 1/64 inches is placed around the entire outside of the box. The pallet was designed to allow the battery packs to be slid in from the top. This allows any pack to be removed from the pallet at any time. It also allows for the load to be lightened without loosing an entire battery pallet. Figure 13-2 shows the basic dimensions of the battery pallet and how the battery packs are loaded. The design constraints on the battery pallets were to hold the loads and to stay within the given area of the Delta II secondary payload.

13.3.2 Battery Pack

The battery packs are made from Delrin which is a light weight, easy to machine, and inexpensive material. Delrin has also been flight proven. The Delrin will be machined into a honeycomb style as shown in figure 13-3. The figure also shows the basic dimensions of the Delrin packing. The honeycomb style allows the maximum amount of batteries to be placed in the smallest area. The Delrin is machined leaving a thickness of 3/32 inches to withstand the loads implied. Also machined in the Delrin packing will be small channels for the battery wires.

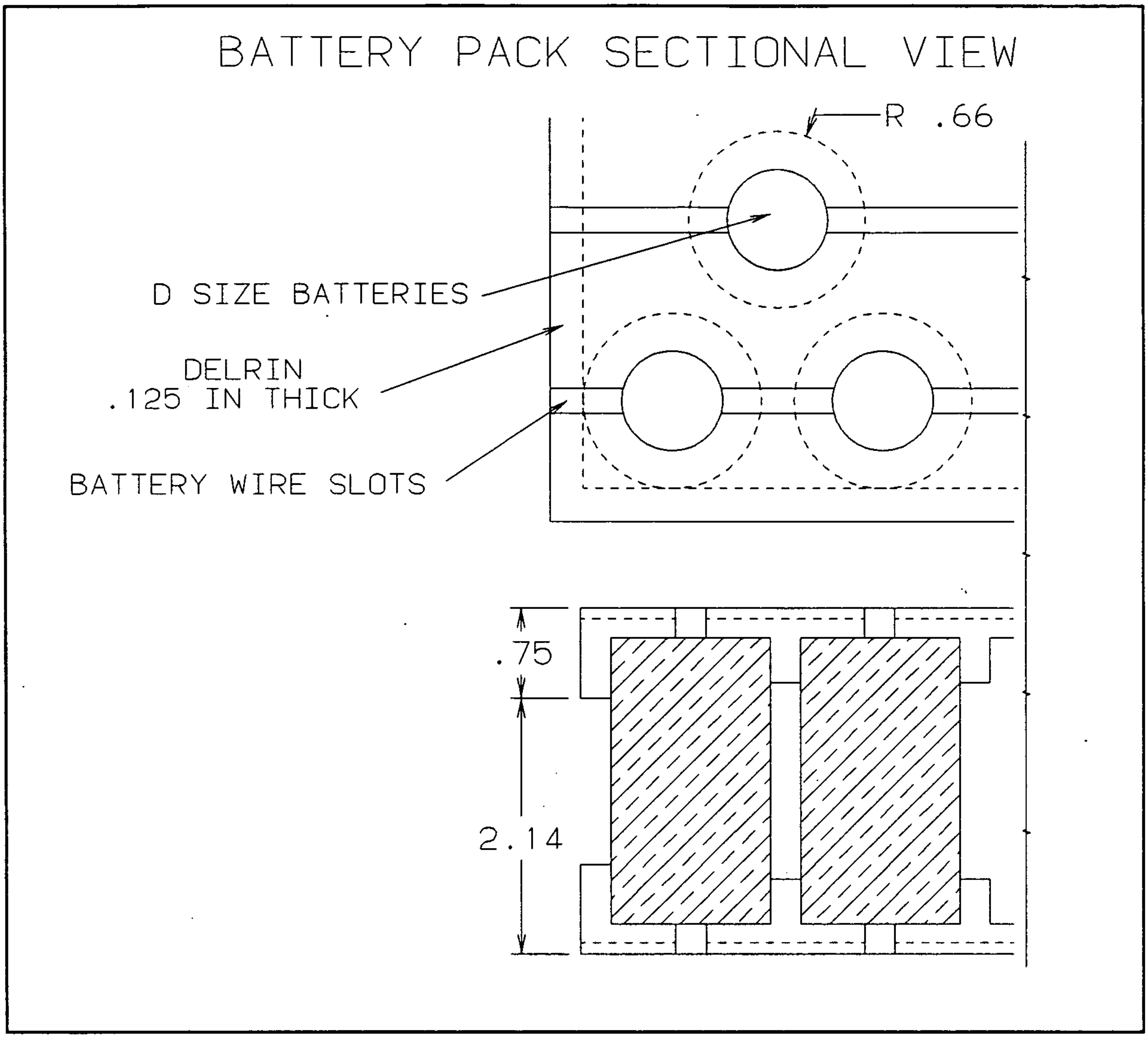

Figure 13-3 Battery pack.

13.3.3 Heat Pipe Mounting

The heat pipe is mounted by a stainless steel arm that runs from the bottom of the insulation, up along the backside of the insulation, and then cantilevers out to the mounting disk of the heat pipe. Stainless steel was used as the arm material because the mounting disk is also made of stainless steel, and the coefficients of thermal expansion will match. The bracket that holds the heat pipe is a spring loaded clamp that wraps around the mounting disk of the heat pipe. This spring type clamp allows the mounting apparatus to expand as the heat pipe thermally expands, yet holds it from moving during launch and whenother loads are applied. The heat pipe mounting arm holds the heat pipe in a position, such that when the third stage has detached, the heat pipe radiates to free space in all directions.

Figures 13-4a & b show both a top and a front view of the mounting bracket. The top view shows how the mounting arm is mounted down on to the aluminum plate. The front view shows the mounting arm going up along the backside of the insulation and attaching to the heat pipe. Spring loaded brackets are commonly used and therefore is not shown in detail in these drawings. The heat pipe insulation in figure 13-4 is simply shown as a black box and is shown in detail later in section 13.3.4.

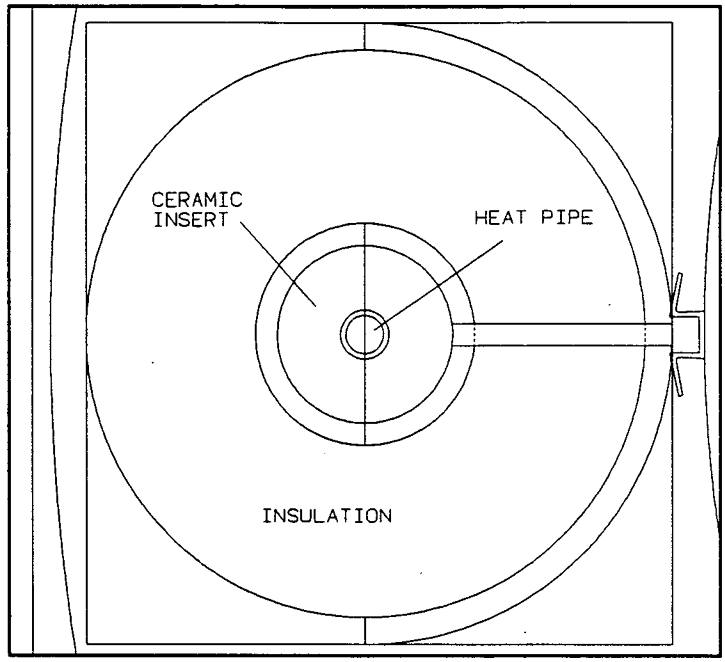

Figure 13-4a Top view of heat pipe mounting bracket.

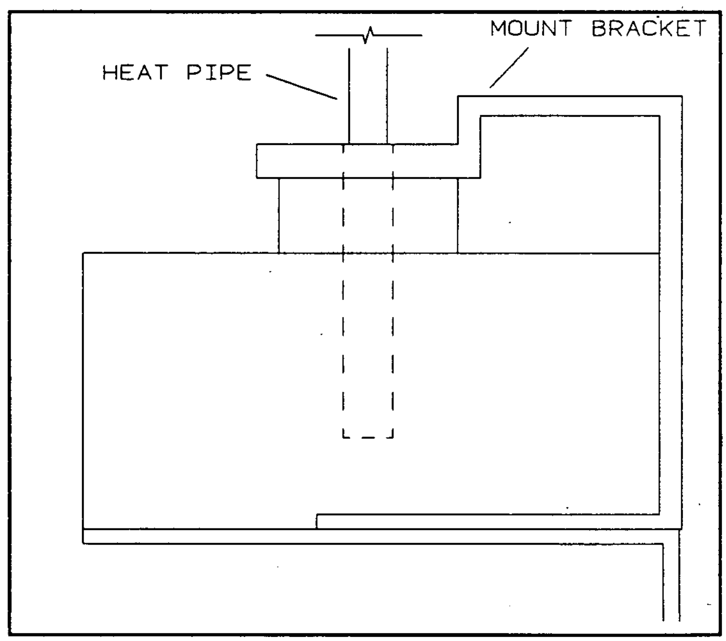

Figure 13-4b Side view of heat pipe mounting bracket.

13.3.4 Insulation System

Two insulating systems were designed. Due to the temperature restrictions of the ceramic material used in the initial design there are possible problems in material failure. Figure 13-5 depicts the initial heat pipe insulation system design. The inner round circles are tungsten heating coil. Kaolite 3300 is a thermal insulating castable ceramic, manufactured by Thermal Ceramics. LI-2200 is a silica fiber, low thermal conductivity insulation for high temperature applications manufactured by Lockheed (see Appendix M for material properties). The steel clamp surrounds the insulation system holding the two ceramic cylindrical halves together and mounts the system to the aluminum frame. The outer members are stainless steel radiation shields; the inner surfaces will be lined with Nomex Felt, and the outer surfaces will be lined or

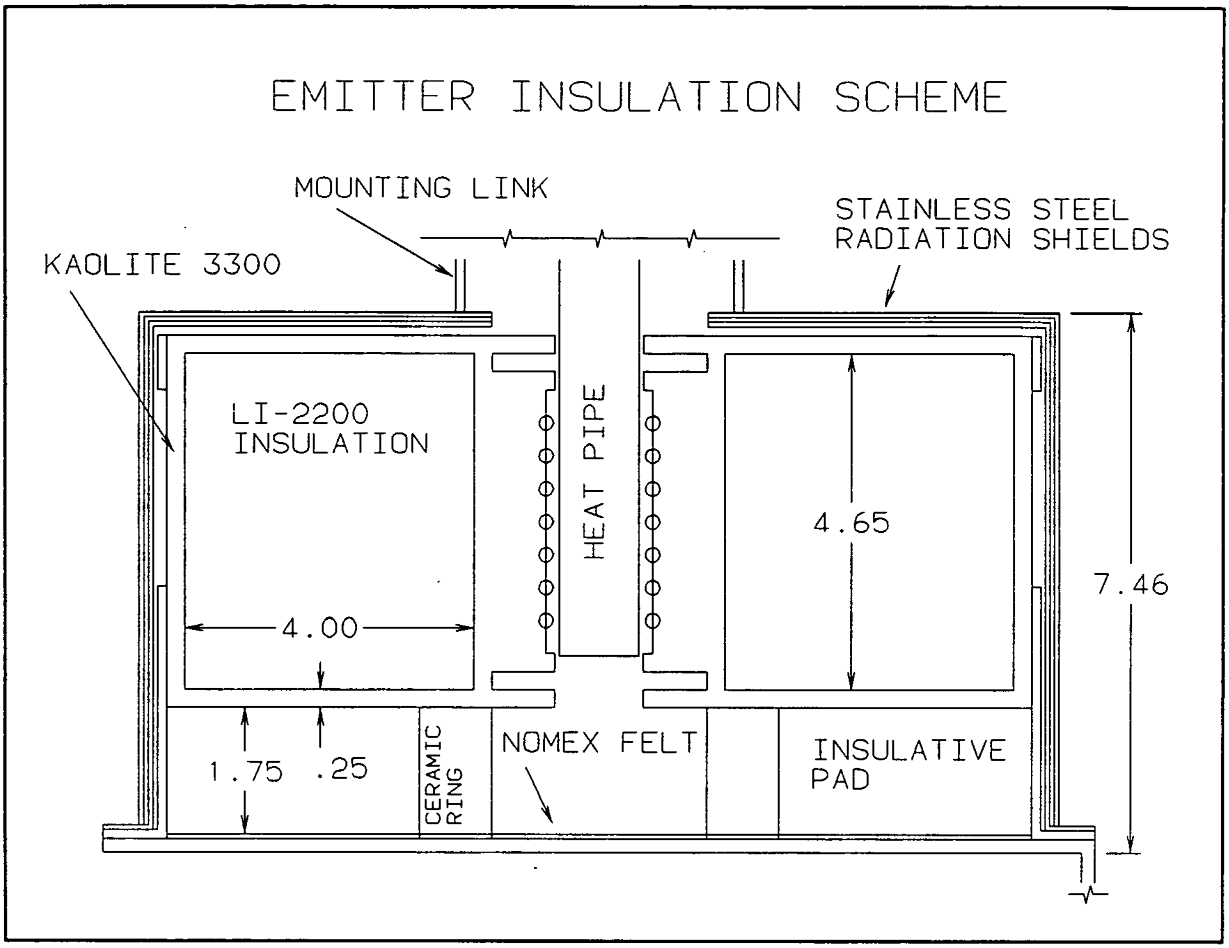

Figure 13-5 Initial insulation design.

painted with a low emissive material. This design has an operating heat loss of approximately 80 watts or 4.5% of the total heat input.

The second insulation design consists of four carbon-carbon arms holding concentric, tungsten foil, radiation shields around the emitter section of the heat pipe. The arms will also hold radiation shields above and below the emitter section of the heat pipe. Surrounding this apparatus is a set of concentric, molybdenum foil, radiation shields that sit on a ceramic mount which sits on the aluminum frame.

13.3.5 Payload Overview

Figure 13-6 shows the side view of the entire payload. The Delta II second stage dynamic envelope is represented by a dashed line, and as shown the payload fits well within the dynamic envelope. The battery pallets will be attached to the rails designed by McDonnell Douglas for their Small Expendable-Tether Deployer System (SEDS Program). The computer box for the payload is located under one of the

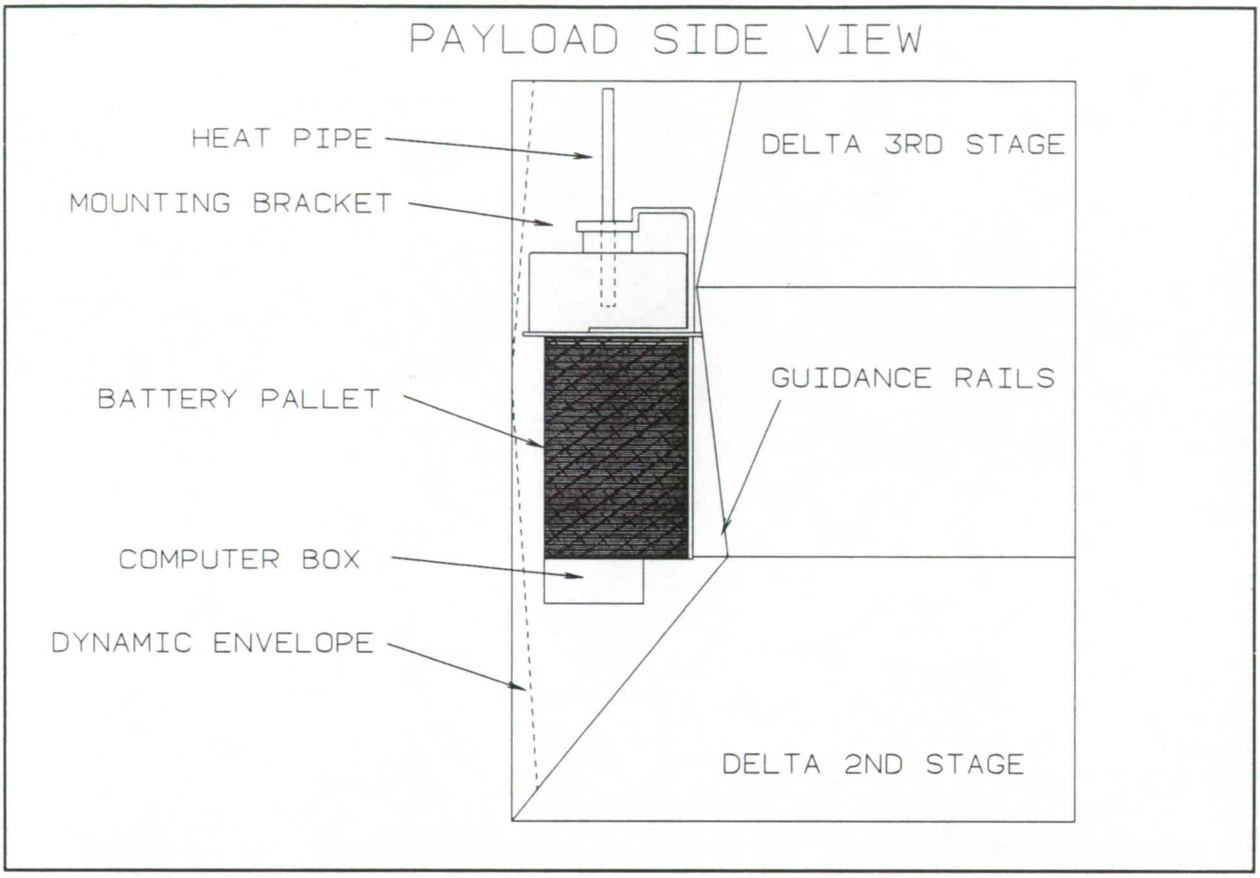

Figure 13-6 Payload side view.

battery pallets. The heat pipe arm bracket and insulation are mounted on the top and in the middle of the two battery pallets. They are mounted to the aluminum plate that comes up from the battery pallets.

The mass breakdown for THERMION-II is as follows:

• BATTERIES	253 kg	556 lb
• PACKAGING AND MOUNTING	36 kg	79 lb
• CPU	3 kg	7 lb
• HEAT PIPE AND INSULATION	5 kg	11 lb
• POWER CONTROL	3 kg	7 lb
TOTAL	300 kg	660 lb

REFERENCES

1. Garvey, J.M.: Delta II Secondary Payload Opportunities. McDonnell Douglas Space Systems Company. August, 1990.

2. Energy Science Laboratoratories, Inc: Small Expendable-Tether Deployer System Interface Control Document. Contract no. NAS8-37885. October, 1989.

14.0 DATA MANAGEMENT & COMMUNICATIONS

The data management system provides control and processing for the following: data acquisition, control power to heat pipe, communications control, and various housekeeping tasks.

This chapter will describe the data management and communications hardware components. Thought was given for software design, but no software has been written to handle any of the data management tasks.

14.1 DATA MANAGEMENT REQUIREMENTS

Data management's major task is data acquisition. Fourteen temperature sensors (thermocouples) line the heat pipe. The experiment will only last a little over 20 hours. Therefore, the data will be collected continuously over this time and then down-linked to a ground station. The data management system was required to select a system that has low power, minimal mass, and a small volume.

14.2 SYSTEM COMPONENTS

The system components will include a CPU, expanded memory card, two data acquisition cards, prototyping card, and storage container. These are the same components as used in THERMION-I and can be found in Section 5.0. Detailed information on these components can be found in Appendix E.

14.3 COMMUNICATIONS

The Small Expandable-Tether Deployer System (SEDS) uses the Delta II communications. THERMION-II will also use the Delta II communications, but limited information on this system has been found. The communication specifications known are as follows:

- TRANSMITTER
  - Frequency 2241.5 MHz
  - Modulation +550 KHz at 3 dB
  - Bandwidth 180 KHz at 69 dB
  - Stability +67 KHz

- ANTENNA
  - TYPE Cavity Backed Slot
  - Gain -2 dB min
  - Omnidirectional

Four watts of power will also be needed to run this system.

5.4 CONCLUSIONS

The computer system that was selected has flown several space shuttle missions and has sat on the bottom of the ocean for two-year periods. This system offers low power, a small volume and weight, and is inexpensive (see Section 14.0 for cost). Software needs to be developed in future work.

The communications system on the Delta II needs further investigation. Following the SEDS program can lead to an inexpensive system.

REFERENCES

1. Elwell, John, Private communication, Utah State University, 1991.

2. Elwell, Jim, Phone conversation, QSI corporation, Utah, 1991.

3. Microspacecraft and Earth Observation: Electric Field (ELF) Measurement Project. M.E. 595 Class, Utah State University, 1989-1990.

4. Griffin, Michael D. and French, James R.: Space Vehicle Design, AIAA, Washington, DC, 1991.

15.0 THE BOTTOM LINE - COST

15.1 HOW MUCH WILL THERMION-I COST?

The total costs were broken down by subsystem, calculating both labor and materials for each. An overhead rate of 100% was assumed, and an additional fudge factor of 1.5 was also multiplied in. Finally, a launch cost of $1,000,000 was added in. Realistically, we expect THERMION-I to cost $2,188,000. The cost breakdown is in Appendix N.

15.2 HOW MUCH WILL THERMION-II COST?

Thermion-II being simpler than THERMION-II will cost less. Our best estimate is $1,651,000. It is calculated using identical overhead, fudge factors and launch costs as THERMION-I. The cost breakdown is in Appendix N.

15.3 COST BREAKDOWN

In accounting the cost, it was assumed that both THERMIONs would be one year programs, from program start to delivery to the launch site.

15.3.1 Overhead Rates And Prime Contractors

There are two ends of the cost spectrum: the customer may choose a very large and experienced aerospace contractor, but pay a high price (overhead) or on the other end of the cost spectrum are the recently arrived small satellite companies, who make up for lack of experience and disorganization in low overhead and extremely dedicated employees. Both companies are capable of building a satellite like THERMION-I or THERMION-II because it is designed simple, capitalizing on existing technologies such as off-the-shelf computers and attitude control systems.

15.3.2 Fudge Factors

Without fail, unique high tech instruments (especially satellites) cost more than they were expected to, take longer to fabricate, and are more complicated than originally believed. It is difficult to account for undiscovered problems from limited experience, and we felt a fudge factor of 1.5 was appropriate.

15.3.3 Launch Costs

The integration and launch of either THERMION-I or THERMION-II is assumed to cost $1,000,000. John Garvey of McDonnell-Douglas (Delta II) said this is the low end of the cost. This figure is negotiable and will be influenced by who owns the primary payload. Assuming both THERMION and the primary payload are from the same organization (ie AIR FORCE), then the launch costs may be kept low.

APPENDIX A

INTRODUCTION

INEL
Idaho National Engineering Laboratory

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October 10, 1990

Dr. Frank J. Redd
Director, Center for Space Engineering
Professor, Mechanical Engineering
Utah State University
Logan, UT 84322-4140

INEL/UTAH STATE UNIVERSITY SPACE EXPERIMENT COLLABORATION - JAL-69-90

Dear Dr. Redd:

It was a pleasure to meet with you on September 28th to discuss INEL's activities in the design and development of a small, ex-core heat-pipe-thermionic space nuclear reactor power system (SEHPTR) for USAF space surveillance missions. This development activity creates, we believe, an interesting opportunity for collaboration between our Laboratory and the Center for Space Engineering at Utah State.

In particular, we propose that Utah State undertake the conceptual design of a space flight-demonstration test vehicle for a solar-powered, thermionic-heat-pipe element for your 1990-91 student design course. Such a conceptual design would call upon several technical disciplines in which Utah State is well qualified, and thereby would make for a good fit to your multidisciplinary student design course. INEL will provide several lectures to lay out the thermionic-heat-pipe design and performance requirements, describe the conceptual design of the nuclear reactor system, and provide the technical information necessary to set the background for your student group to conceive the small space test vehicle design. I currently envisage an introductory lecture in late October, followed by a series of lectures through the Fall school term on issues like the reactor core design, materials issues, thermionic-heat-pipe performance, system reliability. electrical and mechanical design, and space nuclear safety. Other visits to Utah State could be arranged during the Winter term while the design is ongoing on an as-needed basis. We would propose that the final review of the Utah State design be held at the INEL in April of 1991 where we could assemble all of the INEL design team and perhaps arrange for our Dept. of Energy customer and Air Force sponsor to attend.

The objective of the Utah State effort should be to produce a conceptual design of a micro satellite. The mission of this satellite would be to demonstrate the performance of an integrated thermionic-heat-pipe device in space; specifically, verifying the operation of the liquid metal heat pipe and the cesium reservoir concept in micro-gravity. The conceptual design should be of sufficient scope to include not only the space flight test vehicle, but also the design and deployment of the solar collector system, the satellite attitude control system, and the test instrumentation and telemetry system to relay test performance data. Your final design report would need to be sufficiently definitive to allow us collectively to identify the key technical issues and to determine scope, schedule and cost for a proposed joint INEL/Utah State effort to build and fly the test.

We are excited about the prospects for such a demonstration test, and we are eager to work with you and your student design group in the coming months to produce both a successful conceptual design and a meaningful and fulfilling student project.

Sincerely,

James A. Lake

James A. Lake, PhD
Manager, Nuclear Engineering

cc: C. Noble, DOE-ID, MS 1134
P. North, EG&G Idaho, Inc., MS 2509
J. O. Zane, EG&G Idaho, Inc., MS 3600

THERMACORE, INC.

SPI

INEL EXCORE REACTOR CONCEPT

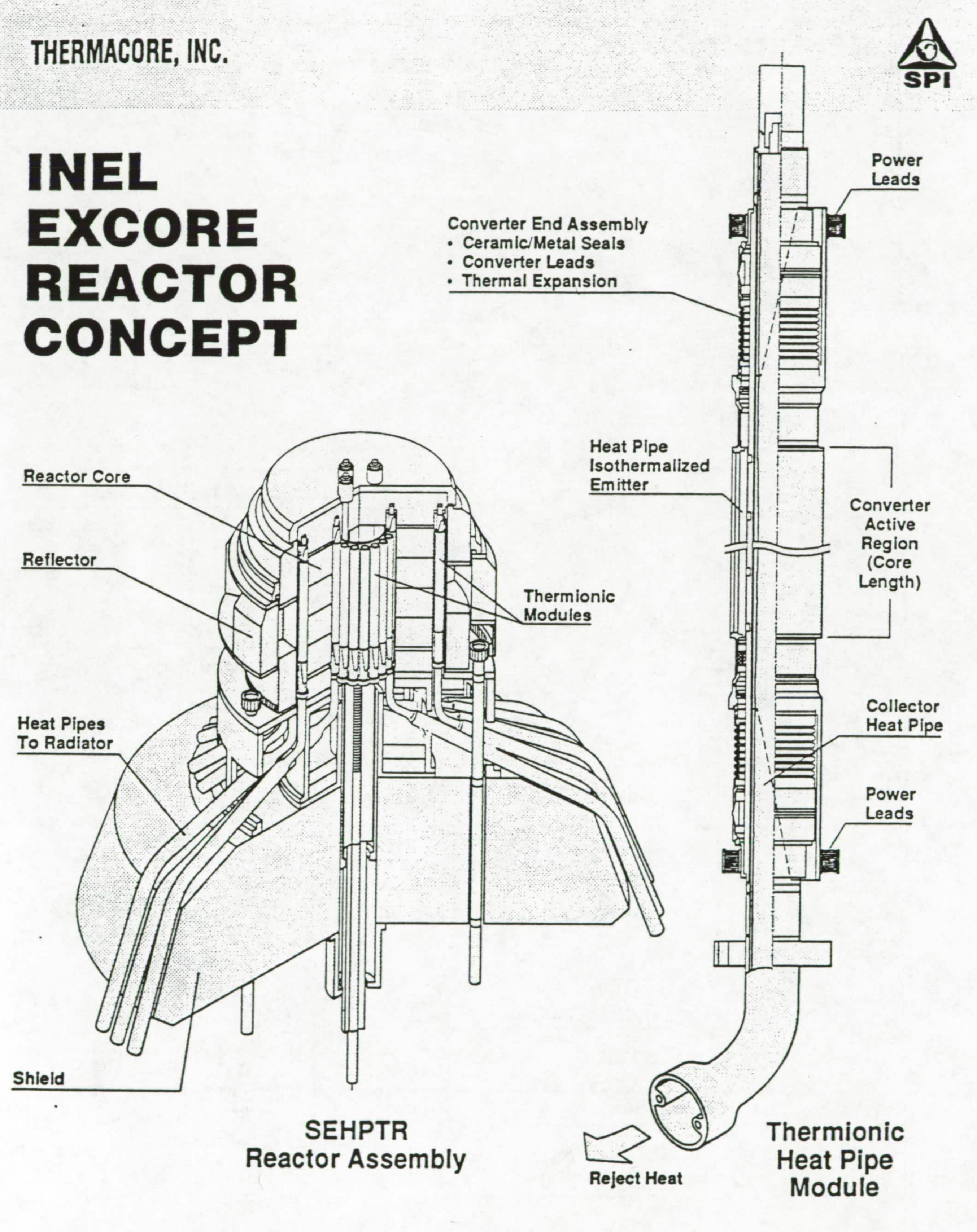

Figure 1. INEL SEHPTR concept

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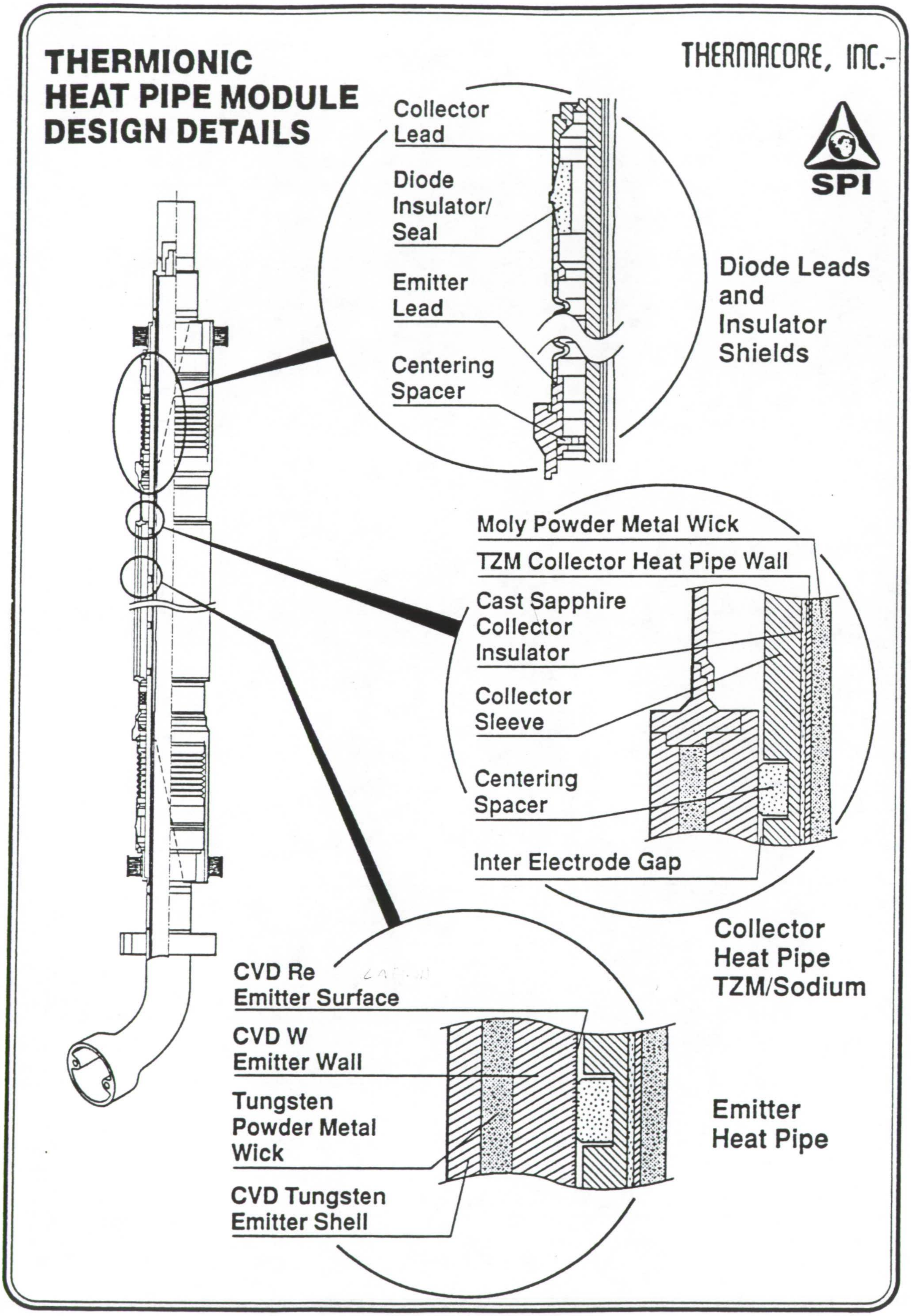


Figure 2. Thermionic heat pipe module design details

II

EXECUTIVE SUMMARY

The Idaho National Engineering Laboratory (INEL) has developed a preliminary conceptual design of a Small Ex-Core Heat Pipe Thermionic Reactor (SEHPTR). The design concept uses integrated Thermionic Heat Pipe Modules to produce 40 kilowatts of electric power. This report documents work done by Thermacore, Inc. of Lancaster, Pennsylvania, and its subcontractors to analyze the conceptual design of a thermionic heat pipe module for application to the SEHPTR power system.

Under INEL Subcontract Number C91-103269-DAJ-1791 a thermionic heat pipe module, conceived by Thermacore, Inc., was designed for use with the SEHPTR power system. A conceptual design was established and analyzed for thermal, mechanical, and electrical performance. Detailed thermionic performance modeling was done by Space Power Inc. of San Jose, California, under subcontract to Thermacore.

Thermionic heat pipe module system modeling and performance trade studies resulted in a baseline conceptual system design and several alternative configurations. The baseline design uses core-length, single cell thermionic elements for electric power generation. The design provides more than 42 kWe (BOL) at 10.2% system efficiency with a maximum SEHPTR fuel temperature of 2619 K.

1.0 INTRODUCTION

Thermacore has developed a conceptual design for a Thermionic Heat Pipe Module (THPM) for use in space nuclear power systems. The THPM has been developed for use with The Idaho National Engineering Laboratory's Small Ex-Core Heat Pipe Thermionic Reactor (SEHPTR) power system. The baseline design produces greater than 40 kilowatts of electric power and offers significant advantages for potential military customers, including simplicity, reliability, and testability. The SEHPTR concept using THPM's may also be scalable over a wide range of power levels.

This report documents work done by Thermacore and its subcontractors under a Phase I Program to develop and analyze a conceptual design for the Thermionic Heat Pipe Module. Objectives of the program were as follows:

1. To identify a conceptual design of an integrated Thermionic Heat Pipe Module which is compatible with the SEHPTR power system.
2. To identify a preliminary design of a proof-of-performance test article and test program that will validate key feasibility issues associated with the Thermionic Heat Pipe Module.

Conceptual design work of the THPM has been completed by Thermacore. A baseline concept was identified and analyzed for thermal and mechanical performance. Detailed thermionic performance modeling was performed by Space Power, Inc. of San Jose California under subcontract to Thermacore. The baseline design uses core length, single cell, cylindrical thermionic elements to meet performance requirements of the SEHPTR power systems.

Section 2 of this report highlights the conclusions and recommendations resulting from this conceptual design study. Section 3 describes the conceptual design of the THPM, including overall performance of the system and compatibility of the design with the SEHPTR concept. Performance analysis of the THPM concept including thermionic and systems design work provided by Space Power Inc., is documented in Section 4. A description of key feasibility issues and the conceptual design of a proof-of-principle test program and hardware is found in Section 5.

APPENDIX B

PAYLOAD

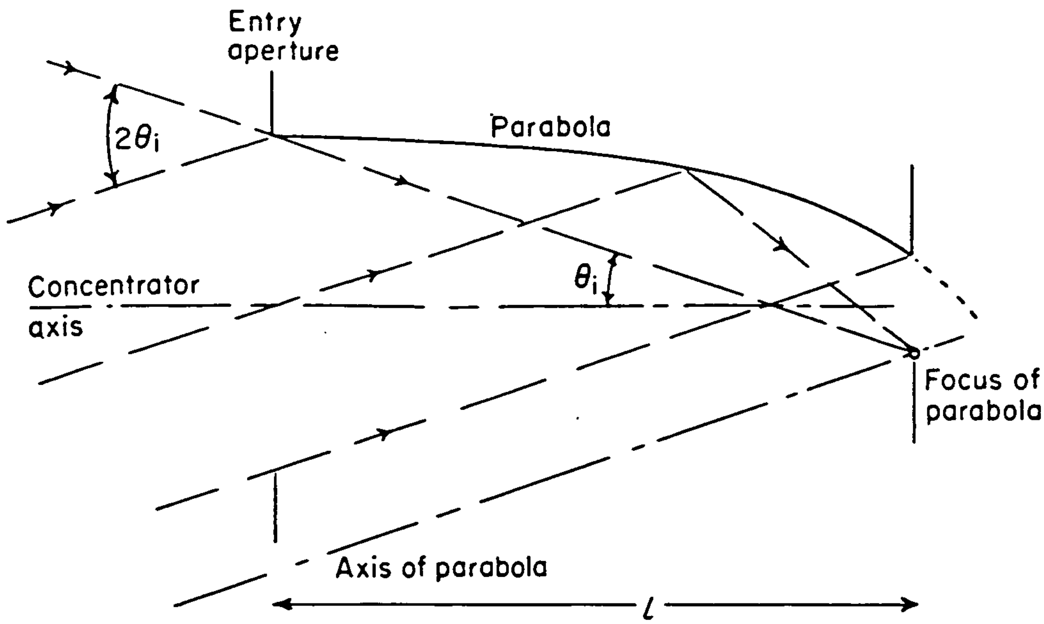


FIGURE Construction of the CPC profile from the edge-ray principle.

Focal length of the parabola is: $f = a'(1 + \sin\theta_i)$

Overall length is: $L = a'(1 + \sin\theta_i)\cos\theta_i/\sin^2\theta_i$

Diameter of the entry aperture is: $a = a'/\sin\theta_i$

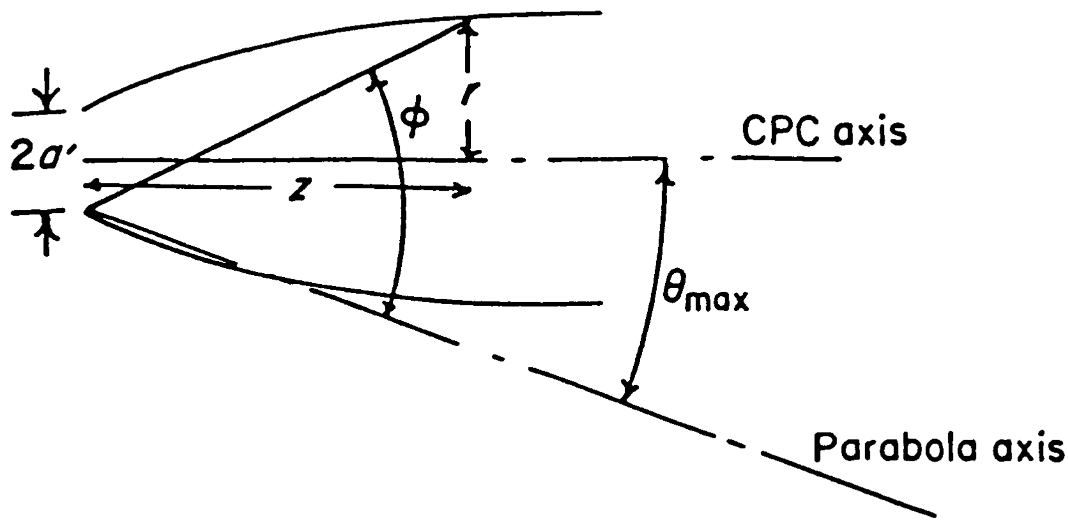


FIGURE The angle ϕ used in the parametric equations of the CPC.

In terms of the diameter $2a'$ of the exit aperature and the acceptance angle this equation is:

$$(r \cos \theta_{max} + z \sin \theta_{max})^2 + 2a'(1 + \sin \theta_{max})^2\, r$$
$$- 2a' \cos \theta_{max}(2 + \sin \theta_{max})\, z$$
$$- a'^2(1 + \sin \theta_{max})(3 + \sin \theta_{max}) = 0$$

PAYLOAD
TEMPERATURE MEASUREMENTS

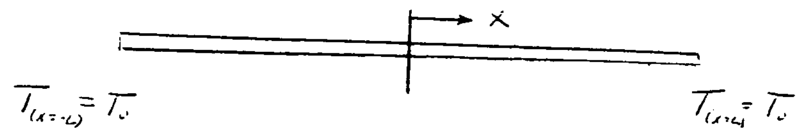
HEAT PIPE - COLLECTOR

- SIX THERMOCOUPLES ALONG LENGTH OF PIPE
  - ANSI type S thermocouples (Pt/Pt - 13% Rh)
    - Capable of temperature measurements to 1720 K
    - 0.1% special limit of error (0.6 K)
  - BeO electrical insulation
  - Tantalum sheath
    - Grounded to thermocouple junction
    - 0.062 inch diameter

HEAT PIPE - EMITTER

- SIX THERMOCOUPLES
  - Tungsten-Rhenium thermocouples (W 5% Re/W 26% Re)
    - Capable of temperature measurements to 3000 K
    - 1.0% limit of error (17 K)
  - BeO electrical insulation
  - Tantalum sheath
    - Grounded to thermocouple junction
    - 0.062 inch diameter (0.16 cm)

Temperature Distribution in Insulated Wire of Length 2L



$$\dot{q} = -K \frac{d^2T}{dx^2}$$

$\dot{q}$ = heat generated per unit volume

$$\dot{q} = \frac{I^2 R}{2\pi D L}$$

$$\frac{d^2T}{dx^2} = -\frac{\dot{q}}{K}$$

integrate once

$$\frac{dT}{dx} = -\frac{\dot{q}}{K}x + C_1$$

integrate twice

$$T(x) = -\frac{\dot{q}}{2K}x^2 + C_1 x + C_2$$

Boundary Conditions

$$T(L) = T_0 \implies -\frac{\dot{q}L^2}{2K} + C_1 L + C_2 = T_0$$

$$T(-L) = T_0 \implies -\frac{\dot{q}L^2}{2K} - C_1 L + C_2 = T_0$$

$$-\frac{\dot{q}L^2}{K} + 2C_2 = 2T_0$$

$$C_1 = 0$$

$$C_2 = T_0 + \frac{\dot{q}L^2}{2K}$$

$$T(x) = \frac{\dot{q}}{2K}(L^2 - x^2) + T_0$$

$$T_{max} = T_{(x=0)} = \frac{\dot{q}L^2}{2K} + T_0$$

Determine Wire Length

$$T_{max} = \frac{\dot{q}\, L^2}{2k} + T_o$$

$$\dot{q} = \frac{I^2 R}{2\pi D L}$$

$$R = \frac{\rho L}{A}$$

substitite $\dot{q} = \frac{I^2R}{2\pi DL}$ $\longrightarrow$

$$T_{max} = \frac{\left(\frac{I^2R}{2\pi DL}\right) L^2}{2k} + T_o$$

substitute $R = \frac{\rho L}{A}$ $\longrightarrow$

$$T_{max} = \frac{\left[\frac{I^2\left(\frac{\rho L}{A}\right)}{2\pi DL}\right] L^2}{2k} + T_o$$

$$T_{max} = \frac{I^2 \rho L^2}{4\pi k D A} + T_o$$

$$L = \sqrt{\left(T_{max} - T_o\right)\frac{4\pi k DA}{I^2 \rho}}$$

$$I = \frac{P}{V} = \frac{(080)(.11)}{0.7} = 169.7\ A$$

$$\rho_{tungsten} = 6.66 \times 10^{-5}\ ohm\text{-}m$$

$$D = 5\ mm$$

$$A = \frac{\pi(.005)^2}{4} = 1.964 \times 10^{-5}\ m^2$$

$$R = \frac{6.66\times10^{-5}}{1.964\times10^{-5}}\ L = 3.391\ L$$

$$k = 138\ \frac{W}{mK}$$

$$T_{max} \approx 2330\ K$$

$$T_o \approx 1000\ K$$

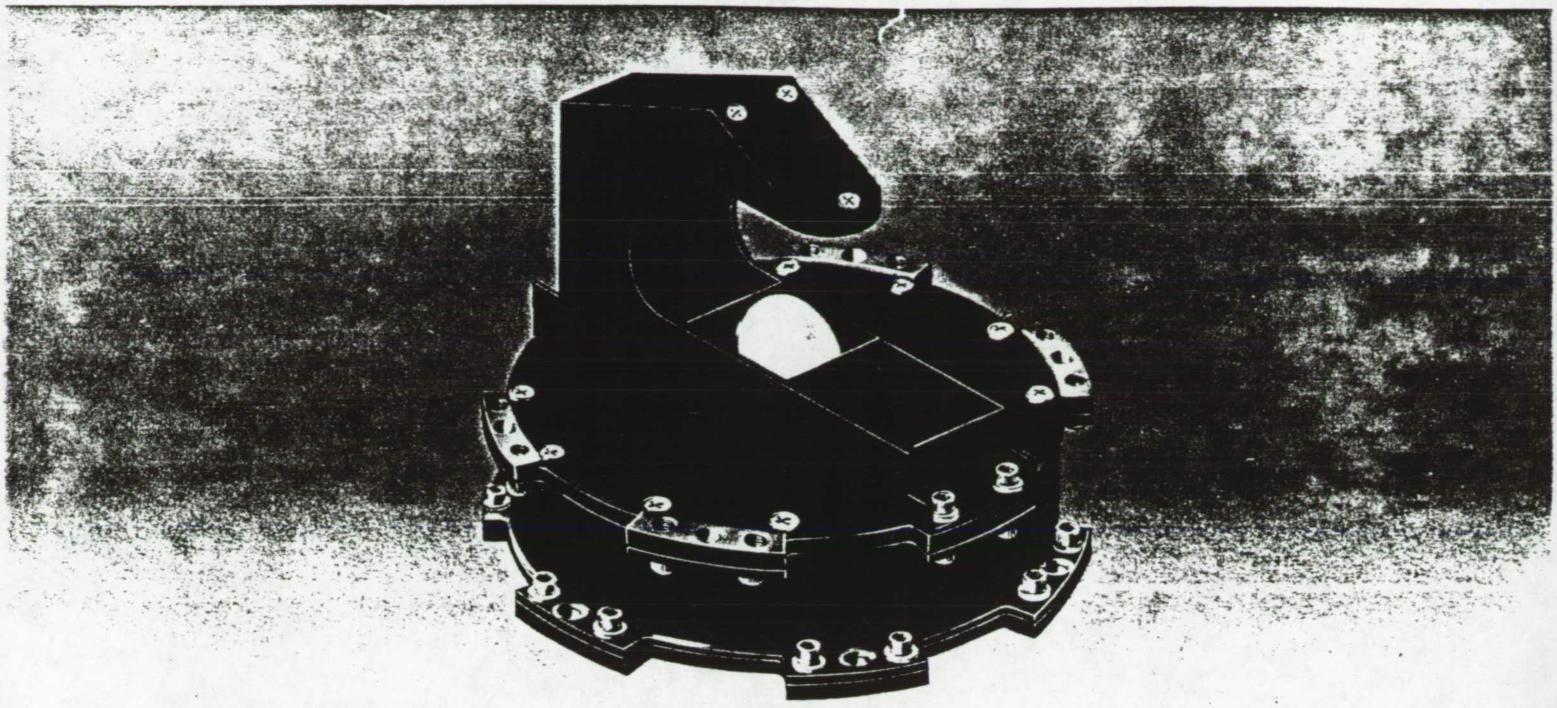
$$L = \sqrt{(2330-1000)\frac{4\pi(138)(.005)(1.964\times10^{-5})}{(169.7)^2(6.66\times10^{-5})}}$$

$$\boxed{\begin{array}{l} L_{wire} = 34.4\ cm \\ L_{wire} = 13.5"\ inches \end{array}}$$

APPENDIX C

ATTITUDE DETERMINATION AND CONTROL

T – SCANWHEEL



APPLICATION

The T–SCANWHEEL® is a momentum/reaction wheel with an integral high accuracy conical earth sensor. The scanwheel provides both angular momentum and control torque, while the conical earth sensor obtains precise attitude information. T–SCANWHEELs are employed on either two or three-axis, stabilized spacecraft where their unique combination of attitude sensing and control capability reduces overall system cost, minimizes weight, and affords an order of magnitude reduction in power consumption.

