Smart Escort Satellite: A Modular Solution for On-Orbit Rendezvous and Inspection

DARPA/SBIR Program

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1.0 Introduction

The SCOUT concept was derived from AeroAstro's Wooden Round Microsatellite (WoRM) proposal submitted to DARPA in December of 2002. WoRM incorporated the following key features:

- *Multi-Mission Compatibility* – Tactical communications; Intelligence, Surveillance and Reconnaissance (ISR); or weather observation are all possible missions for a WoRM.
- *High Payload Mass and Power Fractions* – To maximize their effectiveness, the majority of the mass of a WoRM must be payload. We will aim to improve on the traditional 30% mass fraction by at least 30%, i.e. achieve a payload mass fraction of 60%. Achieving a similarly high payload power fraction is equally important.
- *Long Shelf-Life* – WoRMs need to be storable for several months or years in the field with limited or no maintenance, much like an ammunition round.
- *Rapid Call-Up for Launch* – As with any modern weapon, the WoRM must be able to be readied for use within hours by skilled military personnel. To achieve tactical timelines, the WoRM must require the minimum possible field integration and have built-in test equipment (BITE).
- *Rapid Initialization on Orbit* – The tactical advantage of a WoRM would be lost if check-out and initialization took 30-60 days as with current small satellites. WoRMs must initialize on ascent and be ready to perform their mission on their first orbital pass.
- *Manufacturability* – WoRMs will be built in substantially higher quantities than most spacecraft. While 'mass' manufacturing techniques are not necessarily applicable, the manufacturing approaches typical of military hardware such as a cruise missile are much more relevant than those used on large complex military spacecraft.

Based upon the SCOUT Kickoff teleconference with Major Shoemaker on 02 DEC 02, AeroAstro revisited the WoRM concept and selected its best elements to apply to the SCOUT Phase I approach. This report consolidates the results of that effort.

2.0 Mission Definition

The initial task was to examine a broad scope of missions in order to later establish a requirements set for SCOUT. From there, a single mission was selected to serve as an example of SCOUT's capabilities, and a detailed mission concept for it was developed.

Investigating these potential SCOUT missions included assessing what is realistically and technically achievable, and investigating the corresponding orbits, launch vehicles, and encounter scenarios. The various missions were listed out and prioritized based on DARPA's expressed interest in each and the feasibility of each. The mission considered to be maximally useful and achievable was to have SCOUT equipped with the ESCORT payload, monitoring a primary spacecraft in geosynchronous orbit. The ESCORT package contains the AeroAstro RF Probe instrument for monitoring RF emanations; a COTS visible camera and a COTS laser range finder.

The deliverable for this task, the SCOUT Mission Definition document is included in Appendix A "*Appendix A - SCOUT Mission Definition Final.doc*".

3.0 Technology Assessment

The Technology Assessment task included identifying the key technologies necessary to implement the SCOUT architecture, the existing and projected technologies available to satisfy these needs, and the state of development of the identified technologies. In cases where existing technologies were not expected to be sufficiently mature, AeroAstro assessed alternate or substitute technologies.

The deliverable, the Technology Utilization and Development Plan is provided in Appendix B (*Appendix B - SCOUT Tech Plan-Final.doc*).

4.0 Launch Vehicle Compatibility Analysis

The SCOUT effort proceeded with an investigation of the RASCAL launch vehicle and the development of a set of requirements. Other available launch vehicles were surveyed and requirements developed. Finally, AeroAstro developed a set of launch compatibility requirements for SCOUT that enveloped both RASCAL and a wide range of alternative Launch Vehicles.

The deliverable, the Technology Utilization and Development Plan is provided in Appendix C (*Appendix C - SCOUT LV-Final.doc*).

5.0 Requirements Definition

Based on the Launch Vehicle Compatibility Document, the Concept of Operations, the SCOUT modularity boundary conditions and the Escort payload and mission, AeroAstro developed the Escort-specific mission requirements for SCOUT. The detailed requirements were defined for each subsystem.

The resulting deliverable, the Mission Requirements Document, is provided in Appendix D (*Appendix D - SCOUT requirement Matrix-Final.xls*).

6.0 SCOUT Conceptual Design

The requirements provided the basis for the SCOUT Conceptual Design. AeroAstro developed a system architecture that supported the Launch Vehicle and mission requirements, focusing on modular approaches to the spacecraft bus that provided for evolutionary development consistent with the system architecture. AeroAstro identified modules common to multiple missions and their requirements and identified mission-specific modules and their requirements consistent with the SCOUT architecture. Ongoing technology developments were identified that can be leveraged to enable or improve the capabilities of SCOUT.

The deliverable, the Conceptual Design Review of the mission and spacecraft, is provided in a collection of annotated viewgraph presentations in Appendix E;

- 00 Agenda (Annotated).ppt
- 01 Introduction (Annotated).ppt
- 02 Payload (Annotated).ppt
- 03 Requirements (Annotated).ppt
- 04 System Overview (Annotated).ppt
- 05 Mech-Therm (Annotated).ppt
- 06 C&DH (Annotated).ppt
- 07 Software (Annotated).ppt
- 08 Power (Annotated).ppt
- 09 ADCS (Annotated).ppt
- 10 Propulsion (Annotated).ppt
- 11 RF Comm (Annotated).ppt
- 12 Mission Ops (Annotated).ppt
- 13 I&T (Annotated).ppt

7.0 Conclusion

SCOUT addresses a critical need for rapidly configurable and deployable spacecraft. It's flexible and modular architecture will offer a framework for many applications in and outside of DARPA. Successful development of SCOUT will demonstrate a host of technologies, components and system architectures that could benefit the entire microsat industry. While additional work is needed, the conceptual design of SCOUT appears to meet the immediate objectives of the effort.

AEROASTRO

Appendix A – SCOUT Mission Definition

Submitted on: June 24, 2003

Submitted to: Major James Shoemaker

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1.0 SCOUT Mission Definition

Per section 2.0 of the Work Breakdown Structure, an investigation of potential SCOUT missions was conducted to assess what is realistically and technically achievable using the proposed SCOUT architecture. A prioritized list of missions, based upon interest and feasibility, was developed. After soliciting input from the DARPA TPOC, a mission down-select was conducted and three baselines were identified to be carried forward into conceptual design.

2.0 Identify potential top-level mission types

To define and envelope the scope of potential applications and missions for SCOUT in either a single or multi-element deployment, a comprehensive investigation of candidate functions and opportunities was conducted. Based upon the results of this activity, candidate missions were divided into top-level categories by the function or utility to be provided and the nature of the intended role of the SCOUT vehicle(s). Based upon this scheme, the following six mission types were delineated:

- Satellite inspection and logistics services
- Near-field situational awareness for US space assets
- Extend existing space and terrestrial-based capabilities
- Asset protection
- Surveillance, reconnaissance, and tactical operations
- Rapid access and flight qualification of new technologies

The subsequent sections will address the specific activities, applications, and components associated with each candidate mission genre.

2.1. Satellite Inspection & Logistics Services

A SCOUT or multiple SCOUT vehicles could be utilized for inspection services of a friendly target satellite to aid in deployments, monitor performance, and/or investigate anomalies.

2.1.1. Facilitate In-Orbit Test

With a suitable payload sensor-suite that could include an AeroAstro RF Probe, a SCOUT could enable analysis of antenna gain patterns including primary and side-lobes. In addition, as a maneuverable point-source, SCOUT could also assist in multispectral and hyper-spectral imager calibration.

2.1.2. Anomaly Investigation and Resolution

Employing a diagnostic payload that might include such components as the AeroAstro RF Probe or imaging devices (visible or infrared), SCOUT could assist in the investigation and resolution of on-orbit anomalies. In addition, for deployments (solar arrays, antennas, etc.) that are inhibited or not fully actuated, SCOUT may be utilized to mechanically or thermally assist.

2.1.3. Supply Surrogate or Complementary Functional Element or Capability

In situations where, for example, an uplink receiver of a space asset has been damaged or is otherwise unable to maintain sufficient link with the ground, a SCOUT vehicle may be used as a surrogate for communications of low bandwidth TT&C. Additionally, a SCOUT equipped with sensors could be used to complement the sensors of the primary space asset, including supplying alternate or additional attitude determination measurements to the primary satellite in situations of eclipse or sensor blockage resulting from sun, moon, or earth interference.

2.1.4. Deployable mechanism for sensor or payload shielding

For spacecraft that experience unanticipated solar illumination, either thermal or optical, a SCOUT vehicle could be dispatched to deploy a lightshade or thermal shield. This could provide preferential shading of key bus components, including thermally-sensitive battery systems, or mitigate albedo glint effects or sun intrusion into payload optics or attitude sensors.

2.2. Near-field Situational Awareness

A SCOUT vehicle could be used to monitor the natural or induced local space environment around a target satellite. In benign conditions, a SCOUT payload might include sensors to measure electromagnetic radiation and ESD, or capability to detect contamination generated by out-gassing or propellant leaks. These functions could be extended to include surveillance or sentinel functions, in which a SCOUT could be tasked with monitoring the vicinity for signatures of hostile activities.

2.3. Extend Existing Space and Terrestrial-based Capabilities

A SCOUT or constellation of SCOUT spacecraft could be used to extend US or Allied capabilities by leveraging existing space or terrestrial infrastructure.

2.3.1. Air Traffic Control

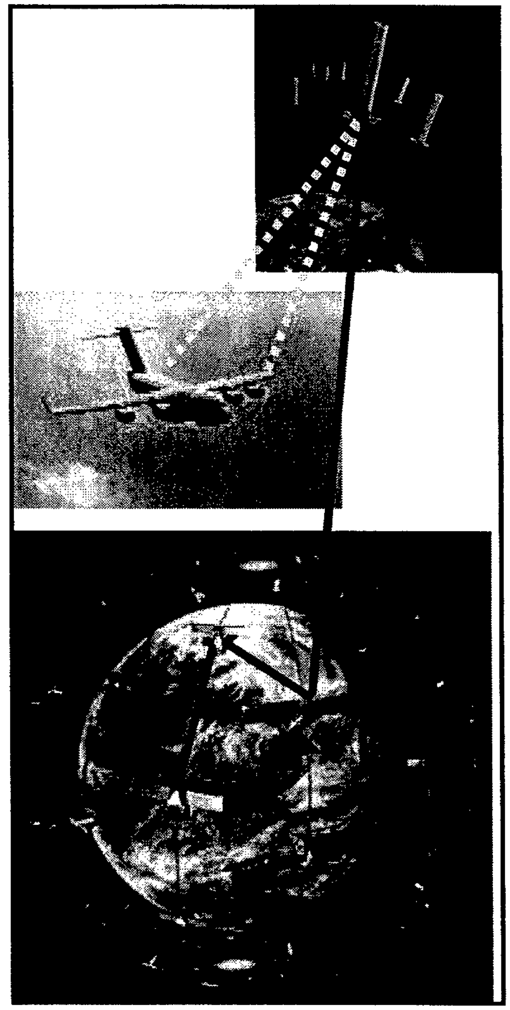
SCOUT satellites could augment current radar-based aircraft tracking systems by monitoring regional traffic and relaying gathered information through global satellite networks.

2.3.2. GEO LaserComm MicroSat

Using small, ultra low power laser transponders on SCOUT satellites in conjunction with a centralized, high power laser transponder for the return ground link, communication with satellites in LEO, MEO or GEO could be enabled.

2.3.3. *Space Surveillance*

The coverage of the NORAD space tracking network could be improved if it were aided by SCOUT satellites able to track Near Earth Objects (NEO) and small orbital debris in orbits of interest.



2.3.4. *Tactical GPS Microsatellite*

A SCOUT or multiple SCOUT vehicles could be utilized in a tactical role to augment or spoof the Global Position System (GPS) network signal. Equipped with a small-scale GPS-compatible transmitter or repeater, a SCOUT could serve as an extra node to reduce the error associated with network access, known as Geometric Dilution of Precision (GDOP), or serve to boost overall broadcast signal power, thereby enhancing the resolution of Position, Velocity, and Time solutions in support of regional, short-duration operations. From an operational perspective, access to the GPS network for navigation and orbit determination purposes, could be further extended by providing re-broadcast of system signals to space assets located at otherwise un-served orbits, such as GEO. Conversely, for strategic purposes, a SCOUT could be dispatched to degrade or spoof GPS available to hostile or foreign targets by sending out a false signals that US and allied forces could ignore by using P(Y) code messages. In the advent of third-generation GPS satellites, SCOUT could provide a means of ensuring access to high accuracy navigation and position data should spectrum become constrained or precluded by the further development of the GLONASS or GALILEO.

2.3.5. *Low-cost LEO Delivery from Auxiliary Payload to GTO via Deployable Aerobrake*

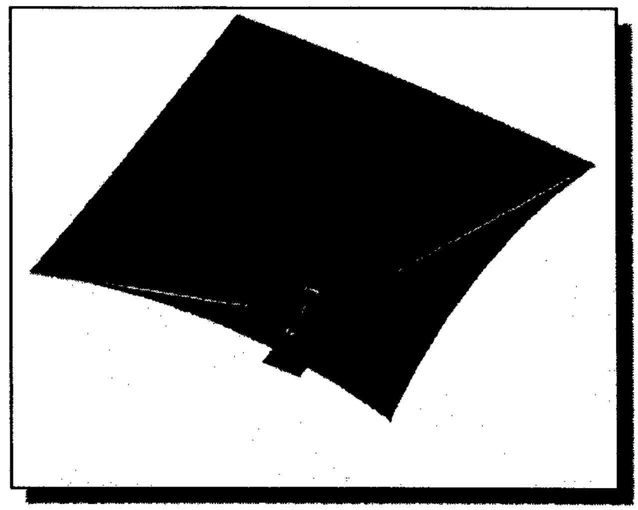
AeroAstro is currently developing significant elements of the first Small Payload ORbit Transfer (SPORT) vehicle under Phase II SBIRs. A SCOUT-SPORT would provide a low-cost launch capability for small spacecraft by exploiting the excess payload capability of large launch vehicles. This system will transfer a microsatellite payload from Geosynchronous Transfer Orbit to Low Earth Orbit, after launch in the piggyback payload slot of a major large launch vehicle. After release into GTO, the SCOUT-SPORT would deploy a large aerodynamic decelerator and begin aerobraking at each perigee pass to reduce the orbit energy and thereby lower the apogee. After completing a series of aerobraking passes, the SCOUT-SPORT propulsion system would be used to circularize the orbit for payload release.

2.4. Asset Protection

SCOUT could be utilized to actively defend an asset against on-orbit threats or attacks by hostile microsatellites. Using an RF or optical system in defense, SCOUT could jam communications, eavesdrop or corrupt TT&C signals of potential threats, or potentially interfere with the GN&C

capabilities of identified targets. If warranted, SCOUT provides an option users for commanded, "sacrificial" intercept of incoming threats. Alternatives, however, also include use of deployable systems, including thin-film shields, nets, or tethered restraints to block, shield, deflect, or otherwise interfere with an aggressor vehicle.

2.4.1. Orbital Debris Deorbiting System



Hostile attacks on space assets may use kinetic or fragmentation projectiles that could potentially result in large strata of orbital debris, threatening US and allied assets in the same and adjacent planes. Replacement satellites can not be safely deployed until the debris is removed. SCOUT-based spacecraft may be capable of mitigating this threat, including a nanosat Orbit Debris Deorbiter, in which SCOUT vehicles intercept and capture small pieces of debris, potentially using an aerobrake to aid collection and deorbit. These SCOUT vehicles could be re-tasked from space surveillance and debris tracking assets, or deployed via a RASCAL launch as needed.

2.5. Surveillance, Reconnaissance, & Tactical Operations

As the first line of space-based counter-intelligence, SCOUT could employ an optical or RF Probe to gather intelligence data for foreign or identified space assets, including evaluation of vehicle data links, payload, capability, and host ground stations. In addition, SCOUT could be used to disrupt the payload or sensors of a foreign asset using an onboard laser or RF source to saturate sensors or command receivers. In general, these activities could be carried out in a way that is largely undetectable on Earth and, as such, disruptive behavior could be temporary in nature and responsive to the tactical needs of the conflict or operation.

2.5.1. Hostile Intent Microsatellite (HIM)

Operators on ground could utilize a SCOUT-HIM to engage an adversarial space asset to temporarily or permanently disable it. A SCOUT-HIM would allow quick and covert removal of a threatening capability from a tactical engagement. A SCOUT-HIM could be stealthy so that opposing ground operators could not detect it and/or react with countermeasures. In this regard, a properly shaped external sheath of radar absorbent material (for example, EccoSorb® with a special formulation of Martin Black paint) would render a SCOUT-HIM virtually invisible from RF, optical, or infrared detection. Possible HIM methods could include forcing an orbit change or degradation or instilling a body spin-up or disturbance torque that could interfere with operations, attitude, or image quality. This could also include jamming, spoofing, or blocking an adversarial spacecraft's bus subsystems at close range, including magnetometers, GPS receivers, RF transponders, imagers, or imaging attitude sensors.

2.6. Rapid Access to Space and Flight Qualification of New Technologies

The modular and configurable SCOUT architecture, in conjunction with the highly responsive deployment capability of the RASCAL system, would also enable rapid access to space for flight qualification and validation of new technologies for demonstration or test purposes.

3.0 Review and prioritize mission profiles

Based upon these mission options, the following table was created that correlates the principal goals and intent associated with each to top-level functional subsystem requirements.

MISSION OPTIONS	SUBSYSTEM							
	Propulsion		ADCS		Power		Communications	
	Co-Planar Transfer	Proximity	Coarse	Fine	Short Term	Long Term	High BW	Low BW
Satellite Inspection & Logistics Services	X	X		X		X	X	
Near-field Situational Awareness	X		X			X		X
Extend Existing Space & Terrestrial-based Capabilities	X		X			X		X
Asset Protection	X	X	X		X		X	
Surveillance, Reconnaissance & Tactical Operations	X			X		X	X	
Rapid Access & Flight Qualification	X		X		X			X

4.0 Mission Baseline Down-select

Based upon the review of these mission types and the direction of the sponsor, the following three missions have been selected for further definition and study.

- Satellite inspection and logistics services
- Tactical GPS microsatellite
- Enable rapid access and flight qualification of new technologies

A detailed design study commensurate with the phase of the program will be presented in the Final Report and will be based on satisfying the following mission:

- Satellite inspection and logistics services

This design will feature a payload including an RF Probe sensor and a visible light imager.

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SCOUT Launch Vehicle Study

Submitted on: February 26, 2003

Submitted to: Major James Shoemaker

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SCOUT LAUNCH VEHICLE STUDY

1. DOCUMENT SCOPE

As part of the work being performed in support of the DARPA Phase I SBIR contract entitled Smart Escort Microsatellite, AeroAstro is developing a modular architecture for small satellites, as well as a conceptual design for a small satellite that could be built using this modular architecture. The modular architecture, as well as spacecraft which are built using it, are referred to as S^3COUT (Small Smart SpaceCraft for Observation and Utility Tasks).

The modular architecture of SCOUT is intended to enable a mass-producible small satellite which is complimentary to the "rapid, responsive and reliable" RASCAL launch vehicle concept. By using the SCOUT architecture, tactical commanders in the field will no longer need to rely on a distant and sometimes oversubscribed homeland-based space capability for the launch and operation of their satellites. Instead, field commanders will be able to field-assemble and configure a customized small satellite from pre-built, pre-qualified modules, integrate it to RASCAL, launch it, and operate it in just a few hours

This document reports on work performed in support of section 5 of the SBIR contract Statement of Work (SOW) as shown below.

5.0 Launch Vehicle Compatibility Analysis

5.1 Identify and analyze RASCAL launch imposed requirements and constraints

- Survey the RASCAL launch vehicle and develop a set of requirements based on that vehicle
- Develop a set of launch compatibility requirements for the overall mission

5.2 Assess alternate deployment options and identify corresponding launch imposed requirements

- Survey other available launch vehicles
- Develop set of requirements based on those vehicles

2. Introduction

Small satellites (10 kilograms to 200 kilograms) are increasingly being utilized in applications such as distributed constellations, satellite inspection and engagement, eavesdropping, and reconnaissance. The ability to get those satellites into operation quickly and efficiently is necessary to maintain information superiority and space control.

Secondary launches are a quick, frequent, low-cost, reliable solution for small satellites. Worldwide, most small spacecraft are launched as secondary, piggyback payloads, aboard larger, more efficient rockets. However, piggyback accommodations in the US are rare, done only on a case by case basis, and far from low cost.

This work will assess current and future US secondary launch capabilities, catalog all requirements, interfaces, and procedures, and create a tool for mission planners to quickly design small satellite missions. We will then develop a Universal Small Payload Interface (USPI), a template for launching small satellites which developers and programs can follow regardless of which launch vehicle is ultimately used.

This tool will deliver rapid access to space. The mission planner can start a program with a template which he/she knows will be applicable to multiple US secondary accommodations. When nearing launch, the planner can select the most convenient launch option, since there will be several launch vehicles usable by the mission. Access to space for small satellites will become rapid, low-cost, and strategically effective.

3. Background and Objectives

The Opportunity

One of the biggest sources of cost and schedule burden lies with the launch phase of the mission. Not only is this phase of the mission costly and often fraught with delays, but the constraints imposed on mission planners by the launch vehicle and the process associated with it have repercussions throughout the program. Small satellites are extremely limited in the number of launch options they have. These options fall into three categories:

- *Space Shuttle* - with a high cost due to safety constraints, a short lifetime (low altitude) orbit, and an inclination not useful for many missions;
- *Dedicated Launches* - with a very high cost and a long launch backlog, often requiring shared launches with custom dual payload adapters;
- *Secondary Launches* - usually "custom" in nature, subject to launch vehicle-specific constraints, and historically unavailable on US launchers, although that situation is improving.

None of the existing domestic launch options are designed for quick, small satellite missions. Small satellites may wait years for a launch opportunity. And those opportunities may cost up to ten times more than the payload The requirements for tactical responses to threats cannot be met under these conditions.

Secondary launches offer an avenue for dramatic improvement to this situation. Worldwide, most small spacecraft are launched as secondary, piggyback payloads, aboard larger, more efficient rockets. However, piggyback accommodations in the US are rare, done only on a case by case basis, and far from low cost. By contrast, the French Ariane V launchers have developed a standard secondary configuration which has enabled a significant number of easily accessible, low-cost launches over the past decade.

A combination of greater availability of standard secondary launch accommodations on US launchers and a degree of commonality among these options will create a revolutionary change in the way small spacecraft missions are developed and launched. Missions can be quickly designed towards a common interface standard,. The need to contract and customize a secondary launch on a specific vehicle at the very beginning of the program will be eliminated, decreasing cost and mission cycle time. Also, the launch will not be dependent on a specific launch of a specific launch vehicle; when the spacecraft is ready, the next launch available will be used, bringing 'launch on-demand' closer to reality.

4. RASCAL Launch Requirements

Table 1 below provides the payload requirements to meet the RASCAL launch vehicle limits.

Property	Limit
Mass	100 kg or less
Static envelope	1.2m diameter x 3.0 m length
Mass moments of inertia	Limited only by mass and static envelope
Center of Gravity (c.g.)	
Axial -distance from c.g. to interface plane	1.5 m or less
Lateral - distance from c.g. to the vector that is normal to the interface plane and that passes through the center of that interface	0.03 m or less
Fundamental frequency when rigidly mounted at LV interface	
Axial	50 Hz or greater
Lateral	40 Hz or greater
Torsional	50 Hz or greater

Table 1 RASCAL Launch Requirements

Explanation:

- Mass—The mass shown is the limit for the total payload, including any needed upper stage.
- Static envelope—This is the physical space in which the payload must stay in the static, unloaded condition. The LV shall provide a dynamic envelope large enough to ensure a payload with the static envelope and fundamental frequencies given above will not make physical contact during launch with any part of the launch vehicle. The dynamic envelope shouldaccommodate rigid-body deflections of the payload resulting from deformation of the mounting structure combined with the elastic deformation of the payload under maximum expected launch loads.
- Mass moments of inertia—self explanatory.
- Center of gravity—These limits are arbitrary and are suggested as a starting point. If these values drive system complexity or cost, it is acceptable to derive reasonable alternatives that can be specified to payload organizations.
- Fundamental frequency—These values are also arbitrary and intended as reasonable lower limits for payloads up to 100 kg. They are suggested as a starting point for designing an LV control system and meeting the environment objectives. Reasonable alternatives acceptable.

5. US Piggyback Capability Assessment

This section provides historical data on piggyback launches and current and future piggyback accommodations. The assessment used the Ariane launcher and its ASAP 5 adapter as a benchmark to highlight the opportunities available for small satellites with standard piggyback payload adapters and proactive launch policies.

5.1 Universal Small Payload Interface

The Universal Small Payload Interface is a design template and an interface standard. The design template provides small and micro-satellite designers the basic design requirements to launch piggyback on a USPI-compatible vehicle[1]. The interface standard provides the designers a set of standard mass-volume classes[2], a standard interface[3], and an accommodation platform. USPI's logistical concept is described in Figure 6 below.

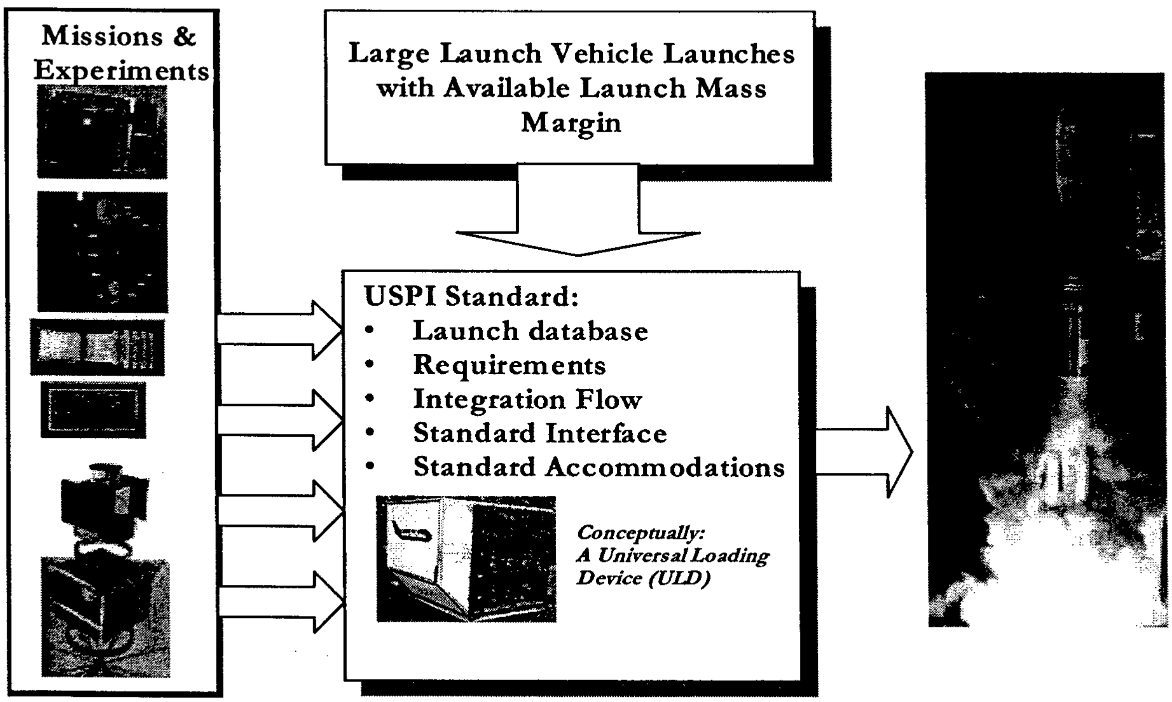

Figure 2: Logistical concept for USPI.
Essentially, we need a process to get missions and experiments on large LVs with capacity using a standard that provides the maximum possible number of missions.

A quick, cost-effective method to get high-risk technologies into space to test the concepts before they have to commit large funds to a full mission is needed.

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[1] Henceforth, this document will use the shorthand USPI standard to mean a design that satisfies the design, interface, and accommodations standards to launch piggyback on a launcher covered by USPI.
[2] The mass-volume class defines the actual physical volume the piggyback payload must fit in.
[3] The standard interface is the separation system and the bolt pattern required of it.

- Have a standard set of requirements to design to – the user knows that if their design fits the USPI requirements, they will fit on a large number of LVs.
- Have a standard accommodation template – allows the piggyback payload to fit in the widest possible variety of LVs.
- Allow LV contractors easy integration – as long as the piggyback payload complies with USPI, the contractors know they can easily integrate the payload onto their LV.
- Allow quick mission turnaround – it does not matter how long the piggyback payload takes to design, build and test. Once it is ready, it can go on the next most convenient mission because the USPI standard allows it to fit on any of the USPI LVs.
- Allows the LV to disengage from the piggyback payload schedule – the LV can take the piggyback payload or a standard dummy mass[4] instead. If the piggyback payload is late, the LV can launch with the dummy mass and not worry about the mass balance of the LV.

The launchers selected for consideration in the USPI standard are shown in Table 3 below.

Table 3: Launch Vehicles Considered in USPI

Ariane 4, 5	K 1	H II A
Atlas II, III, V	Kosmos	Sealaunch
Delta II, III, IV	Minotaur	STS
Eurockot	Pegasus	Taurus

5.2 USPI Requirements Document

The requirements document is a template for a USPI-compatible payload design. The USPI requirements document ensures that, if followed:

- The payload shall fit in the piggyback accommodations provided on any of the USPI-compatible LVs shown in Table 5 above.
- The payload shall comply with the worst-case environmental requirements for the USPI LVs. Therefore, the payload will have the maximum possible LV flexibility.

The requirements document provides the design discipline to make the tradeoffs required for mission success.

The most stringent launcher environmental requirements are needed to qualify a payload on any existing LV. The USPI requirements synthesize all the environments a payload is exposed to on any of the twenty US or European LVs considered. Table 5 is a list of all the launchers considered. Most LVs divide their launch and flight environments into similar categories. Table 4 is a list of the environmental categories used to describe the worst case-loads applied to a payload during launch.

[4] Please see the USPI requirements document for the requirement for the dummy mass. Essentially, the dummy mass is a mass and volume substitute for the actual payload.

Table 4: Environmental Categories Used to Compare LVs.

Acoustic	Electro Magnetic Interference (EMI)
Shock	Fairing Pressure
Low Frequency Vibration	Fairing Wall Temperature
Quasi-Static Load	Testing Factors

5.3 USPI Mass & Volume Classes

The ESPA is not considered in the USPI mass volume classes for a series of technical reasons. The general ESPA configuration is shown in Figure 6 below. The ESPA adapter sits atop the standard EELV PAF. The piggyback payloads are arranged radially from the ESPA adapter.

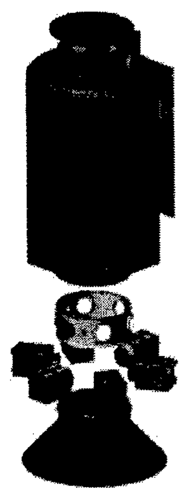


Figure 6: ESPA Piggyback Payload Adapter.
The six blue boxes are the piggyback payloads, attached to the cylindrical adapter. The payload is the large red structure.

The standard ESPA payload is 610 mm x 610 mm x 965 mm with 170 kg mass. The bolt pattern is non-standard, and there seems to be no standard separation system. As a payload accommodation, the ESPA provides a lot of mass and volume for the piggyback payload. But by maximizing the payload mass and volume available, the ESPA payload does not adhere to the ASAP standard.[5] Configured as such, they would also be less likely to fit in other LVs. Therefore, the ESPA configuration is not considered in the USPI mass volume classes.

Three mass and volume classes maintain some standardization without sacrificing too much flexibility. It is therefore possible to have mission flexibility at the expense of optimal use of available mass and volume on a given launch. The three classes are shown in Table 5 below.

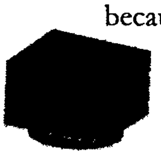
Table 5: USPI Mass and Volume Classes.

Class	Volume	Mass	Comments
Class I	400 mm X 400 mm X 250 mm	25 kg	Smallest class, still can fit ASAP 5 38.5 sep. system
Class II	440 mm X 440 mm X 500 mm	75 kg	Will fit on ASAP 5 sep system Also fit on small launchers

[5] The only other large launch vehicle piggyback payload standard.

Class III	600 mm X 600 mm X 710 mm	120 kg	ASAP 5 Micro piggyback payload standard.

5.3.1 Class I

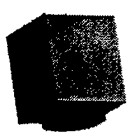


Mass-volume Class I is the smallest USPI standard. The general aspect is flat for this class because the 400 mm X 400 mm footprint is required as a minimum to accommodate the 348 mm ASAP 5 Micro baseline separation system. The bolt pattern is common with the other mass-volume classes. As configured, the Class I USPI payloads will fit as piggyback payloads on the following LVs without modification:

- Pegasus XL DPAF[6]
- Taurus DPAF
- Minotaur – using the OSSS[7] Multiple Payload Adapter
- Delta II Secondary Payload accommodations[8]
- ASAP 5 Standard – Ariane 5, Soyuz ST-Fregat, Eurockot, PSLV, K 1
- STS Hitchhiker
- Kosmos – using a DPAF-like structure in place of the "load bearing satellite"[9]
- Proton – in the non-standard accommodations for piggyback launches on Proton

In addition, the standard accommodation concept presented in section 3.5 can accommodate Class I payloads at a minimum. The Sealaunch vehicle has not shown interest in piggyback launches. Discounting that LV, Class I payloads can fit on any of the LVs considered either without modification on existing piggyback accommodations or on the standard piggyback payload.

5.3.2 Class II



This class has a 440 mm X 440 mm X 509 mm volume with an allowable maximum mass of 75 kg without the separation system. This class also accommodates the 348 mm ASAP 5 Micro baseline separation system and has the bolt pattern common with Class I and Class III. As configured, the Class II USPI payloads will fit as piggyback payloads on the following LVs without modification:

- 50 inch and 63 inch Taurus DPAFs
- Minotaur – atop the OSSS Multiple Payload Adapter
- ASAP 5 Standard – Ariane 5, Soyuz ST-Fregat, Eurockot, PSLV, K 1

[6] Dual Payload Attachment Fitting.
[7] One Stop Satellite Solutions, Ogden, UT 84408-1805.
[8] However, the separation system is not standard for Delta II.
[9] Cosmos 3M Launch Vehicle Users Guide, United Start Corporation, 190 Lime Quarry Road, Suite 106-J, Madison, AL 35758

- Kosmos – using a DPAF-like structure in place of the "load bearing satellite"[10]
- Proton – in the non-standard accommodations for piggyback launches

The standard accommodation concept presented in section 3.5 will also accommodate Class II payloads. Discounting the Sealaunch vehicle, Class II payloads can fit on the LVs above without modification on existing piggyback accommodations or on the standard piggyback payload accommodations described in Section 3.5. However, inserts will be required on the standard PAFs to raise the primary payloads to accommodate the taller Class II payloads on the standard accommodation. The inserts will be less than 550 mm, but the loss – or perception thereof – of height available to the primary

5.3.3 Class III

Mass volume Class III is the same size and mass as the ASAP 5 Micro class payload. Class III has 600 mm X 600 mm X 710 mm volume and 120 kg without the separation system. The bolt pattern and interface remain the same as the other two classes. As configured, Class III USPI payloads will fit as piggyback payloads on the following LVs without modification:

- 63 inch Taurus DPAFs
- Minotaur – using the OSC 63 inch DPAF
- ASAP 5 Standard – Ariane 5, Soyuz ST-Fregat, Eurockot, PSLV, K 1
- Kosmos – using a DPAF-like structure in place of the "load bearing satellite"[11]
- Proton – in the non-standard accommodations for piggyback launches

The following tables shows the worst-case requirements for each launcher considered:

[10] Cosmos 3M Launch Vehicle Users Guide, United Start Corporation, 190 Lime Quarry Road, Suite 106-J, Madison, AL 35758

[11] Cosmos 3M Launch Vehicle Users Guide, United Start Corporation, 190 Lime Quarry Road, Suite 106-J, Madison, AL 35758.

Full Launcher Name	Longitudinal Static Load (G)	Positive Longitudinal Dynamic Load (G)	Negative Longitudinal Dynamic Load (G)	Combined Max. Longitudinal Load (G)	Approx. Longitudinal Frequency (Hz)	Lateral Static Load (G)	Max Lateral Dynamic Load (G)	Combined Max. Lateral Load (G)	Approx. Lateral Frequency (Hz)	Combined Min Lateral Load (G)
Worst Case Levels	**13**	**10**	**-2**	**13**	**35**	**1**	**2.5**	**2.5**	**20**	**-8.25**
ASAP				5.5				6		-6
ARIANE 5 SHORT FAIRING	4.25	1.75	-1.75	6		0.2	1.5	1.7	10	-2
ARIANE 5 LONG FAIRING	4.25	1.75	-1.75	6		0.2	1.5	1.7	10	-2
ARIANE 5 SHORT SPELTRA	4.25	1.75	-1.75	6		0.2	1.5	1.7	10	-2
ARIANE 5 LONG SPELTRA	4.25	1.75	-1.75	6		0.2	1.5	1.7	10	-2
ARIANE 5 SYLDA 5	4.25	1.75	-1.75	6		0.2	1.5	1.7	10	-2
ATLAS II A LPF	5.5	0.5	-0.5	6		0.4	1.2	1.6		-0.8
ATLAS II A EPF	5.5	0.5	-0.5	6		0.4	1.2	1.6		-0.8
ATLAS II AS LPF	5	0.5	-0.5	5.5		0.4	1.6	2		-1.2
ATLAS II AS EPF	5	0.5	-0.5	5.5		0.4	1.6	2		-1.2
ATLAS III A LPF	5.5	0.5	-0.5	6		0.4	1.6	2		-1.2
ATLAS III A EPF	5.5	0.5	-0.5	6		0.4	1.6	2		-1.2
ATLAS III B LPF	5.5	0.5	-0.5	6		0.4	1.6	2		-1.2
ATLAS III B EPF	5.5	0.5	-0.5	6		0.4	1.6	2		-1.2
ATLAS V 400 LPF	5.5	0.5	-0.5	6		0.4	1.6	2		-1.2
ATLAS V 400	5.5	0.5	-0.5	6		0.4	1.6	2		-1.2

Full Launcher Name	Longitudinal Static Load (G)	Positive Longitudinal Dynamic Load (G)	Negative Longitudinal Dynamic Load (G)	Combined Max. Longitudinal Load (G)	Approx. Longitudinal Frequency (Hz)	Lateral Static Load (G)	Max Lateral Dynamic Load (G)	Combined Max. Lateral Load (G)	Approx. Lateral Frequency (Hz)	Combined Min Lateral Load (G)
EPF										
ATLAS V 500 SHORT PLF	5.5	0.5	-0.5	6		0.4	1.6	2		-1.2
ATLAS V 530 SHORT PLF	5.5	0.5	-0.5	6		0.4	1.6	2		-1.2
ATLAS V 550 SHORT PLF	5.5	0.5	-0.5	6		0.4	1.6	2		-1.2
DELTA II 7320 2.9 M FAIRING	6.2	0.6	-0.6	6.8	17-18	0	2	2		-2
DELTA II 7320 3 M FAIRING	6.2	0.6	-0.6	6.8	17-18	0	2	2		-2
DELTA II 7325 2.9 M FAIRING	6.2	0.6	-0.6	6.8	17-18	0	2.5	2.5		-2.5
DELTA II 7325 3 M FAIRING	6.2	0.6	-0.6	6.8	17-18	0	2.5	2.5		-2.5
DELTA II 7920 2.9 M FAIRING	6.2	0.6	-0.6	6.8	17-18	0	2	2		-2
DELTA II 7920 3 M FAIRING	6.2	0.6	-0.6	6.8	17-18	0	2	2		-2
DELTA II 7925 2.9 M FAIRING	6.2	0.6	-0.6	6.8	17-18	0	2.5	2.5		-2.5
DELTA II 7925 3 M FAIRING	6.2	0.6	-0.6	6.8	17-18	0	2.5	2.5		-2.5
DELTA III	3.7	1.5	-1.5	5.2	16-23	0	2	2		-2
DELTA IV MEDIUM	6.5			6.5	35	0	2	2		-2
DELTA IV MEDIUM PLUS 4 M 2 STRAP-ON	6.5			6.5	35	0	2	2		-2

Full Launcher Name	Longitudinal Static Load (G)	Positive Longitudinal Dynamic Load (G)	Negative Longitudinal Dynamic Load (G)	Combined Max. Longitudinal Load (G)	Approx. Longitudinal Frequency (Hz)	Lateral Static Load (G)	Max Lateral Dynamic Load (G)	Combined Max. Lateral Load (G)	Approx. Lateral Frequency (Hz)	Combined Min Lateral Load (G)
DELTA IV MEDIUM PLUS 5 M 2 STRAP-ON	6.5			6.5	35	0	2	2		-2
DELTA IV MEDIUM PLUS 5 M 4 STRAP-ON	6.5			6.5	35	0	2	2		-2
DELTA IV HEAVY	6			6	35	0	2.5	2.5		-2.5
EUROCKOT	7.2	8.35		9.55				0		-8.25
H IIA 202 4S	4.00			0	30			0	10	1
H IIA 202 5S	4.00			0	30			0	10	1
H IIA 202 4/4D	4.00			0	30			0	10	1
H IIA 202 4/4/D-LC	4.00			0	30			0	10	1
HIIA 202 5/4D	4.00			0	30			0	10	1
HIIA 212 4S	4.00			0	30			0	10	1
HIIA 212 5S	4.00			0	30			0	10	1
HIIA 212 4/4D	4.00			0	30			0	10	1
HIIA 212 4/4/D-LC	4.00			0	30			0	10	1
HIIA 212 5/4D	4.00			0	30			0	10	1
K 1 STANDARD	5.6	1	-1	6.6		0	2	2		-2
K 1 EXTENDED	5.6	1	-1	6.6		0	2	2		-2
KOSMOS 3M	6.8	0.2	-0.2	7		0	1.4	1.4		-1.4
MINOTAUR	13.00			13				0		0
PEGASUS XL	13	2	-2	2.7		1	-3.7	-2.7		2
PROTON K	4.3			4.3		0	2.3	2.3		-2.3
PROTON K BLOCK DM	4.3			4.3		0	2.3	2.3		-2.3
PROTON M BREEZE M	4.3			4.3		0	1.35	1.35		-1.35
SEALANCH ZENIT 3SL	4.5	0	0	4.5		0	2	2		-2

Full Launcher Name	Longitudinal Static Load (G)	Positive Longitudinal Dynamic Load (G)	Negative Longitudinal Dynamic Load (G)	Combined Max. Longitudinal Load (G)	Approx. Longitudinal Frequency (Hz)	Lateral Static Load (G)	Max Lateral Dynamic Load (G)	Combined Max. Lateral Load (G)	Approx. Lateral Frequency (Hz)	Combined Min Lateral Load (G)
SOYUZ IKAR	4.80			4.8				0		0
SOYUZ FREGAT	4.80			4.8				0		0
SOYUZ ST	4.80			4.8				0		0
SOYUZ FG	4.80			4.8				0		0
STS				0				0		0
TAURUS 2 (COMM.) 1.6 M	11	0	0	11	35	0.5	0	0.5	20	2
TAURUS 2 (COMM.) 2.3 M	11	0	0	11	35	0.5	0	0.5	20	2
TITAN II 23G	10	10	0.5	10	18		2.5	2.5	6	2

6. Combined Launch Vehicle Requirements

Class I and Class II USPI are compatible with RASCAL LV requirements. The payload should be designed to these worst-case limits:

Property	Limit
Mass	75kg
Static Envelope	440 mm X 440 mm X 500 mm
Longitudinal Fundamental Frequency	35Hz
Lateral Fundamental Frequency	40Hz
Torsional Fundamental Frequency	50Hz
Maximum Longitudinal Load	13g
Maximum Lateral Load	2.5g

SCOUT Technology Development and Utilization Plan

Submitted on: March 18, 2003

Submitted to: Major James Shoemaker

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Date	Authorization	Revision/Change Description	Affected Pages
3/18/03	See Cover Sheet	Initial Release	All

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SCOUT TECHNOLOGY DEVELOPMENT AND UTILIZATION PLAN

1. DOCUMENT SCOPE

As part of the work being performed in support of the DARPA Phase I SBIR contract entitled Smart Escort Microsatellite, AeroAstro is developing a modular architecture for small satellites, as well as a conceptual design for a small satellite that could be built using this modular architecture. The modular architecture, as well as spacecraft which are built using it, are referred to as S^3COUT (Small Smart SpaceCraft for Observation and Utility Tasks).

The SCOUT modular architecture is intended to exhibit the following attributes:

1. **Small and Lightweight** to maximize utility within the limited performance envelope of modest launch vehicle capabilities;
2. **Universally Compatible** to enable using as many different launch vehicles as possible;
3. **Low Cost** to assure that assets can be readily expended as necessary to serve US national security needs effectively;
4. **Rapid Response** to allow US forces to swiftly react to any tactical or strategic situation with readily available assets;
5. **Flexible** to permit a modest complement of off-the-shelf elements to address a wide-ranging set of scenarios;
6. **Field Configurable** to enhance the utility and productivity of the system by allowing the widest possible application;
7. **Modular** to achieve the conflicting goals described above;
8. **Scaleable** to tailor the capability to a range of applications;
9. **Extensible** to allow the system to adapt to future needs.

These characteristics of SCOUT are intended to enable a mass-producible small satellite architecture which is complimentary to the "rapid, responsive and reliable" RASCAL launch vehicle concept. By using the SCOUT architecture, tactical commanders in the field will no longer need to rely on a distant and sometimes oversubscribed homeland-based space capability for the launch and operations of their satellites. Instead, field commanders will be able to field-assemble and configure a customized small satellite from pre-built, pre-qualified modules. By utilizing RASCAL for launch, together with innovative autonomy and ground station concepts for operations, field commanders will also be able to launch and control their satellites within a few hours.

This document reports on work performed in support of section 3 of the SBIR contract Statement of Work (SOW) as shown below.

