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I-Cone® for Rapid Response and Low Cost Access to Space

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ABSTRACT

I-Cone® is an innovative approach to providing payload launch opportunities while at the same time taking advantage of the excess launch vehicle performance available with the Evolved Expendable Launch Vehicle (EELV). The genesis of the I-Cone® concept is the integration of a standard set of space vehicle subsystems into a standard conical launch vehicle adapter, in effect creating an "intelligent cone: or I-Cone. The I-Cone® is capable of providing payloads and small satellites a Fast, Frequent, Flexible and Affordable (F3A) access to space.

The I-Cone® concept is designed for use with the Delta IV and Atlas V (EELV) and is compatible with Delta II and Sea Launch Vehicles. The main I-Cone® structural components are derived from flight heritage payload adapters and separation systems, developed by Saab Ericsson (SE) Space, which minimizes the development risks and production costs. I-Cone® space vehicles can be essentially transparent to the Primary payload of a typical EELV manifest. The launch site processing flow for an I-Cone® has a "no impact" approach on the standard EELV Primary payload processing flow.

The I-Cone® space vehicle concept is suited for a wide variety of technology demonstration and short term operational missions. The baseline concept features typical payload resources of a 100 kg of mass, with 150 Watts of orbit average power, and a standard downlink data rate of 2.0 Mbps. The baseline I-Cone® Space Vehicle is capable of providing a pointing accuracy of 10-50 arc·sec, a propulsion system with 90 kg of mono-propellant Hydrazine, and a mission life exceeding one year. The use of I-Cone® for Low Earth Orbit (LEO) missions is emphasized in this paper, although Geosynchronous Transfer Orbit (GTO) launch can be accommodated by the I-Cone® also.

The modular approach to the I-Cone® space vehicle structure permits an extraordinary level of flexibility for meeting emerging specialized launch requirements. Micro-and nano-satellites can also be accommodated in an I-Cone® variation that incorporates a dispenser. Variations on the I-Cone® dispenser theme include a passive dispenser that provides additional propulsion and attitude control after separation from the launch vehicle. The I-Cone® concept can argument the potential return on investment for any EELV launch as it provides a cost effective and flexible solution particularly for Technology demonstration missions.

This paper will first present what needs the I-Cone® design addresses for access to space. This paper will also provide the generic mission requirements for the I-Cone® design, describe baseline I-Cone® implementation architecture, discuss payload accommodations and provide baseline implementation. Finally this paper will discuss potential mission designs for which I-Cone® can be applied to.

This paper is derived, in part, from a study performed in Reference 1.

ACCESS TO SPACE FOR SMALL PAYLOADS

There is an increasing demand from payload and small satellite providers to achieve space access in a timely and affordable manner. Coupled with this need is the trend, in the last decade, for larger launch vehicles with increasing performance resulting in lower costs per mass to orbit versus existing small launch vehicles which have a higher cost per mass to orbit cost. In conjunction with the launch vehicle trend is the continued reduction in the volume of electronics which provides an opportunity to provide higher performance for payloads in smaller volumes.

Today there are specialized solutions for secondary payloads that require a high degree of coordination for dedicated launches. The European ASAP and the U.S. ESPA are methods to avoid the specialized solutions for secondary payloads. The I-Cone® is a further advancement in accommodating secondary payloads by providing an autonomous satellite/dispenser system offering payloads or micro/nano-satellites volume, power, high data rate and storage with standardized interfaces to the launch vehicle. In addition to this the I-Cone® provides the standard interfaces required for the Primary space vehicle (SV) without impacting the SVs launch site integration.

The I-Cone® is a system to provide fast, frequent, flexible and affordable access to space for small payloads, either satellites (dispenser function) or payloads (satellite platform function). The I-Cone® replaces the satellite adapter that carries the primary manifested SV loads while utilizing the normally empty payload adapter volume for SV subsystems and payloads.

The baseline I-Cone® system provides a standard EELV interface to the launch vehicle and a standard separation interface to the primary SV. The I-Cone® can thus be included in EELV launches without any new interfaces being introduced. The modular approach of the I-Cone® design also allows for implementation on other launch vehicles, including Sea Launch and Delta II.

I-CONE® REQUIREMENTS

The mission of the I-Cone® is to provide a space platform for affordably delivering a secondary payload to Low Earth Orbit (LEO) while providing the primary manifested SV identical physical interfaces as would be accommodated on a dedicated launch. The I-Cone® mission requirements are shown in Table 1 for the baseline I-Cone® design.

The baseline orbit is derived from selecting the Defense Meteorological Satellite Program (DMSP) and the Global Positioning System (GPS) as the potential primary launch partners. These programs were selected because they have multiple launches scheduled, in the National Mission Model, in the upcoming years using the EELV. A propulsion system will be required to finalize orbit altitudes, prevent premature orbit decay and de-orbit. The baseline design will use single string components that can readily achieve acceptable one-year reliability and at the same time minimizing program development costs and schedule.