ITHACO's innovative, modular architecture enables the T–SCANWHEEL to fulfill a variety of mission requirements, functioning as:

- A stand alone momentum/reaction wheel with bolt-on expansion capability
- A combined momentum /reaction wheel with single or dual conical earth sensors

The T–SCANWHEEL is mounted on the spacecraft to achieve proper momentum biasing and storage. A dual T–SCANWHEEL system provides altitude independent pitch and roll attitude information as well as enhanced accuracy. A common application of the T–SCANWHEEL is to replace the earth sensor and the momentum/reaction wheel in a momentum biased ACS. By aligning the T–SCANWHEEL spin axis with the spacecraft pitch axis, gyroscopic stiffness in roll and yaw is provided. A simple pitch lead lag loop controls the T–SCANWHEEL speed so that the spacecraft remains aimed at the earth. TORQRODs then gently torque the spacecraft to align the angular momentum of the T–SCANWHEEL with the orbit normal, thus controlling roll and yaw.

The rotation axis of the T–SCANWHEEL is directed away from the spacecraft to afford the scanning earth sensor a clear field of view. For maximum attitude coverage the scan cone should be free of obstructions, however portions of the scan can be electronically blanked.

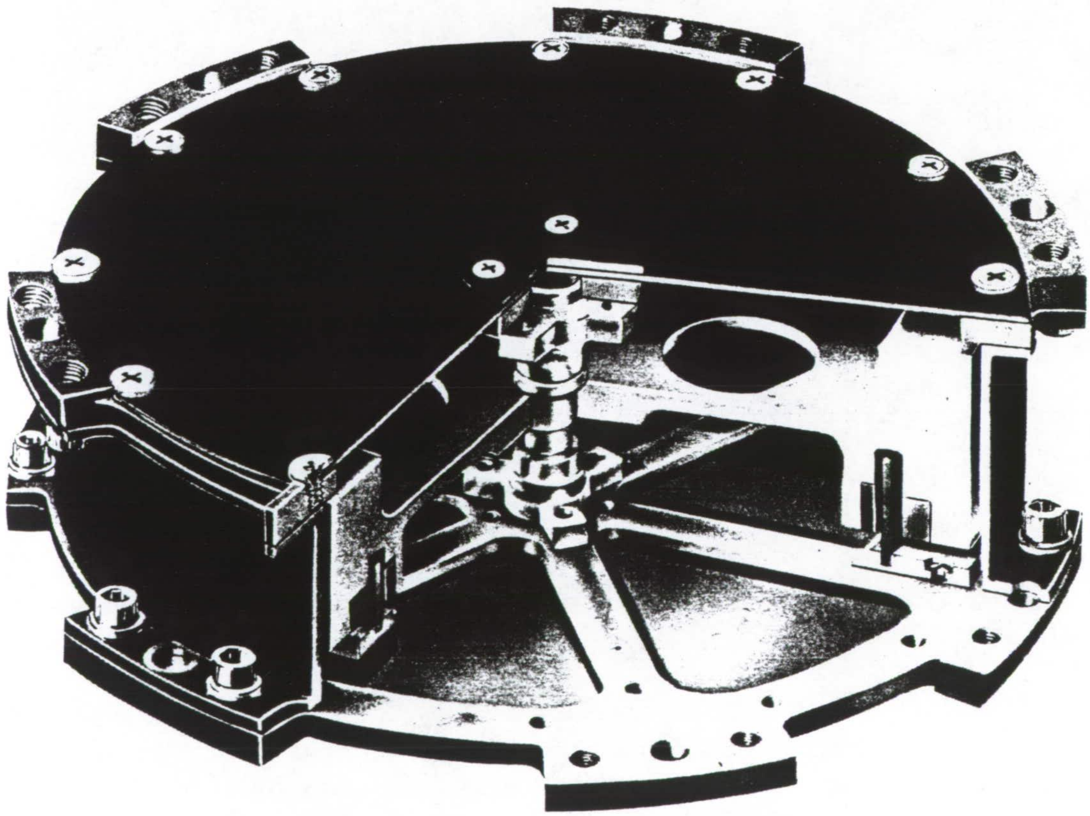
The T–SCANWHEEL's enhanced reliability, low power, light weight, and cost effectiveness make it a superb solution for many attitude determination and control requirements.

ITHACO
SPACE SYSTEMS

SPECIFICATIONS

	REACTION/ MOMENTUM WHEEL	INGEGRATED T–SCANWHEEL
OPERATING SPEED RANGE	0 TO ±6000 RPM	300 to 2000 RPM
ANGULAR MOMENTUM @ 2000 RPM	1.3 N–m–s (1 ft–lb–sec)	1.3 N–m–s (1 ft–lb–sec)
ANGULAR MOMENTUM @ 6000 RPM	4.0 N–m–s (3 ft–lb–sec)	N/A
AVAILABLE REACTION TORQUE	20mN–m (2.8 oz–in)	20 mN–m (2.8 oz–in)
OPTICAL PASSBAND	N/A	14–16μ
FIELD OF VIEW COVERAGE	N/A	270°
SCAN CONE HALF APEX ANGLE	N/A	45° – 85°

Accuracy N/A 0.1°



DESCRIPTION/HARDWARE

The Momentum/Reaction wheel contains an aluminum flywheel suspended on ball bearings and driven by an ironless armature, brushless DC motor. Optimum power efficiency and maximum inertia to weight are realized with the large diameter motor components. The symmetrical housing provides dual mounting surfaces to allow the addition of the T–SCANWHEEL optics module and to expand mounting options on the spacecraft. Increased momentum capacity or redundancy can be obtained by bolting two assemblies together.

The T–SCANWHEEL optics consists of scan and fold mirrors, objective and field lenses, spectral filters, and an infrared detector. The scan mirror reflects light by 45 degrees* causing the Field of View (FOV) of the T–SCANWHEEL to sweep a 45 degree* half apex angle cone in space. When the field of view alternately crosses cold space and the hot earth, the infrared signal produced by the earth is detected, amplified, and sent to a separate electronics box for processing into high precision phase and chord or pitch and roll attitude information. In order to provide the scanning motion, the SCANWHEEL always runs at or above bias speed. The associated electronics box contains the DC-DC converter, signal processor, and motor driver.

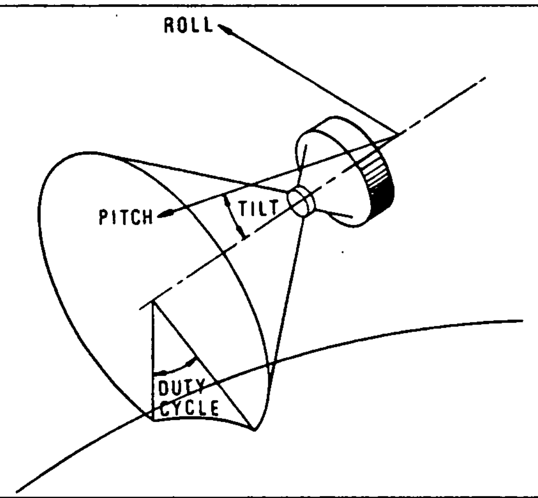
The DC-DC converter provides power conditioning and isolation compatible with single point grounding schemes typically encountered on spacecraft. The high efficiency motor driver controls the motor in the T–SCANWHEEL in response to external analog torque commands. The motor driver also supplies a reference tachometer signal to the spacecraft.

The signal processor conditions, amplifies, and limits the raw detector output which is then subsequently processed into either phase and chord or pitch and roll information. Sun and moon location and/or discrimination and rotor speed compensation functions are also accomplished by the signal processor.

The baseline T–SCANWHEEL configuration may be augmented with additional electronics to perform functions such as attitude computing, TORQROD™ driving, magnetometer conditioning, telemetry conditioning, redundancy management, and mode switching, along with any implementation of the ACS control laws.

*Optional up to at least 85 degrees.

SCANWHEEL
On–Orbit Geometry



$225,000

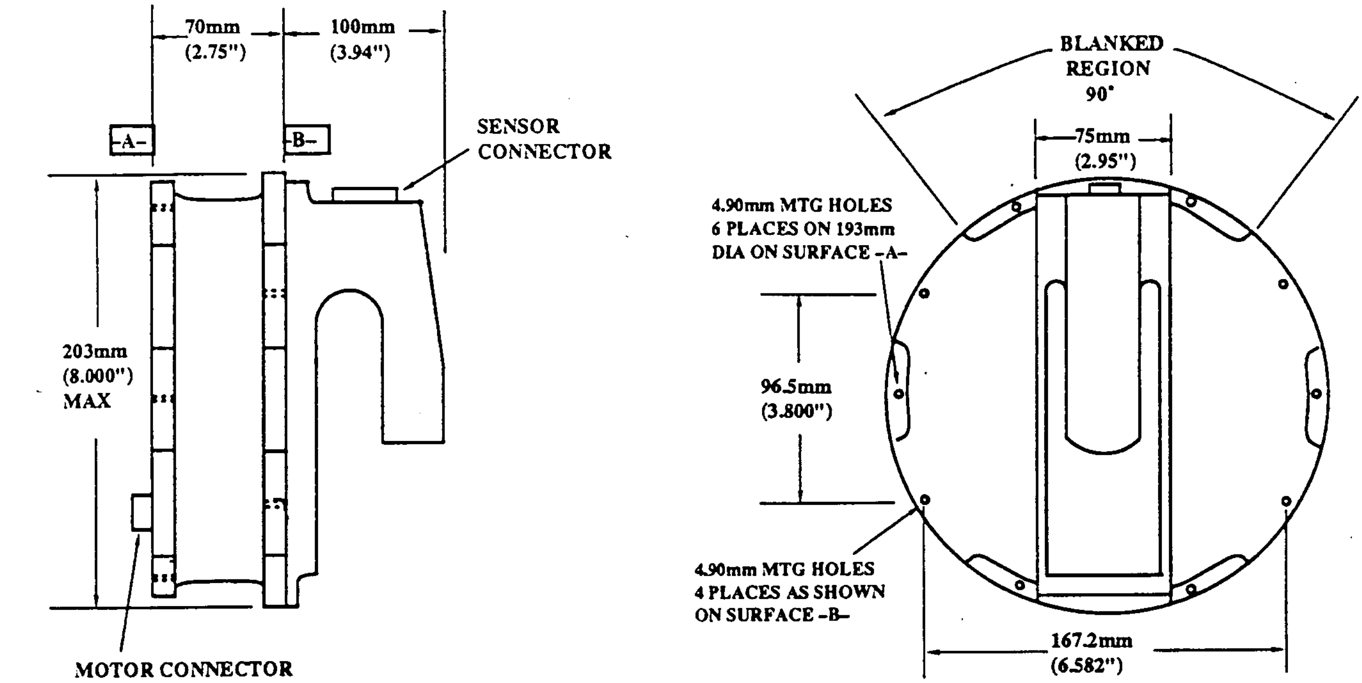
HARDWARE SUMMARY

	WEIGHT	POWER	@ 2000
REACTION/MOMENTUM WHEEL	2.3 Kg (5.0 lbs)	0.5W @ 1000 RPM	1.6W
INFRARED SENSOR MODULE	1.0 Kg (2.2 lbs)	0.1W	.1
ELECTRONICS (Single R/M Wheel)	1.3 Kg (2.9 lbs)	2.0 W	
ELECTRONICS (Single SCANWHEEL)	1.9 Kg (4.3 lbs)	3.0 W	3.2W
ELECTRONICS (Dual SCANWHEEL)	3.2 Kg (7.1 lbs)	5.3 W	
TOTAL SINGLE SCANWHEEL SYSTEM	5.2 Kg (11.5 lbs)	3.6 W	4.3W

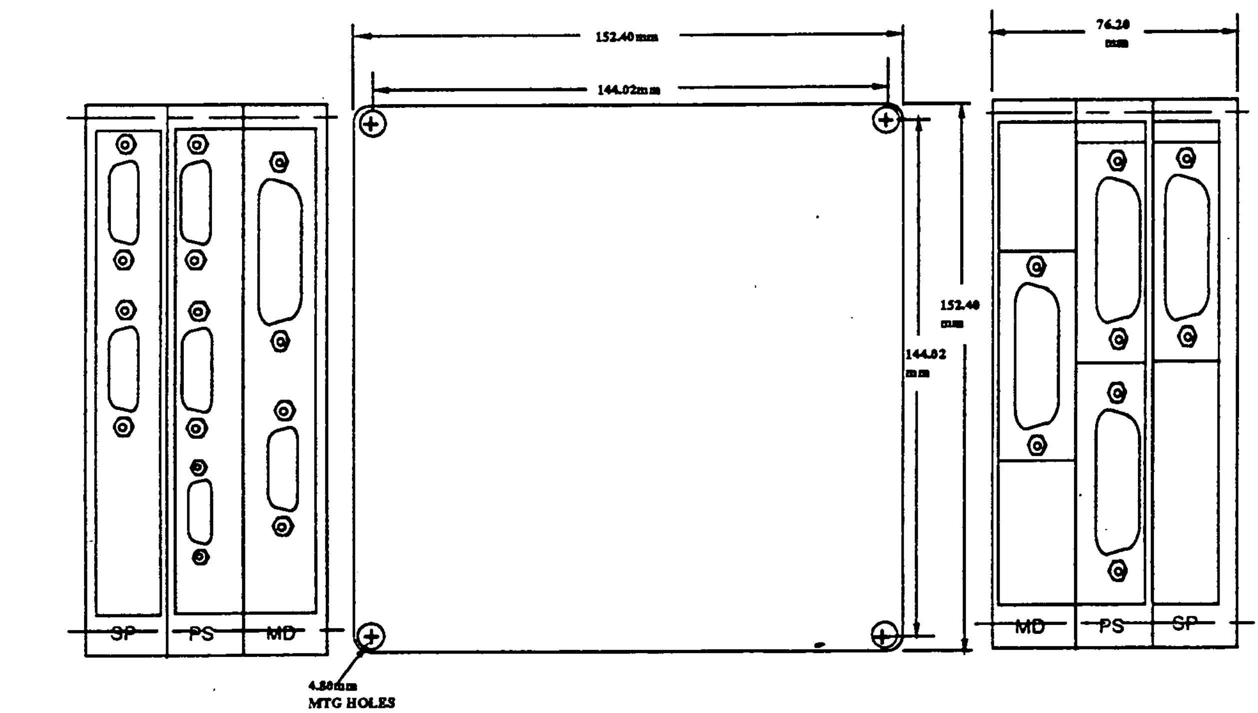
1000 3.6
1250 3.9
1500 4.2
1750 4.5
2000 4.9

[redacted]

SCANWHEEL OUTLINE



ELECTRONICS OUTLINE



ITHACO

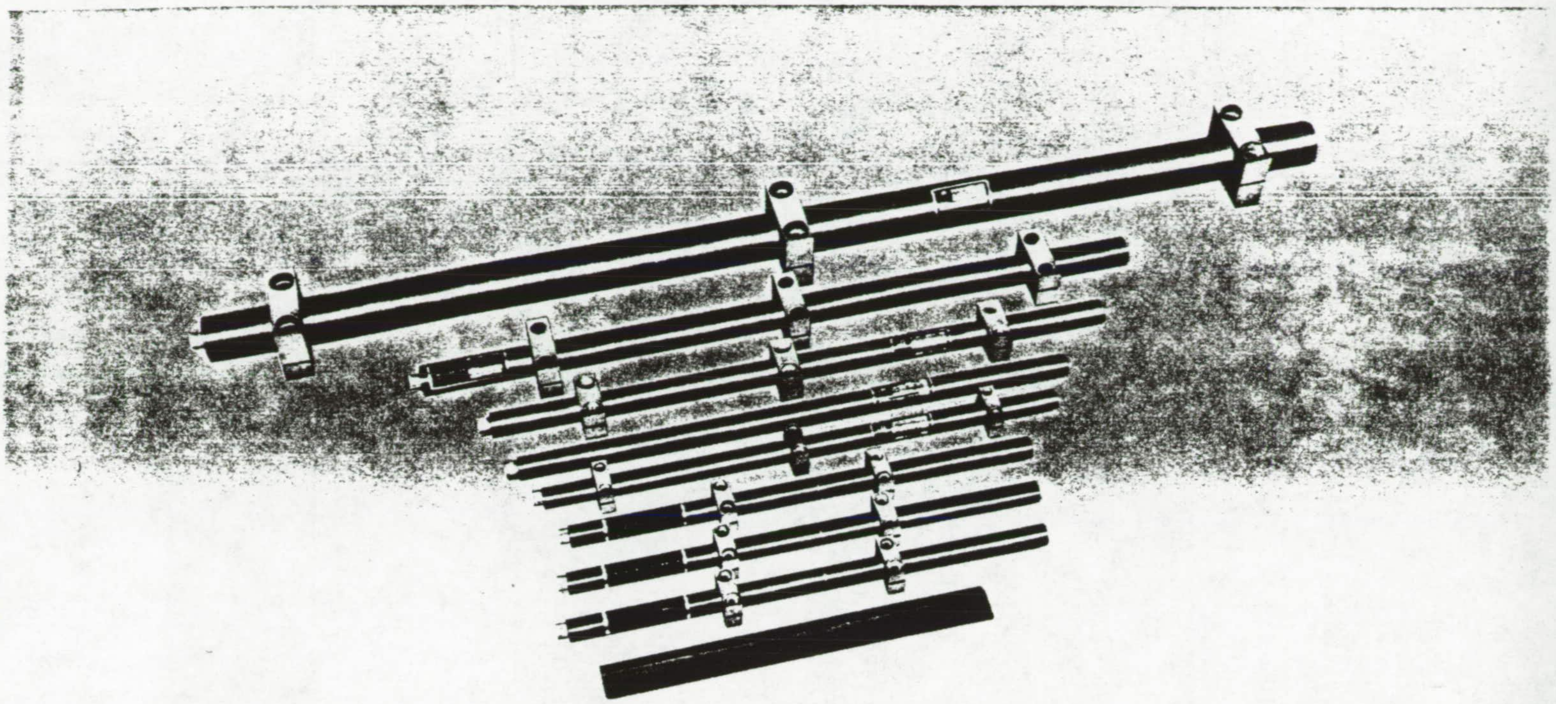
ATTITUDE IS EVERYTHING

735 West Clinton Street, Box 6437, Ithaca NY 14851–6437 USA
Tel: (607) 272–7640 (800) 847–2080 FAX: (607) 272–0804 TWX: 510–255–9307

IPS 13 5/89

[redacted]

TORQROD™



APPLICATIONS

The ITHACO TORQROD is an electromagnet designed to provide complete momentum management on an Earth orbiting spacecraft . Dipole moments developed by the TORQRODs interact with the Earth's magnetic field to generate gentle torques on the spacecraft. Stabilization of a tumbling spacecraft can easily be achieved by a "minus b-dot" control law. The dipole moment (Mx) of each TORQROD is forced according to the law:

$$Mx = -d(Bx)/dt$$

where d(Bx)/dt is the time derivative of the component of the Earth's field along the TORQROD. As a result of this law, the TORQROD acts as a magnetic brake and locks onto the Earth's magnetic grid. Several control laws for momentum management have been developed by attitude control system designers. In all cases, a momentum error vector (H) is determined by the ACS, and the desired dipole moment vector (M) is given by the vector cross product (HxB). The resulting torque (MxB) reduces the error vector H. Specific control laws can solve problems arising from specific mission requirements such as incomplete knowledge of the error vector or the preferential treatment of particular terms in the cross product.

One TORQROD is aligned with each axis of the spacecraft and should be mounted away from any instruments sensitive to magnetic fields. In addition, TORQRODs should be separated from one another in order to avoid cross coupling effects. The use of TORQRODs allows complete momentum management of a spacecraft without consumables in almost any orbit. Even in a geostationary orbit, TORQRODs can make a significant contribution to the total momentum management problem. By eliminating or reducing the need for consumables, the use of TORQRODs can reduce mission costs significantly.

ITHACO

SPACE SYSTEMS

TYPICAL CHARACTERISTICS

REFERENCE NUMBER (Program)	MOMENTS (A–m²) LINEAR	SATURATE	RESIDUAL	MASS [2] Kgm (lbs)	LENGTH Meters (inches)	DIAMETERS [1] Centimeters (inches)	NUMBER OF COILS	RESISTANCE [3] AT 25°C ohms	SCALE [4] FACTOR A-m² /mA	NOTES
B22731 (ETS-3)	10	15	0.05	0.375 (0.83)	0.4 (15.5)	1.8 (0.7)	1	150	0.19	Fiberglass case and 2 mounting blocks
C33008 (SSA)	10	15	0.05	0.4 (0.9)	0.4 (15.5)	1.8 (0.7)	2	280	0.20	Fiberglass case, no connector (pigtail leads), 2 mounting blocks. Coils may be used separately, in series, or in parallel
C32405 (SME/LIPS)	20	30	0.1	0.85 (1.87)	0.5 (19.5)	2.25 (0.875)	2	127	0.19	Fiberglass case, 3 mounting blocks
C32335 (P80-1)	30	45	0.15	0.9 (2.0)	0.5 (19.5)	2.25 (0.875)	2	130	0.20	Same as C32405
C32660 (MOS1)	60	80	0.3	1.5 (3.4)	0.64 (25)	2.5 (1.0)	2	39	0.24	Same as C32405
D42867 (CMM-1)	80	140	0.4	3.2 (7.0)	0.86 (34)	3.3 (1.3)	2	106	1.18	Same as C32405
D42653 (GSTAR)	100	140	0.5	2.0 (4.5)	0.83 (32.8)	2.1 (0.82)	2	157	0.89	Caseless Design for low Weight. Note 5.
D42440 (VIKING)	190	240	0.9	3.36 (7.4)	0.78 (30.6)	3.2 (1.26)	2	67	0.54	Aluminum case. Note 5.
D42733 (SATCOM/KU)	210	350	1.0	4.1 (9.1)	0.92 (36)	2.8 (1.1)	2	127	1.17	Caseless Design for low Weight. Note 5.
D43111 RCA/MIX	500	600	3	7.26 (16.0)	1.27 (50.0)	3.10 (1.220)	2	121	2.19	Same as D42653
D43112 RCA/MIX	850	1000	5	12.1 (26.6)	1.40 (54.98)	3.76 (1.48)	2	83	2.49	Same as D42653
D42910 (GRO)	2000	2700	10	43.2 (95)	1.46 (57.5)	7.5 (2.9)	2	2.5	0.67	Stainless steel case. Note 5.
D42355 (Sp. Tel.)	2800	4000	14	43.2 (95)	2.49 (98)	6.4 (2.5)	1	14	3.96	Stainless steel case. Note 5.

1. Not including mounting feet
2. Mass includes mounting blocks
3. Copper wire, TC .393%/deg C. Resistance is that of a single coil.
4. Scale factor is that for one coil.
5. Metal case or mounting blocks used. Use with onboard magnetometer closed loop feedback systems is not recommended due to "shorted turn" effects of case or mounting blocks.

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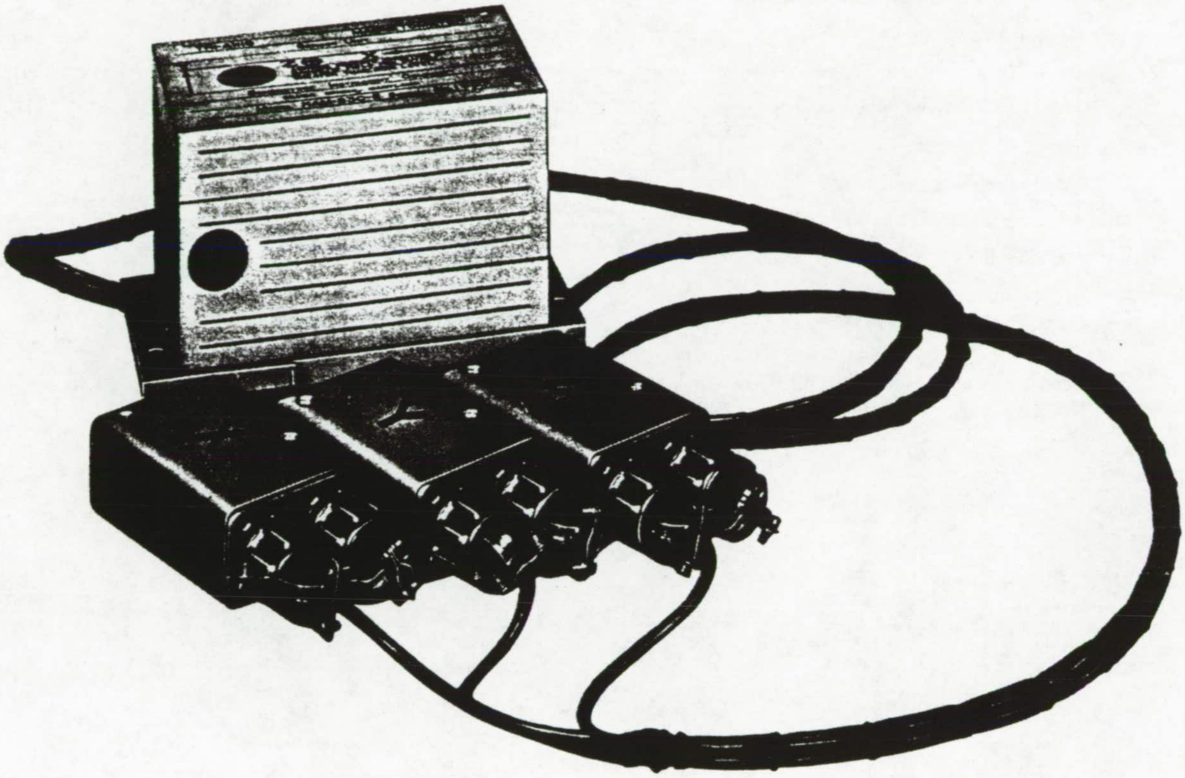
TECHNICAL BULLETIN

MODEL **RAM-53C-2**

SCHONSTEDT

HELIFLUX®

TRIAXIAL MAGNETIC ASPECT SENSOR



DESCRIPTION

The RAM-53C-2 is a triaxial flux-gate magnetometer specifically designed for use in rocket-aspect measurement systems. A field component within the range of ±600 millioersteds is converted to an alalog voltage defined by the equation:

$$E = 2.40 + .004\ H \cos \phi,$$

where E is the output in volts, H is the ambient field in millioersteds, and ϕ is the angle between the magnetic-field vector and the sensor's positive magnetic axis. Since the cosine term can be positive or negative, the output can swing from 0.0 to 4.8 volts.

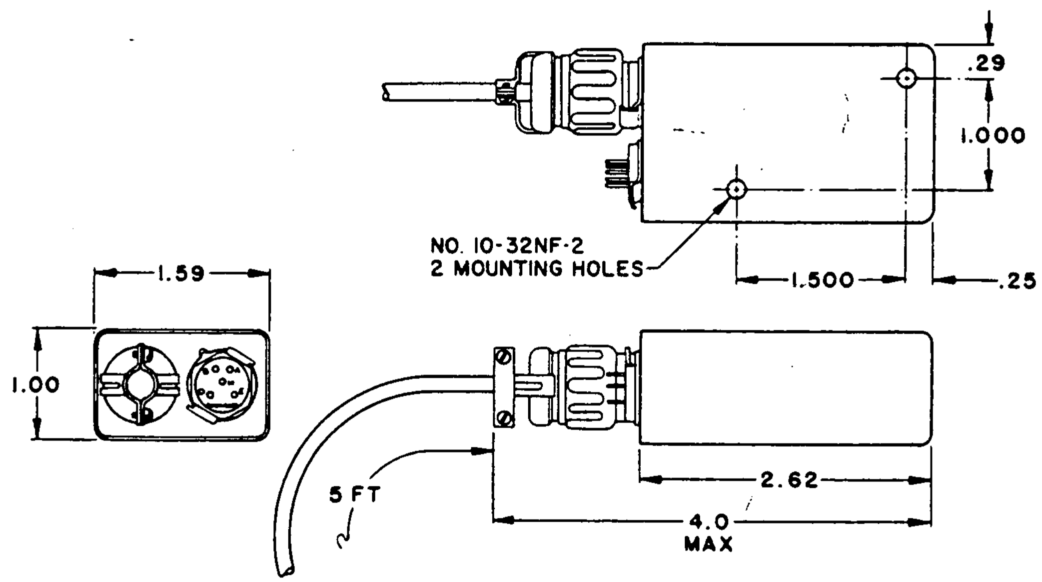
The sensor unit houses three orthogonally aligned flux-gate probes encapsulated in foam and protected in an aluminum case. Three separate electronics packages that are identical to those of the RAM-5C are connected to the sensor unit by three separate cables.

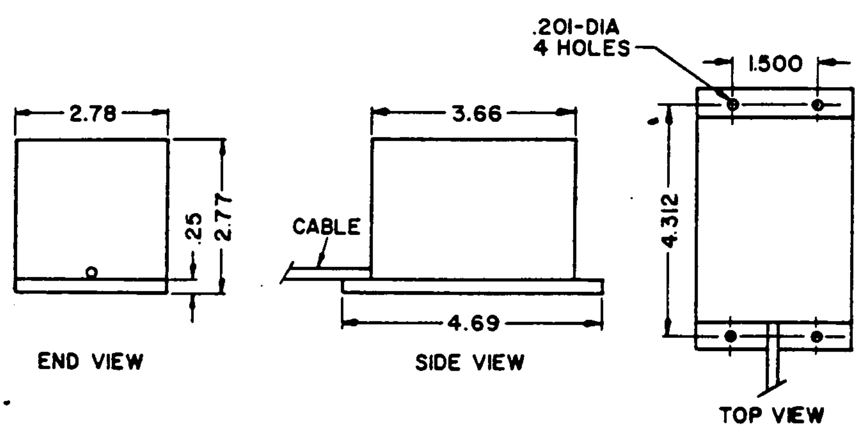
The unique Schonstedt flux gate makes it possible to build the instrument with a minimum number of electronic components, resulting in improved instrument reliability and small, lightweight electronics packages. The instrument can survive the shock, acceleration and vibration environment normally encountered in rocket applications.

RAM-53C-2 SPECIFICATIONS

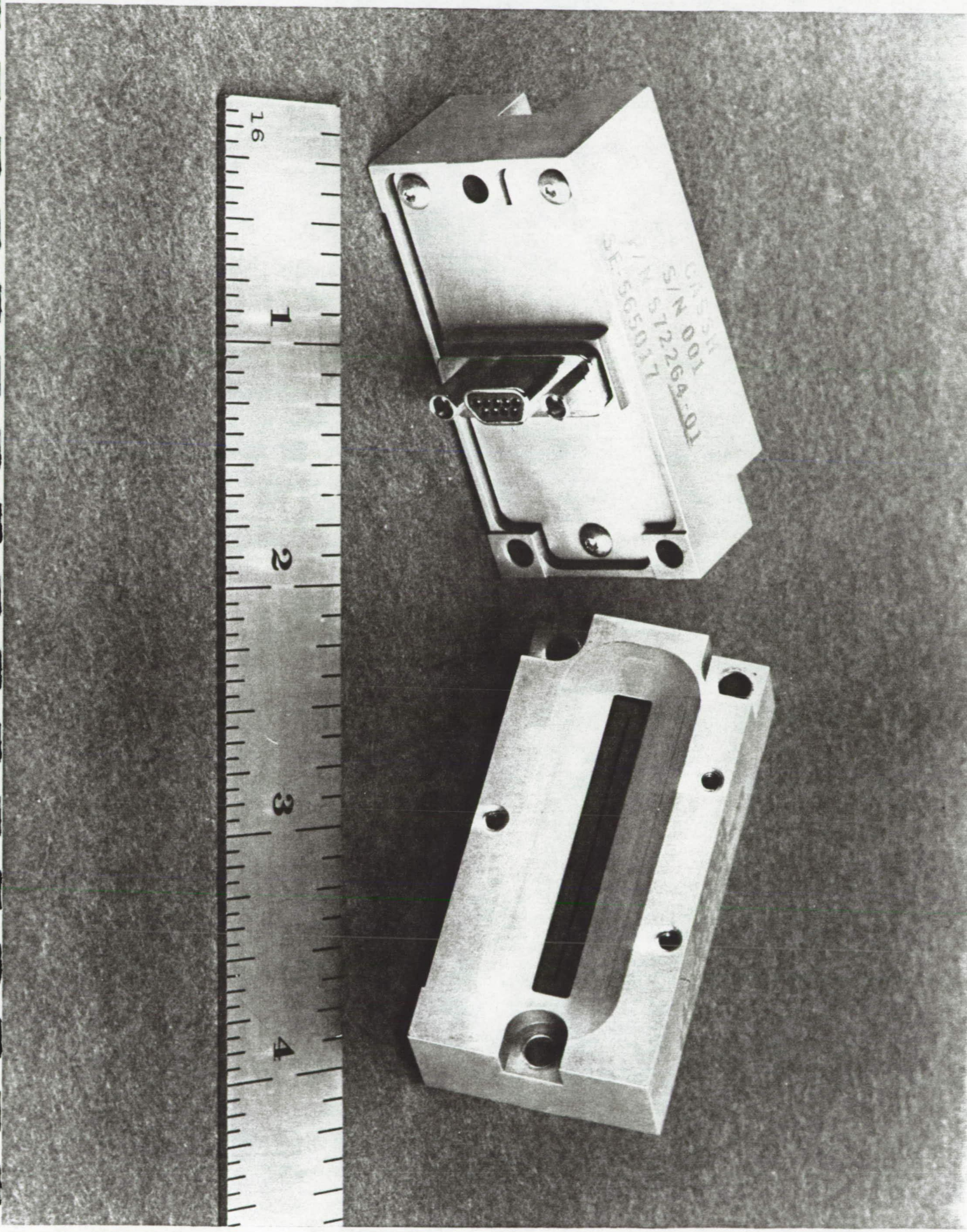
Input Voltage:	24 to 32 volts dc
Input Current:	33 milliamperes total (typical)
Range of Field:	0 to ±600 millioersteds
Sensitivity:	.004 volts dc per millioersted
Stability of Sensitivity:	±3%
DC Output for Zero Field:	2.40 ±.02 volts dc*
Stability of Output for Zero Field:	±0.025 volts dc
Sensor Alignment:	±0.25 degree
Operating Temperature Range:	165°F to 0°F (±185°F to −50°F with reduced accuracy)
Linearity:	±3% of full scale
Frequency Response:	195 Hz (typical)
Output Impedance:	Less than 20K Ω
Output Load (for factory calibration):	100K Ω
Length of Sensor Cables:	5 feet for each electronics
Sensor Weight:	12 ounces including cable
Electronics Weight:	9 ounces total

*Standard units are biased as specified. An instrument, designated RAM-53C-2NB, is available with factory modification to provide 0-volt output for zero field.





SCHONSTEDT INSTRUMENT COMPANY
1775 WIEHLE AVENUE, RESTON, VIRGINIA 22090
Telephone: 703-471-1050



ORIGINAL PAGE
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ADCOLE CORPORATION 669 FOREST STREET MARLBOROUGH, MA 01752

SUN ANALOG SENSOR

PERFORMANCE SPECIFICATION

Prepared by J S Keel **Date** 5 Dec. 86

Approved by ______________ **Date** ______________

Project Engineer

Approved by ______________ **Date** ______________

Manager, Rel and QA

Customer C/PO Number ______________

Application — For Reference Only
C-1952

Revision Status

Sheet	1	2	3	4	5	6	7	8	9	10	11	12	13
Rev.													
Sheet	14	15											
Rev.													
Sheet													
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Rev.													
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Rev.													

ADCOLE CORPORATION 669 FOREST STREET MARLBOROUGH, MA 01752

Sun Analog Sensor,
Performance Specification

SIZE	**FSCM NO**	**DOC NO**	**REV**
A	**18150**	A28361	-

RELEASE DATE

SHEET 1 **OF** 15 **QF-11**

1.0 SCOPE

This specification establishes the design, performance and general test requirements for the Sun Analog Sensor (SAS).

2.0 APPLICABLE DOCUMENTS

The following documents of the issue noted form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

2.1 FACC (issue specified in this list or the subcontract if applicable)

HZ-565362	Sun Analog Sensor Interface Control Drawing
SH-572005	General Design and Test Requirements
SH-551535B	Subcontract Product Assurance Requirements
NHB 5300.4(3A)	NASA Handbook

Copies of documents required by subcontractors in connection with specific procurement functions should be obtained as directed by the FACC subcontract administrator.

3.0 REQUIREMENTS

3.1 General

The Sun Analog Sensor (SAS) is an analog single axis sensor which will be used in conjunction with the X-Ray Positioner.

3.2 Performance Characteristics

On-axis is defined as the axis which contains the optical head linear range. The off-axis is defined as the axis which is insensitive to angular change. The output characteristics shall be defined as the resultant of the transfer function $\left[\frac{I_A - I_B}{I_A + I_B}\right]$ per Para. 3.2.2.4. The equipment shall meet performance requirements as specified herein.

3.2.1 Primary Power Characteristics

The Sun Analog Sensor does not require primary power.

3.2.2 Optical Design

3.2.2.1 Sensor Coordinates

The coordinates for each optical head shall be in accordance with Figure 1 and 2. The sensor linear range shall be in the direction of angle alpha. The relationship between the coordinates of Figure 1 and the optical head physical configuration shall be in accordance with SCD-565362.

3.2.2.2 Field of View

a) Operational FOV

Operational FOV is defined to be when Data Good = 1. Data Good = 1 when $(I_A+I_B) \geqslant 50\mu A$

Data Good Region

On-Axis	$-4.3 \pm 1.0° < \alpha < +4.3 \pm 1.0°$
Off-Axis	$-11 \pm 1° < \beta < +11 + 1°$ (at $\alpha = 0°$)
	$-9 \pm 1.0 < \beta < +9 \pm 1.0$ (at $\alpha = \pm 4.3°$)

b) Unobstructed FOV (Stray Light Limits)

On-Axis	$-8° < \alpha < +8°$
Off-Axis	$-16° < \beta < +16°$

3.2.2.3 Outputs

There shall be two current outputs designated I_A and I_B.

3.2.2.4 On-Axis Characteristics

The on-axis (α) transfer characteristics of $\left\{\frac{I_A - I_B}{I_A + I_B}\right\}$ for FOV designed in 3.2.2.2 shall be in accordance with Figure 3.

