Free-Flying Altimeter Study

Final Report

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From: R. Keith Raney
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ABYSS was a proposal in 2001-2002 to NASA by NOAA and the Johns Hopkins University Applied Physics Laboratory (JHU/APL) for an altimeter to be mounted on an instrument pallet at the International Space Station (ISS) [*Smith*, 2002]. The ISS orbit inclination (~51^{o}) and non-repeat coverage would have been nearly ideal for the ABYSS mission, which was to provide high-resolution maps of oceanic gravity anomalies, from which bathymetric charts can be derived. Data from ABYSS would have provided approximately a four-fold improvement over gravimetry previously gathered by Geosat [*MacArthur et al.*, 1987]. The improvement is due primarily to two factors: the height precision of the delay-Doppler radar altimeter [*Raney*, 1998], and the choice of orbit.

With the ABYSS science and engineering precedent as a foundation, NOAA agreed in the fall of 2002 to fund a small study at the JHU/APL of a dedicated satellite as an alternative version of ABYSS. The principal objective of the Free-Flying Altimeter Study was to derive a conceptual design and cost estimate for a dedicated radar altimeter spacecraft – Abyss-Lite – that would meet the science goals of ABYSS. Our objective was to come up with a small and affordable spacecraft system after the precedent of Geosat whose payload would be a Ku-band delay-Doppler radar altimeter. This objective has been met.

Table 1. Altimeter Parameters Compared

Parameters	ISS-ABYSS	Geosat	Abyss-Lite
Mass (kg)	143	87	27.5
Input Power (W)	138	146	50
Altitude (km)	~400	800	800
Science DR (kbps)	36	10	25
Bandwidth (MHz)	320	320	320
Peak Tx power (W)	10	20	2.5
Avg Tx power (W)	3.9	2.1	1
Antenna area (m^2)	0.35	0.78	0.78
Antenna gain (dB)	35.6	37.6	40
Beamwidth (deg)	3.4 x 1.3	2.0	1.26
Burst length (ms)	2.097	-	4.8
Burst rate (Hz)	188	-	83.3
Interpulse period (usec)	-	980	-
Pulse width (usec)	65	102.4	75
Pulses/burst	32	-	64
Duty factor (%)	39	10	40
# range samples	256	63	128
Range intervals (m)	0.468	0.468	0.468

The study included spacecraft trade-offs, including especially the inventory of rapid development spacecraft maintained in the NASA GSFC RSDO inventory. Our conclusion is that it would be more cost-effective for JHU/APL to build a suitable spacecraft, than to adapt one of the RSDO candidates.

The study also reviewed launch costs. We applied a mass and size restraint on the ABYSS-Lite design with the intent that a Pegasus launch vehicle would suffice. This objective was met, with reserve.

The study hoped to show that the ABYSS-Lite instrumented spacecraft would cost less than ABYSS as originally proposed, which was approximately $60M. This goal was met. The resulting design came in at a cost of ~$56.7M, including the cost of the delay-Doppler altimeter (~$18M). Launch costs and reserves are estimated to be ~$30M and ~$9M, respectively. A propulsion system with sufficient delta-vee to maneuver the spacecraft from a non-repeating orbit to a repeating orbit would add ~$2.2M.

The Abyss-Lite study is summarized in [*Raney et al.*, 2003], which is incorporated as an attachment to this report. Details of the spacecraft design and the underlying trade-off studies and costing may be found in [*Reynolds*, 2003], which also is a part of this report.

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Abyss-Lite: Improved Bathymetry from a Dedicated Small Satellite Delay-Doppler Radar Altimeter

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Abstract - We describe the rationale, scientific basis, and implementation of a mission to map the ocean's bottom topography with a spatial resolution of 6 km based on a high-precision radar altimeter on a dedicated free-flying spacecraft.

INTRODUCTION

Local gravity deflections induce oceanic surface slopes. Slope measurements provided by the Geosat and ERS-1 radar altimeters furnish the best resolution oceanic gravity from space to date. The resulting bathymetric resolution is limited to about 25 km north-south, and to even poorer resolution of east-west slope components.

The spatial resolution of gravity anomalies is degraded in proportion to the distance between the gravimetric source and the observed signal. Hence, the in-orbit gravity measurements of GRACE and GOCE cannot observe anomalies at wavelengths smaller than their altitudes above the Earth, about 485 km and 250 km, respectively. Even though Abyss-Lite is also a space-based instrument, its resolution will be two orders of magnitude better, about 12 kilometers (full wavelength). This is the physical limit for any method (including ship-based gravimetry) that can be achieved through observations at the ocean's surface.

PHYSICAL PRINCIPLE

Gravity anomalies are caused by topographic relief on an interface between two volumes of differing mass density [1]. In the deep ocean, sediments are thin, and the basaltic sea floor crust is internally flat-layered, and so gravity anomalies reflect the topography of the ocean floor. Conversely, at continental margins the sea floor is nearly flat and sediments are generally thick. Beneath these sediments there may be basins or other geologic structures of interest. In such regions, surface slope signals are due primarily to topographic variations at the interface between crystalline rocks and their sedimentary overburden. The sediment/basement interface provides essential reconnaissance information for petroleum exploration. The correlation between slope and existing depth soundings readily distinguishes these two environments [2].

Sea surface slopes reveal gravity anomalies because the primary component of sea surface height is geoid height, the elevation of an equipotential of the gravity anomaly field. The geoid height at a point is the integral of anomalies over the entire earth [3], and one must differentiate the geoid to reveal local anomalies. The horizontal derivatives (slopes) of the geoid indicate anomalies in the direction of gravity called deflections of the vertical.

MISSION REQUIREMENTS

Path Delay

The slope signals required to estimate bathymetry are band-limited (12 km to 400 km full-wavelength). Hence, the height measurements of an altimeter such as Abyss-Lite need to maintain relative accuracy only over this relatively narrow band. It has been shown that the Abyss-Lite signal – geoid slope – is best obtained by taking the along-track derivative of the instantaneous sea surface height signal. In most situations this is within 1 μrad of the geoid slope [4, 5]. Indeed, efforts to correct for path delays usually add noise to slope estimates [5]. Almost all sources of path delay error known to oceanographic radar altimetry have negligible impact to Abyss-Lite science. Hence, Abyss-Lite does not require two frequencies, nor a water vapor radiometer.

Precision

Within the desired spatial passband, excellent precision is required (1μrad) of slope, or only 6 mm height change per 6 km along-track). North and east slope components can be combined to recover gravity anomalies because the three components of the gravity vector are coupled through Laplace's equation [6-8]. A gravity anomaly of 1 mGal (10^{-5} m s^{-2}, or about 10^{-6} of total gravity) corresponds to a geoid slope of 1 μrad (10^{-6} radian, or 1 mm height change per km) [9, 10]. Abyss-Lite should measure slopes to the order of 1 μrad, down to a full-wavelength scale of 12 km.

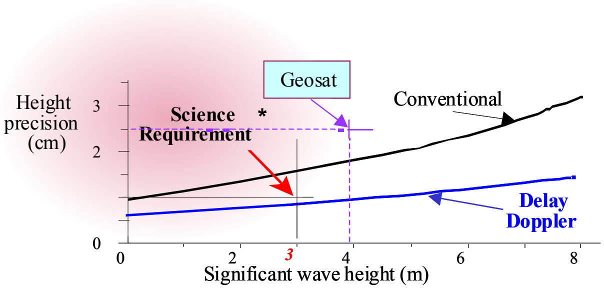
Orbit

The orbit should not repeat for ~1.2 years, to yield an average ground track spacing of 6 km, and should have an inclination near 50°-63° (or 113°-120° retrograde) to resolve north and east slopes nearly equally, and to cover the lower latitudes where existing data are inadequate. Note that oceanographic radar altimeter missions (TOPEX/Poseidon, Jason-1, ERS1/2, Envisat, and Geosat ERM/GFO) normally are placed into exact-repeat orbits (10 to 35 days), and as a consequence have widely spaced (80 km to 315 km) ground tracks. Such orbits cannot resolve the short-wavelength two-dimensional surface slopes required for useful bathymetry.

ALTIMETER PRECISION

Sea surface slope measurements are derivatives of the altimeter's natural measurements, height. Taking derivatives eliminates constant and long-wave height errors, but it amplifies noise at short wavelengths, making Abyss-Lite resolution limits very sensitive to altimeter precision. Using a simple model in which height errors are assumed to be a Gaussian white noise process over the Abyss-Lite band, we find that the one-sigma slope error will be 1.8 μrad if the altimeter's one-sigma height precision is about 1 cm for a one-second averaged height value. Height precision degrades with increasing significant wave height (SWH). As a single point constraint sufficient for altimeter selection, we have stipulated the height precision to be 1 cm at 3 m SWH.

Fig. 1 shows a plot of height precision *versus* SWH for a delay-Doppler altimeter [11] (DDA) and a conventional radar altimeter (RA). The plot shows that the DDA meets the height precision requirement. This result is consistent with previous analyses that show the DDA to be significantly better in precision than an RA [12]. The figure also shows that the DDA is about half as sensitive as an RA to increasing SWH. This is important for Abyss-Lite, as measurement precision degraded by larger significant wave heights proved to be a major source of noise in Geosat surface slope estimates [5, 9].



* Derived from white Gaussian noise process over the 10 km – 200 km band
=> one - sigma 1 microradian slope error ~1- cm height precision

Figure 1. Altimeter precision and significant wave height

ABYSS-LITE DESIGN

The design study worked within the payload constraints of the Pegasus launch vehicle. Table 1 shows the cost and mass to orbit for the Pegasus and Taurus for the baseline prograde orbit, and a typical retrograde orbit. Given the $16 million dollar price difference between Pegasus and Taurus, the motivation to fly on a Pegasus is strong. The baseline mass of the delay-Doppler altimeter is 27.5 kg, representing a modest mass fraction of eleven percent.

We looked at two implementation strategies. The first approach was to have the NASA Rapid Spacecraft Development Office respond to the Abyss-Lite requirements with a set of candidate buses (along with their ROM cost). The second approach was to develop an original spacecraft design within the Johns Hopkins University Applied Physics Laboratory, based on previous missions such as CONTOUR. The in-house approach turned out to be more suitable, and less costly.

Table 1. Launch service characteristics

	60^o	Cost	120^o	Cost
Pegasus	250kg	$31M	160kg	$29M
Taurus	>600kg	$47M	592kg	$46M

The radar peak transmitted power is 2.5W, in contrast to the 20W of Geosat. The Abyss-Lite lower power is due in part to its higher gain antenna, and also due to the delay-Doppler paradigm. Although Doppler ambiguities will be generated by the 1-m antenna combined with the ~13 KHz effective pulse repetition frequency, these can be suppressed through frequency- and range gate selection. In flight configuration, the antenna phase center should be directly below the center-of-gravity, which will minimize height errors that could be induced by spacecraft pitch or roll motions. The antenna just fits within the 1.1-meter dynamic envelope of the Pegasus fairing.

The Abyss-Lite altimeter would be assembled on a dedicated honeycomb deck independent of spacecraft development. Once integrated, the instrument would be environmentally tested, and calibrated before delivery to the spacecraft's integration facility. The integration with the spacecraft would occur toward the end of spacecraft assembly.

The nominal circular 800-kilometer orbit was analyzed for eclipse time characteristics, and for ground contact frequency and duration. The maximum duration for operating completely from battery power is about 35 minutes. The maximum delay between overpasses of a ground station with visibility down to six degrees is 13 hours. On-board data storage was sized to accommodate up to 48 hours of data. These results are independent of orbit inclination.

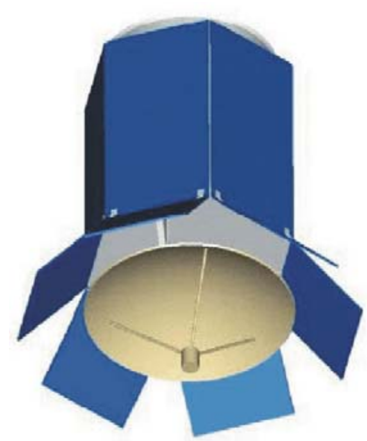


Figure 2. Abyss-Lite on-orbit configuration

CONCLUSIONS

We have completed a conceptual design of a single-frequency high-precision radar altimeter hosted on a small dedicated spacecraft whose mission would be to determine near-global bathymetry to a resolution of 6 km (half-wavelength) by measuring sea-surface slopes. The state-of-the-art in such measurements is no better than about 25 km, based primarily on data from the Geosat altimeter. Improved bathymetry and oceanic gravimetry is needed for better information on deep ocean circulation and mixing, as required for climate models for example, for navigation systems (subsurface, as well as surface and airspace), for primary petroleum exploration, and for trans-oceanic cable routing, among other applications. There are two mission parameters that dominate performance and that are required to achieve the desired resolution. They are (1) instrument precision (which implies a delay-Doppler radar altimeter), and (2) the orbit, which must be non-repeating or a repeat period longer than one year), and at a moderate inclination (either prograde or retrograde). Spacecraft and instrument cost are not sensitive to these parameters, although the launch cost and launch vehicle mass limits do depend on inclination.