3.0 Technology Assessment
- 3.1 Identify key technologies necessary to implement the SCOUT architecture
- 3.2 Identify existing and projected technologies to satisfy these needs
- 3.3 Assess availability and state of development of identified technologies
- 3.4 Identify ongoing technology development that can be leveraged
- 3.5 Assess alternate or substitute technologies where existing technologies are not expected to be sufficiently mature

3.6 Prepare a technology utilization and development plan

2. ARCHITECTURAL CONCEPTS

The following numbered technologies and features are considered important to implement the SCOUT architecture:

2.1 MECHANICAL

1. Stackable plug-and-play component-level module 'slices' for small spacecraft, as shown in Figure 1;
2. A robust modular assembly concept that enables qualified technicians to field-assemble small satellites as needed in a field-deployable facility;
3. A single uniform structural outer shell module housing that can also serve as the primary structure, as shown in Figure 2;
4. A single uniform standard for mechanical interfaces and electrical connectors between modules, as shown in Figure 2;

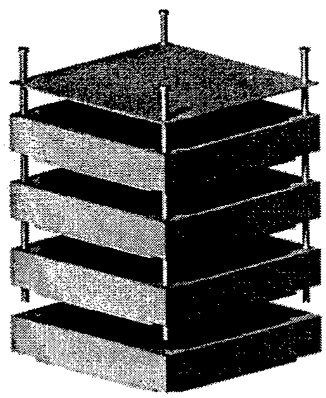

Figure 1: Stack of component-level modules

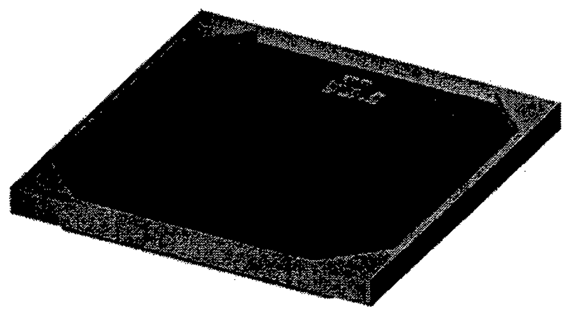

Figure 2: Single module

Figure 2 shows an avionics module consisting of a single printed circuit board (PCB). It should be noted that the modularization concept is not limited to PCBs. As conceived, a single module may also contain sensor, actuator, propulsion, deployable and other elements, as shown in Figure 4.

5. A single uniform keyed footprint and mating joint for the mechanical interfaces between modules, as shown in Figure 3;
6. A rectilinear outline for each module in order to promote efficient packaging of typical electronic component parts, as shown in Figure 3, non-rectilinear PCBs tend to have large amount of unused space;
7. The ability to grow the volume of a module slice by designing increased heights in unit step sizes of approximately 2 cm, as shown in Figure 4;
8. A connecting rod feature at each corner of each module, as shown in Figure 3;

9. The ability of four threaded rods to hold the central tower together, as shown in Figure 1;
10. The ability of the central tower structure composed of the structural outer shells of the modules to carry the launch loads;
11. The ability for larger modules or modules with attachments to extend outside the central tower, as shown in Figure 5;
12. A thermally conductive path along the structural outer shell to isothermalize the bus;
13. The ability to add external structural stiffeners to meet more demanding launch vehicle load requirements without compromising the mass budget for launch vehicles with less stringent load requirements;

2.2 Electrical

14. A uniform pass-through connector in a specified location; preference is to use a single electrical interface connector, as shown in Figure 3;
15. The ability for the stacked modules to form an RF-tight box around the internal components of each module, as notionally shown in Figure 1 and Figure 2 (where the metal plate at the bottom of each module is shown protruding downwards);
16. An electromagnetic interference (EMI) labyrinth seal that is incorporated with the structural outer shell of each module, as shown in Figure 3;
17. A method for providing all data needed to verify proper operation of any module in the field through the pass-through connector;

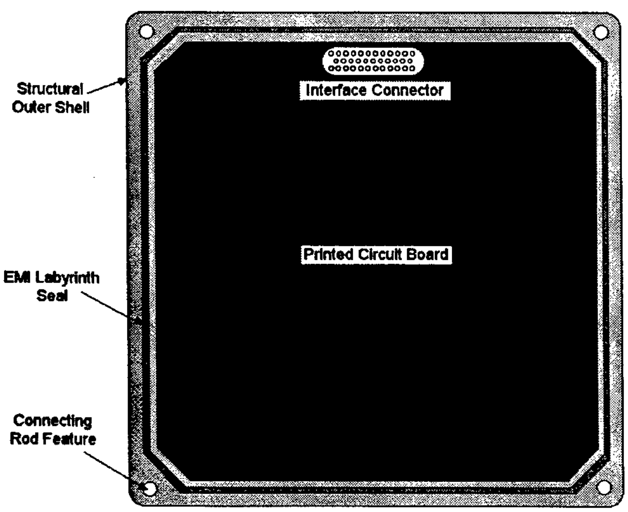

Figure 3: Single module features

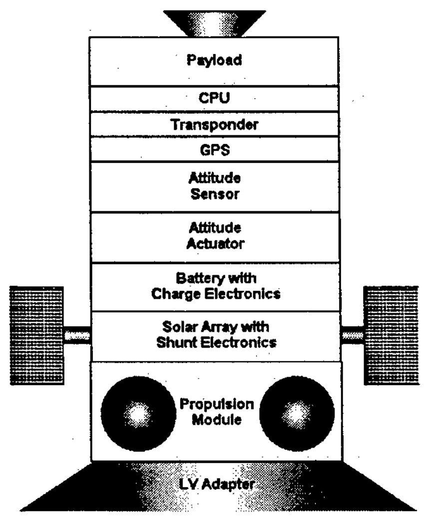

Figure 4: Stack of different modules

18. A data and power bus to join all the modules in the stack together electrically (not all modules will necessarily require interfaces to all elements of the electrical bus, as shown in Figure 6);
    - Power - Most likely 28 ± 6 VDC to keep currents low and increase commonality with conventional spacecraft electronics (TBR);
    - Low Speed Data - Carries command, power enable, current and temperature telemetry, and low-speed data;

- High Speed Data - This is an optional data bus interface which many modules will not use, reserved for high speed data such as imagery, attitude and orbit data, etc.;
- It is possible that some or all of these buses will be redundant for reliability;

19. A method for loading readily available software drivers while on the ground or in space;
20. A method for automatically configuring the attitude determination and control software to work for any combination of stacked modules;
21. A software Built-In-Test (BIT) capability for each module independently as well as the satellite as a whole, for use on the ground and in space;
22. A method for making the high-level software platform-independent, to facilitate future upgrades of computing hardware;

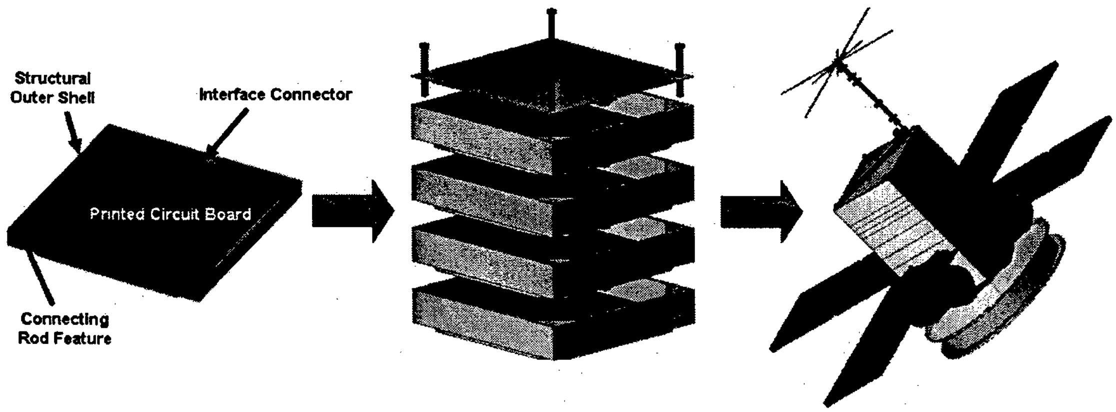

Figure 5: High level diagram of SCOUT modular architecture

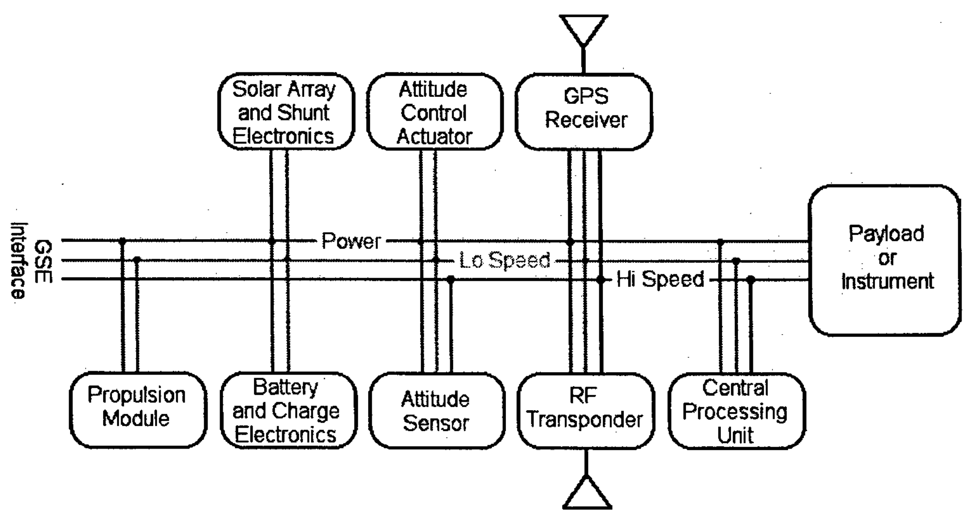

Figure 6: Electrical Block Diagram

2.3 MODULES

23. A computer module (nominally this module would include a Central Processing Unit (CPU) chip, but for some simple missions a hardwired Field Programmable Gate Array (FPGA) logic board may be sufficient);
24. An Electrically Erasable Programmable Read Only Memory (EEPROM) capability on the computer module to allow upload of code during field integration and possibly during flight;
25. A solid-state data storage module for non-volatile storage of large data volumes;
26. An RF communications modules (nominally this would be an X- or S-Band transponder for communicating directly with ground stations, but for increased link availability, communications links to space-based networks such as INMARSAT, Globalstar, Iridium, or TDRS could be used);
27. A conformal or self-contained antenna for the communications module (this could be a conformal antenna such as a belly-band, a patch or a microstrip, or deployable such as a unipole or dipole);
28. The ability to add encryption and decryption to the communications module;
29. A GPS receiver module;
30. An attitude determination module (possibly containing four miniature star trackers with different boresights, three orthogonal MEMS gyrochips, a three-axis magnetometer, and orthogonal sun sensors, as shown in Figure 7);
31. A three-axis magnetic actuator module (featuring magnetic torque rods on the sides and magnetic torque coils printed on one or more stacked PCBs, as shown in Figure 8);
32. A three-axis micro-wheel module for performing attitude actuation, momentum-dumping and power storage, as shown in Figure 9;

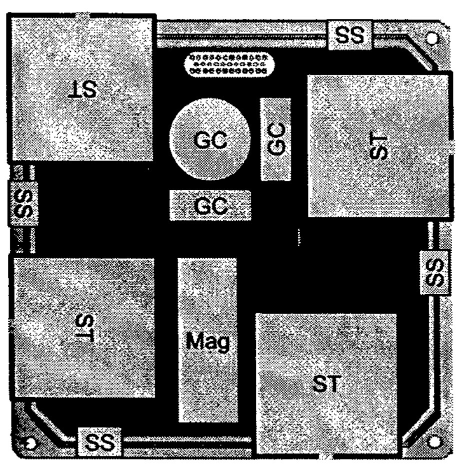

Figure 7: Attitude Determination Module

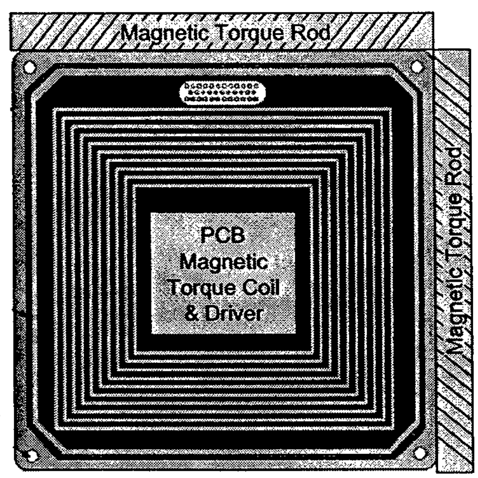

Figure 8: Magnetic Actuation Module

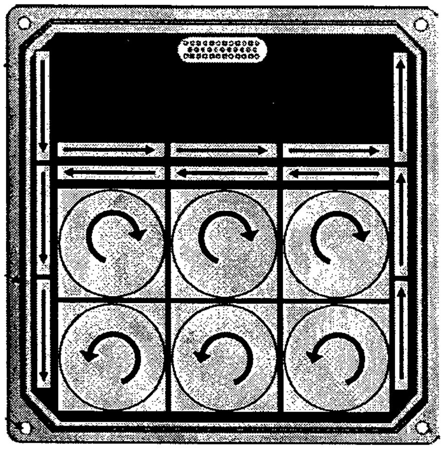

Figure 9: Micro-Wheel Module

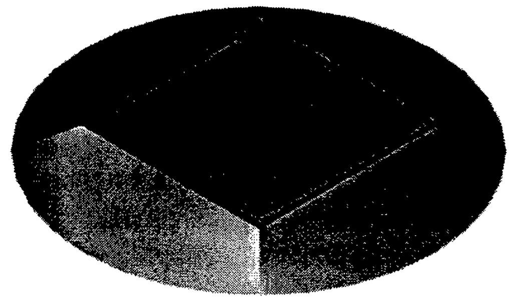

Figure 10: Payload Attach Fitting Module

33. A divert propulsion module with one divert thruster, four attitude control thrusters in a double-V configuration, and tankage, as shown in Figure 5;
34. A maneuvering propulsion module with at least six maneuvering thrusters and tankage, with one of these modules at each end of the stack, but flipped upside-down relative to each other (or otherwise correctly configured), then uncoupled 6-DOF translation and rotation is possible;
35. Toroidal, conformal, or otherwise not rounded propellant tanks for increased propulsion mass efficiency;

Increased performance from propulsion modules could be achieved by allowing tanks or thruster arms to extend above or below the height of their module slice. However, to facilitate modularity no part of the propulsion modules should extend above or below the module height. This is a case where reduced performance should be accepted to facilitate modularity.

36. A chemical battery module, including both primary batteries for short duration missions and secondary (rechargeable) batteries with a charger for longer duration missions (Lithium-Ion cells are a likely candidate);
37. A deployable solar array module with associated power electronics;

The solar array panels should be deployable since body-mounted solar array panels would be difficult to implement using the stacked module concept. High efficiency solar cell technologies should be incorporated for increased power. A tracking array module may be required to get the maximum power utilization from the small deployed arrays. Peak Power Tracker topologies may also be incorporated to provide the maximum power efficiencies from the expected small solar cell areas.

38. A different PAF (Payload Attach Fitting) Interface Module (PIM) for each type of launch vehicle, as shown in Figure 10 (Note that PIMs will probably be truss-work assemblies, unlike the example shown in Figure 10. Numerous PIMs are required to ensure that SCOUT is compatible with as many different launch vehicles as possible. The PIM will

interface with the standard separation switches provided at the LV end and will provide a fail-safe separation indication to SCOUT to initiate deployments);

39. A separation module to allow jettisoning the PIM to maximize use of available propellant;
40. A Master Universal Ground Support Equipment (MUGSE) to enable field-assembly, configuration and limited functional testing, as shown in Figure 11 and Figure 12;

The MUGSE will consist of a laptop computer loaded with the latest software, a base-plate upon which the stack of SCOUT module slices is assembled, and a cable between the laptop and baseplate. MUGSE will have the ability to test out individual modules or a fully integrated SCOUT satellite. The MUGSE base plate will include a standard mechanical footprint with a standard electrical interface connector, as shown in Figure 12. When a module is mounted on MUGSE, it can be powered, commanded, functionally tested, and fully characterized. Additional modules can be added one at a time until an entire satellite is built up and tested in the field. MUGSE will also be used to upload the latest generation of SCOUT software; each version will include data necessary to command every possible module variant in the inventory.

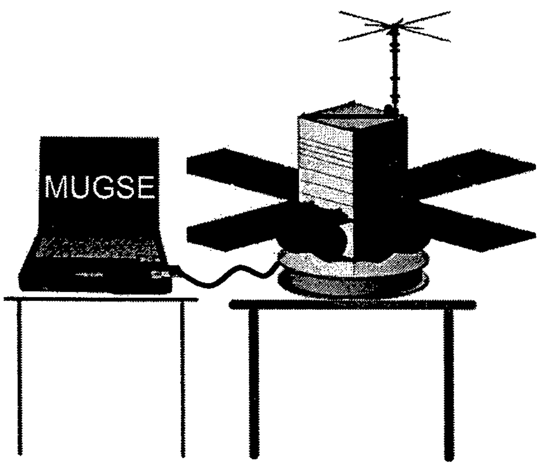

Figure 11: MUGSE Next to SCOUT

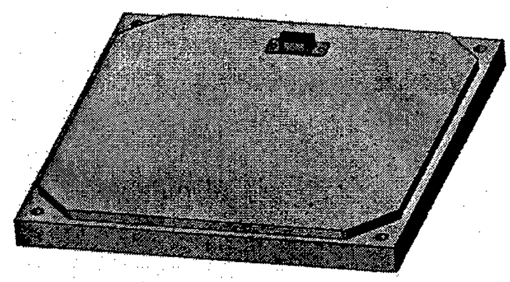

Figure 12: MUGSE Base Plate

41. An efficient production system for low-volume mass-production and qualification of modules;

3. Technology Candidates

Many of the technologies identified in this assessment, particularly mechanical and software technologies, do not exist and are not projected to exist unless AeroAstro is funded to continue development of SCOUT. Nonetheless, some existing and projected technologies may be candidates for incorporation into SCOUT. Some of these identified technologies are described below. The availability and state of development of these identified technologies is also assessed.

3.1 Electrical Bus Technologies

The SCOUT electrical bus architecture consists of two different data buses, one that is suitable for low data rate applications such as commanding and telemetry and one that is suitable for high data rate applications such as imaging. For purposes of this discussion, any bus with a data rate below 1 Mbps is considered to be low data rate, and any bus with a data rate above 1 Mbps is considered to be high data rate.

The following characteristics of different electrical buses were analyzed and traded to determine which were best suited for SCOUT:

- Backplane/Cable: Do all participants on the bus need to be in the same box (as in a VME architecture), or can they be distributed around the spacecraft (like a USB bus)?
- Central/Distributed: Is there one central governing device which controls who can transmit on the bus at any one given time, or all the devices on the bus participate in arbitration to decide who can transmit or have priority?
- Number of Wires: The number wires that are required for the bus
- Max Length: The maximum physical length of the bus
- Data Rate: The maximum data rate of the bus
- Payload Fraction: The maximum number of information bits that are transmitted, versus header or other bits (i.e., transmission efficiency or overhead)

3.1.1 Low Data Rate Electrical Bus Technologies

1. **I^{2}C**: An industry standard serial bus that can link multiple devices off the same 2 wire connections. It is a low data rate bus, ~400 kbps, and is typically used on a single card or over a backplane
   - Pros – Widely used with a lot of devices available to interface
   - Cons – Low data rate, short distance, poor noise immunity (single ended signals)

2. **Bluetooth**: A communications standard for short-distance wireless connections. It replaces the many proprietary cables that connect one device to another with one universal short-range radio link. The data is transmitted at a data rate of no greater than 723.2 kbps over a frequency of 2.4 GHz (unlicensed band). Three power classes are available for Bluetooth:
   - 100 mW, 20 dBm, 100 m
   - 2.5 mW, 4 dBm, 10 m
   - 1 mW, 0 dBm, 10 cm
   - Pros – Widely used industry standard, small form factor, no cabling
   - Cons – EMI/EMC concerns

3. **CAN**: A simple two-wire differential serial bus system, the CAN.bus operates in noisy electrical environments with a high level of data integrity, and its open architecture and user-definable transmission medium make it extremely flexible. Capable of high-speed (1 Mbps) data transmission over short distances (40 m) and low-speed (5 kbps) transmissions at lengths of up to 10,000 m, the multi-master CAN.bus is highly fault

tolerant, with powerful error detection and handling designed in. (Taken from Philips web page.)

- Pros – Excellent noise immunity, well defined electrical, protocol and application layers
- Cons – Power hungry

4. **1553**: An avionics command and control bus that can interface up to 32 devices on a differential serial interface. The specification can handle data rates up to 1 Mbps (Mil-Std-1773, optical version can handle 20 Mbps)
   - Pros – Widely used on military aircraft and spacecraft
   - Cons – Software intense, low data rate not suitable for massive data transfer, significant amount of overhead required in protocol, lots of supporting interface electronics such as termination resistors and transformers

Table 1: Summary of Low Data Rate Electrical Bus Technologies

	Backplane/ Cable	Central/ Distributed	Num Wires	Max Length	Data Rate	Payload Fraction
I^2C	Cable	Distributed	2	~4 m	400 Kbps	~85%
Bluetooth	N/A	Distributed	wireless	100 m	1 Mbps	72.3%
CAN	Cable	Distributed	2	40 m	1 Mbps	59%
1553	Cable	Central	2	~50 m	1 Mbps	75%

After considering the different low data rate buses for SCOUT, I^2C was selected for the following reasons

- Low Overhead
- Commercially Available
- Low Software Loading
- Low power
- Multi-drop

3.1.2 High Data Rate Electrical Bus Technologies

5. **IEEE 803.2 Fast Ethernet**: An industry standard telecommunications interface widely used to integrate telecommunications equipment and computers into local area networks. Embedded system designers are turning to Ethernet to solve the challenge of providing communication links between printed circuit boards (PCBs) in chassis-based networking equipment. Ethernet is an ideal technology for interconnecting multiple line cards and modules within telecommunications equipment backplanes and is widely displacing high-speed serial and proprietary communication protocols. Ethernet can handle data rates up to 100 Mbps and distances of 35 m over twisted pair cable or 1 m over a backplane. Implementing Ethernet in backplane solutions provides many benefits. As a standards-based technology, Ethernet is available from several silicon manufacturers, eliminating concerns of single-source supply. In addition, Ethernet uses innovative filtering and scrambling techniques to provide a high level of data integrity and noise suppression, and has the built-in capability to check for data corruption.

- Pros – Widely used standard, unlimited number of products to interface, eases ground support interface of spacecraft
- Cons – Not generally used in a space environment, would have to use industrial components

6. **VME**: A modular data bus with data bus sizes of 16, 32 or 64 bits and follows the Eurocard form factor. The 16 and 32 bit versions have dedicated address and data signals. The 64 bit version multiplexes the address and data signals on the same nets. A VME bus system consists of a controller, a master (controller and master can be the same device) and a slave and can be capable of data rates of up to 160 MBps.
   - Pros – Modular, industry standard
   - Cons – Bulky form factor, terminations may be required on backplane

7. **SCSI**: A 16 bit wide bus capable of handling data rates of up to 80 MBps. SCSI devices are usually connected to a controller through a cable and not a backplane.
   - Pros – Widely used and recognized interface in the PC industry
   - Cons – More than one industry standard, not all SCSI devices are compatible, only one controller permitted on the bus

8. **USB 2.0**: Also known as Hi-speed USB. The Universal Serial Bus (USB) is a serial interface which can link up to 127 devices on one serial bus. The maximum data rate over the bus is 480 MBps.
   - Pros – High data rate, compatibility with other USB devices
   - Cons – Device designs rely heavily on embedded software development, multiple devices require the use of a hub

9. **RS-644**: A low voltage differential signal (LVDS) equivalent to RS-422. The LVDS signal is uses 1.0 and 1.4 volts as its differential voltage levels. These signals can be transmitted over cables or over a backplane. Because of this lower voltage, much less power is used and a greater data rate can be achieved, up to 655 Mbps. Signals can be bussed to achieve even higher data rates.
   - Pros – High data rate differential, low EMI, low power
   - Cons – Electrical specification only, must use software for communications protocol

10. **cPCI**: The Compact PCI bus (cPCI) follows the PCI bus electrical specification in a Eurocard form factor. Data packets of 8, 16, 32 and 64 bits can be transferred across the bus at rates up to 66 MHz (528 MBps). Data and address signals are multiplexed across the same lines.
    - Pros – Modular, industry standard, any device on bus can become bus master
    - Cons – Bulky form factor, voltage compatibility is an issue (5V, 3.3V PCI versions)

11. **IEEE 1394**: Also know as Firewire, a high data rate point to point interface mainly used for the transmission of digital video signals. The interface is capable of data rates ranging from 100 Mbps up to a maximum of 4 Gbps. The hop distance between each node should not exceed 4.5 m and the maximum number of hops in a chain is 16, therefore the total maximum end-to-end distance is 72 m.

- Pros – High data rate, industry standard
- Cons – Point to point connection (not a multidevice bus)

Table 2: Summary of High Data Rate Electrical Bus Technologies

	Backplane/ Cable	Central/ Distributed	Num Wires	Max Length	Data Rate	Payload Fraction
IEEE 803	Both	Distributed	4	35 m	100 Mbps	96 %
VME	Backplane	Central	128	<1 m	160 Mbps	99.5 %
SCSI	Cable	Distributed	68	6 m	320 Mbps	99 %
USB 2.0	Cable	Central	4	~5 m	480 Mbps	99 %
RS-644	Cable	Distributed	2 min	10-15 m	655 Mbps	99 %
cPCI	Backplane	Central	124	<1m	1000 Mbps	99.5 %
IEEE 1394	Cable	Distribute	4	4.5m	4 Gbps	96 %

After considering the different high data rate buses for SCOUT, IEEE 803.2 Fast Ethernet was selected for the following reasons

- High data throughput
- Well understood standard
- Commercially available
- Easy to interface to development systems
- Well established technology
- Fairly inexpensive to implement

3.2 MODULE TECHNOLOGIES

12. AeroAstro is developing miniature S-Band and X-Band transponders for use on small satellites, as shown in Figure 13 and Figure 14. Flight S-Band transmitters have been delivered for use on the Canadian Space Agency MOST satellite, and S-Band receivers have been deigned up to the layout stage. There are several proposals currently pending which would involve delivering completed S-Band transponders. X-Band transponders for the NASA NMP ST-5 satellites have passed CDR, the prototypes have passed DSN compatibility testing and the flight models will be completed in a few months.
13. AeroAstro has completed a study for NASA Wallops Flight Facility (WFF) which investigated the feasibility of using the Globalstar satellite constellation at 1414 km altitude as a "virtual ground station" to provide more continuous communications coverage for rockets and satellites in LEO. Virtual ground station technology has the potential to be an effective enabler of rapid, responsive and reliable communications freed from the scheduling and availability constraints of the existing military ground station infrastructure. Two more proposals to WFF for further study of this concept are pending, and at least three other interested parties have also been identified. The Globalstar solution is only viable in the long term if Globalstar continues to maintain their constellation. The INMARSAT constellation of GEO satellites has been identified as a potentially better virtual ground station solution than Globalstar. This is primarily because of its wider coverage in LEO and at altitudes above the Globalstar constellation. INMARSAT offers low-bandwidth, low-power services that do not require pointed

antennas which has the potential to make it equally attractive as Globalstar for use on small satellites. INMARSAT has a history of continuously maintaining their constellation by deploying upgraded satellites, they are currently planning the deployment of their fourth generation. A virtual ground station may also be implemented using laser communications. In this scenario each SCOUT satellite is equipped with a miniature gimbaled low-power inter-satellite User Laser Transponder (ULTRA). The SCOUT satellites communicate with the ground through laser communications relay satellites in GEO which are equipped with ULTRAs as well as higher power laser transponders that are capable of penetrating Earth's atmosphere. The advantages of this scenario lie in frequency reuse, more secure communications, much higher data rates than in RF, much better coverage than even INMARSAT and in avoiding frequency regulations. AeroAstro has outlined a conceptual design for the User Laser Transponder and is in the process of submitting a proposal.

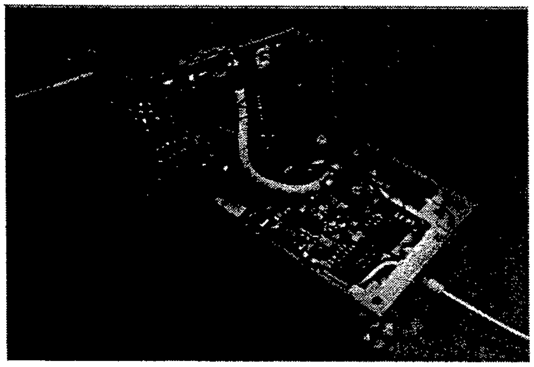

Figure 13:S-Band Transmitter

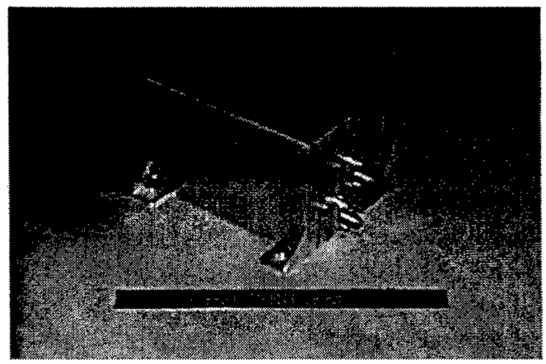

Figure 14: X-Band Transponder

14. The Johns Hopkins Applied Physics Lab (APL) has developed a miniature NAVSTAR/GPS receiver for satellites called GNS-II which is available for licensing. A compatible cross-link ranging transponder is also available. The GNS-II, shown in Figure 15, includes software for on-board orbit determination and orbit-propagation as well as event-based commanding for powerful operational autonomy. Events include specific times, node crossings, other locations, entry/exit from south Atlantic anomaly, eclipse, daylight and ground station visibility. This receiver is currently only intended for use in LEO and does not provide attitude information. Future versions should support orbit determination up to altitudes of approximately 30 Earth radii.

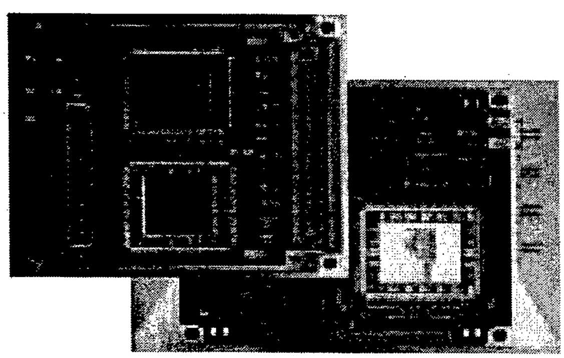

Figure 15: GNS-II **Figure 16:** Miniature Star Tracker

15. A novel method of orbit determination in high Earth orbits including GEO was filed for patent by ITT in 1999. The method requires using a UHF receiver to listen to the AUTONAV crosslink and ranging signals between NAVSTAR/GPS Block IIR satellites. This method of orbit determination requires initialization by another source but may nonetheless be very useful. There are currently eight Block IIR satellites on orbit and another twelve are planned for launch by September 2006. There is no guarantee that the AUTONAV system will be enabled.
16. AeroAstro is currently under contract to MDA to begin developing a low-power, low-accuracy, low-cost Miniature Star Tracker shown in Figure 16. This star tracker features a CMOS imager instead of the more traditional CCD imager, the advantages of the CMOS imager are that it requires much less supporting electronics and is more radiation tolerant. This star tracker also features pinhole optics, the absence of a lens and baffle assembly greatly reduces cost and volume of the star tracker at the expense of performance.
17. The Systron Donner BEI Gyrochip Model QRS11, shown in Figure 17, or future derivatives thereof, represents a suitable very low cost military grade MEMS angular rate sensor for use in an attitude determination module. This Gyrochip is currently available as a standard product.
18. The AeroAstro Medium Sun Sensor, shown in Figure 18, or future derivatives thereof, represents a suitable very low cost space-qualified sun sensor for use in the attitude determination module. This sun sensor is currently available as a standard product.

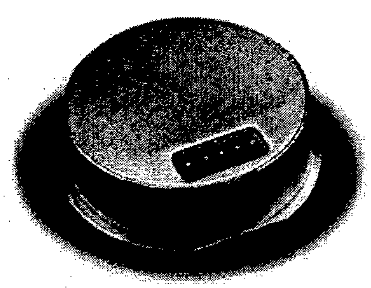

Figure 17: MEMS BEI Gyrochip

Figure 18: AeroAstro Medium Sun Sensor

19. The Billingsley Magnetics TFM100G2 satellite magnetometer, shown in Figure 19, or future derivatives thereof, represents a suitable very low cost space qualified magnetometer for use in the attitude determination module. This magnetometer is currently available as a standard product.
20. The Microcosm torque rods, shown in Figure 20, and AeroAstro torque rod driver boards, or future derivatives thereof, represent suitable low cost space qualified components for use in the magnetic actuation module. These are both currently available as standard products, the driver boards will be validated in space on board STPSat-1 ~2007.

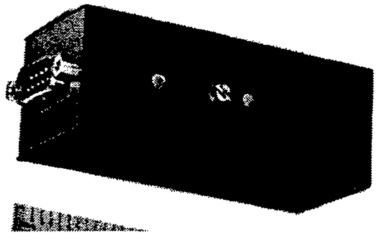

Figure 19: TFM100G2 Magnetometer

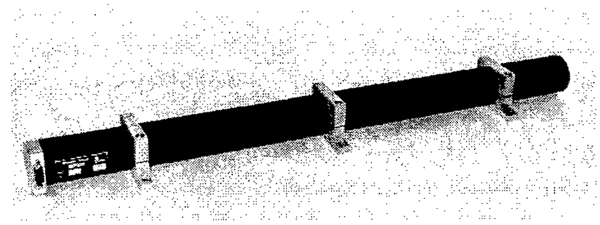

Figure 20: Microcosm Torque Rod

21. The Honeywell Micro-Wheel, shown in Figure 21, represents a suitable component for use in the micro wheel module for attitude actuation, momentum dumping, and power storage. In the Honeywell design using single crystal silicon wafer wheels the angular momentum stored per kg and per m^3 is much higher than what is possible with traditional wheels. The integrated energy storage system is competitive with Lithium-Ion battery cells even if the multifunctionality is not accounted for. When micro wheels are arranged as shown in Figure 9 the multifunctional nature of the wheels does not cause conflicts and even allows momentum to be dumped to power instead of using magnetics or thrusters. This technology was demonstrated under an Air Force contract but has not yet been qualified, Honeywell is currently pursuing more funding for the micro-wheel project through DARPA.
22. AeroAstro is currently under contract to NASA JSC to develop a Nitrous Oxide Propulsion System. This is a suitable system for use in the divert as well as the divert and maneuvering propulsion modules. When used in a hot-gas type system where the liquid Nitrous Oxide propellant is decomposed within each thruster the Isp is ~ 120- 200 seconds, which is in between cold gas and Hydrazine. An advantage over cold gas is that the propellant is stored in liquid form which greatly reduces volume. An advantage over Hydrazine is that the propellant is non-toxic and much safer and cheaper to work with. The propellant is also stored at much lower pressure than either cold gas or hydrazine allowing the use of conformal or otherwise not rounded propellant tanks for greatly increased volumetric efficiency.
23. The Lightband from Planetary Systems Corporation, or future versions thereof, has been identified as a suitable separation system for the separation module, it is shown in Figure 22. Lightband is currently the standard separation system for the EELV Secondary Payload Adapter. Lightband is currently available as a standard product.

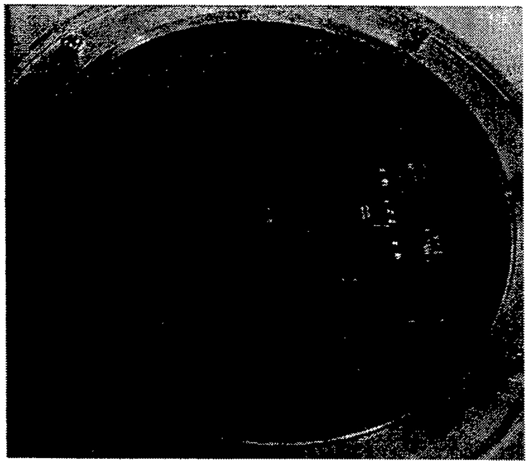

Figure 21: Honeywell Micro Wheel

Figure 22: Lightband Separation System

4. Summary

This document has described conceptual key technology elements that are required to implement the SCOUT architecture. This document has also described actual existing and in-development candidate technologies that may be capable of meeting the requirements if leveraged. The availability and state of development of the candidate technologies was assessed. Many of the technologies listed are alternatives to existing technologies which are not mature enough in terms of miniaturization or low cost to be of use for the SCOUT architecture.

AeroAstro, Inc. SCOUT Requirements Definition Matrix
Last Revised: 3/10/2003
Section 2.0 Spacecraft Subsystems

Pre-CoDR
DRAFT

Requirement Number	Requirement Description	Satellite Inspection and Monitoring Services	Tactical GPS	Enable Rapid Access and Flight Dem-Val of New Technologies	Units	Notes	More Notes
2.1	**Orbit Determination & ADCS**						
	Orbit Definition						
2.1.1	Perigee altitude (km)	GEO (~35,786)	575 <= alt <= 1000	LEO, ~550	kms		
2.1.2	Apogee altitude (km)	GEO (~35,786)	575 <= alt <= 1000	LEO, ~550	kms		
2.1.3	Inclination (deg)	0	any	~50, go through SAA	deg		
2.1.4	Notes	Assume release from primary or drop off within 1 km by other vehicle	Any orbit could be desireable from payload point of view, stay below rad belts and above atmopshere, lower and upper limits help constrain payload design (recommend cone w/ 68 deg half angle)	Maybe they would prefer GTO, aerobraking, or GEO?			
2.1.5	Overall orbit determination type	GEO slot must be known for ground comms, based on assumed known drop off, in emergencies ground ranging can be used	Relative to ECEF coordinates, required for regional duty-cycling and pseudorange accuracy, autonomous is preferable	Relative to ECEF coordinates, ephemeris too			
2.1.6	Overall orbit determination accuracy	+/- 624 km, 3 sig, based on 1st try successful contact using 2 deg beam width dish on ground	+/- 4.5 m, 3 sig (pseudorange is driver, Enge p35)	10 km prediction 1 week ahead, 1 km post processed 1 week later			
2.1.7	Detailed orbit determination type	Relative to primary, performed autonomously on-board vehicle and downlinked	None	None			
2.1.8	Detailed orbit determination accuracy	+/- 1 m, 3 sig	NA	NA			
	Attitude and Proximity Maneuvering					321 rotation sequence, local reference frame is ECI with y=anti-orbit-normal & z=nadir, Euler angles are null when payload faces nadir and body-fixed ram face points into the wind	All requirements are in deg or deg/sec and are 3 sigma, pointing is zero-to-peak and not peak-to-peak

2.1.9	Attitude type	3-axis control, arbitrary direction	3-axis control, nadir-pointed for full ground coverage at low altitude, yaw controlled for predictable SRP and Aero orbit disturbances	Uncontrolled		
2.1.10	Rapid or frequent slew required?	Yes	No	No		Defined as >= 0.05 deg/sec or more than once per orbit
2.1.11	Three-axis translation required?	Yes	No	No		
2.1.12	Translation-free torque required?	Yes	No	No		
2.1.13	Overall attitude determination type	3-axis, ECI	3-axis, ECI	3-axis, ECI		
2.1.14	Detailed attitude determination type	3-axis, relative to primary body-fixed frame	None	None		
2.1.15	Pointing knowledge roll	1.5	~1/3 of control, 0.3	1		pointing knowledge and control requirements hold for both overall and detailed types
2.1.16	Pointing knowledge pitch	1.5	~1/3 of control, 0.3	1	deg	
2.1.17	Pointing knowledge yaw	1.5	~1/3 of control, 0.3	1	deg	
2.1.18	Rate knowledge roll	0.1	None	None	deg	
2.1.19	Rate knowledge pitch	0.1	None	None	deg	
2.1.20	Rate knowledge yaw	0.1	None	None	deg	
2.1.21	Pointing control roll	10	1, no coverage loss for <=1.5 using 68 deg half angle beam at 575 km or higher	None	deg	
2.1.22	Pointing control pitch	10	1, no coverage loss for <=1.5 using 68 deg half angle beam at 575 km or higher	None	deg	
2.1.23	Pointing control yaw	10	1, to minimize SRP and Aero orbit disturbances	None	deg	
2.1.24	Rate control roll	0.3	None	None	deg/sec	
2.1.25	Rate control pitch	0.3	None	None	deg/sec	
2.1.26	Rate control yaw	0.3	None	None	deg/sec	
2.1.27	Mission lifetime (days)	90	365	365	days	
2.1.28	Mission duty cycle (1 = full time)	1	1	1		
2.1.29	DV margin (%)	30	NA	NA	%	
2.1.30	Margined attitude control DV (m/s, iff thrusters clearly required)	126.4	NA	NA	m/s	
2.1.31	Margined proximity maneuvering DV (m/s)	61.4	NA	NA	m/s	
2.1.32	Margined NSSK (m/s)	14.7	NA	NA	m/s	
2.1.33	Margined EWSK (m/s)	0.5	NA	NA	m/s	
2.1.34	Margined Total DV (m/s)	203.1	NA	NA	m/s	
	Divert Propulsion					
2.1.35	Delta V	0	0	0	m	
2.1.36	Thrust/mass_init (based on maneuver time)	NA	NA	NA		

2.2 Power

2.2.1	S/C Power Required		53.1	40.1	40.1 Watts	
2.2.2	Payload Power Required		20	5	30 Watts	
2.2.3	S/C Bus Voltage	28 +/- 6	28 +/- 6	28 +/- 6	VDC	
2.2.4	Payload Bus Voltage	Unregulated	Unregulated	Unregulated	VDC	28 +/- 6VDC
2.2.5	Battery Capacity		240	100	180 Watt-hours	
2.2.6	Peak Power		152	121	121 Watts	
2.2.7	Nominal Power Load		73.1	60	54 Watts	
2.3	**Thermal**					
2.3.1	max survival temperature			WSSP: +70 C	deg C	
2.3.2	min survival temperature			WSSP: -40 C	deg C	
2.3.3	max operating temperature			WSSP: +40 C	deg C	
2.3.4	min operating temperature			WSSP: -25 C	deg C	
2.4	**Command and Data**					
2.4.1	Mass Storage Capacity		165	4	4 MB	
2.4.2	Processing Power		40 <<40	<<40	Mips	
2.4.3	RF Uplink rate		9600	76800	500000 Kbps	
2.4.4	RF Downlink Rate		76800.0	153600	1000000 bps	
2.4.5	Payload Data rate		5 TBD		0.000256 MB per second	
2.4.6	Payload Data Compression	25:1	No	Yes		
2.4.7	Time Source Accuracy			10 msec to UTC	msec	
2.4.8	S/C Telemetry Gathering Rate		1	1	1 Kbps	
2.4.9	Analog lines required		64	64	64	0 for WSSP
2.4.10	Discrete Lines Required		16	16	16	2 for WSSP

AeroAstro, Inc.
Last Revised: 3/10/2003

Section 3.0 Concept of Operations

SCOUT Requirements Definition Matrix

Pre-CoDR
DRAFT

Requirement Number	Requirement Description	Satellite Inspection and Monitoring Services	Tactical GPS	Enable Rapid Access and Flight Dem-Val of New	Units	Notes	More Notes
3.1	**Link Budget**						
3.11	Frequency	2250.00	2250.00	2250.00	MHz	Sband	
3.12	Altitude	34784	1000	500	km		
	Elevation	5	5	5	degrees	Worst case elevation	
	Elevation (radians)	0.087266463	0.087266463	0.087266463	radians		
	S/C antenna look angle	8.879628775	59.44847305	67.48447048	degrees		
	S/C antenna look angle (radians)	0.154978758	1.03757159	1.177826204	radians		
3.13	S/C Antenna Gain (omni)	3	3	3	dBi		
3.14	HPA O/P	5	1	0.2	W		
	Feed Loss	0.5	0.5	0.5	dB		
	EIRP	9.489700043	2.5	-4.489700043	dBW		
	Range	40112.78627	3194.454235	2077.93954	km		
	FSL	-191.5499935	-169.5722703	-165.836995	dB		
3.15	Polarisation Losses	3	3	3	dB		
	Add losses	0.93	0.93	0.93	dB		
	Boltz	-228.6	-228.6	-228.6	dBW/Hz/K		
	Ant Temp	350	350	350	K		
	Ant Gain (5m Dish)	39.2	39.2	39.2	dBi		
	Ant feed loss	0	0	0	dB		
	loss factor	1	1	1			
	Ambient Temperature	293	293	293	k		
	Rx NF	0.41	0.41	0.41	dB		
	Te	29.00871095	29.00871095	29.00871095	k		
	Tsyst	379.008711	379.008711	379.008711	k		
	Rx G/T	13.41350808	13.41350808	13.41350808	dB/k		
	C/No at Demod	56.02321467	71.0112378	67.75681303	dBHz		
	Chip Rate (Mcps)	0.0384	0.0384	0.0384	Mcps		
3.16	Bandwidth (MHz)	0.15	0.3	2	MHz		
	(C/N estimated)	4.262302075	16.24002525	4.74651307	dB		
	Ec/No	10.18	25.17	21.91	dB		
3.17	Bit Rate (bps)	76800	153600	1000000	bps	1 pic per 10 seconds is hi rate/hi res	
	Processing Gain (Rc/Rb)	-3.01	-6.02	-14.16	dB		
3.18	Eb/No	7.17	19.15	7.76			
	Eb/No Required	5	5	5	dB		
	Implementation Loss	1.5	1.5	1.5			
3.19	**Eb/No Margin**	**0.67**	**12.65**	**1.26**	dB		
	PFD (W/m^2)	4.39739E-16	1.38675E-14	6.55473E-15	W/m^2		
	E-Field	4.07163E-07	2.28649E-06	1.57198E-06	V/m		
	PFD dB[(W/m^2/4KHz]	-169.3083642	-157.330641	-168.8241532	dB[W/m^2/4KHz]		

