Preliminary Design of a High Performance Solar Sailing Mission

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Abstract. The Team Encounter participants, Team Encounter, LLC, AeroAstro Inc., and L'Garde, Inc. have recently completed the preliminary design of the Team Encounter spacecraft, Humanity's First StarshipTM. The spacecraft, intended to be launched as a secondary payload on an Ariane 5 launcher in the first or second quarter of 2004, consists of two parts: the *Carrier*, which transports the Sailcraft beyond Earth's gravity well and the *Sailcraft* which transports its payload of 3 kg out of the solar system. The Carrier must provide a stable platform for deployment and separation of the 4900 m² solar sail, and will provide for video streaming of the Sailcraft as it begins its journey. The Team Encounter Sailcraft is the first spacecraft with the capability to self propel itself out of the solar system with a combination of performance-enhancing tacking maneuvers and an areal density of 3.4 g/m² including payload. The mission analysis made use of an innovative unified Matlab / Satellite Tool Kit model developed by AeroAstro that simulates geometry, attitude and trajectory concurrently. This model was used to optimize motor firing, Sailcraft deployment and mission phasing. The Sailcraft ADCS control scheme was validated via stability analysis and simulation, and the results will be presented in this paper.

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Introduction

The Team Encounter PDR was held February 28 and March 1, 2002 in Houston, TX. This paper provides a technical summary of the design status at PDR especially with respect to the Carrier design (along with its major subsystems), Sailcraft attitude determination and control system, systems engineering, trajectory and mission simulation, assembly, integration, test, launch vehicle integration and the ground segment.

The scope of effort for the mission consists of a *Space Segment* and a *Ground Segment*. These

are in turn broken down into the lower level mission elements, which are identified in Fig. 1. As in any significant engineering project, the PDR effort began with identification of requirements. From the top level product requirements, provided by the Customer, the technical team derived and flowed mission requirements, which in turn flowed into the Space and Ground Segments. The requirements traceability process was flowed down to the subsystem and component level, in order to allow for preliminary identification of spacecraft components and vendors. The requirements flow

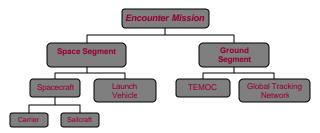


Figure 1. Mission Element Breakdown

and traceability process used on the Encounter program is shown in Fig. 2.

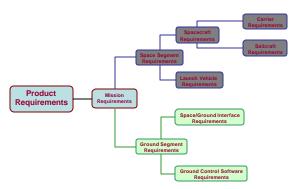


Figure 2. Requirements Flowdown

Space Segment Overview

The space segment consists of the *Spacecraft* and *Launch Vehicle*. As shown in Fig. 1-1, the spacecraft consists of two major elements: the *Carrier* and the *Sailcraft*. The Carrier propulsively transports the Sailcraft beyond the earth's gravity well. Therefore, the Sailcraft is the Carrier's payload until they separate. The Sailcraft, which is completely autonomous after separation from the Carrier, transports its 3kg payload out of the solar system. It is fully dependant on the solar sail to provide the propulsion needed for its mission. These major elements are shown in Fig. 3.

The Carrier provides typical spacecraft resources to the payload, namely ADCS, power, propulsion, and a protective structure. It also provides several mission unique services, such as a stable platform for deployment of a 76 m x 76 m solar sail, jitter-free imaging of the solar sail separation and initial trajectory, as well as accommodation of secondary payloads.







Carrier (Deployed View)

Figure 3. Spacecraft Architecture

The Sailcraft is an atypical spacecraft, utilizing an extremely light-weight gossamer structure, a suspended membrane. and deployment mechanism. The Sailcraft utilizes autonomous attitude determination and control system that features dual-pitch attitude modes for trajectory and visibility optimization. It generates its own power through solar sail mounted thin-film solar array panels, and power distribution and control is handled via an onboard avionics system.

The major mission phases for the Carrier spacecraft are:

- Launch
- Separation
- GTO/Secondary Payload Mission
- Earth Escape
- Sail Deployment
- Sailcraft Separation
- Sailcraft Cruise

The Encounter spacecraft will be launched on an Ariane 5, accommodated by the flight proven Ariane Structure for Auxiliary Payloads (ASAP). As a result, all launch operations will take place at the Centre Spatial Guyanese (CSG)

in Kourou, French Guiana. At CSG pre-launch spacecraft testing, hazardous operations (such as pressurization of the cold-gas propulsion system and installation of the solid rocket motor), and combined operations that follow installation on the ASAP ring adapter will take place.

Ground Segment Overview

The Ground Segment is composed of two major elements: the Team Encounter Operations Center (TEMOC) and the Global Tracking Network. TEMOC will provide all the command and control resources for the Encounter mission, and will be located at the Team Encounter headquarters in Houston, TX. It will also provide for communications connectivity between the global tracking network and the launch site, as shown in Fig. 4.

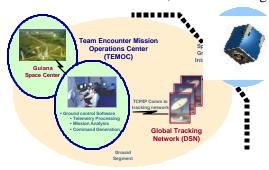


Figure 4. Ground Segment Architecture

The global tracking network will make use of the proven capabilities of the 26 m antenna resources of the Deep Space Network (DSN) managed by JPL.

Carrier Description

The Carrier primary structure (and load carrying member) is the centrally located solid motor, baselined as an ATK Star 12G. Lightweight secondary structure carries all support equipment including ADCS, communications, command and data handling (C&DH), propulsion, imaging, and power subsystems. Deployable solar panels not only meet Encounter's power requirements, but also provide for stabilization during the high-rate spin mode necessary for the motor burn phase.

The top plate of the Carrier structure acts as the integration platform and interface plane for the Sailcraft and associated hardware. These components may be found in Fig. 5.

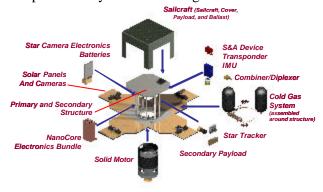


Figure 5. Encounter Exploded View

The Encounter spacecraft total mass allocation is constrained by Ariane. The maximum mass allowed for an ASAP payload is 120 kg. The mass estimate as of PDR is 111.3 kg, which allows for only 8.7 kg margin (7.2%) during the ongoing detail design process. The Carrier mass breakdown is provided in Fig. 6.

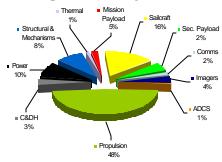


Figure 6. Carrier Mass Breakdown

The power generation capability is in turn driven by the peak power requirement. The total power capability (peak load) is approximately ~295 Watts. The power usage distribution is provided in Fig. 7.