TABLE 1
MISSION REQUIREMENTS FOR BASELINE I-CONE® DESIGN

Mission Requirements Description	Baseline Requirement
Orbit	LEO (500-800 km)
Launch Vehicle	EELV
Mission Life-time	1 year post-orbit checkout
Mass (NTE – not to exceed)	545 kg
Propulsion	Required for operational orbit insertion, maintenance and de-orbit
Downlink Contact Time	60 minutes per day
Uplink / Downlink Encryption	National Security Space (NSS) payloads
Redundancy	Single string with selected redundancy

Some of the most essential and basic requirements of the baseline I-Cone® system are described below:

- The I-Cone® design shall be such that it has a minimal impact on the launch vehicle and the primary space vehicle
- The I-Cone® shall provide for a primary space vehicle load of at least 5000 kg with a Center of Gravity (CG) at 2 m above the separation plane
- The design of I-Cone® shall minimize additional height, over and above the standard adapters used today
- The design of I-Cone® shall, as far as possible, use standard and commercial off-the-shelf (COTS) components and standard interface
- The Atlas 5 and Delta 4 shall be the baseline launch vehicle options
- The baseline launch sites considered in this paper are Vandenberg and Cape Canaveral
- The inner structure design shall have a modular approach to make the system flexible for different payloads

- The I-Cone® platform shall provide sufficient payload capacity, regarding mass, volume and power to enable a reasonably wide range of applications
- The baseline attitude control shall be performed in three orthogonal axes
- The I-Cone® must comply with the stay-out zones required by the LV, Separation Systems and the Primary SV.

PAYLOAD PARAMETERS

The baseline I-Cone® is an autonomous space vehicle capable of supporting payload sensor(s) and electronics. The baseline payload parameters chosen to develop the I-Cone® design assumed for this paper are summarized in Table 2 and represent a nadir viewing imaging payload. These Payload parameters represent an envelope of candidate payloads that Swales Aerospace assessed based on a number of public domain mission requirements and represent in part high performance nadir viewing payloads. These payload parameters where recognized as the major drivers for the I-Cone® performance.

TABLE 2
DRIVING PAYLOAD PARAMETERS FOR BASELINE I-CONE® DESIGN

I-Cone® Payload Requirement Description	Baseline Requirement
Payload - Sensor Volume	720 x 660 x 320 mm
Payload - Electronics Volume	220 x 220 x 180 mm
Total Mass	100 kg
Field of View	±15 deg min
Available Power (Orbit Avg.)	150 Watts
Supply Voltage	+28 ± 6 V
CC&T Data Handling I/F	RS-422 and/or MIL-STD-1553B
Discrete and Analog Inputs	6 each
Discrete and Analog Outputs	6 and 12
Clock/Time Reference	10 mSec w/r UTC
On-Board Data Storage	5 Gbits min.
On-Board Data Transfer Rate	48 kbps average
Max Bit Error Rate	10^{-5}
Temp. Limits Operational	+40 to 0 °C
Temp. Limits Survival	+60 to -20 °C
Launch Mode Power Configuration	Off, Survival Power enabled
Pointing Control (3 axes)	±0.3 ° (3-σ)
Pointing Knowledge (3 axes)	±0.1 ° (3-σ)

ORBIT CHARACTERISTICS

The I-Cone® concept specifically targets use of the Evolved Expendable Launch Vehicle (EELV). Based on the requirement that I-Cone® be manifested with a U.S. National Security Space (NSS) primary space vehicle, only these launch opportunities were examined. It is certainly possible to consider other civil (NASA/NOAA) or commercial primary space vehicles if EELV performance is adequate and ride sharing agreements are developed.

Other factors that were considered when selecting target I-Cone® launch opportunities include the ability to achieve low earth orbit, regular launch schedule at predictable intervals and the Primary SV orbit.

The two programs that met these criteria and were selected as candidate launch partners: The Defense Meteorological Satellite Program (DMSP) and the Global Positioning System (GPS). DMSP operates in a high inclination, sun-synchronous orbit with an altitude of approximately 805 km. A minimum of two operational DMSP space vehicles are maintained at all times; resulting in regular replenishment launches. The GPS constellation requires a greater number of satellites. Twenty-four operational GPS space vehicle are required at all times. Current minimum GPS space vehicle lifetime is specified to be 7.5 years. Both these missions meet the criteria established and form the baseline for the orbits use to develop the I-Cone® Configuration.

The modified GPS transfer orbit and DMSP orbit were studied in order to determine worst case sun and eclipse conditions, driving the power and thermal subsystem design. In particular, the solar beta angle and eclipse times were assessed over a two-year period for the DMSP orbit, and over 4 years for the GPS orbit. Although the targeted mission life for I-Cone® is only one year, these extended scenarios were examined to capture the worst case phasing between the seasons and pertinent orbit parameters.

The solar beta angle, given for each orbit, is defined as the angle between the orbit plane and the sun line. These beta angle profiles prove to be very important when considering the basic steering configuration and coordinated solar wing articulation. For the DMSP orbit that is the 805 km circular sun synchronous configuration, a 90-degree beta angle is the best case given the solar array remains oriented in the orbit plane. For the GPS orbit with a zero degree beta angle the solar panels are oriented along the orbit normal and articulated with orbit position.

For these two orbits, various options exist for space vehicle pointing. Assuming a nadir-oriented payload, these options include orienting the solar array axis along the orbit normal or anti-velocity, or instituting sun/nadir steering. The latter involves coordinating the solar array angle with the space vehicle yaw angle.

I-CONE® CONFIGURATION

In order to establish the basic interfaces and nomenclature for I-Cone® Figures 1a and 1b presents an elevation view of the stacking of the space vehicles within the EELV fairing.

