Space Operations, Inc.

Proposal response to Announcement No. NASA-CCiCap
Commercial Crew Integrated Capability

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1 Executive Summary

Space Operations, Inc (SpaceOps) is pleased to provide this proposal for a fast-track solution to the U.S. capability for a safe, reliable, and cost effective solution for a Commercial Transportation System (CTS) to Low-Earth Orbit (LEO). Although not particularly novel, our solution provides a means to stand upon the shoulders of our predecessors and 50 years of research, development, and operation conducted by our Space Agency. This program is as aggressive as any that have been attempted by our nation since the 1950's. With the goal of arriving at a certifiable orbital crewed flight demonstration within twenty-two months will require either one of two options. The first being the final push of an established system that has enjoyed the initial commercial crew initiative, and the second being an approach where a collection of companies come together to re-establish a system that has already been proven. SpaceOps, along with a group of companies each specializing in a particular aspect of the program, offer our Eclipse Program proposal.

Space Operations, Inc. is small business owned and operated in the Huntsville, Alabama area with no current or planned foreign interest. SpaceOps was established in 2011 to bring the concept of combining legacy spacecraft designs with currently available and proven technologies together which will satisfy the eminent demand for a reliable, affordable, and high quality spacecraft in the shortest possible time frame.

1.1 Concept

SpaceOps proposes to use the proven flight rated technology developed on the Gemini program to form the basis for the spacecraft configuration and operation. The Gemini program's mission, goals, and allotted timeframe are very similar to those of the Commercial Crew Integrated Capability (CCiCap) program that most of the proposed development can be taken directly from the NASA's Gemini Project history files. SpaceOps will incorporate state-of-the-market materials, electronics, avionics, Guidance, Navigation and Control (GNC), Life Support, Reaction Control, and emergency systems to take advantage of the advances in technology that have taken place over the past 50 years. These systems, subsystems, components, and materials are available and have been demonstrated to exhibit increased reliability, durability, and manufacturability over the legacy Gemini configuration.

Although the challenges faced by those original designers involved systems that had never existed prior to the Gemini program, our challenge will be to integrate that system to the extent possible and reasonable with current technology while ensuring System Safety and Mission Assurance (SSMA) policies have all been addressed.

SpaceOps' overall concept will match the goals and objectives set forth in the announcement; however, we will use a stepwise approach to first, establish the same basic capability and, second to extend that capability to the full crew requirements. If selected, SpaceOps with the assistance of NASA centers of excellence would enter the basic period with a significant engineering challenge to layout two configurations, identify all systems necessary for integration, review all design delta's that are shown to be necessary, all test plans, and a fully evaluated and accepted option period test and acceptance plan within six months. This can be achieved because we have located and accessed all recorded Gemini configuration and test records and have access to an existing CAD model of the actual flight hardware. At that point, the first major milestone will be achieved where the CCiCap program's option could be effectively evaluated for execution. If

the option period is elected, then we will begin integration development, and SSMA analysis, long-lead hardware procurement, test article fabrication, and conduct of ground and range testing. This program will culminate in first-flight hardware fabrication and an unmanned demonstration flight.

This parallel approach allows hardware integrators to move forward while mission assurance activities are allowed to become solidified and woven into the manned configuration.

1.1.1 Configuration A (ConA) – Two Crew and Cargo Delivery System (Gemini)

Using the Gemini spacecraft configuration along with the necessary transition sections to allow mating to [redacted] two-stage booster allows for a sensible arrangement that will be able to carry two astronauts and maximum cargo payload potential (up to 10,000 lbf) into LEO. Figure 1.1-1 depicts the basic notion of our CTS concept.

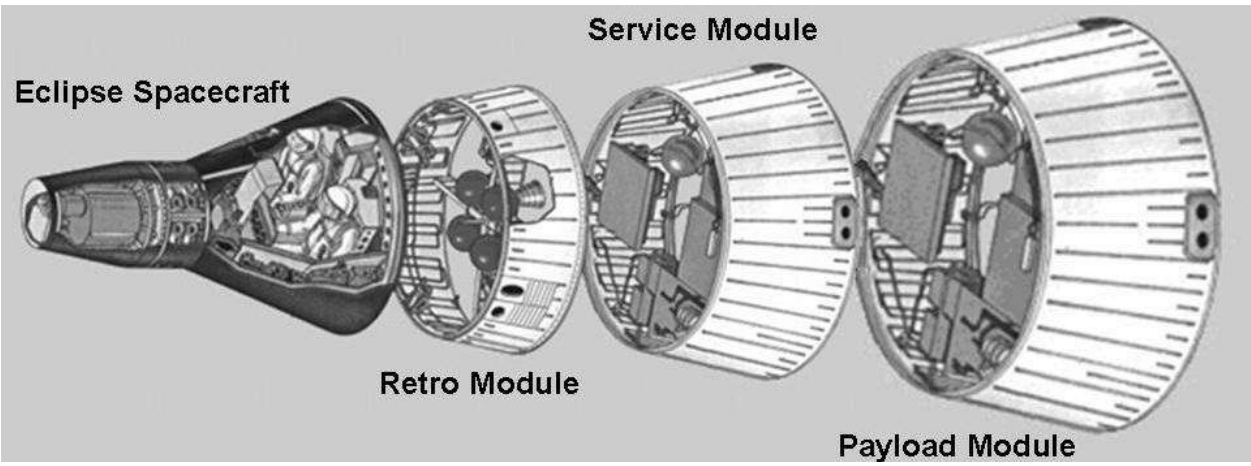

Figure 1.1.1-1 Eclipse ConA Baseline Configuration

Since Gemini successfully launched thirteen times (10 manned flights), the baseline structural configuration is man-rated and proven. In order to minimize development time, the Eclipse program will require strict adherence to the original Gemini configuration that is available in the NASA archives. Adopting this configuration and carefully applying currently available technology such as, avionics, GNC, improved materials (e.g., 4-D composites and newer metallic alloys), and making use of advanced systems such as TDRSS, Global Positioning System (GPS), Reaction Control Systems (RCS), and advanced crew safety and life support systems provides a significant 'jump start' to CTS development. This will allow the main program to be focused on risk management through rigorous testing and adoption of lessons learned in subsequent NASA programs.

