The IRIS Project: A new architecture for a small satellite imaging mission.

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ABSTRACT: Utah State University students have developed an earth-imaging satellite mission designed specifically for the promising, low-cost RASCAL launch platform. Designed over a semester-long space system design class, the mission is intended to capture high quality images of earth from a low-Earth orbit. The total mission cost is estimated at $10 million, including satellite design, fabrication, launch, and mission operations. Students used the university's new Space Systems Analysis Laboratory, a concurrent design center, to assess design trades and develop the system architecture. The mission architecture, mission elements, and satellite design are presented.

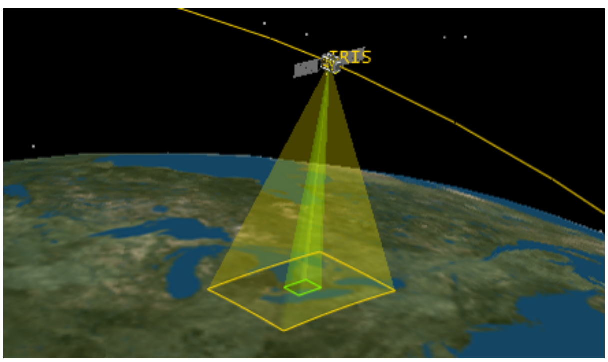

Figure 1: IRIS performing its mission.

MISSION OVERVIEW

Recent and pending technological advances are opening up new opportunities and markets for space-based imagery. With the potential inexpensive access to space possible with the Responsive Access Small Cargo Affordable Launch (RASCAL) vehicle currently in development, and the current advancements in digital imaging technology, the opportunity is open for new spacecraft missions that produce inexpensive imaging of earth. The Innovative RASCAL-launched Imaging Satellite (IRIS) mission was developed for a space systems design class at Utah State University (USU) to address this opportunity. Specifically, the IRIS mission is to generate high quality color images of earth from a low Earth orbiting (LEO) satellite and distribute the images over the Internet using a standard image format.

The flexible, innovative design uses new technology while leveraging existing design elements from heritage USU satellite programs. It is designed for a 19 month development and a launch in December 2006. IRIS has a 3-year mission life and is operated using two ground stations with a primary operations center in Logan, UT and a secondary facility in Bedford, MA. Figure 1 shows a Satellite Tool Kit (STK) image of IRIS performing its mission.

The IRIS mission is composed of all the elements required for image generation to image distribution. The mission architecture, shown graphically in Figure 2, includes imaging, spacecraft, launch system, ground station, operations, and distribution segments. Table 1 describes key technical characteristics of the spacecraft.

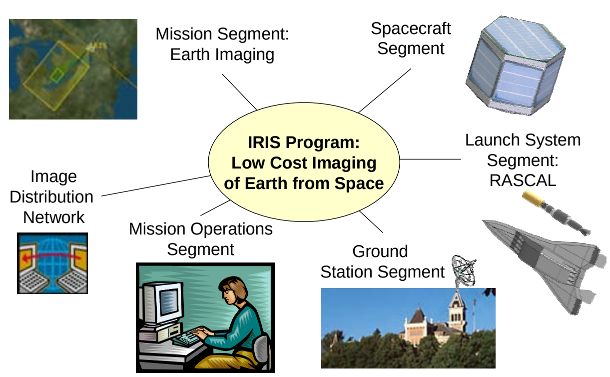

Figure 2: The IRIS mission includes all segments necessary to generate and supply images.

Table 1: IRIS Spacecraft Overview.