The cost of the altimeter and spacecraft would be less than $60M. Added capabilities (such as propulsion for orbit maneuvers, gyro stabilization, or instrument redundancy) would increase that cost.

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ABYSS-LITE SPACECRAFT BUS COSTING STUDY

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7-374

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1.1 Study Objectives

The Abyss radar altimeter concept is being considered for future APL missions as a free flying small spacecraft bus – referred to here as "Abyss-Lite." The type of science and the sponsor constraints dictate that such a concept will compete for selection within a cost constrained/cost capped environment. The defined task was to come up with a basic spacecraft design including a ROM cost.

The payload for this spacecraft is a radar altimeter whose most noticeable feature is a nadir pointing parabolic dish antenna. It was desirable for this antenna to be placed below the spacecraft's on-orbit center-of-gravity. The cost estimate for the altimeter development and delivery have been estimated separately by Paul Marth. The top-level requirements for the spacecraft bus are given in Table 1.

Table 1. Abyss-Lite Top Level Requirements

Parameter	Value
Mass:	27.5 kg
Power:	50 watts
Altitude:	800 km
Science Data Rate:	25 kbps
Antenna Diameter:	1 meter
Inclination:	50 to 63 degrees
Pitch Control:	+/- 0.5 degrees
Roll Control:	+/- 0.5 degrees
Yaw Control:	no requirement
Navigation:	GPS
Mission Duration:	5 years (@ 0.85)

1.2 General Approach

The study started with the Pegasus launch vehicle. The Pegasus is available as a launch service within NASA's small expendable launch vehicle (SELVS) contract along with the Taurus and the Delta II. Table 2 shows the cost and mass to orbit for the Pegasus and Taurus for the baseline 60-degree inclination along with several retrograde orbits; the Delta II is more capable than the Pegasus and Taurus but its costs are substantially higher. Given the $16 million dollar price difference between Pegasus and Taurus, the motivation to fly on a Pegasus is strong. For the desired orbit of 800 kilometers, 60-degree inclination, the Pegasus will deliver 250 kilograms to orbit. The baseline mass of the altimeter is 27.5 kilograms representing a mass fraction of eleven percent – a modest number for a LEO orbiting spacecraft.

Appendix A documents the launch service costs for the Pegasus and Taurus vehicles. Appendix B shows the mass to orbit performance of these vehicles for different orbit inclinations.

There were two approaches in meeting the objectives. The first approach was to have the NASA Rapid Spacecraft Development Office look at the "Abyss-Lite" requirements and respond with a set of candidate buses (along with a ROM cost). The second approach was to develop a spacecraft design – after surveying APL capability and several accepted small satellite approaches. With the design in hand, a development timeline and a cost estimate were formulated using existing spacecraft development data (mostly CONTOUR) that was properly adjusted.

1.3 The Altimeter Instrument

The natural arrangement for this payload design is shown in Figure 1. The altimeter would occupy the nose of the launch vehicle's fairing. The altimeter components are shown in blue. Altimeter components, including the antenna, mount to the top and bottom sides of the instrument's primary structure, an aluminum honeycomb deck. This deck interfaces with the spacecraft's primary structure. The size of the altimeter antenna is constrained by the maximum diameter of the Pegasus dynamic envelope. At one-meter, the antenna just fits within the 1.1-meter dynamic envelope of the Pegasus fairing.

Regardless of approach, the Abyss-Lite instrument would be assembled on its honeycomb deck independent of spacecraft development. Once integrated, the instrument would be environmentally tested, and calibrated before delivery to the spacecraft's integration facility. The integration with the spacecraft would occur toward the end of spacecraft assembly – shortly before spacecraft level environmental testing.

1.4 Abyss-Lite Orbit Characteristics

The nominal circular 800-kilometer orbit was analyzed for eclipse time characteristics and for ground contact frequency and duration. Figure 2 shows the eclipse time durations for the orbit over a two-year period. The graphic identifies the maximum duration for operating completely from battery power is 2,111 seconds (35.1 minutes). This result is the same regardless of inclination. Given the five year lifetime desire, the spacecraft's battery should be sized for depths-of-discharge no greater than 30 percent, regardless of battery technology.

Table 2. Launch Service Cost and Mass to Orbit for Different Inclinations

	Orbit: 800 km circular					
Vehicles	60 deg (WAPS)	Cost	113 deg (VAFB)	Cost	120 deg (VAFB)	Cost
Pegasus	250 kg	31 M$	170 kg	29 M$	160 kg	29 M$
Taurus 2210		47 M$	609 kg	46 M$	592 kg	46 M$

For ground contact frequency and duration, the orbit was studied through a one-week period of time. A ground antenna elevation of six degrees was used as the cut-off point for a viable pass. For the 60-degree inclination, passes occur in one large cluster of up to seven per day (some are not usable) followed by about a 13-hour wait.

For a 120-degree inclination orbit, there are fewer total passes but the overall quality is about the same. Also the clustering is different; there are two clusters of three within a 12-hr period, with the two clusters separated by three orbits (about 5 hours). Then about a 13 hr wait.

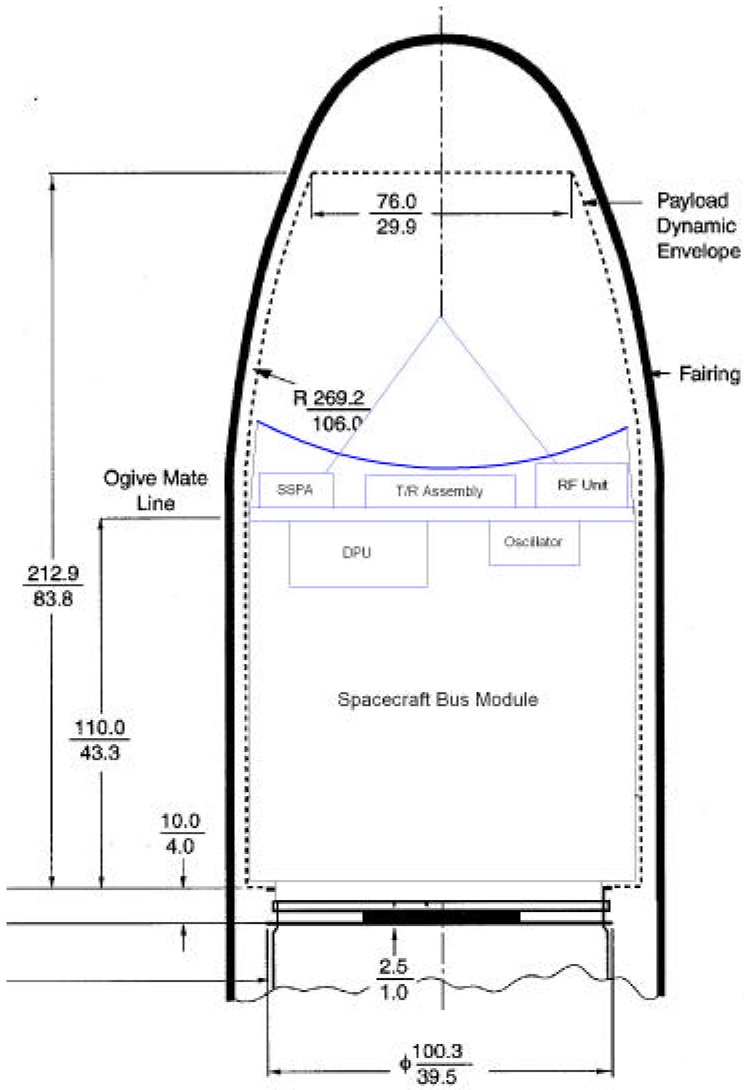


Figure 1 – Pegasus Shroud Showing General “Abyss-Lite” Satellite Concept

Regardless of which orbit is used, there are at least three passes per day that are 10 minutes in duration or longer. The spacecraft’s four megabit per second downlink rate will completely empty the data recorder in less than 10 minutes (1 pass).

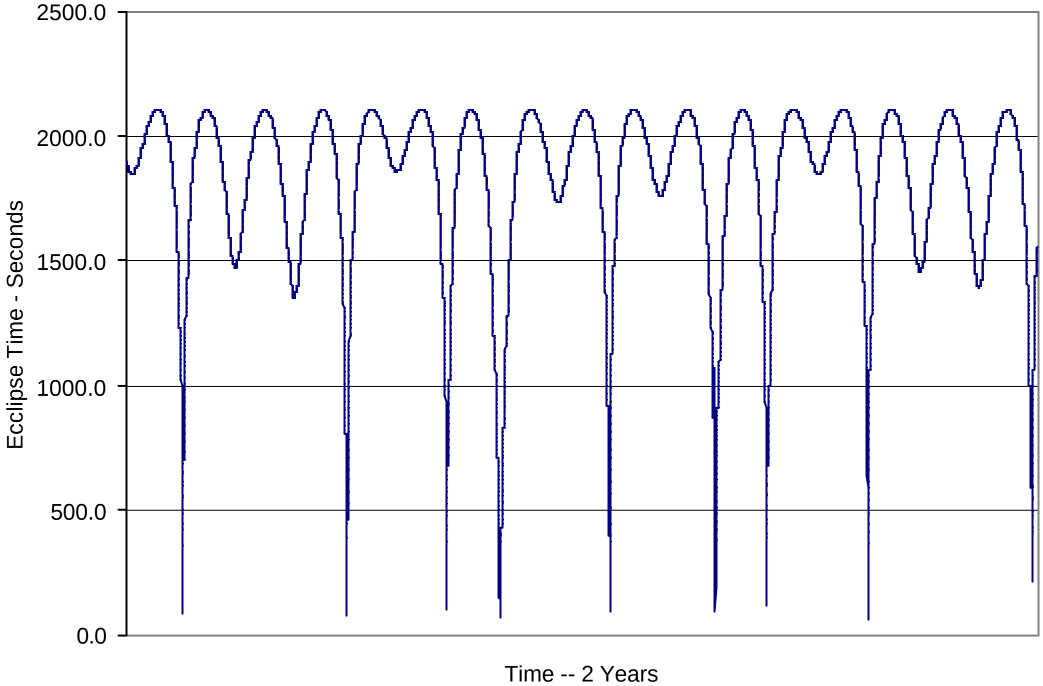


Figure 2 – Orbit Eclipse Time Over a Two-Year Period.

1.5 GSFC's Rapid Spacecraft Development Approach (RSDO)

The NASA Goddard Rapid Spacecraft Development Office (RSDO) was setup in response to complaints that it could take over a year to award a competitive contract for a spacecraft bus. The challenge was to reduce that time down to 30 days. Presently, 90 days is the cycle time from initiation of a competitive procurement to a signed contract. The RSDO business model consists of NASA pre-approved vendors that offer existing, heritage-based designs in a streamlined competition. To be a qualified, the vendor must: be ISO9000 compliant, have taken the bus design through launch processing, and have adequate facilities to perform the work. There are currently 8 vendors offering a total of 21 bus designs. The bus designs are modifiable to meet the unique needs of the customer's payload. Payload mass can range from 1 kg to 800 kg and payload power requirements can range from 1 watt of power to 800 watts. The RSDO vendors operate within a well defined statement-of-work that enforces many of NASA's current practices. Of the 21 bus designs, there appear to be five that would be compatible with the Abyss-Lite payload requirements and the Pegasus launch vehicle. One positive aspect of the RSDO contract is the clause for bus replacement in the event of an on-orbit bus failure. This is usually accommodated by the vendor through an insurance policy whose cost is applied within the cost of the contract's price.

Contracts are fixed price and are structured so that payments are based on real task milestones. A bus contract includes integration of the payload, environmental testing of the system, launch processing and on-orbit commissioning. If desired, the contracted vendor can operate the spacecraft during its lifetime and essentially provide a 'data spigot' to the customer.

1.6 Abyss-Lite and RSDO

As part of this study, contact was made with NASA's RSDO office for a ROM price quote. After a review of the RSDO approach and the standardized RSDO statement-of-work, a completed RSDO template was forwarded to RSDO. Appendix C contains the Abyss-Lite template. The RSDO worked within their office (not going out to the vendors) using cost data provided by the vendors to come up with a list of suitable buses and a ROM estimate for a simple bus procurement – including some required modifications and enhancements. Three buses (TRW T100, TRW T200A, and Ball BCP 600) were presented and their range of estimated cost varied between 38 to 43 million dollars. RSDO also identified the Spectrum Astro SA-200B as a viable bus candidate – if enhanced with more redundancy to improve its lifetime. The response letter and bus description matrix are included in Appendix D.