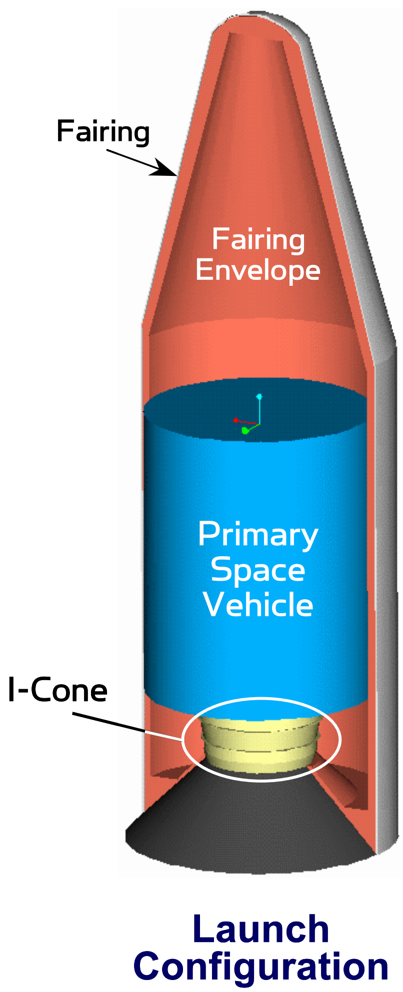

FIGURE 1A
I-CONE® LAUNCH CONFIGURATION

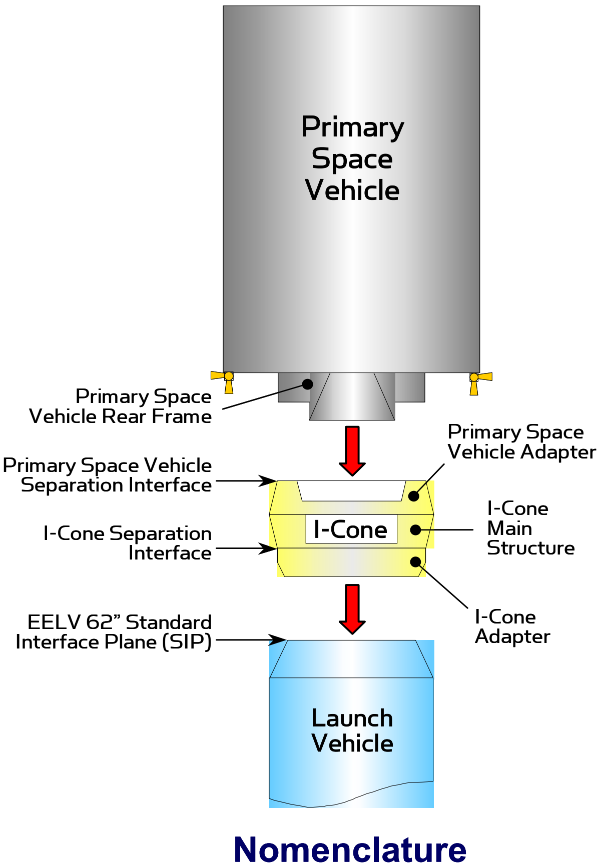

FIGURE 1B
I-CONE® LAUNCH CONFIGURATION

As Figure 1 illustrates the Primary SV is stacked on the I-Cone® standard SV adapter which incorporates an existing SE separation system. The Primary SV is hard mounted to the I-Cone® Main Structure which accommodates the Secondary Payload and I-Cone® Bus. The I-Cone® then interfaces with the I-Cone® Adapter via the I-Cone® Separation system which is again based upon a heritage SE separation system. The I-Cone® adapter is hard mounted to the EELV interface. All of these interfaces are based on existing designs leveraging extensive structural qualification. The overall I-Cone® stack has a small overall impact to the elevation position of the Primary SV and represents a small percentage of the overall vertical volume for the Primary SV. The launch site integration flow for the stacking of the I-Cone® is essentially the same in the standard flow with the exception that the Primary SV is stacked unto the I-Cone® system in lieu of the Primary SV adapter.

The mission timeline for separation of the Primary SV from the EELV is unaltered in the I-Cone® configuration. The Primary SV will in all cases be released first by initiating separation at the Primary SV to Adapter interface. The umbilical harness for the Primary SV passes through the I-Cone® stack similar to the current system. The I-Cone® subsequently will remain attached to the EELV until a separation command is sent to the I-Cone® separation system following standard maneuvers or specific mission profile adjustments as required.

I-CONE® IMPLEMENTATION

The basic structural building blocks of the I-Cone® rely on existing hardware elements from SE standard product lines as illustrated in Figure 2.