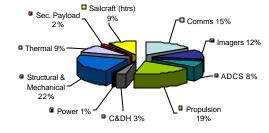


Figure 7. Carrier Power Breakdown

Carrier Mission Phases

The Carrier mission begins at separation from the launch vehicle, and ends after completing to downlink imaging data of the Sailcraft separation event. However nothing in its design precludes the possibility of an extended mission, including secondary payload operations after this time. The Carrier mission may be broken down into the following four major mission phases:

- 1. Post-Launch Check out and Configuration
- 2. GTO Operations
- 3. Escape Burn
- 4. Sailcraft Deployment

The 1st Carrier Phase, Post-Launch, will begin at the ASAP separation event and continue up to the GTO portion of the mission. The significant events of the Post-Launch phase include separation, solar array deployment, post-deployment monitoring, rate nulling, and sun search. It is anticipated that the events from separation to sun lock will require approximately 45 minutes to complete.

After the initial configuration activities of the Post-Launch phase, the 2nd Carrier Phase, GTO Operations, takes place. GTO phase activities basically prepare the Carrier for its injection into heliocentric orbit with the firing of its solid motor. The GTO phase activities will include spacecraft checkout, mission rehearsals, and also the opportunity for conducting an auxiliary mission with a secondary payload. The GTO phase duration will be between 30 and 90 days. depending on the orbit configuration and intended trajectory. Because of this extended duration, several eclipse periods will be observed, during which only essential spacecraft systems will be operating in order to conserve power. Depending on requirements of an auxiliary payload, two operational attitude modes will be available in GTO. In Sun Lock mode the spacecraft is 2-axis stabilized and will be allowed to drift in yaw. In Star Lock mode the spacecraft is 3-axis stabilized.

In the 3rd Carrier phase, Escape Burn, the spacecraft is transitioned from GTO to the

appropriate perigee burn attitude. Operations will include 3-axis stabilization, slewing of the spacecraft to burn attitude, spinning up to 120 RPM (about the motor thrust vector axis), final pre-burn orbit determination and telemetry verification, and the perigee motor burn. The total duration of the Escape Burn phase is estimated at 300 minutes, though the perigee burn itself will take only about 15 seconds to complete.

In the 4th and final Carrier phase, Sailcraft Deployment, the most critical mission activities take place. These activities must be completed per a strict mission timeline, which is initiated as soon as the escape burn is terminated. The major operations during this phase include:

- De-spinning the Spacecraft from perigee burn
- 3-axis stabilization
- Slew to cover release attitude
- Cover release
- Slew to Sailcraft release attitude
- Sailcraft inflation and rigidization
- Sailcraft release
- Sailcraft Imaging
- Video Streaming

The Sailcraft Release phase duration is estimated at 150 minutes. Sailcraft release operations will complete the primary Carrier portion of the Team Encounter mission, and will also initiate the Sailcraft mission phases. The Carrier may continue to operate for an extended mission to support an auxiliary payload.

Carrier Subsystems Design

Communications Subsystem

The communications subsystem (CSS) accommodates Telemetry Tracking and Control (TT&C) during the Carrier mission phases. Other key requirements are that it provide uplink at 2 kpbs and downlink at >64 kbps. It must be compatible with pseudo random noise (PRN) ranging, and communicate at any Carrier attitude during both GTO and heliocentric orbits. The component suite includes the transponder,

antennas, combiner, diplexer, and waveguides. Major components were selected as driven by the mission architecture requirements, as well as demonstrated heritage. The L3 S-band transponder was selected because it has flown successfully on over 80 missions. It is compatible with our ranging requirements, consumes 35 Watts, and transmits at over 5 Watts, and will easily meet our uplink and downlink transmission requirements.

For the antenna, high and low-gain antennas were traded. It was possible to select the low gain antenna once the DSN 26 m ground resources were baselined, and they were also chosen because of their relative simplicity and lower cost. The baselined antenna configuration will be a quadrahelic omni-directional type, providing both broad hemispherical coverage and a low back lobe radiation pattern. The CSS block diagram is shown in Fig. 8.

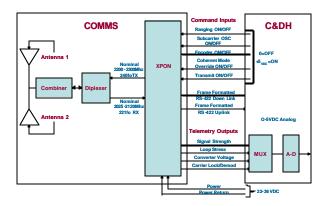


Figure 8. Comm. Subsystem Design

The uplink and downlink budgets each easily closed with adequate link margin, as can be seen in Fig. 9 and 10. In each, the baseline case is the DSN-26 m antenna.

Team Encounter Up Link Budget	Units	Link Budget 13m	Link Budget DSN-26m	Link Budget DSN-70m
G/S Antenna Elevation	deg	5	5	5
Link Operating Frequency	MHz	2300	2300	2300
Bit Rate	bps	2000	2000	2000
Link Bandwidth	Hz	1660	1660	1660
S/C Height	km	49000	49000	49000
S/C RF Output	Watts	50	50	50
Eb/No for 1e-5 BER	dB	9.6	9.6	9.6
Diameter of G/S Antenna	m	13	26	70
Link Evaluation Parameters				
G/S Antenna Gain	dBi	47.32	53.34	61.94
G/S Antenna Beam Width	deg	0.71	0.356	0.132
Noise at Receiver Input	dBm/Hz	-136.65	-136.65	-136.65
S/C Transmitter Output	dBw	63.31	69.33	77.93
Total Link Losses	dB	15.08	14.15	15.05
Slant Range	km	54456.59	54456.59	54456.59
Path Attenuation	dB	194.35	194.35	194.35
Atmospheric Attenuation	dB	0.4589	0.4589	0.4589
Rain Attenuation 10 mm/Hr	dB	1	1	1
Signal at Receiver Input	dBm	-100.1	-104.06	-85.46
Signal to Noise Ratio at Receiver	dB	36.56	42.58	51.18
Link Margin	dB	21.5	28.27	36.12

Figure 9. Up-Link Budget

Team Encounter Down Link Budge	et Units	Link Budget 13m	Link Budget DSN-26m	Link Budget DSN-70m
G/S Antenna Elevation	deg	5	5	5
Link Operating Frequency	MHz	2300	2300	2300
Bit Rate	bps	64000	238,635	1,761,000
Link Bandwidth	Hz	56320	210,000	1,550,000
S/C Height	km	49000	49000	49000
S/C RF Output	Watts	5	5	5
Eb/No for 1e -5 BER	dB	9.6	9.6	9.6
Diameter of G/S Antenna	m	13	26	70
Link Evaluation Parameters				
G/S Antenna Gain	dBi	47.32	53.34	61.94
G/S Antenna Beam Width	deg	0.71	0.35	0.132
Noise at Receiver Input	dBm/Hz	-131.8	-127.25	-118.56
S/C Transmitter Output	dBm	36.99	36.99	36.99
Total Link Losses	dB	13.6	13.6	13.6
Slant Range	km	54456.59	54456.59	54456.59
Path Attenuation	dB	194.35	194.35	194.35
Atmospheric Attenuation	dB	0.4589	0.4589	0.4589
Rain Attenuation 10 mm/Hr	dB	1	1	1
Signal at Receiver Input	dBm	-110.42	-104.06	95.46
Signal to Noise Ratio at Receiver	dB	21.75	23.18	23.101
Link Margin	dB	6.69	8.12	8.04

Figure 10. Down-Link Budget

Imager Subsystem

While all subsystems are necessary for a successful mission, the Imager subsystem is arguably the most essential since, without it, it would be impossible to fully verify successful deployment of the solar sail, given the Sailcraft has no independent communications capability. As shown in Fig. 5, the 10 imaging cameras are arranged on the deployed solar array panels.

