Designing Solar Storm Microsatellite for Predicting Space Weather[1]

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Abstract – Solar flares can cause serious damage to satellites and disruptions to power grids. To provide warning of impending solar storms to satellite operators, power companies and other users, a sun- and earth-viewing microsatellite located in orbit about the L1 libration point was designed by the Aerospace Corporation's Concept Design Center (CDC). The CDC's collaborative design methodology provided a comprehensive investigation of the trade space to support the accelerated procurement timeline.

Not all solar storms inflict damage. The polarity of the solar flare, available only minutes before the storm approaches the earth, determines the potential for damage. The integrated dual payload concept of hemispherical imager and magnetometer provides up to three days warning of an impending solar storm with 30-60 minutes warning of confirmation that the solar storm has the potential to inflict damage to both ground- and space-based systems.

Eight satellite configurations were developed to study the impact of varying key trade space parameters: spacecraft lifetime; data compression; direct injection into the L1 transfer orbit versus transferring from Geosynchronous Transfer Orbit (GTO); varying the number of hemispherical imagers; effects of technology freeze date and redundancy; and smallsat technology and components.

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1. INTRODUCTION

Satellite operators and terrestrial power companies have suffered losses in the past due to damage caused by geomagnetic storms. Several satellites, such as Telstar 401 [1], are believed to have been lost due to solar storm damage. Not all solar storms will interact with the Earth's magnetic field but those with appropriate polarity and magnitude to do so can induce geomagnetic currents that may damage power networks. The province of Quebec suffered economic losses, which have been estimated at over one billion dollars, after the solar flare in 1989 caused extensive damage to the power grid and left six million people without power for nine hours. Solar storm data is also of great value to space weather observers. The National Oceanic and Atmospheric Administration (NOAA) Space Environment Center (SEC) has expressed a need for smaller, lighter, less power hungry and less costly systems for operational use. This need was one of the drivers for designing a this solar flare predicting microsatellite system.

The existing system, the NASA Advanced Composition Explorer (ACE) satellite, is currently located at the L1 location. With a magnetometer as a payload, it currently provides only 30 – 60 minutes warning of geomagnetic storms. It was launched in 1997 and has a two year mission design life [2]. The author is unaware of plans to replace this satellite when it fails. Since 1999-2003 is the peak of the next solar cycle, a solar storm warning system would provide maximum value during this period, but ACE is unlikely to be available. Furthermore, its short warning period provides limited usefulness to potentially affected parties.

The Aerospace Corporation is a California non-profit corporation that operates a Federally Funded Research and Development Center (FFRDC) and has over thirty years of experience in design and development of space systems in support of the United States military space program. A Corporate Strategic Initiative to develop microsatellite technology was seeking a candidate mission to demonstrate implementation. Through knowledge of the Solar Mass Ejection Imager (SMEI) designed by University of California San Diego's (UCSD) Center for Astrophysics and Space Sciences [3] and innovation of the proposed concept of the dual payload at L1, the Solar Storm Microsatellite was born. This concept combines the best features of the two payloads to give up to three days warning of a potential geomagnetic storm, which allows preparations to be made to mitigate damage, followed by confirmation 30-60 minutes before an event so that the contingency plans can be implemented only if needed. The spacecraft and integrated imager concept is an ongoing Aerospace initiative currently under development.

The Concept Design Center at Aerospace

Aerospace's Concept Design Center provides an interactive, real-time conceptual design environment that allows customers to work more closely with engineering experts from the corporation's Engineering and Technology Group to provide conceptual design for space systems two to twenty years in the future. [4].

[1] 0-7803-5846-5/00/$10.00 © 2000 IEEE

The CDC consists of a team of space system engineering experts working with a set of collaborative processes and tools that facilitate rapid design iteration. The CDC facility allows the experts to interact with the customer and each other and promotes the exchange of information through face-to-face contact. The result is a real time interactive design environment in which experts actively discuss assumptions, problems and implications as each works to understand the requirements for their subsystem and their impact on other subsystems and the system as a whole. Desktop personal computers and spreadsheet software are utilized to manage the sharing of information between experts and to help ensure consistency across the entire system design. A CAD system is used to generate spacecraft configurations.

To facilitate interaction, contributions from the design experts are coordinated by a systems engineer. The systems engineer also captures the system's internal and external interfaces. This process enables each of the design experts to prepare their contributions simultaneously and in the presence of the other team members and the customer.

The CDC provides several fundamental advantages over more traditional conceptual design methodologies. Most notable is the short turnaround time. The Solar Storm Microsatellite conceptual design was completed in about six weeks. The consistency achieved by the real-time collaborative approach and information sharing system exceeds that of traditional methods. The CDC provides the ability to rapidly change the direction of the design study. In the case of the Solar Storm Microsatellite, the initial configurations required a Taurus launch vehicle. One week later with some additional research, the team was able to come together again and redesign several configurations that could be launched on a Pegasus launch vehicle with comfortable margins.