The commercial application of this configuration will be for scientific payload deployment, International Space Station (ISS) supply, satellite delivery and/or maintenance missions.

1.1.2 Configuration B (ConB) – Four Crew Delivery System

SpaceOps' solution to handle larger personnel capacity for crew transfer missions will be accomplished by extending the cabin length to accommodate two additional astronauts along with the additional life support elements. [redacted] will also be used to launch the Eclipse ConB into LEO; however, payloads will be limited ISS re-supply due to the mission assurance equipment requirements.

The commercial application for this configuration is simply to provide ISS crew transportation.

1.2 Program Overview

The Eclipse program must be a team effort and would naturally require support from NASA centers of excellence. The smaller basic size of the Eclipse should provide the following attributes:

- Low risk to accomplish program goals due to starting with a proven design
- Reduced cost due to the compact size and minimized development/design
- Greater confidence in SSMA due to reduced overall mass
- Reduced impact to the ISS orbit due to lower overall mass.

1.3 Top Level Schedule

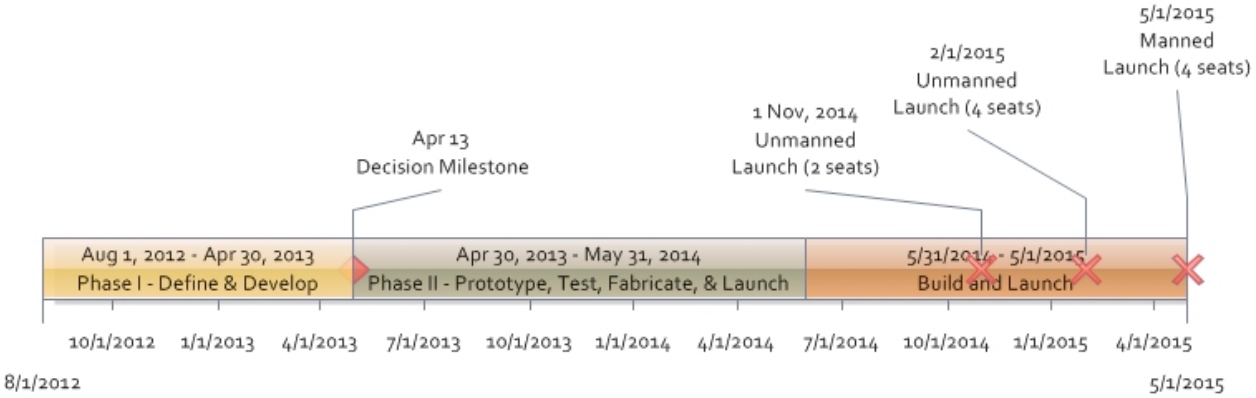

Figure 1.3-1 Top Level Eclipse Program Schedule

1.4 Commercial Operations

The goal of the CCiCap program is to provide a reliable, cost effective method for U.S. agencies and commercial entities to be able to reach LEO with personnel and/or interesting payloads. In the wake of the Shuttle Program closure, scientific endeavors for microgravity experimentation have become costly and problematic. Current vehicles for microgravity study are limited to very little time exposure in the sounding rocket program or the incurred expense of an ISS mission due to the extensive planning or heavy boost operations. We hope to develop a middle ground for scientific research institutions and principle investigators. With the two-man lift, we believe that slightly higher orbits will be attainable and that satellite maintenance or repair similar to the operations performed on Hubble may be of interest to satellite owners.

1.4.1 Transition from NASA Facilities

In order to bring this concept to fruition, SpaceOps will need to acquire its own integration, fabrication, testing, and reclamation facilities so that dependence on NASA is reduced to those services that only NASA can provide. Our goal will be to obtain or reclaim these facilities at or near current spaceport locations where the possibility of obtaining experienced staff is of the greatest potential.

1.4.2 Capsule

The initial fabrication and testing of the Eclipse spacecraft will require the use of specialized NASA facilities and expertise. However, once developed and certified, the production of new

vehicles and the refurbishment of returned vehicles is planned to become a commercial endeavor handled by SpaceOps. Although optional sites can be found, final assembly, inspection, and calibration are hoped to be performed by our subcontractor [redacted] through a usage agreement at the Michoud facility near New Orleans.

1.4.3 Booster

[redacted] has made enormous strides over the past five years and has developed a dependable booster that is necessary for our program to place Eclipse into orbit. This program does not compete with [redacted] current development of the [redacted] spacecraft because their goal is to provide a much larger crew into orbit. If we are able to provide an alternative method for smaller crews and for a different size payload, then the Eclipse vehicle represents a viable alternative that increases their business potential for the [redacted].

[redacted] has agreed with SpaceOps to provide boosters and launch support, as required.

1.4.4 Re-Use

The Eclipse vehicle is planned to be refurbished after each mission. Re-use of the vehicle provides a method to reduce the overall cost of the program and will allow faster turnaround for quick reaction launch requirements. Again, this activity will be to be performed by a SpaceOps subcontractor, [redacted], and at the Michoud facility.

1.5 The Team

SpaceOps has compiled a team of highly motivated companies that share the vision of having a U.S. launch capability for manned spaceflight. The three main technical contributors to this re-engineering effort are, [redacted], [redacted], and [redacted]. Clearly, none of this would be possible without the efforts that have been made by the [redacted] on the new capability in two-stage booster technology. All of these companies regularly do work for the U.S. Government and have compliant accounting systems.