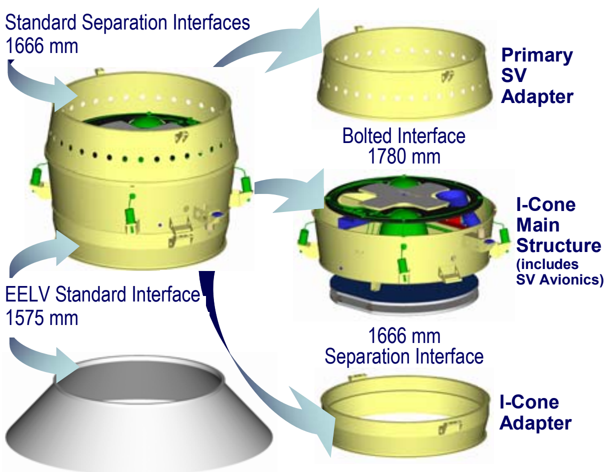

FIGURE 2
I-CONE® IMPLEMENTATION

The I-Cone® in the standard configuration accommodates all elements of the secondary payload and I-Cone® Bus internal to the I-Cone® main Structure with exception of the hydrazine thrusters and patch antennas. The layout of the I-Cone® Payload and subsystems has incorporated all of the standard stay out zones for both the Atlas 5 and Delta 4, standard separation system and the allotted existing Primary SV envelopes (Ref. 2, 3, 4)

Structural Subsystem

The structure subsystem of the I-Cone® is based on qualified adapters developed by Saab Ericsson for various launch vehicles with the addition of an inner structure for equipment and payloads.

The structure design objectives are as follows:

1. Use qualified adapters as outer structures to the maximum extent possible
2. Allow for further adapter extensions (e.g. other primary space vehicle interface)
3. Use standard separation systems
4. Maximize payload volume and attachment surfaces (see Table 2)
5. Minimize the influence on launch vehicle and Primary SV
6. Compatible with EELV requirements.

The structure subsystem of I-Cone® is also developed to meet the specific system requirements generated by the function of I-Cone® as adapter and satellite/dispenser. The inner structure concept consists of a cross of vertical panels enabling the use of the entire inner volume for equipment and payloads. The inner structure also incorporates a Propulsion panel to support the hydrazine tanks and Propulsion system components. The cross-panel solution combines two important features; it is very stiff and gives a large mounting area for equipment. The Inner Structure has the following functions:

1. Provide attachment for satellite equipment, harness and payloads.
2. Provide a radiating surface and transport vehicle to dissipated heat to that surface.
3. Transfer all mechanical loads from equipment and payloads to the I-Cone® Main Structure during ground operations, launch, separation and mission.

The Inner Structure consists of a cruciform panel structure comprised of four vertical aluminum honeycomb panels. These cross panels are attached in the middle to each other via extruded profiles bonded to the panels and bolted together. The cross panels are also attached to the Outer Structure by bonded and bolted profiles. A horizontal H-Panel is attached to the cross panels from below.

The heat dissipated from the satellite equipment is conducted to a dedicated spacecraft Radiator Panel placed on top of the Inner Structure.

Figure 3 provides an illustration of the inner structures with the mounted subsystem components.

I-Cone Subsystems

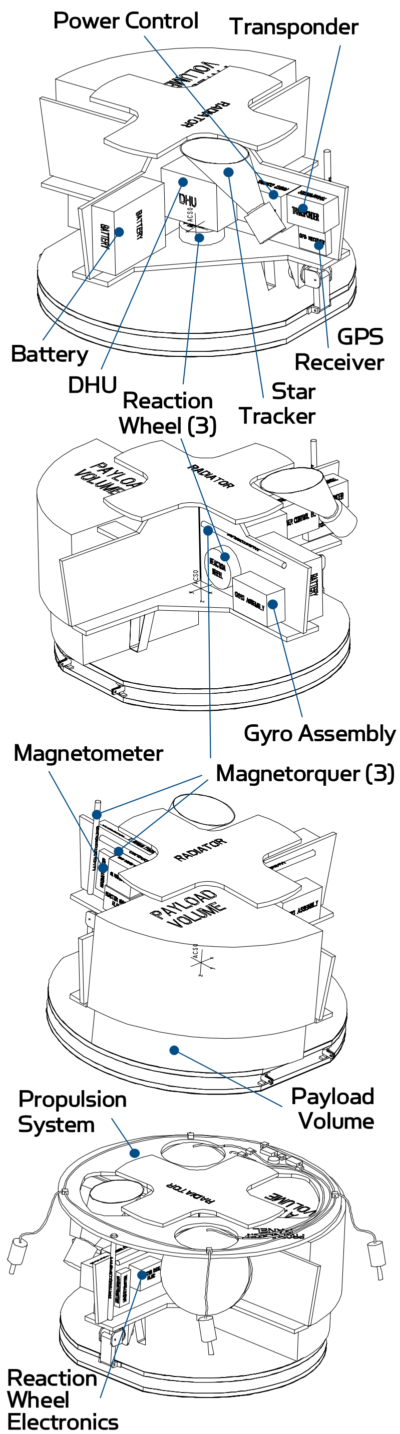

FIGURE 3
I-CONE® INNER STRUCTURE AND COMPONENT LAYOUT

Thermal Subsystem
The I-Cone® Thermal Control Subsystem (TCS) utilizes a passive, cold-biased thermal design consisting of dedicated I-Cone® equipment and payload radiators. This approach minimizes design, analysis and development time while meeting I-Cone® requirements.

The baseline thermal control coatings for the Radiator Panel mounted on the –Z side of the I-Cone® and the payload radiator surface located on the +Y side of the space vehicle is a high emittance white paint. The radiator areas are sized for hot environmental conditions and End-of-Life (EOL) properties. Silver Teflon or Optical Solar Reflectors (OSRs) can be used for missions that expose the radiators to extensive UV environments. In addition, this passive design philosophy allows the radiators to be easily adjusted as the design matures. Furthermore, thermostatically controlled heaters have been allocated to augment the passive TCS if necessary.