Mission:	Earth Imaging
Orbit:	900 km, 60° inclination
Mass:	49 kg
Size:	50 cm by 60 cm diameter
Structure:	Aluminum Isogrid
Imaging:	2x 11 Megapixel cameras 30 m & 150 m resolution
TT&C:	10 W S-band transmitter 1.9 Mbps Downlink
C&DH:	Hitachi SH7709 CPU
ADCS:	3 Reaction Wheels 3 torque coils Magnetometer Horizon Sensor Sun Sensor
G&N:	GPS
Thermal:	Active heater control
Power:	39 W orbit average power 4 GaAs Solar Panels 62 W-hr Lithium Ion battery

DESIGN CENTER

Utah State University, realizing the potential of concurrent engineering facilities in the academic setting, established the Space Systems Analysis Laboratory (SSAL), a state-of-the-art conceptual design center similar to design centers at government agencies such as NASA's Jet Propulsion Laboratory and Goddard Space Flight Center as well as The Aerospace Corporation. To augment Utah State University's growing space engineering capabilities, this facility will be used for teaching space systems courses as well as for engaging in space systems designs. The SSAL concurrent engineering facility is fully equipped and well suited for teaching and space systems design. Its capabilities are on par with concurrent engineering facilities in industry.

The core of the SSAL facility consists of nine PCs, connected together through a Local Area Network and file sharing server, each one outfitted with a plethora of tools for space systems design, ranging from standard software such as Microsoft Office to trajectory analysis software such as STK. The computers are laid out in a U-configuration to allow easy interaction among team members and to allow individuals at each workstation to easily turn their attention to the team lead when necessary. If focus is needed on any one specific subsystem at any time, a projector at the front of the room can display the output from any one of the computers instantaneously.

The design of the IRIS mission was the first major application of the SSAL. The IRIS design team, which included seniors as well first year graduate students, is shown using the SSAL in Figure 3. In the process of using the SSAL in designing the IRIS mission, the team also validated proper set-up of the facility. While the MAE 5530 course is primarily aimed at teaching students about the end-to-end design of a space system, a valuable component of the course is to provide students with the opportunity to perform their work in a concurrent engineering setting. Towards that end, students performed their design work in a manner similar to that in which design teams in industry produce their conceptual designs. Many class meetings were analogous to design sessions in industry. The IRIS team divided themselves into subsystem teams and located themselves at specific workstations in the lab. The subsystem teams then collaborated with one another during these sessions, openly sharing ideas and discussing with each other aspects of the design and any tradeoffs that needed to be made or alternative approaches that could improve the design. Using this facility allowed the students the opportunity to rapidly study various design options as well as to ensure completeness and integration. More information on the design center can be found in reference 1.

Figure 3: The IRIS design team used the Space Systems Analysis Laboratory in the design of the mission.

IRIS IMAGING SYSTEM

The heart of the IRIS mission is the imaging system. Based on two Redlake ES 11000 cameras, the system is designed to produce 11 Mega pixel images with either 30 m or 150 m resolution. The larger resolution will be used for regional targets, such as weather, while the lower resolution is intended for localized targets, such as cities, forests, or coastal areas. The images are produced by RGBG CCD arrays with 4008 by 2672 active pixels. This solution produces large color images using off-the-shelf technology that requires only modest changes to adapt to space applications.

The two-camera approach increases operational flexibility and reduces mission risk. Failure of one camera will limit but not eliminate the mission. The number of images taken at each of the two resolutions can be adapted to market needs.

The imaging system can produce high-quality images of either regional or local points of interest. Table 2 lists a number of potential targets, divided into market categories. With the large pixel size, these images could be subdivided into smaller pictures without significantly reducing the image quality.

Table 2: Potential markets for IRIS images.

Market Category	Potential Targets
Science	- Weather: weather patterns, weather images
Government (non-military)	- Disaster response: weather imaging, disaster evaluation - Forestry Service: forest data, forest fires - Land use: population growth, deforestation - Coastal: shoreline data, shipping routes
Military	- Weather: cloud cover, snow locations, storms - Situational Awareness: location of cities or encampments - Cueing: find targets for other assets to investigate
Civilian	- Media: image locations of interest - Education: geography, student mapping and imaging

Images will be compressible on-board the spacecraft with either pre-planned compression or post-imaging compression. For pre-planned compression, the operator will define the compression to take place during on-board image storing. The other option is to store the full, uncompressed image on the spacecraft then determine the compression level after viewing a thumbnail-style image.