I also looked at the publicly available information regarding the RSDO bus catalog. One bus that looked compatible with Abyss-Lite was the OSC LeoStar2 bus. RSDO confirmed that the bus was compatible but was 'overkill' for the requirements and this was reflected in the price tag of 50 to 55 million dollars. My own survey of the catalog yielded similar results which are shown in Table 3.

Of the three buses presented by RSDO, two were TRW designs. One has flown on a Pegasus (TOMS-EP) but that was over seven years ago. This is a concern from the point of view of parts availability and obsolescence. The other TRW design is meant for an Athena launch vehicle –

which is not available under NASA's SELVS contract. The Ball design (BCP-600) does claim to be compatible with the Pegasus – but to date that bus has only flown on larger launch vehicles and the buses themselves could not have fit on a Pegasus without major modifications.

Ball Aerospace does have a design that is currently under development (funded by in-house IRAD) referred to as the RS300. Because it is not fully developed, it is currently not in RSDO's catalog. The RS300 is applying recent Ball miniaturization advances – particularly from its NASA Discovery Deep Impact mission – and is targeting to put high capability into a Pegasus sized bus. In a conversation with Ball representatives, it is unclear whether the miniaturization would lead to a fully redundant bus for the Pegasus. Once developed, expect to see the RS300 on-ramped into the RSDO catalog.

The figures below [3 – 8] show different Pegasus designs that could be made suitable for an Abyss-Lite payload. A common theme that runs through the different designs is the modularization of the spacecraft bus and a clean interface with the scientific payload. This works well for the Abyss-Lite concept.

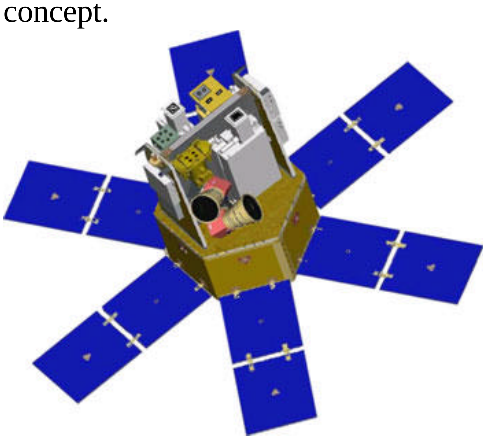


Figure 3. OSC LEOSTAR-2 (SORCE)

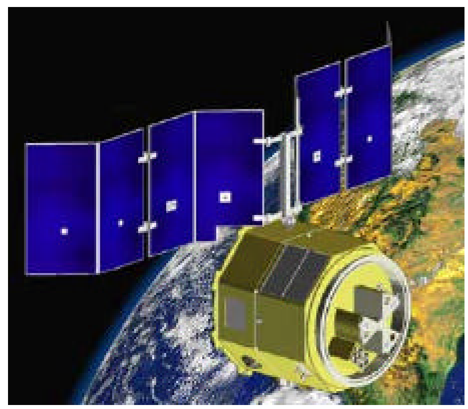


Figure 4. BALL RS-300 Bus

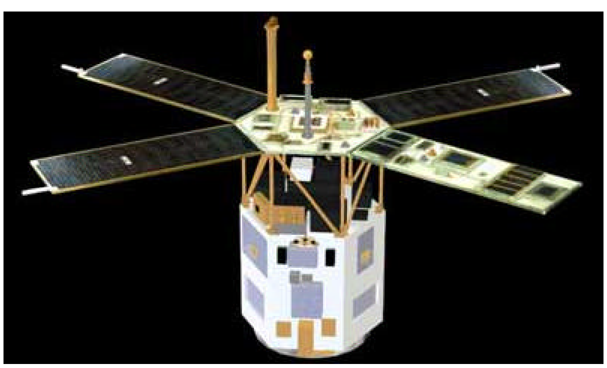


Figure 5. OSC's PegaSTAR

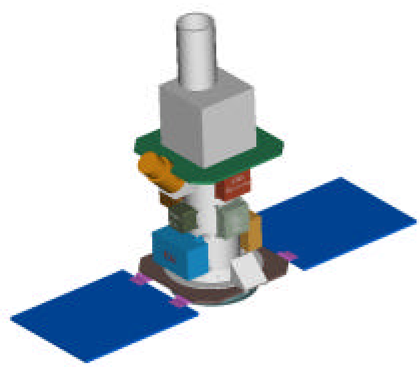


Figure 6. APL SBSS Concept Spacecraft

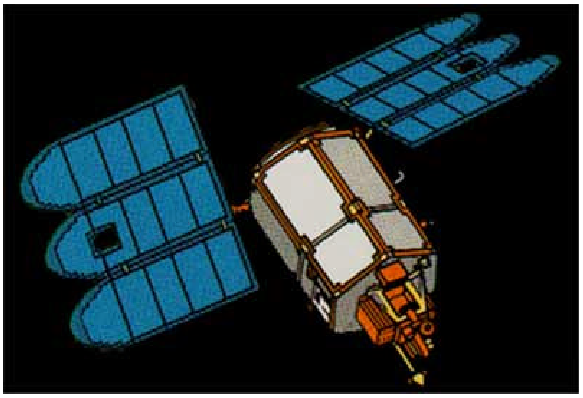


Figure 7. TRW T100 Bus (TOMS-EP)

Figure 8. Ball BCP-600

Table 3 – Ad hoc survey of the RSDO Catalog APL Approach

Candidate Buses	OSC LeoStar-2	TRW T100	Ball BCP600	SA-200S
Cost Scale	50-55 M$	38-43 M$		
Construction	Hexagonal module, Honeycomb	Hexagonal, Aluminum, Multideck	Graphite-epoxy Octagonal Unibody	Irregular, Aluminum half frames, honeycomb
Bus Dry Mass (w/o payload)	178 kg (w/o redundancy)	185 kg	170 kg	Unknown
Propulsion	Hydrazine, blowdown up to 43.7 kg of prop.*	Hydrazine, up to 73 kg.	Hydrazine, Up to 28 kg*	Hydrazine, up to 21.3 kg
Battery Size	16 Amp-Hr NiH	9 Amp-Hr	20 Amp-Hr, NiH2	15 Amp-Hr
Array Size	Fixed, 3 m2	Fixed, 3.75m2	Fixed, 2.4 m2	Fixed, 1.62 m2
Array Technology	Triple Junction	Silicon	Dual Junction	Dual Junction
Payload Power Acc	118 Watts at 29 deg inc.	25 Watts OAP	Up to 125 watts OAP.	66 W
Attitude Control	3-Axis ZM	3-Axis, PM bias	3-Axis, ZM	3-Axis ZM
Knowledge	0.03 deg	.25 deg	0.0034 deg	778 udeg.
Control	0.042 deg	.3 deg P/R, 1 deg Y	0.004 deg	0.01 deg
GPS	Yes	No	No	Yes
Redundancy	Yes - 6 years*	Yes, Fully	Yes, 6 years	Requires enhancement*
Delivery Time ARO	30 months	18 months	32 months ARO	36 months
Data Storage	Scalable to 32 Gb*	16.7 Mbytes	*2 Gbits*	*2 Gbit*
Downlink rate	>4 Mbps*	200 kbps	*1 to 2 Mbps*	*2.8 Mbps carrier with 128 kbps sc.*
* option required to meet requirement.				
Red means does not meet requirement				
Italic means: does not meet requirement but can be made to work by operational change.				

Much of the work in developing an APL Abyss-Lite system is based on Mark Perry's work in defining a small bus for the military Space-Based Space Surveillance (SBSS) concept. Additional insight from other current design practices have also been applied to the design. At the heart of an APL development approach is the necessity to use a focused 'project team' similar to those used for Scout sized projects like PolarBEAR and HILAT.

The spacecraft bus would most likely be a modular structure with six or eight sides. This report assumes that the module is a regular hexagon that is one meter tall. Most of the bus components would fit within the structure. The star trackers would stare out through holes in the sidewalls. The outside of the structure would be populated with solar cells and four sun sensors. Historically, mass estimates of structure have been 15 percent of dry mass. This assumes construction using basic aluminum and honeycomb aluminum materials. For this report, 16 percent of the rolled up mass has been allocated for spacecraft bus primary structure – including fasteners.

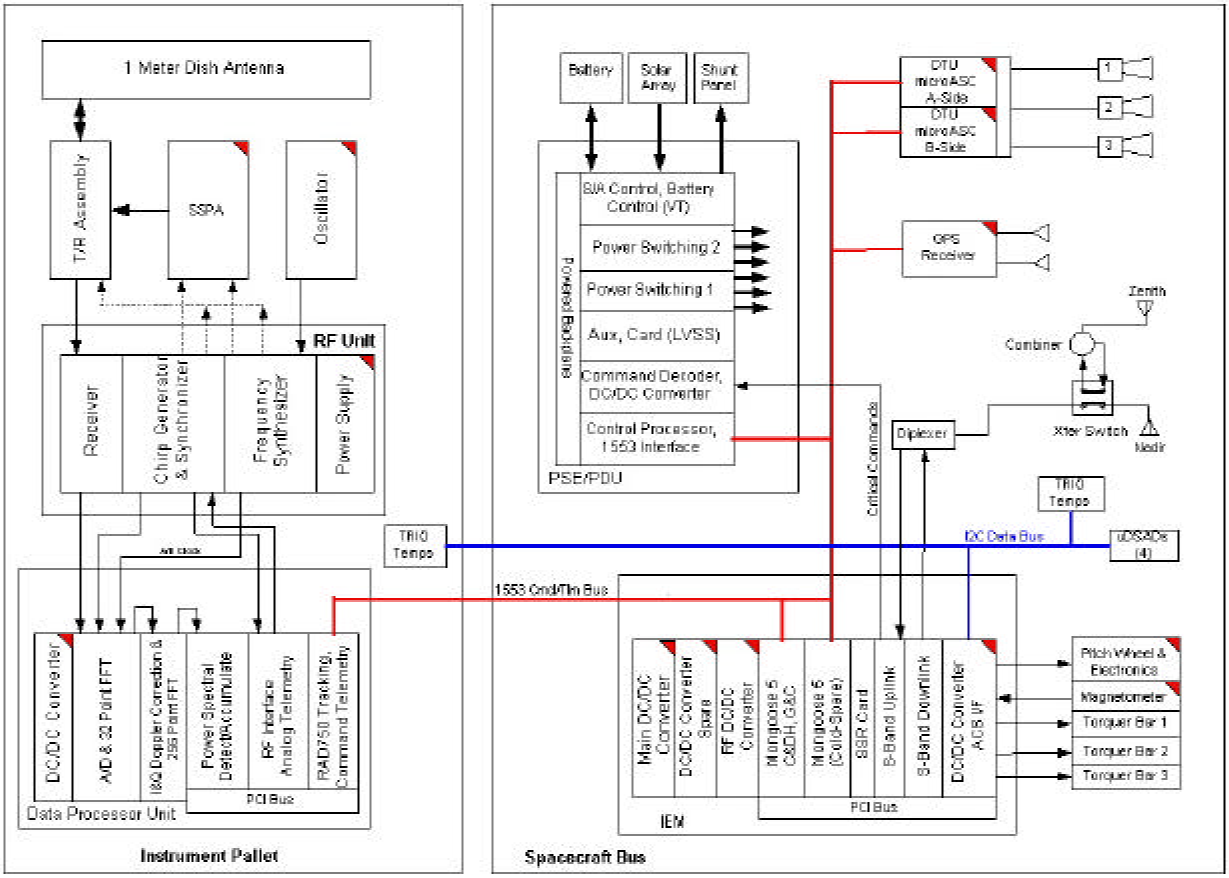
1.7 Baseline Architecture

A baseline architecture was developed that incorporates existing APL practices and techniques. The approach in this study is to develop an absolute minimum configuration necessary to meet (or try to meet) Abyss-Lite performance requirements and then to augment the configuration with more capability and robustness. In each of the configurations studied, the components are given mass growth allowances that are based on technical maturity or the certainty that the sizing of the component to the Abyss-Lite requirement is correct. The masses (including the growth allowances) are tallied and the overall number is compared with launch vehicle capability – which is reported as margin. For configurations in which the margin is greater than 15 percent, then the development risk is considered low and no additional money reserves are deemed necessary to counter mass risk issues. For configurations when the margin is less than 15 percent, then the development risk is considered high and funding reserves, and mitigation plans, really do need to be considered.