A number of tests and analyses were performed to determine suitable image quality requirements, namely resolution, frame rate, color depth, and compression. The results of those trades are shown in Table I, below.

Table I. Imager Subsystem Trades

Trade	Options	Justification	
	640x480 (VGA)	Options considered based on CCDs	
Resolution	752x582	in COTS cameras. Standard "VGA"	
	1792x1200	computer resolution chosen.	
	Custom	Higher speed serial link will be	
Interface Protocol	R S-422	required for Imager read out. Will be used for camera control.	
	1 Frame/Second		
Frame Rate	3 Frame/second	1 & 3 frames are best suited to sail velocity and bandwidth.	
	30 Frame/second	velocity and bandwidth.	
	JPEG	A modified JPEG was selected for	
Compression	Modified JPEG	best compression ratio of typical sail	
	MPEG	image and color space.	
Color Depth	4 Bit	0.0%	
	8 Bit	8 Bit color space was selected due to the limited color range of sal.	
	24 Bit	are minica color range of sail.	

The initial velocity of the Sailcraft is expected to be approximately 3.7 m/sec. The motion of a 4900 m² object moving at this velocity in one second is virtually un-detectable. Therefore a frame rate of 1 fps is deemed adequate. This greatly relieves the transmission bandwidth demand normally associated with broadcast quality video at 30 fps.

The three primary components of the Imager subsystem are the lens, camera, and image processor. For the lenses, a range of focal lengths from 2.8 to 50 mm are provided to ensure adequate coverage of the Sailcraft as it begins its trajectory out of the solar system. The image processor will multiplex power and image data for up to 10 cameras. Compression is applied via a color look-up table (LUT), Huffman, and Direct Cosine Transformation (DCT).

Attitude Determination and Control System

The attitude determination and control system (ADCS) requirements, while not very demanding compared to many NASA science missions, still pose a significant challenge for a low cost space mission. It must maintain the satellite in a power positive and thermally viable attitude, it must support 2-axis, 3-axis, and spin stabilization modes, and maintain a jitter-free platform during the imaging phase of the mission. The component suite includes a star tracker for 3-axis attitude determination, an IMU for attitude propagation, a medium sun sensor (MSS) for simple 2-axis sun-referenced attitude determination, and cold-gas thrusters actuators.

At PDR, preliminary selection of ADCS sensors was completed. The Terma star tracker was selected, mainly for its superior packaging characteristics. The AeroAstro manufactured medium sun sensor was selected for its demonstrated performance, strong heritage, and low cost. Similarly, the Litton LN-200 IMU was selected for its low power consumption, heritage, and low cost.

The ADCS modes are defined in Table II and the mode usage by phase is shown in Fig. 11. Mode switching is accommodated autonomously via on-board timer, or is otherwise initiated via ground command. Implementation of the ADCS modes requires a robust flight software implementation. The ADCS flight software control logic has been developed, and is shown schematically in Fig. 12.

Table II. ADCS Mode Definitions

Mode	Definition	
Sleep	All sensors and actuators disabled.	
Attitude Monitor	Monitor sensors; propagate attitude.	
Rate Nulling	Bring 3-axis rates to near zero and hold.	
Sun Search	Sequence of slews to locate sun.	
Sun Lock	Point nozzle to sun; null yaw rate.	
Star Search	Slow Rotation about sun line until in-track.	
Star Lock	Point nozzle to sun; hold yaw angle.	
Inertial Hold	Hold 3-axis attitude based only on IMU data.	
Inertial Slew	Slow 360 ° rotation about an axis; IMU only.	
Slew to Target	3-Axis rotation to selected target; IMU only.	
Yaw Spin	Rapid spin about thrust axis.	

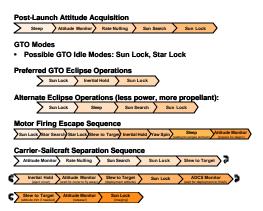


Figure 11. ADCS Modes by Phase

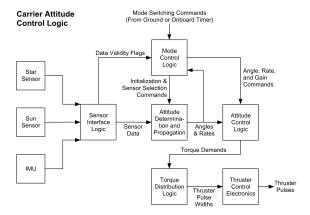


Figure 12. ADCS Control Logic Flow

Propulsion Subsystem

The Carrier primary propulsion system consists of a solid rocket motor. The motor is needed to provide sufficient ?V to the Carrier to achieve earth escape velocity. In order to minimize non-impulsive efficiency losses, high thrust between

4000 and 9000 N is needed, and a minimum Isp of 270 seconds. The ATK Star 12G solid motor was chosen because it meets all performance requirements, and has a successful and repeatable performance record. Because of its location in the Carrier, the solid motor casing serves a dual purpose, acting as the primary structure and load bearing member. Figs. 13 and 14 provide views of the primary propulsion subassembly and a Star 12G solid rocket motor during lot-test firing, respectively.



Figure 13. Solid Motor Assembly



Figure 14. Star 12G Motor Test Firing (Courtesy ATK)

The Carrier secondary propulsion subsystem provides the necessary ?V for pointing, spin-up, and de-spin for all phases of the mission. A cold-gas propulsion subsystem was baselined in order to meet the fine adjustment requirements for the imaging phase, to avoid contamination by thruster plumes, and to prevent damage to the delicate solar sail during deployment and after Sailcraft separation.

The secondary propulsion subsystem block diagram is shown in Fig. 15. As can be seen in this figure, in addition to propulsive services, the cold gas supply is also tapped into for the boom

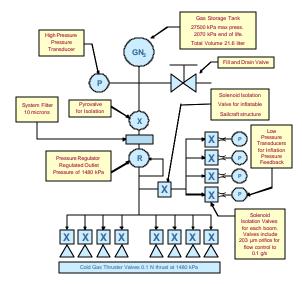


Figure 15. Secondary Propulsion System Block Diagram

structure inflation system, accommodating the Sailcraft structure deployment requirements. A series of valves allow for isolation and independent inflation of each of the four structural booms at a controlled flow rate.