2. Mission Overview

The solar storm mission has been proposed as a means of providing a warning of solar storms to organizations that might be adversely affected by them, such as power companies and satellite operators, as well as to those involved in space weather forecasting. The mission consists of two primary payloads developed by University of California San Diego's (UCSD) Center for Astrophysics and Space Sciences: one or more imagers and a magnetometer. The hemispherical imager [5] views approximately one half of the sky, from 4° from the edge of the sun to 1° from the edge of the earth. It will detect coronal mass ejections (CMEs) and provide about three days warning that magnetic storms may be headed towards earth. However, most CMEs will either not interact with the Earth's magnetic field or will be of insufficient strength to result in a geomagnetic storm. Furthermore, the imager cannot distinguish between CMEs that are headed towards earth and those that are headed away. Thus, an imager alone would produce a high rate of false alarms. Therefore, a magnetometer is also used on the spacecraft, to measure the magnitude and polarity of the magnetic field as it passes the spacecraft. If the CME is determined to have the potential to interact with the Earth's magnetic field, a message is immediately relayed to earth, providing 30 – 60 minutes warning of the impending geomagnetic storm. The magnetometer alone would not provide sufficient warning time to be effective. However, the two instruments together provide up to three days advance warning of a potential problem, which allows for preparations to be made and communications to be established. Appropriate steps can then be taken rapidly if a severe event is confirmed.

In order to achieve adequate reliability of a solar mass ejection prediction, near-full 4π steradian coverage of the sky is required. However, the hemispherical imager baselined for the study would provide only half this coverage. Two means of achieving full coverage were considered. One option considered was to use two hemispherical imagers, placed on opposite sides of the spacecraft. Another option was to use one hemispherical imager and then rotate the spacecraft 180° for successive images. An image would be taken once per hour; in the case of the configuration with two imagers, this would be from alternating imagers. Both of these options were included in the study. Ten exposures lasting two minutes each would be taken consecutively over a twenty-minute period. [After the study was completed, it was determined that twenty exposures over a forty-minute period would be required. The impacts of this have not been investigated and are not included herein.] These images would then be averaged on board to eliminate noise, and the resulting imagery would be downlinked to the ground. Configurations were studied which downlink data that is compressed on board the spacecraft, as well as uncompressed data. Estimates of the number, duration and size of exposures, and of the amount of data compression that could be anticipated, were provided by Dr. Andrew Buffington of the Center for Astrophysics and Space Sciences at the University of California, San Diego.

Imager Design

The imager observes a hemisphere of space, ranging from just above the Sun's limb to just above the Earth's limb. By keeping these relatively bright objects out of the imager's field of view, the coronal mass ejections can be readily observed against the dark background. The hemispherical imager is designed by University of California San Diego's (UCSD) Center for Astrophysics and Space Sciences [5], [6]. The imager incorporates a corral baffle system to keep out stray light, a crystal ball and a toroidal mirror to collect the light within the hemispherical field-of-view, and a CCD camera to convert the collected light to electrical signals. A schematic of the imager is shown if Figure 1. The CCD camera and its electronics are based on a design developed by Rutherford-Appleton Laboratories in UK for the SMEI experiment that UCSD has designed. The CCD camera has an image size of about 2.5 M pixels, or 20 Mbits per image. The combined CCD camera and electronics weigh about 0.7 kg and use 5 W of power. For the solar storm mission, UCSD has estimated that the camera can be reduced in both mass and power, with the power consumption being about 1 W. UCSD has also estimated that the baffle and optics can be reduced in size, at the expense of some degradation in angular resolution, to about 40cm baffle diameter and a total imager mass of about 3 kg. For Configuration 5 only, two imagers were used to allow for imaging both sides of the Sun without rotating the spacecraft for every image. The imagers would alternate taking images, so the overall data rate would be unaffected. For all configurations except Configuration 1, data compression algorithms were used to compress the data transmitted by a factor of ten. The imager parameters used for the study are summarized in Table 1.