Figure 9 shows the Abyss-Lite (minimum configuration) system block diagram. This is a single string design – and would be greatly challenged to meet the five-year lifetime requirement. On the left of the diagram is the altimeter instrument and on the right is the spacecraft bus. With the exception of switched power (designated by red triangle in the block corners), a 1553 bus interface, and an I2C bus interface, the two systems are well isolated from each other.

The power system is a direct energy transfer system that packages the electronics responsible for power generation, regulation, storage, and distribution into a single box. The box is constructed using packaging techniques developed for the APL MESSENGER power system.

The solar array is assumed to be fixed panels populated with triple junction GaAs cells of 26.5% efficiency. The spacecraft's power requirements exceed the power generation capability of a body mounted solar array requiring additional panels to deploy from each sidewall as shown in Figure 10.



Abyss Lite Spacecraft Concept Block Diagram

Figure 9. Abyss-Lite Minimum Configuration System Block Diagram

The battery is a nickel-hydrogen common pressure vessel sized for a maximum on-orbit depth-of-discharge of 30 percent.

The IEM uses the same card architecture and card size as the IEM design used on TIMED and CONTOUR. For this study, it is assumed that the overall box mass can be reduced. On TIMED and CONTOUR, the per card mass of the IEM was 1.2 kilograms – this includes cards, motherboard, housing, and connectors. The five card MESSENGER IEM has a per card mass of 1.1 kilograms. For 'Abyss-Lite', a goal of 1 kilogram per card is assumed feasible. Two additional cards have been added to the IEM and designated as 'cold spares' to either improve redundancy or as development reserve.

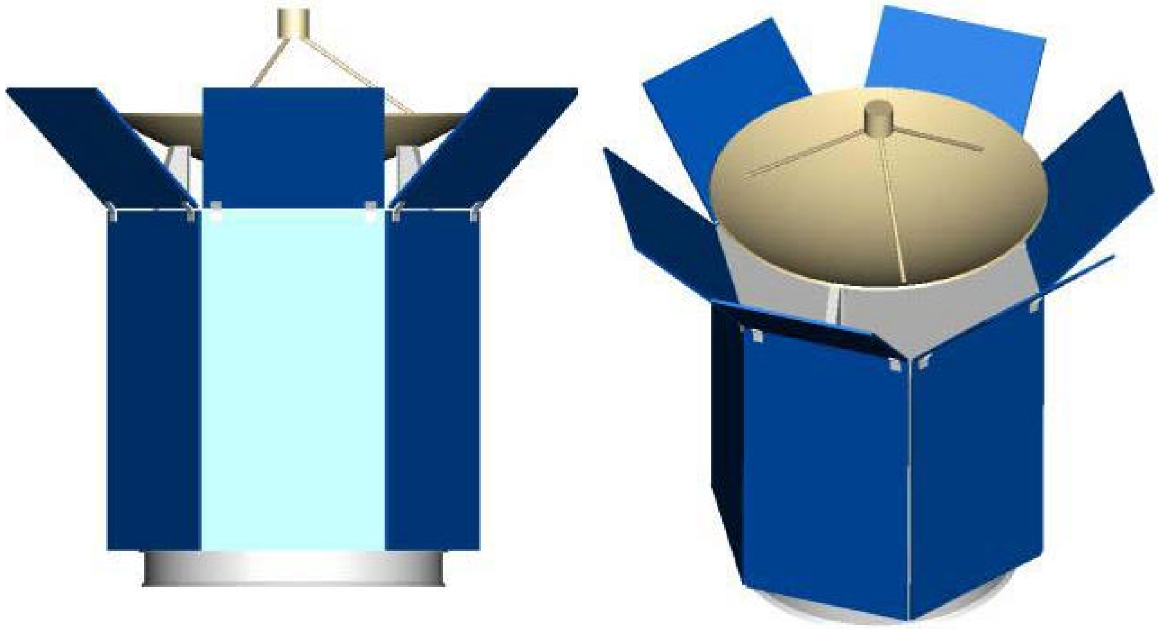


Figure 10. APL Abyss-Lite Spacecraft Concept

Within the IEM, the data cards all communicate with each other over the PCI backplane. A single Synova Mongoose V processor board is used for both Command & Data Handling and for Attitude Determination & Control. Also within the IEM is a 2.5 Gbit solid-state recorder card representing 27.7 hours of science data storage.

The telecommunications system uses S-band cards that reside in the IEM. These cards represent the next generation S-band design that was flown on TIMED. More of the cards functionality is implemented in digital circuitry and hence operate at a lower power. The transmitter card outputs 3 watts of RF energy providing a 4 Mbps downlink to the ground – assuming TIMED geometries and ground station. When not transmitting, the downlink card is in a low power standby mode. At 4 Mbps, a single 10-minute ground pass can downlink 26.6 hours of science data. The orbit used provides several of the passes every day. The spacecraft has a nadir and zenith antenna arrangement similar to TIMED. The use of a transfer switch and combiner allow the antenna selection to change from nadir only (for operational use) to both nadir and zenith simultaneous use for near omni-directional coverage (for spacecraft commissioning and emergency situations).

The primary attitude determination sensor is a multi-head star tracker developed by Danish Technical University. It is the advanced stellar compass (ASC) architecture that has flown on PROBA, OERSTED, and CONTOUR. It has been repackaged into the 'microASC'. The new package is smaller, less power, and more reliable than the earlier design. It is fully redundant. There are three optical sensing heads to the star tracker; each is independently oriented to insure that even if two are blinded by the Sun and Moon, the third head is looking toward space. The microASC provides 21 quaternion attitude solutions per second, enabling the system to operate in a 'gyroless' mode while still meeting attitude requirements. Gyroless mode systems have been flown before, generally as a result of gyro's failing in orbit. With this tracker, and its fast update rate, it is feasible to think in gyroless terms during the conceptual development of the project.

As backup attitude sensors, there are four coarse (~3 degree) Sun sensors and a three-axis magnetometer.

Attitude control is through a pitch bias system. The single wheel provides gyroscopic stiffness to the system and controls agility about the pitch axis. The other axes are controlled via magnetic torquers. Given the inclination of 60 degrees, it is uncertain whether there is enough torque authority of the magnetic torquers to guarantee control within the requirements. Such an analysis is beyond the scope of this study. The interfaces to the wheel, magnetometer, and the magnetic torquers are via an attitude interface card that resides within the IEM and communicates with the spacecraft processor over the PCI backplane.

The mass and power rollup of this single string design fit well within the Pegasus' capability. The drawback to the configuration is lifetime since there is almost no redundancy and performance capability given attitude control uses torquer bars. Because the system relies on torquer bar for control, it has no agility to move the spacecraft in short timescales. To counter this, the solar array configuration needs to provide an almost 'omni-directional' power generation capability to handle emergency situations and anomalous attitudes.

1.8 Enhancing the Architecture

A more capable design is shown in Figure 11. See Appendix F for the Master Equipment List for this approach. It adds a gyro and two more wheels into the system. The addition of the gyro removes any performance uncertainty of operating in a gyroless environment and the three wheels provide for a complete, single string, complement of zero momentum, three axis attitude control capability that is not dependant on magnetic field strength.

The gyro does not come for free; its cost is nearly five hundred thousand dollars and its power consumption is twenty watts. Later, as the system gets developed further, the gyro will be deleted from the configuration in favor of a 'gyroless' control system. Because the wheels were added for primary attitude control, the torquer bar functionality was reduced from 'attitude control' to dumping momentum from the spinning reaction wheels. The result is that smaller torquer bars can be used.

These augmentations work to improve the capability of the system but still do not address lifetime reliability. Ironically, they work to reduce reliability.

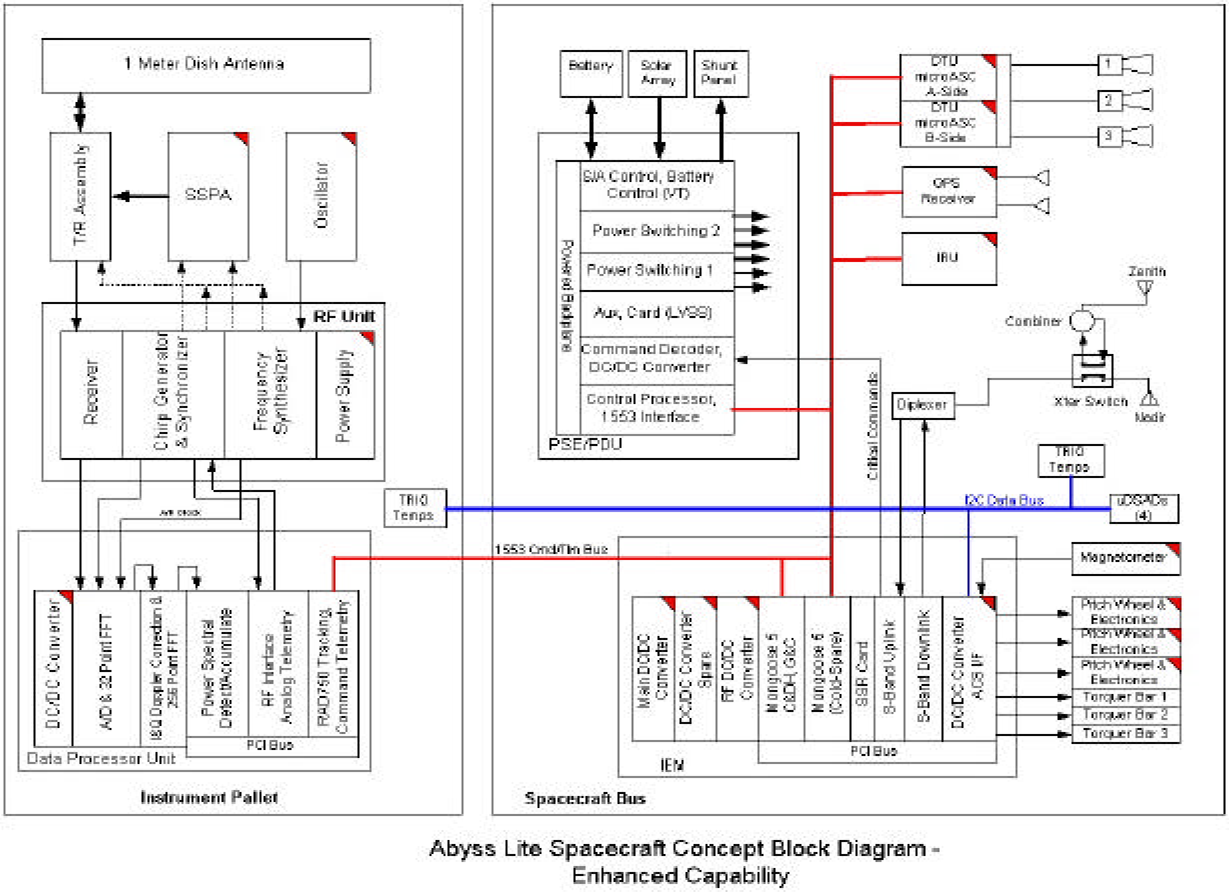


Figure 11. Abyss-Lite Enhanced Capability Single Sting Block Diagram

Two more incremental variations of the APL spacecraft were developed for evaluation. These included an onboard propulsion system and a final version that included the propulsion system and more redundancy. In developing the mass spreadsheets for these systems, it became clear, that while the architecture was compliant to accept additional components, the mass margin was being reduced to the point that substantial development risk would begin to be a factor. Table 4 shows a summary of the mass, mass margin, and power.

Table 4. Summary of Mass, Mass Margin, and Power

	Minimum Configuration	Added Gyro, two more wheels	Added propulsion system	Added another gyro, GPS, wheel, IEM
Rolled Up Mass	178	197	233	262
Mass Margin %	29%	21%	7%	-5%
Rolled Up Power	160	192	212	244

1.9 Developing a Final Configuration

Table 4 shows that it is not possible to have an agile, capable spacecraft that meets performance, has propulsion, and arguably meets the five year lifetime requirement. A change in architecture or a descoping of requirements is needed.

The biggest change to the architecture is the adoption of a 'gyroless' attitude system. Because it is a departure from traditional techniques, it should be modeled and analyzed early in the projects development. If successfully implemented, then the trade space is between lifetime or propulsion.

To meet the project's five year lifetime requirement, there are two approaches: (1) a comprehensive parts screening, testing, and analysis project – similar to the STEREO effort or (2) a more relaxed, traditional parts screening, testing, and analysis approach that incorporates brute force component redundancy. The approach taken here is to employ component redundancy and to have onboard 'cold' spares. Table 5 and Figure 12 summarize this approach. Appendix I contains the master equipment list, inc luding a mass and power rollup for this final version. Note that the basic rolled up mass is 176 kilograms. It is the application of the component growth allowances that take the mass up to 218 kilograms. That leaves 13% margin between the 213 kilograms and the launch mass. A prudent approach would be to refine the design early in the project to lower the uncertainty in the growth allowance and if the outcome is positive, then to insert a modest propulsion system. If the full growth allowance appears to be required, then the propulsion system would be descoped.