[redacted].

[redacted]

[redacted]

[redacted]

[redacted]

2 Technical Approach

The development of the Eclipse system, although simplified in construct, will require considerable coordination among a host of well qualified support elements. In order to minimize program risk, SpaceOps is taking a teamed approach where companies are called upon to provide very specialized products and services that are necessary for timely and professional execution. In addition, SpaceOps is working to enter into one or more Space Act Agreements (SAA) with MSFC, GSC, KSC, and JSC to provide specialized services that cannot be found in commercial industry.

The SpaceOps Commercial Crew Integrated Capability (CCiCap) concept will springboard off the successes of the NASA Gemini program and the technological advances and improvements made over the last 50 years to provide a technically advanced and economical transport to Low Earth Orbit (LEO); this capsule is named the Eclipse. This approach will facilitate a low risk development effort resulting in a two seat unmanned first flight within twenty seven (27) months after contract award, an unmanned four seat Eclipse flight three months later with a four seat manned flight three months after that. Although the crew size for the basic concept is limited to two individuals, parallel development efforts to include two additional crewmember accommodations aft of the existing positions will offer passage for four personnel. Unmanned launches of each configuration prior to a manned flight will reduce risk. The insertion of current technology to replace larger heavier electronic components will also aid in the human factors and provide increased space and comfort, and additional payload capability in the future.

On Dec. 4, 1966 Gemini 7 demonstrated the capability to maintain orbit for almost 14 days and performed a docking maneuver with Gemini 6A, depicted in Figure 2.0-1. Clearly, International Space Station (ISS) docking is beyond the scope of the demonstration flights; however, once the basic capability to approach and release control to the ISS is successfully demonstrated the docking mechanisms that have already been designed by NASA will be modified for interface compatibility resulting in a capability to transfer crew and cargo to the ISS.

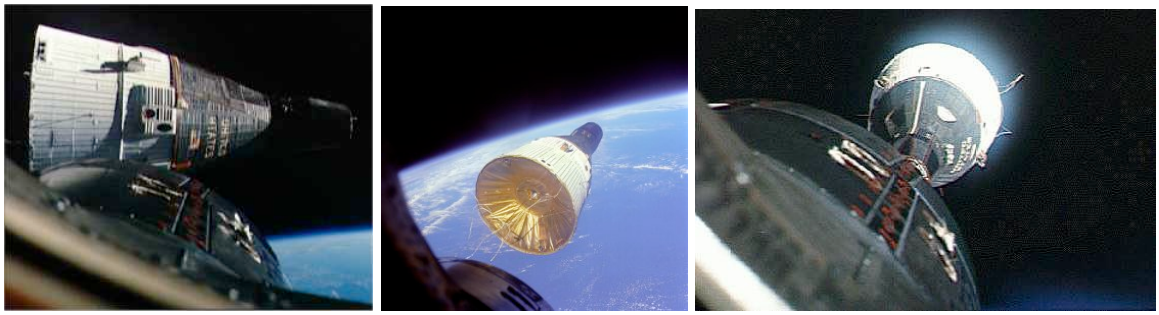

Figure 2.0-1 Gemini 7 Performing Docking Maneuvers with Gemini 6A

Unlike the Gemini the Eclipse reentry and recovery system is designed to be land based making the Eclipse reusable. The heat shield will be updated to take advantage of current development, a parasail or paraglide landing system which was initially developed and tested by NASA, will be completed and incorporated. This updated reentry system will not eliminate the capability of a water recovery in the event of an emergency situation.

In order to ensure program success SpaceOps has assembled a team of highly motivated companies that bring a wide range of expertise to ensure mission success ranging from a proven launch vehicle, [redacted], to spacecraft design updates, to spacecraft and launch vehicle integration, launch operations, mission operations and recovery. The members include: [redacted], [redacted], [redacted], and [redacted]. [redacted] will provide the launch vehicle, Eclipse to [redacted] integration processing support and facility, and lead the launch operations. [redacted] will provide the manufacturing of Eclipse structural components, electrical harnessing, procurement of components and materials, and refurbishment of the Eclipse vehicles. [redacted] will provide engineering and analysis support as needed with [redacted] providing the expertise in orbital mechanics, mission planning and operations, and engineering support in guidance, navigation and control functions.

A "build-a-little test-a-little" approach has been defined to mitigate development risks and ensure program success. As illustrated in the notional schedule 2.0-1 below, the initial Eclipse spacecraft developed as an updated Gemini 2-seat vehicle will undergo the first launch. This will ensure that the updates made to the original design and the modifications to the launch vehicle interface meet all the aerodynamic, mass property and environmental performance requirements. This first launch will also demonstrate the launch operations, mission operations, and re-entry and recovery processes are complete and error free. This test launch is anticipated to be a one day flight consisting of data gathering; orbital maneuvering and systems prove-out.

In parallel to this effort a design team will be concentrating on updating the docking mechanisms and converting the crew Cabin Module to a four seat design which may have an impact on the life support systems, attitude and control system, and propulsion systems. The second unmanned launch will include a pallet of instrumentation at each seat location to collect data to confirm safe human environmental conditions throughout the launch, mission operations and recovery. This test launch will also demonstrate the aerodynamics and mass properties of the modifications and updates. It is anticipated that this test flight will be a three day flight that will include an ISS fly-by and a maneuver control hand-over to the ISS.

In parallel to this activity astronaut selection and training will be conducted. The third launch will be a four crew manned flight for a five day mission flight. It is anticipated that this test will include a docking with the ISS and demonstration of the docking mechanisms and evaluation of cargo and crew transfers between the Eclipse and ISS.

As illustrated in the notional schedule a preliminary design review (PDR), critical design review (CDR), integration readiness review (IRR), and test readiness review (TRR) is conducted on each Eclipse prior to shipping to SpaceX integration facilities. After shipment to Cape Canaveral the milestone reviews will be conducted IAW the launch schedule, delineated in Table 2.2.2.1-1.