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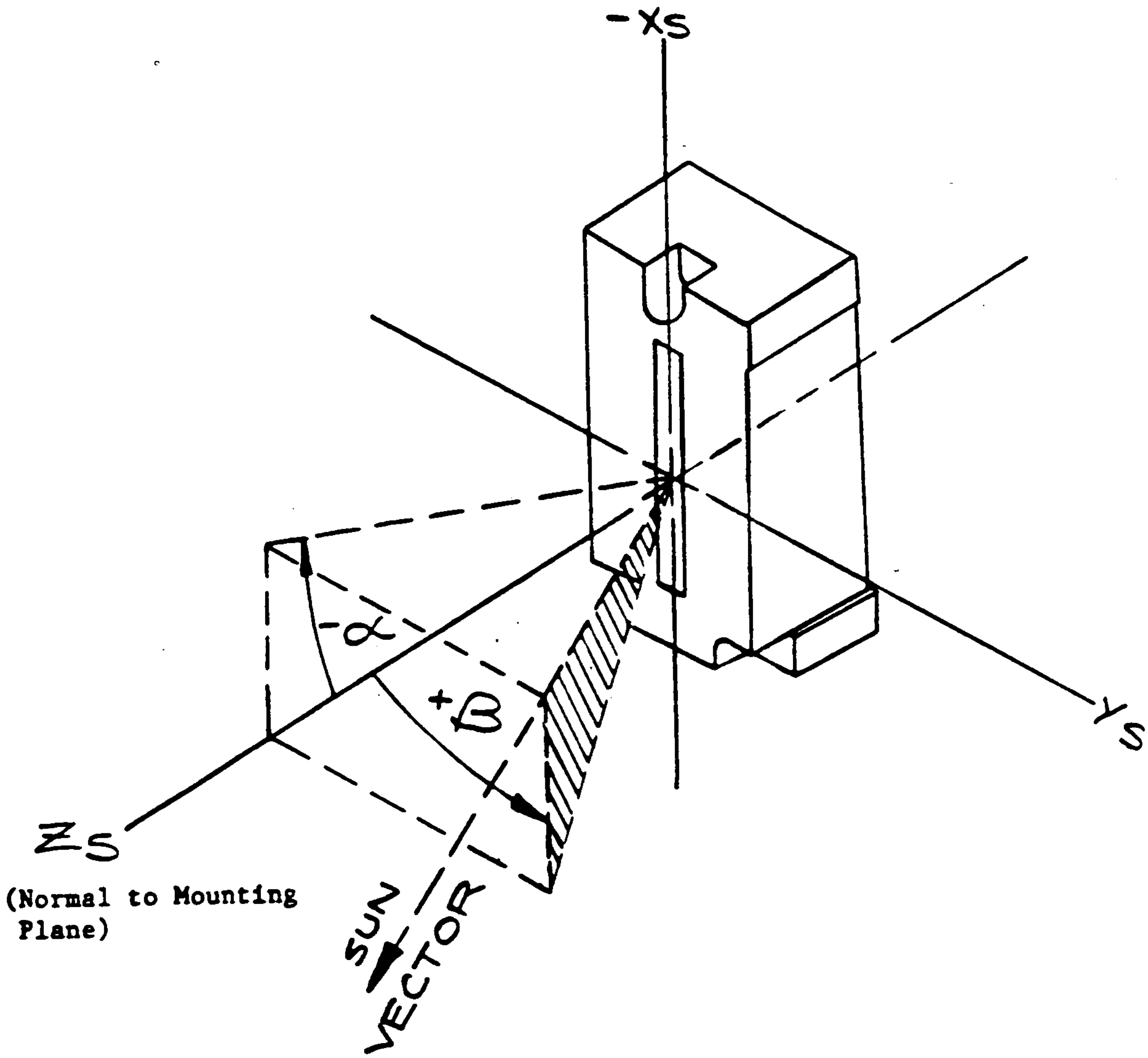

FIGURE 1

Coordinate System for Optical Head

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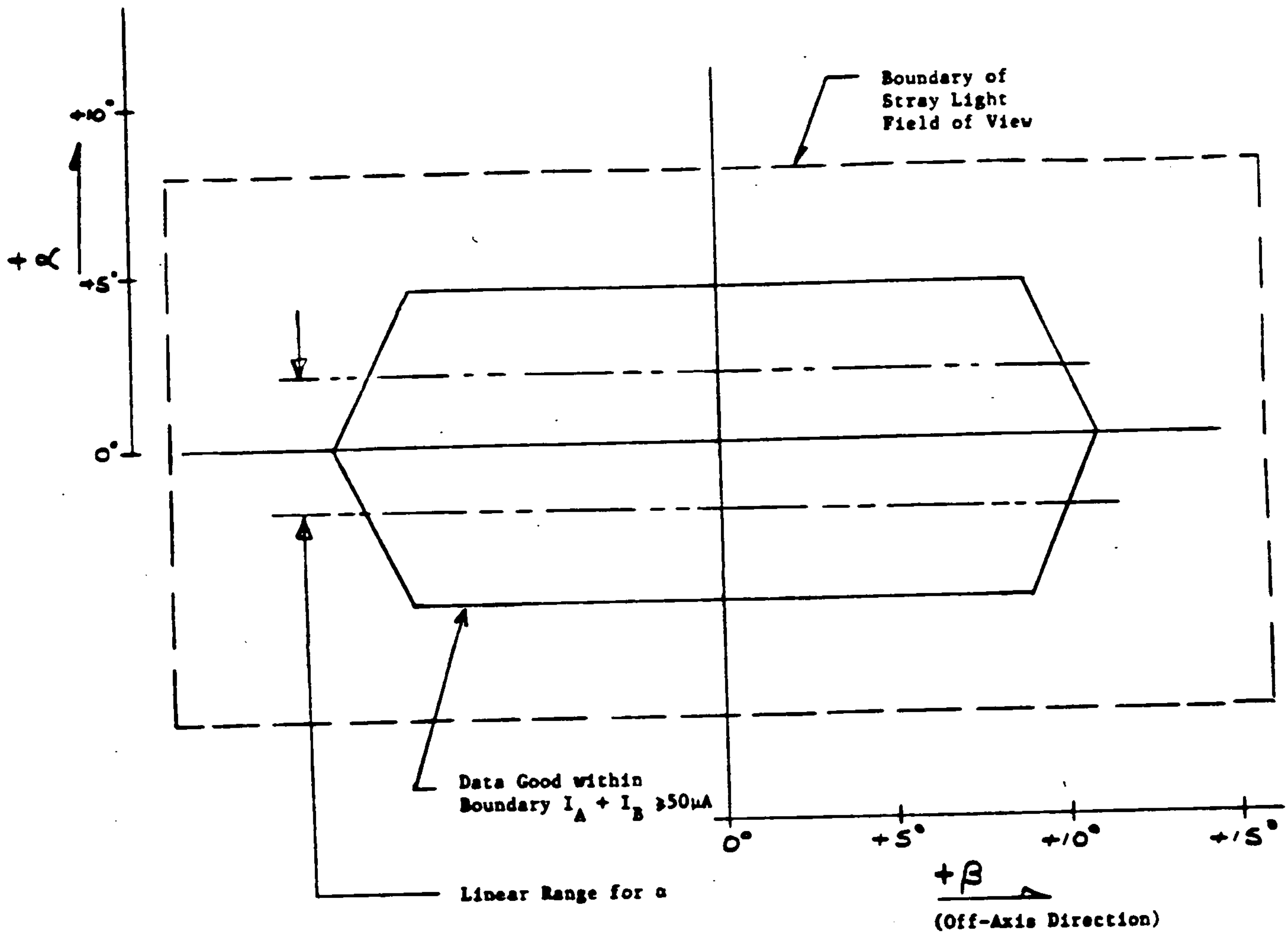

FIGURE 2

SAS Characteristics

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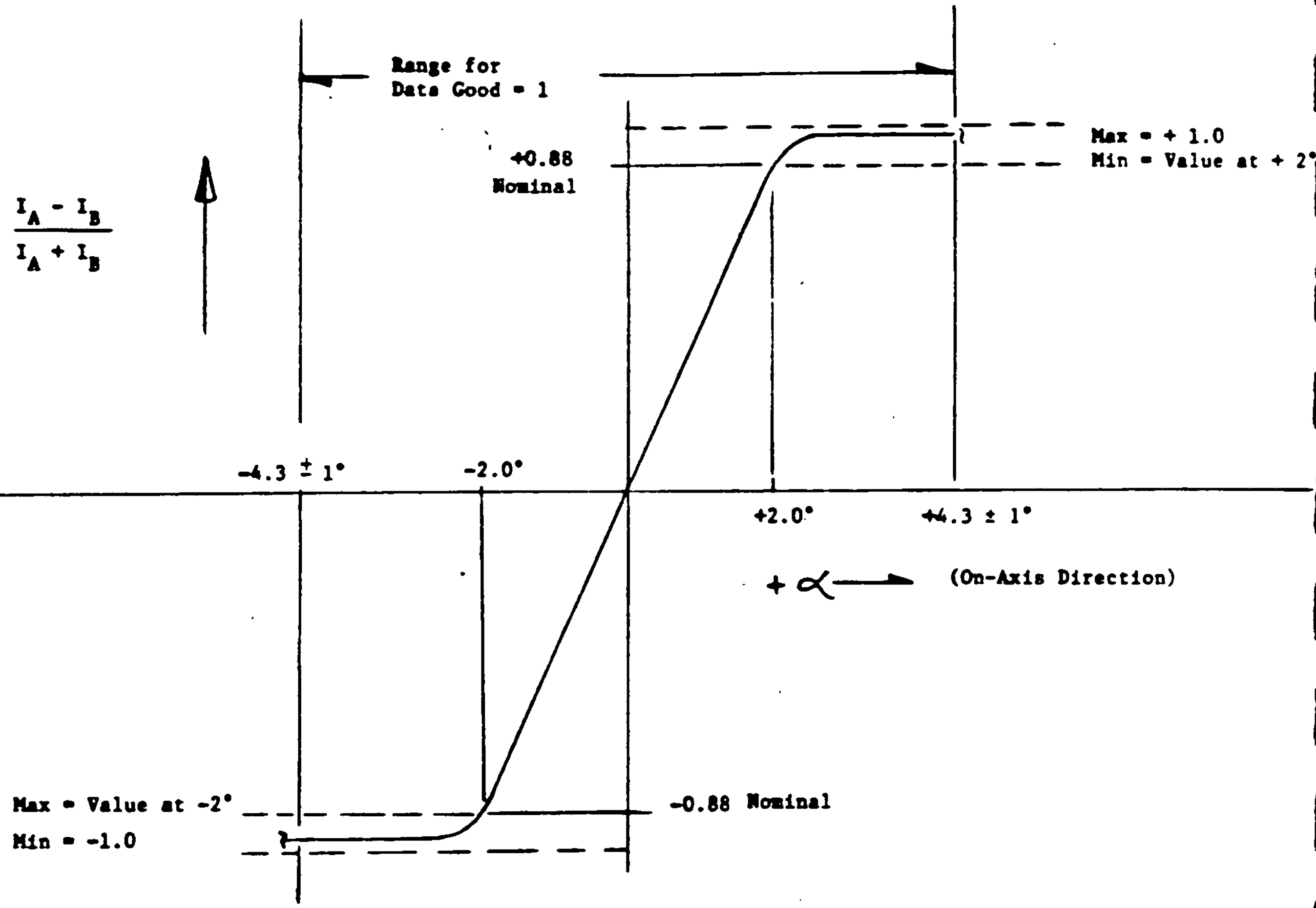

FIGURE 3

On-Axis transfer Function $\left[\frac{I_A - I_B}{I_A + I_B}\right]$

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3.2.2.4 On-Axis Characteristics (continued)

Additional on-axis characteristics are as follows:

a. Linear Range: ±2.0° minimum (reference Figure 3)

b. Saturation Range: From end of linear range to at least the end of Data Good indication. See Paragraph 3.2.2.7.

c. Linear Range Scale Factor: 0.44 ± 5% per degree (best fit straight line between ±2°).

d. Linearity: The deviation from the best fit straight line shall be equal or less than 10% of nominal full scale.

e. Zero Offset: The optical axis shall be perpendicular to the mounting surface within ±0.05° in the α, on axis plane at room temperature.

f. Zero Offset Over Operating Temperature: The zero offset shall not change from ambient condition by more than 0.05°.

g. Data Good: See Figures 2, 3 and Para.3.2.2.7.

3.2.2.5 Off-Axis Characteristics

The on-axis characteristics of paragraph 3.2.2.4 shall not change for $-8.0 < \beta < +8.0$ except for a maximum change of ±5% in on-axis linear range scale factor relative to the linear range scale factor at $\beta = 0°$.

NOTE: For $\beta > \pm 8.0°$ the linear range and sun presence may be less than in 3.2.2.4a and 3.2.2.2).

3.2.2.6 Output Polarity.

The sensor head output polarity shall be in accordance with Figure 1 and 3.

3.2.2.7 Data Good

Data good is defined to be = 1 when $(I_A + I_B) \geq 50\mu A$. Data good shall occur as specified in 3.2.2.2a).

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The Transfer Function $(I_A - I_B)/(I_A + I_B)$ shall meet all specification requirements when Data good = 1.

3.2.3 Electrical Characteristics

3.2.3.1 Grounds

The grounding system of the unit shall include two separate grounds:

a. Signal Ground

b. Chassis Ground

DC isolation between the two grounds shall be maintained: DC isolation shall be greater than 1 megohm, when measured at 50 volts DC.

a. Signal Ground

Signal Ground is used as the zero reference for the two current output signals, I_A and I_B.

3.2.3.2 Output Signals

A schematic of the SAS is shown in Figure 4. Positive current shall be defined in the direction of the arrows.

3.2.3.3 Output Level

$(I_A + I_B)$ maximum < 110μA.

3.2.3.4 Dark Leakage Current

The dark leakage current at a forward voltage of 10mV shall be less than 1μA at 65°C for each output.

3.2.3.5 Cell Output Signal

The interface parameters shall be as follows:

a. Output amplitude for the solar intensity at 1.0AU as used in 3.2.3.3.

b. Output impedance:

Resistance: as specified in 3.2.3.4 when cell voltage maintained below ±10mV.

Capacitance: 1700 pF nominal

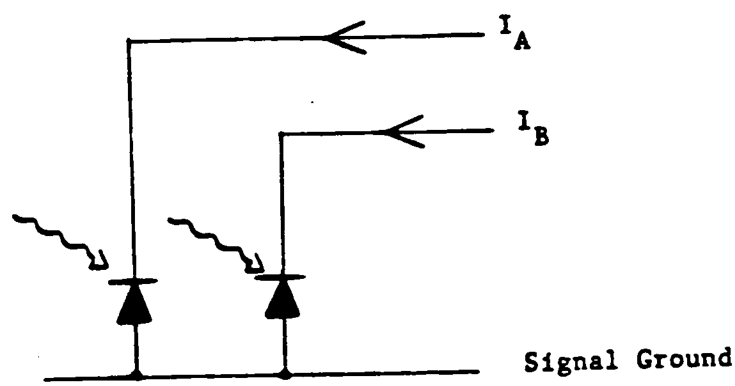

FIGURE 4
SAS Schematic

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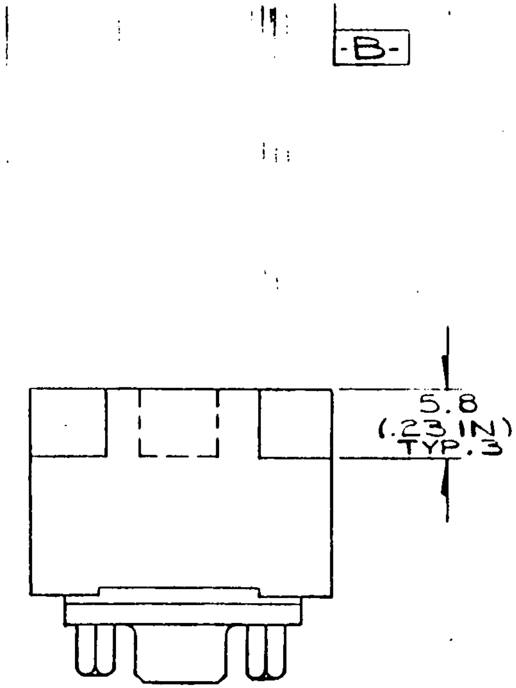
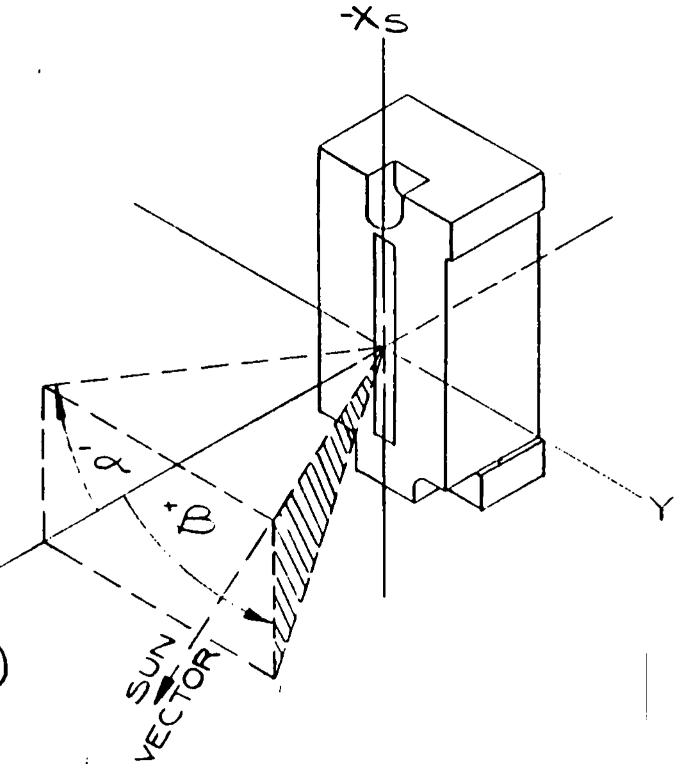
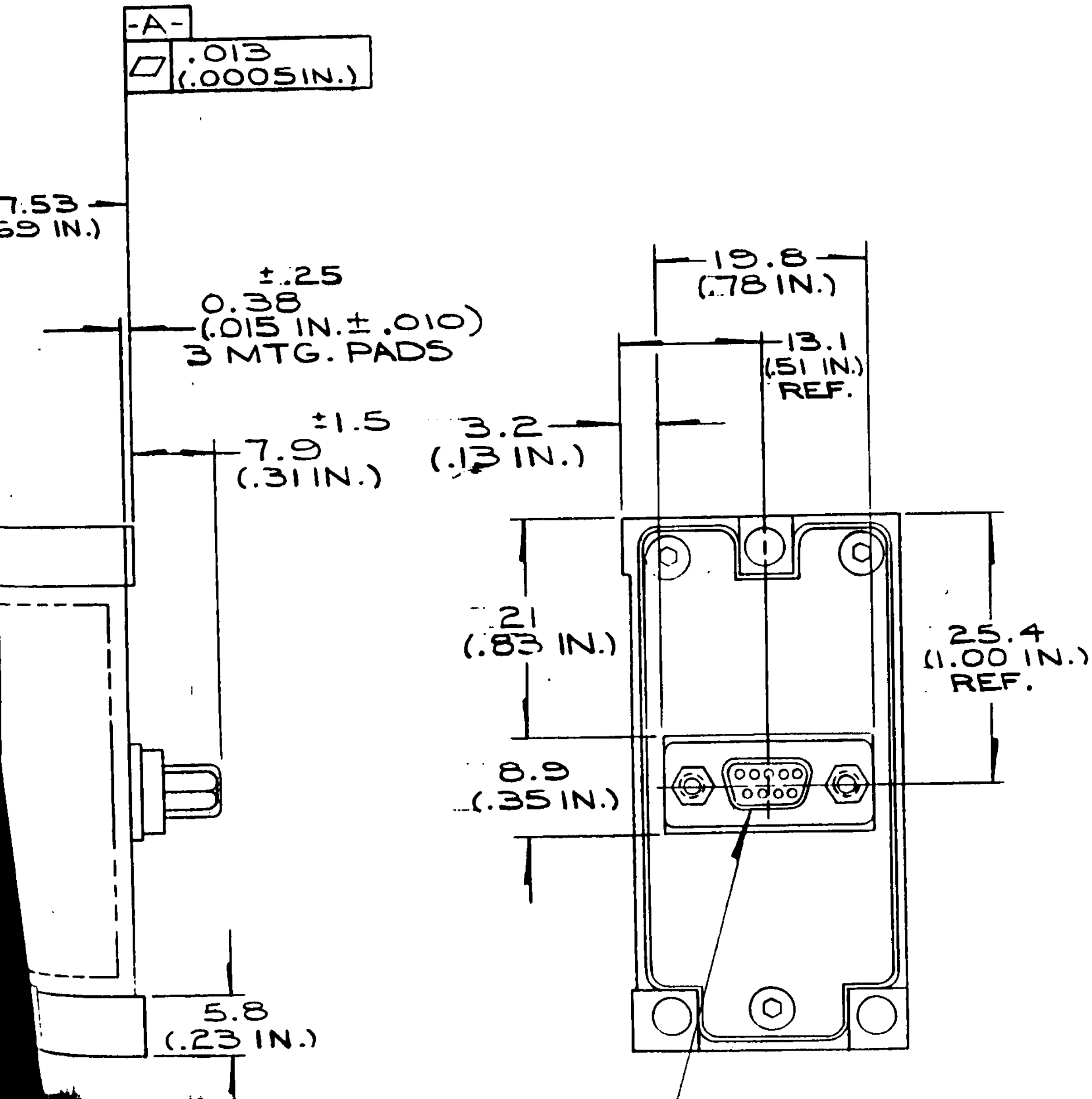


UNIT IDENTIFICATION
(BOTH SIDES)
SEE NOTE 5



COORDINATE REFERENCE SYSTEM

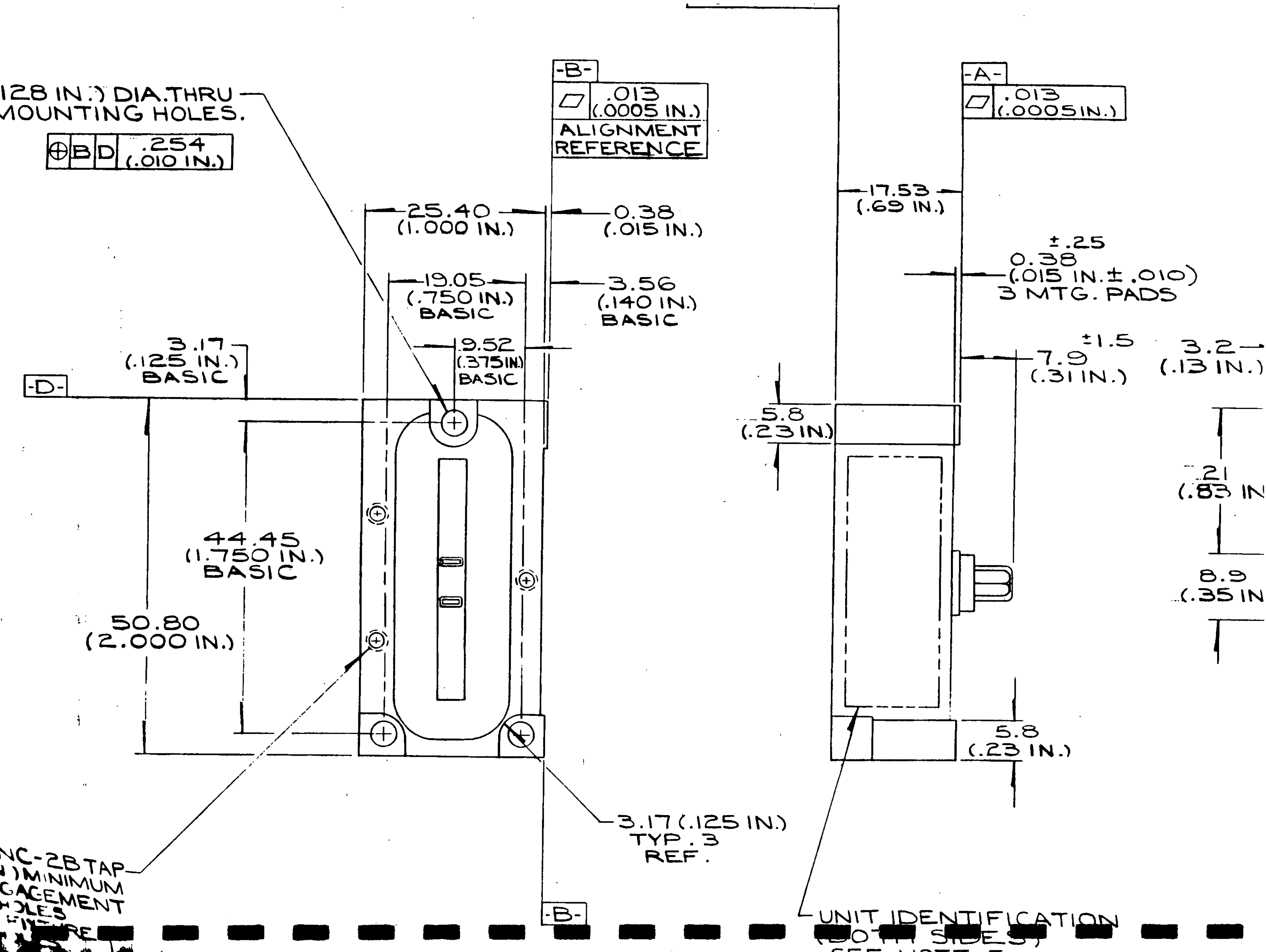


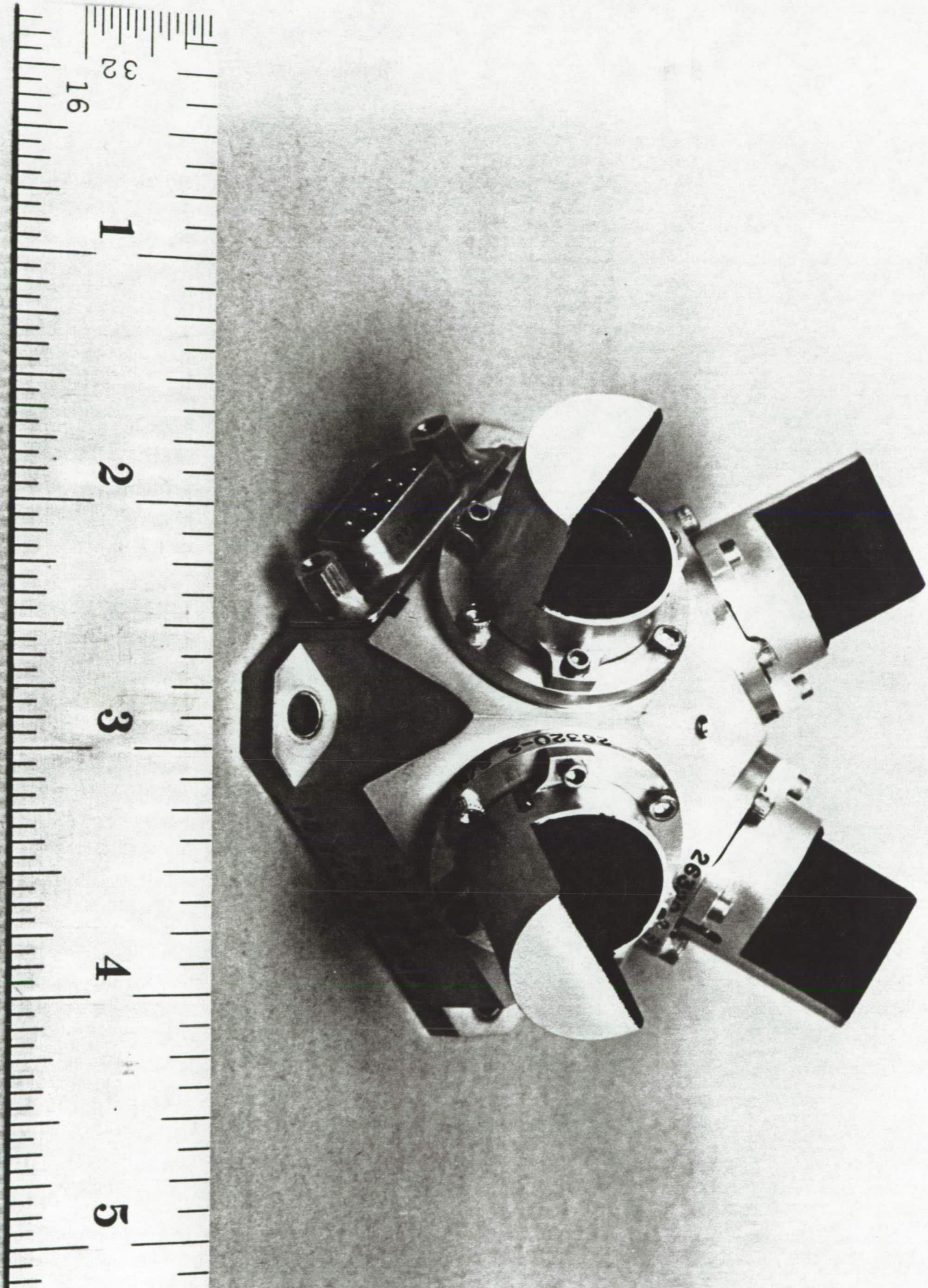
CONNECTOR
PIN FUNCTIONS

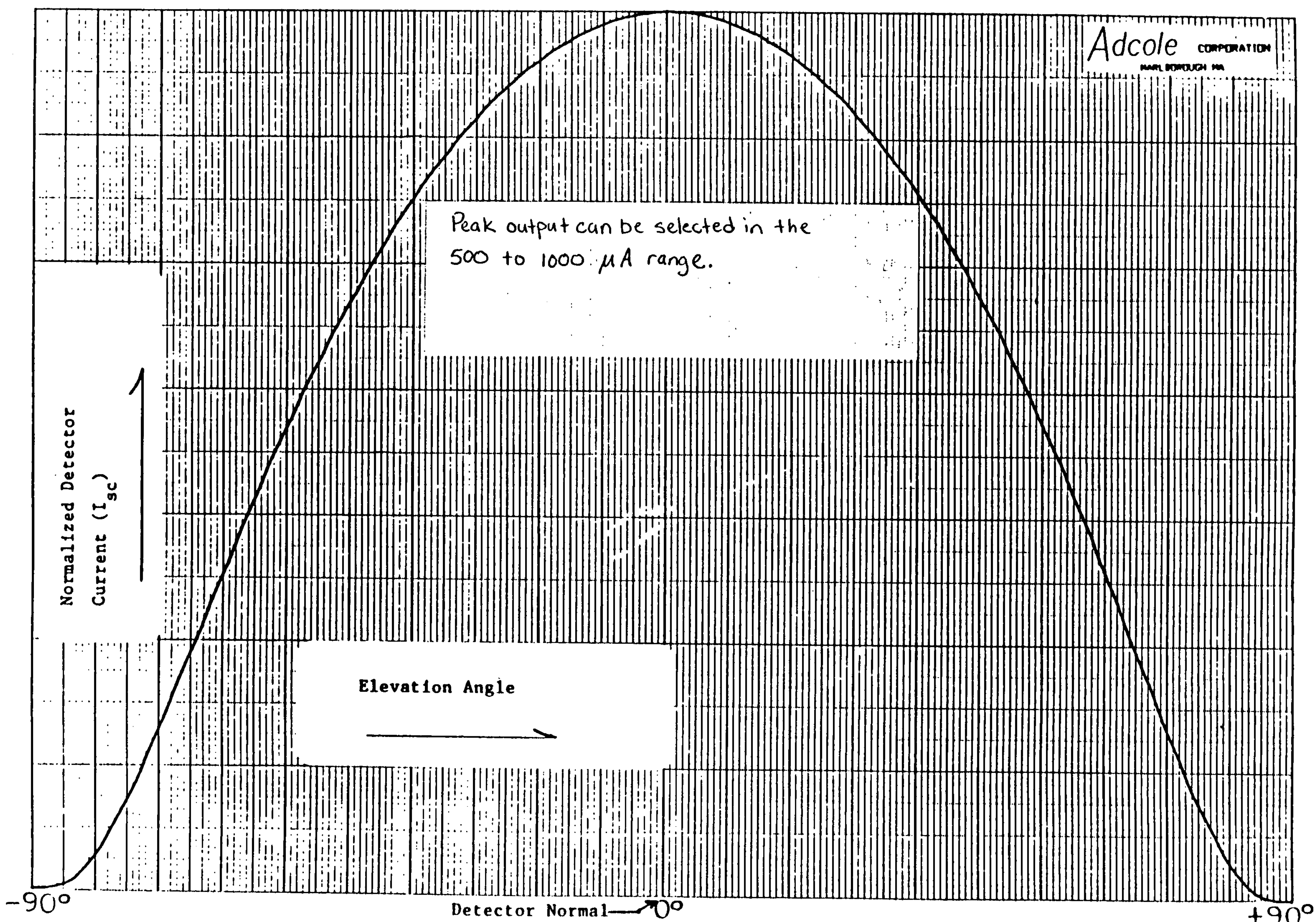
1	SIGNAL RETURN
2	SPARE
3	SPARE
4	SPARE
5	PHOTOCELL "A"
6	PHOTOCELL "B"
7	SPARE
8	SPARE
9	CHASSIS GND.

NOTES:

1. MATERIAL; HOUSING
   T651 PER FEDERAL
2. FINISH; HOUSING: C
   CLASS 1A.
3. UNIT WEIGHT: 75 C

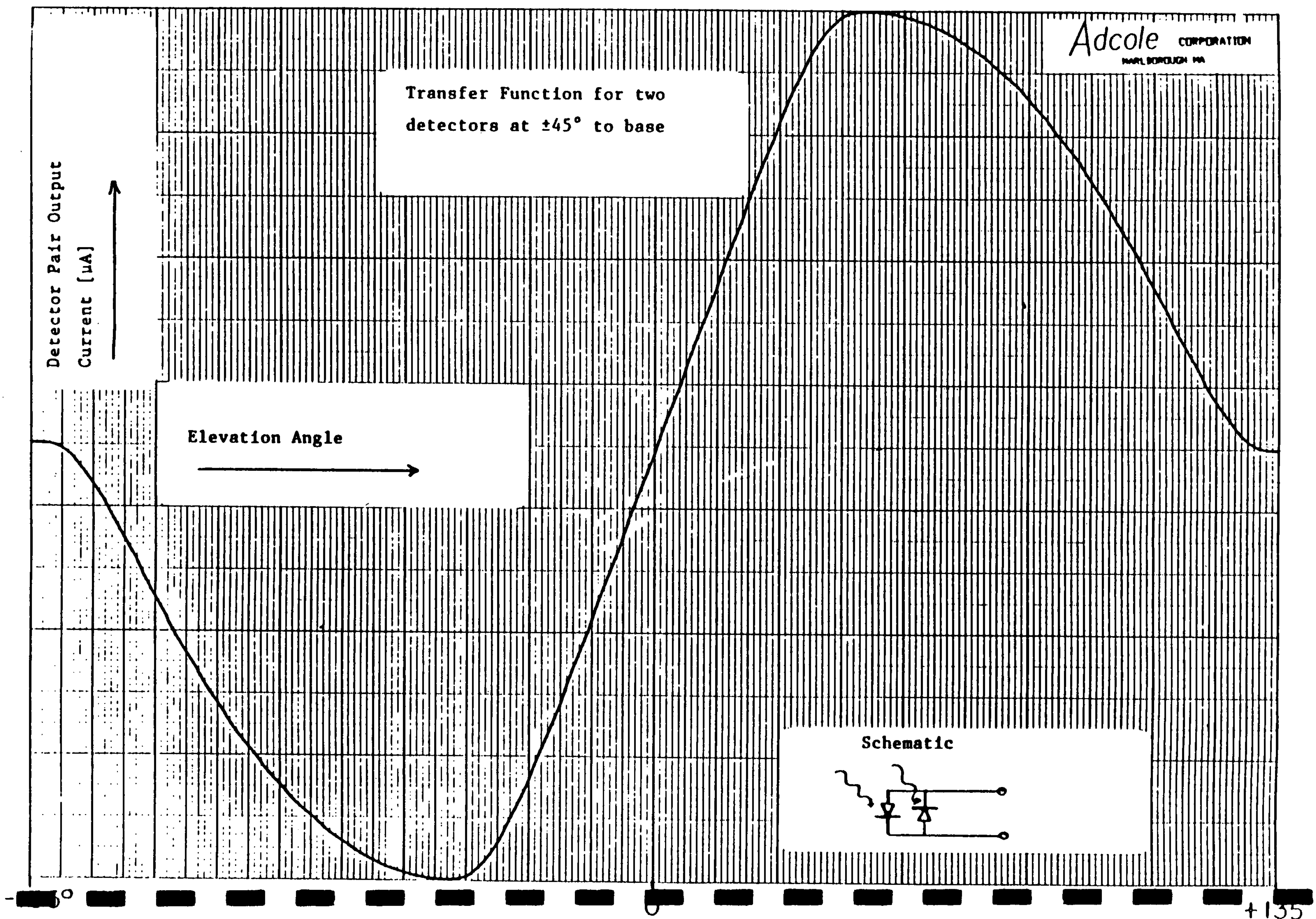

128 IN.) DIA.THRU
MOUNTING HOLES.
⊕ B D .254 (.010 IN.)
-B-
.013 (.0005 IN.)
ALIGNMENT REFERENCE
-A-
.013 (.0005 IN.)
25.40 (1.000 IN.)
0.38 (.015 IN.)
19.05 (.750 IN.) BASIC
3.56 (.140 IN.) BASIC
9.52 (.375 IN.) BASIC
3.17 (.125 IN.) BASIC
-D-
17.53 (.69 IN.)
±.25 0.38 (.015 IN. ±.010) 3 MTG. PADS
±1.5 7.9 (.31 IN.)
3.2 (.13 IN.)
5.8 (.23 IN.)
.21 (.83 IN
8.9 (.35 IN
44.45 (1.750 IN.) BASIC
50.80 (2.000 IN.)
5.8 (.23 IN.)
3.17 (.125 IN.) TYP. 3 REF.
NC-2B TAP
MINIMUM
GAGEMENT
-B-
UNIT IDENTIFICATION




Adcole CORPORATION
MARLBOROUGH MA
Peak output can be selected in the 500 to 1000 μA range.
Normalized Detector Current (I_{sc})
Elevation Angle
-90°
Detector Normal
0°
+90°

K·E 20 X 20 TO THE INCH • 7 X 10 INCHES KEUFFEL & ESSER CO. MADE IN U.S.A.

46 1240



PROFESSIONAL VERSION OF TUTSIM

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Timing:    0.5000000   ,DELTA   ;    3.000E+03   ,RANGE
PlotBlocks and Scales:
Format:
      BlockNo,  Plot-MINimum,   Plot-MAXimum;   Comment
Horz:       0 ,   0.0000      ,   3.000E+03   ; Time
  Y1:      10 ,  -0.0131000   ,   0.0131000   ; psi
  Y2:      26 ,  -0.0131000   ,   0.0131000   ; phi
  Y3:      18 , -20.000E-06   ,  20.000E-06   ;
  Y4:      34 , -20.000E-06   ,  20.000E-06   ;


  0.0062070         1 CON                              ;I3R
 16.2000            2 CON                              ;I1
183.2600            3 GAI      1                       ;omegaR
  6.1111            4 CON                              ;I2
                    5 DIV      3       2               ;omR/I1
                    6 DIV      3       4               ;omR/I2
                    7 MUL      5      21      25
                    8 SUM      7      34               ;psidd
  0.0000            9 INT      8                       ;psid
 -0.0174500        10 INT      9                       ;psi
500.000E-06        11 GAI      9                       ;kv
200.000E-06        12 GAI     10                       ;kp
                   13 SUM     11      12
                   14 SUM    -35      19
                   15 SUM     35     -20
  1.0000           16 SRS     15      14
                   17 MUL     13      16      37
 -5.555E-06        18 LIM     17
  5.555E-06
  0.0034907        19 CON
  0.0069813        20 CON
 -1.0000           21 CON
  0.0000           22 CON
                   23 MUL      6       9
                   24 SUM     23      18               ;phidd
  0.0000           25 INT     24                       ;phid
  0.0174500        26 INT     25                       ;phi
500.000E-06        27 GAI     25                       ;kv
200.000E-06        28 GAI     26                       ;kp
                   29 SUM    -27     -28
                   30 SUM    -36      19
                   31 SUM     36     -20
  1.0000           32 SRS     31      30
                   33 MUL     29      32
-14.700E-06        34 LIM     33
 14.700E-06
                   35 ABS     10
                   36 ABS     26
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  0.0000           39 CON
  0.0000           40 CON
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APPENDIX D

STRUCTURES

```
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xz          Ixz
=====================================================================
=====================================================================
=================
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0E+01  -4.13400E+01
  2     1  battery                         +1.41200E+01  +1.15000E+01  +1.30000E+0
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0E+02  -4.87140E+02
  3     1  torque1                         +1.00000E+00  +8.00000E+00  +8.00000E+0
0  +5.00000E+00  +8.90817E+01  +1.10374E+02  +1.49374E+02  -6.40000E+01  -4.0000
0E+01  -4.00000E+01
  4     1  torque2                         +1.00000E+00  +1.60000E+01  +8.00000E+0
0  +6.50000E+00  +1.27624E+02  +2.98332E+02  +3.41374E+02  -1.28000E+02  -5.2000
0E+01  -1.04000E+02
  5     1  torque3                         +1.00000E+00  +1.70000E+01  +1.40000E+0
1  +4.00000E+00  +2.33374E+02  +3.26374E+02  +4.95082E+02  -2.38000E+02  -5.6000
0E+01  -6.80000E+01
  6     1  base                            +6.27200E+00  +8.00000E+00  +8.00000E+0
0  +1.25000E-01  +5.35341E+02  +5.35341E+02  +1.07042E+03  -4.01408E+02  -6.2720
0E+00  -6.27200E+00
  7     1  sides                           +1.75000E-01  +8.00000E+00  +8.00000E+0
0  +3.50000E+00  +1.33729E+01  +1.33729E+01  +2.24292E+01  -1.12000E+01  -4.9000
0E+00  -4.90000E+00
  8     1  top                             +1.00700E-01  +8.00000E+00  +8.00000E+0
0  +7.00000E+00  +1.35274E+01  +1.35274E+01  +1.71861E+01  -6.44480E+00  -5.6392
0E+00  -5.63920E+00
  9     1  bracket                         +1.12800E-01  -5.00000E+00  +8.00000E+0
0  +4.00000E+00  +1.03776E+01  +5.97840E+00  +1.27464E+01  +4.51200E+00  -3.6096
0E+00  +2.25600E+00
 10     1  heatpipe                        +8.00000E+00  -3.20000E+01  +8.00000E+0
0  +3.20000E+01  +8.71631E+03  +1.63873E+04  +8.71631E+03  +2.04800E+03  -2.0480
0E+03  +8.19200E+03
 11     1  arm                             +2.00000E-01  -1.00000E+01  +8.00000E+0
0  +1.60000E+01  +8.68177E+01  +9.40177E+01  +3.28021E+01  +1.60000E+01  -2.5600
0E+01  +3.20000E+01
 12     4  earthsensor                     +1.14000E+01  +1.20000E+01  +4.00000E+0
0  +3.50000E+00  +4.14200E+02  +1.87340E+03  +1.91520E+03  -5.47200E+02  -1.5960
0E+02  -4.78800E+02
 13     1  power                           +4.00000E+00  +1.30000E+01  +1.00000E+0
1  +2.00000E+00  +4.21667E+02  +7.09333E+02  +1.08833E+03  -5.20000E+02  -8.0000
0E+01  -1.04000E+02
 14     1  magneticsen                     +2.25000E+00  +3.50000E+00  +6.50000E+0
0  +4.50000E+00  +1.43813E+02  +8.60633E+02  +9.12945E+02  -5.11875E+01  -6.5812
5E+01  -3.54375E+01
 15     1  electrons                       +5.62500E-01  +1.70000E+00  +1.30000E+0
1  +1.00000E+00  -9.71484E+01  +2.86781E+00  +9.86803E+01  -1.24313E+01  -7.3125
0E+00  -9.56250E-01
 16     1  rxtx                            +4.00000E+00  +3.00000E+00  +7.50000E+0
0  +2.00000E+00  -2.51667E+02  +6.93333E+01  +2.78333E+02  -9.00000E+01  -6.0000
0E+01  -2.40000E+01
 17     1  mirror                          +2.50000E+01  +3.20000E+01  +8.00000E+0
```