Table 5. Recommended Architecture Mass, Mass Margin, and Power Summary.

Rolled Up Mass	176 kg
Mass+Growth Allowance	218 kg
Mass Margin %	13%
Rolled Up Power	203 watts

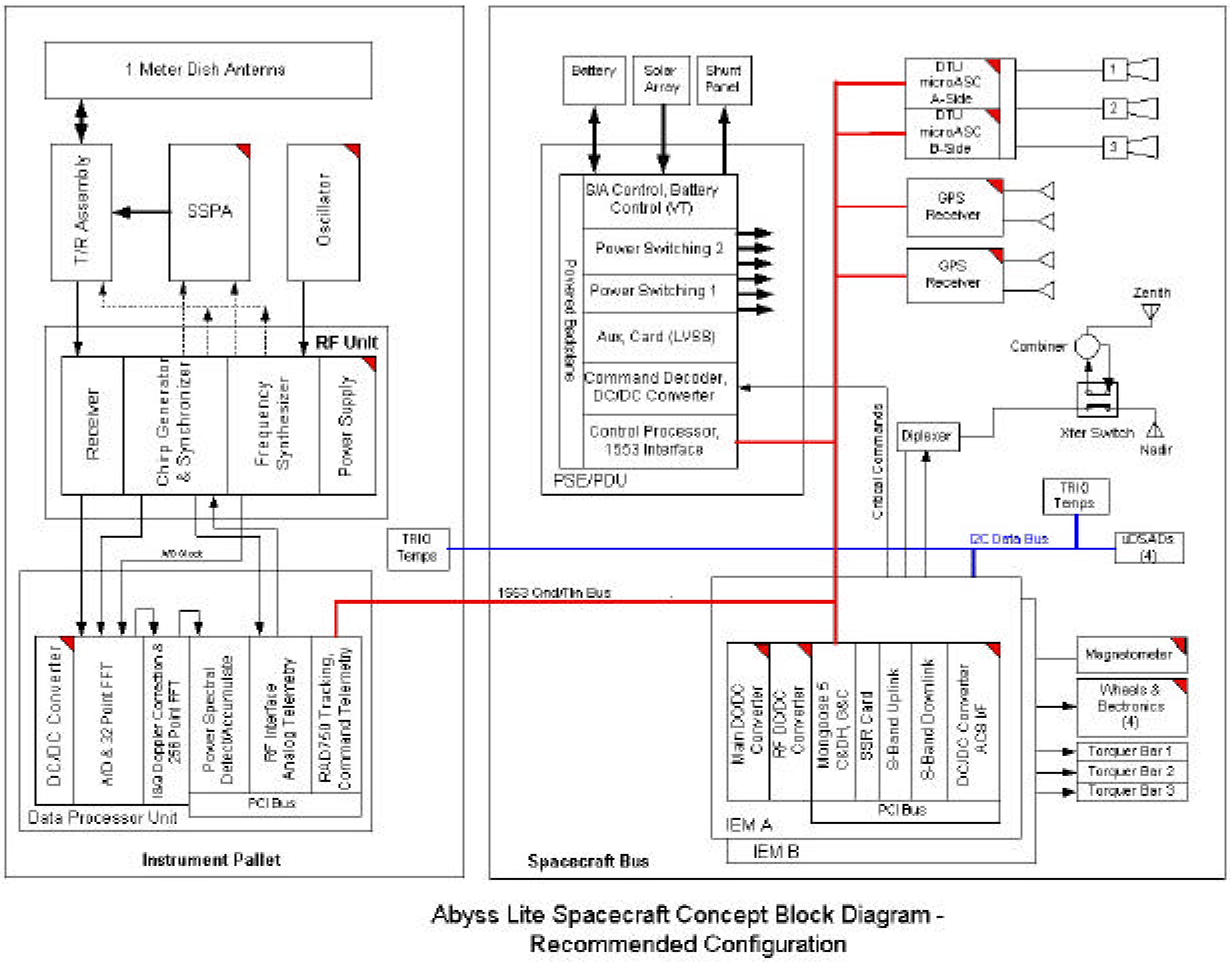


Figure 12. – A Final Recommended Configuration

1.10 Costing the APL Approach

A cost model was developed for Abyss-Lite using CONTOUR "actuals" and appropriately scaling the data within a work-breakdown-structure. This effort was performed without soliciting the groups for new inputs. The following WBS was used in the model is shown in Table 6. Given the scope of work envisioned and a familiarity with the CONTOUR mission, weighting factors were developed and applied to the staffing costs to generate the Abyss-Lite staffing estimates. For Phase A/B, the weight factors were all unity value (1.0). For Phase C/D up through integration, the weighting factors are shown in red in column two of Table 6. The weighting factors used during the I&T phase are shown in blue. The weighting factors reflect the rationale stated in Table 7. In reviewing the comparison chart it is clear that Abyss-Lite is a smaller spacecraft, with less instruments, and less required capability. In some cases, it did not make sense to fractionate below what was felt to be a critical level of staffing. The total staff months for these WBS elements are identified in the third column of Table 6.

A three-year development cycle was selected – from project initiation up through launch plus thirty days. Three years is within the historical range of past APL spacecraft development projects. NEAR developed within 27 months and had greater capability than Abyss-Lite.

Table 6. Abyss-Lite WBS for an APL Built Spacecraft.

WBS Element	Weight Factor	Staff Months	WBS Task Description
Project Management/PA (1**)	0.90 0.90	245.6	Includes: Project Manager, Administrative Assistant, Resource Manager, Mission System Engineer, Verification Engineer, Mission Manager, Product Assurance Engineer, Autonomy Engineer, and G&C lead engineer.
Flight Software Dev & Testing (58*)	0.60 0.50	178.4	Includes: Lead Software Engineer, G&DH Software, G&C Software (non-automated code), Boot Software, and Software Testing.
MOC Software (760)	0.40 0.40	63.3	Includes: Ground Software and its testing.
Mech, Structural , Thermal, Prop. (510-530)	0.80 1.00	192.6	Includes: Mechanical Engineer, Mechical Technician, Thermal Engineer, Thermal technician, Structural analyst, Shop labor in support of the primary and secondary structures, shop labor in support of the fabrication of harness, rf, and thermal mockups.
G&C Procurement Oversite (535)	1.00 1.00	25.4	Includes: writing performance specs and RFPs, reviewing proposals, and technical contract management.
IEM (all cards & TRIO) (54*+522)	0.80 0.70	241.9	Includes: Development, fabrication, testing and delivery of all the cards in the IEM including RF cards, motherboard, chassis. Includes integration and testing of the assembled IEM. Also includes fabrication and testing of the flight TRIOs.
Power Subsystem (55*)	0.60 0.70	152.9	Includes: the development, fabrication, testing and delivery of the power electronics box and its GSE. The writing performance specs and RFPs, reviewing proposals for the battery and the solar array and their technical contract management. The development of a solar simulator and battery GSE.
RF Subsystem (w/o cards) (56*)	0.40 0.40	59.3	Includes the antenna development. The RF switch plate assembly. RF GSE (uplink rack and downlink rack). Any any subsystem testing that precedes delivery to spacecraft integration.
Harness Design (570)	0.80 0.70	28.4	Includes: the development of harness drawings and wire-run lists prior to harness fabrication.
Parts Engineering & SC QA (59*)	0.60 0.60	115.3	Includes: parts procurement labor, parts engineering labor, parts testing labor – also includes quality assurance engineering labor.
System GSE (71*)	0.60 0.60	53.1	Includes: the development of the MOC and miniMOC hardware. The development of the command and telemetry databases.
MOC Preparation (75*)	0.50 0.50	68.6	Includes: the development of the MOC facility. The development and testing of operational scripts.
Spacecraft Assembly, Test, Launch Ops (72*)	0.80 0.60	177.4	Includes: I&T Team, All other subsystem team members once their subsystem has been delivered to the integration facitiy. Also includes harness fabrication and the harness tech.

Table 7. CONTOUR – Abyss-Lite Comparison of Effort

CONTOUR	Abyss-Lite
4+ Year mission	5 Year mission
4 instruments, high pointing requirements, instrument data rates of 20 Mbps, special interfaces. SSR card, Instrument interface card modifications	1 instrument, modest nadir pointing requirements. Standard 1553 interface for instrument.
Dual model attitude control. Spinning and 3-axis control.	Nadir pointing attitude control
Compatibility with DSN infrastructure	Simple S-band RF for LEO mission
Required complex mission simulations	Simple operational verification
400 kilogram dry mass spacecraft, dust shield, solid rocket motor, three antenna types	200 kilogram spacecraft, single hinge solar array deployment.

Figure 13 shows the "month-to-month" staffing profile projected to design, assemble, test, and launch the Abyss-Lite spacecraft bus. Each one of the vertical bars represents a month in the project's 36-month schedule. This staffing profile does not include the instrument costs. Note that the project phases and significant milestones are shown across the bottom. The legend for the WBS elements within the vertical bars is shown at the top of the graphic. The project peaks at 70 staff-months per month midway between CDR and the beginning of Integration and Test (I&T). At this point, the shop staffing is high due to the fabrication of the spacecraft's components and structure.

The total spacecraft bus development staffing, without reserves, is 1,602 staff months – or 133.5 staff years. The total current dollar value of this labor is just over $28 million. A roll-up of the staffing, subcontracts, and procurements is shown in Table 8. The procurements and subcontracts costs are rolled up from an itemized listing that is based on recent historical expenses. The values have been adjusted to incorporate recent past inflation and are stated in current year dollars. For a future mission, they would need to be adjusted for future inflation. Included in the rollup are the costs of a hydrazine propulsion system. These values also do not carry an explicit reserve. At this point in the design -- and at the current level of mass margin – I would recommend a reserve of 10 percent on the APL labor and 20 percent for the procurements and subcontracts. By incorporating the reserves, the total spacecraft bus costs would be $46.85 million, compared with the $38 to $43 million estimated by the RSDO approach. Given, that the roll-up of the RSDO approach appears several million dollars less than an APL built bus, it would make sense to take RSDO to the next step and solicit the actual vendors for price quotes. RSDO has the advantage of a fixed price contract and also bus replacement for on-orbit failure. If the RSDO price quotes are off, then the APL price is almost as competitive – but with no bus replacement clause.

Table 8 – Spacecraft Bus Rollup. (Without Reserves)

Bus Development Staffing Costs	$28,038,576
Bus Procurement & Subcontracts	$13,341,830
Total	$41,380,406

Once operational, the staffing required to operate this spacecraft should be similar to the TIMED mission – seven people.

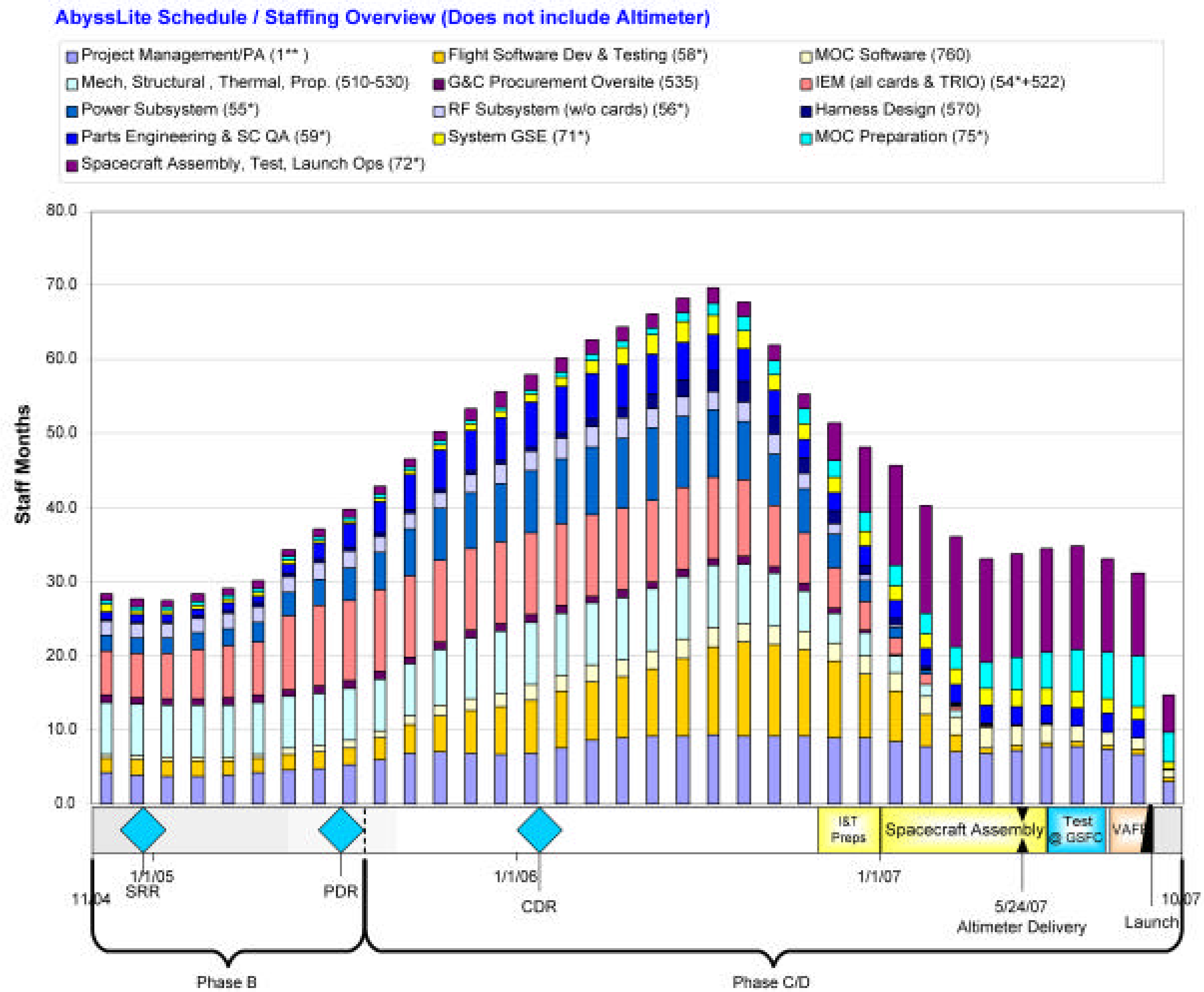


Figure 13. APL In-House Staffing Over the Abyss-Lite Development Cycle.