JPEG was selected as the compression and imaging format. This standard provides various levels of compression while maintaining a high image quality. Table 3 shows some of the common compression options planned for use with the IRIS mission.

Table 3: Compression Options.

JPEG Image Information Loss	Average Final Image Size
5%	25%
25%*	13%
90%**	5%

* Typical of images viewed on the Internet.

** Typical of thumbnail images viewed on the Internet.

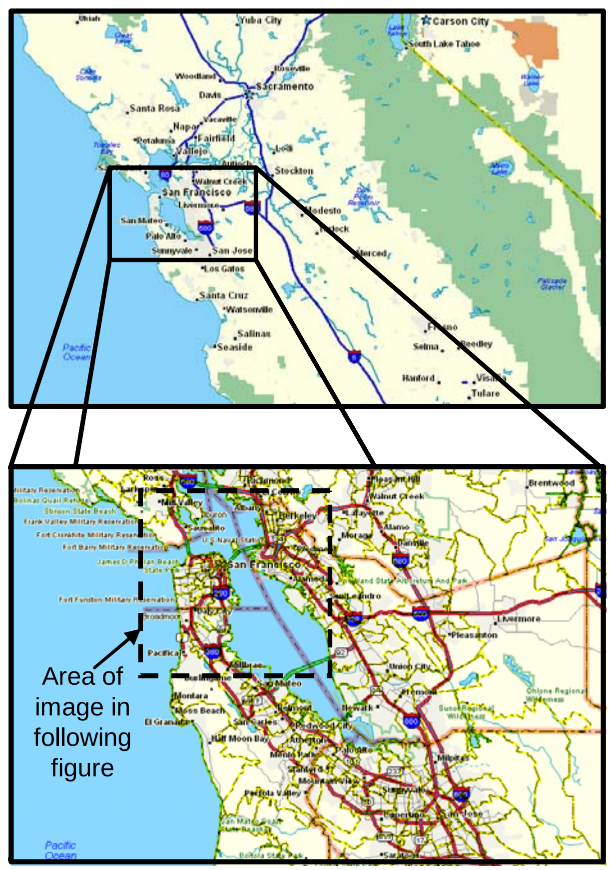

Figure 4: Representative image sizes for the 150 m resolution camera at 400 x 600 km (top) and 30 m resolution at 120 x 80 km (bottom). The dashed line inset in the lower map shows the approximate size of the images shown in the next figure.

Figure 5: Taking images at 150 m resolution allows crisp and clear pictures of large areas. To zoom in, a higher resolution image is required. Compare the second 150 meter resolution with the corresponding 30 meter resolution image at the bottom.[2]

ORBIT

The orbit selected for the IRIS mission is a 900 km, 60º inclination, low-earth orbit. This orbit balances the needs of imaging, communications, and power generation. The orbit has a period of 103 minutes, with approximately 14 orbits per day. The satellite revisits the same location within 6 days, improving imaging opportunities. In addition, slewing the spacecraft will allow the same point of interest to be imaged on three successive orbits. The 60º inclination allows coverage of nearly all major population centers. The altitude and inclination also provide sufficient communications access and power generation.

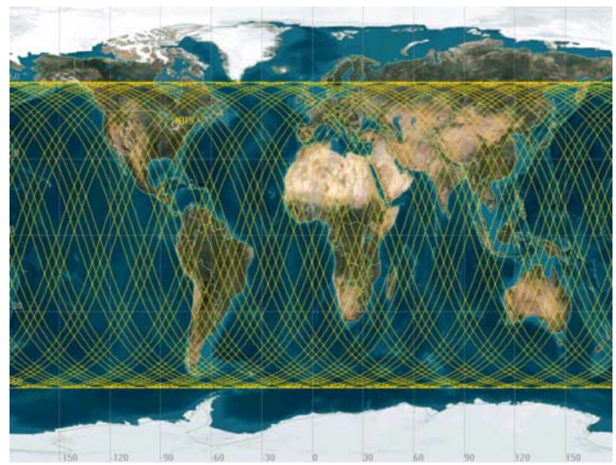

Figure 6: The IRIS orbit ground track shows coverage over nearly all major population centers.