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OF POOR QUALITY

```
+... ....E+.. +..85..E+.3 -2.61204E-04 -..57417E-.4 +...... ..E+.. +.. ..
E-02 -2.....E+.6
================================================================================
================================================================================
===================
                 Overall Values:         +8.51930E-01  +1.00019E+01  +8.05370E+0
. +3.36950E-00 -2.08605E+04 +5.06996E-04 -5.53634E-04 +8.55406E-03 -2.6080
.E-03 +9.00577E+03
================================================================================
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Electrically Initiated, Single-Function Reefing Line Cutters

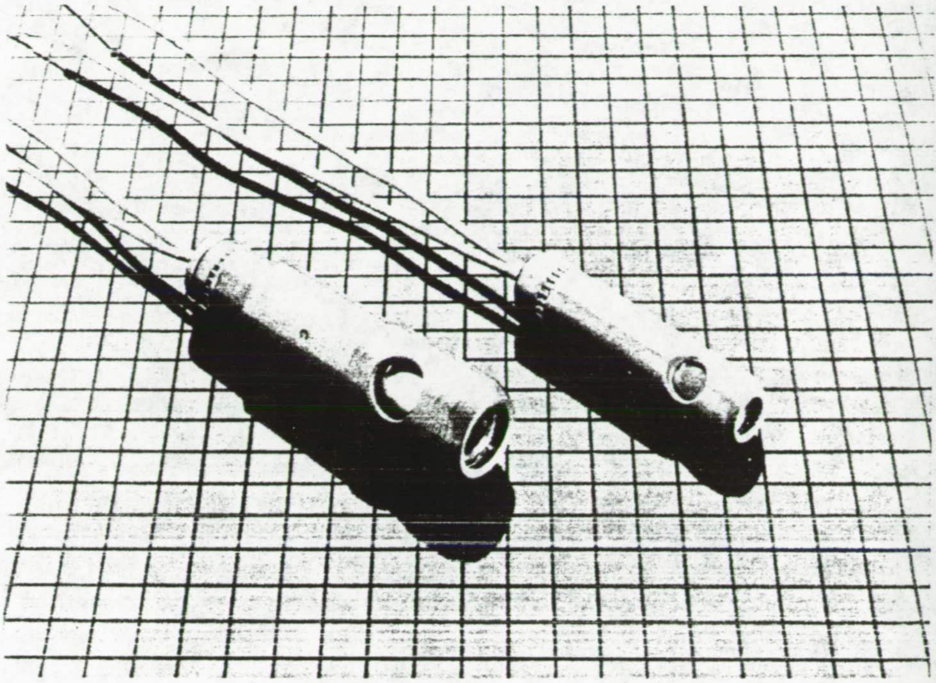
Electrically initiated reefing line cutters are specifically designed as single-function (not reusable) devices for parting nylon reefing lines in aerospace applications.

These cutters deliver a very high level of energy, and are extremely compact and lightweight in comparison with other types of cutting devices. All are characterized by high reliability.

In operation, a small amount of electric current initiates a contained pyrotechnic charge. This generates high-pressure gas which drives a cutting blade through the reefing line.

Variations

The 1SE166 and 1SE167 shown are designed to cut 750 lb and 2000 lb nylon lines respectively. Electrically initiated cutters with a variety of cutting strengths and electrical sensitivities are available or can be made. ICI Aerospace will also design and manufacture cutters to meet customer requirements.



Grid Scale: Actual Size = ¼″.
1SE167, left. 1SE166, right.
Both are shown after functioning.

Characteristics

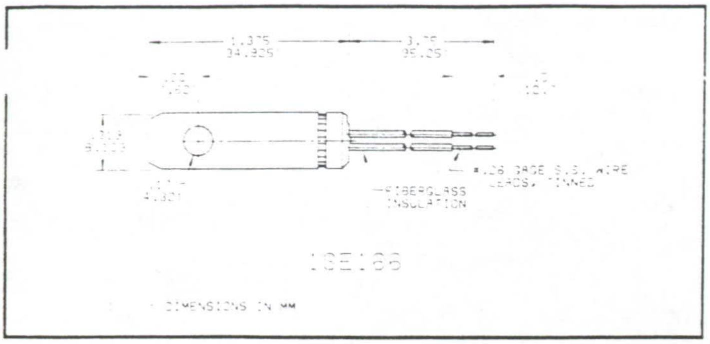
Some of the characteristics listed here are nominal; others are levels to which the units have been tested. They are not limits on design capabilities. Please consult an ICI Aerospace representative before using this data as a specification.

All characteristics are for both 1SE166 and 1SE167 cutters unless otherwise indicated.

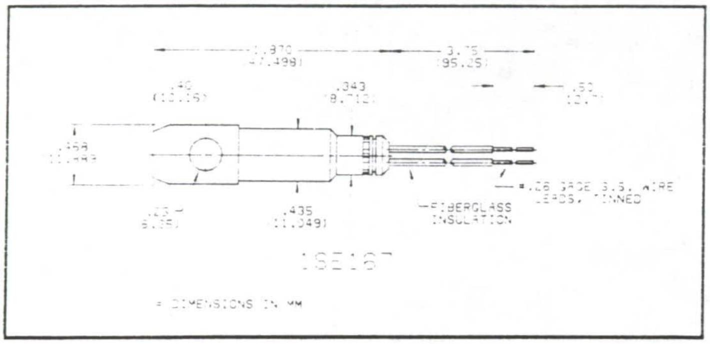
Electrical

Insulation resistance @ 325 Vdc, leads to case:
50 megohm

Squib	Bridge resistance @ 70°F (21°C)	All-fire current @ -65°F (-54°C) 25 ms	No-fire current @ 160°F (71°C) 5 min
Type	Ohm	Amp	Amp
B	4.0 – 5.0	0.550	0.1
C	1.6 – 2.0	1.0	0.1
E	1.3 – 1.5	2.0	0.5



1SE166



1SE167

Mechanical

Body material:
Aluminum

Size:
See drawings

Weight, Approx.:
1SE166: 14 gm
1SE167: 22 gm

Cutting capability:
1SE166: 750 lb line per MIL-C-7515, Type III or 1000 lb line per MIL-C-7515, Type IV

1SE167: 2000 lb line per MIL-C-7515, Type VI

Function time:
Less than 15 ms

Mounting bracket for 1SE167:
Order 3SE244

Environmental

Temperature:
Operating range: -65° to +160°F (-54° to +71°C)

Acceleration:
MIL-STD-810B, Method 513

Dust:
MIL-STD-810B, Method 510

Salt spray:
MIL-STD-810B, Method 509

Shock:
MIL-STD-810B, Method 516

Vibration:
MIL-STD-810B, Method 514

Chemical

Ignition compound:
Lead styphnate

Freight Classification

Shipping name:
Explosive cable cutter

Hazard classification:
Class C

Safety

Maximum pyrotechnic weight:
80 mg

Warning:
Reefing line cutters are self-contained when functioned under normal operating and testing conditions. If they are exposed to temperatures above 165°F (74°C), an electrical charge exceeding the specified no-fire current, or cut open before functioning, the cutters may fire and rupture with considerable force.

If your company does not have a safety program, it is essential that one is established before explosive items are handled or used. For a brief overview of safety precautions, see Data Sheet A103.

ICI Aerospace typically includes a Material Safety Data Sheet (MSDS) with the first shipment of every potentially hazardous product. For more information, contact your ICI Aerospace representative.

ICI Aerospace
P.O. Box 819
Valley Forge, PA 19482
(215) 666-8636
(800) 523-1763

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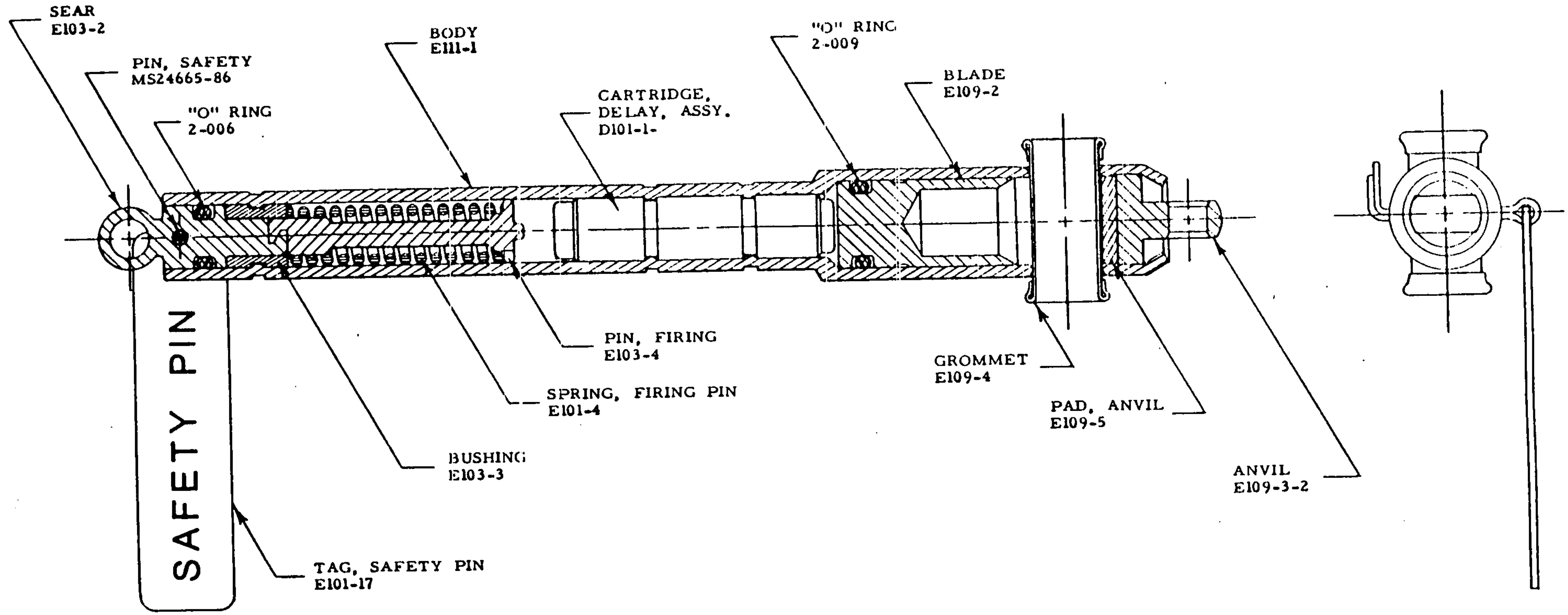
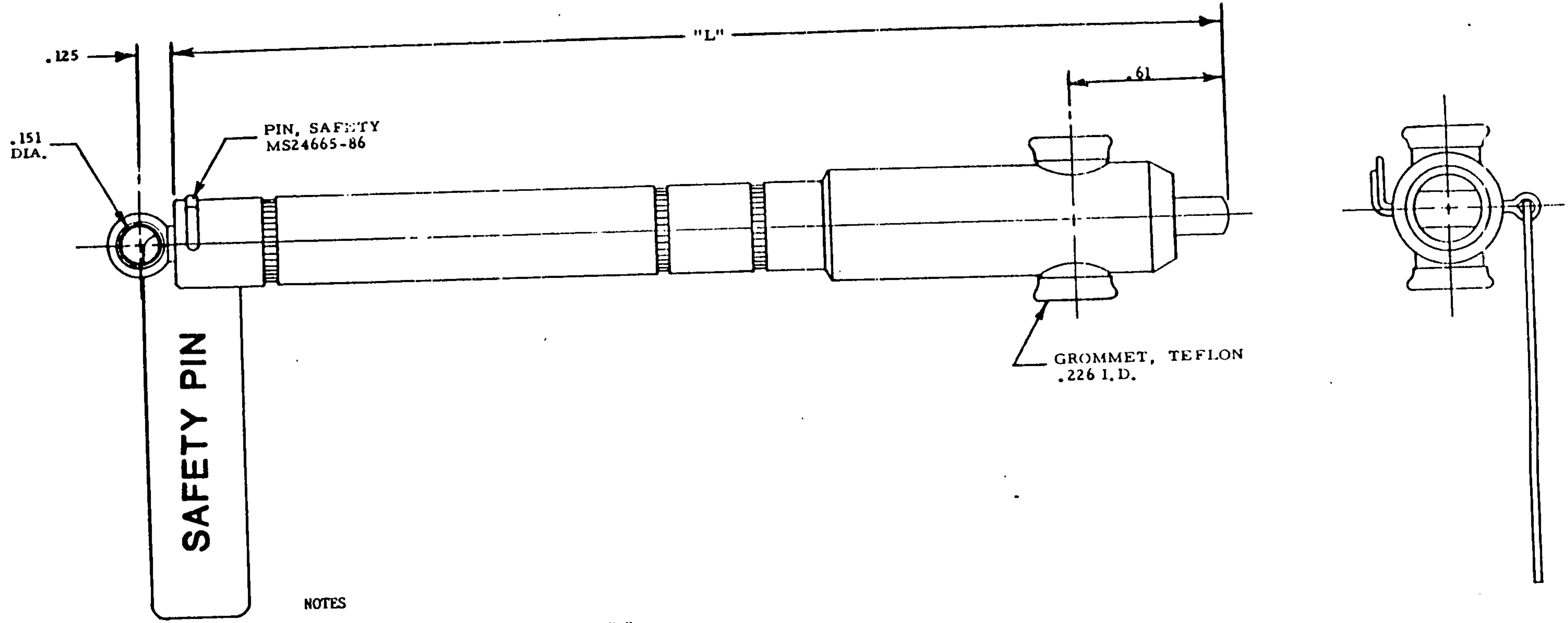
SPECIALIZED EXPLOSIVE & PYROTECHNIC DEVICES

FOR THE AIRCRAFT AND MISSILE FIELD



DRAWING TOLERANCES UNLESS OTHERWISE SPECIFIED DEC. ± ____ ANGLES ± ____ DEG. CONCENTRICITY ____ T.I.R. FINISH ____ R.M.S.	ROBERTS RESEARCH LABORATORY TORRANCE, CALIFORNIA DRAWN BY: acr — SCALE: 10/4 APPROVED BY: acr — DATE: 8/7/79
CUTTER, CARTRIDGE ACTUATED, MODEL C3 - xx SEC.	DRAWING NUMBER C111-1



NOTES

1. TIME DELAY / WEIGHT / "L"

TIME DELAY SEC.	WEIGHT OZ.	"L" IN.
0 - 6	0.97	4.20
7 - 12	1.04	4.60
13 - 18	1.11	5.00
19 - 24	1.19	5.40

2. INITIATION: 35 ± 20 lb. PULL IN 60° CONE ANGLE.

3. CUTTING CAPACITY: 2,250 lb. NYLON LINE.

4. TEMPERATURE RANGE: -65° F. TO +200° F.

5. ALTITUDE: NO LIMIT.

6. EXTERIOR: ALUMINUM, ANODIZED & DYED RED.

7. IDENTIFICATION: CUTTER, CARTRIDGE ACTUATED,
MODEL C3 - xx SEC.
LOT NO. (INCLUDES LOADING DATE)
ROBERTS RESEARCH LABORATORY
MFR. CODE 60880

DRAWING TOLERANCES UNLESS OTHERWISE SPECIFIED	ROBERTS RESEARCH LABORATORY 20202 S. NORMANDIE AVE. TORRANCE, CALIFORNIA 90502	
DEC. ±	DRAWN BY: aae	SCALE: 10/4
ANGLES ± ___ DEG. CONCENTRICITY ___ T.I.R. FINISH ___ R.M.S.	APPROVED BY: aa?	DATE: 8/12/79
CUTTER, CARTRIDGE ACTUATED, MODEL C3 - xx SEC.		DRAWING NUMBER AIII-1

APPENDIX E

DATA MANAGEMENT

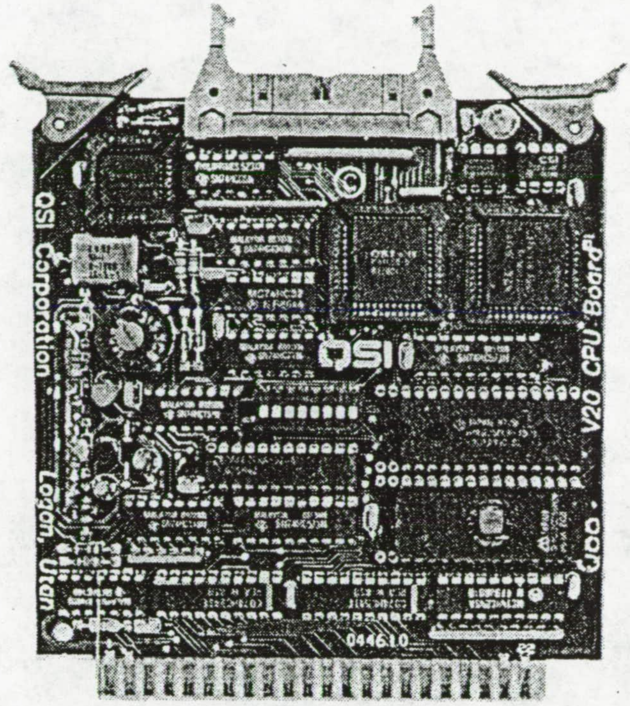
QSI CORPORATION

Q88

V20 (8088) CPU BOARD
FOR THE C-44 BUS

FEATURES

- NEC V20 CPU running at 4.9152 MHz (enhanced 8088-compatible CPU).
- 32 Kbytes EPROM and 32 Kbytes RAM installed; each expandable to 128K.
- 128 bytes of EEPROM.
- Real-time clock (RTC) with 256 bytes of RAM.
- Powerful monitor for easy program development.
- Pre-written BIOS routines for on-board hardware.
- Can access up to 1 Mbyte of off-board memory.
- 22 digital input/output lines.
- Hardware UART, 150 baud to 38,400 baud.
- Four off-board and three on-board maskable interrupts; one non-maskable interrupt.
- Two programmable timers and one programmable alarm.
- Low power consumption: 270 milliwatts while running and 24 milliwatts when powered down.
- Hardware single-stepping allows debugging in ROM.
- Enhanced 8088 instruction set includes fast multiply and divide, string move/compare and BCD string arithmetic, and 8080/8085-compatible mode.



The QSI Corporation Q88 CPU board is a complete, stand-alone computer on a single C-44 board. It can be used with just one unregulated power supply to form a powerful computing system with RAM, EPROM, EEPROM, a RTC, two timers, an alarm, a serial communications port, and 22 digital Input/Output (I/O) lines (Figure 1). It is fully compatible with the C-44 bus mechanical and electrical specifications.

V20 CPU

The NEC V20 CPU was chosen over the 80C88 for this CPU board because it offers greater execution speed and versatility. The integer multiply and divide instructions average about four times faster execution than the 8088 and overall throughout is about 30% faster. The V20 also contains an enhanced instruction set and an 8080 mode. Using sophisticated clock control circuitry, the power consumption of the V20 (which is not a static part) is as low as the 80C88.

MEMORY AND I/O

The Q88 comes with a 32 Kbyte static RAM and a 32 Kbyte EPROM installed. Either or both of these memory ICs can be replaced by 128 Kbyte devices, making a total of 256K of on-board memory possible. Figure 2 shows the memory map with both 32K and 128K parts.

The Q88 is also available with on-board memory disabled, allowing a contiguous 1 Mbyte of off-board memory.

Figure 2 also shows the Q88 I/O map. The Q88 uses the first 32 (20H) of 256 available I/O addresses for communicating with on-board peripherals. The remaining 224 (E0H) addresses are available for off-board peripherals.

82C52 UART

The serial data lines of the 82C52 UART are accessed through the 26-pin I/O connector. The UART operates at baud rates from 150 to 38,400 baud. The hardware UART supports full interrupt-driven I/O.

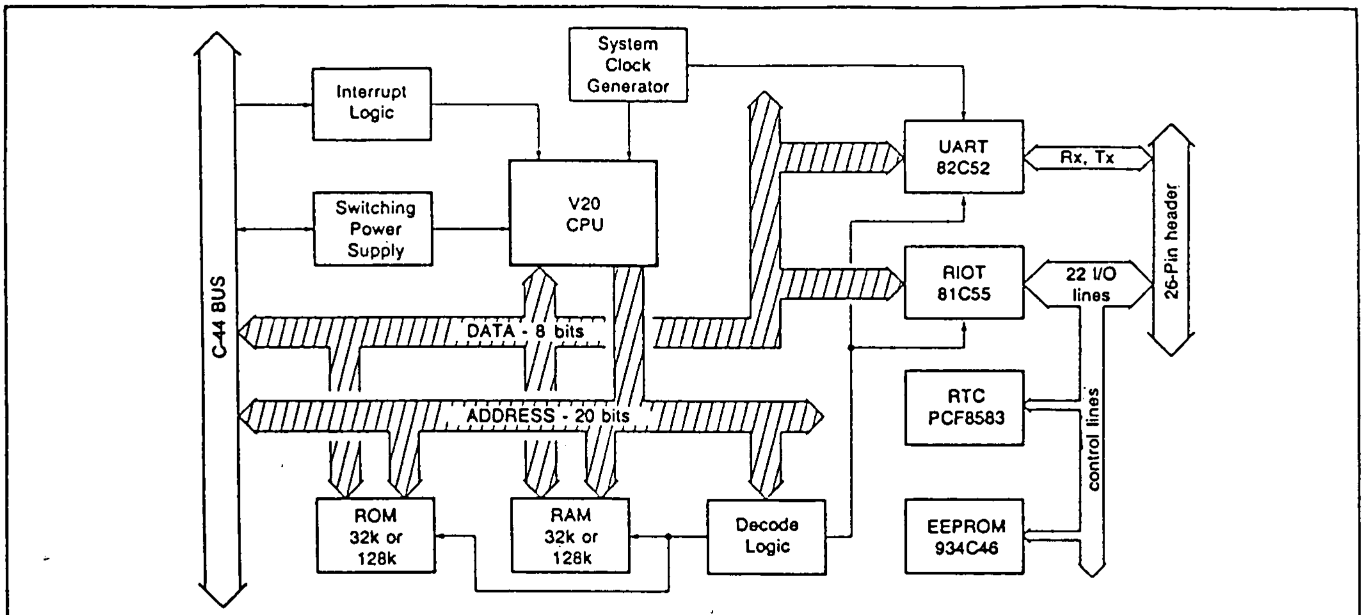


Figure 1. Q88 CPU board block diagram.

81C55 INPUT/OUTPUT/TIMER

The 81C55 I/O/Timer IC provides 22 digital input/output lines in three ports (A, B and C). Seven of these lines have optional on-board uses: the EEPROM uses three lines and the RTC uses four lines. If the EEPROM or RTC is not needed, the corresponding I/O lines are available for other uses.

The 81C55 also provides a 14-bit programmable timer, which can be used to provide interrupts at intervals from 208 microseconds to 3.4 seconds.

REAL-TIME CLOCK

The PCF8583 real-time clock can operate as a simple polled device to retrieve the time, or it can be used to generate two different types of interrupts: timed and dated. The PCF8583 includes an 8-bit timer which can issue interrupts in multiples of hundredths of seconds to days. The alarm function issues an interrupt when a user programmable mask matches the current date and time. The RTC also contains 256 bytes of static RAM.

A small battery can be used to maintain the RTC timekeeping operation and internal RAM. See the *OPTIONS* section for more information.

EEPROM

The Q88 has an EEPROM that can store up to 128 bytes of non-volatile data. This data remains intact even if power to the Q88 is turned off.

LOW-POWER MODES

Two operational modes are available to lower the Q88 power drain below the 270 milliwatt active level. In the WAIT mode the CPU stops all activity, lowering the typical power consumption to about 50 milliwatts.

In the STOP mode the CPU stops all activity and lowers the regulated voltage to 3 volts. This allows the on-board RAM to retain data and the RTC to operate, but reduces power drain to 24 milliwatts.

The Q88 can exit either low-power mode by any of the available interrupts, the WAKE line, or by a system reset.

MONITOR SOFTWARE

A monitor is included in the on-board EPROM of the Q88 board. This monitor communicates through the serial data lines on the I/O connector. These serial data lines can be connected to a terminal (through a level converter such as the QSI Q232A or QCOM-1), creating a stand-alone computer that allows a user to do complex system development.

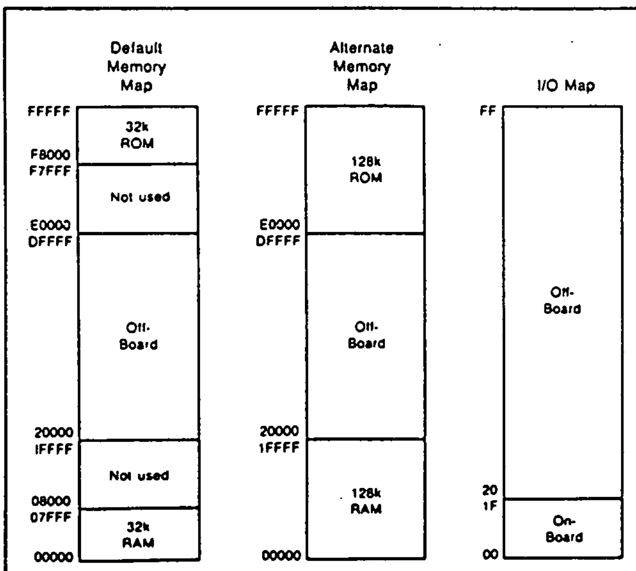


Figure 2. Memory and I/O maps.

through either USER line, to provide battery-backup when the CPU power is off.

POWER SUPPLY

The Q88 power supply is a switching regulator which will accept a DC voltage from about 4.5 to 18 volts, and which can supply 100 mA to the Vreg line. As Table 3 indicates, the minimum battery voltage depends on the maximum Vreg load which will be encountered.

ELECTRICAL SPECIFICATION

Table 3 lists a variety of specifications for the Q88 board. Battery current specifications do not include the power required to drive the serial or digital I/O lines.

The Q88 crystal oscillates at 4.9152 MHz. This gives a minimum instruction execution time of 813 nanoseconds. The system clock provided to the C-44 bus is 614.4 kHz.

ORDERING INFORMATION

The Q88 is supplied with a User's Manual and a disk which includes source code for the on-board monitor (in C and assembly) and for the QBIOS functions.

Order the Q88 with the following part numbers:

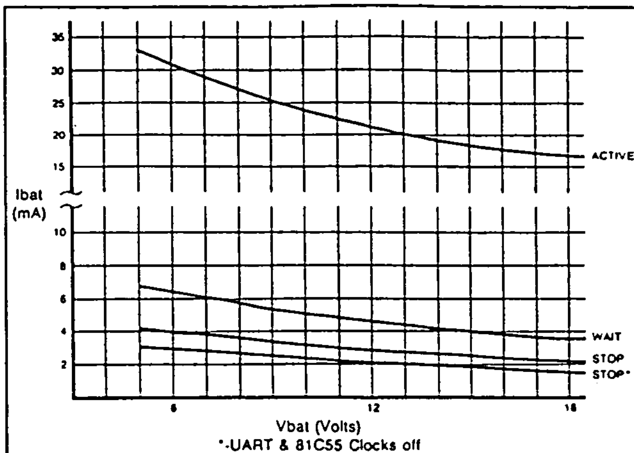


Figure 3. Power supply current.

Q88	with 32K memory ICs
Q88/B	with 128K memory ICs
Q88/N	no memory ICs and on-board memory disabled

Two different cables are available to mate with the 26-pin header on the Q88. CABLE-4 has a mating 26-pin socket connector and 1 meter of 26-conductor ribbon cable. CABLE-5 is also 1 meter long, with 26-pin socket connectors on both ends, to enable connection to the Q88 on one end and the QSI Corporation Q232A or QCOM-1 on the other end.

Table 3. Miscellaneous Q88 Specifications.

Parameter	Min	Typ	Max	Units
Operating temperature range*	-40		85	°C
Storage temperature range	-40		85	°C
Maximum humidity (non-condensing)			95	%RH
Minimum battery voltage, no load*		4.3	5.0	V
Minimum battery voltage, 50 mA load*		4.4	5.1	V
Minimum battery voltage, 100 mA load		5.3	6.1	V
Maximum battery voltage			18	V
Battery current, ACTIVE, Vbat = 12 V*		23	33	mA
Battery current, WAIT, Vbat = 12V*		4	6	mA
Battery current, STOP, Vbat = 12V*		1.9	2.5	mA
Vreg, active or WAIT mode:				
Vbat = 12V, Temp = 25°C*	4.85	5.0	5.15	V
Vbat = 6 - 18 V, Temp = -40° to 85°C	4.75		5.25	V
Vreg, STOP mode:				
Vbat = 12 V, Temp = 25°C*	2.70	2.9	3.10	V
Vbat = 6 - 18 V, Temp = -40° to 85°C	2.60		3.30	V
STOP mode timing				
Enter, 0 - 100 mA load		15		mS
Exit, 0 - 100 mA load		15		mS

* = limit tested on all units

QSI CORPORATION

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2212 South West Temple, #46
Salt Lake City, UT 84115 • USA
Telephone: 801-466-8770 • FAX: 801-466-8792

0021 - Printed in USA

Table 1. Q88 monitor commands.

Cmnd	Name	Use
?	help	list monitor commands
@	set segment	change the current segment
A	assemble	assemble 8088/V20 code
B	breakpoint	set/clear a breakpoint
C	continue	continue execution of program
D	display	display memory
E	enable	enable interrupts
F	fill	fill memory
G	EEPROM	read/write data to EEPROM
H	help	list monitor commands
I	input	input from an I/O port
J	jump	jump (call) to a subroutine
K	kill	disable (kill) interrupts
L	load	load an Intel HEX file
M	mem test	perform memory test
N	next	execute the next instruction
O	output	output to an I/O port
P	clock	view/alter real-time clock
Q	version	display monitor version
R	repeat step	execute machine instructions
S	substitute	substitute values into memory
T	transfer	transfer a block of memory
U	unassemble	unassemble 8088/V20 code
V	view/alter	view/alter CPU registers/flags
W	write	write memory in Intel HEX
X	wait	CPU wait mode test
Y	step over	single step over calls/ints
Z	stop	CPU stop mode test

The monitor in the on-board EPROM has twenty-seven different commands (Table 1). Some of the most useful commands are:

Assemble/Disassemble: In-line assembly directly into memory, and disassembly directly from memory, is supported. The assembler/disassembler commands support the NEC V20 extended instructions as well as the standard 8088 instructions.

EEPROM: Reading from and writing to the on-board serial EEPROM is supported by the monitor.

RTC: Set or read the time and date.

Breakpoints, Single-Stepping, Tracing: These commands allow for quick and easy debugging of application programs. Complete control of all registers and flags is also provided.

Included with the Q88 User's Manual is a disk which has the source code for this monitor, as well as for the QBIOS functions (below).

QBIOS SOFTWARE

The Q88 includes the QSI Basic Input Output System (QBIOS) in the monitor, and in a disk file for applications programs. This QBIOS includes over thirty useful functions (Table 2), all of which are very easy to use, and which will relieve applications programmers of many hours of programming and debugging.

All QBIOS functions are used by loading a CPU register with a function value, loading parameters (as needed) into other registers, then executing a software interrupt. Examples of using the functions in RAM (during debugging) and EPROM (in the final application) are included.

Table 2. QBIOS functions.

Name	Description
SETURT	Set the UART baud rate
GETCH	Get a single ASCII character
GETBT	Get a hexadecimal byte
GETW	Get a hexadecimal word
GETS	Get a string
GETD	Get a decimal number
GETAD	Get a segment: offset address
SENDCH	Send a single ASCII character
SENDBT	Send a byte
SENDW	Send a word
SENDS	Send a string
SENDD	Send a decimal number
CRLF	Send a carriage return/linefeed
SENDSP	Send a space
SEND2S	Send a double space
ERASE	Erase the EEPROM
WRTB	Write a byte to the EEPROM
READB	Read a byte from the EEPROM
WRTE	Write a butter to the EEPROM
READE	Read the entire EEPROM contents
HLOAD	Load an Intel HEX format
HWRITE	Write memory in Intel HEX format
SETTIME	Set the RTC time
STOPRTC	Stop the RTC counters
GETTIME	Get the RTC time
STRTRTC	Start the RTC counters
SETALARM	Set the RTC alarm
SETTIMER	Set the RTC timer
READRTC	Read contents of RTC memory
WRTRTC	Write to RTC memory
GETRTC	Get source of RTC interrupt
STOP	Enter STOP mode
WAIT	Enter WAIT mode

PROGRAMMING THE Q88

Numerous companies provide compilers, debuggers, assemblers, linkers and other support software which can be used with the Q88 board. At QSI we use the Borland Turbo C and Turbo debugger, along with software from Paradigm which allows Turbo C code to be used in ROM-based systems.

QSI's "Software" data sheet provides detailed information on these and other products for programming the Q88 board.

ON-BOARD OPTIONS

The Q88 has several DIP-switch-selectable options:

- Either C-44 bus USER line can be connected to 81C55 I/O bits (USER1 to B0 and USER2 to A0).
- The CLOCK line can be halted (pulled low), to reduce noise and power in the majority of C-44 applications which do not use the bus CLOCK line.
- An external battery (1.0 to 3.0 volts) can be connected to the RTC through the OLDBAT line or

QSI

CORPORATION

QMEM-2

4 MBYTE MEMORY BOARD FOR THE C-44 BUS

FEATURES

- Up to 4 Mbytes on one QMEM-2 board (4 Mbit ICs, four memory banks).
- Each bank can be independently configured for 256 Kbit to 4 Mbit EPROMs, RAMs or EEPROMs.
- Up to eight QMEM-2 boards in one C-44 system allows up to 32 Mbytes of memory.
- Provides programmable non-volatile data storage when used with 5-volt-only EEPROMs.
- Two banks of four 32-pin memory sockets.
- 64K and 256K EEPROMs can be mixed with RAMs in one bank.

The QSI QMEM-2 board is a versatile memory board for C-44 bus systems. The QMEM-2 supports two independent memory banks, with four 32-pin sockets in each bank. Each bank can be configured for 256 Kbit to 4 Mbit EPROMs, RAMs or EEPROMs, which gives the QMEM-2 a capacity of 256 Kbytes to 4 Mbytes.

Up to eight QMEM-2 boards can be installed in one C-44 system. This gives a total C-44 system memory capacity of 32 megabytes.

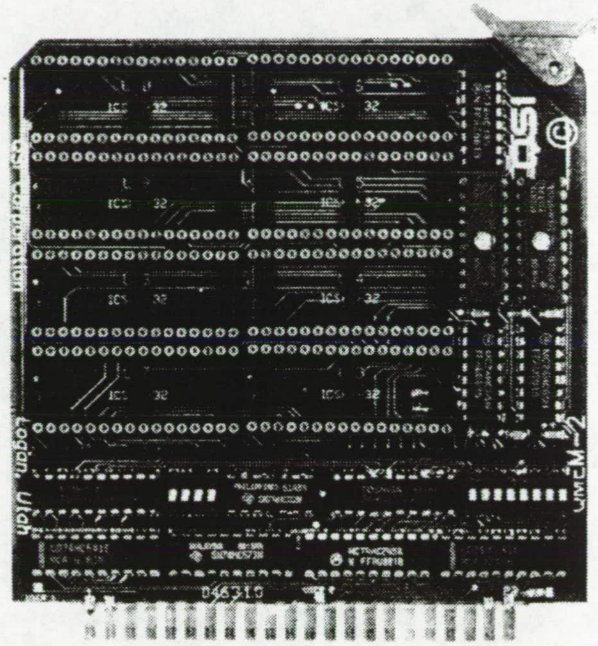
CIRCUIT DESCRIPTION

The QMEM-2 fully buffers all C-44 bus lines. The board includes one 8-bit I/O-mapped latch which is used for configuration. The output from the latch is compared to the upper address bits to determine when to enable a bank of memory for read or write access.

Each bank of memory has its own 20-pin configuration plug, which is actually a PAL that has been programmed for the type of memory in the bank. This PAL performs several functions: it routes the proper address lines to the 1-of-4 bank decoder; routes the particular address and signal lines to the pins of the memory ICs which vary between memory types; keeps all output lines in a static state except when the bank is being addressed to reduce power consumption); provides for half-bank enable/disable for use with 4 Mbit ICs.

MEMORY TYPES

The QMEM-2 can accept a variety of memory types and sizes; the only restriction is that all four ICs in one bank must use the same configuration plug. Table 1 shows the possible memory types and sizes, examples of manufacturers and part numbers, and the configuration plug which must be used. As the plug



designations show, 64K and 256K EEPROMs can be mixed with 256K RAMs in one bank.

There are many minor variations in memory ICs from different manufacturers, some of which are inconsequential, and some of which make a part unusable (particularly with EEPROMs). When in doubt, check with QSI before buying your memory ICs.

Table 1. Example memories for the QMEM-2.

Memory Type	Size (Bits)	Config. Plug	Typical Manufacturers
EPROM	256K	A	HN27C256 - Hitachi
	512K	B	µPD27C512 - NEC
	1M	C	TC571000D - Toshiba
	2M	D	27C020 - INTEL
	4M	E	??
RAM	256K	J	HM62256LP - Hitachi
	512K	K	?????
	1M	L	TC551001PL - Toshiba
	2M	M	??
	4M	N	RAMBO - Onset
EEPROM	64K	J	DQ28C64 - SEEQ
	256K	J	28C256 - SEEQ
	1M	R	X28C010 - Xicor
	2M	S	??
	4M	T	??

CONFIGURATION

Configuring the QMEM-2 for a particular size and type of memory requires three steps: install the correct configuration plug, set the on-board DIP switches properly and write the desired bank address byte.

Two configuration plugs (one for each bank) are ordered with the QMEM-2. Select the plugs for the particular memory ICs you intent to use (see Ordering Information).

There are two on-board DIP switches. The first switch set the I/O address of the QMEM-2. This address can be set to any even address from E0H through FEH, and applies to both banks of memory.

The second switch connects or disconnects address lines from the base-address decoder. The setting of these switches (one set for each bank) is based only on the size of the memory ICs in the memory bank.

The final step is to write one I/O-mapped byte to select the desired base address, and disable/enable either (or both) banks. Table 2 shows what function each bit of this byte performs. Note that both banks can be disabled, which enables multiple QMEM-2 boards in one system.

The "half-bank" select feature allows the full use of 4 Mbit ICs. If the QMEM-2 is loaded with eight 4 Mbit ICs (i.e. 4 Mbytes of memory), this memory will appear as four 1 Mbyte pages on the C-44 bus 20-bit address space.

ELECTRICAL SPECIFICATIONS

The QMEM-2 meets all electrical specifications of the C-44 bus, including reliable static operation in the STOP mode (where Vreg is lowered to 3 volts).

Table 2. QMEM-2 configuration byte.

Bit	Bank	Function
0	A	enable bank
1	A	A17 select (256K)
2	A	A18 select (256K - 512K)
3	A	A19 select (256K - 1M) or half-select (4M)
4	B	enable bank
5	B	A17 select (256K)
6	B	A18 select (256K - 512K)
7	B	A19 select (256K - 1M) or half-select (4M)

The power consumption of the QMEM-2 is highly dependent upon its operating mode, as well as what types of memory are installed. Table 3 shows typical current consumption (from Vreg) using 256 Kbit ICs and a Q85-2 CPU board.

ENVIRONMENTAL SPECIFICATIONS

The operating and storage temperature range for the QMEM-2 is -40 to 85°C. Operating and storage humidity range is 0 to 95% non-condensing relative humidity. As with all of QSI's C-44 boards, the QMEM-2 is burned in for 24 hours, including two cold thermal cycles.

ORDERING INFORMATION

The QMEM-2 is supplied with a User's Manual and two configuration plugs (additional plugs are available). Order the QMEM-2 with configuration plugs by specifying the plug letter from Table 1:

```
QMEM-2xy
       |└──────── Bank B configuration plug
       └───────── Bank A configuration plug
```

For example, the part number QMEM-2CJ would be used to order the board with one 1-Mbit EPROM configuration plug and one 256-Kbit RAM configuration plug.

The QMEM-2 is normally shipped unstuffed. Contact QSI for pricing and availability for boards which are configured, stuffed and burned-in with memory before shipment.

Table 3. QMEM-2 current consumption.

Mode	Ireg	Units
Executing in RAM	13	mA
Executing in EPROM	12	mA
Programming EEPROM	55	mA
Not accessed	120	μA
WAIT mode	16	μA
STOP mode	8	μA

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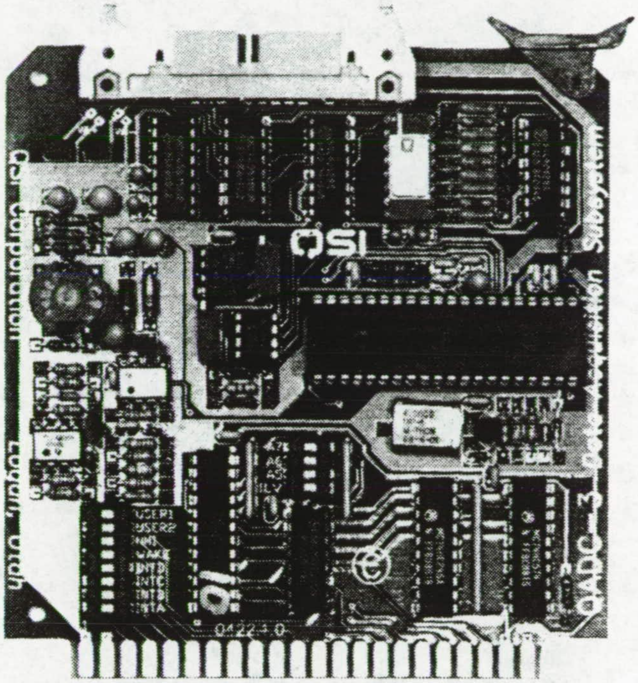
QADC-3

14-BIT, 18 μS ANALOG-TO-DIGITAL CONVERTER BOARD FOR THE C-44 BUS

FEATURES

- 14-bit resolution, unipolar or bipolar inputs.
- 18 μS conversion time.
- Available with or without an instrumentation amplifier (IA).
- No-IA version: lower power, high signal bandwidth, lower cost, 16 single-ended inputs, and software-selectable 0-to-5 volt or ±5 volt input ranges.
- IA version: 16 single-ended or 8 differential inputs, or combination (software selectable) and bipolar or unipolar input ranges of 0.5, 1.0, 2.5, or 5.0 volts.
- Self-calibrating A/D converter IC.
- Low power: no-IA version uses less than 350 mW total power consumption while operating.
- Both versions use less than 50 μW when off.



The QSI Corporation QADC-3 board is a complete 14-bit data acquisition subsystem on a single C-44 card. The board is capable of 18 μS conversions, allowing conversion rates up to 55.5 kHz.

The QADC-3 is available in two versions: with or without an instrumentation amplifier (IA). The QADC-3/N (no IA) offers full-power signal bandwidth well beyond the Nyquist frequency of 27 kHz. This version has 16-single ended inputs, and software selectable input voltage ranges of 0 to 5 and ±5 volts, with automatic or software-controlled ADC calibration.

The QADC-3/I (with IA) does not have as high a signal bandwidth, but the IA allows for more flexible input configuration. The QADC-3/I has software-selectable input channels (16 single-ended, 8 differential, or a combination), software-selectable bipolar or unipolar input ranges (0.5, 1.0, 2.5, or 5.0 volts), and automatic or software-controlled ADC calibration.

Both versions of the QADC-3 offer the low power consumption typical of QSI's C-44 boards: the QADC-3/N uses only 350 mW while active and the QADC-3/I uses only 550 mW. Either version can be turned off, and will use less than 30 μW of power.

CIRCUIT DESCRIPTION

The block diagram in Figure 1 shows the different sections of the QADC-3 board. The eight differential or sixteen single-ended inputs first enter the input multiplexers. Following these is a single differential multiplexer. This multiplexer can select one of sixteen single-ended inputs, one of eight differential inputs, or ground inputs. The output of this multiplexer goes directly to the ADC (/N) or the IA (/I).

By selecting ground inputs to the differential multiplexer, the QADC-3/I can use software calibration to remove zero drift (offset) errors. This is not needed on the QADC-3/N, since this version does not have significant zero (offset) errors.

The IA and related components are not included in the QADC-3/N; the differential multiplexer routes the selected single-ended input directly to the A/D converter.

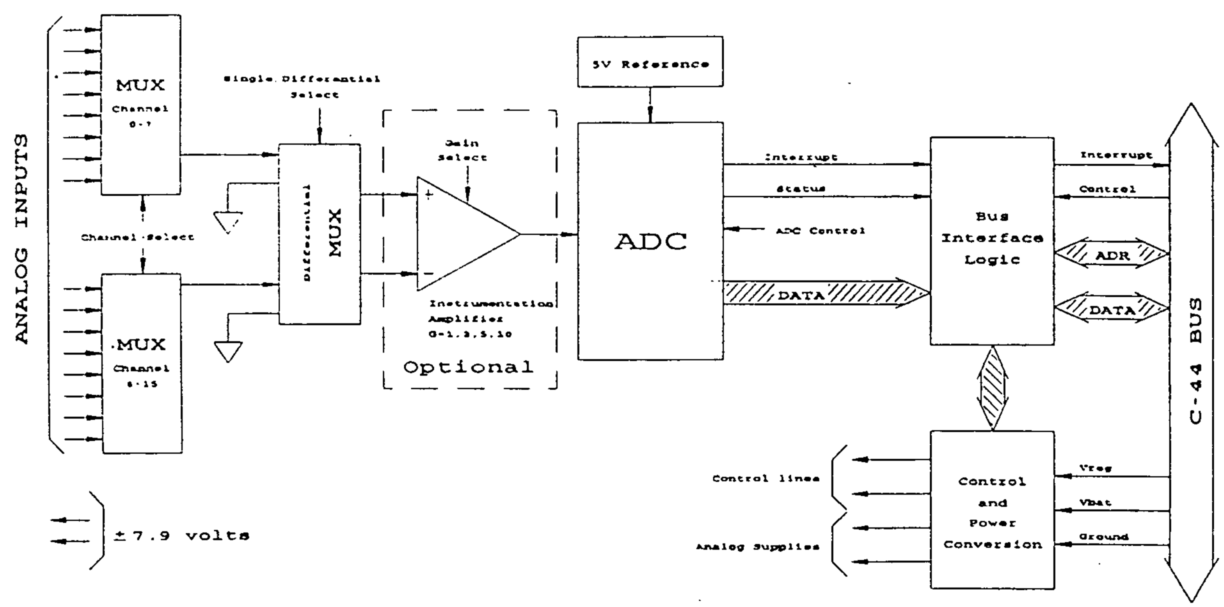


Figure 1. QADC-3 Block diagram.

The IA is software-programmable for gains of 1, 2, 5 or 10, which gives unipolar input ranges of 5.0, 2.5, 1.0, and 0.5 volts, respectively, or bipolar ranges are ±5.0, ±2.5, ±1.0, ±0.5 volts.

Following the IA is the Analog-To-Digital Converter (ADC) chip. This ADC is capable of self-calibration to eliminate drifts that would degrade accuracy. A +5.000 volt external reference voltage is supplied to the converter. The ADC also has an internal sample-and-hold to allow conversion of dynamic input signals.

Other circuitry is responsible for converting the C-44 bus power supplies (Vbat and Vreg) to those needed by the analog circuitry on the QADC-3 board, switching the power supplies on and off, controlling the ADC, and interfacing to the C-44 bus.

INPUT MULTIPLEXERS

The input multiplexers of the QADC-3 will pass any analog signal of less than ±5.3 volts with respect to the analog ground. The common mode range available for differential signals (/I only) is specified in Table 2.

The input multiplexers are not protected against overvoltage conditions. When the board is powered, all signals must be less than ±5.7 volts; when the board is off, all signals must be less than ±0.4 volts.

QADC-3 ACCURACY

For the QADC-3/N, the only items which contribute significantly to conversion errors are the A/D Converter (ADC) and the voltage reference. The ADC has self-calibration capability which effectively removes its own zero error, and keeps its non-linearity error to less than 1 LSB under all conditions.

The gain error of the QADC-3/N is dependent on the stability of the voltage reference. The reference used provides typical drifts of less than 4 ppm/°C, and worst case drifts of less than 8 ppm/°C. Table 1 gives complete specifications for the QADC-3/N accuracy.

Because of the additional components on the QADC-3/I, it has slightly higher conversion errors than the QADC-3/N (Table 1). The zero error can be removed at any time by grounding the IA inputs (with the differential multiplexer), performing a conversion, and subtracting the resulting value from subsequent conversions.

SIGNAL BANDWIDTH

The QADC-3/N has a much higher signal bandwidth (BW) than the QADC-3/I. Table 2 shows the capabilities of both versions of the board. The parameter titled "1 count BW" refers to the frequency at which the signal will be attenuated by one count (0.006%) by the components on the QADC-3. For the QADC-3/N, this exceeds the highest Nyquist frequency (27 kHz) by a large margin.

The signal bandwidth for the QADC-3/I is much lower, and depends strongly on the input voltage swing. The BW is shown for several different input voltages.

A detailed analysis of the signal-to-(noise+distortion)

Table 1. QADC-3 performance specifications.

Parameter	25 °C Typical	25 °C Maximum	0 to 70°C Maximum	-25 to 85°C Maximum	-40 to 85°C Maximum	Units
Zero Error						
/N	±100	±180	±210	±310	±390	µV
/I	±0.6	±3.1	±5.7	±6.7	±6.8	mV
Linearity Error						
/N	±0.3	±0.5	±0.7	±0.8	±0.9	lsb
/I	±0.5	±0.7	±0.8	±1.0	±1.1	lsb
Full-scale error						
5 V range (/N,/I)	±0.5					lsb
2.5 V range (/I)[1]	±2					lsb
1.0 V range (/I)[1]	±3					lsb
0.5 V range (/I)[1]	±4					lsb
Full-scale drift						
5 V range (/N)					8	ppm/°C
5 V range (/I)					12	ppm/°C
2.5 V range (/I)					12	ppm/°C
1.0 V range (/I)					14	ppm/°C
0.5 V range (/I)					16	ppm/°C

[1]The QADC-3/I has one full-scale adjust, which can be used to adjust any one of the four gain ranges to zero full-scale error.

capabilities of the CS5014 converter function over frequency can be found in the Crystal Semiconductor data sheet for the part. This analysis includes the effects of the CS5014 sampling distortion, slew rates, etc.

SYSTEM NOISE

Figure 2 shows the typical noise for the QADC-3 in a form which includes all components of the board, and includes the entire bandwidth over which the QADC-3 can respond to noise.

The histogram shown in Figure 2 is generated by supplying a low-noise, low-impedance signal to the board, then performing 4096 conversions. The histogram shows how many conversions return the correct value and how many are off by one count.

SOFTWARE

User software has direct control over the input channel select bits, the gain control bits, ADC calibration, unipolar/bipolar selection, and starting the ADC conversion. The QADC-3 can generate any C-44 bus interrupt upon completion of a conversion (including the WAKE line). The QADC-3 manual includes a disk with example code for using the QADC-3.

Also available is an external trigger, which allows an external device to start a conversion once the system CPU has set up the proper gain and channel.

The QADC-3 occupies a 32-port block of the C-44 bus I/O space. This can be set to any 32-byte boundary.

ANALOG INPUT CONNECTOR

Analog inputs are connected to the QADC-3 board through a 34-pin header connector. Each analog input has an associated ground line for use in single-ended applications, for a total of 32 lines. The two remaining pins are connected to the ±7.9 V analog supplies. These power supplies can provide ±10 mA (typical) to off-

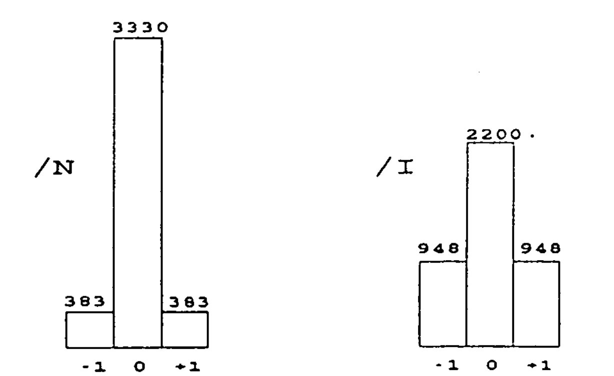


Figure 2. QADC-3 System Noise.

Table 2. QADC-3 Operational Specifications.

Parameter	Typical	Maximum	Units
1-count bandwidth			
/N, 4 V p-p	40		kHz
/I, 4 V p-p	800		Hz
/I, 1 V p-p	2000		Hz
3 db bandwidth			
/N, 4 V p-p	310		kHz
/I, 4 V p-p	105		kHz
Power-on time	380		ms
Common mode analog input range (/I)	±3.0		volts
Power supply rejection:			
Vbat=7 to 18V	0	±½	lsb
Vreg=4 to 6V	0	±½	lsb
Absolute maximum analog input voltage:			
board on		±5.7	volts
board off		±0.4	volts
Conversion time	18		μs
DC input current[1]			
/N selected channel	±30	±70	nA
/I selected channel	±30	±50	nA
unselected channel	±0.01	±50	nA

[1]ON current is bias for the IA or ADC; OFF current is leakage through the input multiplexer.

board user circuitry. They are automatically turned on and off along with the QADC-3 supplies.

QSI cables CABLE-3 and CABLE-12 are specifically intended for use with the QADC-3.

ELECTRICAL SPECIFICATIONS

The QADC-3 board uses the regulated 5 volt supply (VREG) and the unregulated 6 to 18 volt supply (VBAT) of the C-44 bus. Power supply current drains depend on the state of the QADC-3 board and are summarized in Table 3. Since the board has a switching regulator, battery current varies with battery voltage.

Table 3. QADC-3 Power Requirements.

Conditions: Vreg=5V, Vbat=12V, temperature=25 °C.

	Ireg		Ibat		
State	Typ	Max	Typ	Max	Units
/N Converting	5	12	27	38	mA
/I Converting	5	12	44	59	mA
/N Inactive	1	2	27	38	mA
/I Inactive	1	2	44	59	mA
Off	6	20	0	0	μA
Off (WAIT)	6	20	0	0	μA
Off (STOP)	3	10	0	0	μA

The QADC-3 is power switched to minimize its power consumption when conversions are not actually being performed. When the board is turned on, it takes about 20 ms for the power supplies to stabilize, then the converter performs a self-calibration, which requires about 360 ms. Therefore, a total delay of 380 ms is required between the turn-on command and the execution of the first A/D conversion.

ENVIRONMENTAL SPECIFICATIONS

Operating and storage temperature range for the QADC-3 is -40 to 85 °C. Accuracy specifications are given for several different temperature spans (Table 1). Operating and storage humidity range is 0 to 95% non-condensing relative humidity.

ORDERING INFORMATION

Order part number QADC-3/N (for the no-IA version) or QADC-3/I (for the IA version). Either version is supplied with a User's Manual which includes a disk of example software. QSI cables CABLE-3 and CABLE-12/34 are available for use with the QADC-3. See the CABLES data sheet for more information.

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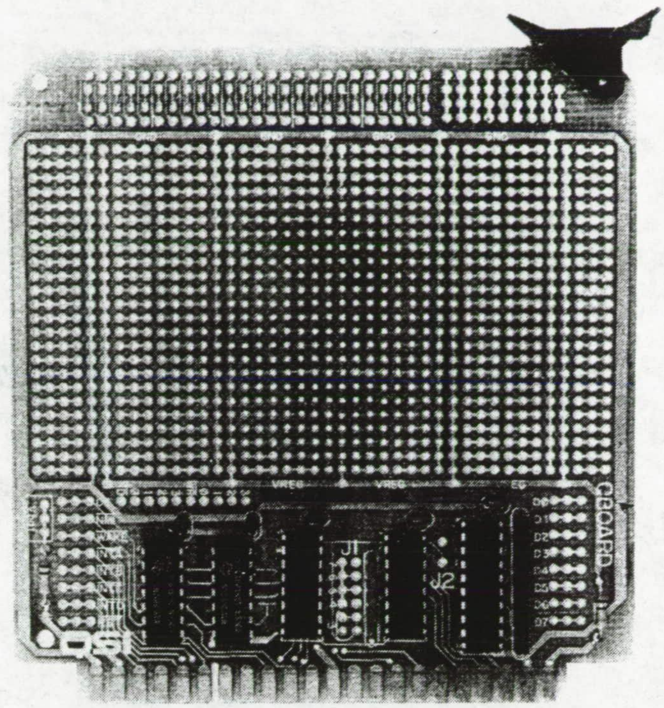
QBOARD DATA SHEET

PROTOTYPING BOARD FOR THE C-44 BUS



FEATURES

- All C-44 bus interface electronics are provided.
- Four I/O write strobes and four I/O read strobes are decoded on-board.
- Easy access to C-44 bus interrupt, power, reset, and clock lines.
- Can be mapped to any 8-byte boundary in the C-44 I/O space.

The QSI Corporation QBOARD is a prototyping board for the C-44 bus. It simplifies the job of designing and prototyping new C-44 bus designs by providing all the bus interface electronics needed to make an Input/Output (I/O) mapped electronics assembly.

QBOARD ELECTRONICS

Figure 1 is a block diagram of the QBOARD bus interface electronics. The QBOARD decodes the C-44 I/O port address space to provide four read strobes and four write strobes in an 8-byte block of the I/O space. This block can be set by a jumper to any 8-byte boundary. The default address of the QBOARD is 60H to 67H.

In addition to the eight fully decoded read and write strobes, the QBOARD provides block decode lines for both 8-byte and 16-byte I/O blocks. These lines can be used to support user components which have on-board address decoding.

PROTOTYPING AREAS

There are three prototyping areas on the QBOARD (Figure 2). The large area has five columns for integrated circuits, and can accept 0.3, 0.4, and 0.6 inch wide ICs. Vreg and Ground traces are run through each column, making them available to ICs anywhere in the column.

The top of the board has a header area large enough for any single or double row header up to 50 pins. The header pins are brought out to solder pads for easy access. Finally, a small area for discrete components is located at the upper right portion of the QBOARD.

QBOARD EXAMPLES

The QBOARD can be used to prototype most circuits which are accessed through the C-44 I/O space. The QBOARD Manual has numerous examples of interfacing to real-world devices using the QBOARD, including digital-to-analog and analog-to-digital converters, data latches, data buffers, stepper motor drivers, opto-isolators, and high current load switches.

ELECTRICAL SPECIFICATIONS

The QBOARD is powered by the Vreg output from the CPU board. All other C-44 power lines are brought to pads from the C-44 bus for use by the prototyped electronics. Table 1 shows the current used by the QBOARD electronics in each of four available operating states.

ENVIRONMENTAL SPECIFICATIONS

Operating and storage temperature range for the QBOARD is -40° to 85°C (-40° to 185°F). Operating and storage humidity range is 0 to 98% non-condensing relative humidity.

ORDERING INFORMATION

The QBOARD is supplied with a User's Manual which contains many examples of using the QBOARD.

Table 1. QBOARD Power Requirements

State	Typ	Max	
Active	0.76	1.14	mA
Inactive	115	160	μA
WAIT	11	16	μA
STOP	6	10	μA

REVISION 1.1

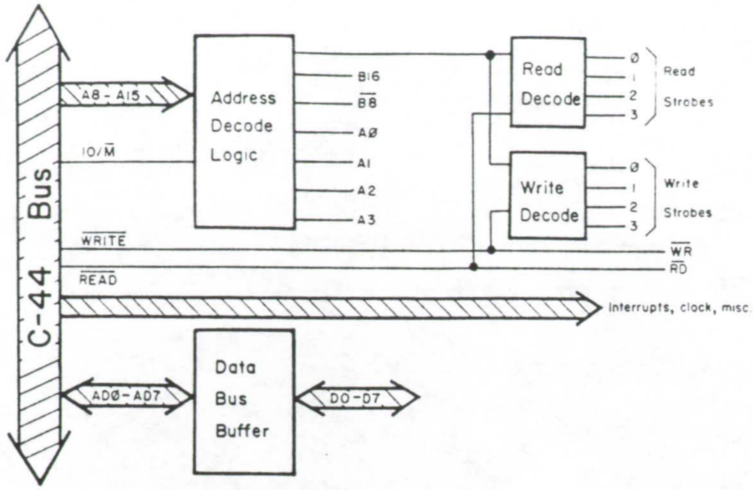


Figure 1. QBOARD block diagram.

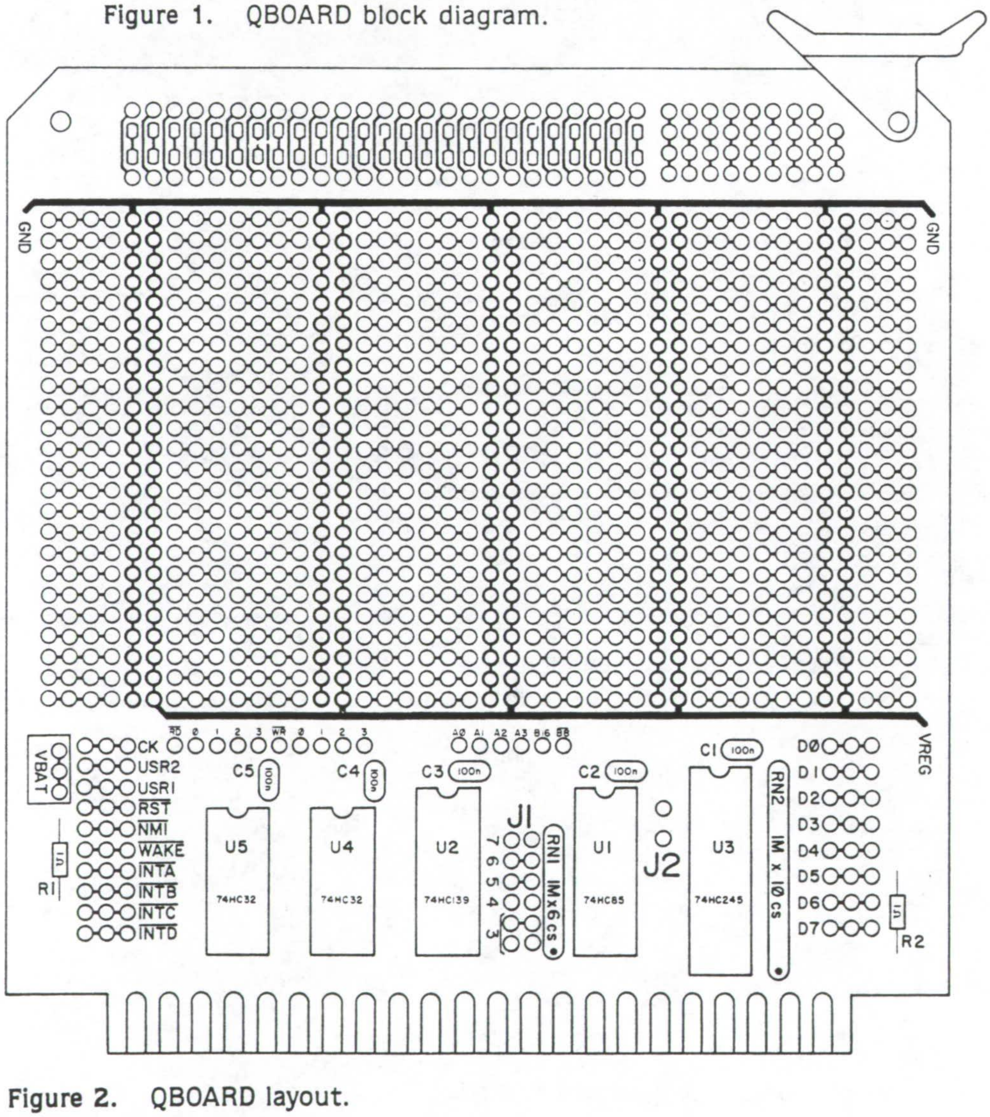


Figure 2. QBOARD layout.

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C-44 BUS ACCESSORIES

CARD CAGES, MOTHERBOARDS, CARD RETAINERS EXTENDER CARDS, PROTOTYPE BOARDS

This data sheet describes accessory products which are available for the C-44 bus. Each of these products is sold separately.

CARD CAGES

There are four card cages available for C-44 Bus systems: 2, 3, 6 and 10 slot. All four cages include card guides and PEM nuts for mounting.

The two and three slot versions can be mounted horizontally against a panel (Figure 3) or, with the addition of a bracket set, vertically (Figure 4). These cages are one-piece blue anodized aluminum. Figure 1 shows a horizontally mounted 3-slot cage and a vertically mounted 2-slot cage (using the bracket set).

The two and three slot cages also have a provision for a user-supplied card retaining pin, which can prevent cards from being shaken or jarred from the cage.

The six and ten slot cages are green anodized aluminum, and consist of four pieces: two side plates with card guides, and two end panels. These cages can be assembled so that the cards enter from the top or from the side. Mounting dimensions are shown in Figure 5.

Figure 2 shows a ten-slot cage with cards inserted from the top, and a six-slot cage with cards inserted from the side.

MOTHERBOARDS

There are four motherboards (2, 3, 6, and 10 slot) available to be used with the four card cages. The dimensions of the motherboards when attached to the card cages are shown in Figures 3, 4 and 5. Figure 6 shows all four motherboards.

For the two and three slot versions, the required interface wires (power, ground, reset) are soldered directly to the motherboard. The six and ten slot motherboards have a small terminal strip into which the wires are inserted and secured with a screw. Wires can be soldered directly to the six and ten slot motherboards if desired.

All motherboards include polarization keys, which can be used with some C-44 boards to prevent the board from being inserted incorrectly.

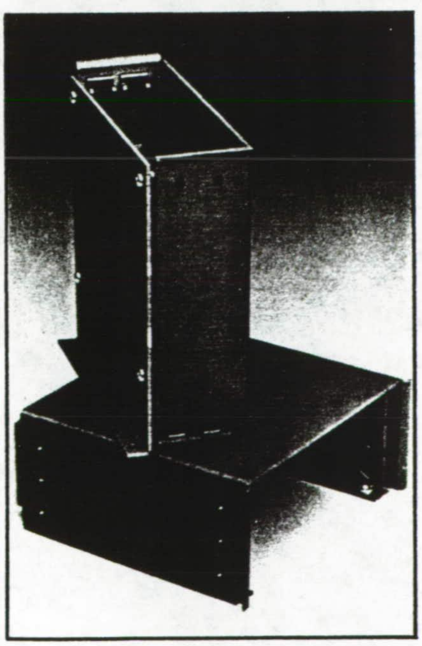

Figure 1. 2- and 3-slot card cages.

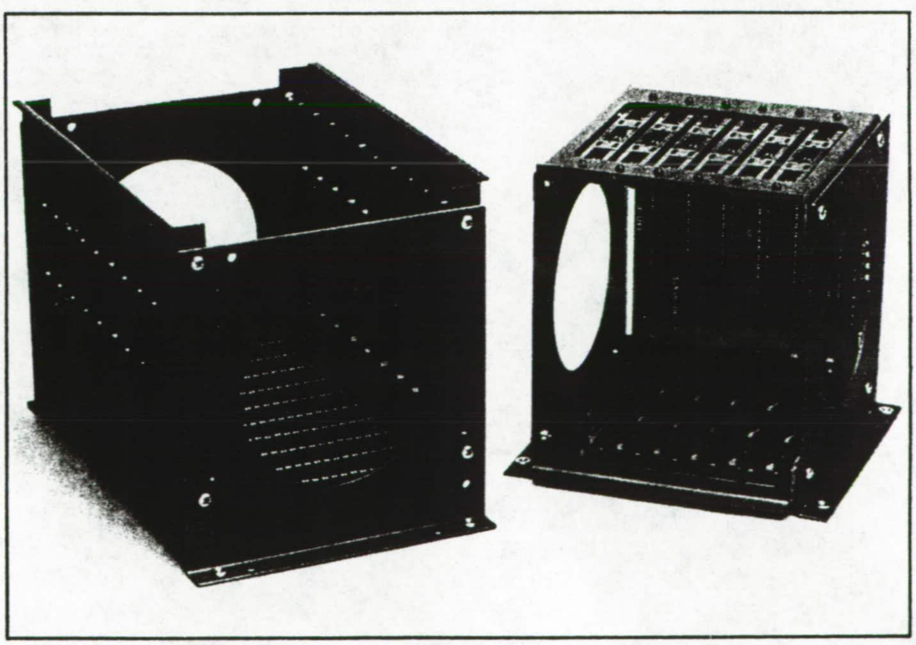

Figure 2. 6- and 10-slot card cages.

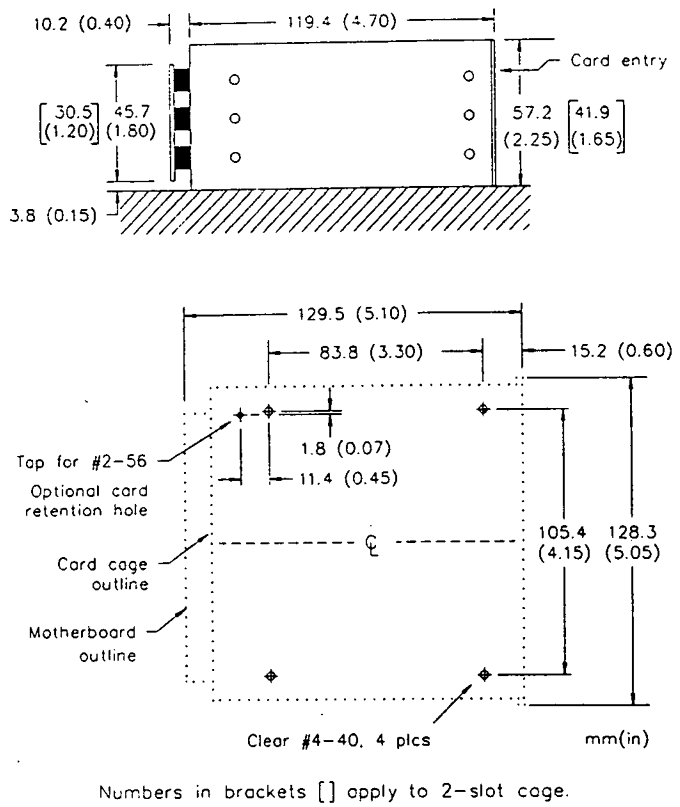

Figure 3. Horizontal 2- and 3-slot card cage dimensions.

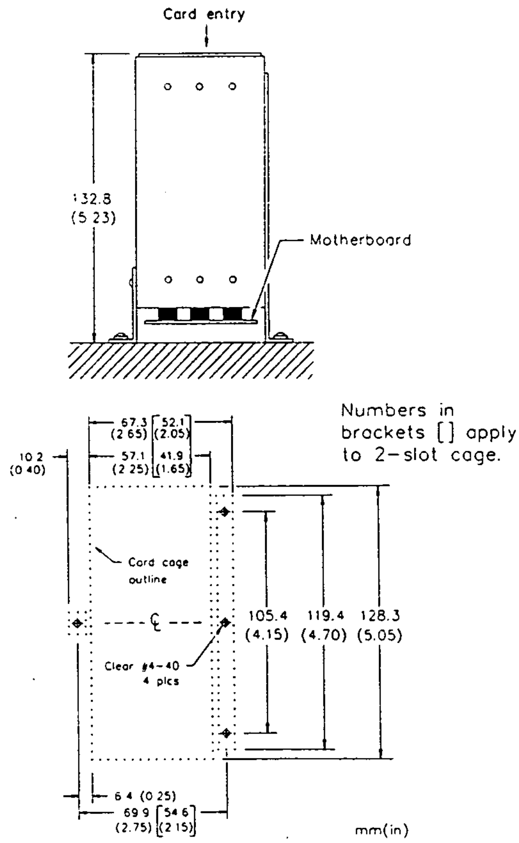

Figure 4. Vertical 2- and 3-slot card cage dimensions.

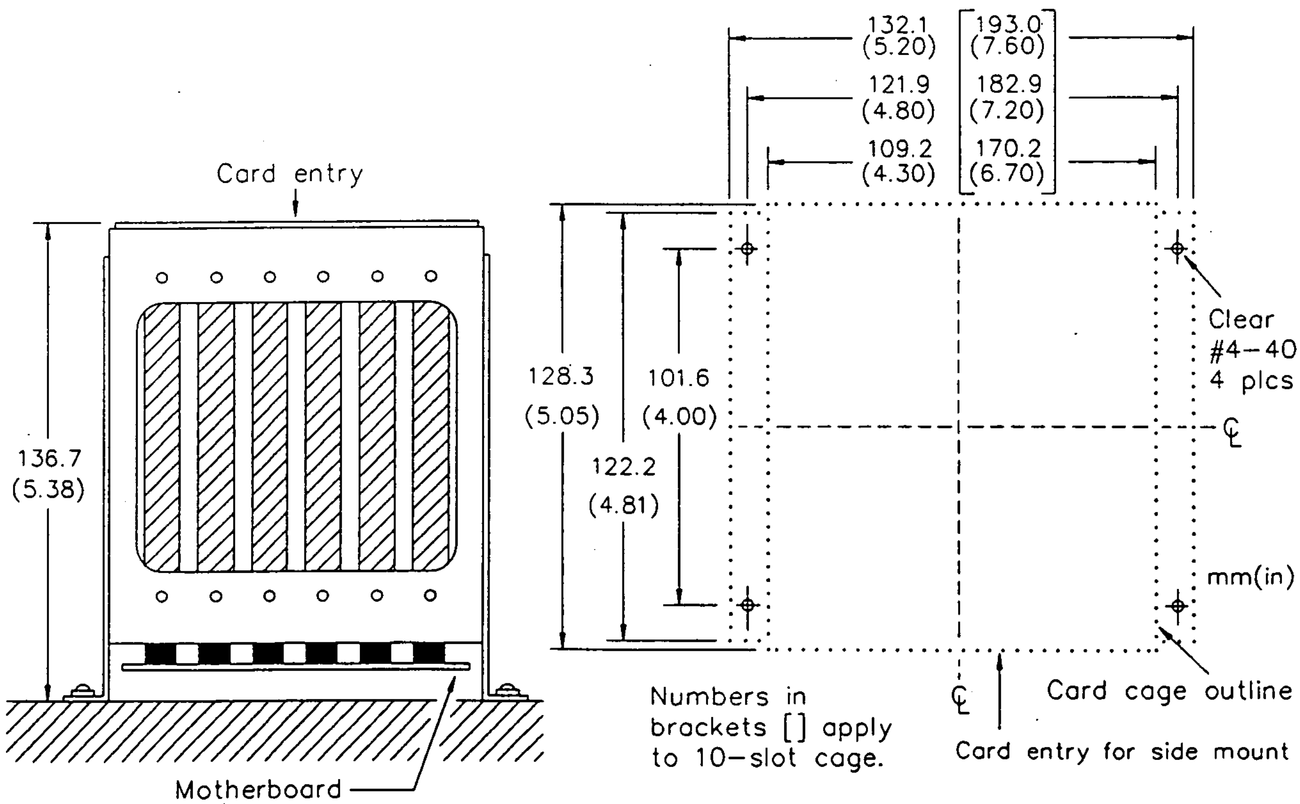

Figure 5. 6- and 10-slot cage dimensions.

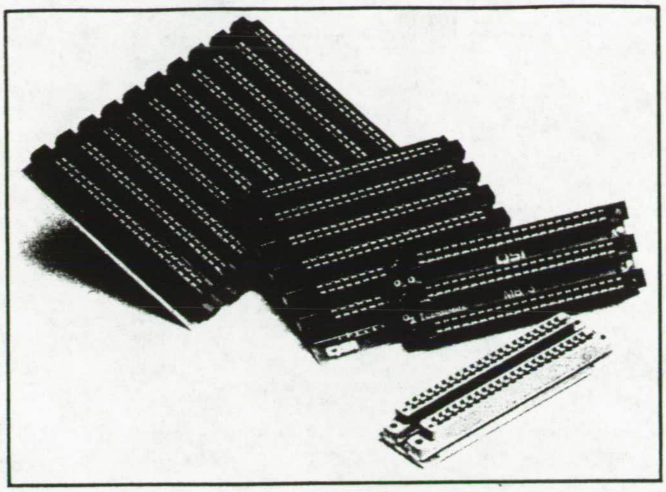

Figure 6. 2-, 3-, 6- and 10-slot motherboards.

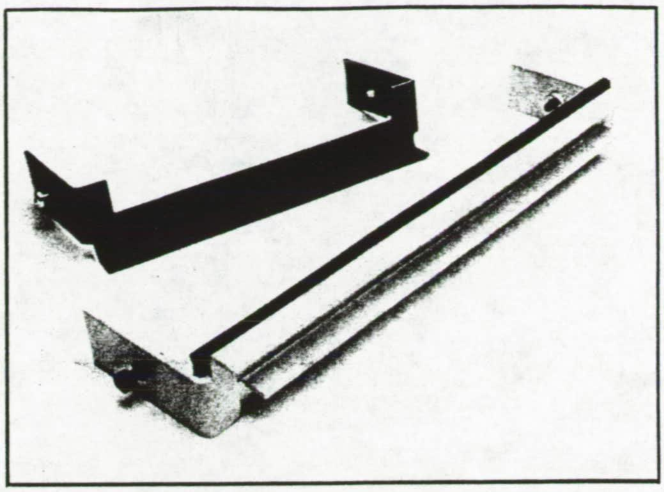

Figure 7. CR-6 and CR-10 card retainers.

CARD RETAINERS

The CR-6 and CR-10 are card retainers for the six and ten slot card cages, as shown in Figure 7. These retainers are secured with two screws (included) over the right-side card ejectors on the C-44 cards, using holes in the card cages. The retainers extend about 10 mm (0.39") above the card ejectors.

Two and three slot card cages have a hole which allows a rod to be used to secure the PC boards. This rod can be a long #2-56 screw, or a 2.25 mm (.090") diameter rod.

EXTENDER CARD

The EXT-44 is a 190 mm (7.5") long extender card which extends a C-44 board out of the card cage for easy access during development. As Figure 8 shows, all 44 bus lines are connected to terminal pins, for easy access with an oscilloscope or voltmeter probe.

The EXT-44 is a four-layer PC board for minimal noise and crosstalk.

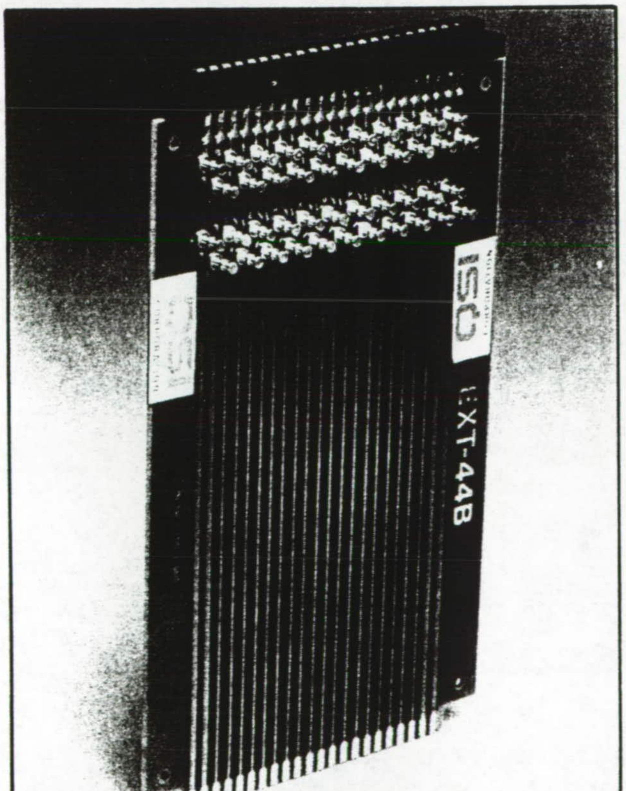

Figure 8. EXT-44 extender card.

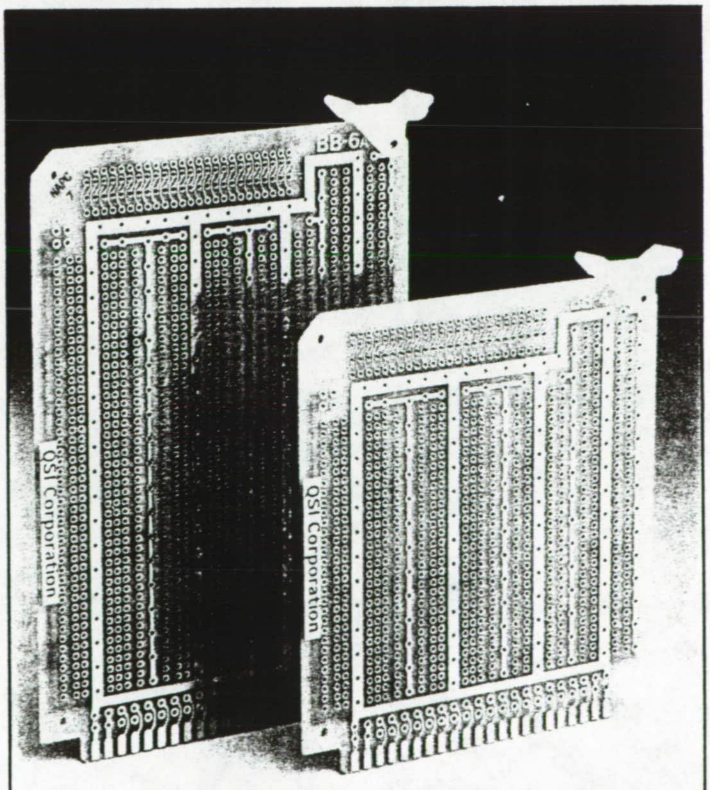

Figure 9. BB-4.5 and BB-6.5 prototyping boards.

PROTOTYPE BOARDS

The BB-4.5 and BB-6.5 (Figure 9) are prototyping boards (breadboards) for the C-44 bus. The BB-4.5 is the same height as a standard C-44 card (120 mm/4.5" high). The BB-6.5 is 165 mm (6.5") high, and will accommodate more circuitry. Both boards have Vbat, Vreg and ground traces running through sets of IC pads. All other C-44 bus lines are accessible from pads near the edge connector.

If you are building an I/O-based C-44 board, the QSI QBOARD is similar to these prototyping boards, but has the bus decoding logic already installed. See the QBOARD Data Sheet for more information.

ORDERING INFORMATION

Below are the order numbers for C-44 accessories:

Order Number	Item
BB-4.5	4.5" breadboard
BB-6.5	6.5" breadboard
BRACKET-1	Vertical bracket set for CAGE-2 and CAGE-3
CAGE-2	2-slot card cage
CAGE-3	3-slot card cage
CAGE-6	6-slot card cage
CAGE-10	10-slot card cage
CR-6	6-slot card retainer
CR-10	10-slot card retainer
EXT-44	Extender card
MB-2	2-slot motherboard
MB-3	3-slot motherboard
MB-6	6-slot motherboard
MB-10	10-slot motherboard

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SOFTWARE

3rd PARTY SOFTWARE FOR USE WITH QSI PRODUCTS

INTRODUCTION

There is a large amount of software offered by third-party vendors which can be used with QSI computers. This data sheet is a list of most of those of which we are aware.

We have neither purchased nor evaluated most of the products listed here, and assume no responsibility for these products. We welcome any comments you may have about how any of these products work with QSI's computers, or about products to add to this list.

If you are doing embedded systems development, we highly recommend the magazine *Embedded Systems Programming* (415-397-1881). In addition to informative articles, the magazine provides up-to-date information about the large variety of development tools which various companies are producing.

80C85 SOFTWARE (for the Q85-2)

C Compiler

MANX Software (AZTEC C)
PO Box 55
Shrewsbury, NJ 07701
800-221-0440

PLM Compiler

BSO
411 Waverly Oaks Road
Waltham, MA 02154-8414
800-458-8276

FORTH Compiler

Forth Incorporated
111 North Sepulveda Boulevard
Manhattan Beach, CA 90266
800-553-6784

Cross Assembler/Simulator-Debugger

2500 A.D. Software, Inc.
109 Brookdale Avenue
PO Box 480
Buena Vista, CO 81211
800-843-8144

Floating-Point Math Package

U.S. Software Corporation
14215 NW Science Park Drive
Portland, OR 97229
503-641-8446

V20 SOFTWARE (for the Q88)

C Compiler

Intermetrics
733 Concord Avenue
Cambridge, MA 02138
800-356-3594

C Compiler

Microtec Research
2350 Mission College Boulevard
Santa Clara, CA 95054
800-950-5554

Development System

ZAX Corporation
2572 White Road
Irvine, CA 92714
800-421-0989

8086/8088 SOFTWARE (for the Q88)

These products can be used with the QSI Q88 CPU board, but will generate only 8086/8088 code. The instructions unique to the V20 (used on the Q88) will not be generated by these compilers.

ADA Compiler

Alsys
1432 Main Street
Waltham, MA 02154
714-472-2410

ADA Compiler

DDC-I Incorporated
9630 North 25th Avenue, Suite 118
Phoenix, AZ 85021
602-944-1883

C Compiler

MANX Software (AZTEC C)
PO Box 55
Shrewsbury, NJ 07701
800-221-0440

C Compiler

Avocet Systems, Inc.
120 Union Street
PO Box 490
Rockport, ME 04856
800-448-8500

C, PASCAL, PLM Compilers

BSO
411 Waverly Oaks Road
Waltham, MA 02154-8414
800-458-8276

C Compiler

Intel Corporation
800-548-4725

C Compiler

Lattice, Inc.
2500 South Highland Avenue, Suite 300
Lombard, IL 60148
800-444-4309

FORTH Compiler

Forth Incorporated
111 North Sepulveda Boulevard
Manhattan Beach, CA 90266
800-553-6784

FORTH Compiler

Laboratory Microsystems, Inc.
PO Box 10430
Marina Del Rey, CA 90295
213-306-7412

Cross Assembler

2500 A.D. Software Inc.
109 Brookdale Avenue
PO Box 480
Buena Vista, CO 81211
800-843-8144

Cross-Compiler/Assembler

American Automation
2651 Dow Avenue
Tustin, CA 92680-7207
714-731-1661

Floating-Point Math Package

U.S. Software Corporation
14215 NW Science Park Drive
Portland, OR 97229
503-641-8446

Microsoft QuickBASIC Development System

It is possible to do software development for the QSI Q88 board using Microsoft's QuickBASIC compiler. An article discussing the process appeared in *Embedded Systems Programming* (415-397-1881) in the March 1990 issue. The required runtime library (called PDQ) and further information are available from:

Crescent Software
32 Seventy Acres
West Reading, CT 06896
203-438-5300

C Development Systems

It is possible to do software development for the QSI Q88 board using native C compilers designed for MS-DOS computers. The advantages of this are that these compilers (specifically, Microsoft *Quick C* and Borland *Turbo C*) are very powerful, economical and widely used compilers, with powerful debugging tools.

The requirement to run code generated by a native MS-DOS compiler on a target computer is a second program, called a *locate* utility, which converts the .EXE file generated by the compiler into a ROMable .HEX file which can be programmed into an EPROM.

Other programs (debugger interfaces) allow the debugging capabilities of the native compiler to be used on a target system. See the next page for details of such an MS-DOS development system.

C Compiler

Borland International
Turbo C Professional Package
(widely available)

C Compiler

Microsoft Corporation
Microsoft C or Quick C
(widely available)

Locate and Debugger Interface Utilities

Paradigm Systems Incorporated
PO Box 152
Milford, MA 01757
800-537-5043

Locate Utilities

Systems and Software Incorporated
18012 Cowan, Suite 100
Irvine, CA 92714-6809
714-833-1700

Aldia Systems Inc.
PO Box 37634
Phoenix, AZ 85069
602-866-1786

Linker/Locators

Genesis Microsystems Corporation
13300 East Sunset Drive
Los Altos, CA 94022-9002
415-941-9002

Phar Lap Software, Inc.
60 Aberdeen Avenue
Cambridge, MA 02138
617-661-1510

Soft Advances
10811 Washington Boulevard
Suite 205
Culver City, CA 90232
213-559-7015

Systems & Software Inc.
18012 Cowan
Suite 100
Irvine, CA 92714-6809

Locator/Debugger

Datalight
17505 - 68th Avenue NE
Suite 304
Bothell, WA 98011
206-486-8086

THE TURBOPAK: TURBO C ON THE Q88

From the list above, the choices seem to be without end. At QSI, we have tried the combination of Borland's Turbo C (and Turbo C++) and Turbo Debugger with Paradigm's Locate and Turbo Debugger Remote utilities with very good results. This combination impressed us so much that we now provide a disk of support software (called the **TurboPak**) with all Q88 boards.

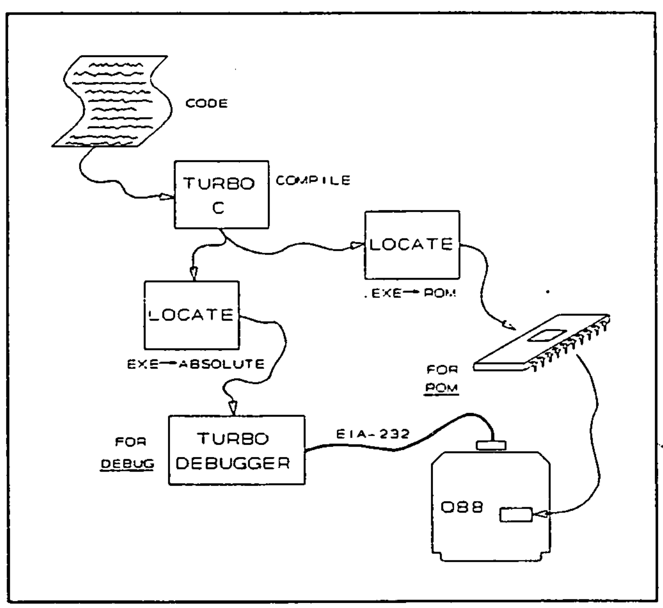
The process of using Turbo C on the Q88 goes something like this:

1) Write a C program with your own editor or using the editor provided with the Turbo C integrated environment.

2) Compile and link the program, using the Turbo C compiler and linker.

3) Use Paradigm's Locate program to convert from the .EXE format to the required debug format.

4) Debug the program with it actually running on the



Q88 using the Turbo Debugger and Paradigm's Turbo Debugger Remote (TDREM) program.

5) Once the program is bug free, use the Locate program a second time to make a ROMable version, then program your EPROM.

This set of programs handles all of the problems you would normally encounter when trying to run Turbo C on an embedded computer, including:

- Turbo C generates code intended to run under the MS-DOS operating system. Paradigm's Locate handles this by taking the .EXE file and relocating all segments to the appropriate RAM and EPROM areas of your CPU board.

- Hardware support for most of the Turbo C I/O functions does not exist (for example, the Q88 does not have a keyboard for use with the *scanf* function). Paradigm supplies a skeleton of the MS-DOS operating system that allows many of the functions to be used. Here at QSI we have modified this skeleton to match the hardware found on the Q88. For example the *scanf* and *printf* functions are directed through the Q88's UART.

- The Turbo Debugger is not intended to be used with an embedded CPU board such as the Q88. Paradigm has solved this with their Turbo Debugger Remote, which fools the Turbo Debugger into thinking that the Q88 is a remote PC. This allows you to debug your code while it is running on the Q88, using all the capabilities of the source-level Turbo Debugger.

  Again, we provide the necessary files to allow you to compile and program a debug EPROM in just a few minutes. Plug this into the Q88, and you are ready to begin Turbo Debugging.

- Finally, Turbo C does not have functions to communicate with all of the Q88 hardware, such as the real-time clock and EEPROM. We have solved this problem by providing the QBIOS with the Q88. This is a set of assembly-language routines which access all of the Q88 on-board hardware. These routines are easily called from your Turbo C program.

The only thing left for you to do is write your program, test it for bugs, and program the EPROM. What could be easier!

The various files mentioned above are available from QSI as the **Q88 TurboPak**. The TurboPak supports both Turbo C 2.0 and Turbo C++. You can optionally use a QSI QDART or QDIO-1 board for the serial interface to your PC, which frees up the Q88 UART for use by your applications program.

The Q88 TurboPak is shipped free-of-charge with all Q88 boards. You must have your own copy of Borland's Turbo C and Turbo Debugger, and Paradigm's Locate and TDREM utilities.

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QSI CORPORATION

C-44 BUS SPECIFICATIONS

ELECTRICAL AND MECHANICAL GUIDELINES FOR DESIGNING ON THE C-44 BUS

C-44 BUS FEATURES

- Designed for battery-based applications.
- Wide operating temperature range: -40° to 85°C.
- Powered from one unregulated 8 to 18V supply (ideally suited for batteries).
- Ultra-low power STOP mode.
- Allows development of very cost-effective systems, even for line-powered operation.
- 8-bit multiplexed bus supports a variety of different microprocessors.
- 20-bit addressing allows up to 1 Mbyte of memory.
- Small boards (4.5" x 4.7") allow design of compact systems.
- Two user lines allow customizing to your application.

C-44 Bus products have seen a lot of the world in the last five years. These products have been into space several times aboard the Space Shuttle, have sat at the bottom of oceans for two-year periods, have floated around on the top of oceans, and have been on mountain tops, into coal mines, and on many factory floors.

C-44 Bus products excel in these types of applications which require a computer to operate from batteries and to operate over wide temperature ranges.

This Specification is designed to help C-44 Bus users build custom boards to operate on the bus, and to take advantage of the unique features of the bus.

GENERAL DESCRIPTION

The C-44 Bus consists of printed circuit cards which are 4.5" wide and 4.73" high. The bottom edge of the cards has a 44-pin edge connector, with double-readout contacts on standard 0.156" spacing. The available card cages space the cards 0.6" apart, which leaves room for 0.4" high components.

The 44 bus interface lines provide ground and power lines, a total of 8 data and 20 address lines, 5 different interrupts, and bus control lines.

The C-44 Bus also supports two low power modes: WAIT and STOP. These are described in detail in the **CPU Boards** section.

C-44 BUS HISTORY

The C-44 bus was developed by Onset Computer Corporation of North Falmouth, Massachusetts. Onset designed the bus specifically for battery-based, harsh environment use, and this is where it has found its greatest use.

Onset introduced the first C-44 boards in 1981; QSI introduced its first C-44 boards in 1983. The bus specifications underwent minor changes over the next two years, but has been completely defined since 1985.

BUS LINE DEFINITIONS

Table 1 lists the 44 bus lines on the C-44 Bus, along with the associated bus pin. The pins are numbered from 1 to 22 on the front of the PC board (left to right), and from A to Z on the back of the board; pin A is opposite of pin 1, and pin Z is opposite of pin 22.

In Table 1, lines which are in parenthesis, such as (A17), are optional, and may not be supported by all CPU boards. Lines preceded by a dash, such as -READ, are active low. Each of the bus lines is discussed separately below.

GROUND is on four different bus pins, and is the system ground as well as the negative terminal of the incoming power supply.

VBAT is on two bus pins, and is the positive terminal of the incoming power supply. Vbat must be from 8 to 18 volts DC. Many C-44 boards can operate on lower voltages, and some C-44 boards are limited to 16 volts maximum. See the data sheet for a given board to see the exact voltages allowed.

OLDBAT was originally used for the battery power supply, and is still used on some older boards. Do not use this pin for any purpose on new designs. If you are using older boards, you must connect OLDBAT to VBAT on your motherboard. All production motherboards do this.

VREG is on two bus pins, and is the 5-volt supply generated by the CPU board. VREG can be used by any peripheral boards. The section **CPU Boards** discusses the requirements this supply must meet. Note that Vreg is *not* supplied by an external voltage regulator.

Table 1. C-44 Bus line functions.

Pin	Function	Pin	Function
1	GROUND	A	GROUND
2	VBAT	B	VBAT
3	(-INTD)	C	-EXT RST
4	USER1	D	-RES OUT
5	(-INTC)	E	(-INTB)
6	-WAKE	F	-INTA
7	(A17)	H	A13
8	(A16)	J	A12
9	(-NMI)	K	A11
10	CLOCK	L	A10
11	USER2	M	A9
12	OLDBAT	N	A8
13	(A19)	P	AD7
14	(A18)	R	AD6
15	-READ	S	AD5
16	-WRITE	T	AD4
17	ALE	U	AD3
18	IO/-M	V	AD2
19	A15	W	AD1
20	A14	X	AD0
21	GROUND	Y	GROUND
22	VREG	Z	VREG

AD0 to AD7 are eight multiplexed address/data lines. During the first part of a read or write cycle (I/O or memory), address bits A0 to A7 appear on these lines. During the last part of a read or write cycle (I/O or memory), data bits D0 to D7 appear on these lines. Detailed timing is shown later in this document.

A8 to A15 are the upper eight address lines. These lines are valid throughout a bus cycle. During a memory access, these lines have memory address bits A8 through A15. During an I/O access, these lines have a duplicated port address; i.e. the same address as on address bits A0 to A7.

A16 to A19 are the extended address lines, which allow up to one megabyte of physical memory on the C-44 Bus. These lines are ignored by I/O-mapped devices. CPU boards which do not support these address lines must keep them at ground level. Note that some older C-44 memory boards do not decode these lines.

-INTA, -INTB, -INTC, -INTD are four maskable interrupt lines. Only -INTA is supported by all CPU boards; the others are optional. The operation of the CPU when one of these lines is asserted by a peripheral board depends on the specific CPU board being used. All CPU boards exit their WAIT and STOP modes when any supported interrupt line is asserted. All four interrupt lines are open-collector for wired-OR operation. The pullup resistor is on the system CPU board.

-NMI is a non-maskable interrupt line, and is optionally supported by the CPU board. As with the maskable interrupt lines, the actual CPU operation when this line is asserted by a peripheral board depends on the specific CPU board being used. All CPU boards exit their WAIT and STOP modes when any supported interrupt line is asserted. The -NMI line is open-collector for wired-OR operation. The pullup resistor is on the system CPU board.

-WAKE is similar to an interrupt line in that it wakes the CPU board out of its WAIT or STOP modes (described below). However, it does not cause any type of interrupt response; the CPU just resumes executing code from where it was when it entered the WAIT or STOP mode. The -WAKE line is open-collector for wired-OR operation. The pullup resistor is on the system CPU board.

-EXT RST is an external reset line which is an input to the CPU card. No other boards should be connected to this line. When this line is asserted, the CPU enters its reset state. When the line is disasserted, the CPU will begin its start up procedure.

-RST OUT is generated by the CPU card, and is the reset line by which all peripheral cards should be reset. This line will generally stay asserted for a few hundred milliseconds longer than the -EXT RST line.

USER1 and USER2 are two lines which are not used for any specific purpose, and can be used as desired by systems integrators. Many newer C-44 boards have optional uses for these lines, but none require the use of the lines.

CLOCK is a clock signal generated by the CPU board. It is from 0.5 to 3.0 MHz, depending on the specific CPU board. This line is infrequently used by peripheral boards, so some newer CPU boards have a switch to prevent the clock signal from reaching the bus, which reduces power and noise. The CLOCK line is always stopped by the CPU when the system is in the STOP mode.

-READ is the read-cycle strobe line, which indicates that the CPU is expecting the addressed peripheral device to send data to the CPU.

-WRITE is the write-cycle strobe line, which indicates that the CPU is sending data to the addressed peripheral device.

ALE is the Address-Latch-Enable line, which indicates when address bits A0 - A7 are valid on the AD0 - AD7 bus. Boards which need these address lines should latch them into a register on the falling edge of the ALE signal.

IO/-M is the Input-Output/-Memory line, which indicates whether a bus cycle is an I/O cycle or a memory cycle. The main difference between I/O and memory cycles is that an I/O address is 8-bits long and appears duplicated on A0 - A7 and A8 - A15 during I/O cycles. Address bits A16 - A19 are ignored by I/O-mapped peripheral cards during an I/O cycle.

TYPES OF C-44 BOARDS

All C-44 boards are one of two types: CPU boards or peripheral boards. Due to the battery-based design of the C-44 bus, CPU boards on this bus have some unique requirements as compared to more common buses.

CPU Boards must perform the following tasks:

- Generate the VREG system supply from the battery. This supply must operate from an 8 to 18 volt DC supply, and generate two output voltages: 5 volts $\pm 5\%$ and 3 volts $\pm 10\%$. A minimum of 75 mA of current must be available in the 5V mode, and a minimum of 10 mA in the 3V mode. Most CPU boards can operate from battery voltages lower than 8 volts.
- Support a **WAIT** mode, where the CPU halts all activity, stops bus operation, and waits for a signal to continue from an on-board device (such as a timer) or from any interrupt line or the -WAKE line.
- Support a **STOP** mode, which is similar to the WAIT mode except that the CLOCK line on the bus must stop, and VREG is lowered to its 3-volt state. The CPU is brought out of the STOP mode in the same fashion as for the WAIT mode.
- Drive all required bus lines with 6 mA sink or source capability.
- Ensure that all unused lines are held at an appropriate disasserted state.

Peripheral Boards for the C-44 Bus also have a few restrictions which must be met:

- They may not use more than 50 mA from the VREG supply in the 5-volt mode when they are accessed, or more than 10 mA from this supply when not accessed. If they need more VREG current than this, they must generate their own 5V power from the VBAT line.
- They may not use more than 30 μA from the VREG or VBAT supplies when in the STOP mode (VREG = 3V). If a peripheral board can remain active while the C-44 bus is in the STOP mode, then the peripheral board must not use more than 5 mA from VREG in this state.
- They must generate all other necessary on-board voltages from the VREG or VBAT supplies.
- They must not present more than three standard HCMOS loads on any one bus line.

BUS TIMING

The C-44 bus is not designed for ultra-high-speed processing; rather, it is designed for low-power operation. Because of this, the bus operates relatively slowly when compared to today's PC buses, such as the PC AT or Macintosh bus. In fact, memory boards on the C-44 bus can use ICs as slow as 450 nS; it is difficult to find memory ICs this slow anymore!

Figure 1 is a basic C-44 Bus timing diagram; the different symbols are defined in Table 2. These times represent the worst-case; i.e. peripheral boards have a minimum of 450 nanoseconds to provide data to the CPU during a read cycle, or to retrieve data during a write cycle.

No timing specifications are stated with respect to the CLOCK line. This line is asynchronous to all other bus operations, and should not be used for bus timing purposes. As indicated earlier, the CLOCK line is rarely used by peripheral boards in C-44 systems.

The manner and timing with which a given CPU respond to the various interrupt lines is dependent upon the actual CPU and the board design. Refer to the CPU board manual for information regarding this timing.

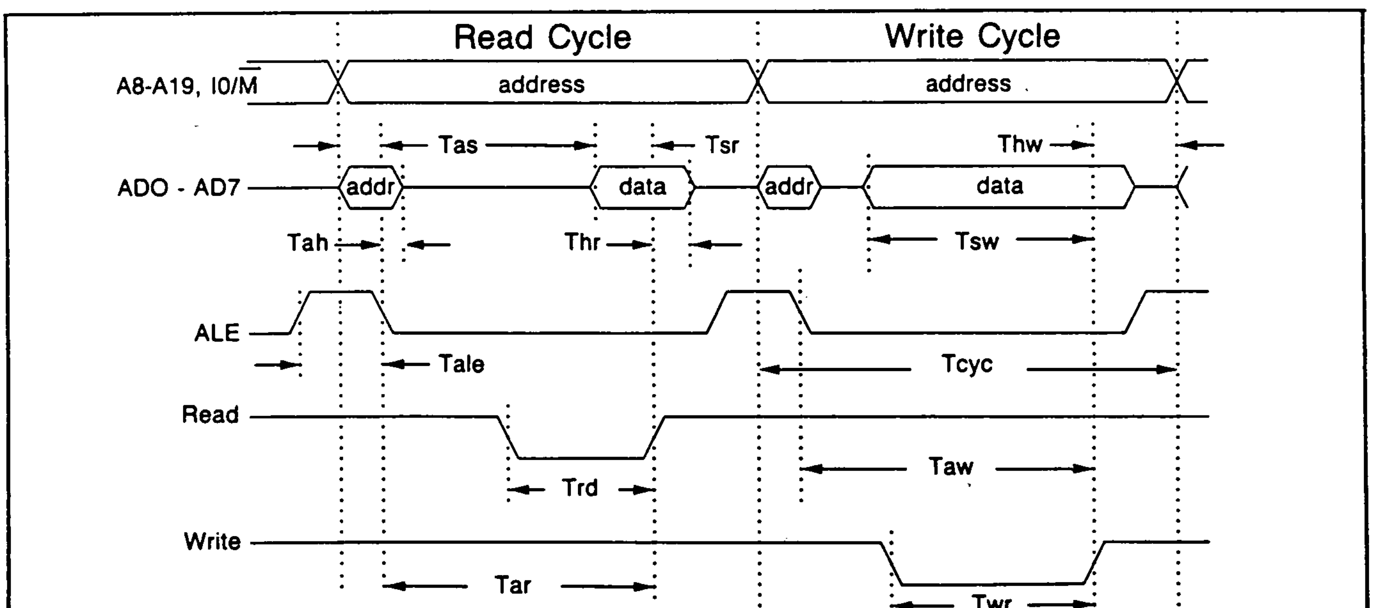

Figure 1. C-44 Bus timing diagram.

MECHANICAL REQUIREMENTS

C-44 Bus boards should be made from standard 1/16" (1.6 mm) thick printed circuit material. Figure 2 shows the physical size of the boards, the location of the connector and its polarizing slot, and the location of the four corner holes.

The holes marked "tooling" should be included on all C-44 boards. The upper right hole is used for the one required card ejector, and the lower left hole for a card-retaining rod in 2- and 3-slot card cages.

The holes marked "optional" may be included as required. The upper left hole is used for the optional card ejector. This ejector should be installed if there is adequate room on the board. The lower right hole is only for compatibility with older boards, and may be omitted.

The polarization slot is also optional. It is not present on some older C-44 boards, but is on all more recent production boards. We recommend its use, since it is the only feature which can help prevent inserting a board into a cage backwards, which usually has fairly disastrous consequences.

HINTS AND HELPS

Below are a number of ideas and suggestions for those who are producing C-44 boards and systems. These ideas have been accumulated from QSI's years of experience in this field.

Burn-In. There are many ways to help improve the reliability of an electronic product. Burn-in is one way to screen production products to ensure that reliability requirements can be met. An excess number of burn-in failures will indicate either design failures, or a component not meeting specification.

All of QSI's C-44 bus products are burned in for 24 hours, in this fashion: 3 hours at 85°C, 6 hours at -40°C, 6 hours at 85°C, 6 hours at -40°C, and 3 hours at 85°C. We recommend that all C-44 products destined for the field go through this or a similar burn-in process.

Testing at Temperature Extremes. If you are designing a board for use only at room temperature, this may not be necessary. However, if your product will be subjected to varying temperatures, it is advisable to test the product, during the development stage, to the expected temperatures. This is particularly true if your product will be running at very cold temperatures. It is far easier to design a C-44 Bus board to run at 85°C than at -40°C.

PAL Devices. A PAL is a device which allows an engineer to program a lot of logic into one IC. There are presently a number of CMOS PAL devices on the market. Unfortunately, the vast majority of these do not meet the criteria necessary for use on the C-44 bus. The main problem is that just because they are CMOS does not mean they use nearly zero power in the static state. Most "CMOS" PALs use from 10 to 60 mA of current in the static state, and even more when operating. This is far too much for most C-44 systems; such systems often use less than 10 mA *total*!

At the time of this writing, we know of four PALs which have near-zero standby current (less than 100μA), and use a few mA/MHz in operation: AMD/SEEQ DQ20RA10Z, Texas Instruments TICPAL16xx series, Altera EP600, and Harris HPL-16xx series. More are sure to be introduced in the future.

The two of these which QSI has used with success are the Altera EP600 and the AMD/SEEQ DQ20RA10Z. The Altera EP600 has better operational current consumption than the AMD/SEEQ part, and is substantially less expensive, but is less flexible and cannot be used in the STOP mode (it does not work at 3 volts).

The AMD/SEEQ DQ20RA10Z uses a little more power than the Altera EP600 when active, but it works fine in the STOP mode, and it is a more flexible part. If your application can justify the cost, the AMD/SEEQ part is well suited to many C-44 Bus designs.

If you use either of these (or any PAL for that matter), remember that their power consumption goes up significantly with the frequency of the inputs. For example, the Seeq part uses about 20 mA if it has a 1 MHz input, but only about 3.5 mA with a 100 kHz input. By carefully selecting which bus lines are used by the PAL, you can help minimize its power consumption.

For example, if you need to clear a flip-flop by inputting from or outputting to an I/O address, use the -WRITE line to decode for an output, since the -WRITE signal toggles much less frequently than the -READ signal.

Table 2. C-44 timing specifications.

Symbol	Description	Limit
Tcyc	Overall cycle length	750
Tale	ALE width	100
Tas	Address to ALE setup	75
Tah	ALE to address hold	75
Tar	Read access time	450
Trd	Read pulse width	300
Tsr	Data to Read setup	300
Thr	Read to data hold	0
Taw	Write access time	450
Twr	Write pulse width	300
Tsw	Data to Write setup	300
Thr	Write to data hold	50

All times are in nanoseconds, and are worst case for the most restictive cycle.

Card Ejectors. The card ejector used on C-44 boards is the Scanbe S-200, Bivar CP-06, or an equivalent made by many different manufacturers. Always install the right ejector, and when possible install the left ejector.

Current-sense resistors. Since C-44 systems are designed to operate from the absolute minimum current possible, a lot of effort is sometimes spent trying to determine how much power is being used and where it is going. Properly-located current-sense resistors make this task much easier. We recommend always putting 1Ω or 10Ω resistors in series with the VBAT and VREG lines for a board. It is often helpful to also put them in-line with the power pins of major ICs.

Finally, even if you do not wish to install the resistors, put two PC pads where they are accessible, and short them with a PC trace. This makes it easy to cut the trace and install the resistor if it becomes necessary to do so.

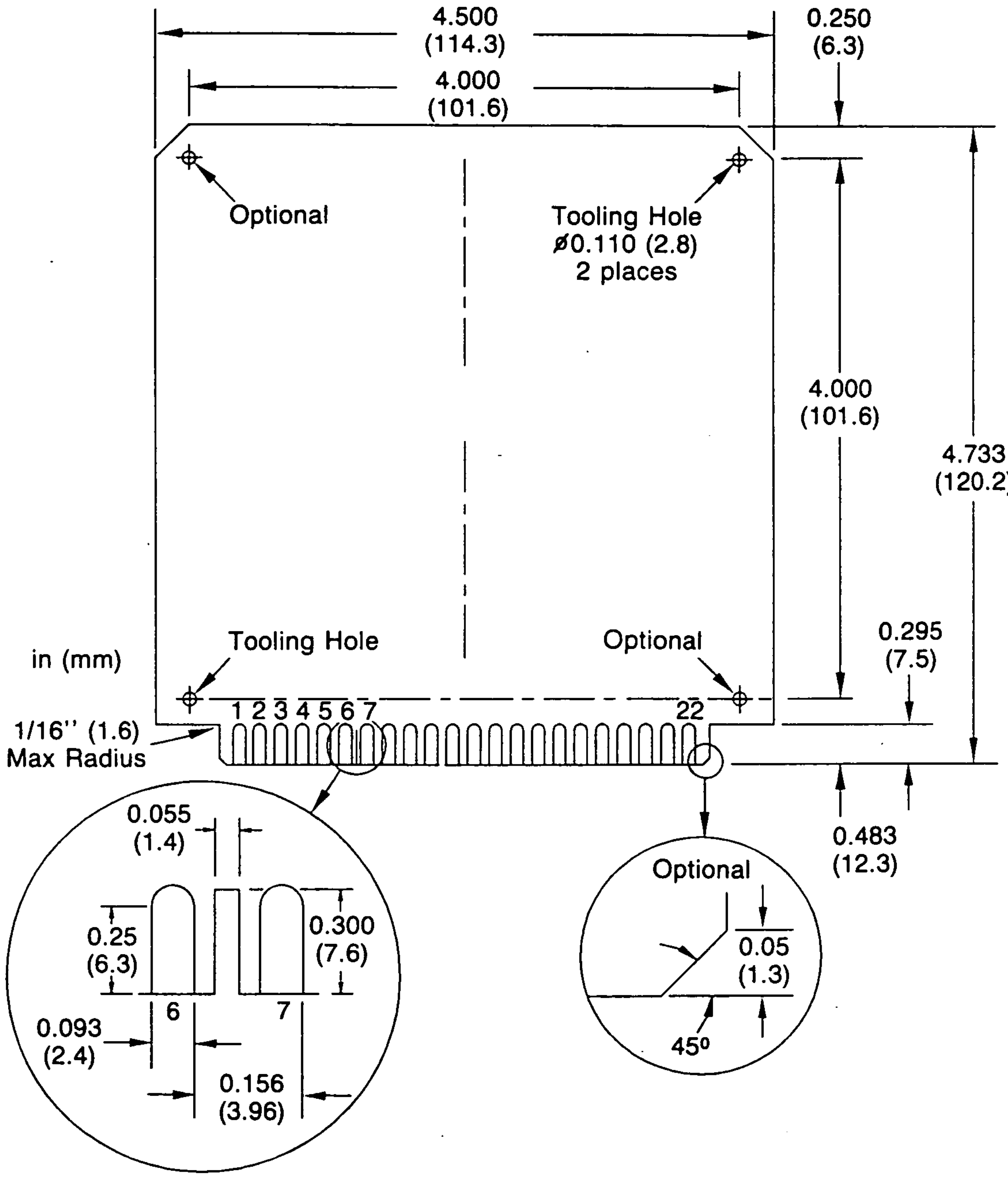

Figure 2. C-44 Bus mechanical requirements.

Edge Connectors. There are many manufacturers of edge connectors which can be used with the C-44 bus. The connectors must have dual-readout contacts on 0.156" centers, and usually have 0.2" row spacing. Compatible connectors are available with printed-circuit tabs, wire-wrap pins, and solder eyelets. Two representative connectors with printed-circuit tabs are:

Elco Corporation	00-6007-044-451-012
Edac Corporation	357-044-526-202

74XX Logic Families. There are a number of CMOS logic families available which can be used on C-44 bus boards. The most common by far is the 74HCxx "high speed" logic family. The older 74Cxx and CD4000 families can be used for on-board logic where speed is not critical, but are not fast enough for bus interface logic. The TTL-compatible 74HCTxx logic will operate well in all bus areas, but, since the input logic threshhold has been lowered to be TTL-compatible, the noise susceptibility is higher. Finally, the new "advanced" CMOS logic families such as 74ACxx and 74ACTxx (TTL compatible) will work well on C-44 Bus boards, but there is generally no need to pay the cost premium for these parts, since their high speed is not necessary for C-44 bus operation.

USER1 and USER2 Lines. The two USER lines have no required uses on any C-44 board, although some C-44 boards offer optional uses. We recommend that *every* C-44 board bring these two lines from the edge connector to solder pads on the board, so that they can easily be connected to if the need should arise.

16-bit CPU I/O Port Addressing. During an I/O access on the C-44 bus, the 8-bit port address is duplicated in both the A0-A7 lines and the A8-A15 lines. When using a CPU such as the 80C88, which has a 16-bit port address, be sure to use only port addresses in which the upper and lower bytes are the same. For example, if you need to access the (8-bit) port at address 37H, your 80C88 port address would be 3737H (16 bits). Since some peripheral boards decode the upper 8 bits and some decode the lower 8 bits, this is required to ensure proper operation.

ONSET COMPUTER CORPORATION

For more information about Onset Computer Corporation and their C-44 bus products and accessories, contact them at:

Onset Computer Corporation
199 North Main Street
PO Box 1030
North Falmouth, MA 02556-1030
Telephone: 508-563-9000
FAX: 508-563-9477

COMPARISON WITH ONSET

There are a few minor differences between QSI's and Onset's nomenclature with respect to the C-44 bus.

What QSI calls the WAIT mode is also called the WAIT mode by Onset. However, QSI's STOP mode is called the HYBERNATE mode by Onset. (HYBERNATE is an Onset trademark.)

The -WAKE line is called -WAKEUP by Onset and in some older QSI literature. We are moving to the -WAKE nomenclature with our newer literature.

We are now calling pin 12 the OLDBAT line, whereas Onset calls it the AUX BAT line. Older ONSET literature also refers to the two BATTERY lines as HCBAT (High-Current Battery); their newer literature uses the same name as we do.

In regards to actual use of the C-44 bus, all Onset C-44 products are compatible with all QSI C-44 products. About the only specification which is undergoing any change is that we are moving to a 16-volt maximum for VBAT, rather than the original 18-volt maximum.

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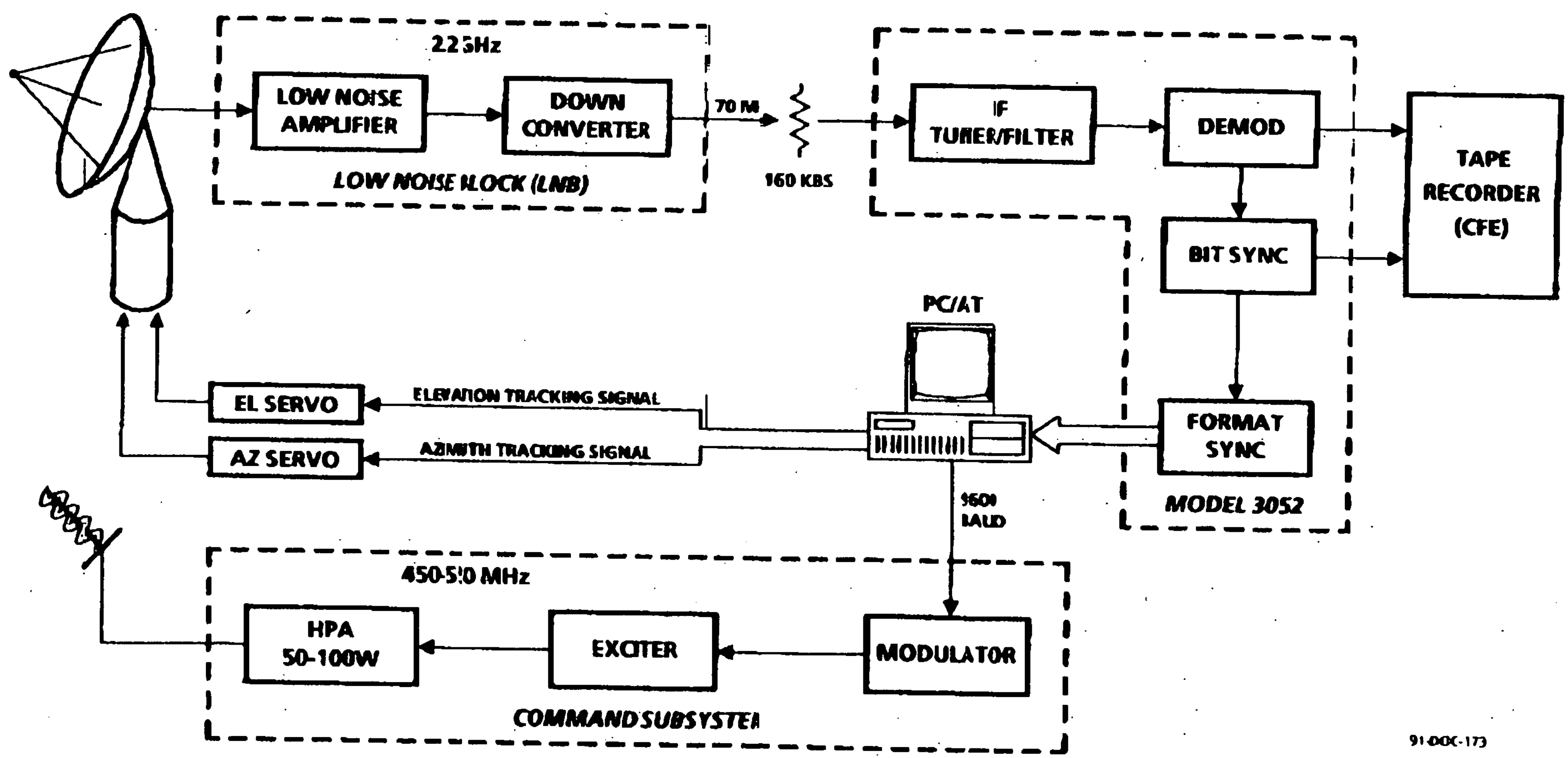
APPENDIX F

COMMUNICATIONS

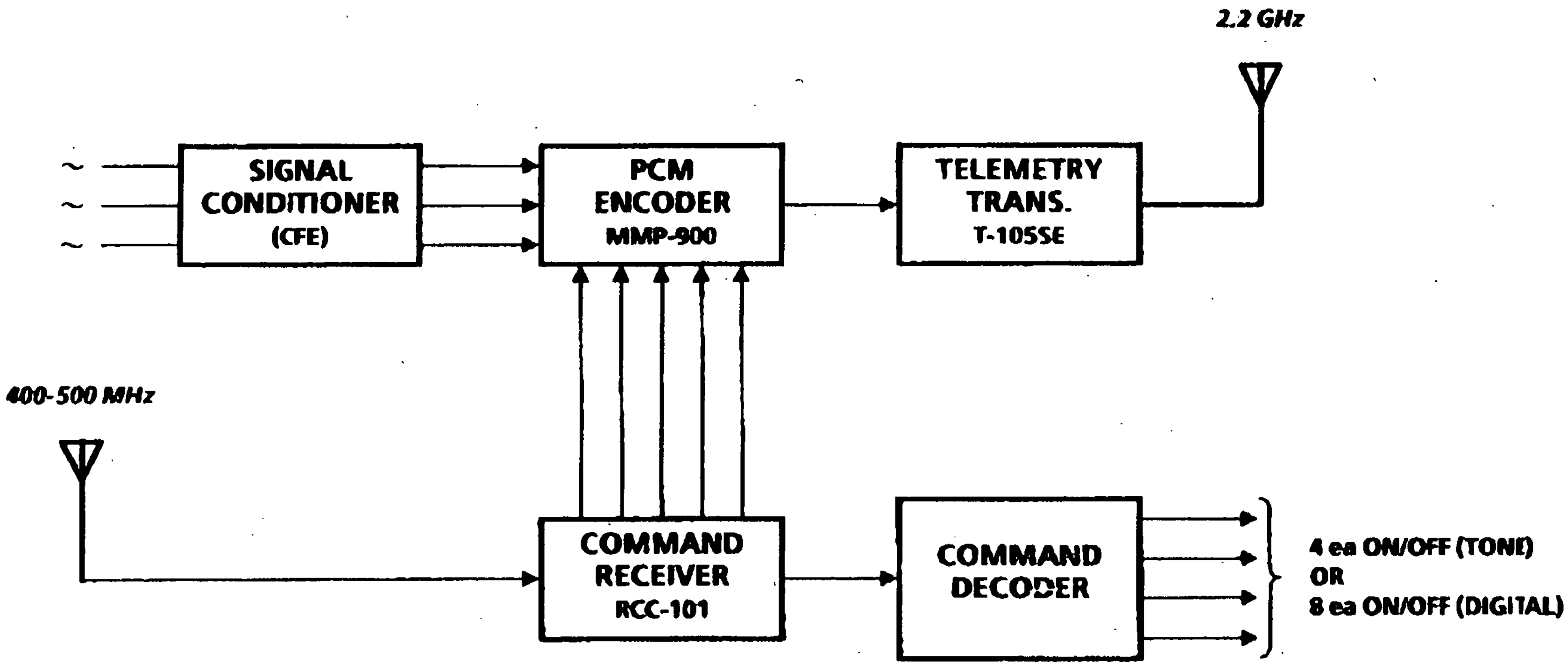
Tactical Telemetry, Tracking and Command (TTT&C) Station

OVERVIEW

Tactical Telemetry, Tracking & Conmand (TTT&C) Station Block Diagram

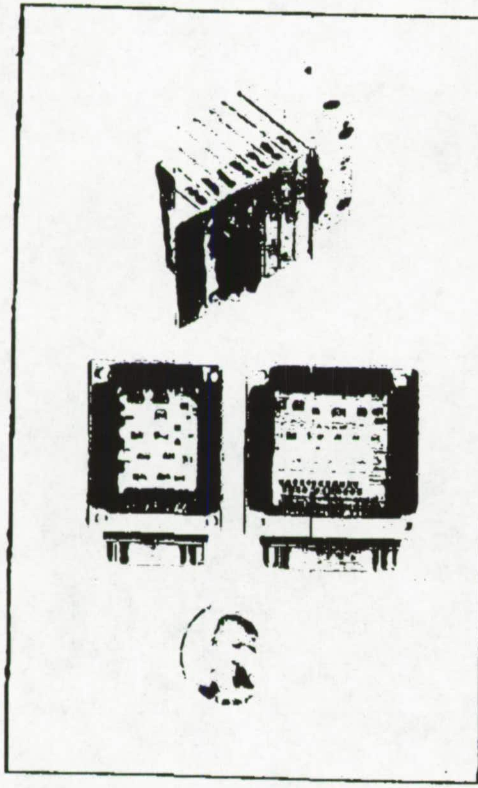


TELEMETRY & COMMAND SYSTEM (Flight Items)



PCM ENCODER

900-SERIMICROIINIATURE PROGRAMMABLE DATA SYSTEM



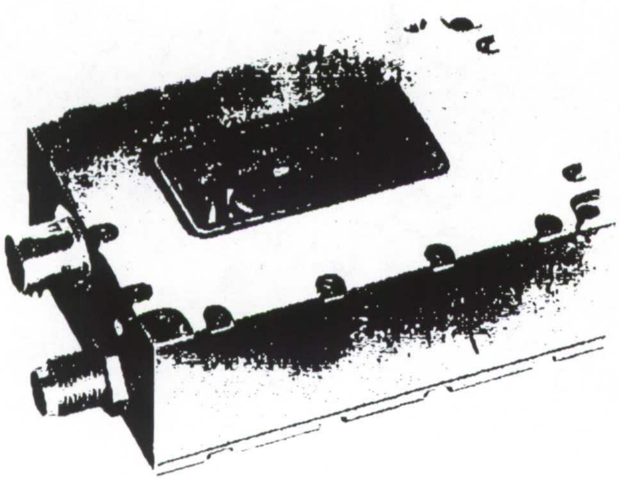
The 900 Series Data System has been designed to accommodate the various data inputs typically encountered in today's test and development activities. 900 Series Signal Conditioners and Encoders are configurations of individual-function hybrid circuit assemblies. The hybrids can be mounted in stackable frames or hermetically sealed packages. System packaging options are: MMP-900, stackable frames; MDS-900, hermetic packages on PC cards in customer-specified enclosures; and MDS-900HS, high shock packaging.

900 Series Encoders will directly accept these inputs: Low-level Analog, High-level Analog, Over-range Analog (Attenuated), Thermocouple Resistive Temperature Devices, Strain Gage, Accelerometer (Piezoresistive), Discrete Digital, Counted/Accumulated Digital, Serial Digital, Custom Interface.

User programmable parameters: Data Sampling Format, Data Sampling Rate, Analog Channel Gain and Offset, Frame Sync Code, System Bit Rate.

Data output options include any of these modulation codes: NRZ-L, M or S; Bi-Phase L, M or S; DM-M or DM-S; Randomized NRZ or Bi-Phase; PSK or FSK; Premodulated filtered or unfiltered.

TELEMETRY TRANSMITTER



T100S/L Series

- 2, 5, 8 or 10 watts minimum power
- Models in L and S telemetry bands
- Wideband response - dc to 1 MHz
- Minimum size and weight
- Power amplifier modules available for increased power output
- Meets latest IRIG specifications

Selected Specifications

RF Power Output:	2, 5, 8 or 10 watts minimum
Frequency Range (in MHz):	L-band, 1435 to 1540 S-band, 2200 to 2300
Frequency Stability:	±0.003%
Modulation Type:	True FM (PM available)
Deviation Sensitivity:	L-band, ±300 kHz/V rms (to ±750 kHz/V rms available) S-band, ±500 kHz/V rms (to ±1 MHz/V rms available)
Frequency Response:	dc to 500 kHz ±1.5 dB (to 1 MHz available)
Linearity:	L-band, 1% BSL for ±300 kHz deviation S-band, 1% BSL for ±500 kHz deviation
Modulation Distortion:	L-band, 1% max. ±300 kHz deviation S-band, 1% max. ±500 kHz deviation
Input Voltage:	28 ±4 Vdc

Input Current:

L-band	S-band	
1.0 amps	1.0 amps	for 2 watts out
*550 mA	650 mA	for 2 watts out
2.0 amps	2.0 amps	for 5 watts out
*1.5 amps	1.5 amps	for 5 watts out
2.5 amps	2.5 amps	for 8 watts out
*2.5 amps	-	for 10 watts out

*high efficiency

Size and Weight: 3.50" x 2.50" x 134" (excluding connectors)
16 ounces max.

TELEMETRY SYSTEM
ANTENNA SUBSYSTEM

The antenna parabolic reflector has the following characteristics:

CHARACTERISTIC	SPECIFICATION
Size	5 meter (16 ft.)
Gain	38.2 db @ 2.2 GHz, 55% efficiency
G/T	24.08 dB @ 20° elevation with 100°K LNA
Beam width @ 3 dB	< 1°
F/D Ratio	.375
Reflector	- Computer designed flattened expanded aluminum shaped to exact parabolic contours. - Aluminum prefabricated trussed ribs for shape rigidity at designed wind loads up to 125 mph. - Aluminum structural LNA mount, adjustable for focus and alignment. - 180° coverage through motorized polar mount.
Weight	500 lbs including mount
Base	- Fabricated structural steel welded members, designed for 125 mph winds normal to dish. - Elevation adjustments for exact latitude, rough and fine. - Azimuth correction adjustment built-in for low rise satellites. - Motor driven horizon to horizon polar tracking system.
Feed	Dipole element
Reflector Type	Mesh (1 cm)
Feed	Focal-Point mounted feed
Wind Operating	60 Mph (96 Kph)
Survival	125 Mph (200 Kph)
Temperature	0°F to 122°F (-17°C to 50°C)
Rain Rate	Up to 4" per hr (10 cm/hr)

TELEMETRY SYSTEM

FEED HORN

- The feedhorn is designed to operate in the 1600 to 2400 MHz frequency range.

- The feedhorn and pre-amp (LNB) are mounted in front of the reflector using a button hook or tripod type mount. The RF cables from the LNB are routed to the rear of the reflector through the mounting tube.

Polarization .... Linear
Feed .... Dipole element, focal point mounted

PRE-AMP POWER SUPPLY

Input Power .... 115 VAC, 1 amp
Output Power .... + 15 VDC
Size .... 6 x 8 x 8 inches (15.2 x 20.3 x 20.3 cm)
Temperature .... 0°F to 122°F (-17°C to 50°C)
Rain Rate .... Up to 4" per hour (10 cm/hr)

TELEMETRY SYSTEM

Pre-Amp and Down Converter

The Pre-Amp and Down Converter are mounted side-by side and are referred to as a LNB (Low Noise Block). The LNB designation indicates that the Low Noise Amplifier (LNA) and the Block Frequency Down Converter are packaged as a unit. The LNB has a gain of +91.6 dB with a Noise Figure of less than 1.0 dB. The Down Converter converts the S-band frequency to 70 MHz IF centered on 2210.0 MHz.

The LNB design provides the following features:

Size	2"H x 4"W x 8"D (5 x 10 x 20 cm)
Weight	3 lbs (1.36 Kg)
Operating Voltage	+ 15 VDC supplied by Pre-Amp Power Supply
Operating Current	1 amp
Filter Frequency	2200 to 2220 MHz
L.O. Stability	± 20 PPM over -10°C to +30°C
Temperature	0°F to 122°F (-17°C to 50°C)
Rain Rate	Up to 4" per hour (10 cm/hr)

TELEMETRY SYSTEM

IF Receiver

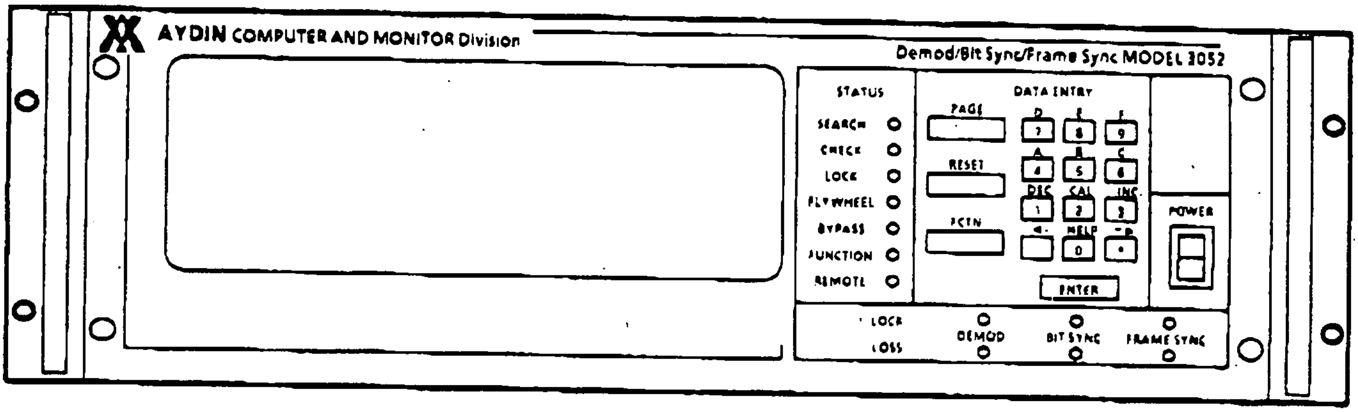
Operation of this receiver is completely automatic with no operator adjustments required. This receiver will acquire and demodulate a BPSK modulated signals in the intermediate frequency range 58-82 MHz. Frequency tuning is under synthesizer control and is programmable in 25 kHz increments from 58 to 82 MHz. A customer selected default frequency is provided with each receiver.

The receiver has an acquisition time of less than 2 seconds. Usable data is typically obtained immediately upon carrier lock. A meter type display provides a signal strength indication. The Receiver accepts the IF frequency of 70 MHz via a BNC type coaxial connector. The Receiver output is a RS-422 digital interface to a bit/frame synchronizer. The receiver mounts in a standard 19" equipment rack.

Size	1.5"H x 10"W x 12"D (3.8 x 48.3 x 30.5 cm)
Weight	5 lbs (2.3 Kg)
Operating Voltage	115 VAC, ± 10%, 47-63 Hz
Power	4 watts
IF input	70 MHz IF via RG-58 coax cable
Signal Output	RS-422 TTL 2 wire (twisted pair)
Temperature Range	Normal office environment - 59°F to 86°F (15°C to 30°C)
Humidity	Up to 80% non-condensing
Finish	Black urethane with white engraving
Mounting	19" wide rack or desk-top

MODEL 3052

Demodulator/Bit Synchronizer/Frame Synchronizer



FEATURES

- BPSK Subcarrier Demodulator
- PCM Bit Synchronizer
- IRIG 106-86 Format Synchronizer with dual output interfaces
- Test-Simulator/Modulator
- Gas Plasma Front Panel Display
- IEEE-488 or RS-232/422 Remote Programming Interface

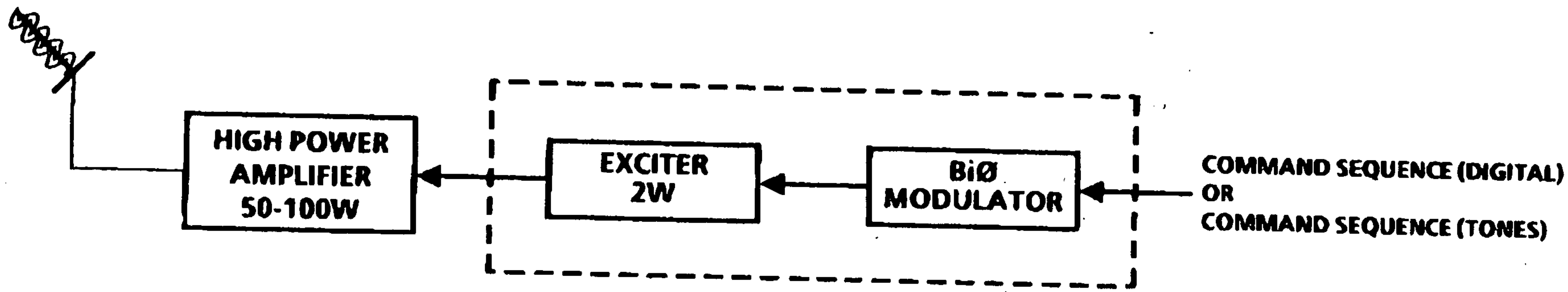
DESCRIPTION

The Model 3052 Demod/Bit Sync/Frame Synchronizer combines the functions of BPSK Subcarrier Demodulation, PCM Bit Synchronization and Format Synchronization into a single remotely -programmable unit. Per customer specifications, the unit accepts a fixed subcarrier frequency in a range of 10 KHz to 2 MHz, at a data bandwidth of up to 6 times the bit rate and at a fixed bit rate in the range of 500 bps to 500 kbps. A fully programmable frame and subframe synchronizer accepts IRIG 106-86 PCM telemetry formats.

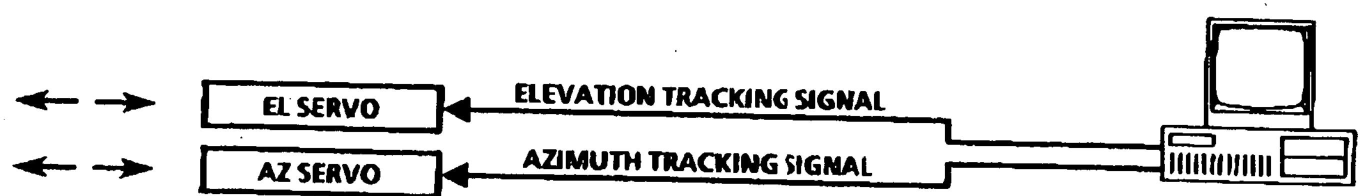
The Model 3052 is ideally suited for use in Telemetry Command and Control (TC&C) Links, Fixed Rate Satellite Downlinks and Standard Telemetry Data Receiving Station applications.

AYDIN COMPUTER AND MONITOR Division

COMMAND SUBSYSTEM



TRACKING SUBSYSTEM



AYDIN COMPUTER AND MONITOR DIVISION

COMMAND SYSTEM

UHF MULTIFREQUENCY TRANSMITTER SPECIFICATIONS

- Uplink frequencies of 430, 547, 550, and 553 as a minimum.
- All Solid State electronics.
- Capable of FM modulation of at least ±200 KHz.
- Output power of from 50 to 200 watts.
- Built in Deviation meter and Power meter.
- Crystal controlled frequencies accurate to ±0.003% of the selected frequency.
- Harmonic and Spurious outputs in accord with IRIG Doc. 106-73.
- Deviation Sensitivity of ±125 KHz/volt RMS.
- Deviation adjustment on front panel of unit.
- Frequency response from DC to 500 KHz.
- Total Harmonic distortion of 1% maximum at ±125 KHz deviation.
- Built in bandpass filter covering desired range of frequencies.
- Operation into any load condition without damage to unit.
- High and Low power switch on front panel with Low power output of around 2 watts.
- Power requirements of 110-130/210-240 VAC, 45-65 Hz, at less than 10 Amps (depending on final output power).

COMMAND DECODER SPECIFICATIONS

PHYSICAL DIMENSIONS

SIZE	6.25" L x 5.25" W x 2.25" H
WEIGHT	Approximately 2 lbs

ELECTRICAL

POWER	+ 28 Vdc
INPUT	BIØ-L PCM CODE FROM VIDEO OUT OF AIRBORNE RECEIVER, 3.906 KBITS, MIN OF 50mV AMPLITUDE (Generated from command encoder)
OUTPUT	TTL OR CMOS LEVELS (0 TO + 5V), 8 PARALLEL BITS PLUS GROUND.
CONNECTORS	CANNON "D" TYPE

ENVIRONMENTAL

OPERATING TEMPERATURE	0°C TO + 65°C
ENVIRONMENTAL TESTING	IN ACCORDANCE WITH "AFGL AIRES ENVIRONMENTAL TEST SPECIFICATIONS".

AYDIN COMPUTER AND MONITOR DIVISION

COMMAND SYSTEM

COMMAND RECEIVER (FLIGHT ITEM) SPECIFICATIONS

- Fix tuned Frequency range -- 400 MHz to 550 MHz
- Internal demodulator for FM
- Input noise figure 6 db maximum
- Frequency stability ±.002% (-20 to +71°C)
- Time delay variation: ±100ns (-90 dbm to -20 dbm)
- Safe RF input level of up to 2 volts RMS
- Video Bandwidth (±1.5 db) of 330 KHz
- 3db IF bandwidth of 550 KHz
- Adjustable video output -- up to 5V peak/peak ±.5V at ±200 KHz deviation into a 10K ohm load
- Image rejection 65db minimum
- Squelch adjustable from 0 to 5 microvolts
- Input power requirements 22 to 36 volts DC (1.5 watts at 28V DC)
- Vibration: 30g random (20 to 2000 Hz)
- Shock: 100g for 11 milliseconds
- Altitude: Unlimited
- Temperature: -20°C to +71°C
- Size: approx. 4 X 2.5 X 1.5 inches (LWH)
- Weight: approx. 8 ounces
- Signal Strength output - approx. 3 volts DC with a -75 dbm carrier level -- variable with RF carrier level
- SMA RF input connector

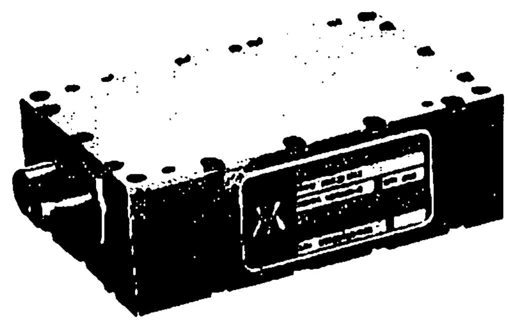
COMMAND SYSTEM

COMMAND ANTENNA (FLIGHT ITEM) SPECIFICATIONS

- Stub Type Quadraloop Antenna
- 215 to 575 MHz - user specified when ordering
- Nominal bandwidth: 4 MHz
- Antenna Material: Stainless Steel
- SMA connectors for RF cable
- Suitable for various payload diameters - spec when ordering
- RF cable assembly included
- Nominal impedance of 50 ohms

AYDIN COMPUTER AND MONITOR DIVISION

COMMAND RECEIVER



RCC-100 Series

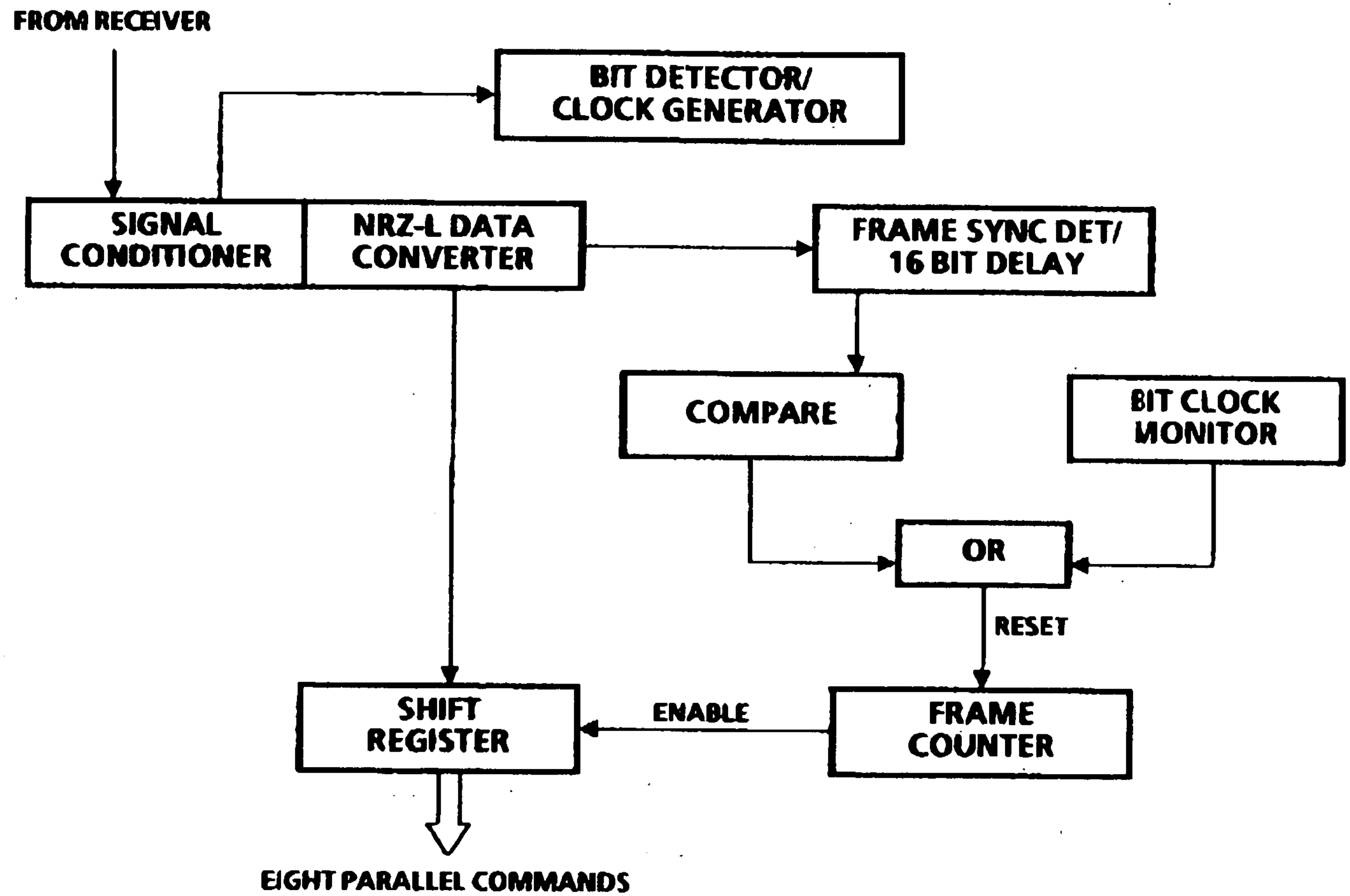
- Designed for guidance and subsystem actuation applications
- Models in three frequency ranges
- High sensitivity and stability
- Mounting provisions for stacking TDC-100 Tone Decoders

Selected Specifications

Frequency Ranges, Model Designations and Noise Figures:	200-290 MHz, RCC-101, 4 dB max. 290-400 MHz, RCC-102, 5 dB max. 400-550 MHz, RCC-103, 6 dB max.
Frequency Stability:	±0.005%
Image & Spurious Response Rejection:	60 dB minimum
Maximum Signal Level:	2 V rms
Dynamic Range:	80 dB minimum
Pre-detect IF Bandwidth:	one of six standard filters is used to provide a 3 dB signal bandwidth of 200, 400, 500, 800, 1200 or 2400 kHz. Other bandwidths are optional.
Audio Output Load:	600 ohms
Audio Response:	±1.5 dB, 100 Hz to ¼ of the Pre-detect IF Bandwidth minimum
Audio Output Sensitivity:	0.01 V peak per kHz deviation (9.0 V p-p max.)
Audio Output Distortion:	2% max. for a p-p deviation of ¼ the Pre-detect IF Bandwidth
Signal Strength Analog:	0 to 4 Vdc nominal into a 10 kilohm load
Input Voltage and Current:	24 to 36 Vdc at 35 mA max.
Size and Weight:	3.87" x 2.50" x 1.32" (excluding connectors) 13 ounces max.

271-0003 TTT

UPLINK COMMAND DECODER



- ONE FRAME = 8 BITS FRAME SYNC + 8 BITS COMMAND (16 BITS)
- COMPARES 32 FRAMES BEFORE COMMAND IS ENABLED. IF ANY BIT FAILS TO COMPARE, COMMAND IS DISABLED UNTIL 32 PERFECT FRAMES ARE SEEN

AYDIN COMPAND MONITOR DIVISION

TELEMETRY L CALCULATIONS

The telemetry link calculations are based onnum of 5 Watts (EIRP) output power at the airborne transmitter and 160kB/Sec NRZ-L dtes. Command calculations are based on 100 watts output power at the ground station.

Transmitter carrier frecuency (MHz) .... 2200 S-Band
LNA noise figure (dB) 1
Receiver Noise Figure (dB) 12
RANGE .... 650 Km
GAINS: Airborne Transmitter Power: .... 5 W 37 dBm (EIRP)
Transmitter antenna gain: .... 0 dB
Receive antenna gain: .... 38 dB
LOSSES: L1 = 1 (dB) Cable Loss betweeving antenna and pre-amp
LM = 3 (dB) Misc. losses like w, fading multipath etc.
LP = 3 (dB) Polarization losses
LT = 1 (dB) Losses between trer and airborne antenna
L2 = 14 (dB) Losses between Ld receiver
Free Space Loss: LS(dB) = 32.5 + 20hge in km) + 20 log(F in Mhz)
LS(dB) = 33 + 56 + l
LS = 156 dB
Bit Rate and Signal Type for Calculations .... 160 Kb/s NRZ-L
Received Signal Level (at Preamp input) .... -79 dBm
Calculated Peak Deviation for NRZ-L sign.............................. 59150 Hz
Calculated IF Bandwidth of the receiver .... 177450 Hz
NOISE: Carrier Noise (Nc) is (dBm): .... -122
Total system noise is N + N Feq .... -120
For FM improvement desired level is min 12 dB above noise level
Therefore ; Carrier to Noise Ratleceived Signal Level - Noise Power
C/N (dB) 41
LINK MARGIN (dB): 29
Based cal + 12 dB C/N ratio for full FM improvement

COMMAND LINK CALCULATIONS

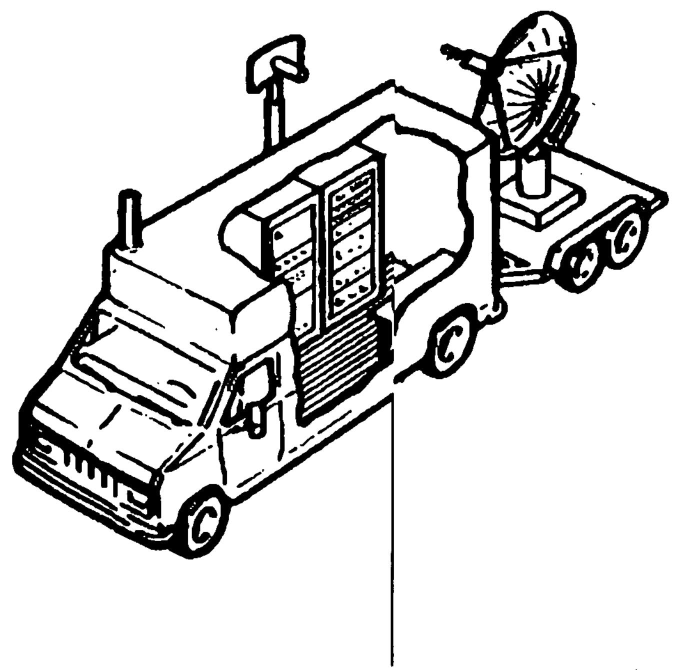
Transmitter carrier frequency (MHz) .... 550 P-Band
Receiver Noise Figure (dB) 6
RANGE .... 650 Km
GAINS: Transmitter Power .... 100 W 50 dBm (EIRP)
Transmitter antenna gain .... 17 dB
Receive antenna gain .... 0 dB

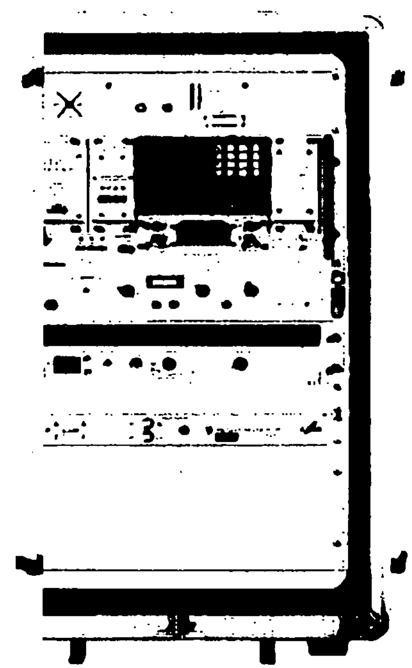
LOSSES: L1 = 3 (dB) Cable Loss between transmitter and antenna
LM = 3 (dB) Misc. losses like weather, fading multipath etc.
LP = 3 (dB) Polarization losses
LT = 1 (dB) Losses between airborne antenna and receiver

Free Space Loss : LS(dB) = 32.5 + 20 log(Range in km) + 20 log(F in Mhz)
LS(dB) = 33 + 56 + 54
LS = 143 dB

Receiver Noise Figure (NF) .... 6 dB
Noise Level .... -114 dBm + 10 log (0.4) + NF
= -114 dBm + 4 dB + 6 dB
= -112 dBm
Receiver Threshold .... -92 dBm for 20 dB for C/N
Received Signal Level (at receiver input) .... -86 dBm

TYPI(AL CONFIGUFION





Where a vehicle under test or conpl does not lie within a direct linet to a central facility due to adverse topography or physical distance, viable alternative must be provid: mobile telemetry and communication system supplies this alternate. The mobile system may simpl the signal to the central site or may perform sophisticated processing :pendent upon user need. Aydines turnkey mobile systems for telecommunication and telemetry prossing that are fully customized tos specific requirements.

APPENDIX G

POWER SYSTEM

THERMION-I MODEL
THIS MODEL RELATES ORBIT AVERAGE POWER TO SOLAR CELL AREA,
THE NUMBER OF CELLS IN THE BATTERY, AND SATELLITE LIFE TIME WITHOUT CHARGING
FROM THE SOLAR CELLS.
NOTE: Thermion-I uses alot more cells than it needs steady state, but initially it will take time to orient towards the Sun.

VARIABLE NAME	VALUE	UNITS	DESCRIPTION
POWER GENERATION:			
n=	0.115		Efficiency of solar cells
degradation/ year	0.06		degradation of the cells performance in a year due to exposure
Sun Angle off normal=	±6.5	degrees	angle between incident rays and array normal
solar intensity=	1350	W/m^2	
Temperature coefficient=	-0.005	/C	degradation in performance of array due to temperature
Temperature of SC=	50	C	
Temperature degradation=	-0.11		
Packaging eff.=	0.90		fraction of array taken up by semiconductor and not electrical paths, etc
Array voltage desired=	28.00	V	
Array voltage designed=	33.60	V	
POWER STORAGE:			
Cell Chemistry	Nickel Cadmium		
Cell Type	'D'		
Cell Voltage=	1.20	V	
Cell Mass=	0.155	Kg	
# Of Cells=	36	Cells	# of cells on Thermion-I
Battery Mass=	5.58	Kg	
Cell Capacity=	5.40	A•hr	Energy Capacity of one cell
Cell Capacity=	6.48	W•hr	Energy Capacity of one cell
DOD=	0.70		Depth of discharge of the cells
Total Battery Capacity=	233.28	W•hr	Battery energy Capacity At 100% DOD
Actual Battery Capacity=	163.30	W•hr	Battery energy Capacity At rated DOD
Time to discharge=	21.77	Hr	Time to discharge battery at orbit average load if no charging
Orbit average power=	7.50	W	Average amount of power used by the satellite
Orbit period=	1.5	hrs	Time for satellite to circle Earth
Time in shadow=	0.60	hrs	Time array is not seeing sun/orbit
Total Capacity=	0.23	A•Hr	Total capacity required
Battery Charge =	12.86	W	Power into battery during charging
SOLAR ARRAY DEFINITION:			
Total Power=	20.36	W	Power Needed from Solar Array
Array Capacity (BOL)=	24.92	W	Beginning of life array capacity
Total Cell Area=	0.160	m^2	area taken up by solar cell array
Actual Array Size=	0.178	m^2	Actual area taken up by solar cell array including packin efficincy
Square array Side Length=	0.422	m	Length of a square array of solar cells
Square array Side Length=	16.625	in	Length of a square array of solar cells

(Ratings of CADNICA Batteries)

■General-use CADNICA batteries (Standard Series)

Type	Model	Nominal voltage (V)	Capacity (mAh) at 0.2C rate		Standard charge		Quick charge		Internal resistance (mΩ)	External dimensions (including tube)		Weight (approx.g)
			Minimum	Typical	Current (mA)	Time (hr.)	Current (mA)	Time (hr.)		Diameter (D) (mm)	Height (H) (mm)	
	N-50AAA	1.2	50	55	5	14~16	15	4~6	55.0	$10.5^{0}_{-0.5}$	16.0^{0}_{-1}	3.5
	N-110AA	1.2	110	120	11		33		30.0	$14.5^{0}_{-0.5}$	17.0^{0}_{-1}	8
	N-120TA	1.2	120	130	12		36		34.0	$7.8^{0}_{-0.5}$	42.5^{0}_{-1}	7
	N-150N	1.2	160	170	15		45		27.0	$12.0^{0}_{-0.5}$	29.5^{0}_{-1}	9
	N-200AAA	1.2	200	220	20		60		21.0	$10.5^{0}_{-0.5}$	44.5^{0}_{-1}	10
	N-200A	1.2	200	225	20		60		20.0	$17.0^{0}_{-0.5}$	17.0^{0}_{-1}	11
	N-270AA	1.2	270	305	27		81		15.0	$14.5^{0}_{-0.5}$	30.0^{0}_{-1}	14
	N-500A	1.2	500	525	50		150		9.0	$17.0^{0}_{-0.5}$	28.0^{0}_{-1}	20
	N-600AA	1.2	600	650	60		180		12.0	$14.2^{0}_{-0.5}$	50.0^{0}_{-1}	24
	N-650SC	1.2	650	700	65		—		6.0	23.0^{0}_{-1}	26.0^{0}_{-1}	29
	N-1000SC	1.2	1000	1100	100		—		4.8	23.0^{0}_{-1}	34.0^{0}_{-1}	42
	N-1100C	1.2	1100	1200	110		—		4.6	26.0^{0}_{-1}	30.0^{0}_{-1}	44
	N-1300SC	1.2	1300	1450	130		—		4.2	23.0^{0}_{-1}	43.0^{0}_{-2}	50
	N-1800C	1.2	1800	2100	180		—		4.1	26.0^{0}_{-1}	50.0^{0}_{-2}	80
	N-4000D	1.2	4000	4600	400				3.3	34.0^{0}_{-2}	61.0^{0}_{-2}	160
	N-6PT	7.2	120	130	12		24	7~8	210.0	17.0(W)×26.0(L)×48.5(H)		42

Operating temperature range ; Charge: 0°~45°C (standard), 10°~45°C (quick); discharge: −20°~60°C; storage: −30°~50°C (−30~35°C for long periods)
Note: Consult Sanyo concerning operating conditions for quick charging of N-650SC or higher models.

■Standard CADNICA batteries (KR Series)

Type	Model	Nominal voltage (V)	Capacity (mAh) at 0.2C rate		Standard charge		Internal resistance (mΩ)	External dimensions (including tube)		Weight (approx.g)
			Minimum	Typical	Current (mA)	Time (hr.)		Diameter (D) (mm)	Height (H) (mm)	
	KR-1300SC	1.2	1300	1450	100	14~16	6.0	23.0^{0}_{-1}	43.0^{0}_{-2}	48
	KR-2000C	1.2	2000	2200	200		5.2	26.0^{0}_{-1}	50.0^{0}_{-2}	75
	KR-2800D	1.2	2800	3200	280		4.5	34.0^{0}_{-2}	44.0^{0}_{-2}	110
	KR-4400D	1.2	4400	4800	440		3.8	34.0^{0}_{-2}	61.0^{0}_{-2}	150
	KR-7000F	1.2	7000	7500	700		3.4	34.0^{0}_{-2}	91.0^{0}_{-2}	230
	KR-10000M	1.2	10000	12000	1000		2.6	43.0^{0}_{-2}	91.0^{0}_{-2}	400
	KR-20000M	1.2	20000	24000	2000		2.5	43.0^{0}_{-2}	146.0^{0}_{-2}	660

Operating temperature range ; Charge: 0°~45°C (standard); discharge: −20°~60°C; storage: −30°~50°C (−30~35°C for long periods)
Note: When using assembled batteries consisting of KR-4400D or higher model batteries, consideration must be given to the problem of cell temperature increase. Sanyo can provide assembled batteries that meet your specific conditions of use.

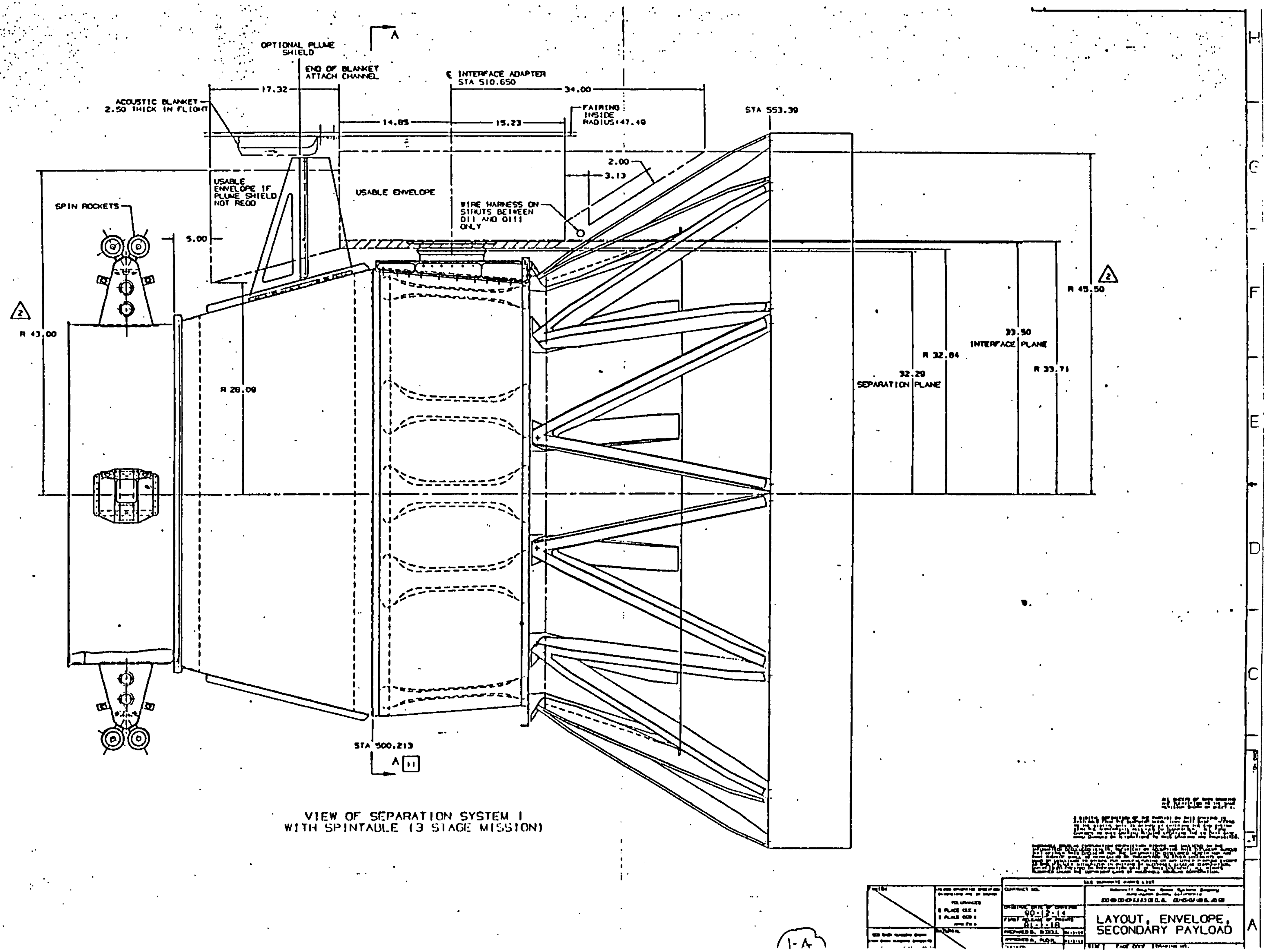
■High-capacity CADNICA batteries (E Series)

Type	Model	Nominal voltage (V)	Capacity (mAh) at 0.2C rate		Standard charge		Internal resistance (mΩ)	External dimensions (including tube)		Weight (approx.g)
			Minimum	Typical	Current (mA)	Time (hr.)		Diameter (D) (mm)	Height (H) (mm)	
	N-225AE	1.2	225	245	23	14~16	20.0	17.0^{0}_{-1}	17.0^{0}_{-1}	12
	N-600AE	1.2	580	630	60		8.5	17.0^{0}_{-1}	28.0^{0}_{-1}	22
	N-700AAE	1.2	700	770	70		11.0	$14.2^{0}_{-0.5}$	50.0^{0}_{-1}	27
	KR-1000AE(L)	1.2	950	1050	100		8.0	17.0^{0}_{-1}	43.0^{0}_{-2}	31
	KR-1200AE	1.2	1200	1300	120		7.6	17.0^{0}_{-1}	50.0^{0}_{-2}	34
	KR-1700AE	1.2	1700	1850	170		7.0	17.0^{0}_{-1}	67.0^{0}_{-2}	45
	KR-1700SCE	1.2	1700	1850	170		5.5	23.0^{0}_{-1}	43.0^{0}_{-2}	53
	KR-2000SCE	1.2	2000	2200	200		5.5	23.0^{0}_{-1}	50.0^{0}_{-2}	63
	KR-2400CE	1.2	2400	2650	240		5.0	26.0^{0}_{-1}	50.0^{0}_{-2}	76
	KR-5000DE	1.2	5000	5400	500		3.5	34.0^{0}_{-2}	61.0^{0}_{-2}	155

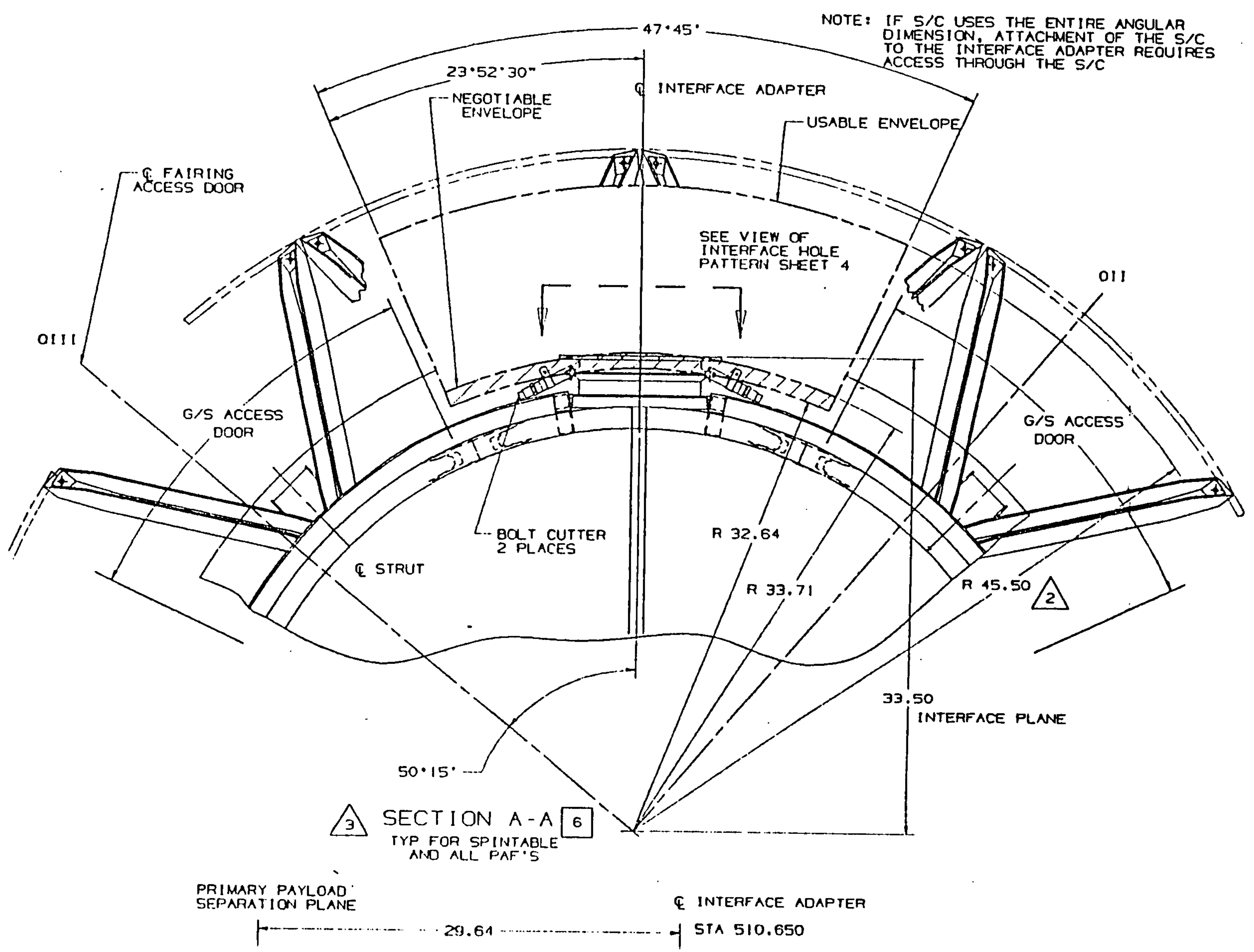
Operating temperature range ; Charge: 0°~45°C (standard); discharge: −20°~60°C; storage: −30°~50°C (−30~35°C for long periods)
Note: Consult Sanyo concerning 1-hour charge.

APPENDIX H

LAUNCH VEHICLE INTERFACE AND DEPLOYMENT

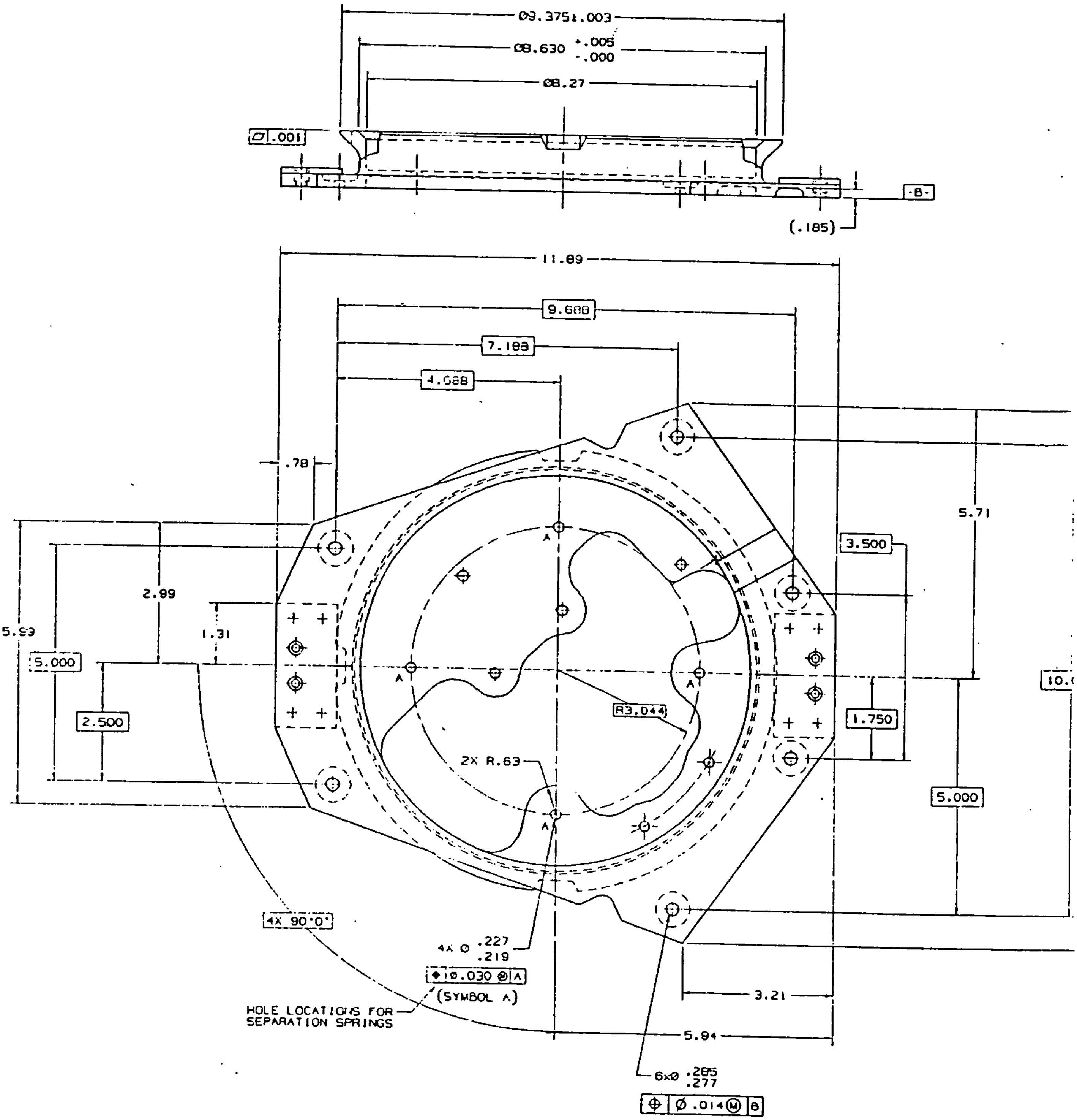


VIEW OF SEPARATION SYSTEM I
WITH SPINTABLE (3 STAGE MISSION)



3 SECTION A-A 6

TYP FOR SPINTABLE AND ALL PAF'S



NOTE: DIFFERENT CONFIGURATIONS OF THE INTERIOR OF THE ADAPTER ARE AVAILABLE TO ACCOMODATE SEPARATION SWITCH DESIGNS.

COUNTER BORE Ø .750 X R.09
.170
.160 DEEP

INTERFACE HOLE PATTERN

VIEW OF PAYLOAD INTERFACE ADAPTER

ROTATED 90° CCW FOR SYSTEM 1
(IE DELTA FWD ⟶)

ROTATED 97° CCW FOR SYSTEM 2
(IE DELTA OUTBOARD ⟵)

ROTATED 94°40' CCW FOR SYSTEM 3
(IE DELTA FWD ↑)

NOTES:

1. ENVELOPES GIVEN ON SHEETS 1 THRU 3 OF THIS LAYOUT ARE FOR STRUCTURAL SYSTEMS OF EXISTING DESIGN. SYSTEM 1 WILL BE FLOWN ON THE LOSAT-X MISSIOIN, SYSTEM 2 WAS FLOWN ON THE UOSAT MISSIONS, AND SYSTEM 3 WILL BE FLOWN ON THE SEDS-1 MISSION.

   ENVELOPES DO NOT ACCOUNT FOR ANY (PRIMARY) MISSION SPECIFIC BRACKETRY. THEY APPLY TO MISSIONS USING A Ø9.5 FT OR A Ø8 FT FAIRING ONLY. THE Ø10 FT FAIRING HAS A RING FRAME THAT SIGNIFICANTLY AFFECTS THE ENVELOPES SHOWN.

   THE STANDARD ACOUSTIC BLANKET CONFIGURATIOIN IS SHOWN. MODIFICATIONS TO THE BLANKETS MAY BE MADE TO ACCOMODATE ENVELOPE PROTRUSIONS.

   ENVELOPES ACCOUNT FOR DYNAMIC DEFLECTIONS BETWEEN THE PAYLOAD AND THE LAUNCH VEHICLE. DEFLECTIONS WERE TAKEN FROM THE LOSAT-X MISSION.

   PAYLOAD MASS AND C.G. REQUIREMENTS WILL BE PROVIDED AS PART OF THE MISSION INTEGRATION SERVICES.

2. MISSIONS FLOWN WITH AN AIR FORCE PRIMARY PAYLOAD REQUIRE A 1.00 INCH MINIMUM CLEARANCE. AFTER ACCOUNTING FOR DEFLECTIONS BETWEEN THE PAYLOAD AND THE FAIRING, THE NOTED DIMENSION MEETS THIS REQUIREMENT. MISSIONS FLOWN WITH NASA PRIMARY PAYLOADS MAY USE AN ADDITIONAL .75 INCH OF ENVELOPE. THE RESULTANT CLEARANCE WILL BE .25 INCH MINIMUM.

3. THE SYSTEM SHOWN IN THE NOTED VIEW CAN BE POSITIONED IN THE ANGULAR LOCATION INDICATED AND/OR ROTATED 180° ABOUT THE GUIDANCE SECTION.

4. THE NOTED ENVELOPE IS NEGOTIABLE BECAUSE THE PRIMARY PAYLOAD MAY EXTEND INTO THIS AREA. MDSSC WILL COORDINATE THE ALLOWABLE LOCATIONS OF PRIMARY AND SECONDARY PAYLOAD HARDWARE IN THIS AREA.

5. THE BOLT CUTTER MAY BE ROTATED TO AVOID INTERFERENCE WITH THE PAF; HOWEVER, ADDITIONAL DESIGN WORK IS REQUIRED TO DETERMINE WHICH ORIENTATION, IF ANY, SOLVES THE PROBLEM. ROTATING 90° DOES NOT WORK BECAUSE THE BOLT CUTTER WOULD THEN INTERFERE WITH THE DEPLOYER SYSTEM.

6. THE FLYOUT CLEARANCE IS NOT CONSIDERED ACCEPTABLE. AN ANGLE OF GREATER THAN 5° IS REQUIRED, BUT THE ACTUAL VALUE CAN NOT BE DETERMINED AT THIS TIME. THE FLYOUT ANGLE IS A FUNCTION OF PAYLOAD MASS, C.G., C.G. OFFSET, M.O.I., AND THE SEPARATION SPRING RATE. IT MAY BE POSSIBLE TO REDESIGN THE PAYLOAD INTERFACE ADAPTER TO PROVIDE ADDITIONAL FLYOUT CLEARANCE.

APPENDIX I

THERMAL MANAGEMENT

SUN SIDE STEADY STATE INPUT FILE