PFD LIMIT	-154	-154	-154	dB[W/m^2/4KHz]		
3.20 **PFD Margin**	15.31	3.33	14.82	dB		
3.2 **Orbit Determination Method**	Track and Range	GPS or Norad	GPS			
3.3 **Ground Access Rate**						
3.31 Average pass duration	150	15	10	minutes	worst case antenna availability	
3.32 Number of available antennas	2	1	1			
3.33 Command load volume	6.4	1		kb		accomodate software code loads
3.34 Upload duration	3	0.5		seconds	2.4kbps each case	assume upload rate = half of download rate
3.35 Download volume (Payload, RT and Stored tlm)	64		4	MB	Payload section CFPL now calls for 32 MB	
3.36 Download duration	112		0.3	minutes		
3.37 Orbit Determination rate	daily	daily	daily			
3.38 Maneuver frequency - eccentricity	once every 12 days	none	none			
3.39 Maneuver frequency - inclination	once every 8 days	none	none			
3.40 Maneuver coordination with target	yes	no	no			
3.5 **Uplink Budget**						
3.51 Frequency	2025.00	2025.00	2025.00	MHz		
Altitude	34784.00	1000.00	500.00	km		
Elevation	**5.00**	**5.00**	**5.00**	**degrees**	worst case elevation	
S/C antenna look angle	8.88	59.45	67.48	degrees		
S/C antenna look angle (radians)	0.15	1.04	1.18	radians		
dipole antenna gain	2.00	-5.27	-7.97	dBi		
GNDSTN Dish Gain	38.30	38.30	38.30	dBi		
3.52 HPA O/P	200.00	10.00	10.00	W	ground site xmit power	
Feed Loss	1.00	1.00	1.00	dB		
EIRP	60.31	47.30	47.30	dBW		
Range	40112.79	3194.45	2077.94	km		
FSL	-190.63	-168.66	-164.92	dB		
3.53 Polarisation Losses	3.00	3.00	3.00	dB	assumes circular xmit, linear rx	
Add losses	0.93	0.93	0.93	dB		
Boltz	-228.60	-228.60	-228.60	dBW/Hz/K		
Ant Temp	180.00	180.00	180.00	K		
3.54 S/C Antenna Gain	3.00	3.00	3.00	dBi	assumes shared uplink/downlink antenna	
Ant feed loss	0.00	0.00	0.00	dB		
loss factor	1.00	1.00	1.00			
Ambient Temperature	293.00	293.00	293.00	k		
Rx NF	9.00	9.00	9.00	dB		
Te	2034.38	2034.38	2034.38	k		
Tsyst	2214.38	2214.38	2214.38	k		
Rx G/T	-30.45	-30.45	-30.45	dB/k		
C/No at Demod	63.89	72.86	76.60	dBHz		
Carrier Power at Antenna O/P	-101.25	-92.29	-88.55	dBm		
Noise Power at Antenna O/P	-165.15	-165.15	-165.15			
3.55 Bit Rate	2400.00	2400.00	2400.00	Baud		
3.56 Bandwidth	40000.00	40000.00	40000.00	Hz		
(C/N estimated)	17.87	26.84	30.58	dB		
Eb/No	30.09	39.06	42.79	dB		

3.57	Eb/No Required	20.00	20.00	20.00 dB
	Implementation Loss	1.50	1.50	1.50
3.58	**Eb/No Margin**	**8.59**	**17.56**	**21.29 dB**

SCOUT Requirements Definition Matrix

Pre-CoDR
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Requirement Number	Requirement Description	Satellite Inspection and Monitoring Services	Tactical GPS	Enable Rapid Access and Flight Dem-Val of New Technologies	Notes	More Notes
4.1	**Spacecraft (SC) General**					
4.1.1	The SC shall be responsible for supplying power (supply and return) and data connectivity to each payload module that requires it.	X	X	X		
4.1.2	The SC shall supply +28 V +/- 4 VDC power to the PL module.	X	X	X		
4.1.3	In order to minimize contamination of the PL, particularly due to organics, all non-sealed SC components shall be low-outgassing, defined as materials that have less than 1% TML, and less than 0.10% CVCM.	X	X	X		
4.1.5	The SC shall sense and report the external temperatures of payload elements as part of the normal vehicle SOH system. Number and location of temperature sensors needed to verify that payload temperature requirements are being met or diagnose a thermal problem on-orbit is TBD.	X	X	X		

4.1.6	The SC shall include sufficient data storage capability to transfer payload data. The SC shall store the data until it can be successfully transmitted to the ground station. The SC shall accommodate any overhead (headers, footers, etc.) that may be required, and the potential need to retransmit the download if a problem should occur.	X	NA	X
4.1.7	Payload data shall be buffered in FIFOs on the C&DH, and read by the flight software fast enough to prevent overflows and/or loss of data.	X	NA	X
4.2	**Payload (PL) Specific**			
4.2.1	**Configuration & ConOps**			
4.2.1.1	The PL module must have a cross-section area of 20 x 20 cm and a palette height of no less than 2 cm and not to exceed (cm):	TBD	TBD	TBD
4.2.1.2	All PL shall meet or exceed the higher-level system mechanical requirements including the sub-component mounting method and thermal and electrical conductivity.	X	X	X
4.2.1.3	All PL in their integrated and pre-integrated states shall either meet or exceed vibration requirements as it pertains to the component's internal structure.	X	X	X
4.2.1.4	Command and operation of the PL shall be controlled by the ground.	X	X	X
4.2.1.5	The PL shall remain in a dormant "sleep mode" unless commanded otherwise by the ground.	X	X	X
4.2.1.6				
	Alignment & FOV			
4.2.2	Provide a FOV exclusion angle (deg)	92.4	NA	Greatest extent possible
4.2.2.1	Provide an unobstructed radiation pattern exclusion angle (deg)	TBD	150 deg	Greatest extent possible

4.2.2.2	Component alignment reference frame to be defined with respect to:	SC Body frame	SC Body frame	SC Body frame
4.2.2.3	Component target coordinate frame to be defined with respect to:	NA	SC Body frame	TBD
4.2.2.4				
4.2.2.7	**Ephemeris Data**			
	Provide SCOUT attitude and ephemeris data versus time wrt UTC.	TBD	X	X
4.2.3	Provide ephemeris predictions of in-track, cross-track, and altitude accuracy within (km)	TBD	300	TBD
4.2.3.1	Provide ephemeris predictions prior to planned observations within specified timeframe (days)	TBD	1	TBD
4.2.3.2	Provide post-processed ephemeris data of in-track, cross-track, and altitude accuracy within (km)	TBD	NA	TBD
4.2.3.3	Provide post-processed ephemeris data following component operation within specified timeframe (days)	TBD	NA	TBD
4.2.3.4				
4.2.3.5	**Environmental**			
	The Thermal Subsystem shall maintain the component survival temperature within a specifed range:	TBD	TBD	TBD
4.2.4	The Thermal Subsystem shall maintain the component operating temperature within a specifed range:	TBD	TBD	TBD
4.2.4.1	The heat flux between the component radiator and the spacecraft bus shall be less than (deg):	TBD	TBD	TBD
4.2.4.2	The PL shall not be exposed to a Peak Radiation Dosage of more than (krad):	TBD	TBD	TBD
4.2.4.3				
4.2.4.4	**C&DH**			
	The SC processor shall communicate with the component via a specified protocol:	TBD	TBD	TBD

4.2.5	The SC processor shall provide a time reference and a time sync line with an accuracy relative to UTC (ms):	TBD	NA	10
4.2.5.1	The SC processor shall provide a time reference and a time sync line with a frequency of (per orbit):	TBD	NA	once
4.2.5.2	Within specified timeframe, the SC processor shall be sized to store collected data not including headers and footers (MB):	60	NA	32
4.2.5.3	The SC processor shall be capable of storing processed data and transmitting it to the ground with a specified frequency of (orbits):	TBD	NA	TBD
4.2.5.4	The SC C&DH shall be capable of accepting component data in a "normal mode" at a rate of (kbits/sec):	29	NA	TBD
4.2.5.5	The SC C&DH shall be capable of accepting component data in a high bandwidth "burst mode" at a maximum rate of (kbits/sec):	2000	NA	NA
4.2.5.6	The SC C&DH shall be capable of operating in a high rate "burst mode" for a maximum specified duration (sec):	30	NA	NA
4.2.5.7.1	The SC shall store telemetry in a specified data format (word bits):	16	16	16
4.2.5.7.2	In order to enable algorithm and parameter patches, the SC shall provide a software upload capability to the component on an as available basis. The software upload will be delivered in multiple packets over multiple passes with the possibility of out-of-order sequences.	X	X	X
4.2.5.8	The SC shall be able to accommodate software uploads of a specified size (kb):	TBD	1	20
4.2.5.9	The PL will be responsible for verifying accuracy, requesting packet retransmission, and integrating the packets into the final software load.	X	X	X
4.2.5.10	After "power on" the component will automatically perform a built-in test.	X	X	X

4.2.5.11				
4.2.5.12	**Deployments**			
	The SC will accommodate deployment of the component boom.	NA	X	TBD
4.2.6	Deployment of the comonent boom is a one-time event that shall be powered, sensed, and controlled by the SC.	NA	X	TBD
4.2.6.1	The SC shall provide a deployment mechanism and a stowage cradle for the component antenna	NA	X	TBD
4.2.6.2				
4.2.6.3				

AeroAstro, Inc. SCOUT Requirements Definition Matrix
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Section 4.0 Payload Component Specifications

Requirement Number	Requirement Description	Satellite Inspection and Monitoring Services RF Probe	Requirement Description	Satellite Inspection and Monitoring Services IR Camera	Requirement Description	Satellite Inspection and Monitoring Services Visual Camera	Requirement Description	Satellite Inspection and Monitoring Services Laser Range Finder	Requirement Description	Tactical GPS	Requirement Description	Enable Rapid Access and Flight Dem-Val of New Technologies
4.7	**Component Specifications**		**Component Specifications**		**Component Specifications**		**Component Specifications**		**Component Specifications**		**Component Specifications**	
4.7.1	Full Width Half-Power Angle (deg)	30	FOV (deg)	15 x 11	FOV (deg)	77	FOV (deg)		Mass (grams)	TBD	TBD	
4.7.2	Max Imaging Rate (Hz)	50	Minimum Imaging Array Active area	160 H x 128 V	Minimum Imaging Array Active area	1024 x 1024	Max Imaging Rate (Hz)	100	Dimensions (cm)	7.6 x 7.6 x 1.9		
4.7.3			Pixel size (µ meter)	51 x 51 microns	Pixel size (µ meter)	10 x 10	Resolution		Power Consumption (W)	0.5		
4.7.4					Dimensions (cm)	63 x 95mm	Spectral Response	860nm	Over/Reverse Voltage	TBD		
4.7.5					Focal Plane Readout (individual focal)	Progressive	Imager Control Electrical Interface	RS-422	Isolation	TBD		
4.7.6			Mass (grams)	120	Operating Voltage	TBD	Mass (grams)	100	Current Limit	TBD		
4.7.7			Dimensions (cm)	3.43W x 3.68H x 6.15D cm	Power Consumption	1.2 W	Dimensions (cm)	TBD				
4.7.8			Focal Plane Readout (individual focal)	Progressive	Sensitivity	TBD	Power Consumption (W)	0.3				
4.7.9			Power Consumption (W)	1.5								
4.7.10			Sensitivity	85 mK/bit								
4.8	**Performance**		**Performance**		**Performance**							
4.8.1	Resolution	TBD	Max Imaging Rate (Hz)	10	Quantum Efficiency	Approx. 65% peak at 520nM						
4.8.2	Spectral Response	50KHz to 18 GHz	Default Imaging Rate (Hz)	0.1	Full-well capacity	Greater than 300,000						
4.8.3	Imager Control Electrical Interface	RS-422	Resolution	14-bit/pixel	Anti-blooming	Overload margin > 800X						
4.8.4	Mass (grams)	1500	Spectral Response	7.5 to 13.5 microns	Integration Period (sec)	0.033						
4.8.5	Dimensions (cm)	17.8 x 7.6 x 5.6	Imager Control Electrical Interface	IEEE-1394 (Fire-Wire)	Max Imaging Rate (Hz)	30						
4.8.6	Power Consumption (W)	6-12										
4.8.7	Sensitivity	TBD			Optical resolution	8- or 16-bit/pixel						
4.8.8	Mass (grams)	TBD			Spectral Response	200 to 1100nm, peaking at 525nm						
4.8.9	Dimensions (cm)	TBD			Imager Control Electrical Interface	RS-644 LVDS						
4.8.10	Focal Plane Readout (individual focal)	TBD			Mass (grams)	250						
4.8.11	Operating Voltage	TBD			Dimensions (cm)	63 x 95mm						
4.8.12	Power Consumption	TBD			Focal Plane Readout (individual focal)	Progressive						
4.8.13	Sensitivity	TBD			Power Consumption (W)	1.2						
4.9	**Environment**		**Environment**			**Environment**						
4.9.1	Operational Temperatures	TBD	Operational Temperatures		-40 to +55 °C	Operational Temperatures	TBD					
4.9.2	Cleanliness	TBD	Cleanliness		TBD	Cleanliness	TBD					
4.9.3	Radiation Dosage: Average	TBD	Radiation Dosage: Average		TBD	Radiation Dosage: Average	TBD					
4.9.4	Radiation Dosage: Peak	TBD	Radiation Dosage: Peak		TBD	Radiation Dosage: Peak	TBD					

SCOUT Requirements Definition Matrix

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Requirement Number	Requirement Description	Satellite Inspection and Monitoring Services RF Probe	Satellite Inspection and Monitoring Services IR Camera	Satellite Inspection and Monitoring Services Visual Camera	Satellite Inspection and Monitoring Services Laser Range Finder	Tactical GPS	Enable Rapid Access and Flight Dem-Val of New Technologies
	RF Probe						
4.1	**Configuration & ConOps**						
4.1.1	The PL module must have a cross-section area of 20 x 20 cm and a palette height of no less than 2 cm and not to exceed (cm):	TBD	TBD	TBD	TBD	TBD	TBD
4.1.2	All PL shall meet or exceed the higher-level system mechanical requirements including the sub-component mounting method and thermal and electrical conductivity.	X	X	X	X	X	X
4.1.3	All PL in their integrated and pre-integrated states shall either meet or exceed vibration requirements as it pertains to the component's internal structure.	X	X	X	X	X	X
4.1.4	Command and operation of the PL shall be controlled by the ground.	X	X	X	X	X	X
4.1.5	The PL shall remain in a dormant "sleep mode" unless commanded otherwise by the ground.	X	X	X	NA	X	X
4.2	**Alignment & FOV**						
4.2.1	Provide a FOV exclusion angle (deg)	NA	18 x 13.2 deg	92.4	18 x 13.2 deg	NA	Greatest extent possible
4.2.1.1	Provide an unobstructed radiation pattern exclusion angle (deg)	NA	NA	NA	NA	150 deg	Greatest extent possible
4.2.1.2	Component alignment reference frame to be defined with respect to:	SC Body frame	SC Body frame	SC Body frame	SC Body frame	SC Body frame	SC Body frame
4.2.1.3	Component target coordinate frame to be defined with respect to:	Component Boresight	Component Boresight	Component Boresight	Component Boresight	SC Body frame	TBD
4.3	**Ephemeris Data**						
4.3.1	Provide SC attitude and ephemeris data versus time wrt UTC.	NA	NA	NA	NA	X	X

4.3.2	Provide ephemeris predictions of in-track, cross-track, and altitude accuracy within (km)	NA	NA	NA	NA	300	TBD
4.3.3	Provide ephemeris predictions prior to planned observations within specified timeframe (days)	NA	NA	NA	NA	1	TBD
4.3.4	Provide post-processed ephemeris data of in-track, cross-track, and altitude accuracy within (km)	NA	NA	NA	NA	NA	TBD
4.3.5	Provide post-processed ephemeris data following component operation within specified timeframe (days)	NA	NA	NA	NA	NA	TBD
4.4	**Environmental**						
4.4.1	The Thermal Subsystem shall maintain the component survival temperature within a specifed range:	TBD	TBD	TBD	TBD	TBD	TBD
4.4.2	The Thermal Subsystem shall maintain the component operating temperature within a specifed range:	-20 to +70 °C	-40 to +55 °C	TBD	TBD	TBD	TBD
4.4.3	The heat flux between the component radiator and the spacecraft bus shall be less than (deg):	TBD	TBD	TBD	TBD	TBD	TBD
4.4.4	The PL shall not be exposed to a Peak Radiation Dosage of more than (krad):	TBD	TBD	TBD	TBD	TBD	TBD
4.5	**C&DH**						
4.5.1	The SC processor shall communicate with the component via a specified protocol:	TBD	TBD	TBD	TBD	TBD	TBD
4.5.2	The SC processor shall provide a time reference and a time sync line with an accuracy relative to UTC (ms):	NA	NA	NA	NA	NA	10
4.5.3	The SC processor shall provide a time reference and a time sync line with a frequency of (per orbit):	NA	NA	NA	NA	NA	once
4.5.4	Within specified timeframe, the SC processor shall be sized to store collected data not including headers and footers (MB):	60	3	101	NA	NA	32
4.5.5	The SC processor shall be capable of storing processed data and transmitting it to the ground with a specified frequency of (orbits):	TBD	TBD	TBD	NA	NA	TBD
4.5.6	The SC C&DH shall be capable of accepting component data in a "normal mode" at a rate of (kbits/sec):	30	1	67	NA	NA	TBD
4.5.7	The SC C&DH shall be capable of accepting component data in a high bandwidth "burst mode" at a maximum rate of (kbits/sec):	2000	115	3355	NA	NA	NA

4.5.8	The SC C&DH shall be capable of operating in a high rate "burst mode" for a maximum specified duration (sec):	30	30	30	NA	NA	NA
4.5.9	The SC shall store telemetry in a specified data format (word bits):	16	16	16	16	16	16
4.5.10	In order to enable algorithm and parameter patches, the SC shall provide a software upload capability to the component on an as available basis. The software upload will be delivered in multiple packets over multiple passes with the possibility of out-of-order sequences.	X	X	X	X	X	X
4.5.11	The SC shall be able to accommodate software uploads of a specified size (kb):	100	20	20	20	1	20
4.5.12	The PL will be responsible for verifying accuracy, requesting packet retransmission, and integrating the packets into the final software load.	X	X	X	X	X	X
4.5.13	After "power on" the component will automatically perform a built-in test.	X	X	X	X	X	X
4.6	**Deployments**						
4.6.1	The SC will accommodate deployment of the component boom.	NA	NA	NA	NA	X	TBD
4.6.2	Deployment of the comonent boom is a one-time event that shall be powered, sensed, and controlled by the SC.	NA	NA	NA	NA	X	TBD
4.6.3	The SC shall provide a deployment mechanism and a stowage cradle for the component antenna	NA		NA	NA	X	TBD

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Last Revised: 3/10/2003

SCOUT Requirements Definition Matrix

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Section 4.0 Payloads

Requirement Number	Requirement Description	Satellite Inspection and Monitoring Services	Tactical GPS	Enable Rapid Access and Flight Dem-Val of New Technologies	Notes	More Notes
	IR Camera	**X**				
4.1	**Configuration & ConOps**					
4.1.1	The PL module must have a cross-section area of 20 x 20 cm and a palette height of no less than 2 cm and not to exceed (cm):	TBD				
4.1.2	All PL shall meet or exceed the higher-level system mechanical requirements including the sub-component mounting method and thermal and electrical conductivity.	X				
4.1.3	All PL in their integrated and pre-integrated states shall either meet or exceed vibration requirements as it pertains to the component's internal structure.	X				
4.1.4	Command and operation of the PL shall be controlled by the ground.	X				
4.1.5	The PL shall remain in a dormant "sleep mode" unless commanded otherwise by the ground.	X				
4.2	**Alignment & FOV**					
4.2.1	Provide a FOV exclusion angle (deg)	18 x 13.2 deg				
4.2.1.1	Provide an unobstructed radiation pattern exclusion angle (deg)	NA				
4.2.1.2	Component alignment reference frame to be defined with respect to:	SC Body frame				

4.2.1.3	Component target coordinate frame to be defined with respect to:	Component Boresight
4.3	**Ephemeris Data**	
4.3.1	Provide SC attitude and ephemeris data versus time wrt UTC.	NA
4.3.2	Provide ephemeris predictions of in-track, cross-track, and altitude accuracy within (km)	NA
4.3.3	Provide ephemeris predictions prior to planned observations within specified timeframe (days)	NA
4.3.4	Provide post-processed ephemeris data of in-track, cross-track, and altitude accuracy within (km)	NA
4.3.5	Provide post-processed ephemeris data following component operation within specified timeframe (days)	NA
4.4	**Environmental**	
4.4.1	The Thermal Subsystem shall maintain the component survival temperature within a specifed range:	TBD
4.4.2	The Thermal Subsystem shall maintain the component operating temperature within a specifed range:	-40 to +55 °C
4.4.3	The heat flux between the component radiator and the spacecraft bus shall be less than (deg):	TBD
4.4.4	The PL shall not be exposed to a Peak Radiation Dosage of more than (krad):	TBD
4.5	**C&DH**	
4.5.1	The SC processor shall communicate with the component via a specified protocol:	TBD
4.5.2	The SC processor shall provide a time reference and a time sync line with an accuracy relative to UTC (ms):	NA

4.5.3	The SC processor shall provide a time reference and a time sync line with a frequency of (per orbit):	NA
4.5.4	Within specified timeframe, the SC processor shall be sized to store collected data not including headers and footers (MB):	3
4.5.5	The SC processor shall be capable of storing processed data and transmitting it to the ground with a specified frequency of (orbits):	TBD
4.5.6	The SC C&DH shall be capable of accepting component data in a "normal mode" at a rate of (kbits/sec):	1
4.5.7	The SC C&DH shall be capable of accepting component data in a high bandwidth "burst mode" at a maximum rate of (kbits/sec):	115
4.5.8	The SC C&DH shall be capable of operating in a high rate "burst mode" for a maximum specified duration (sec):	30
4.5.9	The SC shall store telemetry in a specified data format (word bits):	16
4.5.10	In order to enable algorithm and parameter patches, the SC shall provide a software upload capability to the component on an as available basis. The software upload will be delivered in multiple packets over multiple passes with the possibility of out-of-order sequences.	X
4.5.11	The SC shall be able to accommodate software uploads of a specified size (kb):	20
4.5.12	The PL will be responsible for verifying accuracy, requesting packet retransmission, and integrating the packets into the final software load.	X
4.5.13	After "power on" the component will automatically perform a built-in test.	X
4.6	**Deployments**	

4.6.1	The SC will accommodate deployment of the component boom.	NA
4.6.2	The SC shall provide a deployment mechanism and a stowage cradle for the component antenna	NA
4.7	**Component Specifications**	
4.7.1	FOV (deg)	15 x 11
4.7.2	Minimum Imaging Array Active area	160 H x 128 V
4.7.3	Pixel size (μ meter)	51 x 51 microns
4.7.4		
4.7.5		
4.7.6	Mass (grams)	120
4.7.7	Dimensions (cm)	3.43W x 3.68H x 6.15D cm
4.7.8	Focal Plane Readout (Individual focal)	Progressive
4.7.9	Power Consumption (W)	1.5
4.7.10	Sensitivity	85 mK/bit
4.8	**Performance**	
4.8.1	Max Imaging Rate (Hz)	10
4.8.2	Default Imaging Rate (Hz)	0.1
4.8.3	Resolution	14-bit/pixel
4.8.4	Spectral Response	7.5 to 13.5 microns
4.8.5	Imager Control Electrical Interface	IEEE-1394 (Fire-Wire)
4.9	**Environment**	
4.9.1	Operational Temperatures	-40 to +55 °C
4.9.2	Cleanliness	TBD
4.9.3	Radiation Dosage: Average	TBD
4.9.4	Radiation Dosage: Peak	TBD

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Last Revised: 3/10/2003

Section 4.0 Payloads

SCOUT Requirements Definition Matrix

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Requirement Number	Requirement Description	Satellite Inspection and Monitoring Services	Tactical GPS	Enable Rapid Access and Flight Dem-Val of New Technologies	Notes	More Notes
	Visual Imager	**X**				
4.1	**Configuration & ConOps**					
4.1.1	The PL module must have a cross-section area of 20 x 20 cm and a palette height of no less than 2 cm and not to exceed (cm):	TBD				
4.1.2	All PL shall meet or exceed the higher-level system mechanical requirements including the sub-component mounting method and thermal and electrical conductivity.	X				
4.1.3	All PL in their integrated and pre-integrated states shall either meet or exceed vibration requirements as it pertains to the component's internal structure.	X				
4.1.4	Command and operation of the PL shall be controlled by the ground.	X				
4.1.5	The PL shall remain in a dormant "sleep mode" unless commanded otherwise by the ground.	X				
4.2	**Alignment & FOV**					
4.2.1	Provide a FOV exclusion angle (deg)	92.4				
4.2.1.1	Provide an unobstructed radiation pattern exclusion angle (deg)	NA				
4.2.1.2	Component alignment reference frame to be defined with respect to:	SC Body frame				
4.2.1.3	Component target coordinate frame to be defined with respect to:	Component Boresight				
4.3	**Ephemeris Data**					
4.3.1	Provide SC attitude and ephemeris data versus time wrt UTC.	NA				
4.3.2	Provide ephemeris predictions of in-track, cross-track, and altitude accuracy within (km)	NA				

4.3.3	Provide ephemeris predictions prior to planned observations within specified timeframe (days)	NA
4.3.4	Provide post-processed ephemeris data of in-track, cross-track, and altitude accuracy within (km)	NA
4.3.5	Provide post-processed ephemeris data following component operation within specified timeframe (days)	NA
4.4	**Environmental**	
4.4.1	The Thermal Subsystem shall maintain the component survival temperature within a specifed range:	TBD
4.4.2	The Thermal Subsystem shall maintain the component operating temperature within a specifed range:	TBD
4.4.3	The heat flux between the component radiator and the spacecraft bus shall be less than (deg):	TBD
4.4.4	The PL shall not be exposed to a Peak Radiation Dosage of more than (krad):	TBD
4.5	**C&DH**	
4.5.1	The SC processor shall communicate with the component via a specified protocol:	TBD
4.5.2	The SC processor shall provide a time reference and a time sync line with an accuracy relative to UTC (ms):	NA
4.5.3	The SC processor shall provide a time reference and a time sync line with a frequency of (per orbit):	NA
4.5.4	Within specified timeframe, the SC processor shall be sized to store collected data not including headers and footers (MB):	101
4.5.5	The SC processor shall be capable of storing processed data and transmitting it to the ground with a specified frequency of (orbits):	TBD
4.5.6	The SC C&DH shall be capable of accepting component data in a "normal mode" at a rate of (kbits/sec):	67

4.5.7	The SC C&DH shall be capable of accepting component data in a high bandwidth "burst mode" at a maximum rate of (kbits/sec):	3355	At camera's max rate and max resolution bus would have to support data collection in excess of 20 Mbps so we have reduced the max rate to 5 Hz.
4.5.8	The SC C&DH shall be capable of operating in a high rate "burst mode" for a maximum specified duration (sec):	30	
4.5.9	The SC shall store telemetry in a specified data format (word bits):	16	
4.5.10	In order to enable algorithm and parameter patches, the SC shall provide a software upload capability to the component on an as available basis. The software upload will be delivered in multiple packets over multiple passes with the possibility of out-of-order sequences.	X	
4.5.11	The SC shall be able to accommodate software uploads of a specified size (kb):	20	
4.5.12	The PL will be responsible for verifying accuracy, requesting packet retransmission, and integrating the packets into the final software load.	X	
4.5.13	After "power on" the component will automatically perform a built-in test.	X	
4.6	**Deployments**		
4.6.1	The SC will accommodate deployment of the component boom.	NA	
4.6.2	Deployment of the comonent boom is a one-time event that shall be powered, sensed, and controlled by the SC.	NA	
4.6.3	The SC shall provide a deployment mechanism and a stowage cradle for the component antenna	NA	
4.7	**Component Specifications**		
4.7.1	FOV (deg)	77	
4.7.2	Minimum Imaging Array Active area	1024 x 1024	
4.7.3	Pixel size (μ meter)	10 x 10	
4.7.4	Dimensions (cm)	63 x 95mm	
4.7.5	Focal Plane Readout (Individual focal)	Progressive	
4.7.6	Operating Voltage	TBD	

4.7.7	Power Consumption	1.2 W
4.7.8	Sensitivity	TBD
4.8	**Performance**	
4.8.1	Quantum Efficiency	Approx. 65% peak at 520nM
4.8.2	Full-well capacity	Greater than 300,000
4.8.3	Anti-blooming	Overload margin > 800X
4.8.4	Integration Period (sec)	0.033
4.8.5	Max Imaging Rate (Hz)	30
4.8.6		
4.8.7	Optical resolution	8- or 16-bit/pixel
4.8.8	Spectral Response	200 to 1100nm, peaking at 525nm
4.8.9	Imager Control Electrical Interface	RS-644 LVDS
4.8.10	Mass (grams)	250
4.8.11	Dimensions (cm)	63 x 95mm
4.8.12	Focal Plane Readout (Individual focal)	Progressive
4.8.13	Power Consumption (W)	1.2
4.9	**Environment**	
4.9.1	Operational Temperatures	TBD
4.9.2	Cleanliness	TBD
4.9.3	Radiation Dosage: Average	TBD
4.9.4	Radiation Dosage: Peak	TBD

SCOUT Requirements Definition Matrix

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Requirement Number	Requirement Description	Satellite Inspection and Monitoring Services	Tactical GPS	Enable Rapid Access and Flight Dem-Val of New Technologies	Notes	More Notes
	Laser Range Finder	**X**				
4.1	**Configuration & ConOps**					
4.1.1	The PL module must have a cross-section area of 20 x 20 cm and a palette height of no less than 2 cm and not to exceed (cm):	TBD				
4.1.2	All PL shall meet or exceed the higher-level system mechanical requirements including the sub-component mounting method and thermal and electrical conductivity.	X				
4.1.3	All PL in their integrated and pre-integrated states shall either meet or exceed vibration requirements as it pertains to the component's internal structure.	X				
4.1.4	Command and operation of the PL shall be controlled by the ground.	X				
4.1.5	The PL shall remain in a dormant "sleep mode" unless commanded otherwise by the ground.	NA				
4.2	**Alignment & FOV**					
4.2.1	Provide a FOV exclusion angle (deg)	18 x 13.2 deg				
4.2.1.1	Provide an unobstructed radiation pattern exclusion angle (deg)	NA				
4.2.1.2	Component alignment reference frame to be defined with respect to:	SC Body frame				

4.2.1.3	Component target coordinate frame to be defined with respect to:	Component Boresight
4.3	**Ephemeris Data**	
4.3.1	Provide SC attitude and ephemeris data versus time wrt UTC.	NA
4.3.2	Provide ephemeris predictions of in-track, cross-track, and altitude accuracy within (km)	NA
4.3.3	Provide ephemeris predictions prior to planned observations within specified timeframe (days)	NA
4.3.4	Provide post-processed ephemeris data of in-track, cross-track, and altitude accuracy within (km)	NA
4.3.5	Provide post-processed ephemeris data following component operation within specified timeframe (days)	NA
4.4	**Environmental**	
4.4.1	The Thermal Subsystem shall maintain the component survival temperature within a specifed range:	TBD
4.4.2	The Thermal Subsystem shall maintain the component operating temperature within a specifed range:	TBD
4.4.3	The heat flux between the component radiator and the spacecraft bus shall be less than (deg):	TBD
4.4.4	The PL shall not be exposed to a Peak Radiation Dosage of more than (krad):	TBD
4.5	**C&DH**	
4.5.1	The SC processor shall communicate with the component via a specified protocol:	TBD
4.5.2	The SC processor shall provide a time reference and a time sync line with an accuracy relative to UTC (ms):	NA

4.5.3	The SC processor shall provide a time reference and a time sync line with a frequency of (per orbit):	NA
4.5.4	Within specified timeframe, the SC processor shall be sized to store collected data not including headers and footers (MB):	NA
4.5.5	The SC processor shall be capable of storing processed data and transmitting it to the ground with a specified frequency of (orbits):	NA
4.5.6	The SC C&DH shall be capable of accepting component data in a "normal mode" at a rate of (kbits/sec):	NA
4.5.7	The SC C&DH shall be capable of accepting component data in a high bandwidth "burst mode" at a maximum rate of (kbits/sec):	NA
4.5.8	The SC C&DH shall be capable of operating in a high rate "burst mode" for a maximum specified duration (sec):	NA
4.5.9	The SC shall store telemetry in a specified data format (word bits):	16
4.5.10	In order to enable algorithm and parameter patches, the SC shall provide a software upload capability to the component on an as available basis. The software upload will be delivered in multiple packets over multiple passes with the possibility of out-of-order sequences.	X
4.5.11	The SC shall be able to accommodate software uploads of a specified size (kb):	20
4.5.12	The PL will be responsible for verifying accuracy, requesting packet retransmission, and integrating the packets into the final software load.	X
4.5.13	After "power on" the component will automatically perform a built-in test.	X
4.6	**Deployments**	

4.6.1	The SC will accommodate deployment of the component boom.	NA
4.6.2		NA
4.6.3	The SC shall provide a deployment mechanism and a stowage cradle for the component antenna	NA
4.7	**Component Specifications**	
4.7.1	FOV (deg)	
4.7.2	Max Imaging Rate (Hz)	100
4.7.3	Resolution	
4.7.4	Spectral Response	860nm
4.7.5	Imager Control Electrical Interface	RS-422
4.7.6	Mass (grams)	100
4.7.7	Dimensions (cm)	TBD
4.7.8	Power Consumption (W)	0.3

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SCOUT Requirements Definition Matrix

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Section 4.0 Payloads

Requirement Number	Requirement Description	Satellite Inspection and Monitoring Services	Tactical GPS	Enable Rapid Access and Flight Dem-Val of New Technologies	Notes	More Notes
	Tactical GPS		**X**			
4.1	**Configuration & ConOps**					
4.1.1	The PL module must have a cross-section area of 20 x 20 cm and a palette height of no less than 2 cm and not to exceed (cm):		TBD			
4.1.2	All PL shall meet or exceed the higher-level system mechanical requirements including the sub-component mounting method and thermal and electrical conductivity.		X			
4.1.3	All PL in their integrated and pre-integrated states shall either meet or exceed vibration requirements as it pertains to the component's internal structure.		X			
4.1.4	Command and operation of the PL shall be controlled by the ground.		X			
4.1.5	The PL shall remain in a dormant "sleep mode" unless commanded otherwise by the ground.		X			
4.2	**Alignment & FOV**					
4.2.1	Provide a FOV exclusion angle (deg)		NA			
4.2.1.1	Provide an unobstructed radiation pattern exclusion angle (deg)		150 deg		Goal	
4.2.1.2	Component alignment reference frame to be defined with respect to:		SC Body frame			
4.2.1.3	Component target coordinate frame to be defined with respect to:		SC Body frame			

4.3	**Ephemeris Data**	
4.3.1	Provide SC attitude and ephemeris data versus time wrt UTC.	X
4.3.2	Provide ephemeris predictions of in-track, cross-track, and altitude accuracy within (km)	300
4.3.3	Provide ephemeris predictions prior to planned observations within specified timeframe (days)	1
4.3.4	Provide post-processed ephemeris data of in-track, cross-track, and altitude accuracy within (km)	NA
4.3.5	Provide post-processed ephemeris data following component operation within specified timeframe (days)	NA
4.4	**Environmental**	
4.4.1	The Thermal Subsystem shall maintain the component survival temperature within a specifed range:	TBD
4.4.2	The Thermal Subsystem shall maintain the component operating temperature within a specifed range:	TBD
4.4.3	The heat flux between the component radiator and the spacecraft bus shall be less than (deg):	TBD
4.4.4	The PL shall not be exposed to a Peak Radiation Dosage of more than (krad):	TBD
4.5	**C&DH**	
4.5.1	The SC processor shall communicate with the component via a specified protocol:	TBD
4.5.2	The SC processor shall provide a time reference and a time sync line with an accuracy relative to UTC (ms):	NA
4.5.3	The SC processor shall provide a time reference and a time sync line with a frequency of (per orbit):	NA

4.5.4	Within specified timeframe, the SC processor shall be sized to store collected data not including headers and footers (MB):	NA
4.5.5	The SC processor shall be capable of storing processed data and transmitting it to the ground with a specified frequency of (orbits):	NA
4.5.6	The SC C&DH shall be capable of accepting component data in a "normal mode" at a rate of (kbits/sec):	NA
4.5.7	The SC C&DH shall be capable of accepting component data in a high bandwidth "burst mode" at a maximum rate of (kbits/sec):	NA
4.5.8	The SC C&DH shall be capable of operating in a high rate "burst mode" for a maximum specified duration (sec):	NA
4.5.9	The SC shall store telemetry in a specified data format (word bits):	16
4.5.10	In order to enable algorithm and parameter patches, the SC shall provide a software upload capability to the component on an as available basis. The software upload will be delivered in multiple packets over multiple passes with the possibility of out-of-order sequences.	X
4.5.11	The SC shall be able to accommodate software uploads of a specified size (kb):	1
4.5.12	The PL will be responsible for verifying accuracy, requesting packet retransmission, and integrating the packets into the final software load.	X
4.5.13	After "power on" the component will automatically perform a built-in test.	X
4.6	**Deployments**	
4.6.1	The SC will accommodate deployment of the component boom.	X

4.6.2	Deployment of the comonent boom is a one-time event that shall be powered, sensed, and controlled by the SC.	X
4.6.3	The SC shall provide a deployment mechanism and a stowage cradle for the component antenna	X
4.7	**Component Specifications**	
4.7.1	Mass (grams)	TBD
4.7.2	Dimensions (cm)	7.6 x 7.6 x 1.9
4.7.3	Power Consumption (W)	0.5
4.7.4	Over/Reverse Voltage	TBD
4.7.5	Isolation	TBD
4.7.6	Current Limit	TBD

Requirement Number	Requirement Description	Satellite Inspection and Monitoring Services	Tactical GPS	Enable Rapid Access and Flight Dem-Val of New Technologies	Notes	More Notes
	Customer Furnished PL (CFPL)			**X**		
4.1	**Configuration & ConOps**					
4.1.1	The PL module must have a cross-section area of 20 x 20 cm and a palette height of no less than 2 cm and not to exceed (cm):			TBD		
4.1.2	All PL shall meet or exceed the higher-level system mechanical requirements including the sub-component mounting method and thermal and electrical conductivity.			X		
4.1.3	All PL in their integrated and pre-integrated states shall either meet or exceed vibration requirements as it pertains to the component's internal structure.			X		
4.1.4	Command and operation of the PL shall be controlled by the ground.			X		
4.1.5	The PL shall remain in a dormant "sleep mode" unless commanded otherwise by the ground.			X		
4.2	**Alignment & FOV**					
4.2.1	Provide a FOV exclusion angle (deg)			Greatest extent possible		
4.2.1.1	Provide an unobstructed radiation pattern exclusion angle (deg)			Greatest extent possible		

4.2.1.2	Component alignment reference frame to be defined with respect to:	SC Body frame
4.2.1.3	Component target coordinate frame to be defined with respect to:	TBD
4.3	**Ephemeris Data**	
4.3.1	Provide SC attitude and ephemeris data versus time wrt UTC.	X
4.3.2	Provide ephemeris predictions of in-track, cross-track, and altitude accuracy within (km)	TBD
4.3.3	Provide ephemeris predictions prior to planned observations within specified timeframe (days)	TBD
4.3.4	Provide post-processed ephemeris data of in-track, cross-track, and altitude accuracy within (km)	TBD
4.3.5	Provide post-processed ephemeris data following component operation within specified timeframe (days)	TBD
4.4	**Environmental**	
4.4.1	The Thermal Subsystem shall maintain the component survival temperature within a specifed range:	TBD
4.4.2	The Thermal Subsystem shall maintain the component operating temperature within a specifed range:	TBD
4.4.3	The heat flux between the component radiator and the spacecraft bus shall be less than (deg):	TBD
4.4.4	The PL shall not be exposed to a Peak Radiation Dosage of more than (krad):	TBD
4.5	**C&DH**	
4.5.1	The SC processor shall communicate with the component via a specified protocol:	TBD
4.5.2	The SC processor shall provide a time reference and a time sync line with an accuracy relative to UTC (ms):	10

4.5.3	The SC processor shall provide a time reference and a time sync line with a frequency of (per orbit):	once
4.5.4	Within specified timeframe, the SC processor shall be sized to store collected data not including headers and footers (MB):	32
4.5.5	The SC processor shall be capable of storing processed data and transmitting it to the ground with a specified frequency of (orbits):	TBD
4.5.6	The SC C&DH shall be capable of accepting component data in a "normal mode" at a rate of (kbits/sec):	TBD
4.5.7	The SC C&DH shall be capable of accepting component data in a high bandwidth "burst mode" at a maximum rate of (kbits/sec):	NA
4.5.8	The SC C&DH shall be capable of operating in a high rate "burst mode" for a maximum specified duration (sec):	NA
4.5.9	The SC shall store telemetry in a specified data format (word bits):	16
4.5.10	In order to enable algorithm and parameter patches, the SC shall provide a software upload capability to the component on an as available basis. The software upload will be delivered in multiple packets over multiple passes with the possibility of out-of-order sequences.	X
4.5.11	The SC shall be able to accommodate software uploads of a specified size (kb):	20
4.5.12	The PL will be responsible for verifying accuracy, requesting packet retransmission, and integrating the packets into the final software load.	X
4.5.13	After "power on" the component will automatically perform a built-in test.	X
4.6	**Deployments**	

4.6.1	The SC will accommodate deployment of the component boom.	TBD
4.6.2	Deployment of the comonent boom is a one-time event that shall be powered, sensed, and controlled by the SC.	TBD
4.6.3	The SC shall provide a deployment mechanism and a stowage cradle for the component antenna	TBD
4.7	**Component Specifications**	
4.7.1	TBD	

Section 5.0 Spacecraft Interface & Launch Vehicle

Requirement Number	Requirement Description	Satellite Inspection and Monitoring Services	Tactical GPS	Enable Rapid Access and Flight Dem-Val of New Technologies	Notes	More Notes
5.1	**Spacecraft Interface**					
5.11	The spacecraft design shall include a separation system. The mass and volume of the separation system shall be accounted for in the allowable SV envelope and mass.					
5.12	The separation system shall provide for both electrical and mechanical interface separations and separation verification, and shall be compatible with the LV.					
5.13	The separation system shall be capable of providing a safe separation from the LV, and shall ensure a minimal probability of re-contact.					
5.14	Debris resulting from all separations shall be contained.					
5.15	Spacecraft to LV interface shall include separation sense capability.					
5.2	**Launch Vehicle**					
5.21	mass	75kg	75kg	75kg		
5.22	volume envelope	440mm x 440mm x 500mm	440mm x 440mm x 500mm	440mm x 440mm x 500mm		
5.23	*rigid mount fundamental frequencies*					
5.231	axial	50 Hz	50 Hz	50 Hz		
5.232	lateral	40Hz	40Hz	40Hz		
5.233	torsional	50 Hz	50 Hz	50 Hz		
5.24	*quasi-static loads*					
5.241	axial	13g	13g	13g		
5.242	lateral	2.5g	2.5g	2.5g		

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Last Revised: 3/10/2003

SCOUT Module Selection

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Section 6.0 Spacecraft Modules

Module Category	Module ID	Module Name	Module Description	Satellite Inspection and Monitoring Services	Tactical GPS	Enable Rapid Access and Flight Dem-Val of New Technologies	Notes	More Notes
C&DH	1	FPGA	FPGA-based brain			Y		
C&DH	2	CPU	CPU-based brain	Y	Y			
C&DH	3	Solid State Data Storage	Lots of memory	Y		Y		
Communications	4	S-Band Transponder	Based on MIDI-T and SPORT	Y				
Communications	5	X-Band Transponder	Based on ST5 X-Band Transponder					
Communications	6	INMARSAT-M Flight Modem	Virtual ground station		Y	Y		
Communications	7	TDRS Flight Modem	Virtual ground station					
Communications	8	Intersatellite Laser Transponder	Virtual ground station					
Communications	9	Ground Laser Transponder	Goes on relay					
Communications	10	Cryptography	NSA encryption and decryption	Y	Y	Y		
GN&C	11	GPS Receiver	APL GNS-II+, must work in GEO	Y	Y	Y		
GN&C	12	Attitude Determination	4 star trackers, 4 medium sun sensors, 3 one-axis gyros, 1 three-axis magnetometer, associated circuitry	Y	Y	Y		
GN&C	13	Magnetic Actuation	2 torque rods, 1 torque coil, associated circuitry		Y			
GN&C	14	Wheel Actuation	18 micro wheels arranged in groups of 6, one group per axis		Y			
Propulsion	15	Divert	1 divert thruster and associated tankage					
Propulsion	16	Divert & Proximity Operations	1 divert thruster, 12 smaller thrusters for proximity attitude control and maneuvering, associated tankage	Y				
Power	17	Battery	Chemical battery	Y	Y	Y		
Power	18	Solar Panel Array	4 multiply deployable solar panels, probably non-tracking, associatd electronics	Y	Y	Y		
Launch Vehicle Interface	19	ESPA PIM	Payload Attach Fitting (PAF) Interface Module	Y		Y		
Launch Vehicle Interface	20	SHELS PIM	Payload Attach Fitting (PAF) Interface Module					
Launch Vehicle Interface	21	RASCAL PIM	Payload Attach Fitting (PAF) Interface Module		Y			
Launch Vehicle Interface	22	ASAP5 PIM	Payload Attach Fitting (PAF) Interface Module					
Launch Vehicle Interface	23	Separation	To separate the main SCOUT modules from the PIM	Y				

Launch Vehicle Interface	24	External Structural Stiffeners	For extra stiffness on some launches			
Ground Support Equipment	25	MUGSE Base Plate	Master Universal Ground Support Equipement Base Plate	Y	Y	Y
Payload	26	Proximity Inspection Sensor Array	One or more visual cameras, one more IR cameras, RF Probe, and either laser range finder or near field radar	Y		
Payload	27	GPS-Lite	Generates and transmits GPS signals very similar to real GPS satellites		Y	
Payload	28	WSSP	Wafer Scale Signal Processing, for Space Based Laser			Y

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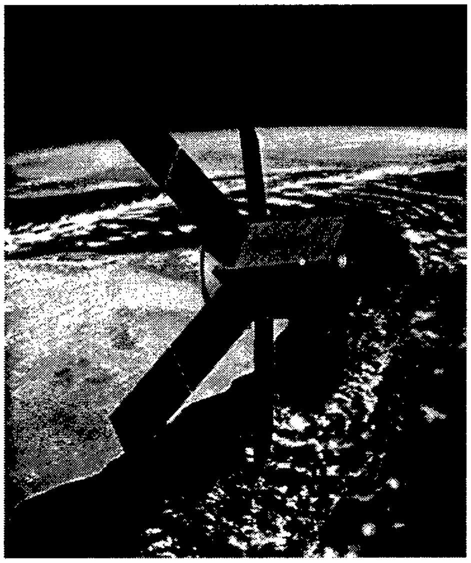

S^3COUT

A Small Smart SpaceCraft for Observation and Utility Tasks

Conceptual Design Review for DARPA

22 MAY 03

The following presentation was developed for the final review of the SCOUT Phase I SBIR program. This review was convened in the form of a Conceptual Design Review (CoDR). The focus of a traditional CoDR is feasibility; the goal of the conceptual design process is to work out enough technical design details to assure that there are no "showstoppers" in the concept. As the conceptual design is presented and discussed in the presentation that follows, it will be seen that the basic concept for the SCOUT microsatellite architecture is sound. No significant issues have been identified, although some elements of the system design are naturally more mature than others.