1.11 Costing the Abyss-Lite Mission

The overall costs to get 'Abyss-Lite' developed and in orbit are shown in Table 9. The instrument costs were developed by Paul Marth in a separate report. A similar reserve strategy was applied to the instrument costs. The Launch Service costs were taken from the NASA SELVS contract. Any mission proposal would need to apply future inflation to the numbers below when developing a spending profile.

Table 9 – Mission Development Cost Rollup (Includes Reserves)

Bus Development Staffing Costs	$30,842,433
Instrument Development Staffing Costs	$13,382,600
Bus Procurement & Subcontracts	$16,010,196
Instrument Procurement & Subcontracts	$7,466,066
Launch Vehicle	$31,000,000
Total A/B/C/D	$98,701,295

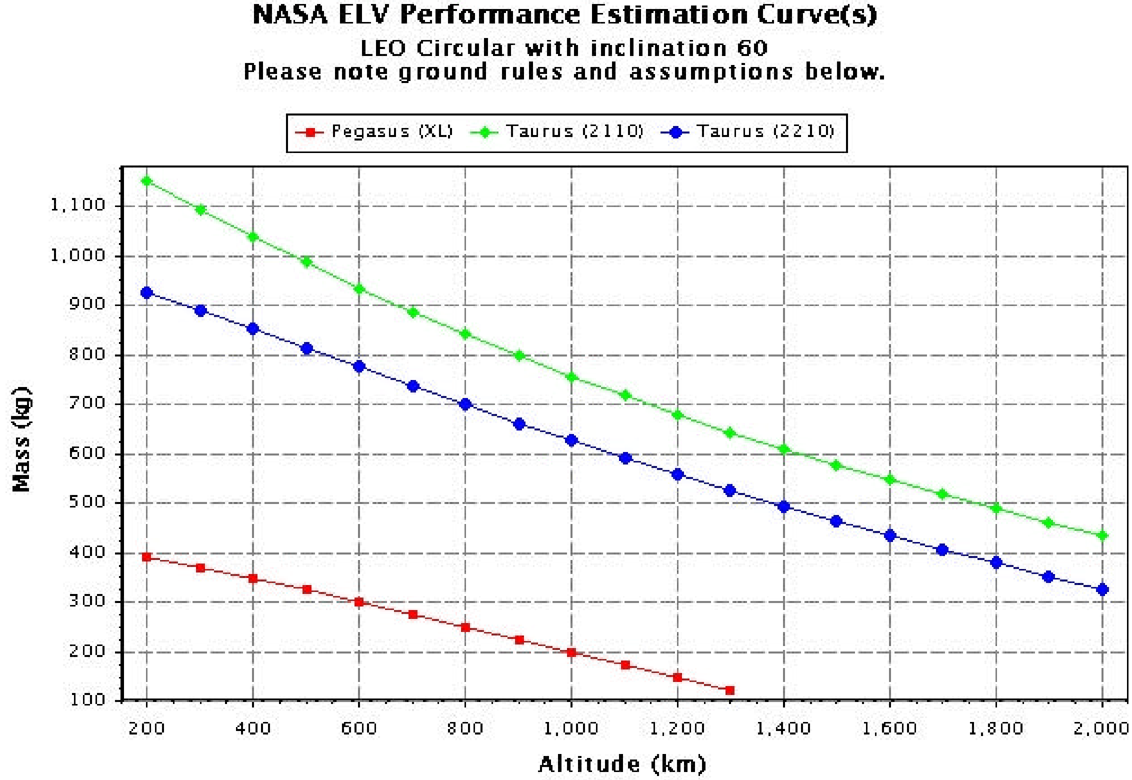
Appendix A – Launch Vehicle Contract Pricing.

NASA SELVS Contract Pricing for a 2007 Launch.

LAUNCH SERVICE	FY'04	FY'05	FY '06	FY '07	TOTAL COST
SELVS-KSC Pegasus (VAFB)	1	13	6	9	29
SELVS-KSC Pegasus (East Coast)	1	14	6	10	31
SELVS-KSC Pegasus (Equatorial)	1	15	6	11	33
SELVS-KSC Taurus (VAFB)	1	22	9	14	46
SELVS-KSC Taurus (CCAS)	1	22	9	15	47

Appendix B – Launch Vehicle Performance.

Important Note: The data contained in these curves are based on ground rules and assumptions located below the plot. Please read this information carefully. This information is intended for NASA customers only.



Assumptions:

Pegasus (XL)
- 38-inch (0.96-meter) separation system.
- Mass of entire separation system is book-kept on the launch vehicle side.
- 220 ft/sec (67 m/sec) guidance reserve.
- Requires VAFB (Vandenberg Air Force Base) waiver.

Last Updated: 5/15/2002 9:57:36 AM

Taurus (2110)
- 38-inch (0.96-meter) separation system.
- Mass of entire separation system is book-kept on launch vehicle side.
- 150 ft/sec (46 m/sec) guidance reserve.
- Launch from South VAFB (Vandenberg Air Force Base).

Last Updated: 5/15/2002 10:04:02 AM

Taurus (2210)
- 38-inch (0.96-meter) separation system.
- Mass of entire separation system is book-kept on launch vehicle side.
- 150 ft/sec (46 m/sec) guidance reserve.
- Launch from South VAFB (Vandenberg Air Force Base).

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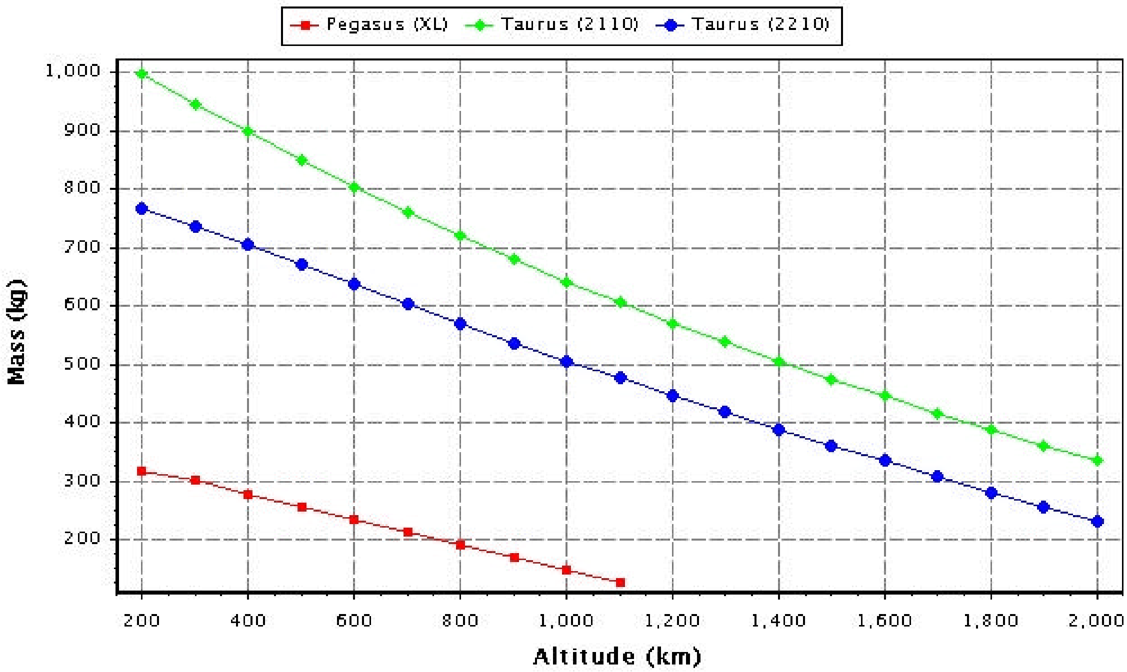
Appendix B – Launch Vehicle Performance (cont'd)

Important Note: The data contained in these curves are based on ground rules and assumptions located below the plot. Please read this information carefully. This information is intended for NASA customers only.

NASA ELV Performance Estimation Curve(s)

LEO Circular with inclination Sun-Synchronous
Please note ground rules and assumptions below.



Assumptions:

Pegasus (XL)
- 38-inch (0.96-meter) separation system.
- Mass of entire separation system is book-kept on the launch vehicle side.
- 220 ft/sec (67 m/sec) guidance reserve.
- Requires VAFB (Vandenberg Air Force Base) waiver.
- Actual inclination for data presented is 98 degrees, not necessarily sun-synchronous for the range of altitudes.

Last Updated: 5/15/2002 9:58:54 AM

Taurus (2110)
- 38-inch (0.96-meter) separation system.
- Mass of entire separation system is book-kept on launch vehicle side.
- 150 ft/sec (46 m/sec) guidance reserve.
- Launch from North VAFB (Vandenberg Air Force Base).
- Actual inclination for data presented is 98 degrees, not necessarily sun-synchronous for the range of altitudes.

Last Updated: 5/15/2002 10:05:32 AM

Taurus (2210)
- 38-inch (0.96-meter) separation system.
- Mass of entire separation system is book-kept on launch vehicle side.
- 150 ft/sec (46 m/sec) guidance reserve.
- Launch from North VAFB (Vandenberg Air Force Base).
- Actual inclination for data presented is 98 degrees, not necessarily sun-synchronous for the range of altitudes.

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Appendix C – RSDO Request Requirements Template

RSDO Rough Order of Magnitude (ROM) Request Form

Mission Name: **Altimeter Lite**

As of: **16 January 2003**

Mission Contact Name	**Edward Reynolds**
Mission Contact Phone	**240-228-5101**
Mission Contact Email	**Ed.reynolds@jhuapl.edu**

Mission Payload Accommodation Requirements:

Payload Mass kg	**34 kg, includes antenna (1 meter diameter circular parabolic dish)**
Payload Power (EOL) Required W	**75 watts, continuous operation**
Science Data Downlink/Band kbps	**25 kbps continuous.**
Science Data Storage Gbits	**5 Gbits, allows for 2+ days of onboard storage.**
Pointing Knowledge arcsecs	**±900 arc-seconds (±0.25 deg.)**
Pointing Control arcsecs	**Instrument is nadir pointed. ±1800 arc-seconds (±0.5 deg.) for pitch and roll. yaw has no requirement.**
Pointing Stability (Jitter) arcsecs/sec	**Instrument not sensitive to jitter.**
Launch Date	**Oct 1, 2006**
Acquisition Date	**Oct 1, 2003**
Mission Life years	**5 years**
Launch Vehicle	**Pegasus XL – 'Wet' spacecraft must be less than 250 kg**
Orbit km	**800 km, circular, 60 degree inclination**
Orbit Knowledge	**1 meter radial, 500 meters cross-track, 500 meters along track**
Radiation Dosage kRads	**10 kRads**
Propulsion requirement	**25 m/s. Trim insertion errors, station keeping, orbit adjust.**
Other considerations	**Instrument interfaces are: 28 volts power (unregulated), a serial command/data interface, and analog temperature sensor wires (5). No deployables. Survival heaters to be provided.**

Other Assumptions Used in the Estimate:

Downlink Communication Band	**S-band or X-band. 4 Mbps data rate required to data downlink. Assume a generic 10 meter ground antenna.**
Redundancy Needed	**75% probability of meeting 5 year mission life**
Propulsion	**Yes. Trim insertion errors, station keeping, orbit adjust.**
Star Trackers	**If necessary for attitude determination, control.**
GPS Receivers	**Required for orbit knowledge and time knowledge**
Schedule Assumption	**Three year maximum for Phases B/C/D**

No. of Spacecraft Types	**1**
No. of Spacecraft	**1**
Instrument positioning on spacecraft.	**Instrument antenna to be below spacecraft C.G. while operational.**
Attitude control desire	**Avoid gravity gradient control**
Thermal interface	**Assume a thermally isolated interface with spacecraft; instrument designed to regulate temperature when nadir pointing and operating. When not operating, spacecraft provided survival heaters maintains temperature above –30C.**
Mechanical interface	**Instrument interface is 1.1 meter circular footprint. Bolt interface at periphery desired. Instrument can be provided as one package or a dish antenna with discrete electronics packages.**

Appendix D – RSDO Response to Request

January 27, 2003

To: Edward Reynolds
Johns Hopkins University, Applied Physics Laboratory

Thank you for your inquiry. You requested a Technical Assessment of RSDO Catalog bus Applicability, and a Rough Order of Magnitude (ROM) estimate of price.