```
        BCD 3THERMAL LPCS
        END
C
        BCD 3NODE DATA
C        NO#,Tinit.,CAPACITANCE
            101,250.0,410.637$              TOP
            1021,250.0,179.654$             SIDES
            1022,250.0,179.654$               "
            1023,250.0,179.654$               "
            1024,250.0,179.654$               "
            103,250.0,2566.482$             BOTTOM
            201,250.0,1887.269$             M. WHEEL/HORIZON SENSOR
            211,250.0,2310.091$             M. WHEEL ELEC.
            202,250.0,2271.740$             CPU
            203,250.0,818.908$              POWER CONDITIONER
            204,250.0,1457.069$             BATTERIES
            205,250.0,1637.452$             TX,CX
            206,250.0,162.770$              INTERNAL TORQ RODS
            207,250.0,162.770$              INTERNAL TORQ ROD
            208,250.0,162.770$              EXTERNAL TORQ ROD
            209,250.0,537.582$              MAGNETOMETERS
            1081,250.0,1221.588$            SOLAR CELLS ON TOP
            1082,250.0,543.987$             SOLAR CELLS ON SUP. STR.
            1071,250.0,354.311$             ARM
            1099,250.0,0.0$                 PAYLOAD NODE
            1031,250.0,800.0$               SUPPORT STRUCTURE
            1051,250.0,5210.008$            MIRROR
            1041,250.0,475.7221$            HINGE
            -98,290.0,0.0$                  EARTH
            -99,4.0,0.0$                    SPACE
        END
C
        BCD 3SOURCE DATA
            1081,146.0$              SUN TO TOP SOLAR CELLS(a=.65)
            1082,65.0$               SUN TO STR SOLAR CELLS(a=.65)
            103,3.75$                EARTH TO BOTTOM (a=.5)
            201,1.0$                 OUTPUT OF M WHEEL
            211,3.5$                 OUTPUT OF M WHEEL ELEC.
            209,1.0$                 OUTPUT OF MAGNETOMETER
            206,2.4$                 OUTPUT OF TORQ RODS
C            207,2.4$                     "
C            208,2.4$                     "
            205,0.0625$              AVE OUTPUT OF TX/RX FOR 40 MIN.
            204,1.4$                 OUTPUT CHARGING BATTERIES
            1099,50.0$               HEAT FROM PAYLOAD
        END
C
        BCD 3CONDUCTOR DATA
            2012,1021,1022,0.42183$       side to side
            2023,1022,1023,0.42183$            "
```