Agenda

SCOUT Conceptual Design Review
22 MAY 03 - SRS Technologies - Arlington, VA

Time	Duration		
8:00	0:15	Introduction	G. Cameron
8:15	0:20	Payload Description	A. Rogers
8:35	0:20	Requirements	S. Kennison
8:55	0:15	System Overview	G. Cameron
9:10	0:30	Mechanical & Thermal Design	J. Miller
9:40	0:20	Command and Data Handling	L. Jordan
10:00	0:15	Software	L. Jordan
10:15	0:15	Power	L. Jordan
10:30	0:20	Attitude Determination & Control System	A. Jacobovits
10:50	0:15	Propulsion	A. Jacobovits
11:05	0:20	RF Communications	G. Cameron
11:25	0:15	Mission Operations	S. Kennison
11:40	0:20	Integration and Test	A. Rogers
12:00		Adjourn	

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The Conceptual Design Review was convened on Thursday, 22 MAY 03 at SRS Technologies, 4001 N. Quincy Street, Suite 275, Arlington, VA. The order of the presentation is shown on this chart. G. Cameron presented the material prepared by S. Kennison in her absence. Due to time constraints, the material was actually presented in three hours rather than the scheduled four hours. Some of the design details that could not be covered during the presentation due to these time constraints will be discussed in greater depth in this document.

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Introduction

Glen Cameron

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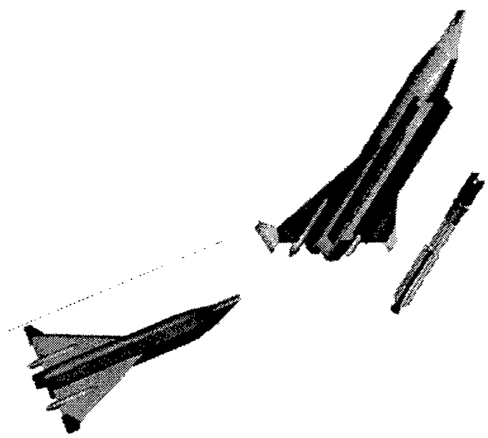
This presentation serves as an introduction to the SCOUT Program Conceptual Design Review.

The Need for SCOUT



- DARPA is developing RASCAL to fill a critical mission need for the US Military Space community
- RASCAL is intended to provide a quick-reaction launch capability to orbit military payloads in <u>days</u>
- Launching a payload in <u>days</u> is of limited utility when it takes *years* to build a typical Space Vehicle
- SCOUT is intended to provide a customized Space Vehicle - built to meet specific mission needs - in <u>days</u>, not *years*

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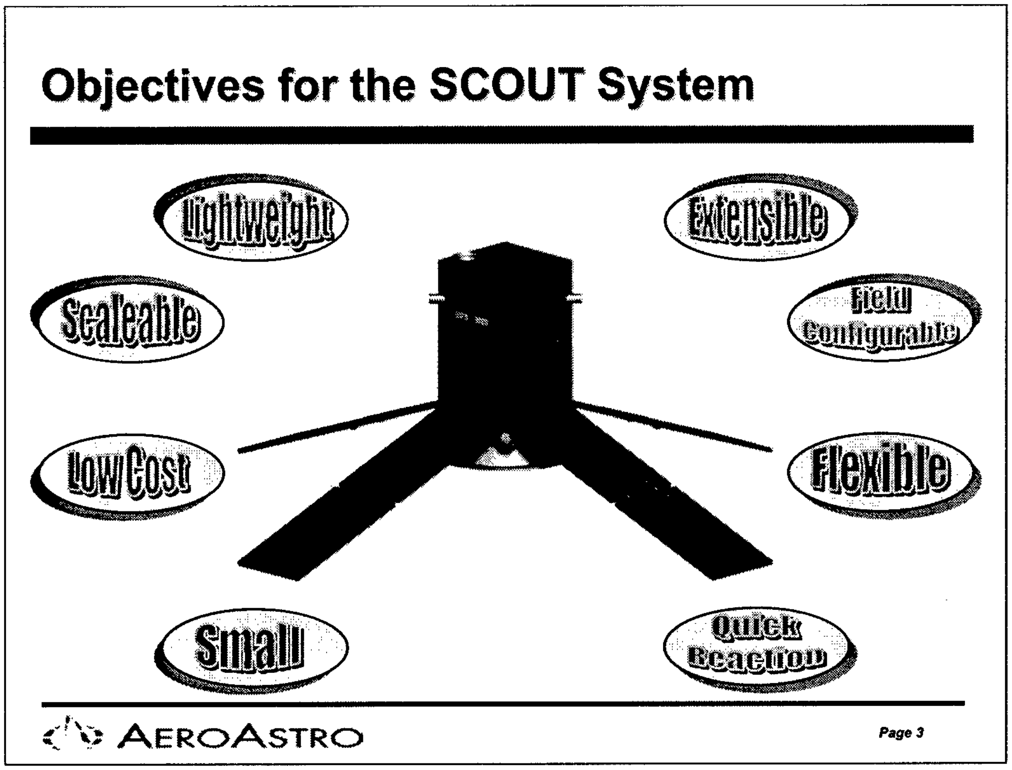
DARPA is currently developing a small launch vehicle called RASCAL for the express purpose of providing cheap, rapid access to space. RASCAL is envisioned as a launch vehicle that can insert a tactical or quick-reaction military payload into earth orbit on an extremely short time scale - ultimately within hours. RASCAL is intended to create a capability for the warfighter to utilize space power projection as easily and responsively as air power is currently employed.

Developing a capability like RASCAL implies that a tactical or quick-reaction military spacecraft must be available for launch on an equally short timescale. The ability to launch a payload within days is of little utility if a spacecraft takes years to develop. The future of military space will be limited unless the warfighter can rapidly configure a spacecraft to meet unique mission needs as flexibly as aircraft are currently configured. The true utility of a RASCAL-like capability can not be fully realized until such a rapid-response, configurable spacecraft is also available.

SCOUT is intended to provide exactly this type of flexible response - to be available in days or even hours, not years. SCOUT will allow the warfighter to configure a customized spacecraft to fill a unique need, assemble it, prepare it for launch, launch it, and put it into operational service rapidly enough to be compatible with the rapid pace of modern warfare.



To fully realize the goal of a truly utilitarian military space capability that will complement the capabilities of the RASCAL Launch Vehicle, SCOUT must meet the following technical objectives:

SCOUT must be SMALL and LIGHTWEIGHT to maximize utility within the limited performance envelope of a small launch vehicle. It is likely that any rapid-response launch vehicle (including RASACAL derivatives) will have a modest mass and orbit insertion capability. To further enhance the utility of the SCOUT architecture, it will be designed for compatibility with a wide variety of potential launch vehicles as both a primary and secondary payload.

SCOUT must be LOW COST to assure that it can be readily expended to serve US national security needs effectively. It is assumed that SCOUT-based spacecraft would need to cost under $1M each (in mass produced quantities) to be useful to US forces. A parallel in the current inventory with regard to the cost vs. capability envelope is the Tomahawk cruise missile.

SCOUT must provide a QUICK REACTION capability to allow US forces to swiftly respond to rapidly changing developments within an evolving battlespace. As recent conflicts have shown, the pace of modern warfare is constantly increasing; a response that takes weeks to implement will be of little or no utility in future engagements.

SCOUT must be FLEXIBLE to permit a managable complement of off-the-shelf hardware to address a wide-ranging set of scenarios. It is axiomatic that unforeseen

Development Process: Initial Assessment

- Mission Definition
  - Investigate potential SCOUT missions
  - Prioritize missions based upon interest and feasibility
  - Solicit input from DARPA to focus mission selection
  - Downselect to a single mission
- Technology Assessment
  - Identify key technologies needed to implement SCOUT
  - Identify existing and projected technologies
  - Assess state of development of new technologies
  - Identify technology developments that may be leveraged
  - Assess alternate or substitute technologies
  - Prepare a technology utilization and development plan

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The next four charts describe the process that was used to develop the conceptual architecture for the SCOUT system.

The development process began by defining a range of possible missions for a SCOUT-based space vehicle. Having identified a number of possible missions for such a vehicle, these missions were prioritized on the basis of sponsor interest in such a capability and the feasibility of implementing a compliant version of SCOUT. This process reduced the likelihood that the SCOUT architecture would be heavily biased by a mission that was unwanted or unlikely to be developed. The list of prioritized potential missions was discussed with the DARPA Technical Point of Contact to solicit input regarding the perceived usefulness of the envisioned configurations. The input of the sponsor was used to downselect to a single mission that served as the basis for a case study of the SCOUT architecture. This mission definition process was documented in the Mission Definition Study that was submitted to DARPA on 07 FEB 03.

In parallel to the mission definition study, although somewhat lagging this process, an assessment was conducted to evaluate the state of technologies that may be key to developing a SCOUT system. First, based upon the evolving mission definition study and the notional SCOUT architecture, a roster of key technologies was developed to provide a catalog of potentially useful capabilities. Having identified a "wish list" of technology needs for SCOUT, the current state of technological advancement was surveyed in a number of related technology disciplines to identify both existing and projected near-term technologies that could potentially satisfy any of the identified technology needs. In those cases where a match was realized

Development Process: Requirements

- Launch Vehicle Compatibility Analysis
  - Investigate RASCAL and develop LV requirements
  - Survey other launch vehicles and develop LV requirements
  - Develop a set of LV requirements for SCOUT that envelopes RASCAL and a wide range of alternative Launch Vehicles
- Requirements Definition
  - Analyze and develop a complete set of mission requirements
  - Develop a detailed Mission Requirements Matrix

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A Launch Vehicle Compatibility Analysis was conducted in parallel to the Mission Study and Technology Study. In this analysis, the planned specifications for the RASCAL Launch Vehicle were assessed and the key limiting parameters for SCOUT were extracted (e.g., envelope, mass, quasi-static loads, minimum first fundamental frequency, etc.). A comprehensive survey of other existing and planned launch vehicles was conducted to gather similar data across a wide range of potential Launch Vehicles, including capacity for both primary and secondary payloads. From this compendium of LV capability and requirements data, a single enveloped set of requirements was developed that would allow SCOUT to launch on ALL of the launch vehicles in the set. For a few selected parameters, a single launch vehicle was significantly more limiting than all of the other launch vehicles. On this basis, selected launch vehicles were excluded from the set to provide for a more meaningful (and less limiting) set of enveloped Launch Vehicle compatibility requirements. The resultant set of enveloped requirements provides a firm base for developing a family of SCOUT spacecraft that can launch on any of 18 current and planned launch vehicles. The results of this Launch Vehicle compatibility analysis were documented in a Launch Vehicle Study submitted to DARPA on 17 APR 03.

Following the Launch Vehicle Compatibility Analysis and Mission Definition Study, an effort was undertaken to define a set of requirements for the SCOUT modular architecture. The Launch Vehicle capabilities and restrictions formed a core for initiating this effort. Other requirements were developed for a number of related disciplines. These requirements included both general requirements that would apply to all SCOUT Space Vehicles and specific requirements that would apply to only the selected candidate missions. These requirements were summarized in a requirements matrix that was submitted to the DARPA sponsor in the form of

Development Process: Conceptual Design

- **SCOUT Conceptual Design**
  - **Develop an architecture that supports the requirements**
  - **Focus on modular approaches to the spacecraft bus that provide for evolutionary development**
  - **Identify modules common to multiple missions**
  - **Identify mission-specific modules**
  - **Develop an overall SCOUT mission design**
  - **Develop a specific SCOUT spacecraft design that meets these mission requirements**
  - **Identify ongoing technology developments that can be leveraged to enable or improve the capabilities of SCOUT**
  - ***Perform a Conceptual Design Review****

Work still to be completed*

AEROASTRO Page 6

Upon completion of the Mission Study, Technology Study, Launch Vehicle Compatibility Analysis, and Requirement Definition process, the SCOUT SBIR program moved into the core task of the Phase I effort: development of a conceptual design for a candidate SCOUT-based Space Vehicle.

Accordingly, a concept for a SCOUT-based microsatellite was developed to execute the "Escort" candidate mission that had been selected during the mission study process. This concept was intended to incorporate the recommended technologies from the technology study and be consistent with the Launch Vehicle restrictions and the requirements developed.

This process began with a set of "brainstorming" meetings in which the proposal and white paper concepts for SCOUT were reexamined in light of the requirements that had been developed. Many of the original concepts were retained in a general sense, but all elements of the design were refined. An architecture was developed that allowed SCOUT to meet or exceed all of the identified requirements. These early discussions focused on maintaining the flexibility, scalability, and extensibility of SCOUT by emphasizing the modularity of the system architecture. In all cases, modularity remained the central pillar of the SCOUT concept. With these basic features in place, the effort began to focus on defining the details of the modules themselves. A rough concept was developed for each of the three missions that had been studied. These concepts were detailed to identify both common and mission-specific modules and features. Via this process, the design details of the common modules were further defined to maximize flexibility and utility of these modules.

With a better understanding of these common modules, the focus returned to the specifics of a SCOUT-based vehicle that could be used to satisfy the Escort mission

Development Process : Documentation

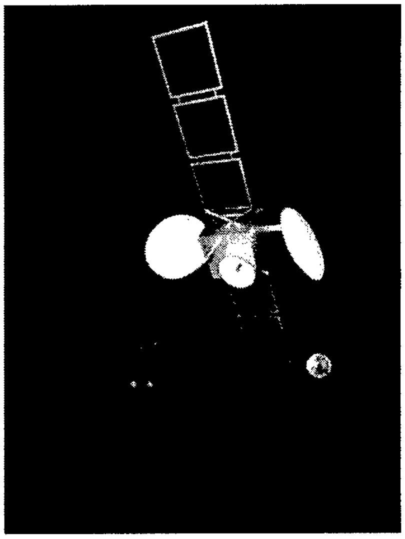
- **Final Report and Documentation**
  - ***Develop and deliver a final report summarizing all investigation, requirement development, and design activities****
  - ***Develop recommendations and plans for Phase II including a roadmap to develop SCOUT to a flight-ready prototype****
  - ***Develop a commercialization report including a roadmap to productize and commercialize the SCOUT system****

Work still to be completed*

AeroAstro

AeroAstro completed the Conceptual Design Review for the SCOUT Phase I SBIR on 22 MAY 03. The remaining task is to document the findings of the SCOUT Phase I SBIR in a final report. In consultation with the DARPA sponsor, it was agreed that the SCOUT Final Report will consist of six elements: 1) the Mission Study; 2) the Technology Utilization Plan; 3) the Launch Vehicle Analysis; 4) the Requirements Matrix; 5) a fully annotated version of the CoDR viewgraph package (of which this is a portion), and 6) an overall "glue" document which ties all of these items together into a cohesive report.

Mission Selection: Escort Vehicle



- Utilizing this mission selection process, it was determined that SCOUT would focus its CoDR on an Escort vehicle mission
- A SCOUT Escort vehicle can be used to:
  - Facilitate In-Orbit-Test activities
    - Analyze electromagnetic emissions
    - Verify antenna patterns
    - Provide confirmation of deployments, alignments, or slewing and pointing
  - Provide intelligence or aid diagnosis of a malfunctioning space vehicle
    - Employ sensor suite to characterize vehicle
  - Monitor the natural or induced local environment in the spacecraft's vicinity
    - Assess potential contamination issues
    - Monitor threats from debris
    - Detect presence of other vehicles

AEROASTRO Page 8

SCOUT is not a spacecraft; SCOUT is an architecture. It's difficult to meaningfully prove the utility of an architecture without testing it in a real-world scenario. If cost was not a restriction, SCOUT would be tested by building a comprehensive selection of modules and testing them in different combinations as solutions to real-world problems. In the constrained environment of a Phase I SBIR, however, we must demonstrate the utility of the SCOUT architecture by measuring it (on paper) against a single mission scenario.

The mission study conducted in Phase I proposed that SCOUT be evaluated by using it to realize an "Escort" mission. The plan was to design a SCOUT vehicle to satisfy the Escort mission needs using "standard" SCOUT modules (i.e., no special modules custom-fitted to the Escort mission would be proposed).

Escort is a microsatellite that flies in close proximity with another satellite (the "primary") to evaluate it's nature, behavior, or performance. For example, Escort can be used to monitor and analyze electromagnetic emissions radiating from the primary, either intentionally or unintentionally. These emissions can be used to diagnose problems or monitor the operating state of the primary. Similarly, Escort can be used to map out the antenna pattern of a satellite in its as-deployed state. There are limitless applications for such a spacecraft. Escort can be used to independently verify the deployment of a boom, array, or antenna, or diagnose an improperly deployed mechanism. Escort can be used to reconnoiter an unknown satellite, or monitor the local environment with respect to particles, fields, waves, or contamination.

AEROASTRO

Mission Operations

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Concept of Operations Introduction

- Two timelines are presented:

Lifetime activity, launch through decommission

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The lifetime timeline is a coarse overview of the expected major transitions of SCOUT, from launch through end of mission, including potential hibernation stints.

The Day in the Life timeline is a more detailed account of the anticipated daily operations scenario, including both space and related ground activities.

Mission Life Timeline I

- Launch into target (GEO) orbit within 1 km of primary
- Deploy arrays and Detumble
- Transition to Sun Pointing Mode
- Perform ground-based track and range collection
  - Ground-based ranging utilized in non-operational periods
  - Relative ranging utilized during operational periods
- Perform On-Orbit Check Out
- Address Anomalies
- SCOUT declared healthy
  - SCOUT orbit known
  - Primary (target) orbit known
- SCOUT is maneuvered step-wise fashion toward the primary to the preferred orbit, a stable, elliptical orbit about the primary at:
  - 250 meter radial distance;
  - 500 meter in-track distance

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Page 3

The assumptions incorporated into the coarse mission timeline are: that a launch vehicle will be available that can place SCOUT into the desired initial orbit; within 1 km of the primary, and that the best stable orbit is as described in the propulsion section; elliptical, with in track axis twice the length of the radial axis about the primary.

Once released from the launch vehicle, SCOUT will be detumbled and the arrays deployed, then the ADCS mode will transition to sun pointing. Sun pointing is expected to be the mode most utilized by SCOUT.

SCOUT will be stepped through its on-orbit check-out with ground ranging being performed through-out, laser ranging being performed upon check-out of the payload. Any detected anomalies will be addressed during check-out.

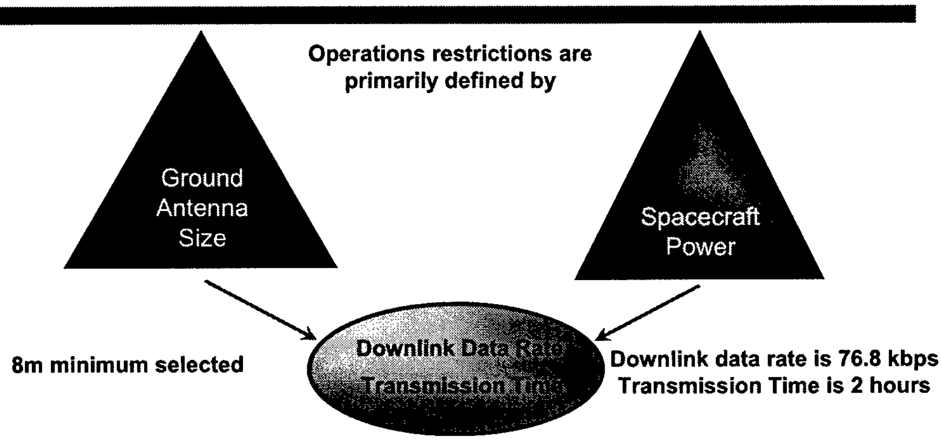
Ground based ranging (absolute with respect to Earth Center) can be performed while SCOUT is in sun-pointing mode but relative (laser) ranging will typically require re-pointing SCOUT to the primary. Once the relative ranging is completed, SCOUT is transitioned back to sun pointing. Relative ranging is the standard ranging method, but in the pre-operational period ground ranging must be performed in order to establish the absolute position with respect to Earth Center. Once the absolute orbit of SCOUT and its primary are well known, the relative ranging data (range between SCOUT and the primary) can be used to update the absolute orbit. It is this adjusted absolute orbit ephemeris that will be used to calculate orbital maneuvers, including the step-in to the preferred orbit about the primary.

Mission Life Timeline II

- ➢Normal Operations Begin
  - Details captured in Day in the Life Timeline
- ➢SCOUT may be placed in Hibernation Mode at any time
  - Slow spin, sun pointed
  - SCOUT powered down except critical loads
  - Once per hour, switch transmitter on:
    - Perform ranging
    - Collect telemetry
- ➢Total mission life is one year including hibernation
- ➢Active mission life is 135 days (based on propellant budget)
- ➢Post-mission, SCOUT is placed in super-synchronous orbit

Hibernation mode - the slow spin provides pointing stability. Should SCOUT be left in hibernation mode for extended periods, the relative position of Sun to SCOUT orbit would change such that it would be necessary to use the propulsion system to realign the SCOUT spin axis to the Sun. Otherwise, hibernation is basically a quiescent mode with minimum thermal control, most loads off except the receiver, flight computer and the transmitter during (ground) ranging only.

Operational Restrictions



These limits define mass storage requirements: 35 Mbytes
Mass storage flows down to payload data collection operations: 1 hour collection

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Spacecraft operations are dominated by power concerns. The main limiter is spacecraft power, though in the case of SCOUT, some power restrictions exist on the ground, as well, in order to keep SCOUT field-operable.

Transmitter power combined with antenna size define the link budget limitations; this, in turn, identifies the maximum downlink data rate achievable. The spacecraft power budget defines how long the transmitter can remain on before the battery depth of discharge exceeds safe limits. Thus, transmitter on-time and data rate define how much data can be downloaded from the spacecraft on a routine basis. This amount of data establishes the data storage requirement (here, 35 Mbytes), which sets the limit on how much data can be collected. Thus, available power ultimately defines payload operations.

Day in the Life Description

- To operate within the identified restrictions, the daily operational cycle has been defined as:
  - Three cycles per day
  - 8 hours per cycle
    - 1 hour of payload activity
    - 5 hours of download and battery recharge
    - 2 hours for maneuver planning, execution, and range data collection
- Delta V budget supports one 100 meter maneuver/day
- When possible, maneuvers will be performed on-node for efficiency, although this is probably not necessary due to the small daily orbital changes anticipated
- Ranging under normal operations is relative (optical) ranging, not ground based (tone) ranging

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The Delta V budget can be broken out as is convenient - for instance, as two 50 meter maneuvers or (for less than 100 meters total) three 20 meter maneuvers.

Day in the Life Timeline I: Space Segment

- SCOUT is slewed from sun pointing to primary pointing
- Relative ranging continues to be performed once per hour (slew to primary as required)
- P/L and S/C health maintenance commands are uploaded
  - Maneuver plans are stored for later execution
- Payload operations are performed for up to one hour
  - Payload data collection is limited to 35 MB; any instrument may be used as desired within that limit.
- Payload is powered down
- SCOUT is slewed to sun pointing
- Transmitter is powered on
- Battery recharge is initiated
- Telemetry and P/L data are downloaded (~ 2 hours)
- Stored maneuver commands are executed

This is a step by step description of the space segment portion of one 8 hour operations cycle.

Note that for relative ranging, SCOUT must first be slewed such that the laser range finder is pointing at the primary and the laser range finder must be switched on; range data is delivered to the on-board storage device.

The payload and spacecraft health and maneuver commands were generated during the previous 8 hour cycle.

The battery recharge is implemented during the download; power subsystem analysis shows this is feasible. Two hours of download (transmitter on time) with battery charging are followed by up to three hours of battery recharge with transmitter off.

Day in the Life Timeline II: Ground Segment

- ➢ Relative ranging data is extracted from telemetry:
  SCOUT orbit is determined
- ➢ Next operation cycle is planned
  - Manevering
  - Health maintenance
    - Ranging/Slewing
    - Battery recharge
  - Payload commands
    - Payload control
    - Transmitter commands
- ➢ Operations plan is converted to command upload and forwarded for vetting and execution
- ➢ Plan is uploaded during the next operations cycle

This is a step by step description of the ground segment portion of one 8 hour operations cycle. Most of these activities occur during the 5 hour battery recharge portion of the space segment operations.

From the space segment operations description, note that telemetry (including ranging data) is downloaded from SCOUT near the end of the previous 8 hour cycle. This ground segment operations description begins at the completion of that download with the extraction of range data from telemetry and the orbit determination.

In the background, the remaining telemetry is reviewed and archived.

The major task of the mission planning software will be to allocate the appropriate amount of payload on-time per payload unit to satisfy data collection requirements within the available on-orbit storage space. Software can automatically calculate the storage allotment per payload on/off sequence and can tally the total to verify the total storage volume is not exceeded.

Once the plan is formatted, checked and forwarded to the control center for upload, separate rules-based checking can be run via the control center software to verify the payload and health commands don't force SCOUT to perform beyond its limits (power, thermal, storage, slew rate, etc). Then the commands can be uploaded to SCOUT at the beginning of the next cycle.

AeroAstro

RF Communications

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The performance and topology of the RF Communications Transponder module is summarized in this presentation.

Requirements

- Uplink
  - Telemetry Types: CMD, Software Upload, 4-tone Ranging
  - Data Rates: At least 2 kbps
  - Protocols: CCSDS Telecommand Packets, COP-1
  - Link Margins: 6 dB at GEO
- Downlink
  - Telemetry Types: P/L TLM, Housekeeping TLM, 4-tone Ranging
  - Data Rates: At least 76.8 kbps
  - Protocols: CCSDS Telemetry Packets
  - Link Margins: 3 dB at GEO

The requirements for the SCOUT Escort Communications subsystem are relatively simple at this conceptual level of the design process.

The uplink must be capable of communicating commands, software uploads, and ranging tones from the ground station to the SCOUT vehicle. The minimum acceptable data rate for this task is 2 kbps. It is assumed that the uplink command protocols would be packetized, most likely using CCSDS standards. If CCSDS is selected, it is assumed that some sort of Command Operations Protocol would be used, probably the COP-1 protocol from CCSDS. Finally, it is expected that the uplink data communications should have a link margin of at least 6 dB in accordance with Good Engineering Practice.

The downlink must be capable of communicating payload or "mission" telemetry, housekeeping telemetry, and return link ranging tones from the SCOUT vehicle to the ground station. The minimum acceptable data rate for this task is 76.8 kbps. It is assumed that the downlink telemetry would be packetized, most likely using CCSDS standards. Finally, it is expected that the downlink data communications should have a link margin of at least 3 dB in accordance with Good Engineering Practices.

Orbit Parameters

Perigee	
hp - Perigee Alt. (km)	35800.00
Rp - Perigee Radius (km)	42178.14
Rp - Perigee Radius (Re)	6.61
Apogee	
ha - Apogee Alt. (km)	35800.00
Ra - Apogee Radius (km)	42178.14
Ra - Apogee Radius (Re)	6.61
Period	
T - Period (sec)	86206.91
T - Period (min)	1436.78
T - Period (hours)	23.95
T - Period (days)	1.00
Orbital Elements	
a - Semi Major Axis (km)	42178.14
e - Eccentricity	0.00
i - Inclination (deg)	0.00
w - Argument of Perigee (deg)	0.00
W - RAAN (deg)	0.00
E - Energy (km^2/s^2)	-4.72520
Ground Station Passes	
Maximum Slant Range (km)	40074.99
Maximum Earth Solid Angle (deg)	66.60
Subsatellite Arc (deg)	16.79808
Minimum ES Look Angle (deg)	15
Constants	
Re - Earth Radius (km)	6378.14
GM or m (km^3/s^2)	398600.50
g (m/s^2)	9.80

- Assumes GEO Orbit
- Assumes minimum Earth Station Look Angle of 15°

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This chart summarizes the orbit parameters for the SCOUT Escort mission, which is expected to operate in Geostationary Earth Orbit (GEO). As shown in this chart, this is approximated as an orbit with an altitude of 35,800 km, an eccentricity of zero, and an inclination of zero.

The key parameter in this chart is the calculated maximum slant range of 40,075 km given an expected minimum Earth Station look angle of 15 degrees. This slant range is carried forward to the link budgets shown on the next two pages.

Uplink Budget

Ground Station Network	USN	USN
Ground Station Location	North Pole, AK	Sturup, Sweden
Ground Station Dish Diameter	13m	8m
Ground Station EIRP (dBW)	68.00	69.00
Signal Frequency (GHz)	2.10	2.10
Path Length (km)	40074.99	40074.99
Path Loss (dB)	-190.94	-190.94
Absorption Losses (dB)	-1.50	-1.50
Polarization Loss (dB)	-0.25	-0.25
Pointing Loss (dB)	-1.00	-1.00
Recieved Power (dBW)	-125.69	-124.69
Recieve Antenna Gain (dB)	0.00	0.00
Feed Losses (dB)	-1.00	-1.00
Filter Loss (dB)	-0.80	-0.80
Received Signal (dBm)	-97.49	-96.49
Required Signal Strength (dBm)	-105.00	-105.00
Link Margin - No Coding (dB)	7.51	8.51

- Universal Space Network Stations at North Pole, AK and Sturup Sweden used:
  - This assumption is made because these stations represent Earth Stations with real performance at S-Band (for uplink) and X-Band (for downlink)
- This shows an uplink margin of at least 7.5 dB (vs. 6 dB need)

AeroAstro

This chart shows a communications Link Budget for the Uplink to the SCOUT Escort vehicle from two different Earth Stations. This link budget does not imply that the Universal Space Network Earth Stations at North Pole, Alaska and Sturup, Sweden would necessarily be used to communicate with SCOUT; these two Earth Stations were selected to provide typical performance numbers for Earth Stations. It is envisioned that SCOUT will use S-Band for uplink communications and X-Band for downlink communications; these two stations both support S-Band Uplink and X-Bank Downlink operations, making them good analogs for an objective system.

Using the stated EIRP of these two Earth Stations at S-Band in combination with the 40,075 km path length leads to a minimum received power of -125.7 dBW at the spacecraft. Combined with the antenna gain and feed loss of the spacecraft, this leads to a minimum input power to the S-Band receiver of -97.5 dBm. This receiver is expected to deliver a Bit Error Rate of 1E-6 at an input power level of -105 dBm, leading to an uplink margin of at least 7.5 dB for this link, which is 1.5 dB in excess of the required 6 dB link margin.

Downlink Budget

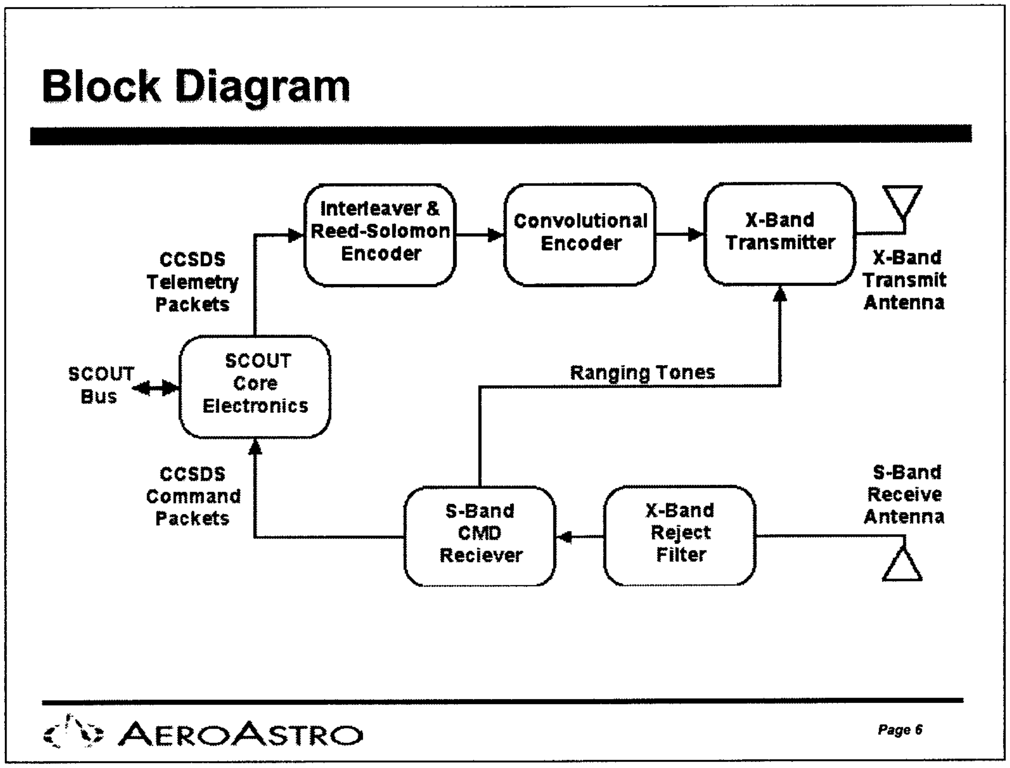
Ground Station Network	USN	USN
Ground Station Location	North Pole, AK	Sturup, Sweden
Ground Station Dish Diameter	13m	8m
Transmitter Output Power (W)	1.50	1.50
Transmitter Output Power (dBW)	1.76	1.76
Feed Losses (dB)	-0.50	-0.50
Transmit Antenna Gain (dB)	0.00	0.00
EIRP (dBW)	1.26	1.26
Signal Frequency (GHz)	8.40	8.40
Path Length (km)	40074.99	40074.99
Path Loss (dB)	-202.98	-202.98
Absorption Losses (dB)	-1.50	-1.50
Polarization Loss (dB)	-0.25	-0.25
Pointing Loss (dB)	-1.00	-1.00
Recieved Power (dBW)	-204.47	-204.47
Recieve System G/T (dB/K)	37.50	32.00
Boltzmann's Constant (dBW/Hz-K)	-228.60	-228.60
Received C/No (dB/Hz)	61.63	56.13
Bit Rate (kbps)	76.80	76.80
Bit Rate (dB-Hz)	48.85	48.85
Recieved Eb/No (dB)	12.77	7.27
Required Eb/No - No Coding (dB)	10.50	10.50
Link Margin - No Coding (dB)	2.27	-3.23
Coding Gain (dB)	7.50	7.50
Required Eb/No - With Coding (dB)	3.00	3.00
Link Margin - With Coding (dB)	9.77	4.27

- Universal Space Network Stations at North Pole, AK and Sturup Sweden used:
  - This assumption is made because these stations represent Earth Stations with real performance at S-Band (for uplink) and X-Band (for downlink)
- Assumes concatenated Rate-1/2 Convolutional plus Reed-Solomon (255,223) (consistent with CCSDS)
- This shows a downlink margin of at least 4.3 dB (vs. 3 dB need)

AeroAstro

This chart shows a communications Link Budget for the Downlink from the SCOUT Escort vehicle to two different Earth Stations. Again, this link budget does not imply that the Universal Space Network Earth Stations at North Pole, Alaska and Sturup, Sweden would necessarily be used to communicate with SCOUT; these two Earth Stations were selected to provide typical performance numbers for Earth Stations. It is envisioned that SCOUT will use S-Band for uplink communications and X-Band for downlink communications; these two stations both support S-Band Uplink and X-Bank Downlink operations, making them good analogs for an objective system.

Using the nominal 1.5 watt output power from the downlink transmitter, combined with the anticipated worst case feed loss and antenna gain, it is expected that the SCOUT Escort vehicle will produce a minimum EIRP of 1.26 dBW. Combining this EIRP with the 40,075 km slant range and 8.40 GHz transmit frequency, the worst case received power is expected to be -204.5 dBW at the Earth Station. Combined with the USN Earth Station G/T performance numbers and a data rate of 76.8 kbps, this yields a minimum Eb/No of 7.3 dB. For a link that uses no coding, 10.5 dB is required to achieve the necessary 1E-6 Bit Error Rate. This means that coding must be used to achieve the desired Bit Error Rate. A rate-1/2 constraint-length K=7 convolutional encoder used concurrently with an interleaved Reed-Solomon (255,223) encoder produces a coding gain of 7.5 dB. If implemented on this link, this coding brings the anticipated worst-case link margin up to 4.3 dB, which is 1.3 dB in excess of the required 3 dB link margin.

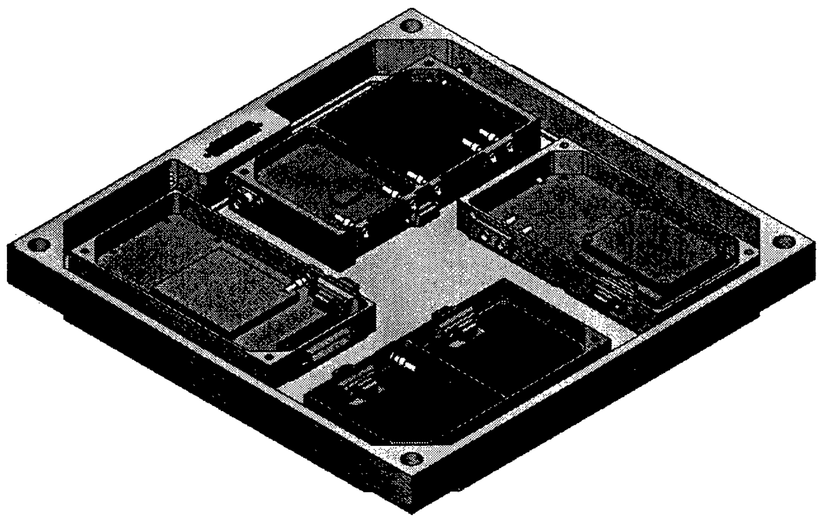


This graphic depicts a block diagram of the SCOUT Communications Transponder module used for the SCOUT Escort Vehicle. The S-Band uplink circuitry is shown in red, the X-Band downlink circuitry is shown in blue, and the common uplink/downlink electronics are shown in green.

The S-Band uplink signal is received by a tuned S-band patch antenna and then (probably) filtered by a small, low-loss X-band reject filter. This filter is intended to keep coupled energy from the X-Band downlink out of the uplink circuitry where it could possibly saturate the Low Noise Amplifier in the receiver Front End. It is likely that this filter will not be needed. This command receiver demodulates the uplink command packets and forwards the packets to the SCOUT Core Electronics Block, which transmits them via the SCOUT Backbone bus to the rest of the spacecraft electronics. The reciever also recovers the ranging tones and sends them over to the downlink transmitter.

The X-Band downlink data is sent to the communications transponder module over the high-speed Ethernet bus in the SCOUT Backbone. This data is forwarded to the downlink circuitry where it is interleaved, Reed-Solomon encoded, Convolutionally encoded, combined with the ranging tone subcarrier, and then modulated onto an X-Band carrier. The modulated carrier is transmitted at 1.5 watts RF power from a tuned X-Band patch antenna.

Module Configuration



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This graphic is intended to demonstrate the feasibility of packaging the S-Band/X-Band transponder into a 2-cm high SCOUT module. The enclosures shown within this module are actual-size models of the transmitter, receiver, oscillator, and diplexer (filter) used in the AeroAstro NMRF X-Band transponder being developed for NASA's Goddard Space Flight Center. From this CAD model, it is apparent that packaging the S-Band/X-Band transponder into a 2-cm high SCOUT module should be relatively straightforward. The rectangles on the side of the module represent the two "patch" antennas that would be used to transmit and receive the electromagnetic signals.

S-Band Uplink Receiver

Receiver Features:

- The receiver is programmable over the S-Band range (2025-2120 MHz)
- The analog transponder channel enables non-coherent tone ranging
- Extremely small and low mass
- Intended for use with CCSDS-compatible ground stations
- Utilizes COTS technology for maximum performance
- Includes Latch-up Detection and Mitigation circuitry (LDM)
- No Diplexer Required
- RX frequency set by embedded micro-controllers, eliminating need for customized crystals

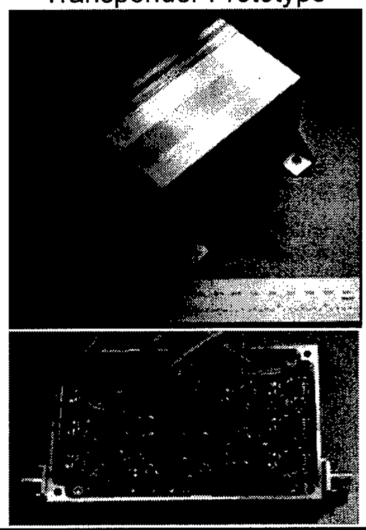
Receiver Specifications:

Modulation Type:	BPSK
DC Input Power:	< 4W
Freq. Range:	2025 - 2120 MHz
Data Rate:	1, 2, or 4 kbps
Sensitivity: (BER=1e-6) @ 2 kbps	-105 dBm
Lockup time:	< 5 seconds
Noise Figure:	< 9dB (with TX on)
Ranging:	4-Tone, non-coherent
Data Interface:	RS-422
Clock Interface:	RS-422
RSSI telemetry:	0-5V analog

This chart summarizes the anticipated characteristics of the S-Band uplink receiver. This conceptual design is derived from another AeroAstro design and is considered relatively low-risk. The Bit Error Rate performance, Sensitivity, and Noise Figure are considered challenging, but not stressing. The RS-422 Data and Clock interfaces would easily communicate with the SCOUT Common Electronics Block, as would the analog telemetry signals.

X-Band Downlink Transmitter

Transponder Prototype



Based on an AeroAstro Transponder developed for NASA; the first three X-Band Transponders will be flown on the NASA New Millennium Program's ST5 mission

Frequencies: 8.40 to 8.50 GHz

Modulation Method: BPSK or QPSK

Data Rates: 10 to 750 kbps

RF Output: 1.5 W transmit power

Power Consumption: 15 W transmit

Volume: 6 cm × 9 cm x 1 cm

Mass: 0.200 kg

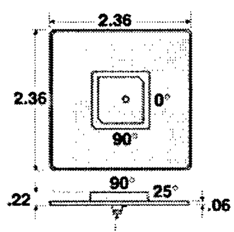
Temperature: -20 to 40°C operational, -40 to 50°C survival

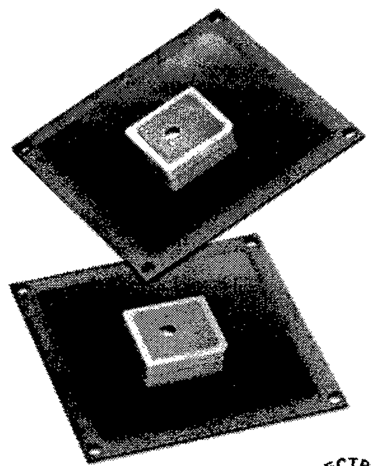
This is a photo of the existing NMRF X-Band Downlink Transmitter. This design can be directly adapted to SCOUT with very little difficulty.

Patch Antennas

Representative Antenna Characteristics:
- Center frequency: 1615 MHz ± 5 MHz
- Left hand circular polarization
- VSWR: 2:1 max (25 MHz bandwidth)
- Gain at zenith: 3 dBic min
- Axial Ratio at zenith: 3 dB max
- Omnidirectional in azimuth
- Half-power beamwidth: 100°
- Directivity variation <5 dB

Dimensions for L-Band Patch Antenna shown at right





Patch antennas have been used on spacecraft for years; the only challenge for these S- and X-band patch antennas would be their relatively small size. The antennas shown above were developed for AeroAstro by Spectrum Microwave and represent a very small size for an L-Band patch antenna. Note that most (approximately 2/3) of the 2.36 inch antenna shown above is the ground plane surrounding the antenna proper. This makes the antenna itself about 0.8 inches, or 2 cm wide. Using this same technology, an S- and X-Band antenna would be even smaller. It is not considered problematic to develop these higher-frequency antennas to fit on the side of a 2 cm high SCOUT module, assuming that the module itself (which is aluminum) can act as the ground plane for these antennas. Again, this is considered a relatively low-risk development.

RF Communications Module Power

- Based upon power consumption for the NMRF Engineering Model, the SCOUT RF Comm Module can be expected to exhibit approximately the following power consumption:
  - Receiver Only: 2.3 to 3 watts (at 6.5 to 8.4 VDC, regulated)
  - Transmitter Only (No PA): 2.3 to 3 watts (at 6.5 to 8.4 VDC)
  - PA Only: 9 watts (at 2W RF output power - capable of 4W RF)
  - DC/DC Converter Efficiency: 80%
- Total Power Consumption:
  - Receiver: 3.75 watts
  - Transmitter: 11.25 watts
    - Power while transmitting: 15 watts
- The power reserved in the budget is very conservative

AEROASTRO

Finally, This chart makes a comment on the power budget used for the SCOUT Escort Vehicle. Referring to the power budget in section 08, it is seen that this budget assumes a communications transponder module that consumes 49.2 watts. This power estimate is based on an early power estimate using an older-technology transmitter. The NMRF X-Band Transponder Engineering Model uses 3 watts of power for its receiver and 9 watts of power for its transmitter (while transmitting 2 watts RF). Using an 80% efficient DC/DC converter, the actual power consumption would be 3.75 and 11.25 watts for the receiver and transmitter, respectively. This adds up to a combined SCOUT communications transponder module power consumption of 15 watts, about 30% of the 49.2 watts carried in the power budget. This savings translates to an orbit average power savings of 8.2 watts, however, as a 24% duty cycle is assumed for the transmitter.