Command and Data Handling Subsystem
The Command & Data Handling Unit (DHU) concept is based on current designs for various programs, i.e. Protheus (currently flying on Jason-1 satellite), Rosetta and new developments. The Command and Data Handling can be customized based on specific mission needs and customer protocols. The DHU contains all the functions needed to perform satellite Command and Data Handling and Attitude Control tasks. It also has the capability of monitoring and controlling multiple payloads and to process telemetry from these payloads.

The DHU provides the following functions:

1. *Telecommand* reception, decoding and handling according to the ESA Packet Telecommand standard
2. *Telemetry* Transfer Frame generation, coding and modulation according to CCSDS 103.0-B-1
3. *Timing and Synchronization* management, providing a stable time reference that can be synchronized to external events
4. *Processing* capability with *Basic Software* to store and execute Application Software and Operating System software
5. *Packet Wire and UART Serial links* for communication with platform and payload equipment
6. *Discrete I/O system* for monitoring and control of various spacecraft equipment
7. *Power Converter* to allow connection of the DHU functions to a $28V_{DC}$ primary bus
8. *Distribution of regulated secondary power* to some spacecraft sensors.

Guidance Navigation and Control (GN&C) Subsystem

The I-Cone® GN&C is based on a three axis stabilization which yields a flexible solution and allows straightforward operations in a variety of typical LEO orbits. The attitude control software can be adapted to the specific mission needs.

The detailed requirements for the GN&C are derived from the mission types previously discussed. The following GN&C specific requirements have been derived:

1. The GN&C shall be able to operate in Low Earth Orbit (LEO)
2. The attitude control shall be performed in three orthogonal axes.
3. The pointing accuracy shall be 0.3° (3-σ).
4. On board pointing knowledge 0.1° (3-σ).

The baseline design of GN&C subsystem has been driven towards a flexible solution, which can be easily adapted to a large variety of missions. The key element is the star tracker, which is by far the most versatile attitude sensor. The star tracker opens up the possibility of designing a high accuracy pointing mission.

The baseline GN&C concept described herein is capable of achieving high accuracy pointing down to the arc-second region. With good alignment of the sensors, a pointing accuracy of 10 arc-sec is achievable.

The following G&NC sensors are baselined:

- Star tracker for accurate attitude reference
- Rate gyro assembly
- Six coarse wide-angle sun sensors
- GPS receiver
- Magnetometer (used for initial acquisition)

The following actuators are baselined:

- Four hydrazine thrusters
- Three Reaction wheels
- Three Magnetic torquer bars
- One Solar Array Drive Mechanism

The GN&C for I-Cone® will run in different modes depending on the phase of the mission.

The following operational modes are foreseen:

1. Safe mode
2. Initial attitude acquisition
3. Orbit acquisition
4. Orbit keeping
5. Normal operation
6. Decommissioning (Boost to lower orbit allowing for passive re-entry in 25 years)

Propulsion System

The I-Cone® requires a propulsion system for orbit acquisition and orbit-keeping. The propulsion subsystem is a monopropellant hydrazine system operating in blow-down mode. The system is an all welded design with the majority of components located on the propulsion subsystem deck.

The following main components are included in the propulsion subsystem:

- Two hydrazine tanks (45 liters each).
- Four 10 N hydrazine thrusters.
- Four thruster brackets
- Two Fill/Vent valves
- One Latch valve
- One Filter
- One pressure transducer

Electrical Power Subsystem

A Direct Energy Transfer (DET) power system, derived from heritage NASA programs, will be used to satisfy I-Cone® requirements. The system consists of a solar array and a battery that are connected directly to a power bus. A Power Control Electronics (PCE) unit performs all the power conditioning functions.

The solar array is a two panel deployable array with a solar array drive. Since the solar array drive is a single axis drive, it is not possible to keep the array normal to the sun under varying beta angle conditions by using the drive alone. The space vehicle attitude control system may also be required to yaw-steer the space vehicle to keep the sun line normal to the array. This will definitively be the case if I-Cone® were to be co-manifested with a GPS satellite. The resulting orbit will be elliptical with a nominal perigee altitude of 800 km and an apogee altitude of 20,200 km. If I-Cone® were to be deployed into a DMSP orbit (805 km, sun-synchronous), the problem is much less severe. In this case, under fixed yaw conditions, the maximum excursion of the sun vector from array normal is 32 degrees. The array power for the baseline design presented here was calculated for this case. The array utilizes 26.5% efficiency triple junction cells that have flown on multiple space vehicles.

A Super NiCd, 21-Ahr, 22-cell battery that has flight heritage and is used to provide power during eclipse. The battery depth-of-discharge (DOD) will depend on the eclipse duration. Using eclipse duration of 36 minutes (which is worst-case for the type of orbits we are considering), the maximum DOD is expected to be 33%. Given that the I-Cone® mission duration is expected to be about a year, battery performance should be excellent at the predicted DOD of 33%.

The PCE regulates power from the solar array, controls charging of the battery, and provides command and telemetry interface to the DHU.

Power regulation from the solar array is performed by the use of digital shunts. The PCE design is flexible and modular in that the number of shunts and the number of strings per shunt can be easily adapted to fit different missions.