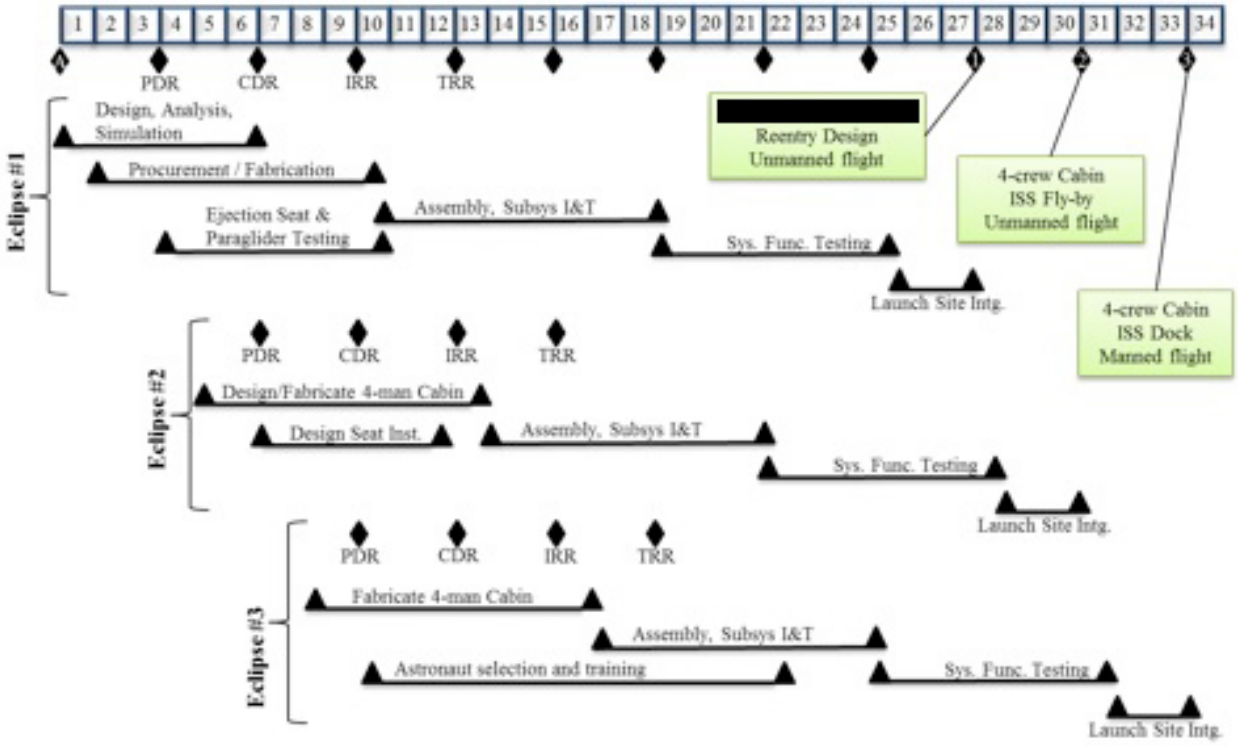

Figure 2.0-2 SpaceOps Eclipse Program Notional Schedule

2.1 Overall Concept and Performance

SpaceOps' Eclipse spacecraft will offer an economical and safe alternative for U.S. Government agencies, prime contractors, satellite manufacturers, and other organizations or individuals having a desire or requirement for space travel to and from LEO. Simply stated, the mission of the Eclipse will be to launch into a narrow range of low earth orbits that include ISS and lower satellite orbits. In order to accomplish this various simulations will be conducted to verify the correct launch parameters are met, station keeping and spacecraft maneuvers can be supported, and vibration and acoustic noise levels have been accounted for in the design. It will also be used to aid in defining the test program criteria, establish and validate emergency procedures, validate the guidance and control requirements, establish the telemetry data requirements and ultimately reduce program risk.

2.1.1 System Simulation

Comprehensive and rigorous proofs now exist that were not available 50 years ago for transforming a nonlinear system into an equivalent linear system; called either feedback linearization or Dynamic Inversion (DI). Synthesis of the component/elements of the many systems will be accomplished by the designer for the tasks under consideration. The task is to be sure that the individual parts are integrated in such a way that the sum of the parts produces an outcome greater than the individual contributions of the parts. This "synergy" is a result of an integrated design. Integration begins with the process used, and depends strongly on the tools available. **This simulation development process and its results are at the core of this phase of development**.

MATLAB and its companion product, Simulink, are used extensively in this study; it has a number of specialized toolkits that directly address matrix algebra and modern control system design, including DI and techniques often used to analyze the robustness of such designs.

MATLAB is one of the most widely used commercial software packages available for control system design. The DI controllers are developed in MATLAB, and simulations are run within the Simulink environment.

The selected space vehicle simulation development tool provides a range of capability, from a stand-alone model development platform to a completely integrated end-to-end customized simulator with hardware-in-the-loop. Off-the-shelf models and code generators are used for required components, such as:

- Orbital and attitude dynamics
- Environment (atmosphere, magnetosphere, solar)
- Electrical power generation and distribution
- Thermal solver (radiation, conduction)
- Guidance and navigation sensors
- Attitude control subsystem
- Communication subsystem
- GPS clocks, receivers, transmitters
- Ground control subsystem (STOL procedures)
- CCSDS telemetry/telecommand encoding

Object-Oriented Modeling - Pre-validated elementary objects are available in object libraries that permit a rapid and reliable build of complex models. The objects are generic and are customizable to the specific equipment characteristics incorporated in the design.

Software-in-the-Loop - Control system software (flight code) can be integrated in the simulator in source code format or as an external library allowing the real control system software to be tested in a fully closed loop, with models of the environment, dynamics, sensors and actuators.