The Moog cold gas thrusters baselined provide a thrust level of 0.1~N and a minimum impulse bit of $375~\mu N$ -s at a working pressure of 1.5~MPa. The working gas is Nitrogen, which provides an Isp of approximately 59 seconds at the minimum expected gas temperature of $3^{\circ}C$. Based on this performance, a propellant budget corresponding to a total Nitrogen supply of approximately 5.9~kg is required. The breakdown of the propellant usage is shown in Fig. 16.

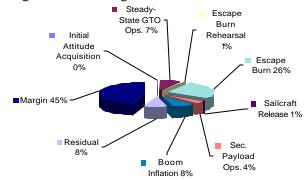


Figure 16. Propellant Budget

For the nominal tank sizing, a healthy 45% margin is provided to accommodate leakage and extended mission requirements.

C&DH Subsystem

The Carrier Command and Data Handling (C&DH) subsystem provides telemetry to the ground, allows commanding of the spacecraft, either autonomously or via ground command, provides for electrical interfaces between other subsystems, maintains the commanded ADCS mode, and controls all imager subsystem functions. The C&DH subsystem is based on AeroAstro's NanoCore Electronics Bundle (formerly Bitsy-DX Kernel) plus additional I/O modules. The NanoCore bundle consists of a power control and telemetry board and an On-Board Computer (OBC). Additional I/O modules that interface with the bundle include an electrical interface board that interfaces Encounter-specific peripherals to the OBC, a propulsion power interface board, and the image processor.

The propulsion power interface board includes valve drivers, propulsion heater control, the pulse width modulation control circuit for thruster control, and control of the Sailcraft deployment and separation mechanisms.

The electrical interface board (EIB) provides electrical interface and routing of Encounter specific peripherals to OBC standard logic levels. It provides RS-422 interfaces to the star tracker, imaging subsystem, and secondary payload, and a RS-485 interface for the IMU. It also provides for electrical isolation to protect the OBC in the event of component latch-ups.

The electrical interfaces between the C&DH subsystem and all other subsystems is shown schematically in Fig. 17.

A primary concern for the Encounter spacecraft C&DH components is their susceptibility to environmental radiation, especially when the spacecraft passes through the Van Allen radiation belts while in GTO. The OBC baselined at PDR is a commercial unit, derived from automotive heritage. Pending test results, it is believed to have a radiation tolerance capability of at least 5 kRad. An analysis was performed to assess the radiation dose as a

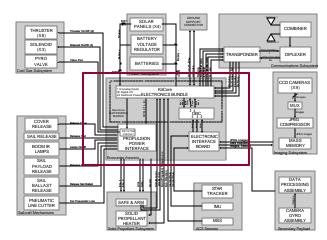


Figure 17. C&DH Subsystem Electrical Interfaces

function of the predicted orbit parameters, using the most conservative results from available NASA models for trapped protons and electrons. The resulting dose-depth curves shown in Fig. 18 provide the effective equivalent Total Ionizing Dose (TID) as a function of exposure time and the thickness of Al absorber material between the sensing surface and the radiation environment.

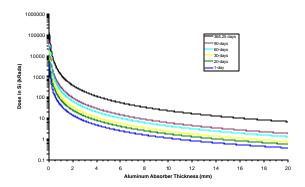


Figure 18. TID vs. Absorber Thickness

From this curve, we see that to maintain the TID at 50% of the rated level (2.5 kRad) for the worst-case 90-day GTO mission requires over 17 mm of aluminum shielding. Because this amount of shielding may present both packaging and mass issues, post-PDR trades were performed to determine the suitability and availability of test verified, radiation tolerant processors capable of operating in significantly higher radiation environments. This study identified two applicable 300 kRad tolerant processors. After further evaluation, one will be

selected and incorporated into the NanoCore Electronics Bundle.

Flight Software

The embedded Flight Software (FSW) must provide for the OBC interface, kernel software, middleware and device drivers, Encounter mission specific application software and all memory requirements. An early trade was to determine the type of operating system to employ. The options considered were a commercially available real-time operating system (RTOS) or development of an OBC specific deterministic kernel. The VxWorks RTOS was chosen because it has significant heritage in a wide variety of aerospace critical applications, including space missions. VxWorks also provides for a relatively seamless approach to code development, debugging, and testing.

The software architecture relies on VxWorks for task scheduling, pre-emption, and queuing. The OBC kernel provides for the bootstrap, memory management, error handling, basic I/O handling. and interrupt handling. The *middleware* provides for basic communications, file handling, external device drivers, and a real-time command handler. Finally, the Encounter application software provides for the imager manager, the Sailcraft manager. command handler. communications handler, ADCS, telemetry, and power management. The software data flow is shown schematically in Fig. 19.

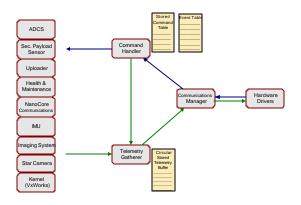


Figure 19. FSW Data Flow

Power Subsystem

The power subsystem provides for all power resources required to operate the spacecraft and recharge batteries from GTO to the end of the Carrier mission. It also provides for sufficient battery power to maintain critical systems during eclipse.

Leading up to the PDR, several significant power subsystem trades were performed. Deployable solar arrays were selected, vs. bodymounted configuration for the larger cell area available and improved power margins and also the improved inertial properties of the spacecraft. Triple junction GaAs cells were chosen for their high efficiencies, and higher structural stiffness versus thin film solar cell panels. Finally, Lithium ion batteries were selected for their improved power density vs. Nickel metal hydride and Nickel cadmium.

The four deployed solar array panels provide 295 Watts at the minimum 24% cell efficiency. The rechargeable Li-ion battery pack provides a total energy capability of 232 Wh. The batteries were sized based on GTO eclipse requirements.

The Power Management System is incorporated into the NanoCore electronics bundle, and its functionality is illustrated in Fig. 20.

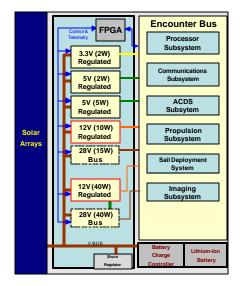


Figure 20. Power Management System

The Power Management System provides for overall control of power to spacecraft subsystems. It has regulated outputs at 3.3, 12, and 28 VDC. Each power output line has a programmable current monitor for latch-up/short circuit monitoring.