```
          2034,1023,1024,0.42183$              "
          2041,1024,1021,0.42183$              "
          1111,101,1021,0.3304$          TOP TO SIDE
          1112,101,1022,0.3304$                "
          1113,101,1023,0.3304$                "
          1114,101,1024,0.3304$                "
          1121,1021,103,0.10965$         SIDE TO BOTTOM
          1122,1022,103,0.10965$               "
          1123,1023,103,0.10965$               "
          1124,1024,103,0.10965$               "
          113,201,103,0.4349$            MW/HS TO BASE
          114,202,103,0.0001059$         CPU TO BASE
          115,203,103,1.0741$            POWER COND TO BASE
          116,204,103,4.5051$            BATTERIES TO BASE
          117,205,103,0.4735$            TX/RX TO BASE
          118,206,1022,0.08652$          INT TORQ ROD TO SIDES
          119,207,1023,0.08652$          INT TORQ ROD TO SIDES
          120,208,1023,0.08652$          EX TORQ RODS TO SIDES
          121,209,205,0.6655$            MAG TO TX/RX
          122,211,103,7.361$             M WHEEL ELEC TO BASE
          3011,1081,101,2274.345$        SOLAR CELLS TO TOP
          3012,1082,1031,1014.451$       SOLAR CELLS TO SUP. STR.
          302,1071,1099,0.0525$          ARM TO PAYLOAD
          303,1041,1051,0.2315$          HINGE TO MIRROR
          304,1041,1071,0.0525$          HINGE TO ARM
          305,1041,1031,0.3412$          HINGE TO SUP. STRUCT.
          306,1031,1024,0.7224$          SUP.STRUCT. TO SIDE
          -981,103,98,4.6822E-9$         RAD FROM BOTTOM TO EARTH (e=.5)
          -9911,1021,99,2.0485E-9$       RAD FROM SIDES TO SPACE (e=.5)
          -9912,1022,99,2.0485E-9$             "
          -9913,1023,99,2.0485E-9$             "
          -9914,1024,99,2.0485E-9$             "
          -9921,1081,99,2.3411E-9$       TOP SOLAR CELLS TO SPACE (e=.25)
          -9922,1082,99,1.0426E-9$       STR SOLAR CELLS TO SPACE (e=.25)
          -993,208,99,4.8863E-10$        RAD EX ROD TO SPACE (e=.5)
          -994,1031,99,4.17122E-9$       RAD SUP STR TO SPACE (e=.5)
          -995,1051,99,0.8779E-7$        RAD MIRROR TO SPACE (e=.5)
          -996,1071,99,2.0685E-9$        RAD ARM TO SPACE (e=.8)
      END
C
      BCD 3CONSTANTS DATA
          NDIM=1000$
          TMPZRO=0.0$               DYNAMIC STORAGE,ABS ZERO TEMP
          NLOOP=1000$               NUMBER OF LOOPS ALLOWED
          ARLXCA=0.5$               ARITHMATIC RELAXATION
          DRLXCA=0.2$               DIFFUSION RELAXATION
C          SIGMA=5.67E-8$           STEFAN BOLTZMANN CONSTANT
      END
C
      BCD 3ARRAY DATA
      END
C
      BCD 3EXECUTION
          STDSTL$
```