COMMUNICATIONS

The primary mission of any satellite communication system is to relay collected data to the ground for analysis. Requirements for the quantity of images to be collected were not given for this design project. Without specific requirements given it was necessary to look at what this satellite is capable of with the resources on hand. Power available on the satellite and the link budget are the main constraints on the communications system. A balance between the limiting factors was found with a 2 Megabit connection using a 10 W S-band transmitter with QPSK modulation. This provides an adequate data rate to allow sixty-five uncompressed 4008x2672 pixel images to be downlinked per day with two ground stations.

Once the link requirements had been established a specific frequency for operation was selected. The mission requirements stated that this satellite should be flexible and allow for multiple customers, waveforms, and even ground stations. No single frequency band is useable by commercial, military, and non-military government entities. In a traditional communications system this flexibility would require distinct transponders for each customer and application. Recent research at Utah State University on a highly flexible Software Defined Radio (SDR) solves this problem.

An SDR capitalizes on the recent technological advances in high speed Digital Signal Processors (DSP) to process the transmitted and received signals in the digital domain. Software can then be changed to adjust the way in which the data stream is handled using different modulations and error-correction codes. This facilitates hardware reuse, thereby reducing risk, cost, and development time. The software can be scaled to meet the changing customer's requirements.

Incorporating a modular hardware design with the SDR allows the same basic radio to be used over multiple frequencies and bandwidths, lending flight heritage to future missions. A standard interface on the SDR attaches to multiple optimized RF modules, allowing a single SDR to function as the equivalent to a number of independent transponders. The SDR can then be reconfigured to change frequency, bandwidth, and modulation even after launch.

The communications system was designed assuming two ground stations. The ground stations are located at Utah State University, in Logan, UT, and the Stewart Radiance Laboratory, a university-owned and operated facility near Boston, MA. Figure 7 shows the overlapping communications coverage provided by these two ground stations.

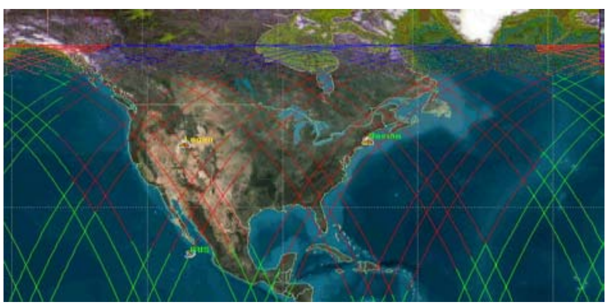

Figure 7: As illustrated by this coverage map, the use of ground stations in Logan, UT, and Boston, MA, gives IRIS sufficient coverage to downlink an average of 65 uncompressed images per day.

Table 4: The communications system allows the following number of images to be downlinked, on average, each day.

Image Type	Number of images per day
24-bit bitmap (uncompressed)	69
12-bit greyscale (uncompressed)	139
JPEG, 75% image (standard JPEG image)	559
JPEG, 10% image (thumbnail image)	11742

COMMAND AND DATA HANDLING

Utah State University has developed a low-power command and data handling (C&DH) subsystem specifically for use inside small satellites. This system is also designed to be radiation resistant and thermally tolerant over a wide range of temperatures. IRIS has some distinct differences from the missions for which this system was originally designed; however, it was determined that with minor modifications on a few subsystem boards this C&DH system is ideal. The use of this in-house system, shown in Figure 8, represents a great cost savings over the purchase of a commercial C&DH system. Furthermore, the system will receive flight heritage with the launch of USUSat II and has already been thoroughly tested to verify its functionality in a hostile environment.