Hardware-in-the-Loop - Actual equipment can be interfaced to the simulator via analog or digital interface objects that call the specific hardware device drivers. In this manner, a hardware component can be stimulated and tested in a fully closed loop, with models of the environment, dynamics, sensors and actuators.

Existing and New Models - Models generated by SpaceOps or provided from equipment vendors can easily be encapsulated and integrated in the simulator. We can integrate models written in FORTRAN, Ada, C or C++. Code generated from Matlab/Simulink™ can also be encapsulated in our simulation environment. This permits the integration of existing models, newly developed models and validated models; this maximizes the re-use of models developed in other phases of the space vehicle development cycle.

Life Cycle Support - The simulator can be used from the beginning to the end of a space vehicle program: System/subsystem requirements verification; system/subsystem design; system/subsystem integration and testing; ground segment validation; operations support; and training

The simulator can run in real-time for testing hardware-in-the-loop, but can also run faster than real-time in order to permit the execution of several days of simulation in a few minutes.

2.1.1.1 Launch

Launch Simulations with the [redacted] and the Eclipse vehicle will require many high fidelity simulations to assure structural integrity and mission success. The [redacted] launch environment is well defined and the data is available for inclusion into the simulations. The

Gemini spacecraft was launched on the Titan II (shown in Figure 2.1.1.1-1) which has demonstrated a more severe environment than the [redacted] Launch Vehicle.

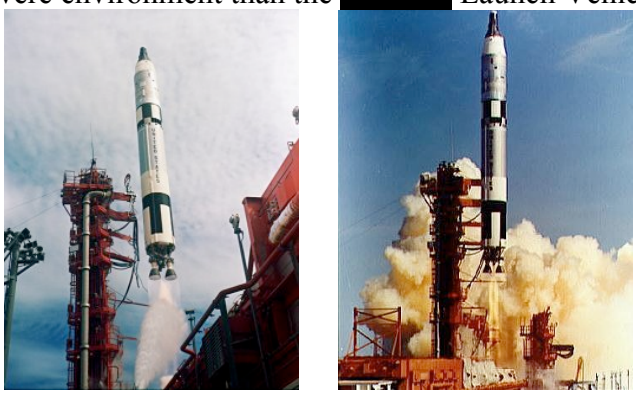

Figure 2.1.1.1-1 Gemini / Titan Launch

Launch Plume: High-fidelity modeling and simulation to provide information on the necessary mesh resolution requirements, physical models, boundary conditions, and turbulence models to accurately predict the quantities of interest. In the context of LV ascent, some of the quantities of interest include aerodynamic force and moment coefficients and load distributions, as well as base heating and surface pressure estimates. To provide acceptable predictions of these quantities, the CFD simulations must accurately model the physics of the exhaust plume as it expands with increasing altitude.

Rockets at high-altitude are subject to a fluid dynamics phenomenon known as Plume-Induced Flow Separation (PIFS). Flow separation occurs when an adverse pressure gradient forces the boundary layer to detach from the surface of the rocket. One cause of the adverse pressure gradient during ascent is the expansion of the exhaust plume as the rocket gains altitude. In a low ambient pressure environment, the high pressure at the nozzle exit rapidly expands the exhaust jet in both downstream and radial directions. This produces an obstruction to the free-stream flow which forms an adverse pressure gradient near the aft section of the rocket. Ultimately, the flow separates and recirculation from the base of the vehicle to the upstream separation point allows convective transport of hot exhaust gas along the surface of the vehicle. The distance between the end of the vehicle and the separation point of the surface is denoted as the PIFS distance. Accurate prediction of the PIFS distance is critical to the design of the thermal protection system. We will make use of Computational Fluid Dynamic (CFD) analysis and wind tunnel testing to minimize design change risks.

Delta-V: Change in velocity, or delta-v is the most important measure of "distance" in space flight. The total delta-v needed is a good starting point for early design decisions since consideration of the added complexities are deferred to later times in the design process. To get from the ground to low Earth orbit (LEO) requires a delta-v of about 8600 m/sec^2. It's the clearest measure of how hard a rocket has to work to change position.

Delta-v is typically provided by the thrust of a rocket engine, but can be created by other reaction engines. The time-rate of change of delta-v is the magnitude of the acceleration *caused by the engines*, i.e., the thrust per total vehicle mass. The actual acceleration vector would be found by adding thrust per mass on to the gravity vector and the vectors representing any other forces acting on the object. The rocket equation shows that the required amount of propellant dramatically increases, with increasing delta-v. Therefore the launch vehicle / spacecraft propulsion systems require study to meeting the total delta-v needed for a given spaceflight,

mission, plane changes, as well as designing spacecraft that are capable of producing a sufficient delta-v.

Increasing the Delta-v provided by a propulsion system can be achieved by: staging, increasing specific impulse, or improving propellant mass fraction.

Specific Impulse: Specific Impulse is the exhaust velocity divided by g, the acceleration due to gravity on Earth's surface (9.80665 m/s^2). It's a common measure of the "mass efficiency" of a rocket engine.

2.1.1.2 Orbital Mechanics

Launch Location: The location of the launch site is extremely important because it usually determines the range of possible orbital inclinations in which to insert a satellite or spacecraft. Most space payloads launched into orbit are considered direct launch satellites. Note that a direct launch from latitude of 28° will by definition have an inclination of at least 28° since the orbital plane must pass through the launch site and the center of the earth. Lower inclinations will require an on-orbit plane change or maneuver, which has significant fuel penalties. The current sites under consideration for the test launches are Cape Canaveral and possibly Kwajalein Atoll.

Launch Window: A launch window is defined as the period of time during which a satellite can be launched directly into a specific orbital plane from a specific launch site. If the orbital plane inclination is greater than the launch site latitude, the launch site will pass through the orbital plane twice a day, producing two launch windows per day. The direction to point is known as the launch azimuth, measured from the north clockwise.