AeroAstro

Propulsion

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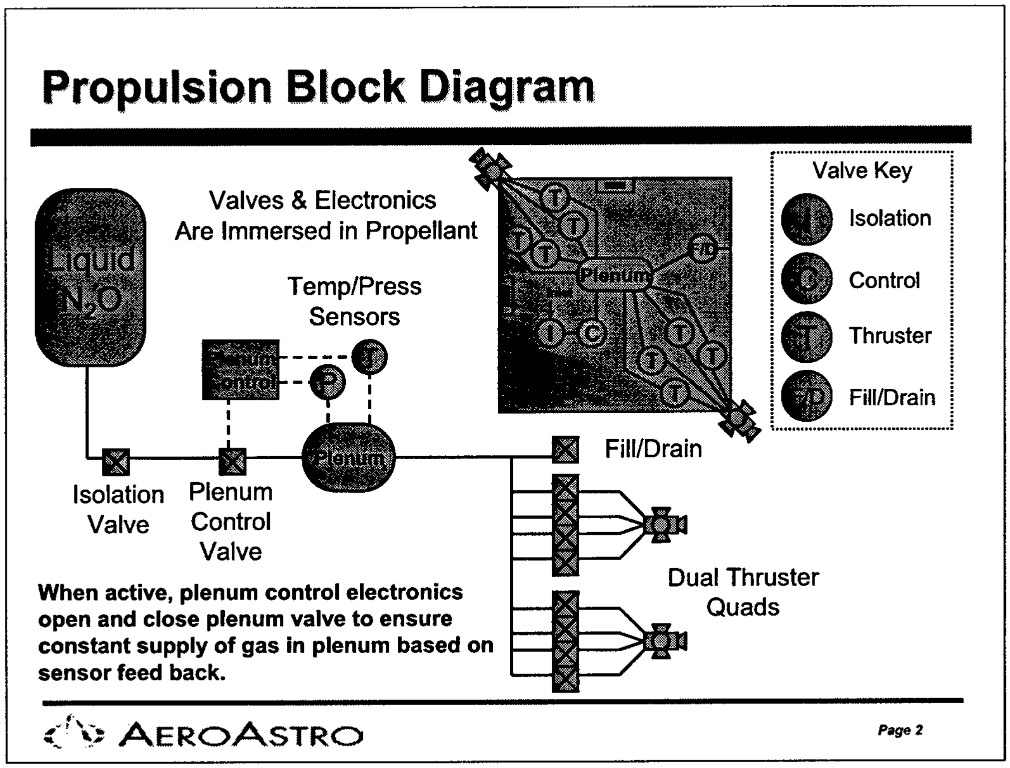
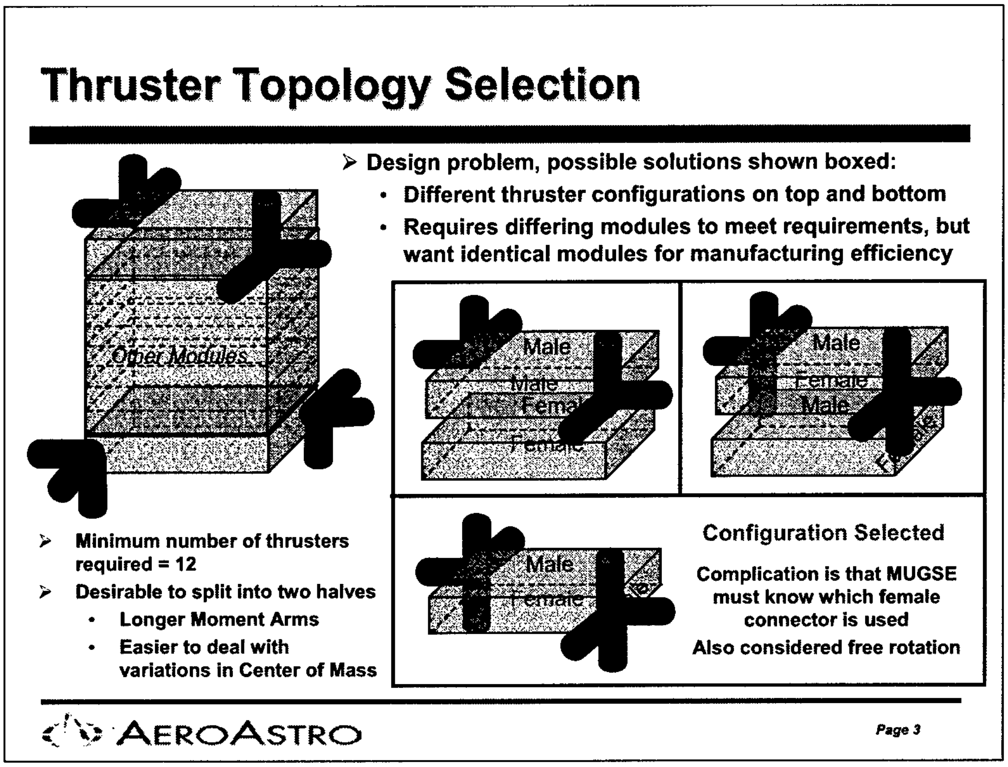


We begin the discussion of the SCOUT Escort propulsion system with a functional block diagram.

N2O propellant enters a micro plenum chamber within the tank through a valve. The pressure and temperature within the plenum is controlled through a closed feedback loop such that all the propellant inside it transforms to gaseous state. MEMS (Micro Electro-Mechanical Systems) valves are used thourghout the system – they are immersed in the propellant along with an electronics driver board on which the propellant feed system is also built. The feed system channels propellant from the plenum to individual hot gas thrusters which are arranged in two quads.

Several other thermodynamic configurations for N2O have been under investigation by AeroAstro, these include cold gas, cold gas with a vaporizer, and warm gas. These alternative configurations were not selected by AeroAstro for further development because of their reduced performance and mostly equivalent risk.



To provide a high level of inspection capability Escort needs to be highly maneuverable in both attitude and translation. In order to provide fully uncoupled six-degree-of-freedom control, at a minimum, a classic twelve-thruster topology should be used. At first it was desired to keep the tanks and all 12 thrusters in the same module for integration ease. However, to keep the thruster moment arms longer, and to provide more flexibility in the center of mass location when stacking modules for different types of SCOUT satellites, it was decided to place the thrusters on separate modules at opposite ends of the stack. In order to avoid the complexities of feeding propellant through different modules it was decided that the thruster module on each end of the stack should be equipped with its own fuel tank and would not share propellant with the other module. An external tube between the two modules was contemplated and not ruled out, but it could lead to undesirable thermal pumping of the fuel.

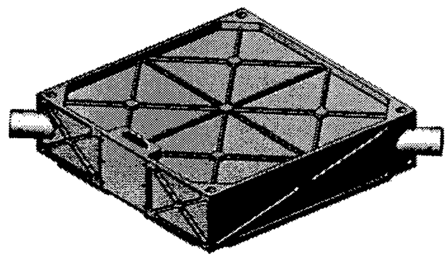
The thruster-clusters at the two ends of the stack need to have somewhat different topologies, in order to meet the requirement for different topologies but also keep the two modules identical, the three different configurations shown were contemplated. The selected configuration was chosen because it did not involve the use of a separate module for rotating or flipping the propulsion module at one end of the stack. The one complexity of the selected configuration is that the MUGSE needs to make sure that the propulsion modules were oriented correctly in stacking, since it is possible to orient them two different ways. A fool proof method of doing this was not settled upon, the current baseline recommendation is place the onus on the human in the loop.

Propellant Selection

N2O Selected

	Dry Mass	Propellant Mass	Thruster Size	Tank Size	Cost
2 Cold Gas	Low	High	Small	Large	Low
2H4 Monoprop	High	Low	Large	Small	High
PT	Low	Very Low	Large	Large	High
2O Hot Gas	Low	Medium	Small	Small	Low

- **SCOUT Implementation**
  - **Low pressure (800 psia) allows non-conventional tanks; fuel inside titanium module**



- **AeroAstro N_2O Propulsion Technology**
  - **Currently under Phase I SBIR from NASA JSC (VACCO is subcontractor)**
  - **N_2O is non-toxic, low-cost, storable**
  - **Isp = 120 to 200 (120 is assumed)**
  - **Based on MEPSI Technology, uses MEMS valves, immersed electronics and integrated feed system**
  - **Propellant is stored as a liquid; N_2O vapor is pressurant - no diaphragm**
  - **Gaseous vapor stored in micro plenum chambers, can enter as gas or liquid**
  - **Pressure drop into plenum so that N_2O inside is gas; process controlled using feedback from plenum P & T sensors**
  - **Ceramic thruster body with titanium shell welded to conformal tank**

AEROASTRO

A propellant trade study was conducted, AeroAstro's N2O technology was selected.

Together with VACCO, AeroAstro has been developing Nitrous Oxide (N2O) hot gas monopropellant propulsion technology for small satellites. There are numerous advantages to this propulsion method:

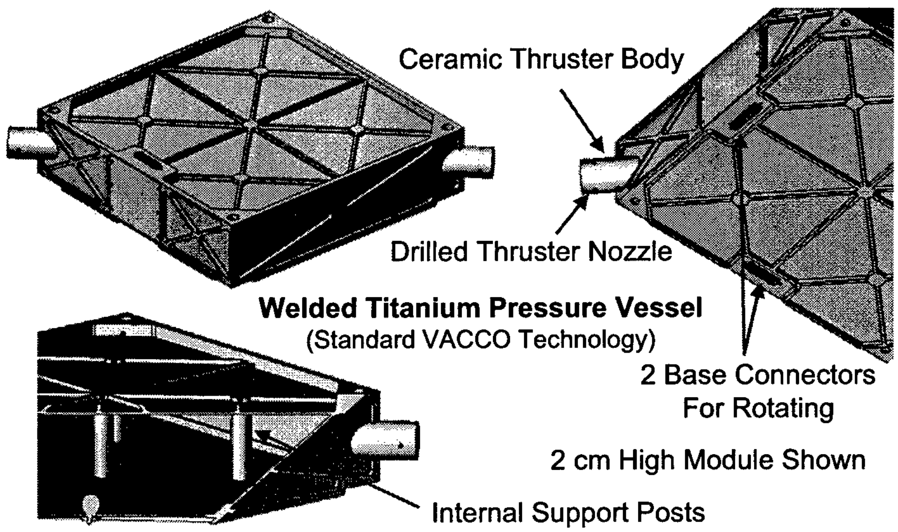
The low propellant storage pressure (~800 psia) allows non-conventional shaped tanks that can fill up more of the available envelope volume than would be possible using a spherical or cylindrical tank, as is required for more common high pressure cold gas or hydrazine systems.

Nitrous Oxide is non-toxic, low-cost, and storable.

The specific impulse (Isp) of at least 120 seconds is considerably higher than cold gas, and may be as high as 200 seconds.

For volumetric efficiency the propellant is stored within the tank as a liquid that is pressurized by its own vapor – no diaphragm is needed.

Proximity Maneuvering Propulsion Module



The diagrams above show some mechanical features of the maneuvering propulsion module. Unlike other modules that are not pressurized, this one is a welded titanium pressure vessel. Titanium is a standard working material commonly used by VACCO and it should pose no problem for them to manufacture. In order to provide extra structural stiffness there are several internal support posts within the module that effectively prevent it from bulging outwards when under pressure. The thrusters quads are cylindrical ceramic blocks with precisely machined combustion chambers and nozzles drilled into them. Because the combustion temperature of N2O is extremely high some titanium may be used around the nozzles for additional heat shielding.

The two connectors on one side of the module are clearly visible. A typical module should only have one such connector but the Maneuvering Propulsion module has two because it needs to be stacked in different orientations at different ends of the stack.

ΔV and Propellant Budgets

- Additional Major Assumptions
  - Thrust: 0.1 N (Based on Chamber)
    - Independent of Isp or Cold vs. Hot
    - Could be reduced if need to tweak mission
  - Specific Impulse: 120 sec (Worst Case)
    - Could be up to 160 sec (Cold N_2O = 60 sec)
  - Orbit Limit Cycle: 2560 sec (10 × ADCS)
  - Orbit Limit Cycle ΔV Margin: 25%
  - Propellant Mass Margin: 30%

Mass Source	Mass (kg)
Dry Structure (25% margin)	60
Impulse (Attitude) Propellant	4.135
ΔV Propellant	2.00
Margin Propellant	1.84
Pressurant Propellant	0.42
Total	68.4

ΔV Source	ΔV (m/s)
Initial Separation	0.0073
Radial & In-Track Maneuvering	0.4345
Cross-Track Maneuvering	0.870
North-South Station Keeping	15.0
East-West Station Keeping	0.6
Orbit Limit Cycle	3.045
Disposal to Super-GEO	10
Thruster Offset Disturbance	6.23
Total	**36.2**

- One 100 m Amplitude Radial and Cross-Track Maneuver Per Day
- 4 cm Effective Thruster Offset
  - Center of Mass Uncertainty
  - Thruster Offset Uncertainty
  - Thruster Misalignment Uncertainty
  - Thrust Balancing Uncertainty

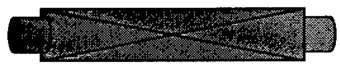
AEROASTRO

The thrust in a N2O thruster is set by adjusting the chamber geometry; a 0.1 N thurst level was selected. The minimum on-time was selected to be long enough for the gas to fully ignite, thereby allowing a higher Isp. The resulting propellant efficiency was found to be greater than what could be achieved by reducing the on-time and accepting a lower Isp closer to that of cold gas, due to incomplete combustion

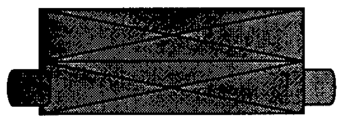
A detailed ΔV budget was assembled that included the entries from the table on the right. The maneuvering (relative to the primary) budget allowed for a single 100 meter amplitude radial maneuver per day and also a single 100 meter amplitude cross-track maneuver per day. The station-keeping budgets were conservative and assumed unusually tight orbit control. The thruster offset was conservatively large to also include the effects of center of mass uncertainty, thruster misalignment uncertainty and thrust balancing uncertainty. Some propellant margin was allocated specifically to the orbit limit cycle ΔV because that was deemed to be a particularly uncertain estimate.

The overall propellant budget was assembled in the table on the left, it includes propellant used for attitude control described in the previous section as well as propellant used for ΔV. Propellant margin was allocated specifically to both attitude control and ΔV at the same time in the form of propellant mass margin. The margined amount of propellant was sized to completely fill up two 4 cm high modules, the resulting mission duration was 119 days.

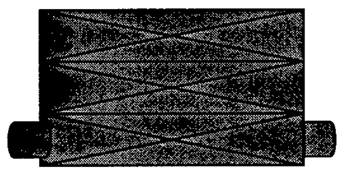
Propellant Trends



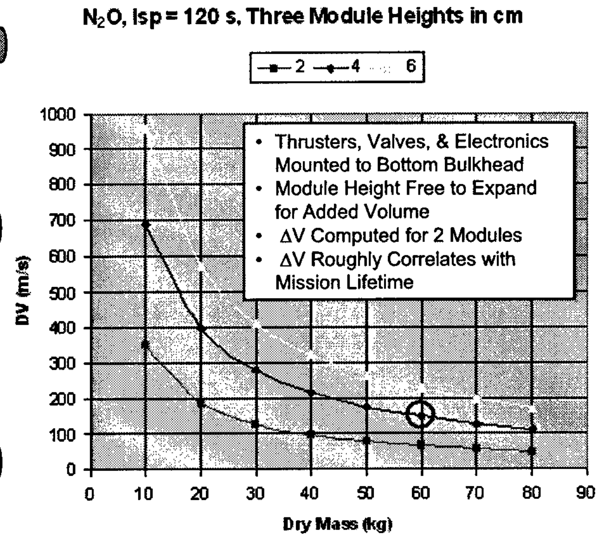
2 cm High Module, 1.71 kg N_2O



4 cm High Module, 3.99 kg N_2O
Selected



6 cm High Module, 6.27 kg N_2O



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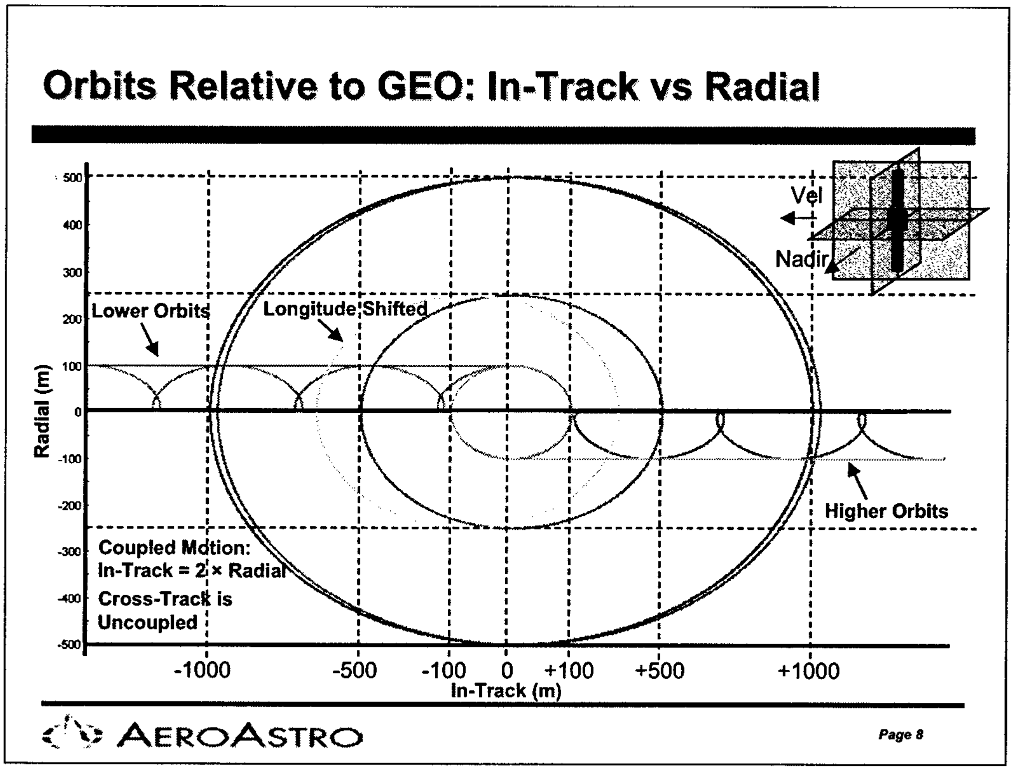
This slide shows the concept of modularity extended to the Maneuvering Propulsion module itself, in that it can be produced in three different yet very similar sizes based on mission requirements. To increase the size the side-wall height and internal support post height are merely duplicated one or two extra times. In each case the immersed board with integrated valves, feed system and electronics is mounted to the bottom bulkhead.

For illustrative purposes the graph shows the available ΔV from two modules as a function of total SCOUT dry mass for each of the three different propulsion module sizes. The green circle shows where on the graph the baseline SCOUT Escort configuration lies and what the effects on performance would be for changes in dry mass and module size. The ΔV shown on the vertical axis is roughly directly correlated with mission duration.

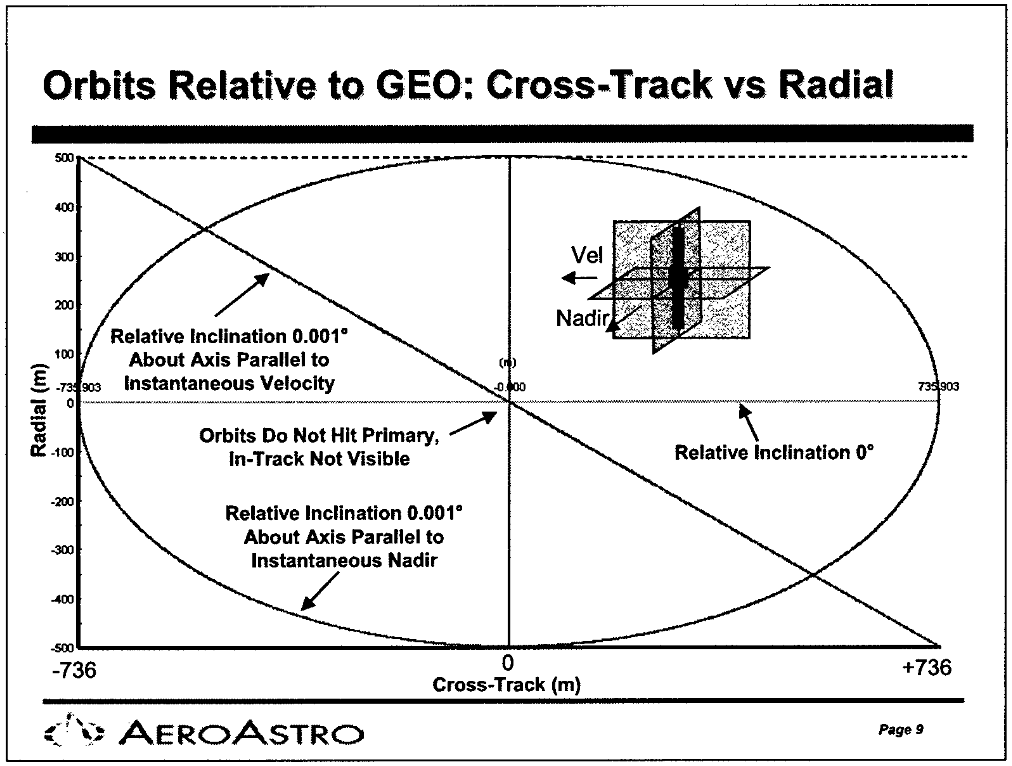


The graph above shows the projection of the Escort's orbit relative to the primary in the primary's in-track and radial directions. The orientation of this plane as it would be seen by an observer on the ground looking at a GEO satellite overhead is shown in the diagram on the top-right.

To minimize propellant usage, Escort spends most of its time passively orbiting its primary satellite. Alternatively, Escort can also lead or trail the primary satellite separated by a fixed true anomaly. The primary satellite is assumed to be in a circular GEO orbit and Escort is slightly offset from that orbit.

A slight true anomaly offset will cause Escort to lead or trail the primary. A slight apogee and perigee (or semi-major axis and eccentricity) offset will cause Escort to orbit relative to the primary in the radial and in-track directions.

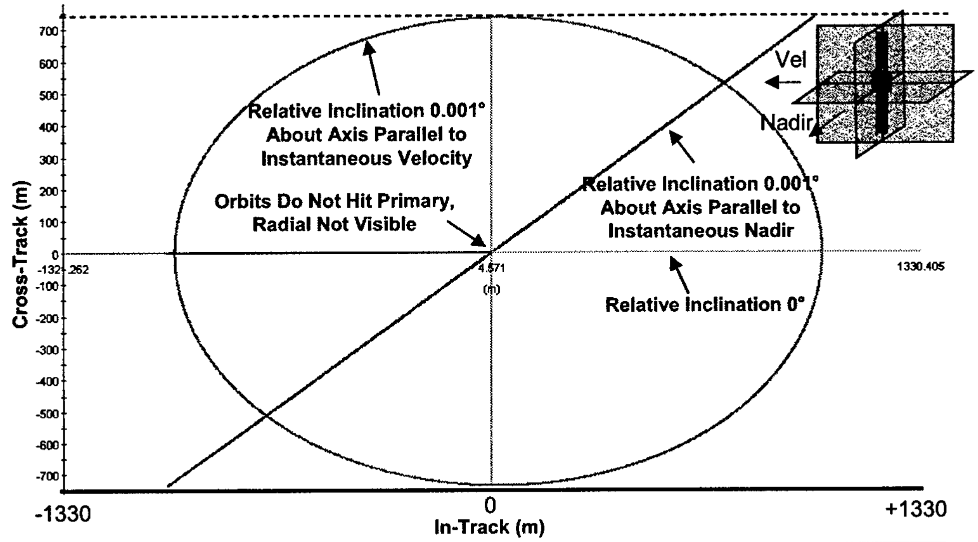
The dynamics of orbits relative to a primary satellite are somewhat counter-intuitive. The radial and in-track motions are inexorably coupled: the in-track radius of the relative orbit will always be twice the radial radius. The cross-track radius is uncoupled and can be independently controlled. Another counter-intuitive point is that the smaller the radius of the Escort's orbit relative to the primary is, the slower the Escort moves relative to the primary. This has advantages from the standpoints that it is safer to move slower when the Escort is closer to the primary, and also that fatures on the primary will still be moving passing through the Escort FOV slowly enough to be inspected.



A slight inclination offset will cause Escort to orbit relative to the primary in the cross-track direction.

Based on preliminary discussion with stakeholders, the minimum distance between the center of Escort and the center of the primary is expected to be on the order of 100 meters. The maximum distance is expected to be on the order of 1,500 meters. The typical maximum length dimension of a large primary is on the order of 50 meters. The typical smallest feature size of interest is on the order of 1 centimeter. The simple imager payload baseline is capable of meeting both the 50 meter and 1 centimeter requirements within the specified operational distances to the target.

Orbits Relative to GEO: In-Track vs Cross Track



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This chart simply shows a merged projection of selected data that was also shown on the previous two charts.

It should be noted that these charts have only shown Keplerian orbits. Non-Keplerian trajectories can also be achieved at the expense of using more propellant and therefore having a shorter active mission duration. A desirable non-keplerian trajectory might be to station-keep at a fixed point relative to the primary with some separation in the radial or cross-track dimension, the further away this point is radially the more propellant it will take. Another good idea might be to fly a sweeping “raster scan” pattern through the main beam of the primary satellite’s communications payload, this would be useful for rapid antenna gain geometry mapping using the RF Probe.

AEROASTRO

Attitude Determination and Control

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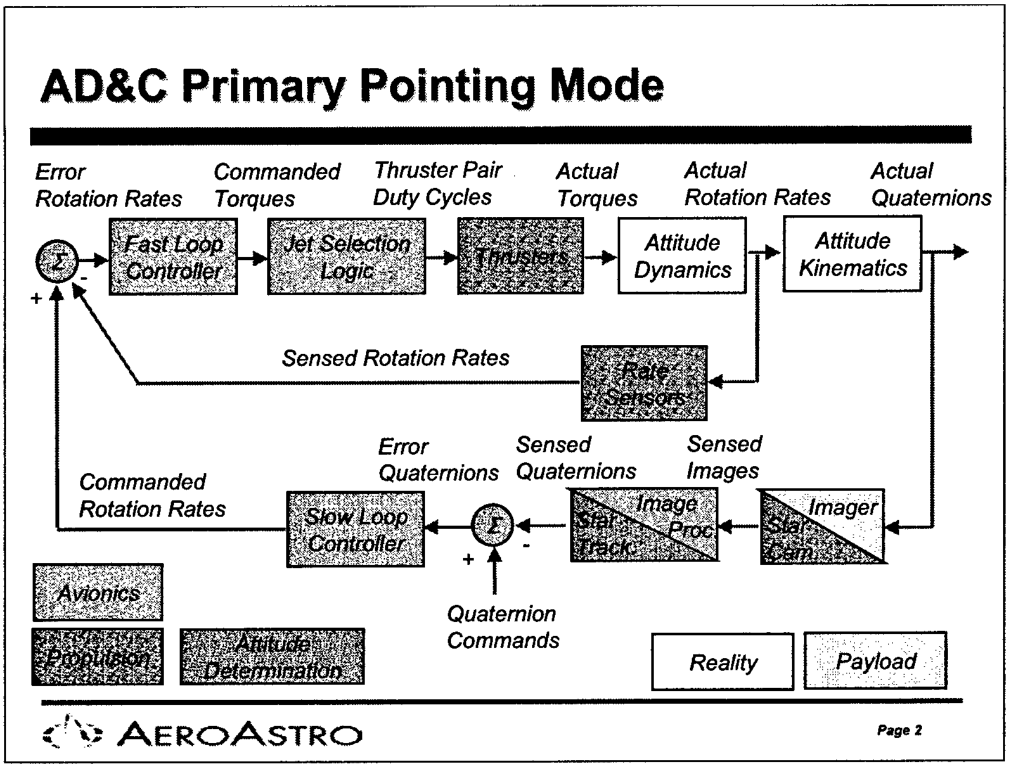


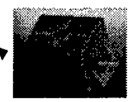
We begin the discussion of the ADCS with a functional block diagram. The control loop type for the Primary Pointing and Primary Acquisition modes is a classical double nested single input vector single output vector control system. The main difference between the two modes is the outer loop sensor, which is the imager in the Pointing mode and the star tracker in the Acquisition mode. The inner loop uses information from the rate sensors and runs at a higher frequency than the outer loop. In a high level sense this control loop is similar to what is used in AeroAstro's STPSat-1 satellite being built for the US Air Force Space Test Program. The fast loop controller would be a traditional bang-bang nonlinear controller using a Schmitt trigger in a feedback loop with a lag. In Maneuver mode during larger translational propulsive maneuvers the optimal jet selection logic can be based on that which was developed for AeroAstro's SPORT Small Payload Orbit Transfer Vehicle. Other control modes would have different architectures, for example, the Detumble and Maneuver modes might only have a single loop using the rate sensors.

The color coding on the blocks indicates which module the functional element resides in. The SCOUT avionics architecture makes integrating different modules very easy. At the time of integration the MUGSE analyzes the module stack up and adjusts settings such as gains and reference vectors in the ADCS software based on the computed moments of inertia, fuel tank locations and loading, and also thruster and sensor topologies.

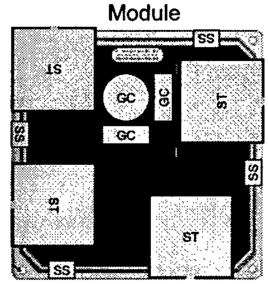
Attitude Sensor Selection

- ➢ Robust Design Meets Requirements for Multiple Frames
  - Multiple ST FOV prevents Sun, Moon, Earth induced loss of lock
  - Insensitive to orbit selection
  - Allows for autonomous initial acquisition and Lost in Space mode
  - Sun and rate sensors allow easy autonomous detumble and power-safe modes
- ➢ Potential Component Vendors
  - AeroAstro Medium Sun Sensors
  - AeroAstro Miniature Star Trackers
  - Systron Donner BEI Gyrochips (Rate Sensors)

Sensor	Axes/Frame	Quantity
Sun	2/Sun	4
Rate	1/Body	3
Star	3/Inertial	4 FOV, 2 CPU
Imaging	3/Primary	1 (PL Module)



Attitude Determination Module



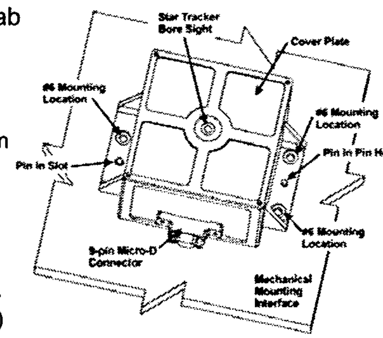
The Escort mission demands a variety of attitude determination sensors:
Star tracker, for when the primary satellite is temporarily out of view
Sun sensor, for when the Escort is in hibernation or battery charging mode
Imager, for attitude determination relative to the primary satellite
Body rate sensors, for when the Escort needs to detumble and to propgate the attitude between environmental sensor updates

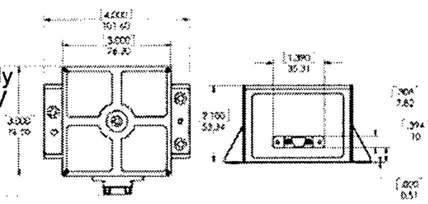
Without advanced technologies such as AeroAstro's coarse star tracker, AeroAstro's medium sun sensors, or MEMS rate sensors produced by a variety of other manufacturers, a sufficiently compact yet capable attitude determination suite would not be possible. It is also fortunate that the imager which can be used for attitude determination serves a dual purpose as a payoad sensor.

As many as four coarse star tracker apertures and two processors should be used to prevent sun, Earth, moon or primary-induced loss of lock. The attitude sensor selection is insensitive to orbit selection, making it usefull for both LEO and GEO, and it is also capable of recovering from a lost-in-space mode.

AeroAstro Miniature Star Tracker

- MDA Phase 1 STTR w/ MIT Space Systems Lab
  - All Information Still Very Preliminary
  - Phase 2 Proposal In Process
- Requirement Goals
  - Mass & Volume: 300 g, 5.1 X 7.6 X 7.6 cm
  - Power & Voltage: < 1 W, 4.5 - 5.5 VDC
  - S/M/E Exclusion: 50° Full Angle Cone
  - Accuracy (3-axes, 3σ): < 300 arc-sec (0.083°)
  - Max Roll Rate: 0.3 deg/sec
  - Autonomy: Acq., Track, Process Att.
  - Update Rate: 1 Hz (increase w/ power)
  - Cost: < ~$100k
- Technologies
  - Pin-Hole Aperture, No Glass, No Baffle, Greatly Reduced Calibration, 25° Full Angle Cone FOV
  - CMOS Imager, No CCD, Greatly Reduced Electronics Mass, Volume and Power
  - <= 4 Pairs of <= 4th Mag. Stars





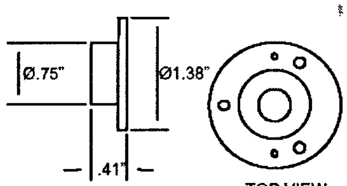
AeroAstro

Page 4

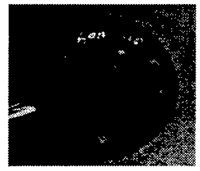
Low cost space vehicles require an affordable, low system-impact sensor for attitude determination. This is especially true when a high degree of navigation is required for high delta-v maneuvers in applications such as on-orbit interceptors. General industry research is focused on increasing accuracy to the exclusion of all else. It focuses on rapid pattern recognition, widening fields of view to process many stars and star patterns simultaneously, and expanding internal star catalogs to well over 20,000 stars ranging from magnitude 0 to magnitude 9. These advances all result in more massive and complex systems. This is contrary to the clear requirement for *decreased* complexity, mass, and power consumption to fit the needs of small, highly maneuverable, low-cost vehicles. AeroAstro and MIT are jointly developing a highly innovative electronic device to solve this challenge: a low-impact star tracker, balancing accuracy with power consumption, mass, and cost. No solution currently exists in this area of the trade space. The design uses a pinhole in place of traditional optics, takes advantage of the processing capabilities embedded in newer CMOS imagers to reduce power consumption and limit the amount of glue logic required, and exploits highly compact pattern recognition algorithms to find star pairs using a minimal star catalog. The solution offers accuracy better than 100 arc-seconds, meeting the unique requirements of small maneuverable space vehicles at a fraction of the cost, mass, and power consumption of larger, higher impact star trackers.

AeroAstro Medium Sun Sensor

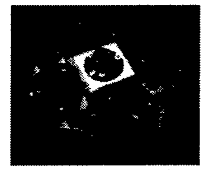
- Heritage: ALEXIS, HETE, CHIPSat, soon MOST
- Detector: Masked Quadrant Photo-Diode
- Output: 2 Axes in Sun Frame
- Calibration Table or Function Stored in Spacecraft Avionics
- FOV: 134° Full Angle Conical
- Accuracy: ~1°, 3σ (Higher Available)
- Mass: 36 g (w/ Harness)
- Power: None
- Cost: < $50 k for two



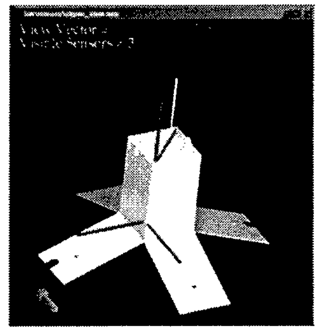
SIDE VIEW   TOP VIEW



Single MSS



4 MSS, 1 CSS



AeroAstro FOV Analysis Software

AEROASTRO

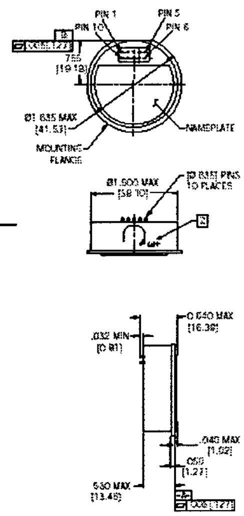
AeroAstro's Medium Sun Sensor is a low-cost, medium resolution sun sensor that was originally designed and built by AeroAstro for the Los Alamos National Laboratory's ALEXIS satellite and has since been used on several others too. The sun sensor is comprised of a small housing containing a set of four photodiodes. The sensor provides four signal outputs and one ground wire. The satellite's computer must store a small table of calibration data and a small interpolation subroutine to compute the sun angle based on four signal outputs.

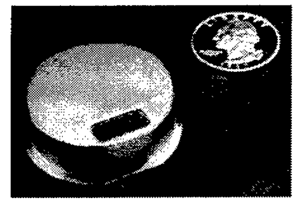
The housing used for the Medium Sun Sensor also serves as the aperture. The accurately machined, integrated aperture reduces mechanical complexity, while serving to precisely mask solar light to provide the desired field of view and illumination properties. The robust aperture also effectively protects the photodiodes during integration and testing. A specially designed mask limits the entry of stray light into the sensor.

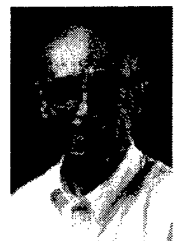
AeroAstro has developed custom light geometry processing and visualization software for determining optimum placement of sun sensors on spacecraft. This is especially useful for AeroAstro's Coarse Sun Sensor that produces attitude output based only on sun intensity measured at different locations on the body. More sophisticated versions of this software are used for solar panel string layout design and power output analysis, as well as for computing solar pressure and aerodynamic disturbance torques.

Systron Donner BEI GyroChip QRS11

- Heritage: Numerous Aircraft, Missile and Space Systems
- Detector: Solid State MEMS Piezoelectric Quartz Tuning Fork
- Output: 1 Axis Rate in Body Frame
- Threshold/Resolution: < 0.004°/sec
  - Not Used for Integration
- Noise: < 0.01°/sec/√Hz
  - 0 to 100 Hz
- Saturation: ±50°/sec
  - Internal Anti Aliasing Filter
- Mass: ≤ 60 g
- Power: 0.8 W
  - ±5 V
- Cost: < $3000







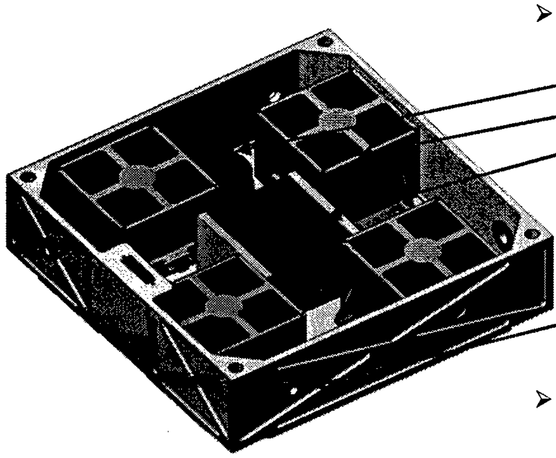
Professor DeBra
Stanford University

The BEI GyroChip™ Model QRS11 is a "MEMS" technology, solid-state "gyro on a chip." This DC input/high-level DC output device is fully self contained, extremely small and lightweight. No external support electronics are required. Since the inertial sensing element is comprised of just one micromachined piece of crystalline quartz (no moving parts), it has a virtually "unlimited" life. The Model QRS11 is a mature product in high volume production. It is fully qualified and used on numerous advanced aircraft, missile, and space systems.

The BEI GyroChip™ Model QRS11 utilizes a one piece, micromachined, vibrating quartz tuning fork sensing element. Applying the Coriolis effect, a rotational motion about the sensor's input axis produces a DC voltage output proportional to the rate of rotation. Use of piezoelectric quartz material simplifies the active element resulting in exceptional stability over temperature and product life. (Above two paragraphs from product data sheet)

This gyro technology was initially conceptualized by a Swiss watch company that was using it for precision oscillators. The watch company later sold the rights to its use in gyros to Systron Donner. After further development Systron Donner determined that the technology would require significant further development to be useful. They retained three expert engineers, one each in the fields of dynamics, electrical engineering, and software to get the gyros working. Professor Dan DeBra of Stanford was the dynamics engineer, he is also a member of AeroAstro's board of directors.

Attitude Determination Module



- Contains All Attitude Determination Components
  - 3 GyroChips
  - 4 Miniature Star Cameras
  - Electronics
    - 2 Star Tracking Processors
    - 4 MSS Calibration Tables
    - Filtering
    - SCOUT Core Electronics Block
  - 4 Medium Sun Sensors
- 6 cm Tall Module

This slide shows a conceptual packaging of the attitude determination module. Each of the AeroAstro Miniature Star Trackers is shown in its current nominal configuration which includes both aperture and all processing electronics, and would actually have the bore-sight pointed out the top. Although they would not be functional as shown, the slide shows that volumetrically they would all fit in. The appropriate engineers have determined that it would be possible to re-package each individual star tracker to point the bore-sight out the side and then also still keep the height below the 6 cm standard module height. This is especially feasible if the processing electronics are offloaded to the centralized board stack.

This module configuration is slightly different compared to what was presented in the SCOUT Technology Plan. The main difference is that the magnetometer was removed. This was done because AeroAstro decided that magnetometers are a poor choice for attitude determination when more accurate sensors are already available within the same module. Instead, the magnetometer should be included with what was previously the magnetic actuation module for use in compensating for the unknown variability of Earth's magnetic field when commanding the magnetic actuators.

Attitude Actuator Selection

	(Thrusters)	(Honeywell Micro Wheels)		Selected
Figure of Merit	Spin Stabilized	Momentum Bias Wheels	Reaction Wheels	Bang-Bang Thrusters
Imager Complexity	High	Low	Low	Low
Pointing Stability	Moderate	Moderate	High	Low
Slew Capability	Low	Moderate	High	High
Power Draw	Low	Moderate	High	Low
Mass	Low	Moderate	High	Low
AD&C Complexity	Low	Moderate	High	Low
Cost	Low	Moderate	High	Low

- Thrusters also very good choice because they can supply needed propulsive maneuvering capability

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A variety of different attitude control designs were evaluated, as shown in Table 3. Bang-bang thrusters were selected, because they provide the best attitude control capabilities and also because they could be used for translational control which is required by the Escort mission. The desired characteristic for each row of the trade matrix is Low, except for Slew Capability, where the desired characteristic is High. Bang-bang thrusters are the only option that meets all the desired characteristics. Even if one of the other concepts was selected thrusters would still be required for translational maneuvering. Concepts using magnetic actuation in any way were discarded because of the low magnetic field strength in GEO.

Attitude Error Budgets

- ➢ Attitude Pointing Control
  - Requirement: Image > 70% of Intended FOV
  - Imager Full Angle FOV = 5.3°
  - 1.5° Error → 72% of Intended FOV is Imaged
  - Requirement: ±1.5°, 3 Axes, 3σ
- ➢ Attitude Rate Control
  - Requirement: Do Not Blur Image
  - Based on Imager Integration Time
  - Requirement: ±0.13°/sec, 3 Axes, 3σ
  - Min Torque Impulse Bit: 0.047°/sec
- ➢ Meet Requirements Based On:
  - Attitude Determination Accuracy
  - Attitude Update Rate (1 Hz)
- ➢ Attitude Pointing Determination
  - Requirement: 1/3 of Control
  - Requirement: ±0.5°, 3 Axes, 3σ
  - Star Tracker: ±0.083°, 3 Axes, 3σ
  - Includes Centroiding & Algorithm Errors
  - Plenty of Margin for Other Errors:
    - Angular Velocity Aberration
    - Mounting & Thermal Misalignment
    - Relative Orbit & Clock Uncertainty
- ➢ Attitude Rate Determination
  - Requirement: 1/3 of Control
  - Requirement: ±0.043°/sec, 3 Axes, 3σ
  - Rate Sensor: < ±0.004°/sec, 3 Axes, 3σ
    - Plenty of Margin for Noise
    - << Minimum Torque Impulse Bit

The baseline imager for Escort has a 5.3° full-angle field of view (FOV). Requirements for attitude pointing control are set based on being able to image a certain percentage of the intended FOV. For example, to image 72% of the intended 5.3° FOV, pointing control accuracy of ±1.5° is required. The requirement was somewhat arbitrarily set to 70%. Requirements for attitude rate control are set based on a requirement not to blur the images. Based on typical imager integration times the attitude rate control requirement is likely to be on the order of ±0.13°/sec. By rule of thumb, the requirements for attitude pointing and rate knowledge are simply set to one-third of the control requirements, which should be sufficient assuming that all numbers are quoted at three sigma. These requirements can be met using the selected suites of attitude sensors and actuators.