Structural Subsystem

The Carrier structural subsystem provides the interface to the Micro ASAP5 separation system, and the support and separation interface to its Sailcraft payload. It must also withstand the Ariane 5 launch environments. The driving environmental factors are:

- Longitudinal stiffness > 90 Hz
- Lateral Stiffness > 45 Hz
- Longitudinal Static Loading -7.5 g's / +5.5 g's
- Lateral Static Loading +6.0 g's / -6.0 g's

The primary structure central core is the Star 12 G rocket motor housing, made from a graphite composite structure. Top and bottom panels carry loads from the Star 12G motor to secondary structure, and provides mounting surfaces for instruments and components. These elements are shown in Fig. 21.

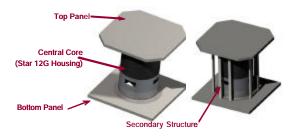


Figure 21. Carrier Structure Design

At PDR the top and bottom panels were baselined as aluminum (Al) honeycomb (11.2 mm thickness) with Al facesheets (0.75 mm thickness). A post-PDR trade examined options available for mass reduction. It was found that one could reduce the panel thickness to 9.5 mm total (with 0.51 mm facesheets), affording a significant mass reduction. The use of graphite / cyanate ester composites was not deemed suitable as further mass reduction was small compared to the additional complexity in fabrication, handling, and additional cost.

The secondary structure surrounds the central core, and is also of Al construction. It consists of vertical support columns that run between the top and bottom panels, as seen in Fig. 21. The secondary structure provides GSE and hardware mounting points for all component brackets.

Finite element analysis (FEA) was performed using Patran/NASTRAN to verify all effective load cases, determine stress margins, and identify the first modes. Even with reduced panel thickness, all structural load cases resulted in margins greater then 4, based on a factor of safety of 1.25. The first lateral and longitudinal modes were 47.7 and 93.8 Hz, exceeding the Ariane requirements of 45 and 90 Hz, respectively. Analysis was also performed to validate the deployed solar panel sizing with respect to the maximum loading conditions associated with solid motor firing (10 G) and the lateral spin loading (6 G). Stress and displacement even under these worst case conditions were well within limits.

The deployable solar array panels are aluminum facesheet (thickness = 0.25 mm) with aluminum honeycomb core. Graphite fiber, cyanate ester composite facesheets may alternately be used for mass reduction purposes. Each panel is retained and released by a Starsys Qwiknut 2500, non pyro-technic release nut. Each panel has eight cup/cone hinges and four cup/cone snubbers of Titanium alloy construction, designed by Planetary Systems Corporation (PSC). The cup cone hinges are self latching and posses very high stowed stiffness, The deployment angle, from stowed to deployed, is 90°. A representative hinge is shown as Fig. 22.



Figure 22. Solar Array Hinge (Courtesy PSC)

The deployed solar array panels also serve double-duty as platforms for the imaging cameras. It is important that these cameras provide sharp pictures of the Sailcraft on its initial trajectory, free of blur induced by jitter from the separation event or thruster use. An analysis was performed to assess this situation based on a derived requirement that the camera boresight does not move more than half a pixel during the worst case camera integration time. Meeting this requirement is most challenging for the longest focal length used. With lenses selected between 2.8 and 50 mm, the corresponding maximum allowed body rates are from 0.9 to 0.05 deg/sec, respectively. So long as camera imaging is time-phased to use the wider angle lenses during the early part of the imaging phase (approximately the first six minutes), the image quality will remain free of jitter induced blur.

Thermal Subsystem

The thermal subsystem is designed to maintain thermal environments for all components within their specified limits. The thermal subsystem also provides special services to the Sailcraft payload, namely IR heaters to prevent premature boom rigidization. Also, the Carrier and Sailcraft interface must be thermally insulating to allow for nominal rigidization after deployment. All critical components will be heater controlled using Kapton strip heaters and temperature sensors. Solid state switching of heaters will be accomplished via logic in FSW operating on the OBC. Multi-layer insulation (MLI) blanketing will be needed at several locations wrapped around the Carrier body, leaving openings for component radiators. MLI will also be placed on the bottom surface of the Carrier, and around the solid rocket motor nozzle. MLI will be placed on the back surface of dedicated camera radiators mounted on the solar panels.

Component boxes are attached to the support structure with insulating spacers to minimize the thermally conductive path. Conversely, radiators are silver epoxied to their respective components to ensure a good thermal path. Radiators will also be coated with a high emissivity but low absorbtivity coating (like Magnesium oxide) to ensure they radiate well but don't absorb significant solar flux.

The thermal design of the spacecraft was made more challenging because the components have very different duty cycles for the mission phases, and correspondingly a wide disparity of heat production. Also, the thermal system must meet all of its requirements in both 3-axis and spin-stabilized operated modes, and be able to handle the heat generated during the solid motor firing.

A Carrier thermal model was constructed in SINDA. The 600 node model was generated in FEMAP and used to determine worst case hot and cold transient thermal timelines. In addition, Mathcad models were used to predict radiator sizing and heater requirements. Temperature gradients throughout the Carrier were calculated for nominal GTO, worst case cold (after eclipse), worst case hot (after engine firing), and after de-spin. The results of this analysis are summarized in Table III.

Table III. Thermal Analysis Results

	Minimum Temp (°C)	Margin (°C)	Max Temp (°C)	Margin (°C)
Nitrogen Tanks	3	23	38	43
Bitsy	9	29	23	27
Batteries	3	3	35	5
Transponder	-13	7	49	1
TERMA Optics	9	49	22	7
TERMA Electronics	8	28	33	17
LoPASS Optics	10	10	28	12
LoPASS Electronics	7	37	26	34
IMU	-23	12	20	45
Solid Propellant	17	12	25	10

The analysis indic ated that all major components will be kept within their thermal limits over the Carrier mission phases. The relatively simple thermal design, utilizing only local radiators, strip heaters, and MLI blankets is sufficient to meet all mission requirements.

Carrier Secondary Payload Accommodation

The Carrier will carry at least one, and likely multiple secondary payloads. Prior to PDR, a case study was performed to assess the impacts of accommodating various candidate secondary payloads.

Depending on the size and configuration of a secondary payload, it may be accommodated on the base panel surrounding the primary structure, or mounted on the vertical struts that compose the secondary structure. Payloads consuming up to 5 W, with a mass of 5 kg, and a packaging volume up to 2800 cm^3 volume are relatively easy to accommodate on the Carrier. The ADCS sensor selection and cold gas thruster actuators are sufficient to provide the $\pm 3^\circ$ attitude knowledge and $0.1 - 1.5^\circ$ /s controlled slew rates required to adequately test optical and attitude sensors, communications payloads, or other instruments.

Sailcraft Subsystems Design

Sailcraft Overview

The Sailcraft must deliver a 3 kg payload out of the solar system, using only solar sail technology for propulsion. It must continue augmenting the ?V of the payload until it reaches a distance of 14.0 AU from the sun, and the Sailcraft active control systems and on-board electronics most be operational for at least the initial 365 days (out to 4.0 AU). The solar system escape mission is extremely challenging, and its successful demonstration will be historic in that no other spacecraft has been able to self propel itself from the solar system. By contrast, Pioneer and Voyager used gravity assists on their missions.