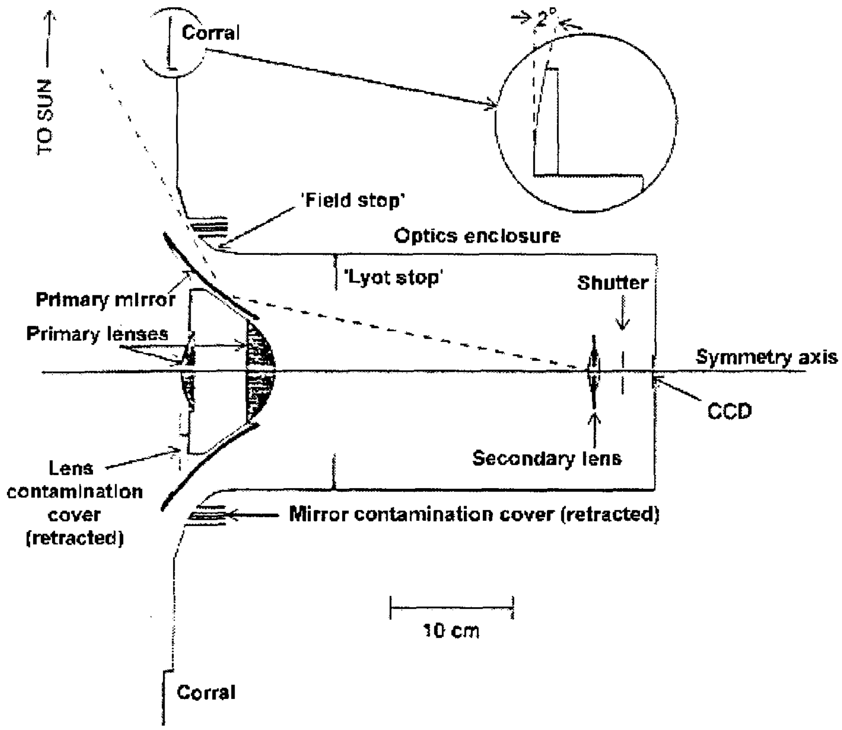

Figure 1 Imager Schematic [5]

Table 1 Imager Parameters

Parameter	Config. 1	Config. 2 - 4, 6 - 8	Config. 5
Quantity	1	1	2
Field of View (deg)	180 x 180	180 x 180	180 x 180
Type of Image	Still	Still	Still
Steerable Field of View	No	No	No
Mass (kg)	3.0	3.0	6.0
Power (W)	1.0	1.0	1.0
Imaging Rate	1 image/hr	1 image/hr	1 image/hr
Imaging Duration	20 min	20 min	20 min
Image Size	20 Mb per image (uncompressed)	2 Mb per image (compressed)	2 Mb per image (compressed)

Magnetometer Design

The magnetometer is used to measure the polarity and magnitude of the solar storm magnetic field as it passes the spacecraft. If the polarity is such that it will couple with the Earth's magnetic field, then geomagnetic currents may be induced and may damage power networks. The information gathered by the magnetometer provides about a half- to one hour warning of the impending activity. The magnetometer selected is a vector magnetometer developed by the Technical University of Denmark for the Oersted satellite. It consists of a 9 cm diameter sphere and an 18 cm by 24 cm by 3.8 cm electronics box. The magnetometer weighs 0.38 kg, and the electronics box weighs 1.7 kg. It is assumed that a 4 meter boom would be used to sufficiently separate the magnetometer from the spacecraft's magnetic fields. This may require that some steps be taken during spacecraft design to reduce the intensity of its magnetic field. The magnetometer would transmit polarity and magnitude data back to the earth once every few minutes; this would be an extremely small quantity of data. Information on the magnetometer is summarized in Table 2.

Table 2 Magnetometer Parameters

Parameter	Magnetometer
Quantity	1
Boom length (m)	4
Magnetometer mass (kg)	0.4
Electronics mass (kg)	1.7
Power (W)	1.1

Payload Mass and Power Estimation

The overall payload mass and power characteristics are summarized in Table 3.

Table 3 Payload Mass and Power Summary

Parameter	Config. 1 - 4, 6 - 8	Config. 5
Quantity of Imagers	1	2
Quantity of Magnetometers	1	1
Mass (kg)	5.1	8.1
Power (W)	2.1	2.1

3. Astrodynamics

The mission orbit for the Solar Storm Microsat is a halo orbit around the Earth-Sun interior libration point, also known as the L1 point. The L1 point is located 1.50 million km (0.01 AU) from the Earth on a line connecting the Earth and Sun. This location always provides an unobstructed view of the sun. It also provides about 1000 to 5000 times the amount of warning time of a Low Earth Orbiting (LEO) satellite, due to the increased distance from the earth and proximity to the sun. The orbit is unique in many respects. It is not centered around a celestial body, rather the orbit is positioned about a (moving) point in space. It is three-dimensional rather than in a 2-D plane and its shape approximates that of a potato chip in appearance. The halo orbit can be described by its dimensions, in this case a 120,000 km by 667,000 km orbit. The long dimension represents the "width" measurement in the y-axis direction, perpendicular to the x-axis direction defined by the line from the Earth to the Sun lying in the ecliptic plane, as is shown in Figure 2. The shorter dimension gives a representation of the "height" of the orbit as it describes the dimension along the z-axis, perpendicular to the ecliptic plane, as shown in Figure 3. The period of this halo orbit is approximately 6 months and it is inherently unstable, requiring small ΔV applications approximately every 50-70 days to maintain, for a total stationkeeping budget of about 10 m/s per year.