Battery charge control is accomplished by temperature-compensated voltage (V/T) and current controller circuits. The batteries will be charged up to a ground selectable VT level until an on-board software-controlled Amp-Hour integrator indicates full charge. The charging is then automatically switched to trickle. Over voltage protection is provided by a hardware circuit that acts as a backup to the normal voltage control performed by the software. Although the battery is permanently connected to the power bus on orbit, a relay is provided to disconnect it from the bus for ground operations. To recover from an inadvertent opening of this relay on orbit, an ON command from the PCE is provided to close the relay (the relay cannot be opened by commands on orbit).

Command and telemetry interface to the C&DH subsystem is provided by a dual redundant MIL-STD-1553 data bus. All normal command and telemetry functions are performed using this bus. However, certain critical functions (such as resetting the processor) are performed by the use of hardware commands that bypass the 1553 bus.

The power bus voltage is determined by the battery voltage, and it varies over a range of 22 to 35 Volts. The power bus is capable of supporting a maximum combined bus/payload orbital average power at EOL (1 year) of 300 W, which includes about 16% margin over the current power requirements.

Communication Subsystem

The Radio Frequency (RF) Communications subsystem provides the RF interface between the I-Cone® space vehicle and ground stations. The Communications Subsystem performs command reception and delivery to the DHU and telemetry modulation and transmission to the ground.

The Communications subsystem as shown in Figure 4 consists of a transponder (with associated diplexer) that is connected to two omni-directional antennas via a hybrid. Placement of the two omni antennas on the nadir and zenith facing decks provide near spherical coverage and allows communications with the space vehicle in almost any orientation.

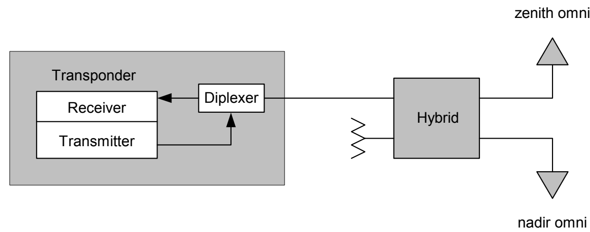

FIGURE 4
RF COMMUNICATION SUBSYSTEM
BLOCK DIAGRAM

All RF components chosen have design heritage over a variety of missions. For the baseline design, the widely used SGLS transponder, with 5W RF output, has been selected. Nothing in the I-Cone® design, however, precludes the use of other S-band transponders (such as the NASA standard STDN transponder, for example).

BASELINE I-CONE CONFIGURATIONS

The above subsystems are shown packaged in Figure 5. In the stowed configuration view the solar array is mounted on the launch vehicle side of the I-Cone®. The payload is mounted in one quadrant of the I-Cone® and allows for +/- 15 degree unobstructed sensor field of view (FOV) once the solar array is deployed. After released from the launch vehicle the solar array is deployed and the I-Cone® will perform initial attitude acquisition and spacecraft bus checkout. Once this is completed orbit acquisition would be performed and the payload would be turned on for checkout and the start of normal operations.

The mass budget for the stowed and on-orbit configuration is provided in Table 3. A contingency margin of 15% is used against the dry mass of the I-Cone® Bus. This is conservative since a number of the components assumed in the baseline I-Cone® configuration are heritage units. It is assumed the payload mass incorporates contingency mass. Obviously propellant mass can be reduce in order to accommodate additional payload mass if the mission delta velocity requirements are less than the baseline I-Cone® performance. It should be noted that a number of mass elements can be credited against the I-Cone® full mass capability since these elements would fly regardless of the use of the I-Cone®. These elements constituted approximately 76.7 kg and result in and "Added Launch Mass" of 506.1 kg which is a fairly small impact to a typical EELV mission.

TABLE 3
I-CONE® MASS BUDGET

SUBSYSTEM	MASS In-Orbit	MASS Launch	
I-Cone Spacecraft Structure	182.3	182.3	
I-Cone Adapter & Separation System	0.0	47.3	
Power Subsystem	44.8	44.8	
Data Handling Unit	11.0	11.0	
Communication Subsystem	4.2	4.2	
Harness	10.0	10.0	
Thermal Subsystem	5.0	5.0	
Attitude Control Subsystem	19.4	19.4	
Propulsion	17.8	14.6	
I-Cone Bus Dry Mass	**294.5**	**341.8**	
Contingency 15%	51.0	51.0	
I-Cone Bus Dry Mass w/Contingency	**345.5**	**392.8**	1
Hydrazine Propellant Consumable	90.0	90.0	
Payload	100.0	100.0	
I-Cone SV – Fully Integrated and Fueled	**535.5**	**582.8**	
Primary space vehicle Adapter		(50.4)	2
Primary SV Separation System		(18.9)	2
Adapter Harness		(5.3)	2
1575 mm mounting bolts		(2.1)	2
Added Launch Mass		**506.1**	

NOTES:

(1) Contingency does not apply to propellant and payload. It is assumed that the payload is either well developed or the payload provider is employing a separate risk mitigation strategy.

(2) These items are flown with the launch vehicle even if the I-Cone is not part of the launch, therefore no added launch mass.