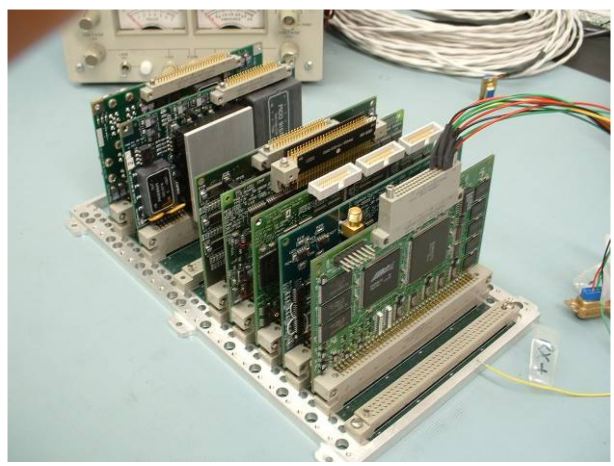

Figure 8: The IRIS spacecraft processing is based on a low-power computer developed for USUSat II.

Modifications to the C&DH system are required on the Telemetry board, and a Camera Data Acquisition (CDA) board must be added. Redundancy in hardware and software will be implemented in the CDA to provide radiation tolerance on the same order of that of the rest of the CDH subsystem, thereby not degrading the overall subsystem performance.

The cameras selected for IRIS require that the CDA be able to capture data at minimum of 257 MHz data rate. A FireWire connection is used between the CDA and cameras to accommodate this high data rate. The clock rate of the CPU board is only 80 MHz and isn't equipped to handle this massive data flow. For this reason a Digital Signal Processor (DSP) will be included on the CDA that will process the image data as instructed by the CPU board. The DSP will also provide a high data rate transfer directly to a modified telemetry board for downlink. This mitigates the problems of the relatively slow bus speeds of the existing C&DH subsystem.

The 220 Mbytes of storage on the CDA allow for various operational scenarios. Compressed images can be stored, using pre-defined compression instructions, or uncompressed images can be stored with thumbnail images that are downloaded for user-selected compression and image download on a subsequent pass.

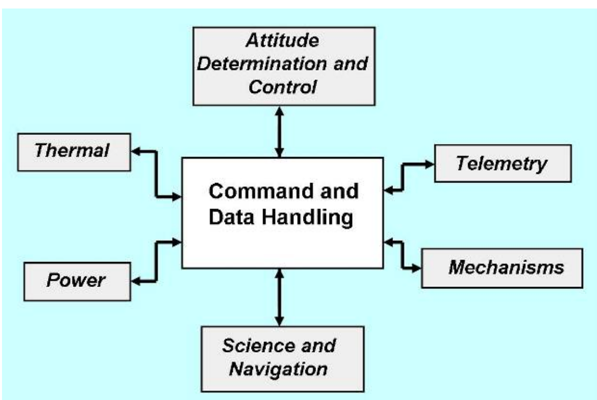

Figure 9: IRIS C&DH subsystem interfaces are shown in this simple block diagram.

ATTITUDE DETERMINATION AND CONTROL

The IRIS Attitude Determination and Control Subsystem (ADCS) is designed to provide full 3-axis control. The high accuracy in pointing required during imaging mode is the key design driver. The IRIS spacecraft has been designed with flight proven ADCS components to ensure this key functionality is reliable and robust. Attitude determination is accomplished using a magnetometer, horizon sensor, and sun-sensors. Momentum wheels are used to control the attitude, while torque coils are used to desaturate the wheels. A high accuracy Global Positioning System (GPS) receiver provides positional and timing knowledge. ADCS subsystem requirements are shown in Table 5.

In order to provide reliability and robustness, IRIS has been designed using flight proven equipment. All equipment can be found off-the-shelf and allow the IRIS spacecraft to take advantage of the expertise of the respective manufacturers. Figure 10 shows the components represented on the IRIS satellite.