Attitude Impulse Sizing

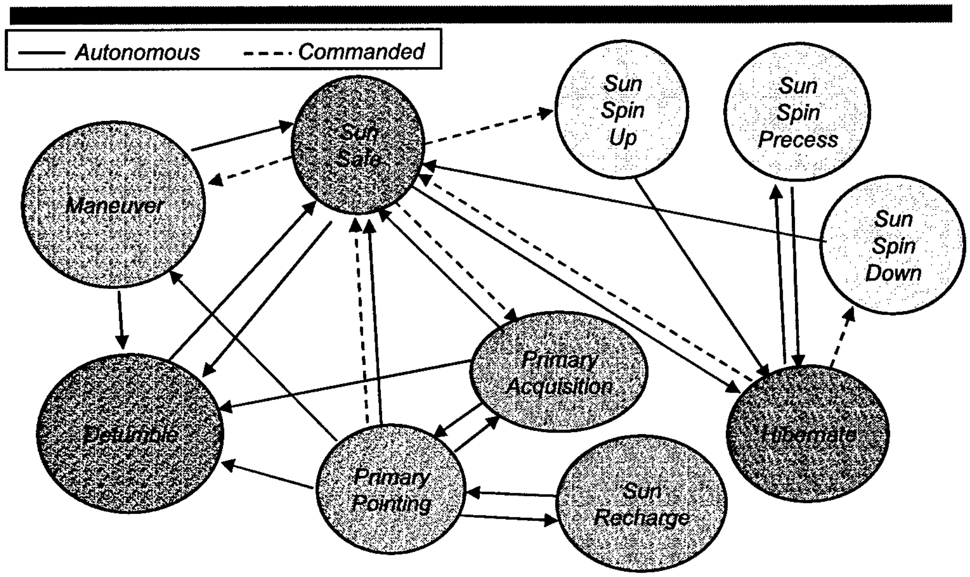
External Disturbance	Torque (Nm)
Solar Pressure	5.9e-7
Gravity Gradient	2.7e-8
Aerodynamic	0
Residual Magnetic Dipole	2.1e-7
Total	**8.3e-7**

Torque Impulse Source	Impulse (Nms)
External Disturbance	45
Attitude Limit Cycle	4820
Total	**4865**

➢ Major Assumptions:
- Altitude and Inclination: GEO
- Solar Pressure Moment Arm: 6.25 cm
- Average Moment of Inertia: 4.67 kg m^2
- Residual Magnetic Dipole: 1 Am^2
- Average Thruster Moment Arm: 0.191 m
- Thrusters Fired in Pairs for Each Axis
- Nominal Control (3σ, 3-Axes): +/- 1.5°
  - Limit Cycle: 256 sec
  - Time Between Firings: 256/2 = 128 sec
  - Jitter: <= 0.047 °/sec (Rigid Body Analysis)
- Minimum Impulse Bit: 0.01 Ns
  - Valve Open Time: 0.1 sec
  - Valve Power Time: 0.2 sec
  - Thrust: 0.1 N
- Two 4 cm High Maneuvering Modules
  - Achievable Mission Lifetime: 119 days
  - Achievable Mission Duty Cycle: 100%

The attitude control momentum impulse requirement is calculated by assuming that Escort has to continuously fight the maximum possible environmental disturbance torque and also continuously perform a bang-bang attitude limit cycle. Over the 119-day active life of the SCOUT Escort in GEO, the limit cycle impulse is approximately two orders of magnitude larger than the disturbance torque impulse. Estimates for solar pressure moment arm and residual magnetic dipole were conservative. The limit cycle period and amplitude was verified by computing it two different ways. The jitter analysis assumed that the SCOUT was a rigid body, at a later stage of analysis flexible body modes including fuel slosh and solar panel vibrations could be included.

AD&C Mode Switching Diagram



The nominal mode of operations is Primary Pointing with periodic excursions to Sun Recharge for simultaneous recharging and payload data downlink. If the lock on the primary needs to be initially acquired or if it is ever lost then the Primary Acquisition mode can be used. In Primary Acquisition mode the star tracker is used to guide a sweeping search pattern for the imager. A separate Maneuver mode is available for larger propulsive translational maneuvers to change the orbit of the Escort relative to the primary. Unless the Escort is spinning on purpose, if it ever enters a tumble then the Detumble mode is activated after which it returns to a non-spinning Sun Safe mode. Because the Escort may not be needed all the time a Hibernate mode is also available in which it is placed into a slow spin with the solar panels pointed at the sun. While hibernating it can periodically wake up and precess its spin vector towards the sun if it drifted away.

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Power

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The following set of slides are for the power subsystem for the SCOUT architecture. This presentation was developed as part of the final review for the SCOUT phase I SBIR program.

Power Requirements

Modules Used	Name	Module Power	Sunlight Duty Cycle	Sunlight Average Power	Eclipse Duty Cycle	Eclipse Average Power	Average Power
0xE01	Escort Payload	15.63	13%	2.03	13%	2.03	2.03
0x302	Propulsion	19.38	0.10%	0.02	0.1%	0.02	0.02
0x101	RF Communications	49.22	24%	11.81	24%	11.81	11.81
0x202	ADCS	9.74	100%	9.74	100%	9.74	9.74
0x002	Avionics II	7.03	100%	7.03	100%	7.03	7.03
0x401	Battery	2.13	100%	2.13	0%	0.00	1.85
0x302	Propulsion	19.38	0.10%	0.02	0.10%	0.02	0.02
0x402	Solar Panel Array	1.11	87%	0.96	0%	0.00	0.84
0xA01	ESPA PIM	0.00	0%	0.00	0%	0.00	0.00
			Sub Total	33.74		30.66	33.35
		Regulation Efficiency	80%	8.44		7.66	8.34
			Total	42.18		38.32	41.69

All powers are shown in watts

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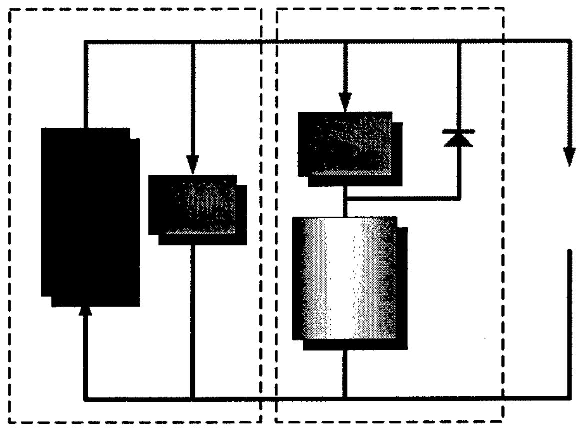
The spreadsheet shown in here is part of a larger spreadsheet that keeps detailed power information for all of the conceived SCOUT modules. This spreadsheet makes it easy to select the desired modules, setup up operation profiles (duty cycles) and tally up their power requirements. The spreadsheet also keeps track of battery and effective solar array area.

Power Architecture

- Direct Energy Transfer power system architecture
- Shunt Regulator located on the Solar Array module
- Batteries and Battery Charger in Battery module
- Battery Module supports up to four independent batteries
  - **Additional battery modules can be added for greater battery power storage capacity**
- Outstanding challenges:
  - **Handling power dissipation from Shunt Regulator**



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The power subsystem has been divided into two independent slices for power storage be it primary or secondary, and power collection.

Power Collection Module will house the deployable solar arrays and the shunt regulator while the power storage module will house the batteries and battery management electronics

Solar Arrays

- Currently investigating two competing solar cell technologies
  - Triple Junction GaAs
    - **High Efficiency (>20%)**
    - **High Cost ($100's K)**
    - **Moderate Power Density (less than 50W per kg)**
  - Thin Film CIS (Copper Indium Diselinide)
    - **High Energy Density (greater than 100W per kg)**
    - **Easy to package**
    - **Relatively new technology**
    - **Low Efficiency (approx. 10% at cell level)**

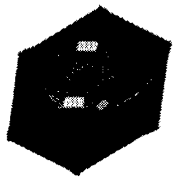
Solar Array Power	Triple Junction	Thin Film	
Solar Flux	1358	1358	W/m^2
Panel Efficiency	25%	8%	
Array Output Power (P_O)	340	109	W/m^2
Sun Incident Angle	0	0	deg
Power Out BOL (P_{BOL})	340	109	W/m^2
Degradation per year	2.5%	2.5%	
Lifetime Degradation (L_d)	97.5%	97.5%	
Power Out EOL (P_{EOL})	331	106	W/m^2
Eclipse Efficiency (X_e)	90%	90%	
Daylight Efficiency (X_d)	80%	80%	
Necessary Power (P_{SA})	59	59	W
SA Effective Area (A_{SA})	0.18	0.56	m^2

Two different solar array technologies are being studied for SCOUT; rigid triple junction GaAS, and Thin film CIS. Selection of the specific technology will be conducted during phase II of the study.

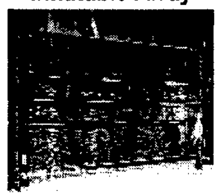
The spreadsheet shown on this slide depicts the effective solar array area required by the ESCORT version of SCOUT

Thin Film Deployment Options

SPORT Concept

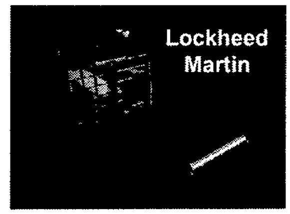


Inflatable Array



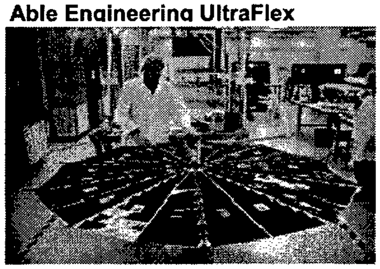
8 to 20 kW (> 200 W/kg)

Roll-Out Array

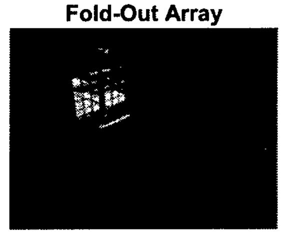


2 to 8 kW (> 175 W/kg)

Able Engineering UltraFlex



Fold-Out Array



to 1 kW (> 100 W/kg)

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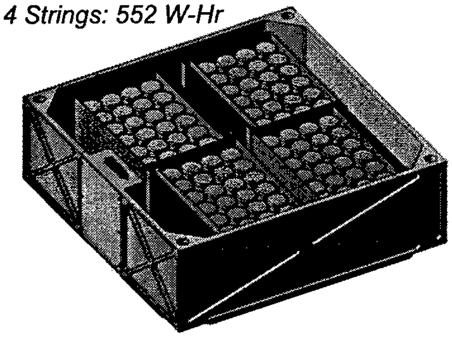
AeroAstro has studied different mechanism for deploying membranes and thin film solar arrays during the Phase I and Phase II Aerobrake SBIRs. Some of the considered concepts are depicted in this slice.

<u>Note</u> Not all of the concepts are AeroAstro proprietary.

Lithium-Ion Batteries

1 String: 138 W-Hr
2 Strings: 276 W-Hr
3 Strings: 414 W-Hr
4 Strings: 552 W-Hr



- **Battery Slice houses up to 4 independent battery strings**
- **String Specifications:**
  - **4.8 A-Hr**
  - **33.6 V OVC**

	SCOUT Shadowed	SCOUT Not Shadowed	
Energy Required (P_e)	94.22	44.32	W-Hr
Max DOD	30%	30%	
Transmission Eff	0.9	0.9	
Battery Size	348.95	164.16	W-Hr
Capacity	12.46	5.86	A-Hr
Cells per String	8	8	
Cell Capacity	1.6	1.6	A-Hr
Strings	8	4	
Number of Cells	64	32	
Cell Mass	42	42	g
Battery Mass (Raw)	2.688	1.344	kg

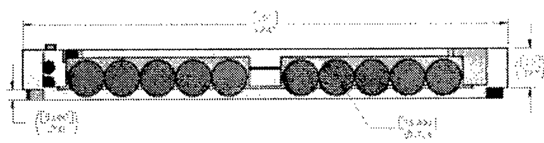
Concepts for two different power storage slices have been designed; a *fat* Lithium-Ion battery box (depicted in this slide) and a *thin* Lithium-Ion battery box (next slide).

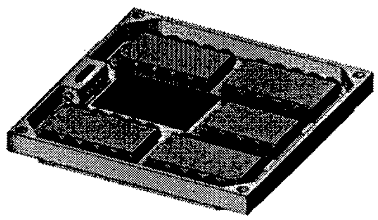
The *fat* Lithium-Ion battery slice can house up to four independent battery strings along with their management electronics. Depending on power needs of the spacecraft one, two, three, or all four battery boxes can be populated. Mass equivalents will be available to drop in place of unused battery strings.

For added power storage capabilities, two or more battery slices can be stacked up in the same SCOUT spacecraft. It will also be possible to add primary batteries If necessary.

Lithium-Ion Batteries (Alternative configuration)

- Battery Slice houses a single battery string
- String Specifications:
  - 4.8 A-Hr
  - 33.6 V OVC
- Flat configuration will allow us to minimize overall spacecraft volume and height when only one or two battery strings are required





- At present, flat battery configuration violates the 2cm slice height
- Plan to evaluate alternative cell manufacturers and dimensions to stay within slice envelope volume

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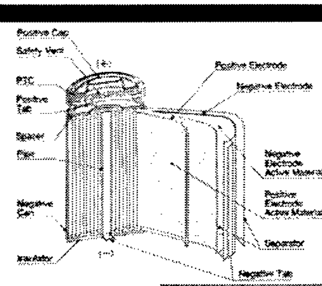
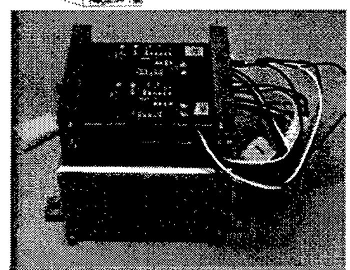
An alternative to the *fat* battery slice that takes up unnecessary volume in the SCOUT configuration in which only a single string is necessary, a *thin* battery slice has been designed. This battery slice will house a single battery string as depicted in this slide. Multiple *thin* battery slices can be stacked together or in combination with the *fat* battery box.

At present the *thin* battery slice violates the 2cm slice clearance. One of the solutions are to investigate different cell manufacturers and/or capacities.

Lithium-Ion Battery Cells

- Commercially available Lithium-Ion Battery Cells (18650 cells 1.6 A-Hr)
  - High Energy Density (~ 130 WHr per kg)
  - Many Safety Features (Positive Thermal Controller, Pressure Vent, Pressure based circuit breaker)
  - Low self discharge (~ 5% per month)
  - No "memory effect" (Cells do not need routine reconditioning)
- AeroAstro has experience qualifying Li-Ion batteries





SPASE Battery Assembly

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Page 8

Commercially available 18650 cells will be used for the construction of the battery slices. This particular cells are widely available from different manufacturers (Sony, Panasonic, Maxell), offer great performance at an affordable price.

A number of safety features have been incorporated in the design of the cells making them one of the safest Lithium-Ion cells in the market.

AeroAstro

Software

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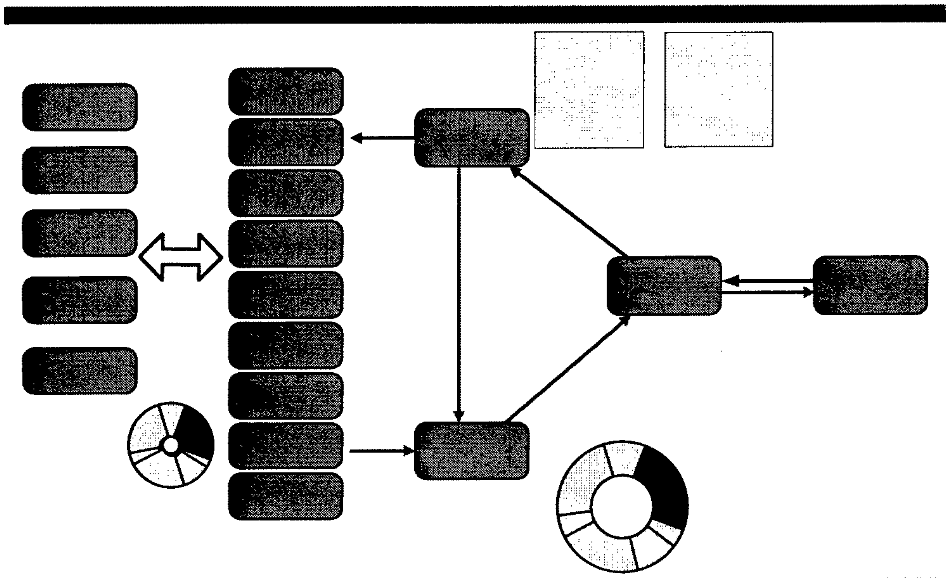
The following set of slides is for flight software for the SCOUT vehicle. This presentation was developed as part of the final review for the SCOUT phase I SBIR program.

Architecture



AeroAstro Page 2

The Flight Software will be divided into a number of layers and modules (within each layer) making it portable, flexible, modular.

Portable; by abstracting the higher layers of the software (Spacecraft managers, C&DH, etc) we can divorce the implementation of this layers from the hardware.

Flexible behavior of the spacecraft revolves around a set of infrastructures that handle uplink, telemetry gathering, communications, and health and maintenance. A simple framework will enable the operator to set the behavioral rules, by which the spacecraft operates.

Modular; each hardware slice will have a corresponding software module, which the application software utilizes for commanding the slice. Slices are hierarchically categorized by the function they perform and the information they provide and or require. By organizing modules in a manner described above we can define high level software ICDs for the modules and identify commands and data formats that each *similar* module will recognize.

For instance, SCOUT has defined three different GN&C actuation modules (micro-wheel actuation, magnetic actuation, and propulsion). The details of how those modules are controlled vary widely; those details are found within the slice software module. However, to the SCOUT application flight software all three GN&C actuation modules perform similar functions, to maneuver the spacecraft. Defining a SCOUT GN&C actuation ICD that specifies the data types, functions, and parameters that these functions require would make swapping the modules seamless to the field integrator.

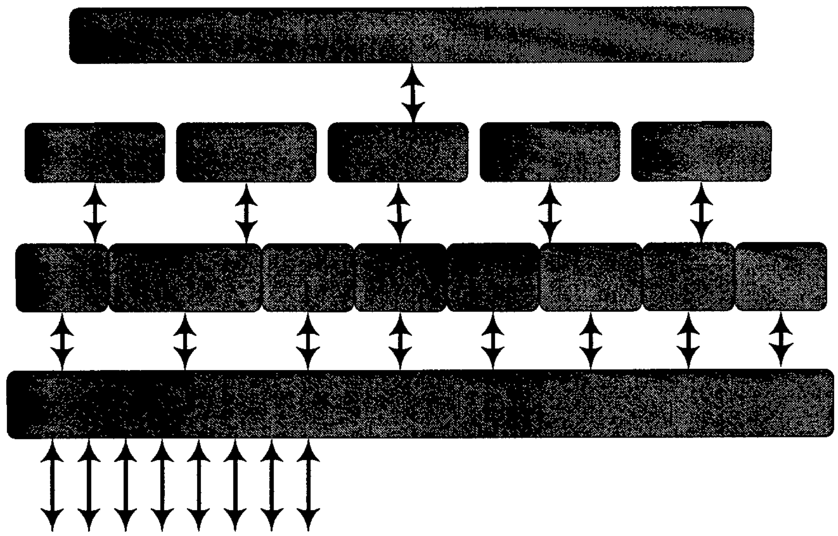
File Uploader

Health & Maintenance

ADCS

Power Management

Modularity



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This graph shows a more traditional view of the flight software architecture. It also depicts the interactions and the data interfaces between the different layers of the flight software.

GN&C
Framew

X-band M
Module

Commanding

- Real-time commanding
  - Commands are forwarded to the appropriate process or module for immediate execution
- Event based commanding
  - Executed at a later time based on detection of an "event"
    - The arrival of a specified time is an "event"; time-tagged commanding is a subset of event-based commanding
  - Implemented utilizing an event handler library
    - Includes table manipulation commands
    - Table telemetry includes:
      - Entry number
      - Major/Minor type
      - Execution time/event
  - Stored command table maintained by command handler
    - Permanent commands loaded at system startup
    - Temporary commands loaded by ground
    - Both are modifiable

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Commanding of the spacecraft is done either by uploading commands to the **stored command table** or by having *permannent* commands loaded during the power-up sequence. All of the command on the command table are accessible and modifiable by the ground operators.

Commanding is event based; that is commands placed on the **stored command table** will be triggered by an event such as time, upload carrier detected, other commands, etcetera.

Telemetry

- ➢ Collects telemetry from vehicle subsystems and modules
- ➢ Services telemetry queue
  - Messages routed to proper stream
    - Real time telemetry sent to ground
    - Stored telemetry sent to circular buffer
- ➢ Table-based telemetry handler
  - Telemetry message
  - Real time rate
  - Stored rate (a multiple of real time rate)
  - Enable/Disable

Telemetry gathering is table based as well, the C&DH system will post the flight software telemetry, spacecraft status, and slice status.

Telemetry (Continued)

- Real time telemetry
  - Enabled by command from ground
    - Continuous telemetry stream
    - No additional commands required
    - Sent to "Low Priority" Queue, not to interrupt payload data
    - Disabled by disconnect from ground or command
- Stored telemetry
  - Collected continuously
  - Stored in circular buffer
  - Configurable to either overwrite buffer or stop collecting telemetry
- Storage Rates
  - Three Rates (H,M,L) for each (Real Time/Stored)
    - High Rate: once every 3 seconds (approximately)
    - Medium Rate: once every 30 seconds (approximately)
    - Low Rate: once every 5 minutes (approximately)
  - Adjusted as necessary to meet storage and mission requirements

As mentioned earlier, the ground operator as the option to get *real time* telemetry updates; he/she can selected to see all or part of the telemetry points.

Telemetry is stored in a circular buffer; the operator will have the ability to set the spacecraft to either overwrite or cease storing telemetry once the buffer is full. The telemetry table will continue to be updated at the regular bases.

The operator can choose what telemetry points it wants stored and at which frequency by setting the storage rates.

Health And Maintenance (H&M)

- H&M Telemetry
  - SWLOG
  - State Of health
    - Collected for entire vehicle and individual subsystems
    - CPU load analysis (task basis)
    - Memory statistics
- H&M Task
  - Centerpiece of H&M system
  - Collects and reports health TLM
  - Runs low-level maintenance
  - Can be designed to detect and possibly act on detection of faults
- Response to On-orbit H&M issues
  - Implemented via event table entries
    - Rules for module reset
    - Rules for module Power cycles
  - Will provide the capability to increase the level of autonomy
- Maintenance Health Subsystem
  - EDAC Scrub
  - Watchdog tickling
  - Collection of health telemetry
    - Memory statistics
    - CPU loading
      - Total load
      - Task level loading
    - SWLOG and error monitoring
- Software Log Messages - "SWLOG"
  - Essentially a "printif" statement
  - Indicates error conditions
  - Used for "SLOW" Errors and Informational Messages
  - Messages can be routed into stored telemetry for later download

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The Health and maintenance system has two responsibilities; to ensure the health of the spacecraft and to report any anomalies.

The H&M has the background responsible to perform memory scrubbing and service watchdog timers.

The most powerful feature of H&M is the **H&M event table** that is implemented via event table (similar to stored command table) entries. Rules for how the system must react (what commands to execute) to an anomaly. This feature will provide the capability to increase the level of autonomy as we learn more about a particular SCOUT configuration and the environment in which it operates.

Software Development Process

- Small design team assigned to each module
  - Follows the AeroAstro design *buddy* approach
  - Design and implementation by primary *buddy*
  - Module test and release by secondary *buddy*
- Code management
  - Version control
    - Utilizing CVS software
    - Baseline tracking
    - Code release management
  - Bug Tracking
    - Utilizing Mantis software
      (an open-source bug tracking and reporting software package)
    - Assign, categorize, track, and provide status on anomalies

AeroAstro

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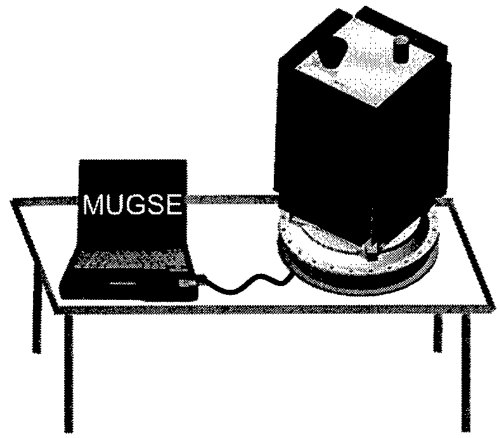
Each SCUOT module will have a small team (preferably 2) of engineers responsible for the design, implementation, and testing of the module.

AeroAstro employs a number of tools for version control of the software and well as tracking anomalies; those tools will be used during the development of SCOUT software as well.

Software Integration & Test

- Ground I&T Support
  - The Master Universal Ground Support Equipment (MUGSE) will be used for all SCOUT I&T
  - MUGSE will be used to upload and troubleshoot the latest version of software and drivers
- On-Orbit software upload
  - Operators can upload software patches or entire image to S/C
  - Operator has the ability to test uploaded software before committing to flash
  - In case of failure during test, the S/C will reset and load last known good software image
  - S/C has capability to have multiple images in flash



A MUGSE will be used for interacting, servicing, and diagnosing the SCOUT spacecraft and/or any combination of slices. The MUGSE will connect to the Spacecraft via an Ethernet network connection.

Software images or a *patch* to a software image can be uploaded at any time by an operator. The Spacecraft can be instructed to execute the uploaded software and *commit* (write it to FLASH) to it once the newly uploaded software has been tested. If an anomaly where to occur during the upload or test periods, the spacecraft will reset and load the last good know software image.

Software Test Approach

- Integration support
  - MUGSE will be used for all SCOUT Integration & Test (I&T) activities
  - MUGSE allows test of individual modules or fully integrated spacecraft
  - MUGSE provides power, CMD, and low- and high-speed TLM interfaces
  - MUGSE will be used to upload the latest versions of software and drivers
- Test Flow
  - Functional unit or module test
  - Software integration and validation
  - SCOUT module test
  - Integration and Test
  - Formal system level I&T
  - Formal acceptance test
- Test Categories
  - Functionality
  - Performance
  - Resource Utilization
  - Adherence to interface specifications
  - Timing constraints
  - Induce error conditions such as subsystem or module power cycle

Testing will be done, witch one exception, in a very traditional way, test functions, subsystem, spacecraft. Testing for functionality, performance, timing issues, etc.

One exception will be that each slice will be individually tested and qualified for SCOUT so that only a quick functional test will be necessary when it is integrated in the field with the rest of the SCOUT vehicle.

Software Diagnostics and Field Testing

- Operators and integrators can easily perform testing and diagnostics by using built-in diagnostic routines
- Each module will be loaded with self-diagnostic routines
- Well-defined diagnostic codes will assist operators and integrators to efficiently identify faulty components
- Built-in test and diagnostics can be executed from either the MUGSE workstation or while in orbit
  - In orbit, diagnostic codes will be transmitted to the ground
- MUGSE will incorporate a Graphical User Interface to simplify selection of tests and interpretation of returned diagnostic codes and error messages

One of the tasks that each CEB must perform is to perform a series of Built-In-Tests (BIT) as defined by the specific design of the slice. Each BIT has pass/fail criteria for every component been tested. These tests return standard diagnostics codes that would allow the field integrator to quickly and efficiently identify, isolate, and replace faulty components. Each module is capable of performing BITs as a standalone, by connecting the module directly to a MUGSE, or in a SCOUT stack with or with out the assistance of a MUGSE. A Graphical User Interface on the MUGSE simplifies test selection and interpretation of returned diagnostic codes and error messages. Built-In-Test can also be performed while in orbit. The return codes are transmitted to the ground as part of the RF stream.

AEROASTRO

Command and Data Handling

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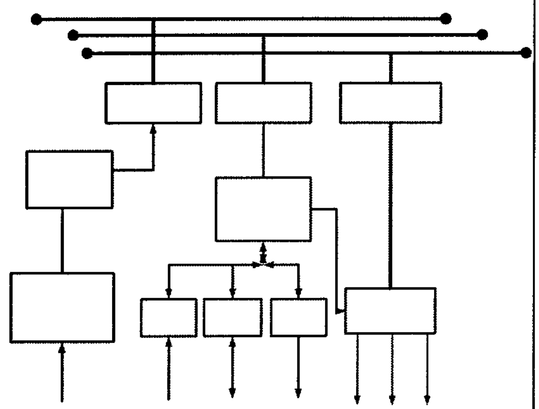
The following set of slides is for the Command and Data Handling subsystem of the SCOUT vehicle. This presentation was developed as part of the final review for the SCOUT phase I SBIR program.

Core Electronics Block

- The Core Electronics Block (CEB) will be required for every module in SCOUT
- The CEB will interface the SCOUT bus to instruments, subsystems, and sensors
- The CEB will have the following functionality:
  - Low-speed data interface
  - High-speed data interface
  - Power distribution, switching, and regulation
  - A/D for TLM, e.g., thermal
  - Discrete (logic) I/O
  - RS-422 full duplex data
  - Buffer



The Core Electronics Block (CEB) is a small electronics board that will reside in each SCOUT *slice*; it will provide the glue required to interface to sensors, actuators, and components in the individual slices. The CEB will offer a great deal of functionality to each SCOUT slice; specifically it will offer analog data collection, discrete I/O, serial interfaces, secondary power regulation and monitoring. In addition the CEB will offer other services to support the SCOUT bus such as performing built-in tests (BIT). At the heart of the CEB will be a radiation tolerant FPGA for telemetry gathering, control, and diagnostics.

Backbone Data Bus Selection

		Backplane or Cable	Central or Distributed	No. of Wires	Max. Length	Data Rate	Payload Fraction	
Low-Speed Data Bus	I^2C	Cable	Distributed	2	~4 m	400 kbps	~85%	← *Selected*
	Bluetooth	N/A	Distributed	wireless	100 m	1 Mbps	72.3%	
	CAN	Cable	Distributed	2	40 m	1 Mbps	59%	
	1553	Cable	Central	2	~ 50 m	1 Mbps	75%	
High-Speed Data Bus	IEEE 802.3	Either	Distributed	4	35 m	10 or 100 Mbps	96%	← *Selected*
	VME	Backplane	Central	128	< 1 m	160 Mbps	99.5%	
	SCSI	Cable	Distributed	68	6 m	320 Mbps	99%	
	USB	Cable	Central	4	~ 5 m	480 Mbps	99%	
	RS644	Cable	Distributed	2min	10 m	655 Mbps	99%	
	cPCI	Backplane	Central	124	< 1 m	1000 Mbps	99.5%	
	IEEE 1394	Cable	Distributed	4	4.5 m	4 Gbps	96%	

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The data bus architecture for SCOUT vehicle has been divided into two groups, one that is suitable for low data rate applications such as commanding and telemetry and the second that are suited for high speed applications such as imaging. A trade was performed for both groups with the cut-off point at 1Mb/sec, i.e. any data bus architecture that supports data transfers higher than 1Mb/sec will be considered high speed. For the purpose of this trade the following characteristics were evaluated:

Backplane/Cable: do all participants on the bus need to be in the same box (VME), or can they be distributed around the spacecraft (USB)

Central/Distributed: Is there one central governing device which controls who can transmit on the bus at any one given time, or all the devices on the bus participate in arbitration to decide who can transmit or have priority

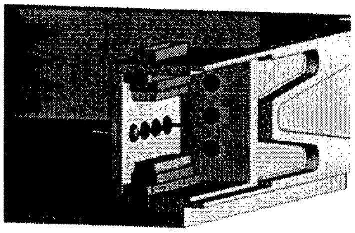
Number of Wires: The number wires that are required for the bus

Max Length: The maximum physical length of the bus

Data Rate: The maximum data rate of the

Payload Fraction: What is the maximum number of bits that transmitted, versus header or other information

Backbone Harness Connector



	Data	Power	Ground
I^2C	2	0	0
I^2C (redundant)	2	0	0
802.3	4	0	0
Power	0	3	4
Power (redundan	0	3	4
Sep Switches	4	0	0
Isolation	4	0	0
Sync	2	0	0
Spares	5	0	0
Total lines	23	6	8
Connector Size >	37		

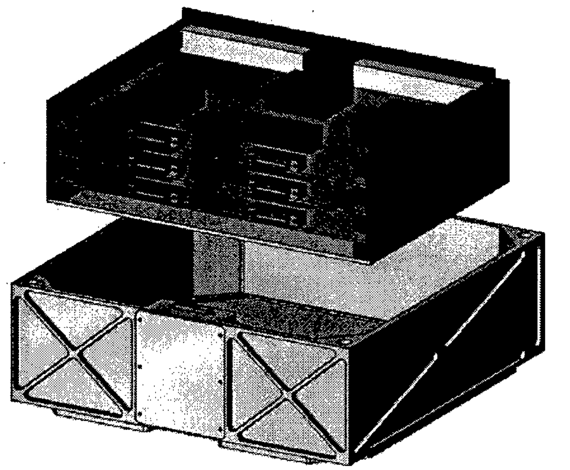
➢ Harness Specifications

- 37-pin MDM inter-module connectors
- 26-AWG Teflon harness
- Power handling capabilities; 1.4A per line.
- Total power handling capability 6 X 1.4A X 28V = 235W
- Isolation pins between power and signal lines

MDM connectors will be used to connect SCOUT slices to the electrical backbone of the vehicle. A 37 pin connector has been chosen to provide the conduit for power, data, and synch signals. This configuration will enable to SCOUT system to handle anywhere from 235W up to 313W (is using spare lines to route power). A redundant set of I2C lines have been added for robustness. Each slice will be able to access the electrical backbone by either connecting trough EMI-filters, a connector, or a cutout. Having isolation pins and proper shielding will alleviate EMI concerns.

The Cable Derating Criteria follows the suggestions of NASA's GSFC Preferred Parts List

C&DH Module (Version III)



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Depicted on this picture is a Version III S&DH slice. Included in this particular version is a Core Electronics Board, a Flight Computer, and a Mass memory board.

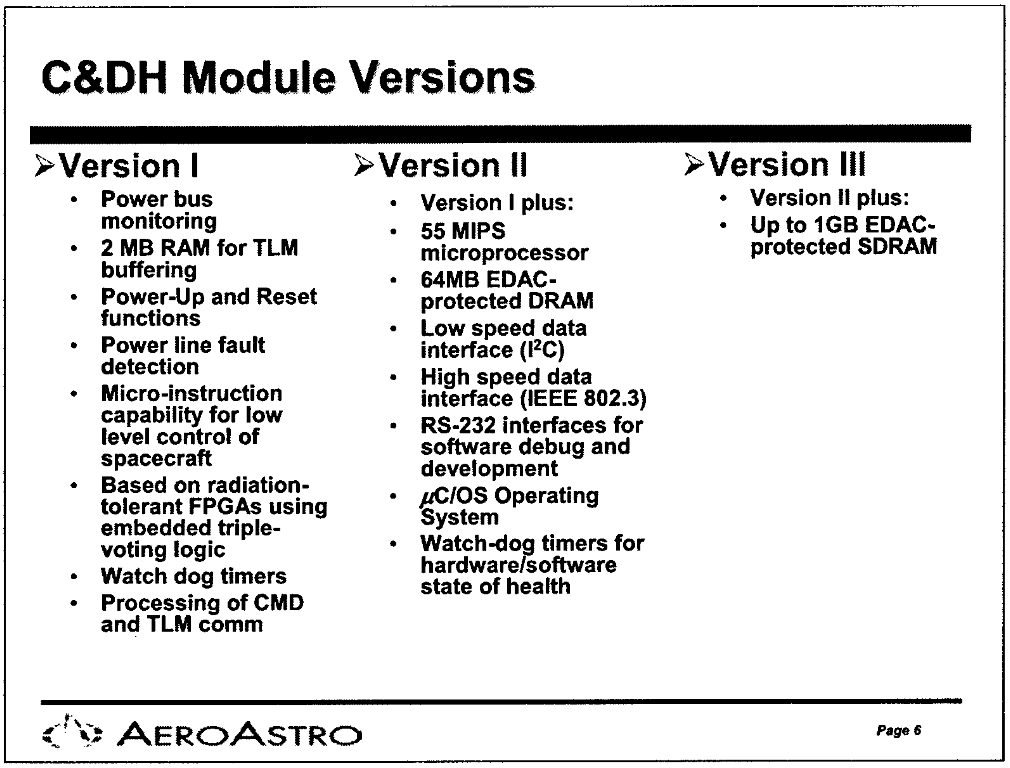
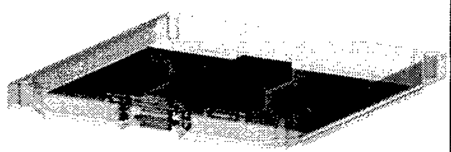


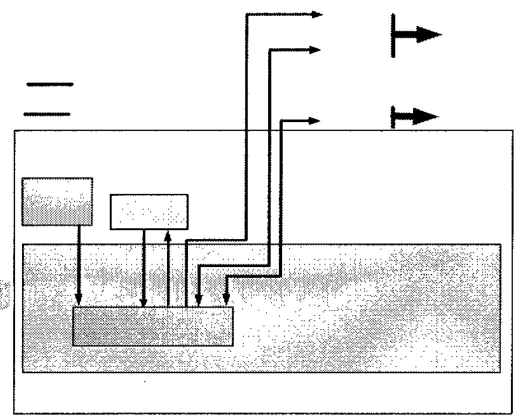
Figure 1 SCOUT Avionics Block Diagram

In order to meet the avionics requirements of the different missions the SCOUT modular architecture could serve; AeroAstro adapted its avionics architecture to provide additional modularity, scalability, and adaptability while retaining a simple design. There are three versions of the SCOUT avionics, version I, II, and III. Each version increases the overall capabilities of the previous version and of the spacecraft bus. By having this modular avionics architecture, SCOUT can be used for a variety of missions ranging from a simple experiment test-bed to complex applications such as high resolution imaging. The capabilities of each version are described in the table below.

C&DH Module - Version I

- Version I is designed for simple missions without active AD&C
- Ideal for applications such as:
  - Microgravity experiments
  - Flight Tech Demo missions





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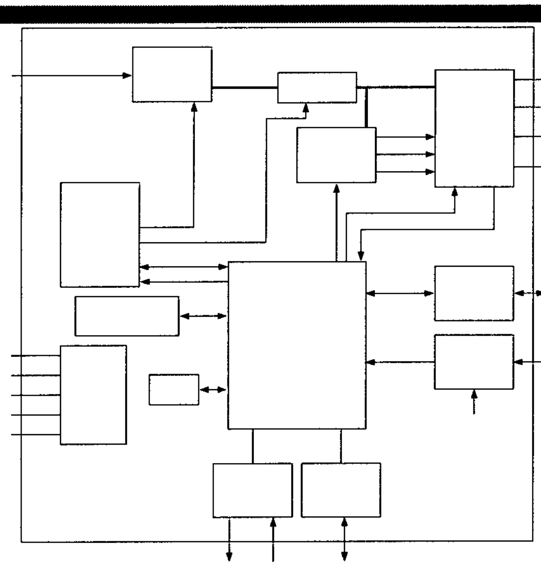
Version I of the C&DH module is the simples avionics module. It is designed to support the simplest of spacecraft needs with out an active Guidance and Navigation system. A Radiation hardened FPGA is at the hear of this module. This particular version of the avionics will be capable of collecting telemetry, interpreting and storing ground commands, monitoring and reporting state of health, and handle some small amount of payload telemetry (about 4 MB). Additionally, the Core Electronics will interface with the Launch Vehicle separation sensors to determine when the Spacecraft has been released and initiated the power-up sequence.

C&DH Module - Version I Block Diagram



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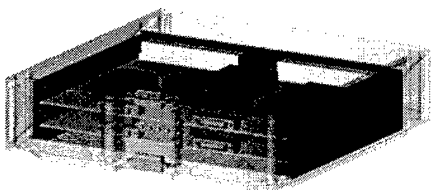
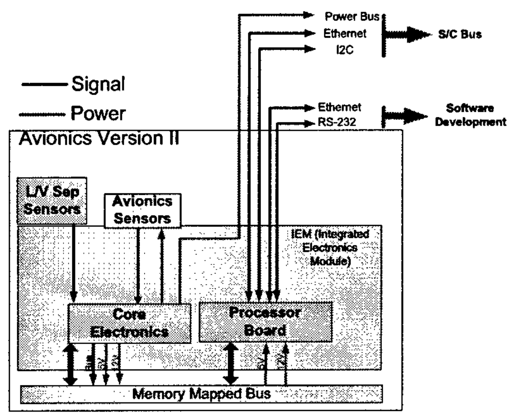
The Power Command and Telemetry Board is an integrated, single-board solution that provides the majority of the capabilities required for a simple spacecraft. A first version of this board was originally developed by AeroAstro for the NASA SPASE (Small Payload Access to Space Experiment) nanosatellite. The PC&T is built around a radiation-tolerant Actel FPGA that contains firmware to issue timed commands, support communications, and command slices to switch power lines. It includes 2 MB of RAM for telemetry buffering. The FPGA provides the algorithm by which the PC&T core provides, power up & reset control, command processing, L/V separation switches, communications, telemetry gathering, telemetry & command storage, and ground support.

C&DH Module - Version II

- Version II adds a microprocessor board to the Version I module
- Designed for applications such as:
  - Space Experiment Testbed
  - Escort
  - Tactical GPS

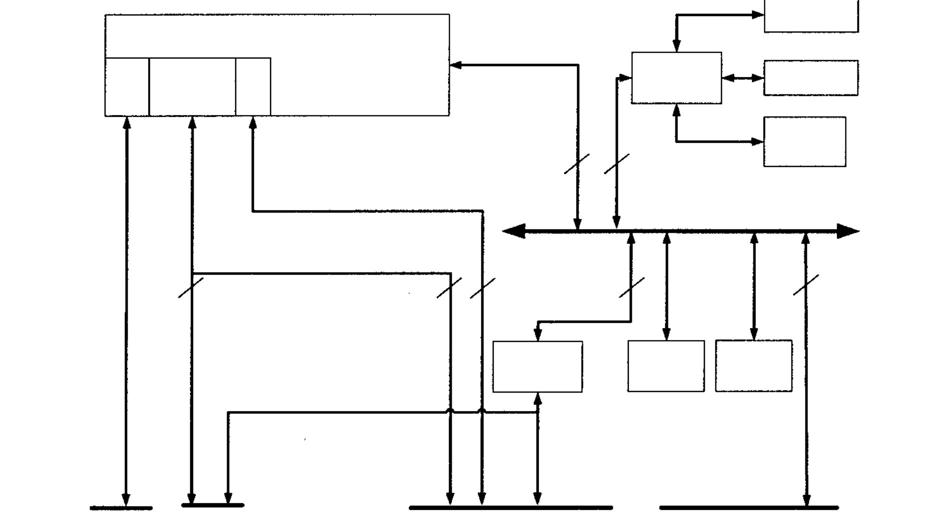




Version II of the SCOUT avionics will add a processor board to the avionics subsystem; the rest of the subsystem (core electronics board) will remain unchanged.

With a microprocessor present, autonomy can be added to the spacecraft; it will also be feasible to perform complex operations such as those required for G&NC operations. This version of the C&DH module will provide 64 to 256MB of ECC protected mass memory.

C&DH Module - Version II Block Diagram



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SCOUT's single board computer is based on the Motorola MCF5307 Coldfire processor. The MCF5307 is a widely used embedded processor which provides a standard memory interface bus, DRAM controller, I2C interface and serial interfaces. Memory is connected to the processor through a Memory/EDAC controller which will be custom designed into an Actel FPGA. The memory/EDAC controller will support boot ROM, Flash memory and DRAM (with help from the MCF5307 DRAM controller). The local memory bus will be brought off card to allow the processor to access peripherals on other parts of the system.

DEBUG

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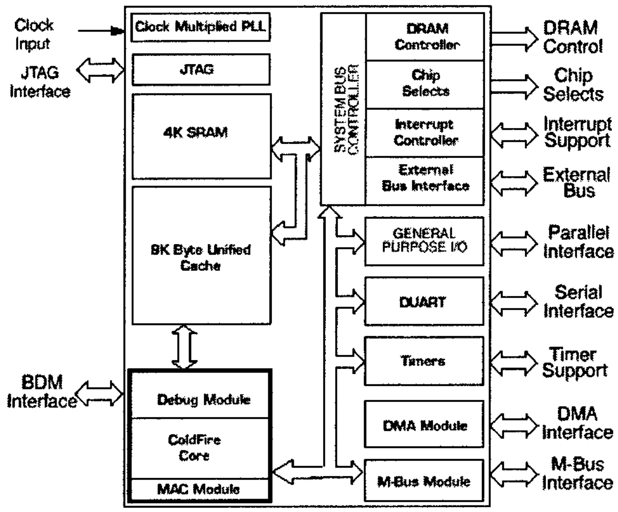
The single board computer will also provide an 802.3 (Ethernet) interface for debug and communication to other peripherals. The 802.3 interface will be based on Standard Microsystems LAN91C96 Ethernet MAC/PHY. The LAN91C96 does not need a PCI bus and provides direct interfacing to the MCF5307 through the local memory bus.

The MCF5307 has two configurable UARTS. These UARTS, with the help of external logic, can be configured as an RS-232, RS-422, or CAN bus interface. These serial channels will be brought off card to interface with other spacecraft peripherals or for testing purposes. The single board computer will provide a direct interface to the MCF5307 debug port for system testing. Additionally we are bringing the MCF5307 processor's I2C bus off card to interface with peripherals such as analog to digital converters, digital to analog converters and EEPROM. Other functions will include power-on-reset, watchdog timer, and real-time clock

C&DH Module - Version II Microprocessor

- The Motorola Coldfire MCF5307 is a low-power, highly integrated processor designed for embedded control applications
- On Chip Features:
  - 8 KB of unified cache
  - 4 KB of SRAM
  - MAC module
  - Hardware divide
  - 4-channel DMA controller
  - DRAM controller with glueless support for 256 MB of synchronous, EDO or page-mode DRAMs
  - Dual 16-bit general-purpose multimode timers
  - Serial and parallel comm interfaces
  - I²C compatible Motorola M-bus
  - Power management modes
- A Radiation - Hardened version is under development



The MCF5307 is a low-cost, highly integrated microprocessor, designed for embedded control applications, which combines a ColdFire® processor core with a Multiply Accumulate (MAC) unit, DRAM controller, DMA controller, timers, and parallel and serial interfaces. The MCF5307 provides interfaces to 8-, 16-, and 32-bit DRAM, SRAM, ROM, and I/O devices. ColdFire® 5307 offers a great deal of functionality making it well suited for applications such as micro and nano-satellites. Its power draw is less than 1W making the 5307 a good choice for a very low power avionics flight computer. Some of the on-chip features include;

•8 Kbytes of unified cache

•4 Kbytes of SRAM

•MAC module

•Hardware divide

•4-channel DMA controller

•DRAM controller with glueless support for up to 256 Mbytes of synchronous, EDO or page-mode DRAMs

•Dual, 16-bit general-purpose multimode timers

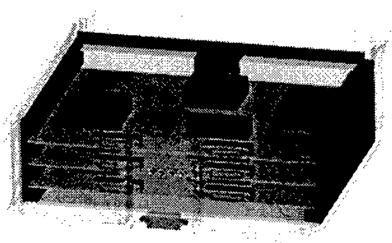
•Serial and parallel communication interfaces

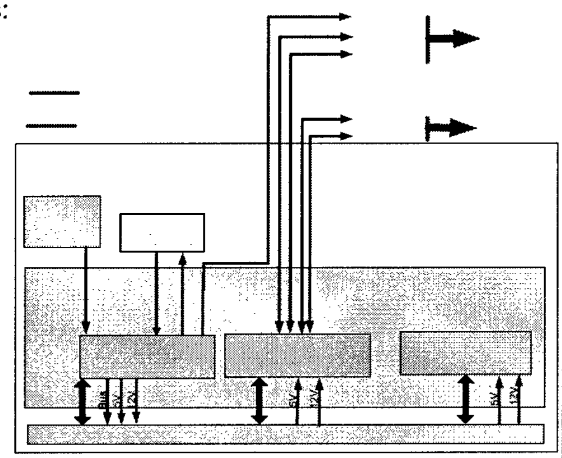
•I2C compatible, Motorola M-bus

•Power management modes

C&DH Module - Version III

- Version III is designed for high data volume applications such as:
  - Hi-Resolution or high frame rate imaging





For those applications that requires great amounts of mass storage; a version of the avionics has been devised that offer up to an additional 1GB of ECC protected RAM. Another Idea that we are studying is to have a mass memory module that could contain several mass memory cards.