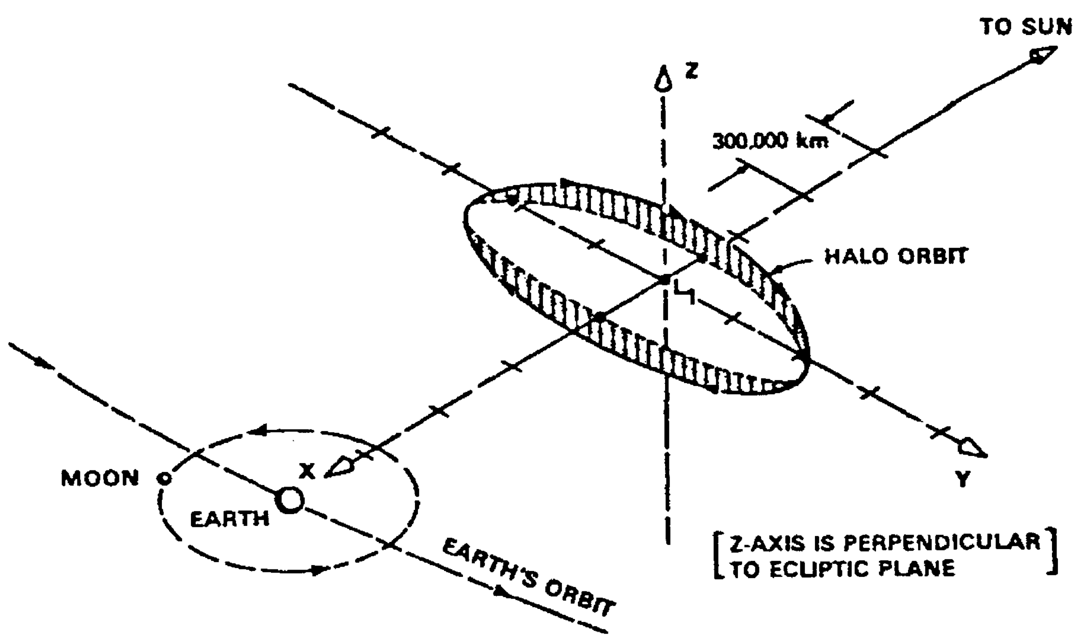

Figure 2. Halo Orbit Orientation and Coordinate Frame [7]

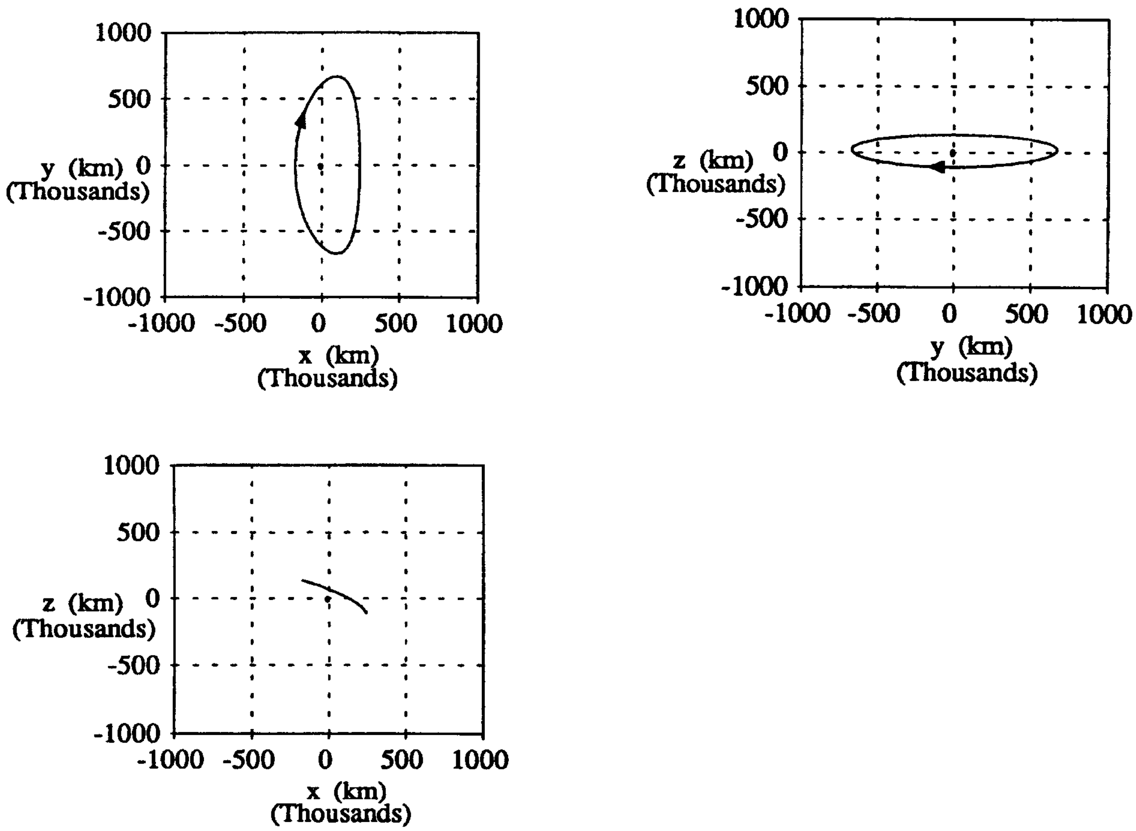

Figure 3 Halo Orbit Dimensions [8]

The selected halo orbit has the intentional characteristic of avoiding solar interference in downlink communications by maintaining a sufficient exclusion angle to ensure that the sun does not appear behind the spacecraft when viewed from the Earth. The maximum communications distance from this halo orbit to Earth is about 1.7 million km.