Inclination: If the inclination of the orbital plane is equal to the launch site latitude, the launch site will be coincident with the orbital plane once a day, producing one launch window per day at a launch azimuth of 90º (due east). If the inclination is less than the launch site latitude, the launch site will not pass through, or be coincident with, the orbital plane at any time, so there will not be any launch windows for a direct launch. A simplified model for determining inclination (i) from launch site latitude (L) and launch azimuth (Az) is: $\cos(i) = \cos(L) \bullet \sin(Az)$. The launch azimuths allowed (in most countries) are limited due to the safety considerations that prohibit launching over populated areas or foreign airspace. This restriction further limits the possible inclinations from any launch site.

Launch Velocity: When a satellite or space vehicle is launched, a tremendous amount of energy is imparted to it. Such forces are necessary to overcome the gravitational force of the earth as discussed previously. To maintain a minimum circular orbit at an altitude of 90–100 miles, the satellite has to travel at about 17,500 mph. Due to the earth's rotation; more or less kinetic energy may need to be supplied, depending on the launch azimuth. The starting velocity at the launch site varies with latitude and can be determined by multiplying the cosine of the latitude by 1,037 mph. For example, at an altitude of 45° north latitude, the starting velocity would be determined in the following manner: $\cos(45) \times 1{,}037\text{mph} = 0.7071068 \times 1{,}037\text{mph} = 733.3\text{mph}$. A satellite or space vehicle launched from the equator in the same direction as the earth's rotation (due east) has an initial speed of 1,037 mph. Therefore, 16,463 mph must be supplied (17,500 mph – 1,037 mph = 16,463 mph) to launch a satellite into that particular orbit (90–100 mile altitude). If launched from the equator in a retrograde orbit (against the rotation of the earth), 18,537 mph must be supplied. Launching with the earth's rotation saves energy and

allows for larger payloads for any given booster. In addition, the above equations show substantial energy savings when locating launch sites close to the equator.

Orbit Maneuvers / Station Keeping: An orbital maneuver is a deliberate change in the size, shape, and/or orientation of a satellite's orbit. The reasons for conducting an orbital maneuver include (but are not limited to) the following: increasing the satellite's field of view, counteracting the effects of atmospheric drag or other perturbations, increasing imaging resolution, rendezvousing with another satellite, or deorbiting a satellite.As previously mentioned, a spacecraft's velocity and position determine its orbit. To change one of these requires the application of force, which then accelerates the vehicle, this acceleration produces an impulsive change in velocity which changes the size of the orbit by either adding or subtracting energy. For any single Δv orbital change, the desired orbit must intersect the current orbit, and the point of intersection is where the change is applied. Otherwise it will take at least two Δv's to achieve the final orbit, one to leave the current orbit and another to join the final desired orbit. The amount of Δv required can be determined by subtracting the present vector from the desired vector.

Mission Considerations: Mission planners must ensure that a satellite is provided with sufficient fuel to perform the planned maneuvers once in orbit. Additional fuel on board a satellite results in a heavier payload and may require a more powerful booster to place the satellite in orbit, so these maneuvers must be planned carefully. There are two types of orbital maneuvers: in plane and out of plane.

In-plane maneuvers are the most common type of orbital maneuvers performed since they require much less fuel and energy to perform. These maneuvers are conducted to change a satellite's period (size), argument of perigee, or true anomaly. The majority of in-plane maneuvers are performed to counter the external forces, or perturbations, that are constantly acting upon the satellite and changing its orbit.

Out-of-plane maneuvers result in a change in inclination or right ascension of the ascending node. This type of maneuver requires a much larger amount of fuel to generate the sufficient velocity vectors (Δv) to change the satellite's orbital plane. For example, a 28º plane change, necessary for a Kennedy Space Center–launched satellite to become equatorial, requires a Δv of about 3.5 km/s. This same Δv applied in-plane would be enough for the two burns needed to raise a LEO satellite to geostationary.

Some orbit maneuvers are done simply to maintain the given orbit in the light of perturbations; all satellites are subject to external forces acting upon a satellite that affect its otherwise constant orbital parameters. These forces have a variety of causes, origins, and effects. For instance, because of drag, the eccentricity of a satellite orbiting the earth can never truly equal zero. These forces are named and categorized in an attempt to model their effects. The major perturbations are:

- Earth's oblateness
- Atmospheric drag
- Third-body effects
- Solar wind/radiation pressure
- Electromagnetic drag

2.1.1.3 Re-Entry Flight Dynamics

A new reentry simulation tool, called the general simulator for atmospheric reentry dynamics (GESARED), was implemented in the MATLAB®/SIMULINK environment. The primary objective of this simulator is to be used as a design environment of GN&C systems for reentry vehicles. As a test bed, GESARED is also designed to facilitate performance evaluations against requirements in both nominal and off-nominal conditions, allowing the entire GN&C system development process to be condensed in one tool based on a compatible, versatile, and easy-to-use environment.