We understand that the ALT-LITE mission comprises one spacecraft in a low earth orbit, carrying a single altimeter payload complement, including a fixed dish antenna. The estimate of mission requirements that you sent to us is enclosed. In the absence of detail knowledge of the instrument, we have assumed easily packaged, benign boxes, with relatively simple interfaces.

RSDO has conducted an in-house Bus Applicability study, identifying at least five potential candidate buses for your mission. The results are detailed in the accompanying spreadsheet. The TRW T100, TRW T200A, and Ball BCP 600 buses appear most interesting.

The Spectrum Astro SA-200B bus may also be of interest: it is identified as having only a 1 year life capability, but we have included their option to incorporate full redundancy. The Orbital MicroStar bus may also be of interest, although the suitability of its configuration for your payload mission requires deeper study.

The Applicability spreadsheet lists your estimated performance requirements and shows the corresponding performance parameters of the buses. Color-coding identifies the following:

- Green Meets requirement
- Light Green Meets requirement with addition of contract option
- Yellow Close to meeting requirement
- Red Clearly does not meet requirement
- Blue Overkill-- much more capability than needed.

Contract options that would be needed are identified in the spreadsheet. Other modifications that would be needed are also identified at the bottom. We have adjusted Catalog prices to include estimates for these modifications.

Please note that three of the buses have fixed solar array wings. Two have single axis drives. We have not conducted an orbital analysis to determine if there are any resulting impacts relative to payload attitude restrictions.

The anticipated ROM price range, when acquired competitively through the Rapid-II contract is expected to be \$38 – 43 M, based purely on the in-house study. The Orbital MicroStar Bus may be somewhat less expensive. These prices are based upon the Rapid-II contracts Statement of Work and Terms and Conditions, including full refund or replacement in the event of spacecraft failure. More accurate ROM prices can be obtained only if RSDO contacts the Vendors with a more detailed Request for Information (RFI). If you would like to pursue an RFI, we will arrange additional time working with you to develop further the mission requirements information.

Rapid Spacecraft Acquisition contracts are performance based, fixed price contracts that baseline utilizing Vendors' practices where appropriate, although all terms and conditions can be modified to suit your needs. Please note that these prices may not be valid for contracts with other terms, conditions, or processes. These estimates are to be used for planning purposes and should not be distributed beyond APL direct employees or government civil servants, except on a need-to-know basis.

For any additional information or discussion, please feel free to contact Bruce Clark, Associate Chief of RSDO, at bruce.w.clark@nasa.gov. 301-286-0404.

Gregory F. Smith

Chief, RSDO
GSFC Code 473

	Rapid-II Summary		TRW	Orbital	TRW	Ball Aerospace	Spectrum Astro
ALT-LITE ROMTS		Units	T100	MicroStar	T200A	BCP 600	SA-200B
75	Max Avg PL Power EOL	W (EOL)	25	50	94	125 w/SA normal to sun	86
34	Payload Mass Limit of Bus	kg	36	68	75	90	100
250 wet mass	Bus Dry Mass (w/o PL)	kg	184.1	58.6	242.4	203	90
25 science gen, 400 downlink	Science Data Downlink Capability	kbps	200	2000	1400	17,000 (two channels)	2500
5000	Science Data Storage Capability	Mbit	16.7	3	2000	2000	2000
900	Pointing Knowldg	arcsec	694	2880	360 - R/P, 324 - Y	12 (3sigma); 17 yaw	317
1800	Pointing Control	arcsec	833 R/P, 2777 Y	2160	1080 - R/P, 1440 - Y	13 (3sigma); 17 yaw	360
No Requirement	Pointing Stability (Jitter)	arcsec/sec	Unknown	36	18 arcsecs per 0.1 sec above 3 Hz	< 4.2 (3sigma)	360
	Slew rate	deg / min	300	60	120	~ 60	240
5	Miss. Design Life	years	3	3	4	6	1 @ .76
Pegasus XL	Compatible LVs	names	Standard Pegasus	Pegasus, Taurus	Athena 1	Pegasus XL, Taurus multiple manifest	Mino, Peg, Taur, Ath, Del, T-II, Atlas
800 km, 60 deg inclination	Nominal Orbit		750 circular sun-synch	580 km, 97.75 deg	600 circular	600 km 60 deg inclin	555 km, 28.5 deg
	Types of Orbits available		N / A	LEO: All inclinations, 580 - 1000 km	LEO 250 - 1000 km, any inclination	450 to 900 km, 0 deg to sun synch	All LEO sun synch, mod PL duty cycle
	External Volume available for PL		6.114 cubic feet atop bus structure	1.5 m dia x 1 m h	80" diameter to LV fairing, 12.7 ft^3	111 dia taper to 70 dia at height of 101 (cm)	.71 x .38 x 1.20 m
	Internal Volume available for PL		N/A	Special (additional PL rings as needed)	N/A	71 cm x 71 cm x 55 cm high	Slots for up to 7 6U VME cards
Earth Pointing	ACS	type	Pitch-momentum bias	3 Axis zero momentum	Pitch mom-biased 3 axis	3 axis, zero net momentum	3 axis zero bias, wheels, mag tqs
YES	GPS	$ receivers		1	1, intl redund		Option #3
	Batteries	type / Ah	NiCd / 9	NiH2 / 10	Super NiCd / 21	NiH2 CPV / 20	NiCd / 4 Ah (3)
	Arrays	cell type, area	5796 N-on-P Si, 3.75 sq. m	Single junction GaAs, 1.57 m^2	Si 1710 x 2 / 2 wings, total 5.71 m^2	GaAs / Ge dual-j, 2.35 m^2	Si, 2.54 m^2
28 VDC	Nominal Voltage	V	28	14, 5, 28	28	28	28
	C&DH Bus Architecture	description	80C86 bus central processor	RS-422 distributed	1553 + 80C86 bus, central processor	HLDC, LLDC, 1553, serial digital	RS422, 1553
	Downlink Formats		STDN	STDN, CCSDS	NASA/STDN & ESA	CCSDS STDN	CCSDS
S or X-Band	Downlink Band		S-Band	S-Band	S-Band	S-Band	S-Band
	Structure	description	Hexagon / Aluminum	Dual faced cyl, AlBeMet/Al h'comb	Hexagon / Aluminum	Rectangular, Al h'comb w/graph face sheets	Rectangular, Al h'comb
Yes	Propulsion	type	Monoprop, Blowdn, N2H4	None	Monoprop, Blowdn, N2H4	None	None
	Propellant Capacity	kg	73	None	73	None	None
25	Max delta v	m / s	548	None	352	None	None
	Heritage mission	name(s)	TOMS-EP	BATSAT, ORBCOMM	ROCSAT	GEOSAT follow-on	MightySat II.1
	nominal schedule	months	18	22	21	32	37
	OPTION	1	Remove Propulsion	Custom struc ring for flexible PL accom	Fine Pointing	Battery Bypass	Full Redundancy
	OPTION	2		Increase power up to 270 w		Monoprop 28.4 kg N2H4	High data: 80 M down, 32 G storage
	OPTION	3		Add 1553/1773 PL data interface			GPS (1)
	OPTION	4		Enhanced data storage 256 Mb			Ground segment integration support
	OPTION	5		Add propulsion module			
	OPTION	6		Reduced pointing accuracy			
	OPTION	7		Operations for 2 years			
	OPTION	8		Delete SW maintain			

Appendix E – Master Equipment List for Minimum Baseline Configuration

Abyss Lite - Base model						3/10/2003
	Qty	Mass (kg)	Growth Allowance (reserves)	Total Mass not to exceed	Ave. Power (watts)	Notes
Abyss Lite		**27.30**	**20%**	**32.76**	**50.00**	
RF		**4.00**	**28%**	**5.20**	**0.00**	
Transceiver	1	0.00	20%	0.00	0.00	Mass and power covered in IEM
S-Band omni antenna	2	1.00	30%	1.30	0.00	Hemispherical low can antennas
Combiner/Diplexer/Xfer Switch Assy	1	2.00	30%	2.60	0.00	Plate mounted components
Cabling	1	1.00	30%	1.30	0.00	Cabling from IEM to antennas
G&C		**16.56**	**14%**	**20.03**	**22.00**	
Star tracker redundant DPU	1	0.86	10%	0.94	4.00	DTU Redundant microASC
Star tracker inner baffles	3	0.21	10%	0.23	0.00	CONTOUR actuals
Star tracker outer baffles	3	1.50	10%	1.65	0.00	CONTOUR actuals
Star tracker heads & cables	3	0.93	10%	1.02	0.00	CONTOUR actuals
Magnetic Torque rods	3	5.10	30%	6.63	6.00	Ithaco Model TR60CFR
Pitch wheel	1	2.55	10%	2.81	5.00	Type-A Ithaco wheel
Wheel electronics	1	0.91	10%	1.00	1.00	Ithaco data sheet
Magnetometer	1	0.50	10%	0.55	1.00	EO-1 TAM mass, Used for momentum dumping
GPS	1	4.00	30%	5.20	5.00	EO-1 mass. Includes antennas
Power		**47.40**	**27%**	**58.82**	**12.00**	
Solar Array - Body mounted cells	6	5.40	30%	7.02	0.00	cells on all six walls.
Deployed panels (six petals)	6	9.00	30%	11.70	0.00	CONTOUR panel density of 3 kg/m2. Assumes 0.5 m by 1 meter area per panel.
Battery	1	14.00	10%	15.40	0.00	12 A-H (30% max orbital DOD)
PSE/PDU	1	12.00	30%	15.60	12.00	Partially redundant, one box; DET system
Shunt Panel	1	1.00	30%	1.30	0.00	
S/A launch restraint&hinges	1	6.00	30%	7.80	0.00	1 kg/panel for hardware
C&DH		**9.15**	**18%**	**10.53**	**33.50**	
IEM	1	9.00	15%	10.35	33.50	Nine card IEM. Duty cycled transmitter
TRIO (Temperture Units)	3	0.15	20%	0.18		45 Temperature readings
Bus Structure	1	**22.56**	**30%**	**29.33**		Note that a 1 inch honeycomb deck is included in the instrument allocation
Thermal	1	**7.00**	**30%**	**9.10**	**20.00**	MLI, heaters, thermisters
Harness	1	**9.23**	**30%**	**12.00**	**2.06**	6.5% of mass + 30 percent growth
Reserve Power					**20.00**	
SC Total		**143.20**		**177.77**	**159.56**	
Launch Mass				**250.00**		Pegasus XL to 800 km; Incl=60 degrees.
Mass Margin- Dry				**29%**		This is in addition to growth