The assumed Earth-L1 transfer trajectory is depicted in Figure 4, being essentially identical to those used by previous halo orbit missions, ISEE/ICE and SOHO. Transfer is initiated from a 185 km, 28.5° inclined LEO parking orbit and requires two midcourse correction maneuvers. Transfer time for this trajectory is about 110 days. If the trajectory originates from GTO, the required ΔV is significantly reduced, but the transfer time is increased to 190 days. The total ΔV budget is tabulated in Table 4, including a contingency of 25 m/s for the LEO transfer case, 50 m/s for the GTO transfer case, and 10 m/s in the no redundancy, single string configuration.

Table 4 Spacecraft ΔV Budget

	Config. 1,2,5,6,7	**Config. 3**	**Config. 4**	**Config. 8**
Transfer Orbit Origination	LEO	LEO	GTO	LEO
Mission Lifetime at L1	2 years	5 years	2 years	2 years
Midcourse Correction ΔV (m/s)	40	40	40	40
Halo Orbit Insertion ΔV (m/s)	60	60	60	60
Mission Stationkeeping ΔV (m/s)	20	50	20	20
Mission Contingency ΔV (m/s)	25	25	50	10
Total Mission ΔV Budget (m/s)	145	175	170	130

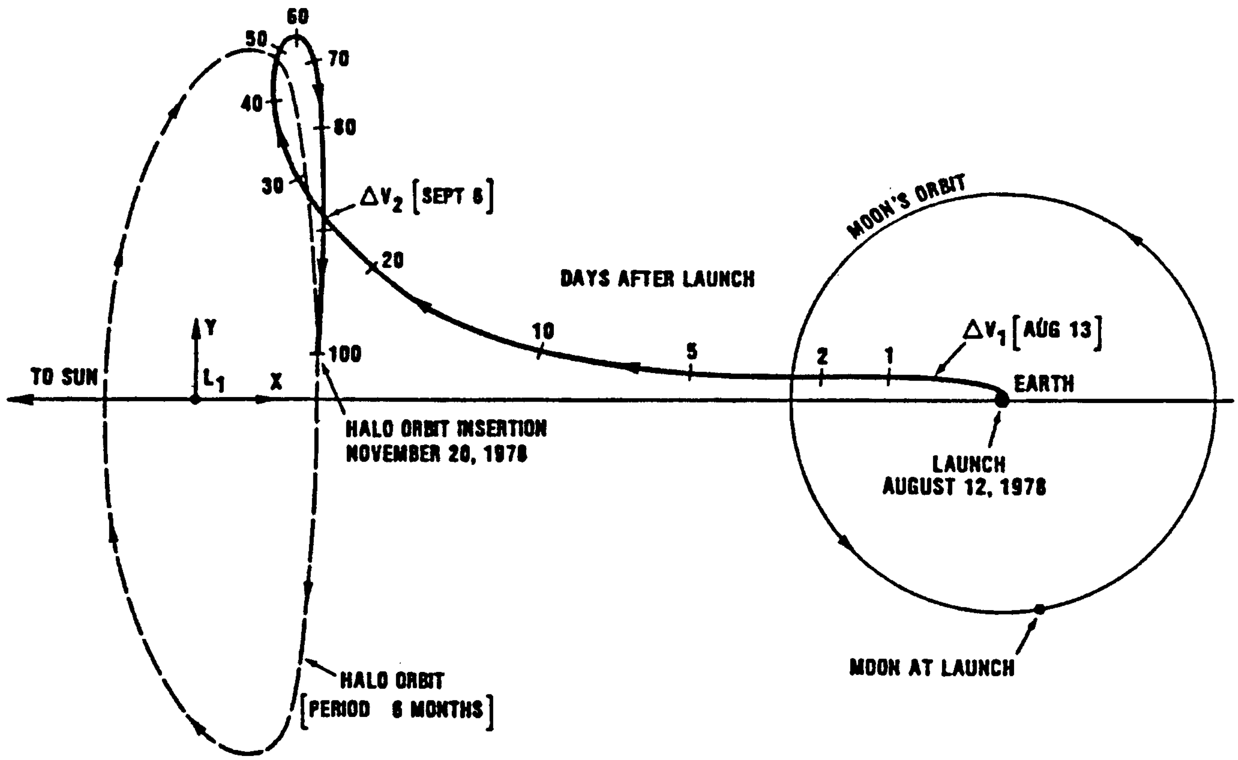

Figure 4 Typical Halo Orbit Transfer (ISEE Example) [9]

Satellite control including launch and early orbit operations, tracking and orbit determination, maneuver planning and commanding is assumed for this concept to be provided by existing ground facilities including the Deep Space Network (DSN) and the Air Force Satellite Control Network (AFSCN). (Feasibility of this concept has not been verified at this time). Though the relatively long transfer times and low ΔV stationkeeping maneuvers could be achieved using very low thrust levels, electric propulsion was not considered for this spacecraft due to its inherent large power requirements in relation to the otherwise minimal power necessary for the rest of the mission.

4. Design Study Trade Space

The primary areas of interest identified were the number of imagers, on-board data compression and data transmission rates, type of orbit injections used, and the design life of the vehicle. Five configurations were studied initially. Due to limitations of the CDC's available database for small spacecraft at the time of the study, additional research was done to improve the database and algorithms and an additional three configurations were studied in a follow-up session. The eight configurations are presented in Table 5. The parameter which was varied for each case has been highlighted in the table.