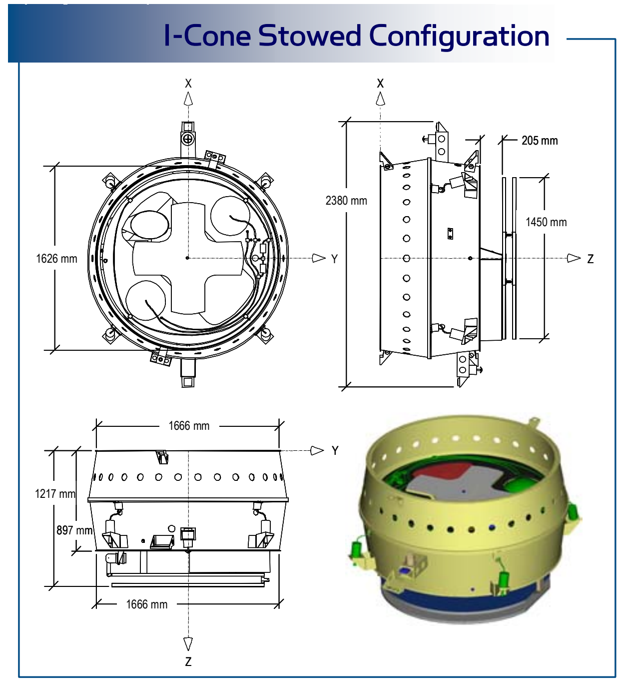

FIGURE 5A
I-CONE® STOWED CONFIGURATION

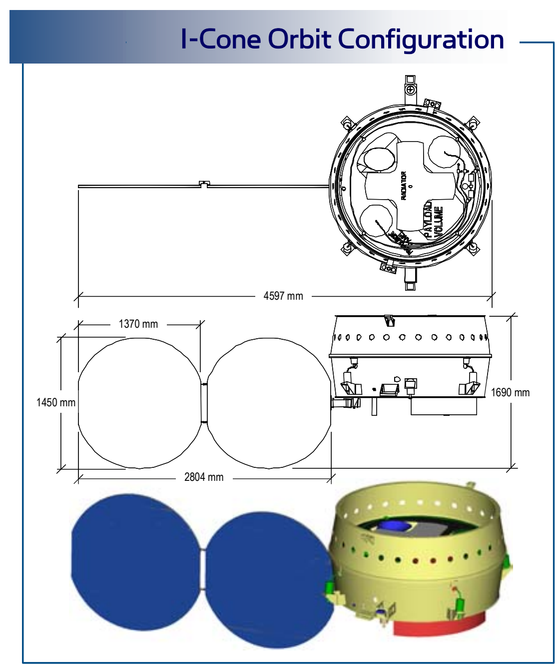

FIGURE 5B
ON-ORBIT CONFIGURATION

PAYLOAD ACCOMMODATIONS

Based on the payload parameters and interfaces established in the previous section and identified in Table 2 the I-Cone® design provides a full complement of resources to satisfy secondary payload needs.

Figure 6 provides an illustration of the accommodation volume for and internal and external payloads. The internal payload volume is approximately 0.30 m2 with a maximum instrument stowed height of 720 mm. Two additional volumes can be accommodated external to the I-Cone® and are approximately 365 mm by 270 mm by 380 mm. This provides significant flexibility to the payload manifest and with the baseline I-Cone® Bus resources the platform can adapt to a number of different mission profiles.

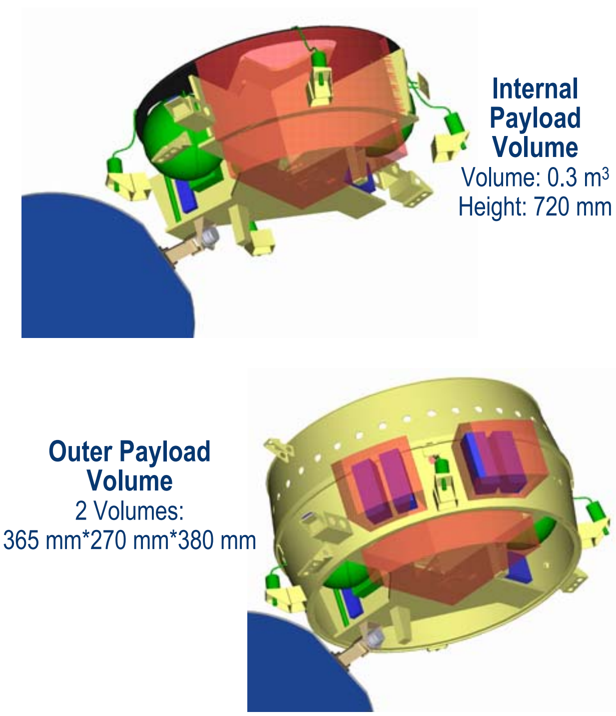

FIGURE 6
PAYLOAD ACCOMMODATION

The baseline I-Cone® configuration can be modified to provide a number of different dispenser platforms. Figure 5 illustrates an active and passive dispenser that can accommodate a number of different payload configurations. In the Active dispenser configuration outboard micro satellites can be deployed using standard release systems. Internal Nano Satellites can be accommodated adjacent to the Primary I-Cone® payload (Mission secondary). Figure 5 also illustrates a passive dispenser where it assumes that the Micro and or Nano satellites are separated from the I-Cone® which remains attached to the Launch Vehicle. In this scenario the I-Cone® is only a mechanical support system for the secondary payloads during launch and the Micro and or Nano satellites provide all of their own bus resources. The I-Cone® provides a highly adaptable platform that can be commissioned quickly to accommodate high priority missions with essentially no impact to the Primary SV.