Sailcraft structure technology, under development at L'Garde, is extremely high performance, reaching an areal density of 3.4 g/m². The main sail material is 0.9 µm thick, and utilizes a proprietary metallization coating process to achieve a propulsive reflectivity over 85%. Its size, approximately 76 m on an edge, providing a reactive area of 4900 m², will represent the largest structure ever deployed in space. It utilizes an innovative control system developed by AeroAstro that provides for passive pitch/roll axis stability, a mass drop for pitch axis control, and active yaw axis control system. The Sailcraft coordinate system is shown in Fig. 24, and the design overview in Fig. 25.

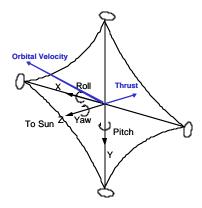


Figure 24. Sailcraft Coordinate System

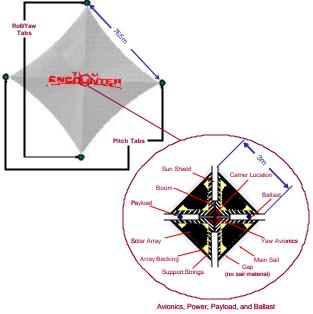


Figure 25. Sailcraft Design Overview

Sailcraft Mission Phases

The Sailcraft mission phases begin when the Carrier mission phases end. The major Sailcraft mission phases are:

- Sailcraft Deployment
- Initial Sailcraft Flight
- Sailcraft Tack
- Sailcraft Cruise

The 1st Sailcraft mission phase, Sailcraft Deployment, overlaps with the corresponding Carrier mission phase and starts after the escape

burn phase and completes with release of the deployed Sailcraft. The event sequence is as follows:

- Cover release
 - IR lamps pre-heat booms
 - Canister jettisoned
- Payload/Ballast/Spreader released from Carrier
- Sailcraft boom inflation (all four simultaneously)
- Boom inflation pulls main sail into position
- Boom deployment check to verify proper inflation
- Boom Rigidization
 - IR lamps turned off
 - EEPROM data uploaded
- Final Check
- Zero Momentum Sailcraft Release from Carrier

This entire sequence of events takes place on a critical mission timeline, and the expected total duration is 30 minutes.

The 2nd Sailcraft mission phase is also the start of the Sailcraft trajectory independent of the Carrier. It begins upon release from the Carrier, and ends when the ballast mass is dropped. For the initial flight phase the Sailcraft is normal to the sun-line. Pitch and roll are stabilized at 0°, and the yaw control system is active, and turning to a 0° attitude. The total duration of this phase is 5 days. The orbit trajectory for this phase is shown below, in Fig. 26.

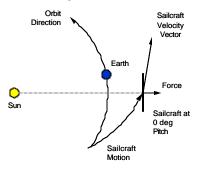


Figure 26. Sailcraft Initial Flight Phase

The 3^d Sailcraft mission phase, Sailcraft Tack begins upon release of the Ballast mass. The ballast is sized and located so that when it is released, the Sailcraft will rotate to a 25° pitch angle. This provides a significant performance advantage, approximately 42% greater than a purely radial trajectory by aligning the force and velocity vectors. The total duration of this phase is one year, during which roll is passively stabilized and yaw is actively stabilized, both at 0°. The orbit trajectory for this phase is shown below, in Fig. 27.

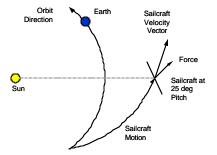


Figure 27. Sailcraft Tack Phase

The 4^h and final Sailcraft mission phase, the Sailcraft Cruise, begins when the payload is dropped (remaining attached to the Sailcraft by a tether), and is completed when the Sailcraft reaches escape velocity. When the payload is dropped, the Sailcraft is restored to a 0° pitch angle. With minimal power available, the yaw control system may still be operable, but is no longer necessary to maintain the desired trajectory. After reaching escape velocity, the Sailcraft will continue through the solar system. It is expected to reach Pluto's orbit after 16.9 years and reach the vicinity of the nearest star after 140,000 years. The orbit trajectory for this phase is shown below, in Fig. 28.

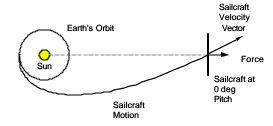


Figure 28. Sailcraft Cruise Phase

Sailcraft Trajectory Analysis

A Matlab/STK model was developed to validate Sailcraft performance and to plot its trajectory. It incorporates a gravity model with all planets in precise orbits (based on JPL data), a force model with absorbed, reflected and thermally reemitted photons, including specular bias and propulsive zenith effects and a diffuse surface model.

The Sailcraft geometry is modeled with unlimited facet capability (typically >15,000) to accurately describe its shape. It is represented below, in Fig. 29.

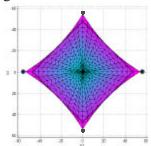


Figure 29. Sailcraft Geometry Model

The Mission Model is propagated in STK. Matlab sets the sail attitude and computes solar pressure force. The solar pressure force vector is sent to STK, where it is combined with the gravity force for orbit propagation. This refined model has been correlated to prior results, and showed excellent consistency.

Analysis has shown that the initial flight is sensitive to the Earth Escape maneuver timing. We found that it is best to wait at least 20 days after launch to initiate the Sailcraft mission in order to avoid reentering the Earth-Moon system. The maximum wait time is generally limited to 90 days, based on radiation susceptibility of electronic components. The optimum wait time may also be driven by Sailcraft visibility from earth. Visibility of the spacecraft from earth is highly dependant on the Sun-sail-Earth angle, Sailcraft pitch angle, distance from Earth, and the surface reflectivity. Results of Sailcraft visibility analysis are presented below, in Table IV.

Table IV. Visibility Analysis Results

Solid Motor Burn (days after launch)	Visibility from Earth, (Magnitude days after release)			
	0 – 7 days†	7 – 14 days ^{††}	14 – 21 days††	21 – 28 days ^{††}
<30 day	None	None	None	None
30-60 day	None	None	+10 to +11	+11 to +12
60-90 day	None	None	+10 to +11	+11 to +12
90-120 day	+2 to +8	+7 to +9	+9 to +11	None

† Assumes Pitch angle of 0 ±5 deg and Earth within a ±20 deg reflection cone †† Assumes Pitch angle of 25 ±5 deg and Earth within a ±20 deg reflection cone ††† This assumes an equivalent Aluminum absorber thickness of 14mm

Sailcraft Power Budget

The on-board powered devices, namely a yaw sensor and actuators (described in a later section) require a total of approximately 8.8 W, accounting for losses. Power generation is afforded by the sail's solar panels. The panels will utilize Copper Indium Gallium Diselenide (CIS) technology. The CIS material will be deposited on polyimide, and will be attached at four locations to the central core of the Sailcraft structure, as shown in Fig. 25. While the CIS panels are only 6.9% efficient, they are extremely efficient on a mass basis at >> 500W/kg, when deposited on thin film. Fig. 30 indicates the power capability and margins.