For the study, eight configurations for a solar storm warning system were developed. A technology freeze date of 1998 and a launch date of 2000 were assumed for all configurations except for Configuration 7, which was assigned a technology freeze date of 2000 and a launch date of 2002. Table 6 presents the goals and requirements for the configurations and Table 7 presents the driving parameters for the first five configurations. From the first five configurations, Configuration 2 was chosen as the initial baseline case. On the third study date, new "small satellite" components and design parameters were added to the spacecraft subsystem models and variations of Configuration 2 were evaluated. Table 8 presents the variations from Configuration 2 for Configurations 6 through 8.

Table 5 Configuration Parameters

Config.	Data Transmission	Life (years)	Orbit Injection	Number of Imagers	Tech. Freeze Date	Redundancy
1	All	2	Direct	1	1998	Selected
2	Compressed	2	Direct	1	1998	Selected
3	Compressed	5	Direct	1	1998	Selected
4	Compressed	2	Transfer from GTO	1	1998	Selected
5	Compressed	2	Direct	2	1998	Selected
6	Compressed	2	Direct	1	1998	Selected
7	Compressed	2	Direct	1	2000	Selected
8	Compressed	2	Direct	1	1998	None

Table 6 Goals and Requirements

Mission	
Payload Description	Hemispherical Imager and Magnetometer
Mission Lifetime	2.3 - 2.5 years (5.3 years for Configuration 3)
Ground Segment Lifetime	3 years (6 years for Configuration 3)
Technology Freeze Date	1998 (2000 for Configuration 7)
Launch Date	2000 (2002 for Configuration 7)
Mission Orbit	L1 Halo (~120,000 x 667,000 km)
Desired Launch Vehicle	Pegasus
Constellation Size	1 spacecraft
Spacecraft	
Redundancy	Selected
Heritage	Commercial Components
Stabilization	3-axis
Reposition Requirements	None
Slew Requirements	180 deg. flip maneuver every hour (or none)
Pointing	Sun-pointing
Pointing Accuracy	0.05 degrees
Attitude Knowledge	0.03 degrees
Environment	Natural

Table 7 Configuration Parameters -- Configurations 1 through 5

Config.	Number of Imagers	Orbit Injection	Data Transmission	Lifetime (Years)
1	1	Direct	All (2 MB / hr)	2
2	1	Direct	Compressed (200 kB / hr)	2
3	1	Direct	Compressed (200 kB / hr)	5
4	1	Transfer from GTO	Compressed (200 kB / hr)	2
5	2	Direct	Compressed (200 kB / hr)	2

Table 8 Configuration Parameters -- Configurations 6 through 8

Config.	Comments
6	Same as Configuration 2, but with new Smallsat components and design parameters.
7	Same as Configuration 6, but with a technology freeze date of YR2000.
8	Same as Configuration 6, but with no redundancy (single string components) in all subsystems.

For this study, it was assumed that the spacecraft would be designed and built with existing off-the-shelf components, and would have a very short development schedule. For these reasons, a mass and power contingency of only 15% was accounted for in the payload and spacecraft bus to reflect this type of design effort.

5. RESULTS

The assumptions, effects of those assumptions, and basic subsystem design are described below.

Ground

Several assumptions were made regarding the operations and ground segment design for the Solar Storm Microsat. Note that these assumptions have not been verified and no discussions have taken place with these organizations. It was assumed that mission command and control would be performed from NASA's Integrated Mission Operations Center. The mission data and telemetry would be continuously received by the Rutherford-Appleton Laboratory in Great Britain and the Communications Research Laboratory in Japan while in view. These two facilities are currently used in this capacity for ACE, and it was assumed that there would be no conflicts between the two missions. When necessary, the Air Force Satellite Control Network (AFSCN) would provide connectivity to the satellite. For configuration 1, which required higher mission data rates due to the uncompressed image data, it was assumed that the Deep Space Network (DSN) would be used for connectivity.

Communications

The AFSCN would be used at S-band for connectivity during launch and early orbit operations. Spacecraft Telemetry, Tracking and Command (TT&C) data would be relayed at S-band to and from one of the two dedicated ground stations during the transfer and operational orbits. For configuration 1 (high data rate), X-band would be used to relay data between the satellite and the DSN to take advantage of higher data rates and reduce the amount of time using the DSN. In the study, it was assumed that an S-band package would need to be carried for connectivity to the AFSCN. However, some of the AFSCN Remote Tracking Stations (RTSs) are being upgraded to include X-band, which might eliminate the need to carry an S-band package for this case.

Attitude Determination and Control System (ADCS)

The attitude determination requirement of 0.03° drives the ADCS to use star trackers, though of very low accuracy. Several lightweight (<5 kg) star trackers were identified. The rest of the sensor suite consists of three two-axis gyros for use during slew and recovery from safe modes and two sun sensors for initial sun acquisition, for sun- or other safe-hold modes, and as a backup to the star trackers in the event of star tracker failure.