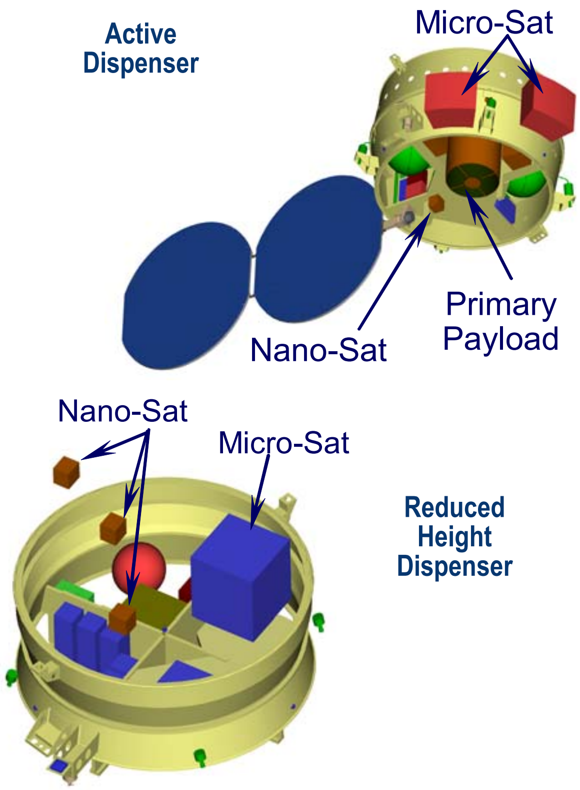

FIGURE 7
I-CONE® DISPENSER

SUMMARY

This paper has outlined the type of mission profiles and payloads that the I-Cone® architecture can accommodate to provide rapid response and low cost access to space via the use of heritage components with standard interfaces. Table 4 summarizes the baseline I-Cone® implementation strategy. The I-Cone® design can meet a wide range of mission profiles and payload needs by modifying the implementation strategy. The I-Cone® design also minimizes impacts to the Primary SV allowing for a streamline mission integration of the secondary payload.

TABLE 4
BASELINE I-CONE® IMPLEMENTATION

Item	Proposed Baseline Implementation
Mission Life	Baseline designed for minimum of 1 year with consumables for 2+ years
Launch Vehicle	Standard 1666 separation interface in baseline approach with optional 1194, 1663, or 937 interfaces available; 1575 bolted interface to launch vehicle included
Mission Profile	Worst-case orbit parameters used in evaluation to bound system design; On-board propulsion system provides for launch insertion errors, small inclination changes, drag makeup, and deboost, ΔV=410 m/s with 20% margin
Launch Readiness	Proposed 24 months for initial unit readiness for payload and 18 months for follow-on units
Payload Mass	Accommodated by the I-Cone inner structure and potential exterior mounted shelves
Space Vehicle Mass	Maximum launch mass limited only by primary mission requirements and launch vehicle, 506 kg baseline estimate
Payload Power	28 VDC, four (4) switched services with over and under current protection
Space Vehicle Power and Margin	EOL 300 W total system load prediction including 15% margin; (1) - 21 Ah battery max 33% DOD during worst eclipse; 2.6 m^2 deployable solar array is driven at orbital rate by single axis actuator
Thermal	Passive, cold-biased thermal design consisting of dedicated spacecraft and payload radiators; robust autonomous thermostatic control of resistive heaters
C&DH Architecture	Time tagged and event driven commands. Telemetry and I/O for payloads along with discrete analog/digital I/O
Payload Data Rate	Payload data rate accommodated: 48 kps with 1553 I/F and 2 Mbs with RS-422 I/F plus 16 kbs for SV housekeeping data
Data Storage	Standard DHU configuration can accommodate 5 Gbit and is expandable to 16 Gbit
CMD Interface	Both 1553B and RS-422 serial I/F provided
GN&C (3 sigma values)	Three Axis Stabilization Employed - Zero Momentum (3 RWA), Gyro and Fixed Head Star Tracker, Pointing Accuracy of 10-50 arcsec achievable with SIRU upgrade
Communication	S-Band System, Omni Antenna; 2 Kbps Uplink: 2 Mbps Downlink Data Rate provides > 6db Link Margin six 8 minute contacts/day provides 25% margin relative to budgeted 60 minute downlink time
Radiation	All avionics are resistant to > 30 Krads dose and are SEU tolerant
Magnetic Cleanliness	2.5 A-m^2 Peak, Single-Axis Transient SV Control Magnetic Moment and a S/C Harness E/M Dipole of 0.72 A-m^2 at the S/C Body Outer Surface are Worst Case Estimated Environments from SV
Redundancy and Fault Tolerance	SV bus is single string; some functional overlap and selective redundancy allows for increased fault tolerance. The over-voltage protection circuitry of the PCE provides for protection against exceeding the maximum bus voltage specifications and a degraded mode of operation for battery charging

REFERENCES

1) Final Report NRO000-01C-4369, NRO Space Accessibility by Using the I-Cone, May 2002.
2) Atlas Launch System, Mission Planners Guide, September 2001
3) Delta 4. Payload Planners Guide, October 2000
4) Evolved Expendable Launch Vehicle Standard Interface Specification Version 6.0; 5 September 2000