Entry Dynamics: The entry phase in Earth's atmosphere corresponds to an altitude range from 120 km to nearly zero. This phase is characterized by large variations of environmental conditions. Therefore, it is necessary to have realistic models of the environment. Because of large variations of atmospheric conditions and aerodynamic angles, the model of the vehicle's aerodynamics has major importance. The main objective is the design of control laws, and thus the models of the sensors and actuators should also be realistic, in order to obtain a reliable overall performance evaluation of the closed-loop system. The entry motion is modeled as of six degrees of freedom (DOF), where uncoupled translational and angular motions are considered. The translational motion represents the point-mass trajectory motion. The generic kinematics and dynamics equations that describe this motion are:

$$F_E = m\,\frac{d^2 r_{cm}}{dt^2}\,, \quad V = \frac{dr_{cm}}{dt}$$

in which F_E denotes external force, r_{cm} denotes the position vector, m is the vehicle mass, and V denotes the velocity vector in the inertial frame. The choice of the coordinate system is of major importance because it will influence the effectiveness of the simulation tool. The objective is to have a minimum number of constraints and maximum generality, while avoiding unduly the complexity of equations. Therefore velocity is expressed in Cartesian coordinates of north, east, and down. Position is expressed in spherical coordinates of latitude, longitude, and distance toward the Earth's center. The resulting equations of translational motion comprise three equations for velocity and three more for position, the external force vector components F_x, F_y, and F_z being expressed in the vertical frame. The only singularity of these equations of translational motion is when the latitude is $\pm\pi/2$. The consequence of this singularity is a constraint in the simulation tool of polar reentry trajectories.The forces acting on the vehicle are the aerodynamic forces, the gravity force and the thrust force, resulting in the total external force F_E. The aerodynamic and thrust forces are computed from both the vehicle and the environment models, given a specific flight condition. The gravity force is computed from the environment model, given the mass and the altitude of the vehicle. The angular motion represents the rigid-body attitude motion. This motion is described by the following equations:

$$M_E = \frac{dH}{dt}\,, \qquad \Omega = \frac{d\theta}{dt}$$

in which M_E denotes an external moment, H is the angular momentum, ω is the angular velocity vector, and θ is the attitude vector. The angular velocities are conveniently expressed in the vehicle's body-fixed reference frame, represented by the common notation of p, q, and r . The attitude position is chosen in order to avoid singularities. If the attitude angles of roll, pitch, and yaw are used, there will be a singularity whenever one particular angle is $\pm\pi/2$, depending on the order of rotation. This constraint might be unacceptable when dealing with planetary entries in

general, and therefore a four element quaternion is used instead. The external moments acting on the vehicle are the aerodynamic moments and the moments resulting from the activation of the thrusters. The resulting equations for angular motion comprise three equations for angular rates and four for the attitude quaternion, where the external moment (vector M_x, M_y, M_z) is expressed in the body-fixed reference frame.

2.1.2 Mission Operations

A SpaceX Mission Manager will facilitate the interface between SpaceOps and [redacted] technical execution staff and all associated licensing agencies in order to achieve a successful mission using the [redacted] launch vehicle. The [redacted] Mission Manager is responsible for coordinating mission integration analysis and documentation deliverables, planning integration meetings and reports, and coordinating all integration and test activities associated with the mission. The Mission Manager will also facilitate SpaceOps insight during the launch campaign. The Mission Manager will coordinate all launch site activities with SpaceOps to ensure program satisfaction during this critical phase.

SpaceOps will deliver a fully assembled Eclipse spacecraft to the [redacted] integration facility for integration onto the [redacted] where it will be moved to the assigned launch pad for pre-launch testing and validation. Two concepts are currently in evaluation to determine the most cost effective method of integrating the Eclipse with the [redacted] Launch Vehicle. The first being a horizontal integration approach were the Eclipse and the launch vehicle are mated on its side and then raised at the pad. The second approach is to raise the launch vehicle at the pad and then perform a vertical integration of the Eclipse to the launch vehicle as performed with the Gemini, illustrated in Figure 2.1.2-1.

Figure2.1.2-1 Vertical Integration

SpaceOps will apply for authorization to use the NASA Tracking and Data Relay Satellite System (TDRSS) network and the Spaceflight Tracking and Data Network (STDN) from NASA Goddard with [redacted] support.

The SpaceOps team will establish the requirements and support from the Mission Operations Center (MOC) which is the focal point of spacecraft operations. The mission operations team working with NASA personnel will schedule requests for support, monitor spacecraft performance and upload control information to the spacecraft (through TDRSS). The SpaceOps MOC support team will consist of principal investigators, mission planners and flight operators. Mission planners provide documentation for the spacecraft and its mission, with flight operators as the final link, sending commands to the spacecraft and performing the operations.

The Gemini program would discard the Service Module (equipment section) at approximately 150 nautical miles altitude; this space debris would eventually reenter earth's atmosphere and burn-up in reentry. In order to remove mass from densely populated orbits, it is recommended by the international Space Congress, the Inter-Agency Debris Coordination Committee (IADC)and ISO that spacecraft and orbital stages are commanded to re-enter the Earth's atmosphere within 25 years of mission completion, if their deployment orbit altitude is below 2000 km (i.e. in the LEO region). Nevertheless SpaceOps does not wish to contribute to the growing space debris problem therefore the service module will undergo a modification to

ensure a controlled reentry over unpopulated regions after being discarded by the Eclipse vehicle. This modification will be demonstrated on the first test launch.

Although docking with the ISS is not a requirement of this effort it is anticipated that the Eclipse will need to provide the capability to dock with various spacecraft. Designers of future spacecraft that need to dock with each other for crew and cargo transfers are likely to use a new set of interface standards just published by the International Space Station partners in the hope they will simplify human exploration beyond low Earth orbit.

The interface standards do not cover the actual technology that would operate a docking system. Instead, they give measurements and force loads for engineers to match as they design their own docking system. For example, the diameter of the opening in the mating plane listed in the documents posted at http://www.internationaldockingstandard.com/ is 1,045 mm., but it is up to individual designers to design the mechanisms that will use that opening to guide and lock the two halves together. The mechanical Androgynous Peripheral Assembly System (APAS) has been modified with elements of the electromechanical Low-Impact Docking System (LIDS) under development at JSC to create a standard for systems that work like APAS but require less force to drive the two spacecraft together.