Table 5: Attitude Determination and Control Requirements.

Mode	Determination Requirements	Control Requirements
Acquisition (de-tumbling)	Accuracy: 0.25º, 3 axis	Accuracy: 10 º, 3 axis Range: full range Drift: 1 deg/hr
Imaging	Accuracy: 0.25º, 3 axis Range: 30º of nadir	Accuracy: 0.5º, 3 axis Range: 30 º of nadir Jitter: TBD
Communication	Accuracy: 0.25º, 3 axis Range: 30º of nadir	Accuracy: 0.5º, 3 axis Range: TBD
Safe (sleep)	Accuracy: 0.25º, 3 axis Range: 30º of nadir	Accuracy: 10 º, 3 axis Range: 20º of nadir

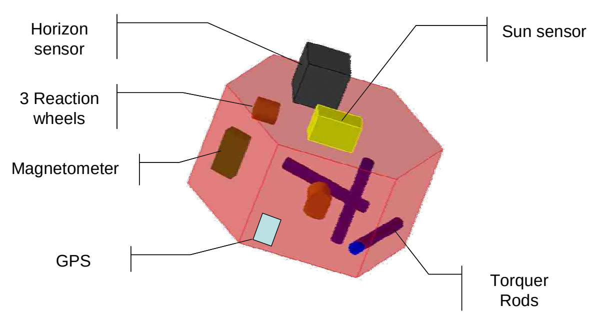

Figure 10: Attitude Determination and Control components.

THERMAL

The IRIS thermal design is based on passive radiators and heat paths, augmented with Kapton film heaters that provide a safety net for sensitive components. These elements produce a simple, reliable, cost effective, and energy efficient design. The thermal subsystem is designed to assure that all satellite components remain within the prescribed operational temperature ranges given in Table 6.

Table 6: Operational Temperature Limits.

Components	Operational Temperatures °C
Cameras	0 to 40
Flight Computer	-40 to 80
Battery	-20 to 60
Solar Arrays	-180 to 95
Sun Sensors	-50 to 80
Magnetometers	-80 to 80
Reaction Wheels	-40 to 70
Torque Rods	-50 to 80
Star Camera	-20 to 50
GPS	-20 to 50
Transmitter	-40 to 85
Antennas	-65 to 95

POWER

The power system for IRIS is a conventional, but reliable system based on body-mounted solar array panels for power generation and a battery for power storage. The IRIS satellite requires an orbit average power of 30 Watts. The design maintains a 30% power margin by producing an orbit average of 39 Watts. This is accomplished using 4 solar panels with a total solar array surface area of 6746 cm^2 distributed on three side panels of 1400 cm^2 each and a top panel of 2546 cm^2. IRIS will use high efficiency (~26%) Emcore triple junction GaAs solar cells. Each cell produces 2.1 V, and will be configured strings of 16. The cells have a convenient size of 30 sq. cm allowing a better than average packing factor.

For storing power, the power system will use Lithium-Ion batteries to maximize performance. Li-Ion has been space qualified in recent missions, and is also used widely in commercial applications. The 1-year mission life and 3-year mission goal resulted in a derived requirement that the batteries have a cycle life of approximately 17500 cycles. The maximum mass allowed for the spacecraft also limits the choice of batteries, which are one of the heaviest satellite components. The number of cells in the battery depends on the battery capacity required and individual cell voltage, as well as the spacecraft bus voltage required. IRIS will use 8 cells of 3.7 V each in series to provide a nominal 28 V bus and a battery capacity of 62 W-hr. Li-Ion batteries have the highest volume and mass energy densities (466W-hr/L and 167 W-hr/Kg, typically), a wide operating temperature range (-20 to 60 deg C), and ample cycle life.

The power electronics is designed around the existing USUSat II design with upgrades and modifications as required. To ensure flexibility, the electronics is designed for various bus voltages, complementing the modularity of other subsystems.