```
      END
C
      BCD 3VARIABLES 1
      END
      BCD 3VARIABLES 2
      END
C
C
      BCD 3OUTPUT CALLS
          TPRINT$
          GPRINT$
          QNPRNT$
      END
C
C
      BCD 3END OF DATA
```

COLD CASE TRANSIENT INPUT FILE

```
       BCD 3THERMAL LPCS
       END
C
       BCD 3NODE DATA
C       NO#,Tinit.,CAPACITANCE
           101,252.,410.637$         TOP
           1021,244.,179.654$        SIDES
           1022,245.,179.654$          "
           1023,243.,179.654$          "
           1024,246.,179.654$          "
           103,278.,2566.482$        BOTTOM
           201,280.,1887.269$        M. WHEEL/HORIZON SENSOR
           211,278.,2310.091$        M. WHEEL ELEC.
           202,278.,2271.740$        CPU
           203,278.,818.908$         POWER CONDITIONER
           204,278.,1457.069$        BATTERIES
           205,280.,1637.452$        TX,CX
           206,273.,162.770$         INTERNAL TORQ RODS
           207,243.,162.770$         INTERNAL TORQ ROD
           208,228.,162.770$         EXTERNAL TORQ ROD
           209,282.,537.582$         MAGNETOMETERS
           1081,253.,1221.588$            SOLAR CELLS ON TOP
           1082,250.,543.987$        SOLAR CELLS ON SUP. STR.
           1071,376.,354.311$        ARM
           1099,1328.,0.0$           PAYLOAD NODE
           1031,250.,800.0$          SUPPORT STRUCTURE
           1051,124.,5210.008$            MIRROR
           1041,214.,475.7221$            HINGE
           -98,290.0,0.0$            EARTH
           -99,4.0,0.0$              SPACE
       END
C
       BCD 3SOURCE DATA
           1081,5.625$               EARTH TO TOP SOLAR CELLS(a=.75)
           1082,2.505$               EARTH TO STR SOLAR CELLS(a=.75)
           201,1.0$             OUTPUT OF M WHEEL
           211,3.5$             OUTPUT OF M WHEEL ELEC.
           209,1.0$             OUTPUT OF MAGNETOMETER
C           206,2.4$                 OUTPUT OF TORQ RODS
C           207,2.4$                      "
C           208,2.4$                      "
           205,0.0625$               AVE OUTPUT OF TX/RX FOR 40 MIN.
       END
C
       BCD 3CONDUCTOR DATA
           2012,1021,1022,0.42183$ side to side
           2023,1022,1023,0.42183$      "
           2034,1023,1024,0.42183$      "
           2041,1024,1021,0.42183$      "
           1111,101,1021,0.3304$   TOP TO SIDE
           1112,101,1022,0.3304$        "
```

```
     1113,101,1023,0.3304$        "
     1114,101,1024,0.3304$        "
     1121,1021,103,0.10965$  SIDE TO BOTTOM
     1122,1022,103,0.10965$    "
     1123,1023,103,0.10965$    "
     1124,1024,103,0.10965$    "
     113,201,103,0.4349$          MW/HS TO BASE
     114,202,103,0.0001059$  CPU TO BASE
     115,203,103,1.0741$          POWER COND TO BASE
     116,204,103,4.5051$          BATTERIES TO BASE
     117,205,103,0.4735$          TX/RX TO BASE
     118,206,1022,0.08652$   INT TORQ ROD TO SIDES
     119,207,1023,0.08652$   INT TORQ ROD TO SIDES
     120,208,1023,0.08652$   EX TORQ RODS TO SIDES
     121,209,205,0.6655$          MAG TO TX/RX
     122,211,103,7.361$      M WHEEL ELEC TO BASE
     3011,1081,101,2274.345$ SOLAR CELLS TO TOP
     3012,1082,1031,1014.451$     SOLAR CELLS TO SUP. STR.
     302,1071,1099,0.0525$   ARM TO PAYLOAD
     303,1041,1051,0.2315$   HINGE TO MIRROR
     304,1041,1071,0.0525$   HINGE TO ARM
     305,1041,1031,0.3412$   HINGE TO SUP. STRUCT.
     306,1031,1024,0.7224$   SUP.STRUCT. TO SIDE
     -981,103,99,4.6822E-9$  RAD FROM BOTTOM TO SPACE (e=.5)
     -9911,1021,99,2.0485E-9$     RAD FROM SIDES TO SPACE (e=.5)
     -9912,1022,99,2.0485E-9$          "
     -9913,1023,99,2.0485E-9$          "
     -9914,1024,99,2.0485E-9$          "
     -9921,1081,98,2.3411E-9$     TOP SOLAR CELLS TO EARTH (e=.25)
     -9922,1082,98,1.0426E-9$     STR SOLAR CELLS TO EARTH (e=.25)
     -993,208,99,4.8863E-10$ RAD EX ROD TO SPACE (e=.5)
     -994,1031,99,4.17122E-9$     RAD SUP STR TO SPACE (e=.5)
     -995,1051,99,0.8779E-7$ RAD MIRROR TO SPACE (e=.5)
     -996,1071,99,2.0685E-9$ RAD ARM TO SPACE (e=.8)
 END
C
 BCD 3CONSTANTS DATA
     NDIM=1000$
     TMPZRO=0.0$             DYNAMIC STORAGE,ABS ZERO TEMP
     NLOOP=1000$             NUMBER OF LOOPS ALLOWED
     CSGFAC=2.0$
     OUTPUT=300.0$
     ARLXCA=0.5$             ARITHMATIC RELAXATION
     DRLXCA=0.2$             DIFFUSION RELAXATION
     TIMEND=1800.0$
 END
C
 BCD 3ARRAY DATA
 END
C
 BCD 3EXECUTION
     SNDUFR$
 END
C
```

```
      BCD 3VARIABLES 1
      END
      BCD 3VARIABLES 2
      END
C
C
      BCD 3OUTPUT CALLS
          TPRINT$
          GPRINT$
          QNPRNT$
      END
C
C
      BCD 3END OF DATA
```

```
LUMPED MASS MODEL OF THE HEAT PIPE PAYLOAD

      INTEGER N
      CHARACTER*20 INFILE
      PRINT*,'ENTER A FILE NAME (LPT1 FOR PRINTOUT)    '
         READ(*,'(A)') INFILE
         OPEN(10,FILE=INFILE,STATUS='UNKNOWN')
C
C******************  SET INITIAL CONDITIONS ****************
      T=1957.0
      AREAH=7.07E-4
      AREAC=2.87E-2
      AREAR=7.339E-3
      SIGMA=5.67E-8
      EM=0.22
      QS=2048.0
      TM=236.8
      EBH=QS/AREAH
C
C***************** OUT OF SUN AT 1957K ***********
C           WRITE(10,*)'OUT OF SUN WITH INITIAL TEMP OF 1957K'
      DO 2 N=1,1801
           T=T+(-QR-QHC)/TM
           QR=EM*SIGMA*AREAR*T**4
           QHC=(SIGMA*T**4)/((1.0-EM)/(AREAC*EM)+1.0/AREAH)
C     IF(I.EQ.60)THEN
C          WRITE(10,*)'TIME=',TIME,'TEMP=',T,' Qh=',QHC,'Qr=',QR
           I=0
C     ENDIF
           WRITE(10,*)TIME,T
           TIME=TIME+1.0/60.0
           I=I+1
2     CONTINUE

C**************** BACK INTO  THE SUN ****************
C          WRITE(10,*)'INTO SUN..COLDEST TEMP=',T,'K'
      DO 4 N=1802,2700
           T=T+(QHC-QR)/TM
           QHC=(EBH-SIGMA*T**4)/((1.0-EM)/(AREAC*EM)+1.0/AREAH)
           QR=EM*SIGMA*AREAR*T**4
C     IF(I.EQ.60)THEN
C          WRITE(10,*)'TIME=',TIME,'TEMP=',T,'QIN=',QHC,'Qr=',QR
           I=0
C     ENDIF
           WRITE(10,*)TIME,T
           TIME=TIME+1.0/60.0
           I=I+1
4     CONTINUE

             END
```

APPENDIX J

TEST AND EVALUATION

TESTING & EVALUATION
SUBSYSTEM TESTS

- ***PAYLOAD***
  - SATELLITE SEPARATION TEST
  - DETERMINE DEPLOYED MASS, CG & INERTIA

- ***SOLAR COLLECTION***
  - VALIDATE HEAT OUTPUT PARAMETER FROM COLLECTOR
  - VALIDATE HEAT OUTPUT FROM RADIATOR
  - PROPER DEPLOYMENT OF SOLAR COLLECTOR AND HEAT PIPE

- ***THERMAL***
  - POSSIBLE VALIDATION OF SOME SUBSYSTEMS SEPARATE FROM OVERALL SYSTEM THERMAL TEST

- ***STRUCTURE***
  - STRUCTURAL TEST OF BUS, SOLAR COLLECTOR & HEAT PIPE
  - CAD FIT CHECK OF THERMION TO DELTA II

- ***POWER***
  - FUNCTIONAL TEST OF TOTAL POWER BUS
  - POWER ON SEQUENCE THAT FOLLOWS BOOSTER SEPARATION

- ***ATTITUDE CONTROL***
  - FUNCTIONAL & ALIGNMENT TEST OF ALL SENSORS AND TORQUE RODS
  - VALIDATE POINTING & ATTITUDE CONTRO ALGORITHMS
  - MAGNETIC INTERACTION/VALIDATION

- ***COMMUNICATIONS***
  - ANTENNA ALIGNMENT
  - VERIFY DOWN LINK AND UP LINK COMPATIBILITY
  - VERIFY SOFTWARE ALGORITHMS
  - VERIFY NON-INTERACTION BETWEEN TORQUE RODS & TRANSCEIVER

- ***DATA MANAGEMENT***
  - VERIFY DATA ACQUISITION AND DATA STORAGE ALGORITHMS
  - VERIFY SYSTEM HOUSE KEEPING FUNCTIONS
  - VERIFY PROPER INTERACTION BETWEEN SUBSYSTEMS
  - VERIFY OUTPUT VOLTAGE/CURRENT MEASUREMENT TECHNIQUE

APPENDIX K

THERMION-II

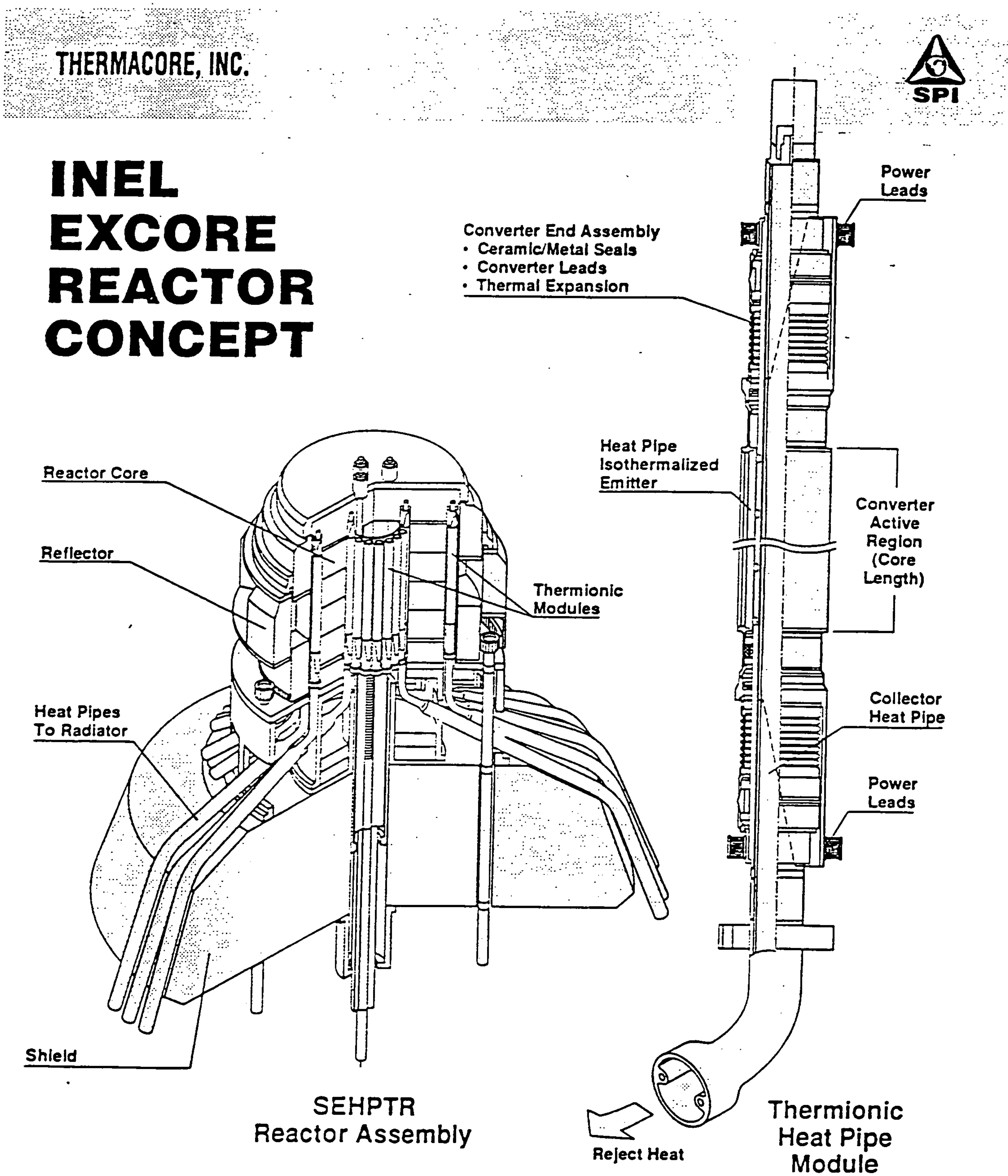
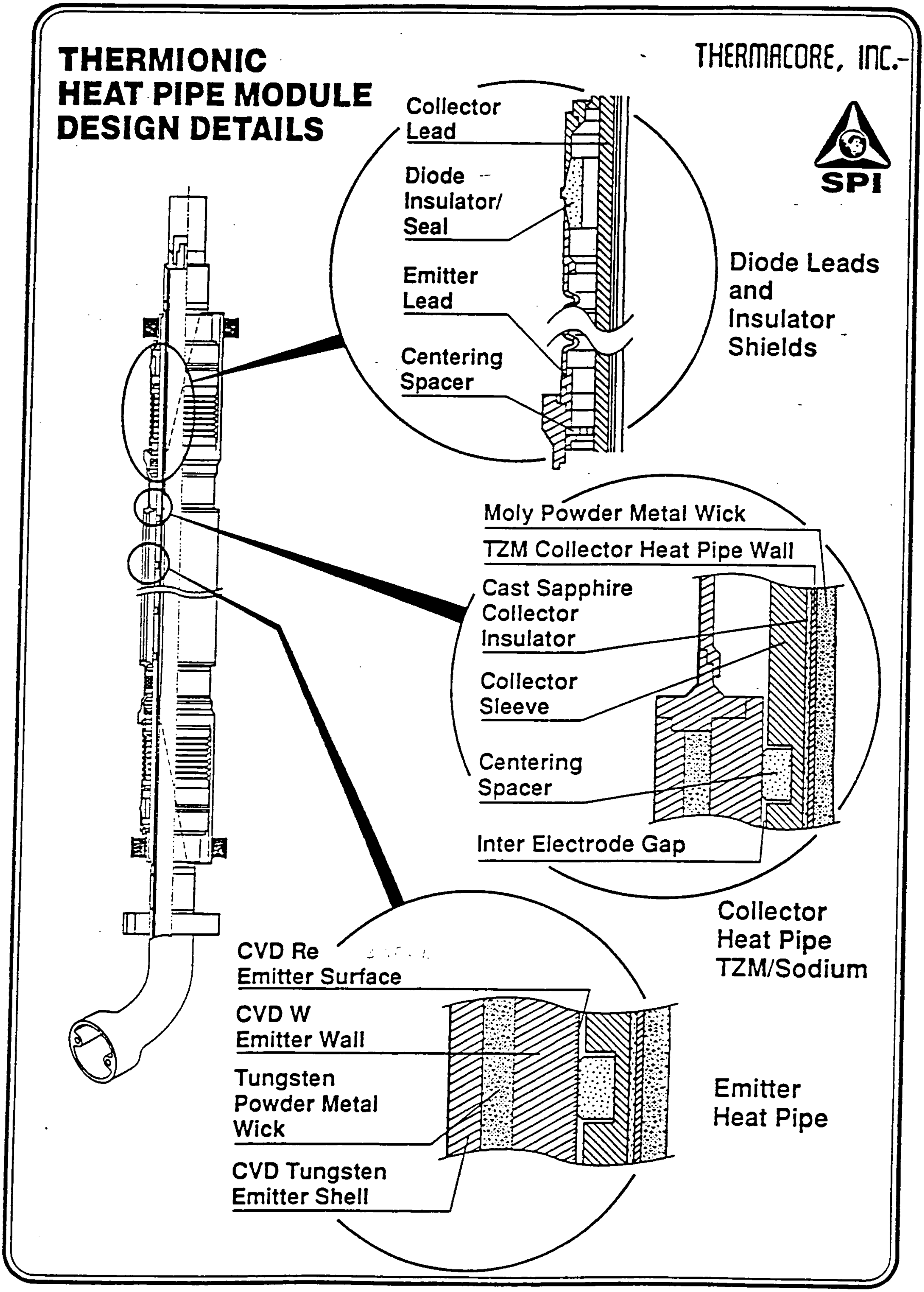


Figure 1. INEL SEHPTR concept



021491-1

Figure 2. Thermionic heat pipe module design details

EXECUTIVE SUMMARY

The Idaho National Engineering Laboratory (INEL) has developed a preliminary conceptual design of a Small Ex-Core Heat Pipe Thermionic Reactor (SEHPTR). The design concept uses integrated Thermionic Heat Pipe Modules to produce 40 kilowatts of electric power. This report documents work done by Thermacore, Inc. of Lancaster, Pennsylvania, and its subcontractors to analyze the conceptual design of a thermionic heat pipe module for application to the SEHPTR power system.

Under INEL Subcontract Number C91-103269-DAJ-1791 a thermionic heat pipe module, conceived by Thermacore, Inc., was designed for use with the SEHPTR power system. A conceptual design was established and analyzed for thermal, mechanical, and electrical performance. Detailed thermionic performance modeling was done by Space Power Inc. of San Jose, California, under subcontract to Thermacore.

Thermionic heat pipe module system modeling and performance trade studies resulted in a baseline conceptual system design and several alternative configurations. The baseline design uses core-length, single cell thermionic elements for electric power generation. The design provides more than 42 kWe (BOL) at 10.2% system efficiency with a maximum SEHPTR fuel temperature of 2619 K.

1.0 INTRODUCTION

Thermacore has developed a conceptual design for a Thermionic Heat Pipe Module (THPM) for use in space nuclear power systems. The THPM has been developed for use with The Idaho National Engineering Laboratory's Small Ex-Core Heat Pipe Thermionic Reactor (SEHPTR) power system. The baseline design produces greater than 40 kilowatts of electric power and offers significant advantages for potential military customers, including simplicity, reliability, and testability. The SEHPTR concept using THPM's may also be scalable over a wide range of power levels.

This report documents work done by Thermacore and its subcontractors under a Phase I Program to develop and analyze a conceptual design for the Thermionic Heat Pipe Module. Objectives of the program were as follows:

1. To identify a conceptual design of an integrated Thermionic Heat Pipe Module which is compatible with the SEHPTR power system.
2. To identify a preliminary design of a proof-of-performance test article and test program that will validate key feasibility issues associated with the Thermionic Heat Pipe Module.

Conceptual design work of the THPM has been completed by Thermacore. A baseline concept was identified and analyzed for thermal and mechanical performance. Detailed thermionic performance modeling was performed by Space Power, Inc. of San Jose California under subcontract to Thermacore. The baseline design uses core length, single cell, cylindrical thermionic elements to meet performance requirements of the SEHPTR power systems.

Section 2 of this report highlights the conclusions and recommendations resulting from this conceptual design study. Section 3 describes the conceptual design of the THPM, including overall performance of the system and compatibility of the design with the SEHPTR concept. Performance analysis of the THPM concept including thermionic and systems design work provided by Space Power Inc., is documented in Section 4. A description of key feasibility issues and the conceptual design of a proof-of-principle test program and hardware is found in Section 5.

APPENDIX L

POWER SYSTEM

POWER BUDGET DIAGRAM

SUBSYSTEM	QTY	OPERATING VOLTAGE, V	OPERATING CURRENT, A	POWER REQUIRED, W	DUTY CYCLE, % OF ORBIT PERIOD	ORBIT AVERAGE POWER, W	% OF TOTAL POWER
COMPUTER							
CPU	1	12	0.0225	0.27	100	0.27	0
DATA ACUISITION	1	12	0.0625	0.75	33	0.2475	0
RECEIVER*	1	28	1	25	1	0.25	0
TRANSMITTER*	1	28	1	25	1	0.25	0
HEAT PIPE HEATER	1	60	29	1750	100	1750	100

TOTAL INSTANT POWER= **1801.020**

*USING EXISTING DELTA-II

TOTAL ORBIT AVERAGE POWER CONSUMTION= **1751.018**

APPENDIX M

LAUNCH VEHICLE INTERFACE AND PAYLOAD

Castable series	Kaolite 2500-LI (cast)	Kaolite 2500-LI (gunned)	Kaolite 2500-HS (cast)	Kaolite 2500-HS (gunned)	Kaolite 2600-LI[1]	Kaolite 2800[1]	Kaolite 3000-C	Kaolite 3000-G[1]	Kaolite 3300
Recommended methods of application (X)									
Cast or rammed	X	X	X	X	X	X	X	–	X
Troweled	X	X	X	X	X	X	X	–	X
Gunned	X	X	X	X	X	X	–	X	–
Lbs. required to place one cu. ft.	74	70	78	85	81	84	68	90	71
Required water, U.S. qt. per bag[2]									
Casting by vibrating	0.5-12.0	–	11.5-12.5	–	13.0-15.0	9.5-10.0	6.5-7.2	–	4.0-4.5
Casting by hand-rodding	0.5-12.0	–	11.5-12.5	–	13.0-15.0	9.5-10.0	6.5-7.2	–	4.0-4.5
Pouring	–	–	–	–	–	–	–	–	–
Recommended use limit, °F	2500	2500	2500	2500	2600	2800	3000	3000	3300
Melting point, °F	2740	2740	2700	2700	2800	3300	3170	3140	3530
Density, lbs./cu. ft.									
Fired at use limit temperature	67	65	82	84	84	85	74	95	73
Modulus of rupture, psi (ASTM C 133-84)									
Dried 18-24 hrs. @ 220F	150-250	150-250	220-300	250-400	300-600	300-500	180-400	400-600	450-650
Fired 5 hrs. @ 1000F	–	–	–	–	–	–	–	–	–
@ 1500F	150-300	150-300	130-200	200-350	200-400	200-300	150-350	200-500	–
@ 1600F	–	–	–	–	–	–	–	–	–
@ 1800F	–	–	–	–	–	–	–	–	–
@ 2000F	–	–	–	–	–	–	–	–	200-300
@ 2200F	–	–	–	–	–	–	–	–	–
@ 2300F	–	–	–	–	–	–	–	–	–
@ 2500F	350-500	250-400	400-600	600-850	–	–	–	–	–
@ 2600F	–	–	–	–	750-950	–	–	–	–
@ 2800F	–	–	–	–	–	1300-1500	–	–	–
@ 3000F	–	–	–	–	–	–	900-1300	700-1000	–
@ 3300F	–	–	–	–	–	–	–	–	700-950
Cold crushing strength, psi (ASTM C 133-84)									
Dried 18-24 hrs. @ 220F	750-800	500-800	700-1000	800-1300	800-1200	950-1200	650-800	750-1000	1000-1400
Fired 5 hrs. @ 1000F	–	–	–	–	–	–	–	–	–
@ 1500F	550-800	500-650	450-700	700-1100	800-1000	600-900	500-750	1200-1600	–
@ 1600F	–	–	–	–	–	–	–	–	–
@ 1800F	–	–	–	–	–	–	–	–	–
@ 2000F	–	–	–	–	–	–	–	–	700-1000
@ 2200F	–	–	–	–	–	–	–	–	–
@ 2300F	–	–	–	–	–	–	–	–	–
@ 2500F	700-1300	700-1000	900-1200	900-1300	–	–	–	–	–
@ 2600F	–	–	–	–	1800-2000	–	–	–	–
@ 2800F	–	–	–	–	–	3500-4200	–	–	–
@ 3000F	–	–	–	–	–	–	1700-2200	2500-3400	–
@ 3300F	–	–	–	–	–	–	–	–	1700-2000
Permanent linear change, % (ASTM C 113-74)									
Dried 18-24 hrs. @ 220F	–	0.0	0.0	0.0	0.0	0.0	–	0.0	0.0
Fired 5 hrs. @ 1000F	–	–	–	–	–	–	–	–	–
@ 1500F	−0.2	−0.1	−0.2	−0.3	−0.3	−0.3	−0.3	−0.2	–
@ Recommended use limit, °F	−0.1	+1.0	−2.0	−0.8	+1.0	−0.5	−1.5	−1.5	−0.3
Chemical analysis, % (fired basis) (ASTM C 573-81)									
Alumina Al_2O_3	44.0	44.0	43.7	43.7	46.0	60.0	60.0	60.0	94.0
Silica SiO_2	35.0	35.0	34.6	34.6	36.0	33.0	34.0	34.0	0.5
Ferric oxide Fe_2O_3	0.9	0.9	2.6	2.6	1.4	0.4	1.0	1.0	0.2
Titanium oxide TiO_2	1.8	1.8	1.0	1.0	1.7	0.7	0.7	0.7	–
Calcium oxide CaO	17.0	17.0	17.3	17.3	14.0	5.0	3.6	3.2	4.6
Magnesium oxide MgO	0.2	0.2	0.1	0.1	0.2	0.1	0.2	0.2	0.1
Alkalies, as Na_2O	1.3	1.3	0.7	0.7	0.7	0.8	0.7	0.9	0.5
Thermal conductivity, Btu • in./hr. • ft.² • °F (ASTM C 417-86)									
Mean Temperature @ 500F	1.78	1.75	2.2	2.3	2.4	3.1	3.2	3.5	6.0
@ 1000F	1.92	1.88	2.3	2.4	2.5	3.2	3.2	3.6	5.0
@ 1500F	2.08	2.05	2.5	2.6	2.7	3.3	3.5	3.8	4.8
@ 2000F	–	–	–	–	–	3.4	3.8	4.2	4.9
Pounds per bag	50	50	75	75	75	75	75	75	50

Data are average results of tests conducted under standard procedures and are subject to variation. Results should not be used for specification purposes.

LI-2200
Lockheed Missiles & Space Co.
silica

Density 22.0 lb_m/ft^3
Maximum Temperature 3160 °R

T (R) — k perpendicular (through the thickness) (Btu/ft hr R)

T (R)	Pressure (Atm)					
	0	0.0001	0.001	0.01	0.1	1.0
210	0.014	0.014	0.016	0.026	0.030	0.032
460	0.017	0.017	0.022	0.031	0.038	0.043
710	0.022	0.022	0.026	0.036	0.047	0.054
960	0.024	0.024	0.032	0.040	0.056	0.065
1210	0.029	0.029	0.037	0.045	0.064	0.076
1460	0.034	0.034	0.043	0.052	0.075	0.088
1710	0.041	0.041	0.050	0.060	0.092	0.104
1960	0.050	0.050	0.059	0.071	0.106	0.119
2210	0.060	0.060	0.070	0.084	0.123	0.139
2460	0.070	0.070	0.083	0.100	0.142	0.160
2760	0.085	0.085	0.100	0.120	0.167	0.189
2960	0.096	0.096	0.113	0.135	0.186	0.212
3260	0.116	0.116	0.135	0.164	0.219	0.250
3460	0.131	0.131	0.153	0.189	0.245	0.280

T (R) — k parallel (in plane) (Btu/ft hr R)

T (R)	Pressure (Atm)					
	0	0.0001	0.001	0.01	0.1	1.0
210	0.028	0.028	0.033	0.038	0.050	0.055
460	0.030	0.030	0.035	0.040	0.054	0.060
710	0.032	0.032	0.037	0.042	0.058	0.065
960	0.040	0.040	0.044	0.052	0.070	0.080
1210	0.045	0.045	0.051	0.058	0.078	0.090
1460	0.052	0.052	0.058	0.067	0.090	0.104
1710	0.058	0.058	0.067	0.077	0.104	0.120
1960	0.067	0.067	0.075	0.088	0.118	0.136
2210	0.075	0.075	0.088	0.098	0.132	0.150
2460	0.086	0.086	0.098	0.113	0.153	0.177
2760	0.100	0.100	0.116	0.133	0.180	0.208
2960	0.112	0.112	0.128	0.147	0.197	0.221
3260	0.130	0.130	0.150	0.173	0.235	0.270
3460	0.146	0.146	0.167	0.193	0.258	0.300

Ref. 2 Aeroassist Flight Experiment Aerobrake Thermal Design Data Book, JSC-23571, May, 1989.
Ref. 18. Space Shuttle Program Thermodynamic Design Data Book., SD73-SH-0226,, Rockwell Int. January 1981.

LI-2200 *(continued)*

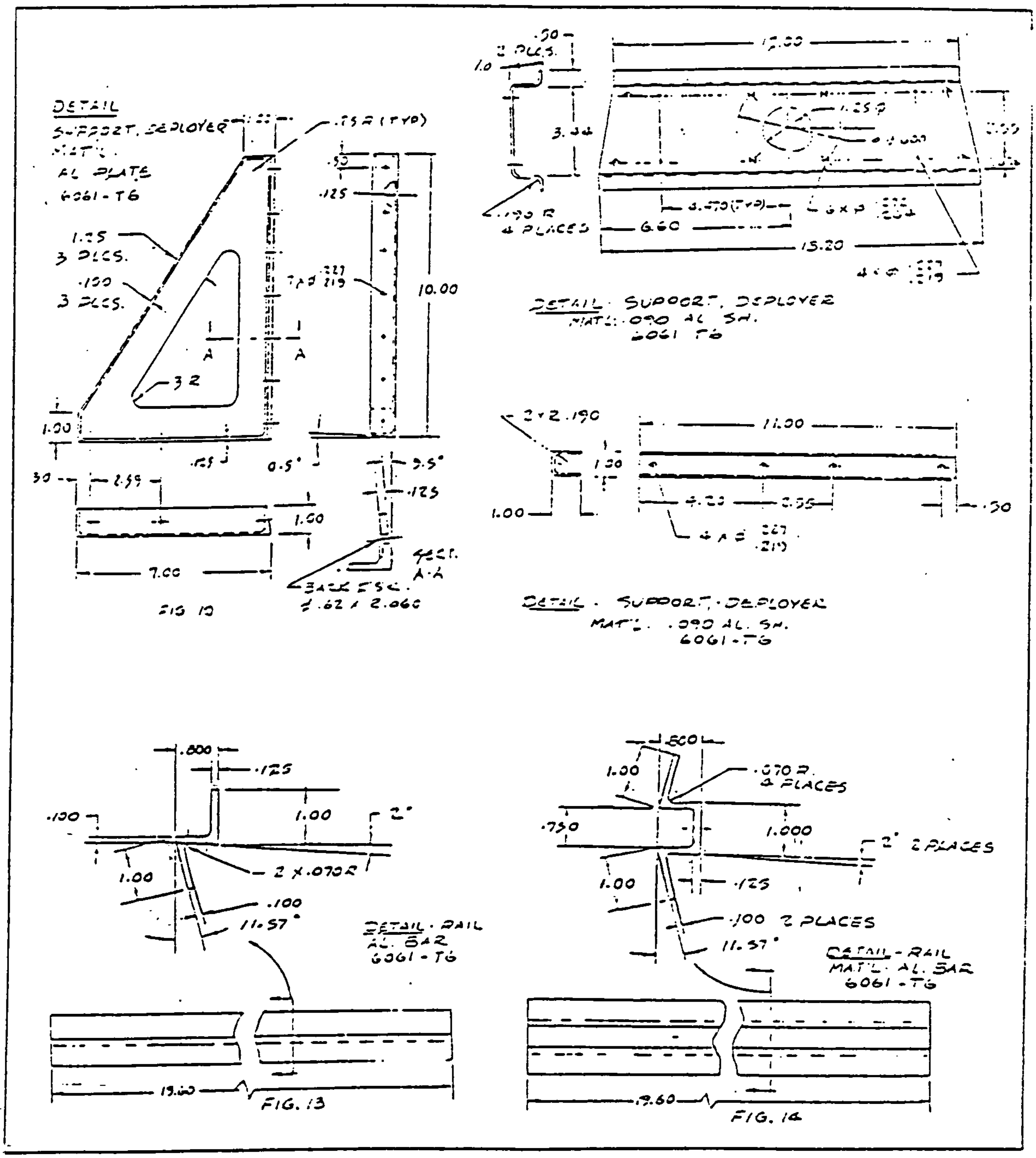
T (R)	C_p (Btu/lb_m R)
210	0.070
310	0.105
460	0.150
710	0.210
960	0.252
1210	0.275
1460	0.288
1710	0.296
1960	0.300
2160	0.302
2210	0.303
2460	0.303
3460	0.303

SEDS/DELTA II MECHANICAL INTERFACES

Attachment Description

Description to be furnished by MDSSC.

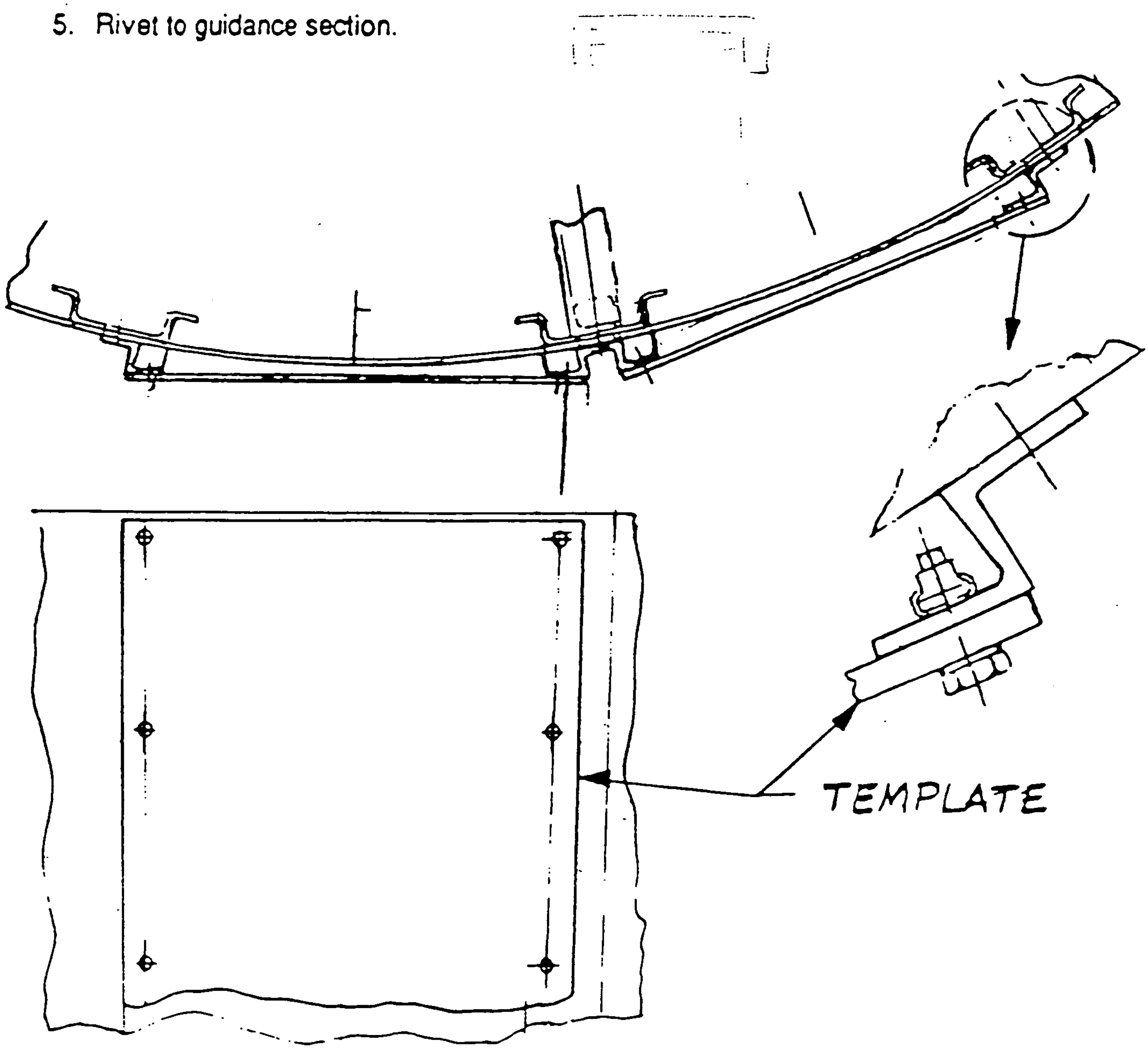
FIGURE: SEDS ATTACHMENT INTERFACE



ATTACHMENT (b)

Method of installing nut plates on Z, and top hat sections prior to installing the sections to vehicle guidance assy.

1. Prepare hole pattern template to nom. dims.
2. Drill holes in Z and top hat sections using template.
3. Install nut plates.
4. Mount to template.
5. Rivet to guidance section.



6. Matching templates to be used for drilling holes in all components to be mounted to guidance assy.

APPENDIX N

COSTS
THERMION-I & THERMION-II

THERMION I

SYSTEM	Task	Time (weeks)	Material Costs	Subsystem Costs
	Mission Planning	6.0		
	Budgeting	2.0		
	Interface Subsystems	36.0		
	Support Help	26.0		
				$98,000
PAYLOAD				
	Mirror Design	6.0		
	Manufacture	8.0		
	Materials		$2,000	
	Cavity Design	8.0		
	Manufacutre	4.0		
	Materials		$3,000	
	Heat Pipe Integration	4.0		
	Payload Assemble	2.0		
	Testing	3.0		
				$54,000
ATTITUDE CONTROL				
	Control Algrorithm	4.0		
	Software Algrorithm	3.0		
	Sensor Selection	4.0		
	Hardware			
	Coarse Sun Sensor		$35,000	
	Fine Sun Sensor		$30,000	
	Horizon/Momentum		$225,000	
	Photo Diodes		$200	
	Torque Rods		$75,950	
	Magnetometer		$9,130	
	Norad Efemeres Data		$1,000	
	Hardware Integration	5.0		
	Testing	4.0		
				$404,280
STRUCTURE				
	Structure Design	8.0		
	Manufacture			
	Base Plate		$1,000	
	Sides		$1,000	
	Top		$500	
	Arm		$1,000	
	Payload Arm		$1,000	
	Marman Band		$10,000	

	Bolt Cutters		$6,000	
	Separation Switch		$10	
	Brackets		$1,000	
	Trorque Rod Deployment		$2,000	
	Hinge		$3,000	
Hardware Integration				
	Internal	3.5		
	External	2.0		
Testing		4.0		
				$51,010

DATA MANAGEMENT

Computer Selection		1.0		
Hardware				
	CPU Card		$510	
	Storage Memory Card		$211	
	Acqusition Card		$695	
	Prototying Card		$69	
	Card Cage		$100	
	Cables & Connectors		$100	
Software Development		10.0		
Soft/Hard Integration		2.0		
Testing		3.0		
				$24,085

COMMUNICATIONS & GROUND STATION

Antenna Selection		1.5		
Ground Station Selection		1.0		
Hardware				
	Sending Antenna		$25,000	
	Receiving Antenna		$25,000	
	Ground Station		$50,000	
Software Development		3.0		
Hard/Soft Integration		2.0		
Testing		3.0		
				$114,700

POWER MANAGEMENT

Power System Design		2.0		
Hardware				
	36 Ni-Cad Batteries		$400	
	Recharging Control		$5,000	
	Solar Cells		$5,000	
	Misc.		$500	

Integration	2.0		
Testing	3.0		
			$20,700

THERMAL MANAGEMENT

Thermal Design	8.0		
Thermal Testing	2.0		
			$14,000

Launch Vehical Interface

Our Integration	8.0		
Delta II Integration		$1,000,000	
			$1,011,200
		Subtotal:	$1,791,975

** Uncertainty **
Times Subtotal 1.5

Grand Total	$2,187,963

THERMION -II COST BREAK DOWN ASSUMING 1 year from kick-off to delivery

OVERHEAD RATE 100 %

FUDGE FACTOR 1.5

LABOR

			Task Time (weeks	Cost, $
PROGRAM MANAGEMENT	Rate,$/Hr=	30		
Negotiate Contract			1	1,200
Mission Planning				0
Negotiate Launch time, orbit and Fee			2	2,400
Determine McDonnel-Douglas responsibilities			3	3,600
Determine Thermion II responsibilities			3	3,600
Budgeting			3	3,600
Interface With Airforce			3	3,600
PDR			1	1,200
CDR			2	2,400
ADMINISTRATIVE ASSISTANT	Rate,$/Hr=	12	52	24,960
MECHANICAL ENGINEERING DESIGN	Rate,$/Hr=	17		
Design of battery pallets				
Super Structure			3	2,040
Battery Boxes			2	1,360
Interface to launch vehicle			2	1,360
Interface to electrical System			4	2,720
Design of Payload				
Structural Support			2	1,360
Insulation Selection & Integration			5	3,400
Interface to THERMACORE design			5	3,400
Integration to launch vehicle				
Structural Loading determination			1	680
Stress Analysis			0.5	340
Thermal Analysis			0.5	340
Satisfy safety concerns				
Design Shipping Containers			3	2,040
			1	680
ELECTRICAL ENGINEERING DESIGN	Rate,$/Hr=	17		
Design of Power System			8	5,440
Design of Communication Interface			2	1,360
Payload Integration				
Thermocouple design			2	1,360
Design of computer			10	6,800
Satisfy safety concerns			2	1,360
Subsytem Bench tests				
Computer			4	2,720
Power Regulator			4	2,720
Comunication Interface			2	1,360

TECHNICAL SUPPORT	Rate,$/Hr=	14		
Produce Fabrication Drawings				
	Mechanical		14	7,840
	Electrical		14	7,840
	Assembly		2	1,120
Mechanical Assembly			6	3,360
Electrical Assembly			2	1,120
TESTING AND VERIFICATION	Rate,$/Hr=	17		
Mechanical				
	Tungsten Heater Calibration		1	680
	Static Load		2	1,360
	Random Vibration		2	1,360
	Thermal/Vacuum		2	1,360
Electrical				
	Electrical System Test			
	Bench		4	2,720
	Random Vibration		2	1,360
	Thermal/Vacuum		2	1,360
	Thermocouple Calibration		1	680
Mechanical/Electrical	Fit check to Launch Vehicle		1	
LAUNCH SUPPORT	Rate,$/Hr=	30		
Unpack			0.3	360
Assemble			0.5	600
Subsystem Test			1	1,200
Full system Test			0.1	120
Hand Over To McDonnell- Douglas			-	
			TOTAL=	123,840

MATERIAL COSTS	QTY	Cost/Unit	Total Cost, $
Fabrication of Payload Support Module	1		10,000
Fabrication of battery boxes	16	200	3,200
Fabrication of battery pallets	4	1000	4,000
Fabrication of power regulator	1	5000	5,000
Fabrication of Computer	1	5000	10,000
Alkaline Cells	4000	1.5	6,000
Thermal Vac & Random Vib. Test	1		15,000
		Total	53,200

TRAVEL/SHIPPING		40,000
	TOTAL OF LABOR AND MATERIALS=	217,040
	OVER HEAD=	217,040
	SUBTOTAL=	434,080
	FUDGE FACTOR * SUBTOTAL=	651,120
LAUNCH COST		1,000,000
	GRAND TOTAL=	1,651,120