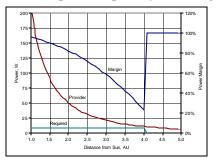


Figure 30. Sailcraft Power vs. Solar Distance

The power available degrades dramatically from a beginning of life (BOL) capability of 210 Watts down to about 13 W after 1 year (at 4.0 AU) – at which time the power demand drops to zero.

Mass Budget and Margin Metric

The Sailcraft mission is extremely mass sensitive. This has dictated a strict mass control program, which is tracked very carefully over the design cycle. Every design change is

rigorously analyzed for potential of mass impact. The total mass estimate for the Sailcraft at PDR is 20.1 kg. The mass breakdown is shown in Fig. 31.

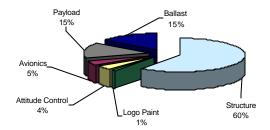


Figure 31. Sailcraft Mass Breakdown

At the start of the Sailcraft tack phase, after the ballast mass is dropped, the Sailcraft mass is reduced by 3 kg to 17.1 kg.

The Sailcraft performance is determined using a calculation where margin against the total mass above which the Sailcraft is able to reach escape velocity after 5 years is tracked as a metric. All system inefficiencies are converted into mass penalties for the purpose of this performance tracking. A reference minimum positive margin of 500 g is set as the design goal.

The sources of error that contribute to mass penalties include uncertainties in solar system mass, propulsive reflectivity losses, pitch, roll, and yaw angle errors, and main sail area losses. The total estimated performance loss at PDR is 510 g

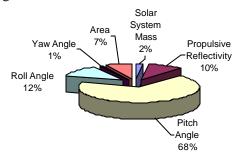


Figure 32. Sources of Error

We can see from Fig. 32 that by far, pitch errors are the dominant contributor to performance losses. The performance tracking metric, is shown in Fig. 33, where at the PDR design level a design margin of 577 grams was available.

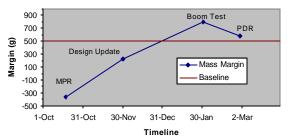


Figure 33. Sailcraft Performance Metric

Sailcraft ADCS Subsystem Design

The ADCS design provides for a passively controlled system about the pitch and yaw axes, and is actively stabilized about the yaw axis. It must provide for normal (0°) and inclined (25°) pitch modes, and return to a normal pitch mode for the initial, tack, and cruise Sailcraft mission phases, respectively.

Two pitch tabs are located on opposing boom tips. They are static (cannot be rotated), and are inclined for stabilization. The roll/yaw tabs are mounted on opposing boom tips. They rotate for yaw control, and provide roll control passively in any orientation. The tab locations are shown in Fig. 34.

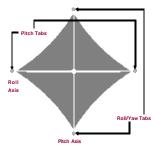


Figure 34. Sailcraft Control Tabs

It is possible to control pitch and roll passively, because a pitch or roll disturbance will result in asymmetrical solar pressure loading on the tabs, that will result in a restoring moment, as shown in Fig. 35.

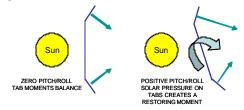
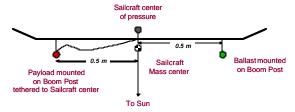
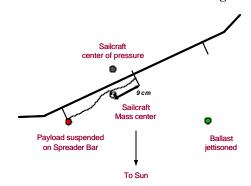


Figure 35. Passive P/R Control

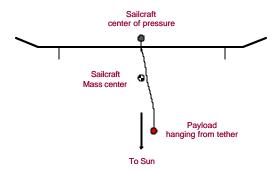
The pitch bias of the spacecraft – from 0°, 25°, and back to 0° is maintained by the relative position of ballast and suspended payload masses. This is shown in Fig. 36, below.



Pitch Control - Initial Sailcraft Flight



Pitch Control - Sailcraft Tack Phase



Pitch Control – Sailcraft Cruise Phase Figure 36. Sailcraft Pitch Control

The yaw angle of the Sailcraft cannot be passively stabilized, as no reference or bias is provided by sunlight in the yaw direction. An active stabilization system is employed, whereby a yaw sensor (very simple star tracker) measures the orientation relative to a fixed star field. If the yaw sensor detects an error above the acceptable threshold (3°), it will command the actuators to one of three positions (neutral, +30°, and -30°). The two yaw tabs counter-rotate, except in the neutral position. Both the yaw sensor and the

yaw actuator are new developments for the Encounter program.

A mission simulation of the yaw control system was performed, and the results of that are plotted in Fig. 37. The requirement – to maintain the yaw angle within $\pm 3^{\circ}$ – is easily met.

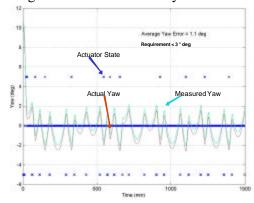


Figure 37. Sailcraft Pitch Control

Sailcraft Avionics Design

The yaw sensor described previously is part of the avionics package on the Sailcraft. Additional components include power converters, an EEPROM storage device, an IR communications link, and release systems for the ballast and payload. The power converters directly draw power from the solar arrays in order to sustain the actuator and yaw sensor requirements. The IR link is used to communicate data across the Carrier/Sailcraft interface prior to the separation event. This accommodates any "last-minute" customer data in the form of text and images that can be stored on the EEPROM.

The payload and ballast separation systems are identical in design, and utilize redundant voting reed relay separation switches to trigger a timer. When the ballast timer elapses, the ballast is released from the Sailcraft, and correspondingly when the payload timer expires, it is released from (but still tethered to) the Sailcraft in preparation for the final cruise phase. This design is shown in Fig. 38.

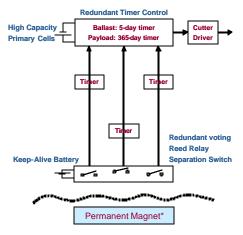


Figure 38. Ballast/Payload Release Electronics

Launch Vehicle Integration

The primary requirements imposed by Arianespace for use of the ASAP5 auxiliary configuration are volumetric and mass constraints. The maximum payload cross section cannot exceed 600 mm x 600 mm, and its height may not exceed 710 mm. The maximum payload mass (without separation system) may not exceed 120 kg. The PDR baseline design is compliant with all of these requirements.