Appendix F – More Agile Single String System

Single String - Enhanced Capability

3/7/2003

	Qty	Mass (kg)	Growth Allowance (reserves)	Total Mass not to exceed	Ave. Power (watts)	Notes
Abyss-Lite		**27.30**	**20%**	**32.76**	**50.00**	
RF		**4.00**	**28%**	**5.20**	**0.00**	
Transceiver	1	0.00	20%	0.00	0.00	Mass and power covered in IEM
S-Band omni antenna	2	1.00	30%	1.30	0.00	Hemispherical low can antennas
Combiner/Diplexer/Xfer Switch Assy	1	2.00	30%	2.60	0.00	Plate mounted components
Cabling	1	1.00	30%	1.30	0.00	Cabling from IEM to antennas
G&C		**27.38**	**15%**	**32.26**	**54.00**	
Star tracker redundant DPU	1	0.86	10%	0.94	4.00	DTU Redundant microASC
Star tracker inner baffles	3	0.21	10%	0.23	0.00	CONTOUR Actual
Star tracker outer baffles	3	1.50	10%	1.65	0.00	CONTOUR Actual
Star tracker heads & cables	3	0.93	10%	1.02	0.00	CONTOUR Actual
Magnetic Torque rods	3	3.00	30%	3.90	6.00	Ithaco Model TR30CFR, momentum dump only
Reaction wheels	3	7.65	10%	8.42	15.00	Type-A Ithaco wheel
Wheel electronics	3	2.73	10%	3.00	3.00	NEAR heritage
Magnetometer	1	0.50	10%	0.55	1.00	EO-1 TAM mass
IRU	1	4.50	20%	5.40	20.00	Honeywell (Clearwater) RLG
GPS	1	5.50	30%	7.15	5.00	EO-1 mass, includes GPS antennas cabling
Power		**46.40**	**27%**	**57.12**	**12.00**	
Solar Array - Body mounted cells	6	5.40	30%	7.02	0.00	cells on all six walls.
Deployed panels (six petals)	6	9.00	30%	11.70	0.00	Assumes 0.5 m by 1 meter area per panel.
Battery	1	16.00	10%	17.60	0.00	Minimum of 14 A-H NiH
PSE/PDU	1	12.00	30%	15.60	12.00	Partially redundant, one box; DET system
Shunt Panel	1	1.00	30%	1.30	0.00	
S/A launch restraint&hinges	1	3.00	30%	3.90	0.00	Placeholder. Need stiff s/a
C&DH		**9.15**	**18%**	**10.53**	**45.00**	
IEM	1	9.00	15%	10.35	45.00	Nine card IEM
TRIO (Temperture Units)	3	0.15	20%	0.18		45 Temperature readings
Bus Structure	1	**27.20**	**30%**	**35.36**		Note that a 1 inch honeycomb deck is included in the instrument allocation
Thermal	1	**7.00**	**30%**	**9.10**	**20.00**	MLI, heaters, thermisters
Harness	1	**10.99**	**30%**	**14.28**	**2.72**	6.5% of mass + 30 percent growth
Reserve Power					**20.00**	
Spacecraft Totals		**159.41**		**196.62**	**203.72**	
Launch Mass				**250.00**		Pegasus XL to 800 km; Incl=60 degrees.
Mass Margin- Dry				**21%**		This is in addition to growth
Unallocated Reserves				**53.38**		

Appendix G – Added Propulsion System

Abyss Lite - OnBoard Propulsion 3/10/2003

	Qty	Mass (kg)	Growth Allowance (reserves)	Total Mass not to exceed	Ave. Power (watts)	Notes
Abyss Lite		**27.30**	**20%**	**32.76**	**50.00**	
RF		**4.00**	**28%**	**5.20**	**0.00**	
Transceiver	1	0.00	20%	0.00	0.00	Mass and power covered in IEM
S-Band omni antenna	2	1.00	30%	1.30	0.00	Hemispherical low can antennas
Combiner/Diplexer/Xfer Switch As	1	2.00	30%	2.60	0.00	Plate mounted components
Cabling	1	1.00	30%	1.30	0.00	Cabling from IEM to antennas
G&C		**42.38**	**16%**	**51.76**	**54.00**	
Star tracker redundant DPU	1	0.86	10%	0.94	4.00	DTU Redundant microASC
Star tracker inner baffles	3	0.21	10%	0.23	0.00	CONTOUR Actual
Star tracker outer baffles	3	1.50	10%	1.65	0.00	CONTOUR Actual
Star tracker heads & cables	3	0.93	10%	1.02	0.00	CONTOUR Actual
Magnetic Torque rods	3	3.00	30%	3.90	6.00	Ithaco Model TR30CFR, momentum dump only
Reaction wheels	3	7.65	10%	8.42	15.00	Type-A Ithaco wheel
Wheel electronics	3	2.73	10%	3.00	3.00	NEAR heritage
Magnetometer	1	0.50	10%	0.55	1.00	EO-1 TAM mass
IRU	1	4.50	20%	5.40	20.00	Honeywell (Clearwater) RLG
GPS	1	5.50	30%	7.15	5.00	EO-1 mass, includes GPS antennas cabling
Propulsion (hydrazine based)	1	15.00	30%	19.50	0.00	Estimate based on other bus designs.
Power		**51.40**	**27%**	**63.22**	**12.00**	
Solar Array - Body mounted cells	6	5.40	30%	7.02	0.00	cells on all six walls.
Deployed panels (six petals)	6	9.00	30%	11.70	0.00	Assumes 0.5 m by 1 meter area per panel.
Battery	1	18.00	10%	19.80	0.00	Minimum of 15.6 A-H NiH
PSE/PDU	1	15.00	30%	19.50	12.00	Partially redundant, one box; DET system
Shunt Panel	1	1.00	30%	1.30	0.00	
S/A launch restraint&hinges	1	3.00	30%	3.90	0.00	Placeholder. Need stiff s/a
C&DH		**9.15**	**18%**	**10.53**	**33.50**	
IEM	1	9.00	15%	10.35	33.50	Nine card IEM
TRIO (Temperture Units)	3	0.15	20%	0.18		45 Temperature readings
Bus Structure	1	**30.24**	**30%**	**39.31**		Note that a 1 inch honeycomb deck is included in the instrument allocation
Thermal	1	**11.00**	**30%**	**14.30**	**40.00**	MLI, heaters, thermisters
Harness	1	**12.29**	**30%**	**15.97**	**2.84**	6.5% of mass + 30 percent growth
Reserve Power					**20.00**	
Spacecraft Totals		**187.75**		**233.06**	**212.34**	
Launch Mass				**250.00**		Pegasus XL to 800 km; Incl=60 degrees.
Mass Margin- Dry				**7%**		This is in addition to growth
Unallocated Reserves				**16.94**		Used for fuel

Appendix H – Enhanced Redundancy

Abyss Lite - Higher Redundancy 3/10/2003

	Qty	Mass (kg)	Growth Allowance (reserves)	Total Mass not to exceed	Ave. Power (watts)	Notes
Abyss Lite		**27.30**	**20%**	**32.76**	**50.00**	
RF		**5.00**	**28%**	**6.50**	**0.00**	
Transceiver	1	0.00	20%	0.00	0.00	Mass and power covered in IEM
S-Band omni antenna	2	1.00	30%	1.30	0.00	Hemispherical low can antennas
Combiner/Diplexer/Xfer Switch As	1	3.00	30%	3.90	0.00	Plate mounted components
Cabling	1	1.00	30%	1.30	0.00	Cabling from IEM to antennas
G&C		**54.54**	**16%**	**65.58**	**80.00**	
Star tracker redundant DPU	1	0.86	10%	0.94	4.00	DTU Redundant microASC
Star tracker inner baffles	3	0.21	10%	0.23	0.00	CONTOUR
Star tracker outer baffles	3	1.50	10%	1.65	0.00	CONTOUR
Star tracker heads & cables	3	0.93	10%	1.02	0.00	CONTOUR
Magnetic Torque rods	3	1.70	30%	2.21	6.00	Ithaco Model TR30CFR, momentum dump only
Reaction wheels	4	10.20	10%	11.22	20.00	Type-A Ithaco wheel
Wheel electronics	4	3.64	10%	4.00	4.00	NEAR heritage
IRU	2	9.00	10%	9.90	20.00	Honeywell (Clearwater) RLG
Magnetometer	1	0.50	20%	0.60	1.00	EO-1 TAM mass
GPS	2	11.00	30%	14.30	20.00	EO-1 mass
Propulsion	1	15.00	30%	19.50	5.00	
Power		**53.40**	**27%**	**65.42**	**12.00**	
Solar Array - Body mounted cells	6	5.40	30%	7.02	0.00	cells on all six walls.
Deployed panels (six petals)	6	9.00	30%	11.70	0.00	Assumes 0.5 m by 1 meter area per panel.
Battery	1	20.00	10%	22.00	0.00	Minimum of 17.8 A-H NiH
PSE/PDU	1	15.00	30%	19.50	12.00	Partially redundant, one box; DET system
Shunt Panel	1	1.00	30%	1.30	0.00	
S/A launch restraint&hinges	1	3.00	30%	3.90	0.00	Placeholder. Need stiff s/a
C&DH		**16.31**	**18%**	**18.77**	**38.50**	
IEM	2	16.00	15%	18.40	38.50	Redundant 8-card IEM
TRIO (Temperture Units)	6	0.31	20%	0.37		45 Temperature readings
Bus Structure	1	**32.00**	**30%**	**41.60**		Note that a 1 inch honeycomb deck is included in the instrument allocation
Thermal	1	**11.00**	**30%**	**14.30**	**40.00**	MLI, heaters, thermisters
Harness	1	**13.00**	**30%**	**16.90**	**3.31**	8% of mass.
Reserve Power					**20.00**	
SC Total Mass		**212.54**		**261.83**	**243.81**	
Launch Mass				**250.00**		Pegasus XL to 800 km; Incl=60 degrees.
Mass Margin- Dry				**-5%**		This is in addition to growth
Unallocated Reserves				**-11.83**		

Appendix I - Recommended Configuration.

Abyss Lite - Recommended Long Life Configuration 3/10/2003

	Qty	Mass (kg)	Growth Allowance (reserves)	Total Mass not to exceed	Ave. Power (watts)	Notes
Abyss Lite		**27.30**	**20%**	**32.76**	**50.00**	
RF		**5.00**	**28%**	**6.50**	**0.00**	
Transceiver	1	0.00	20%	0.00	0.00	Mass and power covered in IEM
S-Band omni antenna	2	1.00	30%	1.30	0.00	Hemispherical low can antennas
Combiner/Diplexer/Xfer Switch As	1	3.00	30%	3.90	0.00	Plate mounted components
Cabling	1	1.00	30%	1.30	0.00	Cabling from IEM to antennas
G&C		**30.54**	**16%**	**36.18**	**40.00**	
Star tracker redundant DPU	1	0.86	10%	0.94	4.00	DTU Redundant microASC
Star tracker inner baffles	3	0.21	10%	0.23	0.00	CONTOUR Actual
Star tracker outer baffles	3	1.50	10%	1.65	0.00	CONTOUR Actual
Star tracker heads & cables	3	0.93	10%	1.02	0.00	CONTOUR Actual
Magnetic Torque rods	3	1.70	30%	2.21	6.00	Ithaco Model TR30CFR, momentum dump only
Reaction wheels	4	10.20	10%	11.22	20.00	Type-A Ithaco wheel
Wheel electronics	4	3.64	10%	4.00	4.00	NEAR heritage
Magnetometer	1	0.50	20%	0.60	1.00	EO-1 TAM mass
GPS	2	11.00	30%	14.30	5.00	EO-1 mass, includes GPS antennas cabling
Power		**49.40**	**27%**	**61.02**	**12.00**	
Solar Array - Body mounted cells	6	5.40	30%	7.02	0.00	cells on all six walls.
Deployed panels (six petals)	6	9.00	30%	11.70	0.00	Assumes 0.5 m by 1 meter area per panel.
Battery	1	16.00	10%	17.60	0.00	Minimum of 15 A-H NiH
PSE/PDU	1	15.00	30%	19.50	12.00	Partially redundant, one box; DET system
Shunt Panel	1	1.00	30%	1.30	0.00	
S/A launch restraint&hinges	1	3.00	30%	3.90	0.00	Placeholder. Need stiff s/a
C&DH		**14.31**	**18%**	**16.47**	**38.50**	
IEM	2	14.00	15%	16.10	38.50	Redundant 8-card IEM
TRIO (Temperture Units)	6	0.31	20%	0.37		45 Temperature readings
Bus Structure	1	**27.68**	**30%**	**35.98**		Note that a 1 inch honeycomb deck is included in the instrument allocation
Thermal	1	**11.00**	**30%**	**14.30**	**40.00**	MLI, heaters, thermisters
Harness	1	**11.25**	**30%**	**14.62**	**2.71**	8% of mass.
Reserve Power					**20.00**	
Spacecraft Totals		**176.47**		**217.83**	**203.21**	
Launch Mass				**250.00**		Pegasus XL to 800 km; Incl=60 degrees.
Mass Margin- Dry				**13%**		This is in addition to growth
Unallocated Reserves				**32.17**		

Note: This configuration does not have a propulsion system in its mass rollup. In the feasibility study (Phase A), the design would be tightened up to see in a propulsion system could be implemented with mass currently allocated to the 'growth allowance'. The system mass margin would still remain a >13%.