The slew requirement of 180 degrees in 30 minutes drives the sizing of the torque and momentum capability of the reaction wheels, though the smallest wheels that were identified still had much larger capability than that required. Only 3 reaction wheels were selected, rather than the usual set of 4+ due to the short mission life requirement (2 years).

Use of magnetic torquers to dump accumulated momentum was not an option for this mission due to the weak magnetic field at this orbit, therefore thrusters were selected. However, also due to the orbit, the spacecraft is subjected to very minimal disturbances, so that momentum accumulation (and dumping) is very small.

Power

Twenty percent efficient Gallium Arsenide (GaAs) multi-junction solar cells were assumed for the Solar Arrays. A single small nickel cadmium battery with a capacity of 0.20 A-hr was sized to supply the backup energy storage. Both the batteries and the spacecraft would operate at 28.0 volts. The solar panel would be the entire aft end of the bus.

Propulsion

A blowdown monopropellant hydrazine system was selected for the spacecraft propulsion system used for stationkeeping and attitude control, due to the small delta-V requirement. Electric propulsion was investigated but would be inefficient due to the high dry/wet mass ratio and the high electrical power requirements.

Thermal

The thermal environment of the Solar Storm Microsat would be fairly constant and benign. The configuration in the halo orbit always keeps one side of the spacecraft pointed at the sun, which facilitates using the spacecraft's sides and earth-facing panel as radiators. The focal plane should be maintained at a temperature of -80°C; this should be easily achievable if the primary sensor has a clear view to space and is shadowed from the sun. The thermal control system would then be a passive system consisting of multilayer insulation (MLI), surface finishes, thermal isolators and heaters.

Summary

A summary of the mass and power budgets for all of the configurations is presented in Table 9 for comparison. Detailed subsystem weight and power breakdowns are included in the appendix. The first five vehicles can be accommodated by the Taurus launch vehicle with large launch margins. Configuration 6 through 8, which incorporated small satellite components and design parameters, can be accommodated by the Pegasus launch vehicle.

Transmitting all of the data, as in Configuration 1, rather than compressed data did not have a significant effect on the overall spacecraft weight. Because of the potential difficulties in using the DSN, transmission of compressed data was selected for the baseline.

Extending the life from two years to five years also had a minimal effect on spacecraft sizing. The only real changes were additional propellant for stationkeeping and the addition of a reaction wheel for the increased reliability required to attain the longer life.

Getting to orbit by doing a transfer from GTO was effective from a weight perspective. However, the penalty of adding almost three months to the transfer time when the mission life is only two years was deemed unacceptably high.

The single imager concept conserved momentum in the rotation, so that there was no significant difference in the ADCS. If the data rates had been much higher, the two-imager might have been required to allow more time for data transmission. Alternatively, a faster slew could have been accommodated with the selected reaction wheels. Consequently, there was no need to go the greater weight of the two-imager concept.

Substantial improvement was made between the first five and final three configurations, when additional research was conducted to identify smaller, lighter components. Key contributors to the weight reduction were the ADCS star trackers and reaction wheels.

Extending the technology freeze date had the effect of reducing the spacecraft mass, but only by a very small amount. Likewise, eliminating all redundancy saved a small amount of weight. However, the increased risk associated with doing this was not considered to be a reasonable trade-off to the small weight savings.

Table 9 Configuration Comparison

	Configuration							
	1	2	3	4	5	6	7	8
Payload Mass (kg)	5	5	5	5	8	5	5	5
Payload Power (W)	2	2	2	2	2	2	2	2
Spacecraft Mass (kg)	130	130	138	139	134	84	77	73
Spacecraft Power (W, BOL)	146	159	163	159	159	127	119	121
Solid Motor Mass (kg)	--	--	--	50	--	--	--	--
Launch Mass (kg)	137	136	145	195	140	87	79	75
Taurus LV Launch Margin (kg)	238	239	230	322	234	288	296	300
Pegasus LV Launch Margin (kg)	-32	-31	-40	-70	-35	18	26	30

Spacecraft Configuration

Configuration 6 was selected as the baseline Solar Storm Microsat. A conceptual layout for the spacecraft is presented in Figure 5 for the deployed configuration. The spacecraft bus is approximately 1.0 m in diameter and 0.5 m in height. When deployed, the magnetometer boom extends 4.0 m towards the sun. Eight thrusters are configured with four towards the sun and four towards the earth. The hemispherical imager is mounted on one of the sides of the spacecraft so as to point perpendicular to the earth-sun line. The star tracker is mounted to another of the sides in a similar fashion. The 0.6 m (2 ft) S-band antenna is mounted on the earth-facing side within the launch adapter ring. Two sun sensors are mounted to the sun-facing panel at 45° angles so as to provide sun acquisition during tumbling. Two S-band omni antennas, one mounted on the earth-facing panel and one mounted on the sun-facing panel, provide earth capture during tumbling. Figure 6 presents the launch configuration of the spacecraft in relation to both the Taurus and the Pegasus fairings. Note that the Pegasus requires a STAR-27 solid motor to deliver to the L1 transfer orbit, which constrains the height of the spacecraft. As designed, the spacecraft just fits within the Pegasus fairing.