AeroAstro

Mechanical & Thermal Design

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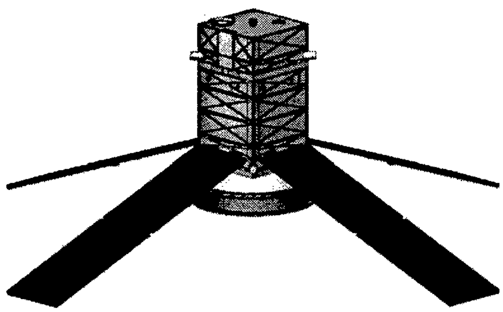
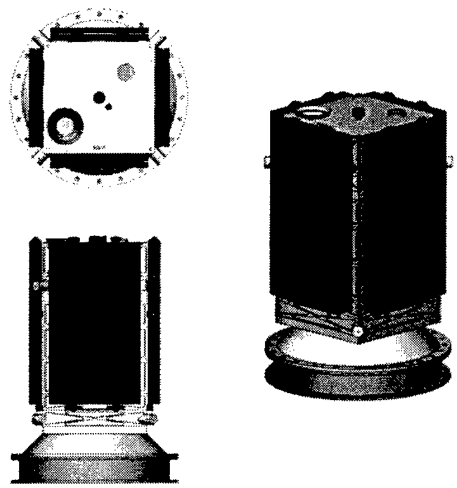
SCOUT Mechanical Concept



- 9 Modules
- Cross section:
  25 cm X 25 cm
  (excluding thrusters and solar arrays)
- Height: 46.7 cm
  (excluding PAF Interface)
- Approx mass: 63 kg

The SCOUT mechanical concept will be demonstrated by assembling a spacecraft with the SCOUT slices. The particular spacecraft depicted here is for a 9 module configuration for a height of 46.7 cm and a mass of approximately 55 kg. Each module has a cross section of 25 cm X 25 cm. The modules are stacked vertically with any peripheral features like solar arrays, thrusters, and sensors violating the 25cm square envelope.

SCOUT Mechanical Concept



- Stowed Views
- Gaps between solar arrays accommodate thrusters and other potential protrusions
- Top module position allocated to payload

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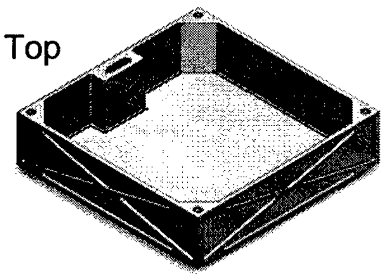
There are a few rules about the order to stack the modules:

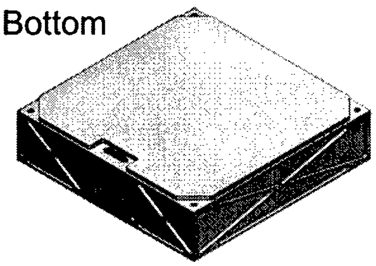
1. Any propulsion should be placed as far from the center of gravity as possible.
2. The top module position is allocated to the payload. This position allows for the most external surface area to accommodate any sensors or cameras associated with the payload.
3. Even though the modules are designed to be independent, there must be some consideration to keep the peripheral features from interfering. As seen here, the stowed solar arrays and the thrusters must be coordinated to avoid interference.

SCOUT Standard Module

Top



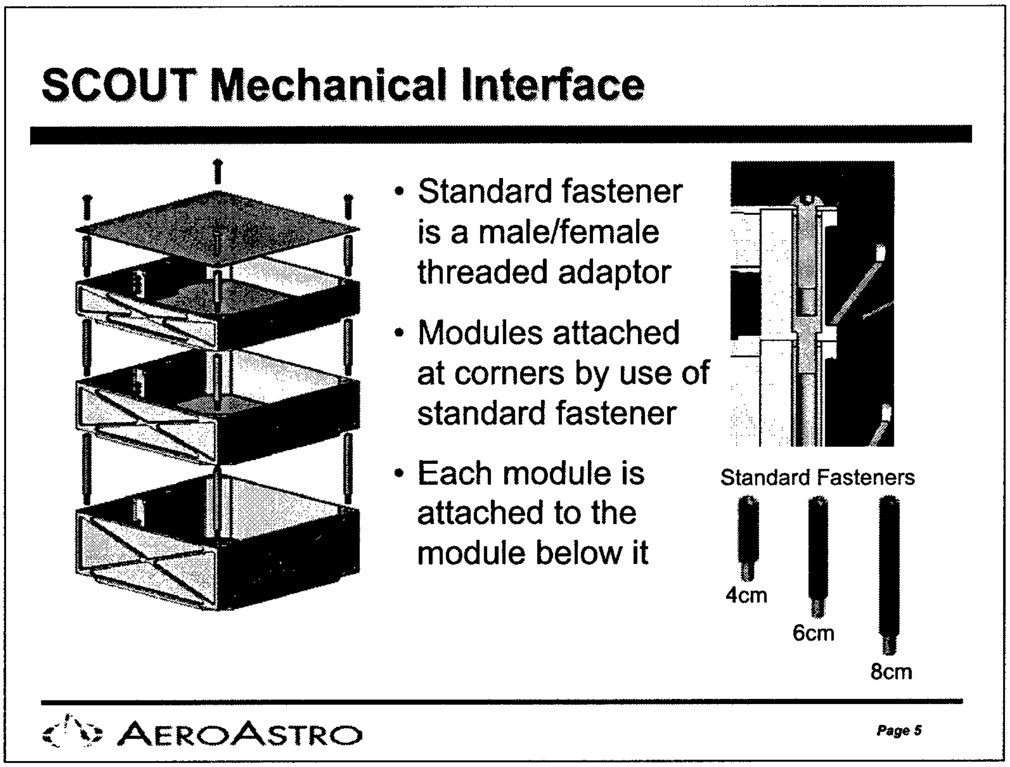
Bottom



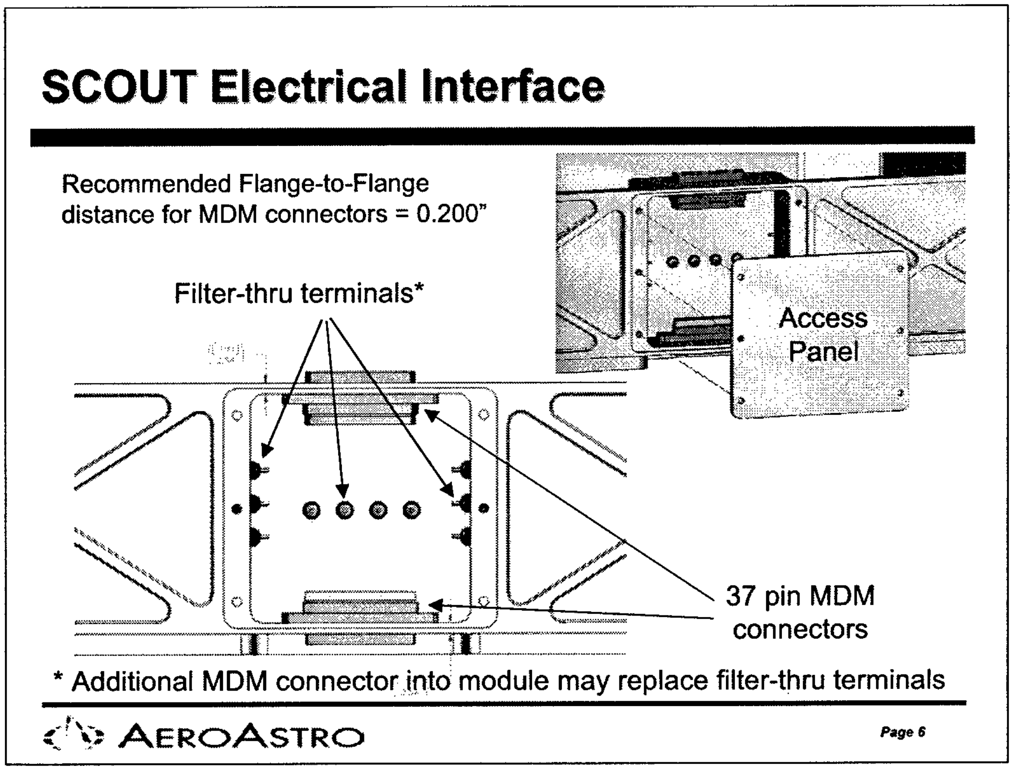
- Standard module contains features common to all modules
  - Same mechanical interfaces
  - Same electrical interfaces
  - 2 cm height increments

Structure Mass Estimate	
Slice	**Mass**
2cm	1.15 kg
4cm	1.30 kg
6cm	1.45 kg
8cm	1.60 kg

Shown here is an example of the standard SCOUT module. Each module will contain identical mechanical and electrical interfaces. One of the features of this design approach is that slices can be assembled in in any order (with a few exceptions) that will best meet specific mission requirements.



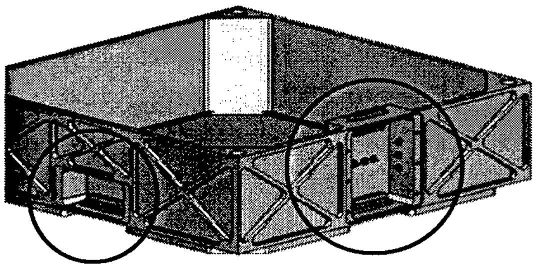
Mechanical interfaces consist of four male/female threaded adaptors, one on each corner of the module. Each module is integrated to the module below it as shown.

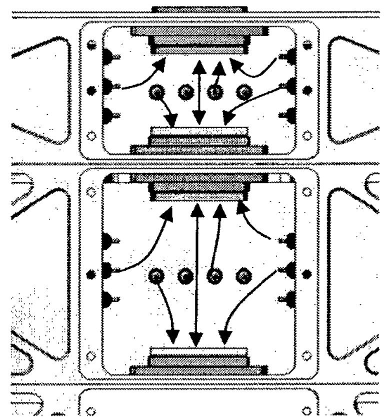


Electrically, each module has a base and a top MDM 37 contact connector. The pins of the base connector are connected directly to the corresponding pins on the top connector. Wires from the filter-thru terminals are then tapped into the harness where appropriate. Once all the connections have been made, a removable access panel is bolted in place to protect the wiring harness.

SCOUT Electrical Interface

Rotating Module – 2 base connectors and 1 top connector allow module to rotate 90 degrees relative to module below





Inter-Module Electrical Stack-up

Note that each module can be wired as needed and still use the same interfacing connector

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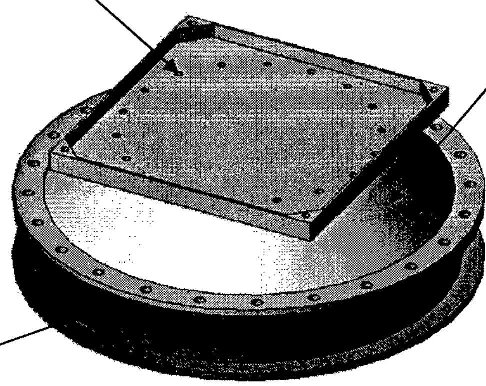
On occasion, there is a need for a model to be rotated 90°. These module will be equipped with a second base connector.

Module 1: PAF Interface Module (PIM)

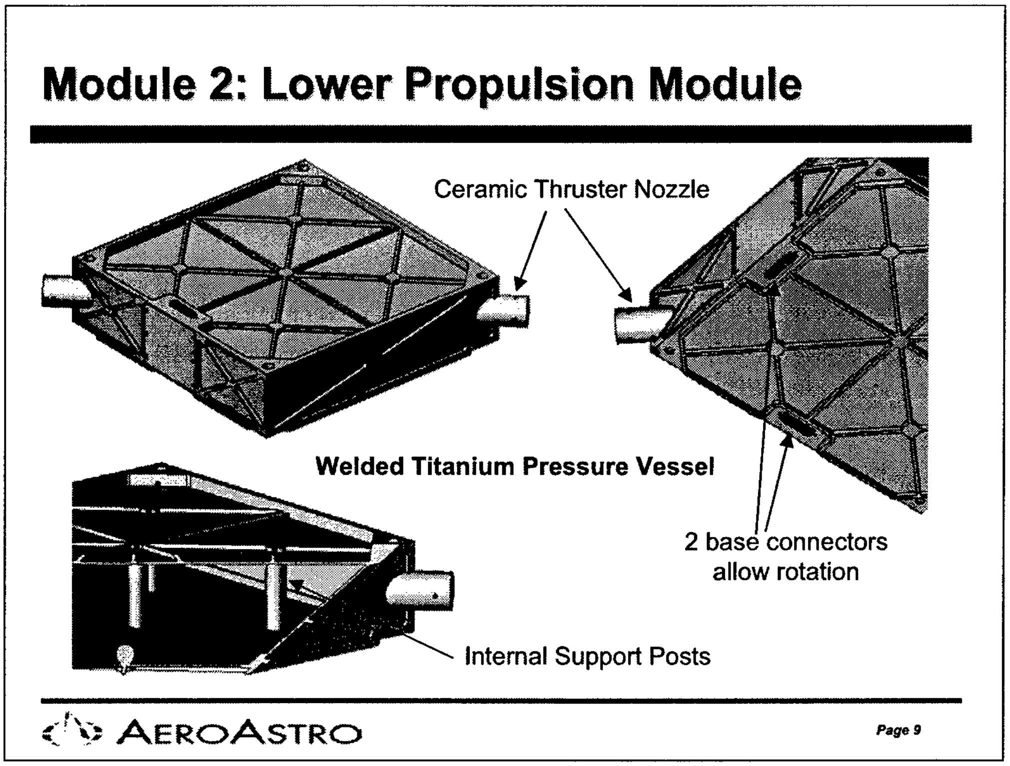


The first module of the S/C is the PAF Interface Module which consists of:

•The separation system. In this case it is a standard 15" Lightband.

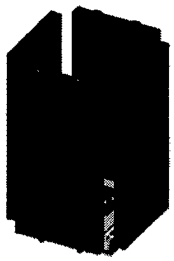
•An Adaptor Cone to reduce the size of the bolt pattern to correspond to the 25 cm^2 SCOUT cross section

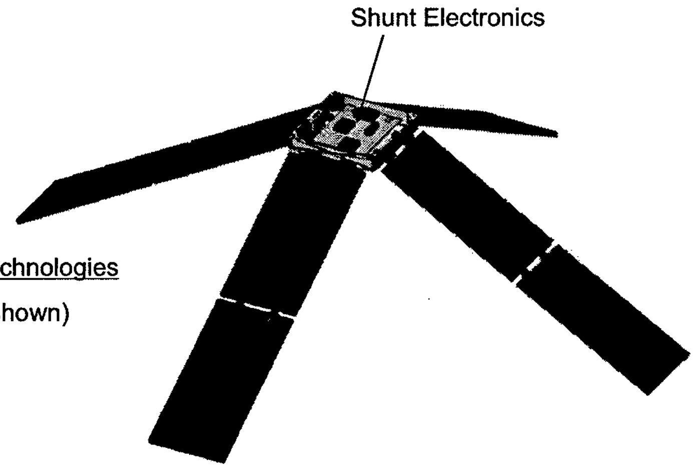
•A Bolt Pattern Adaptor to allow the next module to attach with the standard SCOUT fastener.



Module 2 is the lower propulsion module. The propulsion module is a welded titanium pressure vessel with a ceramic thruster nozzle on two opposing corners. This is one of the modules that requires a second base connector for rotation.

Module 3: Solar Arrays



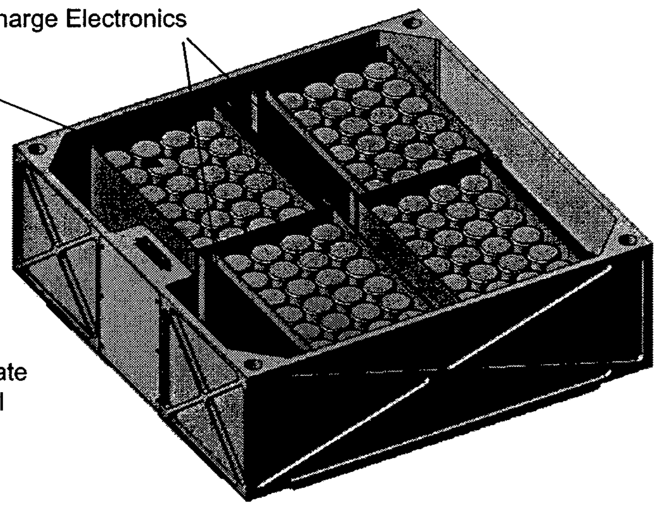


Possible Array Technologies

- Ridged Panels (shown)
- Flexible
- Inflatable
- Other

Module 3 is the solar array module. The array technology that will be used has not been selected. An issue with the ridged panels (shown) is that they can not be held in the stowed position without help from a second module located higher up on the stack of modules. One of the concepts of SCOUT is that the modules are designed to be completely independent from the rest of the spacecraft. Other technologies that may be a better fit for SCOUT are flexible arrays that roll up on each side, and inflatable array structures. The array module will also be equipped with any shunt electronics that are needed.

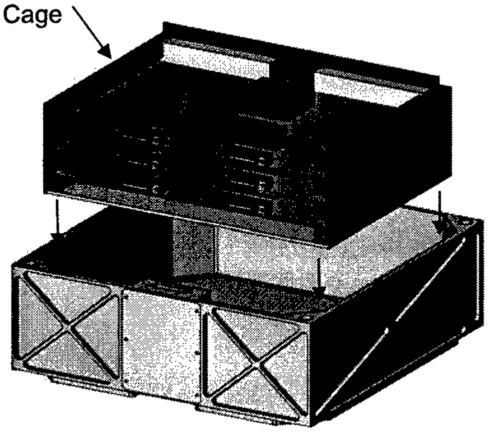
Module 4: Battery

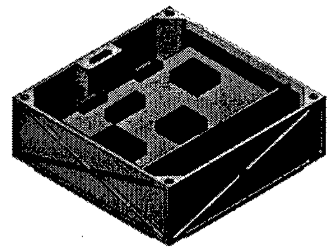


Module 4 is the battery pack. The module we are using can hold up to four 24 cell batteries complete with their charge electronics.

Module 5: Command & Data Handling

Removable
VME Cage





VME secondary
structure allows for
integration and test of
electronics before
module integration

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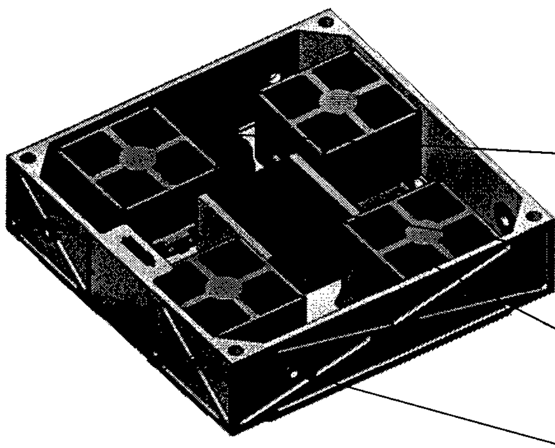
Module 5 is allocated to command and data handling. The concept for this module is to include secondary structure in the form of a removable VME cage. This allows the boards to be integrated and tested easily while still allowing for access to all boards. After all testing is completed, the entire VME cage assembly is dropped into the primary module structure and bolted in place.

Module 6: Attitude Determination

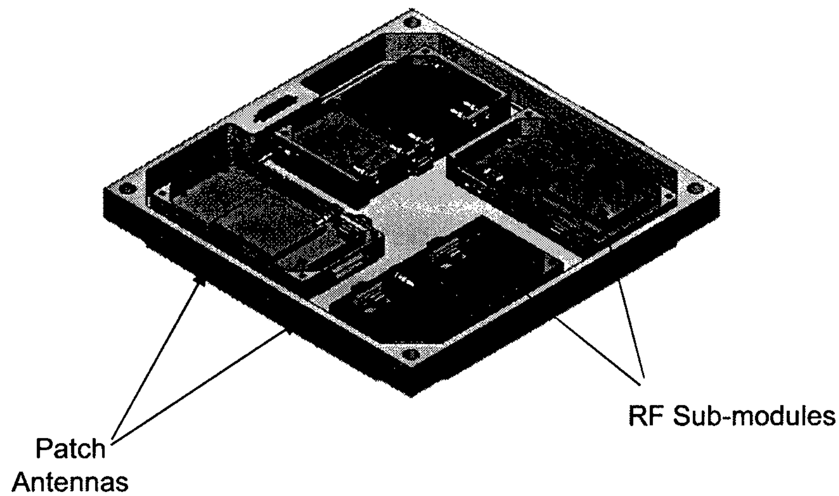


Contains all attitude determination components

- Gyrochip Array
- ADS Electronics
- 4 Star Trackers
- 4 Sun Sensors

The attitude determination module (Module 6) contains all the instruments needed for attitude determination along with the electronics and software required. The module shown here contains star trackers, sun sensors, and gyrochips.

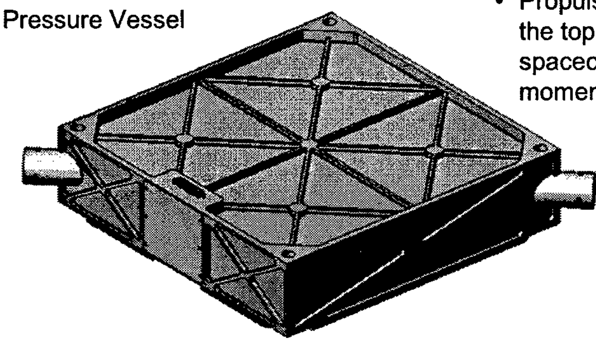
Module 7: RF Communications



Module 7 contains the equipment for RF communications. The components shown are from the NMRF transponder that is being developed by AeroAstro. All antennas needed will be included in this slice (patch antennas are shown).

Module 8: Upper Propulsion Module

Welded Titanium Pressure Vessel



- Identical to Lower Propulsion Module
- Rotated 90 degrees
- Propulsion modules are near the top and bottom of the spacecraft to provide a large moment arm for the thrusters

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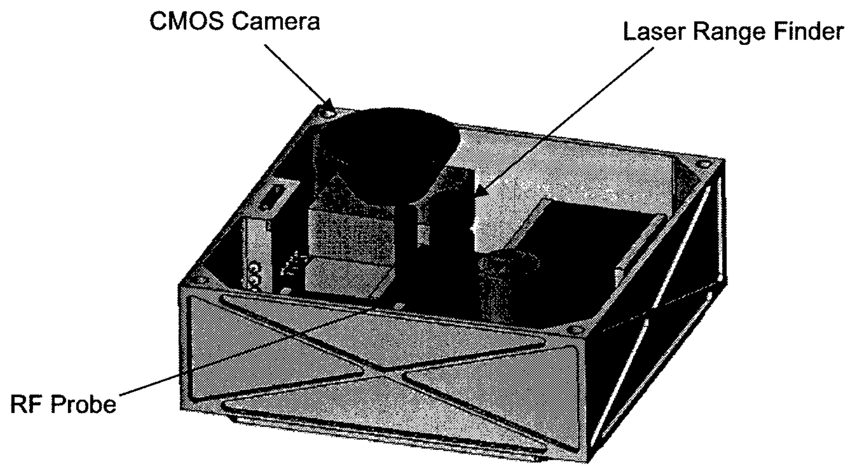
The upper propulsion module is identical to the lower propulsion module but it is rotated 90˚. Notice that the propulsion modules are as far away from the spacecraft CG as possible.

Module 9: Payload



The Payload Module is custom configured for each application

Module 9 is the uppermost module and as such is reserved for the payload. This module can be configured as to meet the customer's needs as long as all electronics communications entering and leaving the module are SCOUT compatible.

SCOUT Mass Budget

Slice Description	Estimated Mass	Margined Mass
Module 1 - PAF Interface	4.78	5.98
Module 2 - Propulsion - 4cm	3.80	4.75
Module 3 - Solar Arrays - 4cm	9.33	11.66
Module 4 - Battery - 8cm	7.37	9.22
Module 5 - VME Cage - 8cm	4.26	5.32
Module 6 - ADS - 6cm	3.47	4.33
Module 7 - Radio - 2cm	2.91	3.64
Module 8 - Propulsion - 4cm	3.80	4.75
Module 9 - Payload - 8cm	4.00	5.00
S/C Dry Mass	43.72	54.65
Propellant	8.4	8.4
S/C Wet Mass	52.12	63.05

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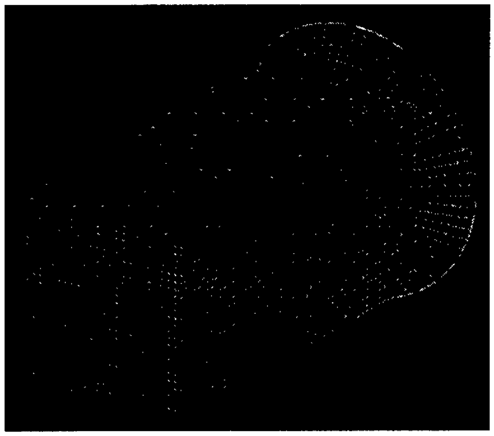
This mass budget shows the approximate mass for this SCOUT based mission.

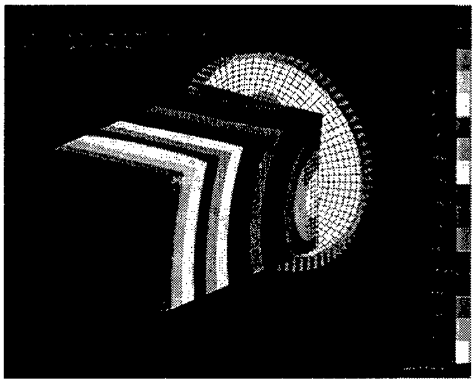
SCOUT Structure Analysis



- All Wall Thicknesses: 0.060”
- Material: Al 6061-T6
- Static Loads: 13 Gs in any axis
- Stiffness Requirement: 40 Hz
- Nonstructural mass evenly distributed
- Analyzed S/C Mass: 75 kg
- Analyzed S/C Envelope: 25 cm X 25 cm X 50 cm

Structure analysis was done with Patran/Nastran. The model assumes that the slices are fastened together in such a way that the spacecraft behaves as if it were made from a solid piece of aluminum. All walls were assumed to be 0.060” thick. Static loads and stiffness requirements were determined to accommodate most existing launch vehicles. All non-structural mass was distributed evenly throughout the spacecraft.

SCOUT Stiffness Results

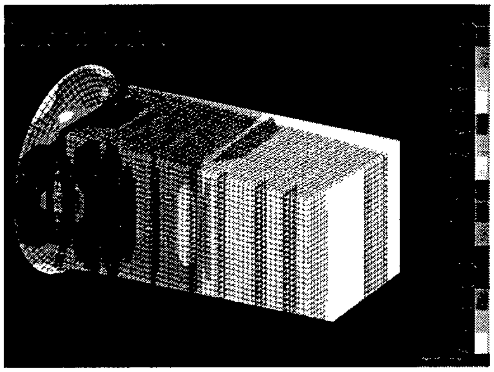


- Stiffness increases directly with baseplate thickness
- Stiffness Requirement: 40 Hz

Baseplate Thickness	Frequency
0.060"	69 Hz
0.200"	165 Hz

The stiffness analysis shows that the structure is in compliance with the stiffness requirement of 40 Hz. It is important to note that the frequency is directly related to the baseplate thickness. This means that if the stiffness needs to increase, the only thing that must be changed is the baseplate. This allows SCOUT to be a truly modular design because regardless of the launch environment or the mass of the spacecraft, the structure of the individual modules do not need to be modified.

SCOUT Stress Results



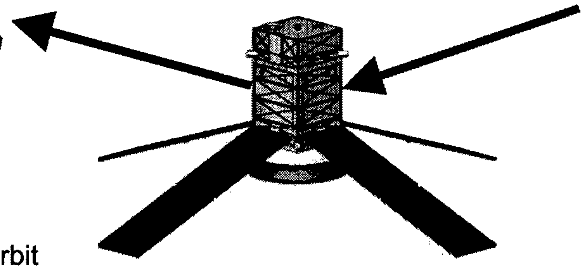
- Static Load Requirement: 13 Gs in any axis
- Safety Factor: 1.25

Baseplate Thickness	Stress (MPA)	Margin	Shear (MPA)	Margin
0.060"	79.69	3.64	47.06	4.51
0.200"	11.87	30.14	6.54	38.64

Stress results are similar to the stiffness results. Again notice that the stress margin increases directly with the baseplate thickness.

SCOUT Thermal Analysis

Spacecraft analyzed as a point mass
Thermal Mass: 20 kg (structure mass)



- GEO Orbit
- Coating: Black Anodize (0.86,0.86)
- Maximum Temperature: + 18.5° C
- Minimum Temperature: - 4.5° C

55 watts dissipated (orbit average)

A preliminary thermal analysis was conducted on SCOUT. The spacecraft was analyzed as a point mass with a thermal mass of 20 kg (approximate structure mass). Radiation absorbed and emitted was estimated using the surface area and the thermal properties of a black anodize coating. Internal heat generation was estimated at 55 watts. This thermal system was cycled though a typical GTO sunlight/eclipse cycle to find the maximum and minimum temperatures.

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System Overview

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This presentation provides an overview of the SCOUT microsatellite architecture with a focus on the characteristics of a SCOUT vehicle configured to fulfill the Escort mission.

Key SCOUT Design Feature: Modularity

- **The SCOUT system hinges on the adoption of uniform mechanical and electrical interfaces to allow continuous expansion of the envelope**
- **Another key feature of SCOUT is that the housing serves as the structure - this avoids the need to develop a custom framework for each mission**
- **This uniform housing provides rigidity and strength and also serves as a thermally conductive path to isothermalize the spacecraft bus**
- **A standard field joint allows any module to interface to any other module**
- **Each module has a "pass-through" electrical connector in a standard location that provides power, low speed data, and high speed data to and from each module for flexible standard "backbone" data interfaces**
- **This uniform electrical interface provides the ability to test any module in the field using the Master Universal Ground Support Equipment**

Modularity enables the Flexibility, Field Configurability, Scalability, and Extensibility of the SCOUT System

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The key to meeting the SCOUT requirements for flexibility, field configurability, scalability, and extensibility is MODULARITY. It is virtually impossible to meet all of these goals without a modular spacecraft bus architecture. The SCOUT modular architecture starts with a uniform mechanical and electrical interface that is common to all modules. This common interface provides the framework for realizing an extensive set of potential modules that can be combined in nearly limitless combinations to tailor a SCOUT-based vehicle to meet a single set of unique requirements.

As the uniform mechanical interface evolved, it gave rise to several features which became core elements of the common mechanical design. The housing of the SCOUT module doubles as the SCOUT load-bearing structure, so there is no requirement for a separate primary or skeletal structure. The SCOUT modular housing is referred to as the EXOSKELETON, which provides both enclosure and structure in a single element. This robust structure is stiffened with rib features to provide an extremely stiff, strong, load-bearing capability. This structure also serves as a thermally conductive sink for dissipating heat from the enclosed SCOUT electronics. Assuming that this structure is aluminum, this highly conductive housing serves to isothermalize the bus, spreading excess heat from those modules which dissipate high power levels to those modules which are inherently low-power devices. This standard field joint is intended to allow any given module to mate to any other module. This feature provides tremendous flexibility in the topology of a selected bus.

Each module has a "pass-through" electrical connector in a uniform location; this

Notional SCOUT Modules (1 of 2)

Module Category	Module Name	Module Description	Escort	Tactical GPS	Tech Dem-Val
C&DH	Avionics I	FPGA			Y
C&DH	Avionics II	FPGA, CPU		Y	
C&DH	Avionics III	FPGA, CPU, Mass Memory	Y		
Communications	S-Band Transponder	Subset of ST5 X-Band Transponder	Y		
Communications	X-Band Transponder	Based on ST5 X-Band Transponder			
Communications	INMARSAT-M Flight Modem	Virtual ground station		Y	Y
Communications	TDRS Flight Modem	Virtual ground station			
Communications	Intersatellite Laser Transponder	Virtual ground station			
Communications	Ground Laser Transponder	Goes on relay			
Communications	Crypto	NSA encryption and decryption	Y	Y	Y
GN&C	GPS Receiver	APL GNS-II+, must work in GEO	Y	Y	Y
GN&C	Attitude Determination	4 star trackers, 4 medium sun sensors, 3 one-axis gyros, 1 three-axis magnetometer, associated circuitry	Y	Y	Y
GN&C	Magnetic Actuation	2 torque rods, 1 torque coil, associated circuitry		Y	
GN&C	Wheel Actuation	18 micro wheels arranged in groups of 6, one group per axis		Y	
Propulsion	Divert	1 divert thruster, 4 ACS thrusters, associated tankage			
Propulsion	Maneuvering	6 or more ACS thrusters, associated tankage	Y		

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The Mission Study conducted in the opening weeks of the Phase I SBIR effort identified three leading contenders for candidate SCOUT missions. These missions were the SCOUT Escort (described previously), the SCOUT Tactical GPS (used to augment or temporarily substitute for operational GPS satellites), and the SCOUT Tech Dem-Val (a general purpose SCOUT spacecraft used for testing new technologies in space). In the Dem-Val scenario, SCOUT serves as a vehicle to gain flight heritage, i.e., to prove that a payload or subsystem can survive launch and operate in space. In some cases, limited on-orbit operations are required.

For each of these three missions, a notional concept was quickly developed to identify the SCOUT modules that would be required to realize each version of SCOUT. This chart (and the following chart) summarizes the results of this survey. The goal of these charts is to identify "common" modules and differentiate them from "mission unique" modules. Some entries may be misleading or confusing, however, so we will quickly review this table.

This chart shows that the Technical Demonstration-Validation mission (TDV) requires only an Avionics I Module, the Tactical GPS mission (TGPS) requires an Avionics II module, and the Escort mission requires an Avionics III module. This leads to an impression that there is no "common" avionics module, howver, this is NOT the case. The Avionics I module is a subset of the Avionics II module, and the Avionics II module is a subset of the Avionics III module. Therefore, the Avionics I module is common to all three missions and the Avionics II module is used by two out of three missions. Furthermore, upon closer examination, it was concluded during the conceptual design effort that Escort only requires an Avionics II module.

Notional SCOUT Modules (2 of 2)

Module Category	Module Name	Module Description	Escort	Tactical GPS	Tech Dem-Val
Power	Battery	Chemical battery	Y	Y	Y
Power	Solar Panel Array	4 multiply deployable solar panels, probably non-tracking, associatd electronics	Y	Y	Y
Launch Vehicle Interface	ESPA PIM	Payload Attach Fitting (PAF) Interface Module	Y		Y
Launch Vehicle Interface	SHELS PIM	Payload Attach Fitting (PAF) Interface Module			
Launch Vehicle Interface	RASCAL PIM	Payload Attach Fitting (PAF) Interface Module		Y	
Launch Vehicle Interface	ASAP5 PIM	Payload Attach Fitting (PAF) Interface Module			
Launch Vehicle Interface	Separation	To separate the main SCOUT modules from the PIM	Y		
Launch Vehicle Interface	External Structural Stiffeners	For extra stiffness on some launches			
Ground Support Equipment	MUGSE Base Plate	Master Universal Ground Support Equipement Base Plate	Y	Y	Y
Payload	Proximity Inspection Sensor Array	One or more visual cameras, one more IR cameras, RF Probe, and either laser range finder or near field radar	Y		
Payload	GPS-Lite	Generates and transmits GPS signals very similar to real GPS satellites		Y	
Payload	WSSP	Wafer Scale Signal Processing, for Space Based Laser			Y

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This chart shows that all three missions would use the rechargeable chemical battery module, which is nominally a Lithium Ion battery. The early concept for the battery module assumed an 8cm tall module that housed one to four "battery packs" that allow the user to scale the battery capacity to the mission. A more recent concept envisions a 2 cm tall battery module that is completely self-contained. In this concept, individual battery modules are incorporated in sufficient number to meet energy storage requirements. Better definition of the battery module(s) for each mission would require a better knowledge of the required capacity.

The chart also shows that all three missions would require the Solar Panel Array module. While it is clear that all would require some sort of solar array, it is not assured that all three would require the exact same module. Scalability of the solar array module is still being studied; as a more scalable design is developed, there may be greater differentiation between these missions with regard to selection of the solar array module.

For purposes of this study, it was assumed that four different variants of the Payload Attach Fitting Interface Module (PIM) would be developed. This chart shows that the Escort and TDV vehicles would both be launched using an ESPA PIM and that the TGPS vehicle would launch on a RASCAL PIM. In fact, no firm requirement exists for any of these missions; the actual selection of the PIM would be subject to Launch Vehicle selection. The important point is that the PIM does not have to be selected until shortly before launch. In fact, the PIM can be changed after integration if there is a change in Launch Vehicle.

SCOUT Modules for Escort

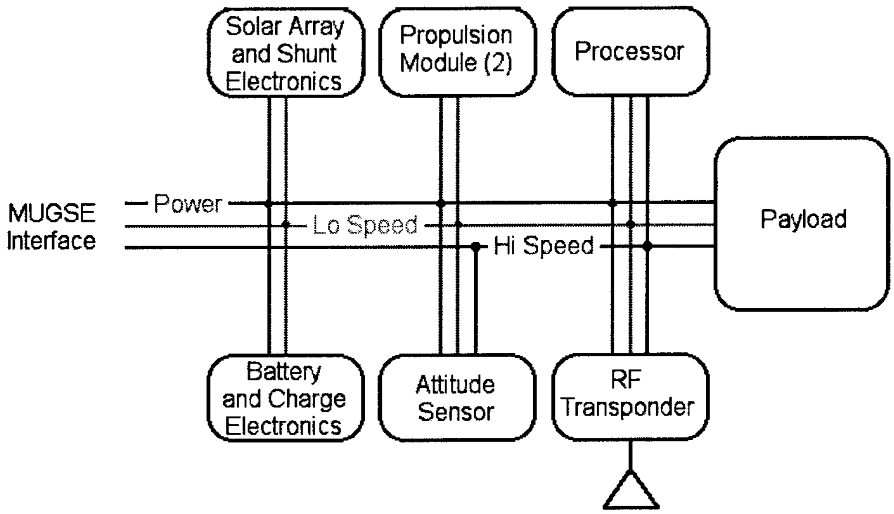
- Payload
- Communications Transponder
- Attitude Determination
- Avionics II
- Storage Battery
- Solar Array
- Propulsion (2)
- LV Payload Attach Fitting Interface

This chart details the precise modules envisioned for the Escort version of SCOUT as conceived during the Conceptual Design phase of the SBIR. The details relating to each module are discussed below.

At the bottom of the SCOUT stack is the LV Payload Attach Fitting Interface Module, or PIM. This module provides the mechanical interface between the SCOUT Escort Vehicle and the Launch Vehicle (LV). The selection of PIM is dependent upon the selection of LV, i.e., there is a unique PIM for each type of Launch Vehicle. The PIM also provides an electrical interface between the LV and the SCOUT Escort for detecting separation (only). Regardless of LV, the unique separation signal from each LV is translated to a uniform signal on the SCOUT Backbone. This Backbone signal is a fault-tolerant 2-for-3 voting scheme designed to provide assured knowledge of the separation event to the SCOUT Vehicle. The SCOUT Escort Vehicle uses this signal to begin its on-orbit initialization process. There is no other electrical connection to the Launch Vehicle, i.e., SCOUT cannot recharge it's battery or provide umbilical telemetry to the LV. This avoids a significant level of time-consuming interaction that accompanies development of a typical LV Interface Control Document. More detail about the PIM is provide in Section 05.

The next module on the stack is the first of two Propulsion Modules. Details of the Propulsion Module will be provided in Section 10. For now, suffice it to say that this module uses two four-way ceramic thruster nozzles located on opposite corners of the module. Nitrous Oxide has been selected as a propellant due to its properties as a safe, storable liquid. The Nitrous Oxide is stored inside of the Propulsion

Escort Block Diagram

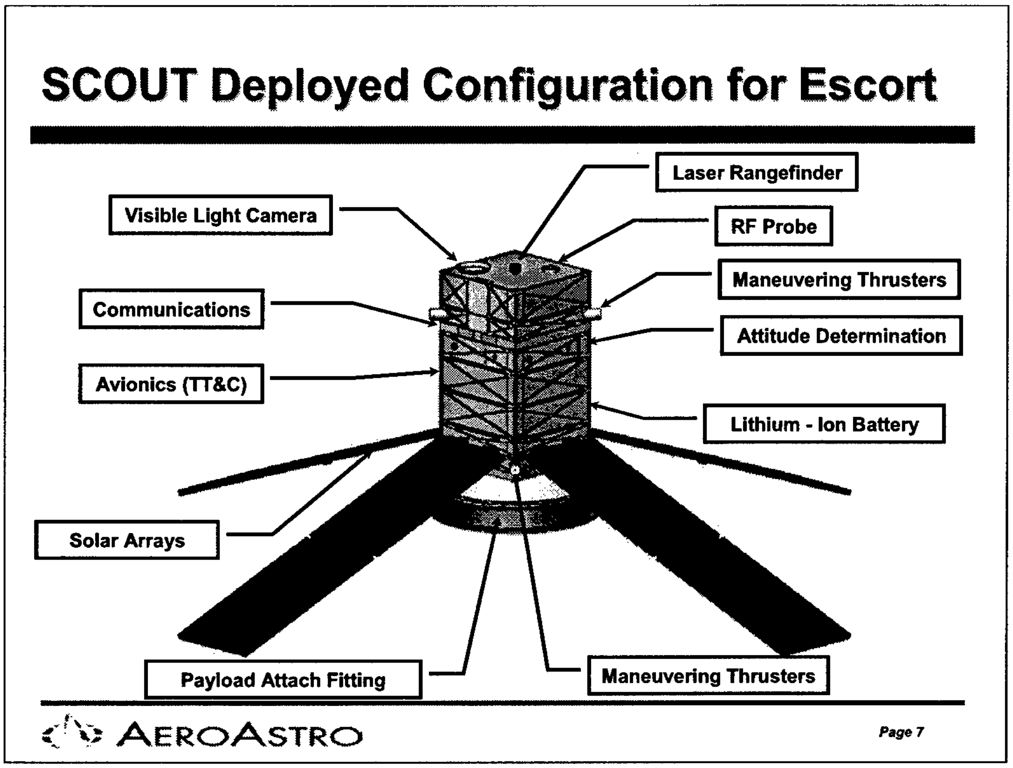


This chart shows a Block Diagram of the SCOUT Escort vehicle. Each of the modules described in the previous chart are depicted functionally in this diagram. Additionally, this diagram shows the interconnecting SCOUT Backbone bus that allows each module to communicate and draw power. It can be seen that the Backbone bus provides three interfaces: power; low-speed data; and high-speed data. More detailed information about the Backbone bus is available in Section 06.

The power bus, shown in red, connects to every module. The Solar Array module sources power to the power bus when the arrays are illuminated. The Battery Module takes power from the power bus when the arrays are producing excess power (storing the energy in Lithium Ion batteries), and sources power to the bus when the arrays are not capable of supplying sufficient power. All of the other modules take power from the power bus. Power switching is done locally in each module as commanded via the low-speed data bus.

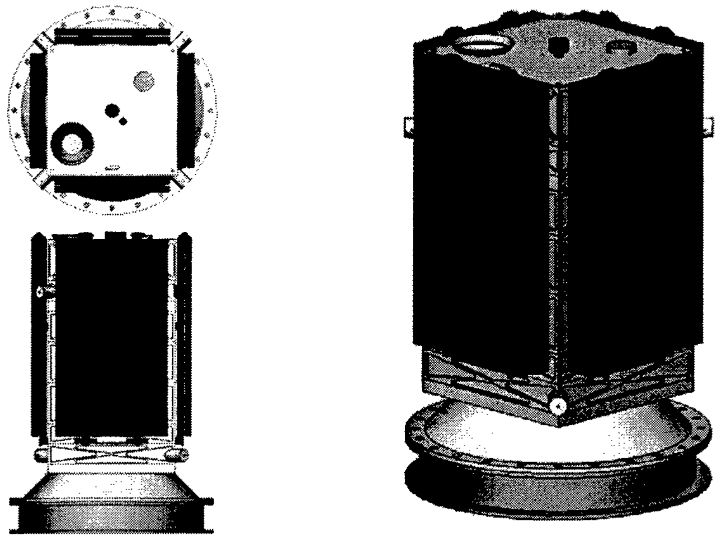
The low-speed data bus, shown in green, also connects to every module. This data bus, capable of handling data rates up to 400 kbps, is used for sending commands to each module and sharing low speed telemetry data from one module to the next. Data is transmitted on this bus using the Inter-Integrated Circuit (I2C) data communications standard.

The high-speed data bus, shown in blue, only connects to selected modules. This bus is capable of handling data rates up to 10 Mbps using the IEEE 803.2 (Ethernet) data communications standard. For Escort, it is anticipated that only the Attitude



This image depicts the deployed configuration of the SCOUT Escort Vehicle. It shows the location and orientation of all modules described on page 5 of this presentation, as well as the location and deployed configuration of the solar arrays, thrusters, and antennas. What is not communicated in this view is a sense of the scale of the vehicle, which is actually fairly small. Note that the scout modules are each 25 cm X 25 cm (10" X 10"), with heights varying from 2 cm to 8 cm.