As few as two and a maximum of eight payloads may populate an ASAP ring. The ASAP payloads must be installed in pairs for center of gravity balancing purposes. Fig. 39 shows the populated ASAP-5 platform mounted in the Ariane 5 SYLDA. The primary payload (not shown) would be mounted on its adapter platform in the center of the SYLDA.

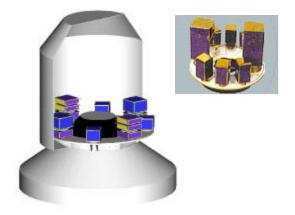


Figure 39. ASAP-5 Platform / SYLDA

Encounter will utilize the standard Ariane 5 flight proven separation system. After separation, approximately 1 kg of residual mass remains on the Carrier, but this is not counted against the ASAP mass allocation. The ASAP5 separation system provides for a 5° half-angle separation cone, has an adjustable separation velocity (1-3 m/s), and provides a max. tip-off rate of 3°/s. The separation system is triggered by the launch vehicle.

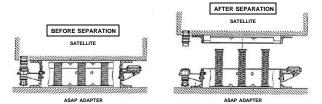


Figure 40. ASAP-5 Separation System (Courtesy Arianespace)

Assembly, Test and Launch Operations

The assembly, integration, and test (AI&T) flow includes activities that take place at four locations. The Carrier subsystems and elements will be fully integrated at AeroAstro. The Sailcraft will be fully integrated and tested at L'Garde, including the avionics subsystem provided by AeroAstro. The final Encounter Spacecraft AI&T activities, where the Carrier and Sailcraft elements are mated will take place at Team Encounter's facility in Houston. Lastly, the hazardous AI&T activities, namely installation of the Star 12G solid motor and ordnance devices and pressurization of the cold gas storage system, will take place at CSG's facilities in Kourou, French Guiana.

A preliminary test plan has been prepared, that identifies subsystem and spacecraft level test activities. In this plan, environmental tests take place only at the spacecraft level, not at the subsystem level. However standard acceptance tests are also performed at the component level that may include environmental tests. The preliminary test plan is shown below in Table V.

Table V. Preliminary Test Plan

Functional Tests	Spacecraft Level Tests
GSE Functional Test	Spacecraft anechoic chamber RF pattern test
Lifting/Handling Load Test and certification	ADCS Hardware-in-the-Loop Test
ADCS Component Testing	Carrier Functional Test (predefined, tests a ll subsystems together)
C&DH Functional Test	Mass Properties measure/Spin Balance
C&DH + Software Functional Test	Team Encounter Environmental Tests
Cold Gas Functional Test (Leak Test)	Sine Sweep/Natural Frequency Measure
Battery charge/discharge test	Random Vibration
Imager Functional Test	Separation Shock
Power subsystem Test	
Structural Mode "Tap" Test	
Comms. Functional Test	
"Foilsat" RF pattern test	
Internal Spacecraft EMC test	
GSE Functional Test	

Ground Segment Design

Space/Ground Interface

The notional baseline global tracking network for Encounter will be the Deep Space Network's (DSN) 26-m antennas located in Goldstone, Canberra, and Madrid. A representative antenna is shown in Fig. 41. The 70-m antenna is also being considered for use, and a back-up service with 13-m antennas may be available.



Figure 41. 26-m DSN Antenna

With the 26-m antennas, an S-band data and telemetry downlink rate of 238 kbps will be possible, far exceeding the 64 kpbs requirement. The DSN network can perform PRN ranging, to provide range, rate, and angle information. The S-band command up-link rate will be fixed at 2 kbps.

Use of DSN requires uplink in the S-band frequency range between 2025 - 2110 MHz and downlink between 2200 - 2290 MHz. The frequency assignment process requires coordination between JPL and NASA spectrum managers, and application through the National Telecommunications and Information Administration (NTIA) 4 stage process. Team

Encounter is working with these organizations to expedite the process, which normally takes 3 to 4 years.

All missions utilizing DSN must also use the Consultative Committee for Space Data Systems (CCSDS) telemetry and commanding communications protocols. The CCSDS packet telemetry format generally requires about 1% increased overhead compared to Time Division Multiplexing (TDM) telemetry, but this can be easily accommodated within Encounter's bandwidth capability.

As was shown in Fig. 4, a TCP/IP link will be maintained between DSN and the Team Encounter Mission Operations Center (TEMOC). TEMOC will provide work-stations capable of displaying and trending real-time and logged telemetry during the course of the Carrier mission phases. Each station will be equipped with a voice communications link to other stations, to DSN, and to the launch site (during mission rehearsals and count-down). Passive telemetry viewing (no commanding) will also be provided to the world wide web, and will be available to members of the Team Encounter community. A schematic of the TEMOC layout is shown in Fig. 42.

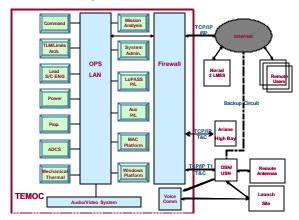


Figure 42. TEMOC Block Diagram

Ground Control Software

A number of Ground Control Software (GCS) packages were evaluated by AeroAstro prior to PDR. The Integrated Test and Operations System (ITOS) was selected because it meets all

of Encounter's T&C requirements, has demonstrated heritage with both the DSN and USN ground control networks, and provided for the lowest risk and cost solution. While ITOS is a NASA product, commercial support is available in order to customize for mission specific requirements.

In addition to ITOS, Satellite Tool Kit (STK) will also be used for mission analysis and maneuver scheduling in the operations center. A block diagram of the ground control software is shown in Fig. 43.

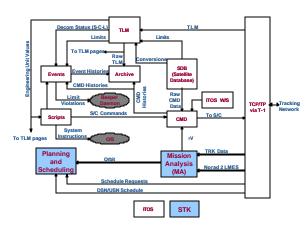


Figure 43. GCS Block Diagram

Conclusions

The preliminary design team, Team Encounter, AeroAstro, and L'Garde, have completed initial development of the interstellar Encounter spacecraft, through the preliminary design phase, satisfying all major mission requirements. A precursor mission, designated Flight One, is now being defined to demonstrate critical Sailcraft deployment and control technologies prior to the interstellar mission, now called Flight Two. The Flight One technology demonstration is meant to reduce program risk for the follow-on mission.

In the next phase, the Team Encounter PDR design will be adapted for the Flight One mission, and continued forward to critical design, manufacture, integration and test of the Carrier and Sailcraft elements with a complete spacecraft ready for launch as an Ariane 5 secondary payload in the summer of 2004.

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- Rogan, J, Gloyer, Paul, et al. "Encounter 2001: Sailing to the Stars.", 15th Annual USU Conference on Small Satellites, Paper SSC0I-II2, Logan, Utah, August 2001.
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