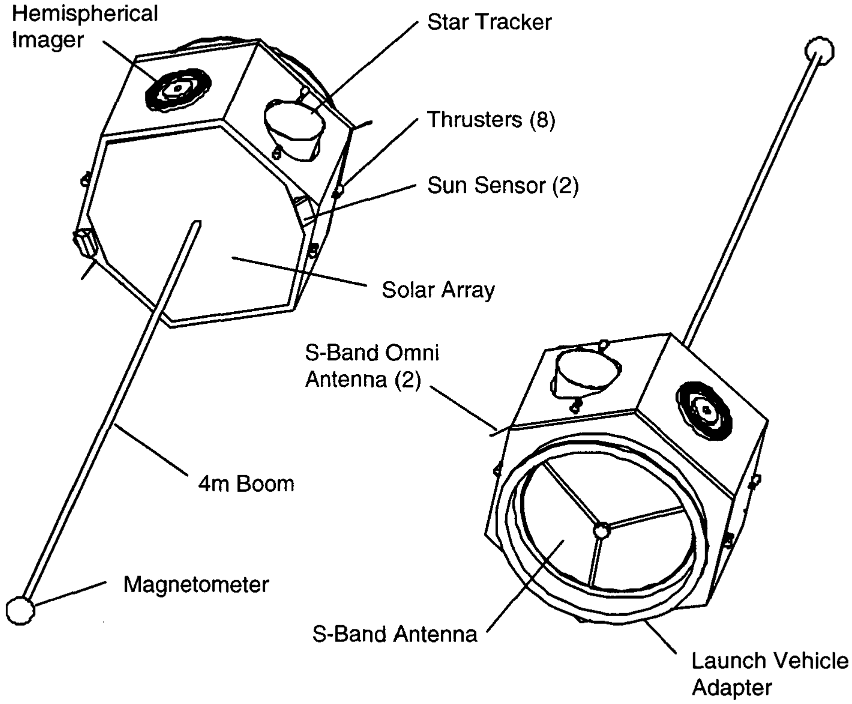

Figure 5 Conceptual Spacecraft Layout

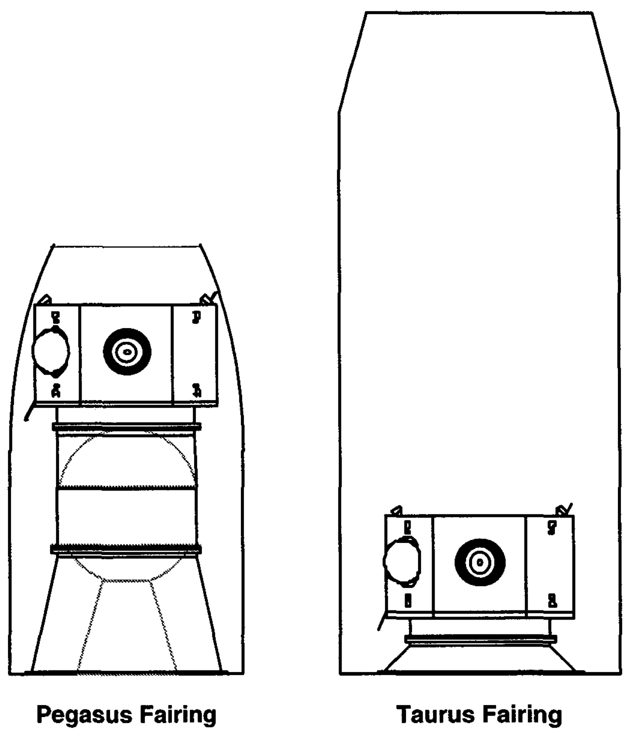

Figure 6 Spacecraft Launch Configuration

6. CONCLUSIONS

It has been demonstrated that a Microsat-class spacecraft (typically defined as less than 100kg) can be designed to perform the Solar Storm warning mission proposed herein. The satellite design fits, both in terms of weight and size, on a Pegasus launch vehicle and requires no new technology. The concept has the potential to provide great benefits for both terrestrial and space applications.

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BIOGRAPHY

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ACKNOWLEDGMENTS

This Solar Storm Microsatellite concept was devised by John Hussey of The Aerospace Corporation. The study was performed for The Aerospace Corporate Strategic Initiative under the Microsatellite Task managed by Ernie Robinson. Leon Gurevich was the CDC Study Co-lead. Hemispherical Imager parameters were provided by Andy Buffington of the Center for Astrophysics and Space Sciences at the University of California, San Diego. The study was conducted with the following Aerospace Corporation experts: Vince Canales (Cost), Douglas Daughaday (Processing), Andrei Doran (Controls), Marilyn Dubas (Ground), David Glackin (Space Science), Terry Hershey (Power), David Gilmore (Thermal), Chris Kobel (Astrodynamics), John Le (Propulsion), Deanna Mains (Astrodynamics), David McKenzie (Space Science), Rhoda Novak (Software), John O'Donnell (Communications), Michael Papadopoulos (Structures) and Christopher Taylor (Solid Modeling).