Subsystem	Average Power
C&DH	3.5
Imaging	2.5
Comm	4.8
ADCS	7.9
Power	4.0
Battery Charging	7.6
Avg power consumption per orbit	**30.4**
30% Safety Margin	9.1
Avg solar power required	**39.5**
Required Average Solar Power During Sunlight	59.2
Required Battery Capacity	11.4
Required Battery capacity with 20% DOD	**56.9**

Figure 11: Power budget by subsystem.

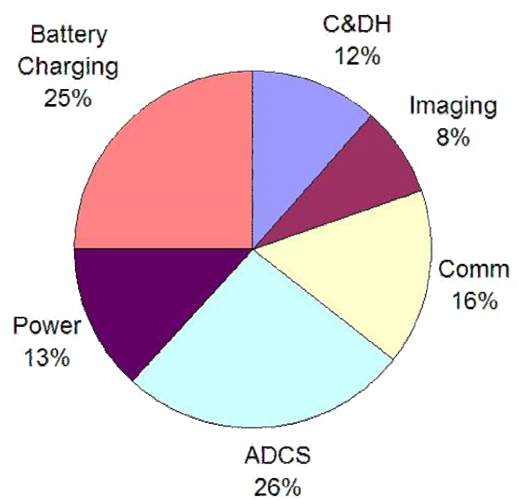

Figure 12: Power budget distribution by subsystem.

STRUCTURE

The IRIS spacecraft design, shown in Figure 13, is a simple hexagonal structure. Each of the six side panels, as well as the top and bottom panels, are of a light-weight aluminum isogrid design. The simple, easily manufactured panels are constructed to ease integration, reduce fasteners, and improve modularity. Each panel is directly attached to its neighbors, removing the framework typical of many satellite structures. The isogrid assembly is stiff and strong enough to ensure survival of launch loads. Although not as light as honeycomb panels, the isogrid panels provide more radiation shielding for sensitive internal components.

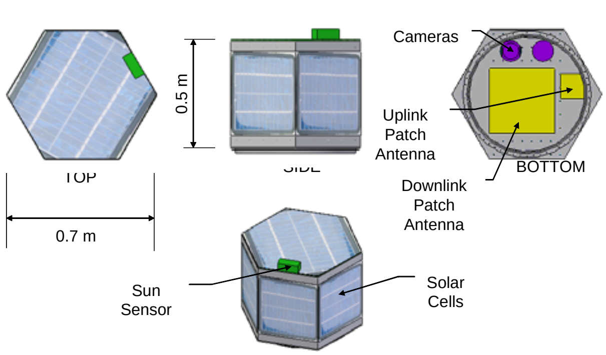

Figure 13: External view of the IRIS spacecraft.

Internal components, shown in Figure 14, are mounted directly to the panels. Each of the panels is also designed to be assembled and tested as an individual unit, reducing assembly, integration, and test time.

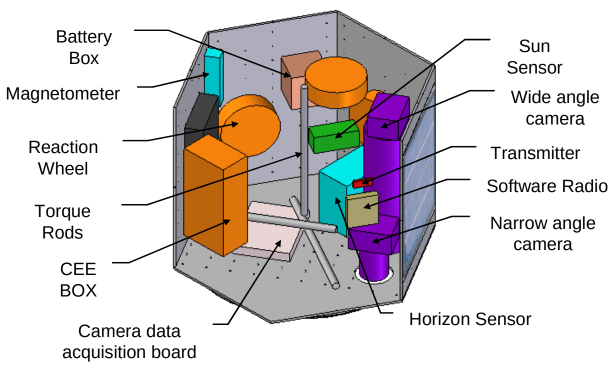

Figure 14: Internal view of the IRIS spacecraft.

SPACECRAFT MASS

Due to the selection of the RASCAL launch vehicle, the maximum launch mass to the selected orbit is approximately 80 kg. The current best estimate (CBE) mass is less than half of this requirement, at 36.5 kg. With a growth margin of 30% the mass climbs to 48.9 kg. The distribution of mass is shown graphically in Figure 15. Tabular data, by subsystem is shown in

Figure 16.