SCOUT Stowed Configuration for Escort



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This graphic depicts the top, side, and orthographic views of the stowed SCOUT Escort vehicle. Looking down from the top of the vehicle, it can be seen that the protruding thrusters on the corners of the two propulsion modules must be accounted for in the layout of the stowed solar arrays. The overall vehicle height is less than 50 cm (20"), not including the Payload Attach Fitting. At 25 cm X 25 cm X 50 cm, the SCOUT Escort microsatellite is about the size of an office wastebasket.

SCOUT Performance Summary for Escort

• Space Vehicle (SV) Dry Mass (w/ 25% margin):	55 kg
• Orbit Average (OA) SV Power (w/ 25% margin):	42 W
• Payload Mass (w/ 25% margin):	5 kg
• OA Payload Power (w/ 25% margin):	2 W
• Pointing Knowledge (3-axis, 3σ):	0.5 deg
• Pointing Control (3axis, 3σ):	1.5 deg
• Mission Lifetime:	1 year
• Orbit:	GEO
• Downlink Data Rate:	76.8 kbps
• Uplink Data Rate:	2 kpbs
• On-Board Data Storage:	256 MB

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This chart summarizes the high level performance characteristics of the SCOUT Escort microsatellite. This configuration of SCOUT shows a dry mass of 55 kg including 25% margin (typical for a conceptual-level design) and an orbit average power consumption of 42 watts, also including a 25% margin. Of this total, 5 kg and 2 watts are allocated for the payload. It is observed that these mass and power fractions would not be considered very commendable for a typical custom-designed satellite. There are several mitigating factors to consider when examining this metric, however. First, There is a mass and power penalty associated with the modularity of the design; a general-purpose modular spacecraft is never going to be as mass and power efficient as a typical custom design. Second, this design includes a great deal of margin due to the very modest level of effort expended on analysis to date; it is anticipated that these very conservative mass and power numbers will be reduced as the design is detailed. The pointing knowledge is expected to be better than 0.5 degrees, with a pointing control of better than 1.5 degrees. The vehicle is expected to have a mission life of better than one year in GEO, however, ultimate mission life would be limited by the frequency and magnitude of propulsive maneuvers, i.e., the vehicle will ultimately be propellant-limited. The Downlink data rate (from GEO) will be 76.8 kbps assuming a transmit power of 1.5 watts. It is possible to increase transmit power and therefore downlink data rate at the expense of power consumption. The uplink data rate is sized at 2 kbps. The onboard data storage is limited to 256 MB using the Avionics II C&DH module. An Avionics III module would have considerably more data storage (with associated mass, power, and dimensional impacts), however, data volume is ultimately limited by the downlink data rate and the time associated with dumping data.

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Requirements

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SCOUT Requirements Basis

- Usually, requirements are based on a single mission
- SCOUT requirements are developed on the basis of:
  - Applicability to a broad range of missions;
  - Low recurring cost;
  - Low mass and small volume;
  - Scalability and Extensibility;
  - Rapid response;
  - Easy field assembly
- All requirements must be consistent with identified boundary conditions, e.g., Launch Vehicle constraints
- Additional requirements are levied for the Escort mission as a case study to validate the SCOUT conceptual design

In other words, SCOUT requirements are, in a sense, worked backwards. Rather than optimizing a bus to a specific mission, SCOUT provides a low-cost bus that can handle a multitude of missions, and assumes that payload providers are willing to subject their instruments to SCOUT interfaces (thermal, mechanical, electrical). To prove the utility of the bus, AeroAstro applied a very strenuous mission to it - the Escort payload monitoring a GEO primary spacecraft. The listed requirements on this document apply to the Escort mission.

SCOUT Boundary Conditions

- Identified boundary conditions are:
  - Launch Vehicle compatibility
  - Restrictions of the SCOUT module architecture which affect:
    - Component size
    - Component placement
    - Thermal Limits
    - Power Limits
    - Subsystem component selection which affects performance
      - Propulsion
      - Attitude Determination and Control
      - Communications (link budgets)

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These are the boundary conditions that ultimately limit SCOUT performance, and thus the missions it can support.

Launch vehicle compatibility limits spacecraft volume and mass, as well as the orbit that can be attained.

Component selection is limited by what can fit into the volume restrictions of a SCOUT module, what is satisfied by the SCOUT power and data provisions, and what can operate within the SCOUT thermal environment.

Component selection inherently limits/defines subsystem performance.

Requirements Development

- SCOUT subsystem requirements are flowed down from restrictions imposed by:
  - Launch Vehicle envelopes and capabilities;
  - Selection of modular architecture;
  - Escort payload instrumentation
- 25% margins have been applied to all requirements

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The requirements presented here take the aforementioned boundary conditions into account, then further define them by application of the Escort payload specifications.

Launch Vehicle Restrictions*

5.2 Launch Vehicle Requirements	
5.21 Maximum Space Vehicle Mass:	75kg
5.22 Maximum Envelope:	
5.22.1 Height	500 mm
5.22.2 Width	440 mm
5.22.3 Depth	440 mm
5.23 Minimum Fundamental Frequencies:	
5.23.1 Axial	50 Hz
5.23.2 Lateral	40 Hz
5.23.3 Torisional	50 Hz
5.24 Maximum Quasi-static Loads:	
5.24.1 Axial	13G
5.24.2 Lateral	2.5G

**Condensed from Launch Vehicle study submitted 17 APR 03*

SCOUT Launch Vehicle Compatibility

➢By satisfying these restrictions, a SCOUT-based spacecraft would be compatible with requirements for launch on:

- Ariane 4 and 5
- Atlas II, III, and V
- Delta II, III, and IV
- Eurockot
- H-IIA
- K1
- Kosmos
- Minotaur
- Pegasus
- RASCAL
- Sealaunch
- STS
- Taurus

Modularity Derived Requirements

- ➢Each SCOUT module will be 25 x 25 cm x Z cm (where Z is a multiple of 2, e.g., 2, 4, 6, 8, etc.)
- ➢Each module is completely self-contained
- ➢Each module will pass through and have access to:
  - Power and electrical ground
  - Low speed data
  - High speed data
- ➢Connectors will enforce correct stacking of modules during assembly; some modules (e.g., propulsion) are rotatable
- ➢MUGSE testing will be modularized

RF Probe Specifications

RF Probe Description	Value
Component Specifications	
Full Width Half Power angle (deg)	30
Mass (kg)	1.50
Dimensions (cm)	17.8 x 8.9 x 6.2
Power Consumption (W)	6
Performance	
Max Data Rate (kbps)	2000
Nominal Data Rate (kbps)	30
Frequency Response	50 KHz to 18 GHz
Control Electrical Interface	RS-422
Environment	
Operating Temp Range (deg C)	-20 to +70

Visible Camera Specifications

Visible Imager	
Description	**Value**
Component Specifications	
FOV (deg)	5.3
Array Size	1004 x 1004
Pixel size(µm)	7.4 x 7.4
Mass (kg)	0.6
Dimensions (mm)	
Lens	63 Diam x 95 L
CCD	6.2 x 6.2 x 3.8
Power Consumption (W)	5.5
Performance	
Quantum Efficiency (p/e)	45% @ 490 nm
Max Data Rate (kbps)	3355
Nominal Data Rate (kbps)	67
Optical resolution	10-bit/pixel

Laser Range Finder Specifications

Laser Range Finder Description	Value
Component Specifications	
Mass (kg)	0.1
Power Consumption (W)	0.3
Control Electrical Interface	RS-422
Performance	
Minimum Resolution	6 cm at 1.5 km
Max Imaging Rate (Hz)	100
Spectral Response	860 nm
Environment	
Oper Temp Range (deg C)	-40 to +55

Subsystem Requirements*

Req. Description	Value	Units
Orbit Determination & ADCS		
Perigee altitude	35,800	km
Apogee altitude	35,800	km
Inclination	~0	deg
Inertial orbit determination accuracy	+/- 1	km
Relative orbit determination accuracy	+/- 10	m

**Note: These are mission specific requirements for the Escort payload*

For the Escort type mission, SCOUT will require a geosynchronous orbit with an inclination that approximately matches the target, near zero. Ground ranging must meet an accuracy of +/- 1 km and laser ranging to the target must meet +/- 10 meter accuracy.

Subsystem Requirements (Cont'd)*

Req. Description	Value	Units
Attitude and Proximity Maneuvering		
Pointing control roll	1.5	deg
Pointing control pitch	1.5	deg
Pointing control yaw	1.5	deg
Rate control roll	0.127	deg/sec
Rate control pitch	0.127	deg/sec
Rate control yaw	0.127	deg/sec
Mission lifetime (days)	60	days
Mission duty cycle	100	%
DV margin	30	%

**Note: These are mission specific requirements for the Escort payload*

The limiting payload element for both pointing and rate control is the visual camera.

As the Escort service is anticipated to be an early mission test and anomaly analysis tool, its lifetime is only required to be 60 days. Note that the allowable propellant budget provides 135 days, exceeding this requirement by over 100%.

Hibernation could extend total on-orbit time to up to a total of 1 year.

Subsystem Requirements (Cont'd)*

Req. Description	Value	Units
Attitude and Proximity Maneuvering		
Stabilization Method	3-axis	
slew required for relative ranging	2 per hour	
Three-axis translation required?	Yes	
Translation-free torque required?	Yes	
Pointing knowledge roll	0.5	deg
Pointing knowledge pitch	0.5	deg
Pointing knowledge yaw	0.5	deg
Rate knowledge roll	0.03	deg/sec
Rate knowledge pitch	0.03	deg/sec
Rate knowledge yaw	0.03	deg/sec

**Note:* These are mission specific requirements for the Escort payload*

Full three axis stabilization is required in order to continue to correctly point the payload as SCOUT moves relative to the primary.

Orbit control is three dimensional (three axis translation) and attitude control is required to be able to occur without coupling into an orbit adjustment (translation-free torque).

Pointing and rate control are 1/3 that of the pointing and rate knowledge requirements.

Subsystem Requirements (Cont'd)*

Req. Description	Escort	Units
Power		
SV Orbit Average Power Required	50	Watts
Payload Power Required	16	Watts
SV Bus Voltage	28 +/- 6	VDC
Battery Capacity	315	W-Hr
Peak Power	110	Watts
Thermal		
Max survival temperature	60	deg C
Min survival temperature	-15	deg C
Max operating temperature	40	deg C
Min operating temperature	0	deg C

**Note: These are mission specific requirements for the Escort payload*

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Orbit Average Power required is based on the operational demands of the payload as well as the spacecraft bus draw. A Depth of Discharge of 40% yields the battery capacity value.

The thermal requirements are based on the Li Ion Battery as the limiting component

Subsystem Requirements (Cont'd)*

Req. Description	Value	Units
Command and Data		
Mass Storage Capacity*	35	MB
Processing Power	40	MIPS
RF Uplink rate	1000	bps
RF Downlink Rate	76800	bps
Payload Data rate	5	Mbps
Payload Data Compression	25:1	
Time Source Accuracy	200	msec
S/C Telemetry Gathering Rate	1	Kbps
Analog lines required	64	
Discrete Lines Required	16	

Storage required for 5 seconds Visible and RF Probe at Burst Rates + 5 minutes Visible at Nominal Rate + 55 minutes RF Probe at Nominal Rate. Also allows downlink in 2 hours at 76.8 kbps rate

**Note: These are mission specific requirements for the Escort payload*

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Payload Data Rate - again, the most demanding element is the Visible camera. With a Burst Rate of 3.35 Mbps, plus margin, it requires a 5 Mbps total data rate capability. The SCOUT high speed line capacities of either 10 or 100 Mbps can easily handle this.

Time Source Accuracy requirement is based on the relative orbit knowledge requirement of +/-10 meters.

Telemetry gathering rate is based on typical small spacecraft state of health data collection rates as is the Uplink (command) data rate.

Environmental Requirements

- Cleanliness
  - Integrate modules in a class 100,000 clean room
- Electromagnetic Compatibility
  - Compliant to MIL-STD-461C
- Radiation Tolerance
  - To 1.25 years at geosynchronous altitude
    (1 year mission life plus 25% margin)
  - Typical radiation dose is 40 krad/year with 60 mil Al shielding*
  - This yields a tolerance requirement of 50 krad TID

* Source: Wertz, Space Mission Analysis and Design

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Escort Payload Description

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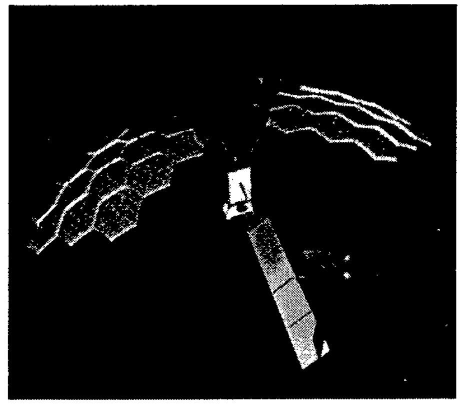
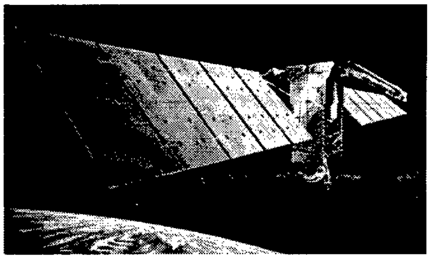
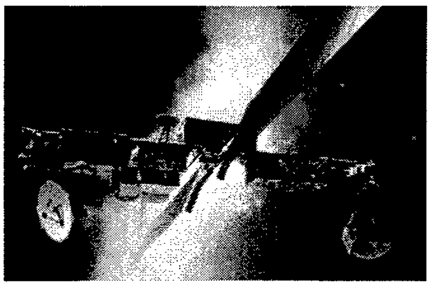
Overview

- Characteristics for GEO Spacecraft
- Potential Activities of Interest
- Potential Features of Interest
- Technical Performance Metrics
- Payload Module Design
  - Visual Camera
    - Specifications
    - Performance
    - Target Illumination
  - RF Probe Performance
  - Laser Rangefinder Performance
  - Module Integration



Characteristics for GEO Spacecraft





- Typical Dimensions:
  - Length (Solar Panels): 30 – 50 meters
  - Width (Antennas): 9 – 17 meters
  - Height (Antennas): 5 – 8 meters
- Typical Broadcast Frequencies (UL/DL):
  - Commercial:
    - L-Band: 1.635 – 1.66 / 1.535 – 1.56 GHz
    - C-Band: 5.9 – 6.4 / 3.7 – 4.2 GHz
    - X-Band: 7.9 – 8.4 / 7.25 – 7.75 GHz
    - Ku-Band: 14.0 – 14.5 / 12.5 – 12.75 GHz
  - Military:
    - L-Band: 1.635 – 1.66 / 1.535 – 1.56 GHz
    - S-Band: 2.65 – 2.69 / 2.5 – 2.54 GHz
    - X-Band: 7.9 – 8.4 / 7.25 – 7.75 GHz

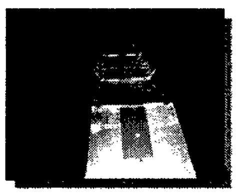
In order to understand the characteristics of the typical GEO spacecraft an Escort SCOUT might be tasked with surveying, an investigation was conducted to envelop the target features. The dimensions of an average communications satellite are dominated by the solar panel array and can be as large as 50 meters, though more commonly closer to 40 meters. Bus dimensions, in width, usually include the large reflectors, that themselves span diameters that average 3-5 meters. Height is largely the cross-sectional area of the bus, with the addition of small Earth uplink antennas. In military applications, such as the Milstar spacecraft shown, this height is much longer—necessary to accommodate the large payload complement.

Information for typical broadcast frequencies is publicly available from resources such as the ITU, FCC, or references such as Space Mission Analysis and Design (Wertz).

Potential Activities of Interest

QuickTime™ and a Cinepak decompressor are needed to see this picture.

- Nominal Operations
  - Deployments
    - Solar arrays (5 – 30 min)
    - Reflectors
      - Primary (5 min – 3 hours)
      - Earth-Deck (5 – 15 min)
    - Antennas, Booms
  - In-Orbit-Test
    - Antenna slew cuts (10 – 30 min)
    - Plume wake-field mapping (RF)
- Anomaly Response
  - Drive motor stuck, impaired
  - Thruster, actuator all-open/closed
  - Errant, foreign RF signals



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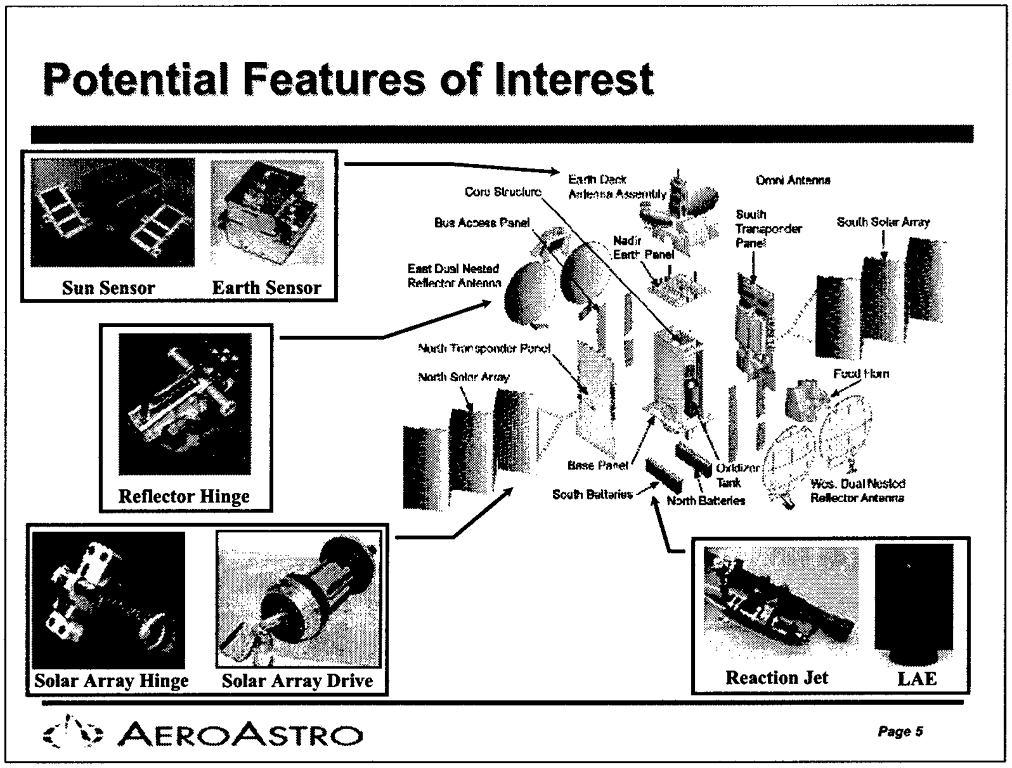
Within the operational lifetime of a GEO spacecraft, there are a number of significant events that would greatly benefit from an observer located in the near-vicinity, that was capable of sampling the events in the visual and/or RF domain. While the majority of activities on station are relatively slow moving and benign, certain key, "rapid" moments, such as deployments, represent critical periods of concern. A good deal of understanding could be realized, for example, on how certain flexible body modes are excited by the pyrotechnic release of a solar panel or the broad sweep of a slew cut taken to define antenna gain patters. This information could be provided as empirical data to design engineers that are responsible for specifying hinge damping ratios or locations of ejector springs. An obvious application of an Escort SCOUT would be to support a ground team's understanding and ability to perform anomaly resolution.



There are myriad potential features of interest on a large GEO spacecraft, including everything from solar array drive and panel hinges that might become impaired due to thermal issues or interaction with a prevalent environmental substance, to attitude determination sensors that may have lost alignment during launch or are experiencing odd heating effects due to an unforeseen sun geometry. Instances of electric propulsion systems becoming disabled on orbit is well documented, though understanding of the plume wake's interaction with the local RF field is not.

Technical Performance Metrics

➢ It is desired that the visible imager be capable of:
- Resolve target with 1 cm detail at 100 m stand-off ➔ Focal Length
- Sufficient light-gathering capability to image target ➔ F-Stop
- View the breadth of a large GEO spacecraft at a stand-off position of less than 1 km ➔ FOV
- Fast imaging times ➔ CCD detector efficiency, integration time
- Mitigate sensitivity to thermal noise ➔ Peak wavelength (pref. blue)

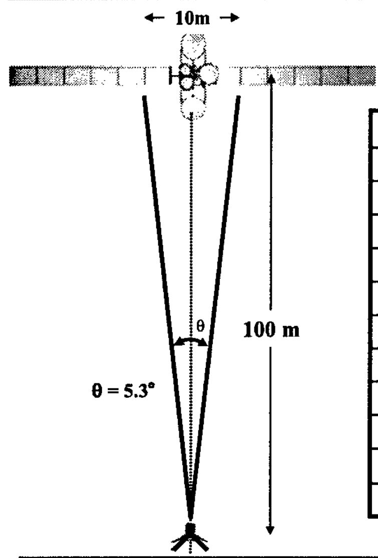
➢ It is desired that the RF Probe be capable of:
- Analyze RF spectrum from 1-15 GHz
- High resolution of discrete channels ➔ Dynamic range > 60 dB

➢ It is desired that the Laser Rangefinder be capable of:
- Determining stand-off position within 1 m accuracy at 100-1000 m

AeroAstro Page 6

In order to perform the conceptual design of an Escort Payload Module (EPM), we need to reconcile the characteristics of the typical GEO target spacecraft against the Concept of Operations in order to define some functional performance metrics. As a consequence of this activity, it was determined that a EPM consisting of a visual camera, AeroAstro RF Probe, and a laser rangefinder would offer a balanced, diverse suite of inspection capability. When the high-level metrics are translated to the EPM components, it is noted that the SCOUT vehicle must be able to conduct its inspection and situational awareness activities at a stand-off position relative to the target that is no closer than 100 m, relative to center of mass. As such, it is desired that the visual camera be capable of both resolving fine detail at the nearest approach and capturing the full extent of the largest dimension of the satellite from a relative location that is less than 1 km. As no additional light source will be available for this imaging activity, the optics and CCD detector must be sized appropriately to allow both strong light gathering potential and short integration times. Similarly, the RF Probe must be capable of analyzing the full spectrum of interest of typical GEO commercial and military spacecraft (previously identified, see chart 3). With a Concept of Operations that is predicated upon a nominal injection within 1 km of the target vehicle and a stand-off position no closer than 100 m, the EPM laser rangefinder must be capable of medium accuracy across this regime.

Visual Camera Specification



Specification	Value
Focal Ratio (f)	2.0
Focal Length/Aperture	40 mm / 20 mm
Field of View (θ)	5.3°
Detector Size	1004 x 1004
Pixel Size	7.4 μm x 7.4 μm
Pixel Depth	10 bits
Airy Disk Diameter	0.43 pixels x 0.43 pixels
Peak Quantum Efficiency (nominal)	45% at 490 nm
Sensitivity (nominal)	13 $\mu V/e^-$
CCD Well Depth (nominal)	60 ke^- (V) x 120 ke^- (H)
Total Camera Noise	42 e^- rms

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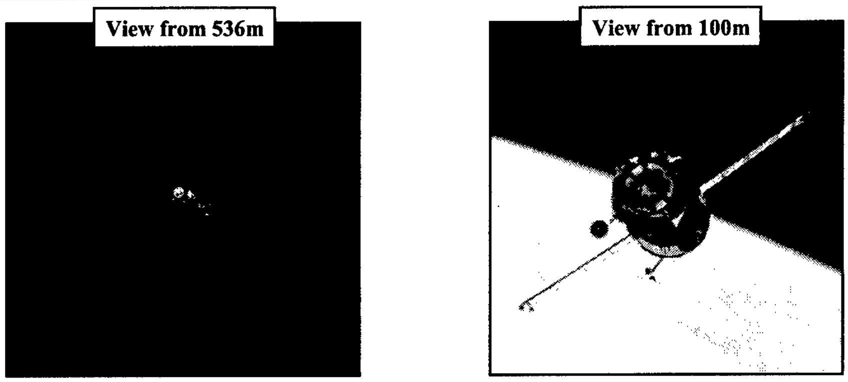
Based upon the delineated technical performance metrics, a visual camera specification was defined. Seeking, foremost, the capability to resolve 1 cm detail from a relative stand-off position of 100m, values for the optics were determined, including the focal ratio, aperture, and effective field of view (FOV). The Basler CCD detector was selected for its excellent sensitivity, low camera noise floor, and a peak quantum efficiency centered about 490 nm—in the blue and hence favorable for low sensitivity to thermal noise.

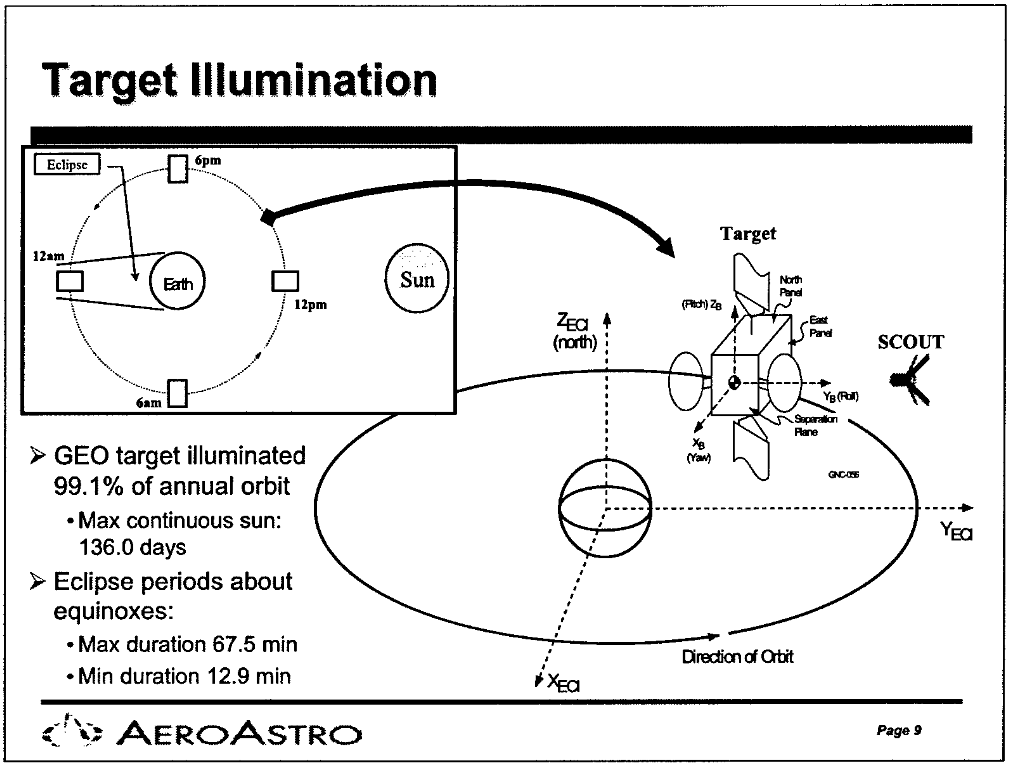
Visual Camera Performance

Specification	Low Resolution	High Resolution
Distance to Target	536.7 m	100.0 m
Size of Image Frame	50.0 m x 50.0 m	9.3 m x 9.3 m
Resolution	49.8 mm x 49.8 mm	9.3 mm x 9.3 mm



Target (Soyuz): 2.7m (Dia) x 7.9m (L) with 9.8m Wingspan

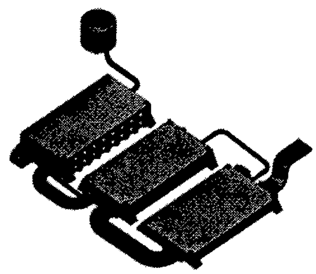
Based upon the optical specifications chosen for the desired resolution at nearest approach, in order to view the entire breadth of the largest spacecraft dimension (taken to be 50 m), the SCOUT vehicle must regress to a relative position of 536.7 meters. From this vantage point, the imaging resolution capability scales linearly and is thus reduced to approximately 50 x 50 mm. An example is provided of the Soyuz spacecraft that illustrates what the difference in stand-off position equates to from a visible perspective.

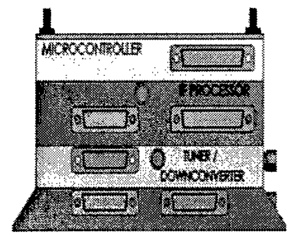


In order to assess the ambient environment of a typical GEO spacecraft, an Satellite Tool Kit (STK) simulation was created and a sun visibility analysis was conducted. As a result of this effort, it was determined that with the exception of a few, brief eclipse periods that occur during the equinoxes, the featured spacecraft is illuminated 99.1% of the annual orbit.

RF Probe Performance

- Frequency Range: 50 kHz -18 GHz
  - Variable IF filter bandwidth tuning
  - Spectral shape compared to templates stored in the MPU
  - Determine modulation type, date rate
  - Emitted signal power (with stand-off distance known)
- Dynamic range: 80 -100 dB
  - Linear channels
  - Logarithmic channels
- Self-calibration
- Ground-controlled or semi-autonomous
- Interface: RS-422
- Low power (6 W)





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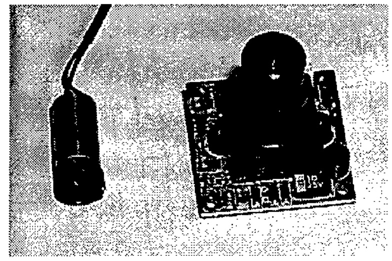
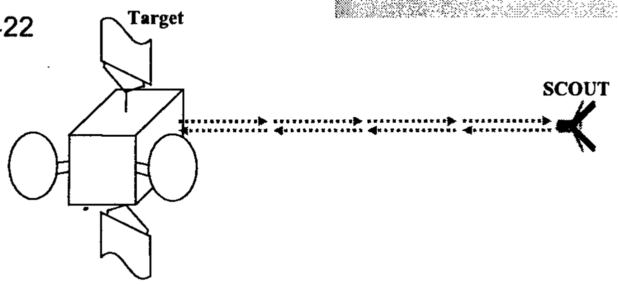
The AeroAstro RF Probe was chosen for its specific capability to resolve, analyze, and define spectral signatures associated with broadcast sources such as those of interest at GEO. The frequency range that can be sampled by the RF Probe spans that of the identified spectrum in chart 3 with great margin. In addition, it is capable of reconciling spectral characteristics against a known template database, determining the modulation type and, in conjunction with the laser rangefinder, quantify the emitted signal power. Within the signal captured by the RF Probe, the component also features a broad dynamic range across both linear and logarithmic channels that allows it to separate individual channels of information. In keeping with a preferred autonomy in the SCOUT architecture, the RF Probe is capable of self-calibration and operation, though designed for ground-in-the-loop control.

Laser Rangefinder Performance

- Operational wavelength: 860nm
- Beam intensity: 30 mW (eye safe)
- Sample rate: 10 - 100 pps
- Accuracy: 6 cm at 1.5 km stand-off
- Range:1 to1.5 km (Earth atmosphere)
- Power: 300 mW
- Interface: RS-422





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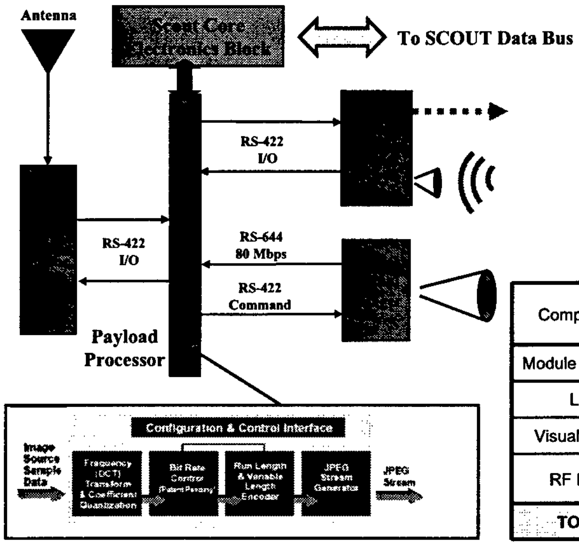
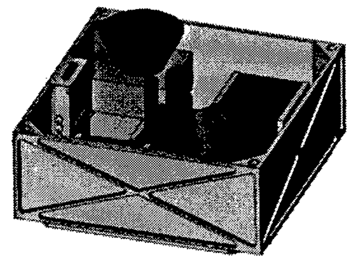
The laser rangefinder (LRF) was chosen for its ability to resolve position with high accuracy, across a broad range of stand-off positions. Because the Concept of Operations of the SCOUT Escort vehicle specifies for an injection from 1 km and a nominal 536 x 284 m proximity orbit, the long range capability is essential. As a power constrained microsatellite, the LRF represents a small impact on the overall budget.

Escort Payload Module Design





Components	Mass	Dimensions		
		Length	Width	Height
Module Structure	1800 g	*25 cm*	*25 cm*	*NA*
LRF	100 g	3.3 cm	3.3 cm	3.0 cm
Visual Imager	600 g	6.2 cm	6.2 cm	3.8 cm
RF Probe	1500 g	17.8 cm	8.9 cm	6.2 cm
TOTAL	**4.0 kg**	**25 cm**	**25 cm**	**8.0 cm**

The three components are easily configured into the nominal cross-section footprint, stacking to a unit-increment height of 8 cm. Utilizing an FPGA to both coordinate and manage the devices, data output from the visual camera, RF Probe, and laser rangefinder are packetized, compressed, and then fed on to the SCOUT data bus via the Core Electronics Block. For the visual camera, a motion-jpg software layer is included on this processor in order to compress both single image and burst video data before being relayed to the SCOUT CD&H system.

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Integration & Test

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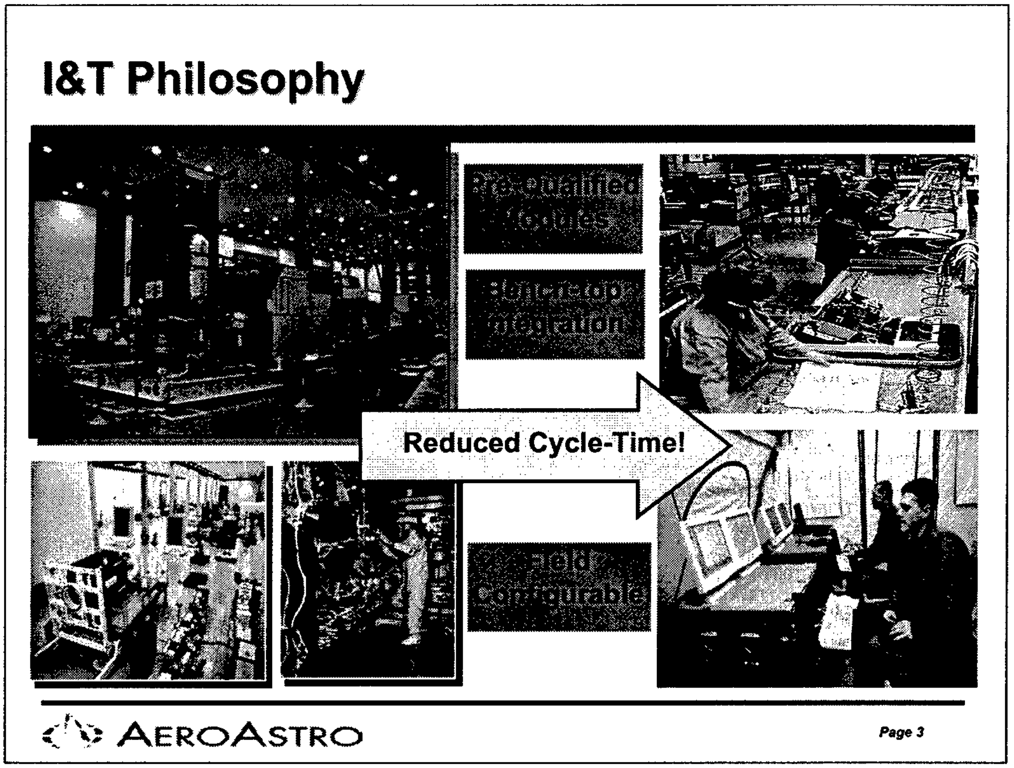
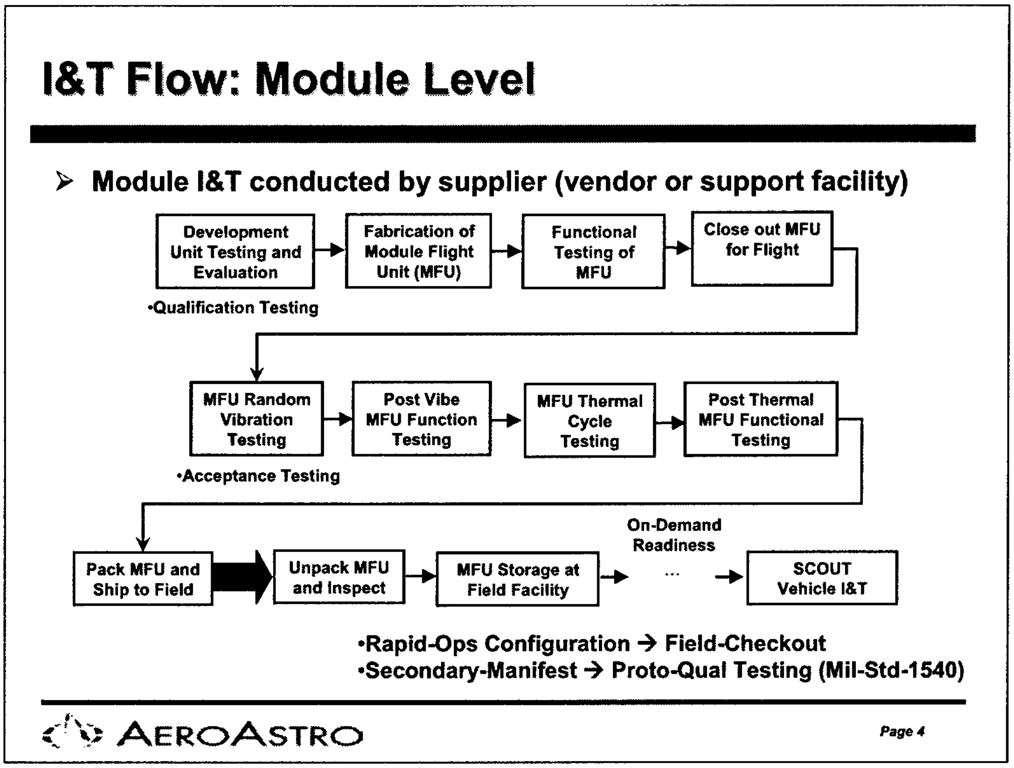
Overview

- I&T Philosophy
- I&T Flow: Module Level
- SCOUT Vehicle Assembly
- Ground Integration and Test
- MUGSE Operations
- Launch Site I&T Flow
- In-Orbit Test and Checkout Operations



One of the best ways to reduce the lead-time and cost of deploying small satellites is to reduce the level of testing and analysis required for launch. At this time, most military satellites are designed and built to the very strict requirements of MIL-STD-1540 and MIL-HDBK-343. While commercial and civil missions are not subject to the same, exacting standards for analyses and tests to be performed prior to a payload being accepted for a launch, they still incur typical design and AI&T programs that average three years. In order to leverage the responsive capability afforded by the RASCAL launch system, a vastly reduced set of testing and analysis standards need to be created for the SCOUT architecture.

SCOUT will utilize an AI&T process that, in much the same way a computer manufacturer, does not require complicated clean-room procedures, nor rely upon custom equipment for different assemblies of components.

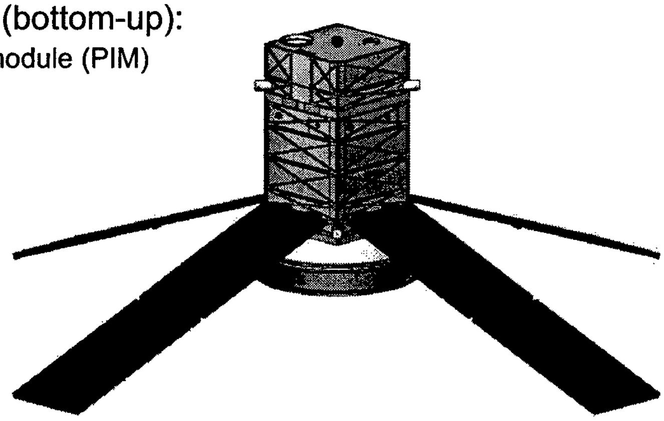


Fundamental to the SCOUT concept of AI&T is the implementation of open architecture standards and robust interfaces that can support a customer payload being integrated and checked out at the launch site. Upon receipt of a "build" order for a specific SCOUT vehicle, a dynamic AI&T process will commence.

The responsive nature of the SCOUT architecture is largely predicated upon pre-qualified component modules that greatly reduces AI&T schedule. As detailed, all qualification and acceptance testing of component modules is conducted by the supplier. Delivery to the Field Facility represents both the vendor's certification of manufacturing and testing quality, with a commensurate acceptance of risk by the customer. Should schedule allow, proto-qualification testing would be conducted per Mil-Std-1540.

SCOUT Vehicle Assembly

- Modules stacked from "bottom-up"
- Stack order determined by configuration
- Proposed order (bottom-up):
  - PAF Interface module (PIM)
  - Propulsion-1
  - Solar Array
  - Battery
  - CD&H
  - ...
  - n-3
  - Propulsion-2
  - Payload



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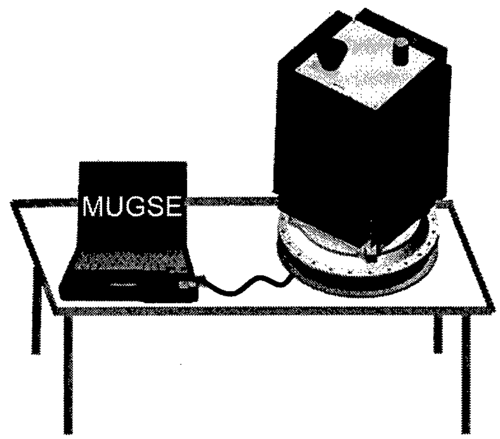
At the time of deployment order execution, the required SCOUT component modules (identified by the MUGSE) are layered from "bottom-up" depending upon the mission-specific configuration, typically beginning with the payload adaptor fairing (PAF) interface module (PIM), and working up towards the payload module(s), which complete the stack.

Spacecraft Integration & Test

- Master Universal Ground Support Equipment (MUGSE)
  - Used in all SCOUT integration and test activities
  - Equipped with an easy-to-use GUI for selecting tests and interpreting response codes
  - Used to upload and test the latest software and drivers
    - Individual modules
    - Fully integrated spacecraft
  - Provides power, commanding, and both low-speed and high-speed telemetry interfaces
  - Executes built-in test and diagnostic routines



SCOUT AI&T will only require use of a portable Master Universal Ground Support Equipment (MUGSE) to provide all process directions to technicians working in a field-deployable clean tent. The MUGSE is equipped with an easy to use GUI, bus power supply, and connectivity to both the low- and high-speed bus. Drivers and configuration templates will be available from a central, online repository or intranet server, that can be accessed by the support technicians as needed to support the specific stack build.

MUGSE Operations

- Bus continuity
  - High-speed data
  - Low-speed data
  - Power
- Module checkout
  - Validate results for Built-In Test and diagnostic routines
  - Certify dictionaries
    - Command
    - Telemetry
  - Functionality
  - Performance
- Stack configuration
  - Mass properties (e.g. MOI, CG)
  - PID control gains, slew rates, pulse widths

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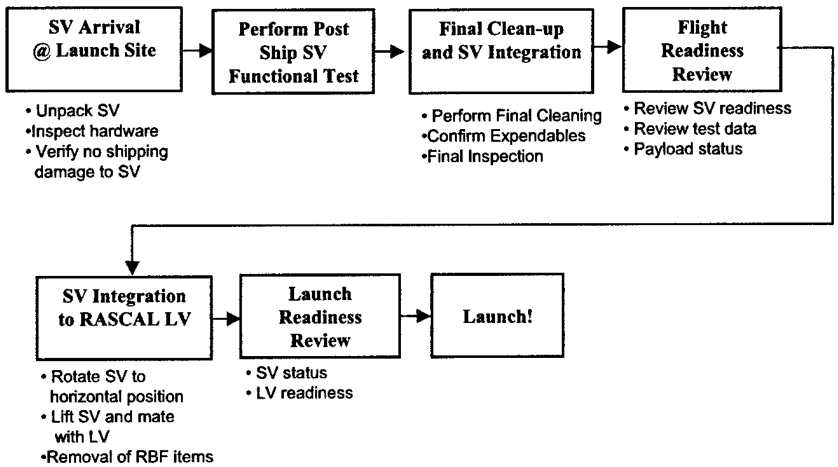
Leveraging the intelligent, self-diagnostic built in test (BIT) software that is managed by the MUGSE, when communicating with the SCOUT vehicle (via an access port), technicians using the MUGSE will conduct all module interrogation and check-out, certify dictionaries (command and telemetry), perform overall stack functional verification, and configure all mission-specific software drivers and parameters (e.g. mass properties and PID control gains).

Launch Site I&T Flow

- **SCOUT Vehicle (SV) Launch Preparations & Launch**



SCOUT launch site I&T is largely conducted in accordance with the specifications of the RASCAL or designated vehicle, however, the architecture is designed to require minimal re-touch and inspection once delivered.

In-Orbit Test Operations

- Execute BIT and diagnostic routines
- Perform module functional checkout
- Verify Spacecraft state of health (SOH)
- Verify Spacecraft configuration
  - Parameter and table uploads
- On-Orbit software upload if necessary
  - Operators can upload software patches or entire image to S/C
  - Operator can test uploaded software before committing to flash

In-Orbit Checkout designed to take less than 12 hours

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Another important aspect of rapid deployment is the ability of the payload to become operational virtually immediately upon reaching orbit. This implies that the SCOUT is active during launch, receiving time and position data from its GPS receiver and preparing to execute pre-loaded commands upon separation from RASCAL. Upon separation, the SCOUT would be able to autonomously de-tumble, acquire attitude, and perform an functional state of health (SOH) check-out of subsystems via the same BIT and diagnostic routines executed during ground AI&T. Precluding the need or desire to upload software patches or re-image the system, the SCOUT vehicle would commence normal operations and begin supplying data according to its operations plan.