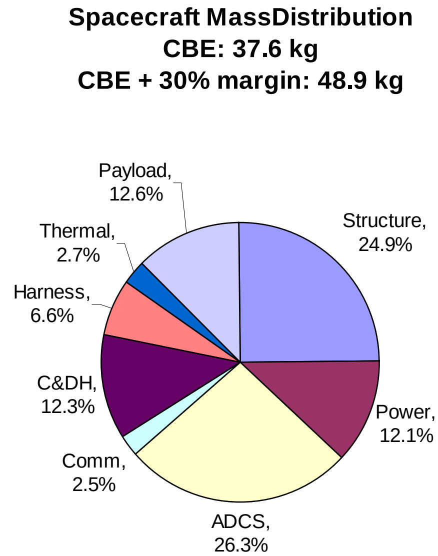

Figure 15: Mass distribution by subsystem.

Subsystem	CBE Mass (kg)	CBE + 30% Margin (kg)
Structure	9.1	11.8
Power	4.4	5.7
ADCS	9.6	12.5
Comm	0.9	1.2
C&DH	4.5	5.9
Harness	2.4	3.1
Thermal	1.0	1.3
Payload	4.6	6.0
Subtotal:	**36.5**	**47.5**
Lightband	1.1	1.4
Total on-orbit:	**37.6**	**48.9**

Figure 16: IRIS current best estimate of mass and estimate with 30% growth margin.

LAUNCH VEHICLE

The IRIS spacecraft was design specifically for the RASCAL launch vehicle. This launch vehicle is a partially reusable, three-stage vehicle. The first stage uses Mass Injection Pre-Compressor Cooling (MIPCC) technology for thrust in an airplane-like configuration. The first stage is fully reusable. The second and third stages are expendable rocket stages, providing the final boost to orbit. This launch vehicle is designed to launch 50 kg payloads into a 1250 km sun-synchronous orbit or 80 kg to the IRIS orbit at 900 km 60º inclination. The launch vehicle provides a fairly soft ride, with estimated 4g lateral and 5 g axial loads. The vibration environment is estimate to be above 50 Hz in the axial and torsional directions and above 40 Hz in the lateral direction.[3]

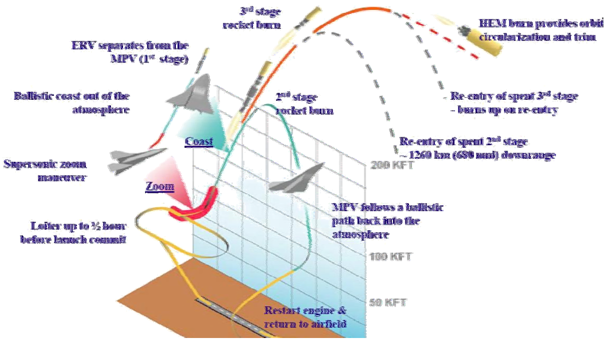

Figure 17: Launch sequence for the RASCAL launch vehicle.[3]

MISSION COST

The IRIS team used a bottoms-up approach to determine a total cost for the mission. Using estimates for components, labor, and manufacturing costs, it was estimated that the spacecraft development, launch, and mission operations costs would be under $10 million ($US). Mission operations costs included the use of existing ground station hardware that requires only modest refurbishment and facility costs to become fully operational.

Table 7: Estimated Mission Cost.

	Cost (millions)
Spacecraft development	$5.1
Launch and Check-out	$2.2
Operations	$2.4

CONCLUSIONS

The IRIS mission design shows the feasibility of generating high quality imaging from a low-cost, micro-satellite in low Earth orbit. The two-camera approach used by the IRIS team allows adaptability to market needs and reduces risk to the mission in the event that one camera fails. The use of off-the-shelf and flight-proven components reduces risk, schedule, and cost.

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