The European Space Agency is at work on a similar system for its proposed Advanced Reentry Vehicle (ARV), a spin-off of its Automated Transfer Vehicle (ATV) that would be able to return cargo from the ISS. Known as the International Berthing Docking Mechanism, it would also be made compatible with the new standard and would allow the ARV to dock at the U.S. end of the ISS. **SpaceOps will comply with this standard once fully defined; in the meantime the Gemini docking mechanism will be modified to mate with the ISS on the third test flight**.

After the Eclipse has landed a recovery team with emergency vehicles (as a precaution) will be on site to assist the crew from the Eclipse, remove and stow the parasail/paraglider, stow the vehicle on a truck for transport and perform post flight evaluation and refurbishment.

2.1.2.1 Re-Use

The modifications that will be incorporated to the Cabin module to accommodate a crew of four will be accomplished by maintaining the same exterior angles, and geometrically enlarging the capsule. This approach will allow a lengthening of the Cabin Module by approximately seven feet resulting in a 144” mating interface to the launch vehicle at approximately the center of the Service Module. This additional seven feet of volume is anticipated to provide sufficient space to relocate the equipment mounted in the Service Module and thus eliminate space debris concerns and maximize the re-use of avionics and propulsion components, thus reducing launch costs.

Additionally implementing a land recovery system versus a water recovery (discussed in paragraph 2.2.1.1.4) will eliminate the concerns of water and salt damage to the Cabin Module and the equipment mounted between the inner and outer shell of the capsule.

The full impact of the reentry heating that the Eclipse will undergo is anticipated to be very similar to the effects documented during the Gemini program. When this is confirmed after the Eclipse first flight SpaceOps anticipates that the capsule may be refurbished and reused. The full refurbishment effort is expected not to exceed the reapplication of the heat shield, re-servicing of the system consumables, and general cleaning efforts. This capability will significantly reduce launch and mission costs.

2.2 Eclipse Development and Demonstration

The development of this system, although simplified in construct, will require considerable coordination among a host of well qualified support elements. In order to minimize program risk, SpaceOps is taking a teamed approach where companies are called upon to provide very specialized products and services that are necessary for timely and professional execution, discussed in paragraph 3.1.2). In addition, SpaceOps is working to enter into one or more Space Act Agreements with MSFC, KSC, GSC, and JSC to provide specialized services that cannot be found in commercial industry.

The SpaceOps Eclipse spacecraft is composed of five major components, including a rendezvous and recovery section, re-entry control section, cabin section, retrograde section, and an equipment section or service module, depicted in Figures 2.2-1 through 2.2-7.

Figure 2.2-1 Eclipse Spacecraft

Figure 2.2-2 Rendezvous and Recovery Module

Rendezvous and Recovery Module - The rendezvous and recovery (R&R) section includes a rendezvous radar, a drogue parachute, and a main parachute. A protective nose fairing made of fiberglass reinforced plastic laminate mounts to the forward end of the R&R section, a horizon sensor fairing is attached to the left side mating point of the Cabin section. The fiberglass nose fairing protects the rendezvous radar during the initial launch phase and is jettisoned about 45 seconds after ignition of the second stage main engine. The Rendezvous Radar System consists of the rendezvous radar range and range rate indicator, and command link encoder and allows the Eclipse Spacecraft to rendezvous with a target vehicle. The guidance system is composed of the Rendezvous Radar System, the Digital Command System, the Time Reference System, and the Inertial Guidance System. The Inertial Measurement Unit provides a stable reference system for use in guidance and control tasks and also provides inertial attitude and acceleration information. The Digital Command System (DCS) consists of one receiver/ decoder package and three relay packages. The DCS receives, decodes, and transfers digital commands transmitted from ground stations. The Time Reference System is the central timing system of the spacecraft.

The rest of the section is covered with beryllium shingles; the beryllium shingles will be evaluated and possibly replaced with titanium shingles.

During re-entry, the rendezvous and recovery section is pyrotechnically separated from the spacecraft. A small pilot parachute pulls the section away, while also deploying the main parachute. Other methods of re-entry provide a land based recovery, these approaches will be evaluated for incorporation into the Eclipse design (refer para. 2.2.1.1.4).

Figure 2.2-3 Re-entry Control System Module

Re-entry Control System (RCS) Module - The RCS section contains fuel and oxidizer tanks, along with 16 attitude control thrusters and consists of two fully redundant propulsion systems, each with eight 25 pound-force (lbf) thruster assemblies operating on storable hypergolic propellants. The legacy RCS section has an internal titanium frame, with eight beryllium panels forming its outer skin; the beryllium panels will be evaluated and possibly replaced with titanium panels. Titanium is more cost effective, more easily procured, and although not as light it does have a slightly higher melting point.

The forward end of the RCS section contains attachment points for the main parachute. This section will be re-evaluated to determine efficiencies, reliability, and the existence of improved performance, size and weight.

The RCS may require modification to ensure re-entry responsiveness and performance as the crew module increases to accommodate a four man crew. The Reaction Control System provides attitude control of the Eclipse from retrograde to deployment of the parachute/paraglide/parasail, subsequent to jettisoning of the Orbit and Attitude Maneuver System.

Figure 2.2-4 Cabin Module

Cabin Module - The cabin section outer shell is shaped like a truncated cone. A smaller, irregular shaped, pressure vessel is located within the outer shell. The space between the pressure shell and the outer shell is divided into several unpressurized equipment bays, housing various electronic and environmental control system components. The inner shell the astronauts occupy is pressurized to 5lb/sq. in (about 1/3 normal earth pressure) with pure oxygen. Components are installed in the space between the cabin and the spacecraft's outer shell so that any failed units can be easily replaced from outside. Each man has his own hatch above his head and the relatively large size allows for EVAs in later flights. The pressure shell has a total volume of 80cu ft. although each man has only 20cu ft. of useable space. The legacy Cabin section is the crew station for two astronauts; this section will be “stretched and widened” to facilitate the addition of two more crewmen behind the existing seat positions. It will be geometrically the same shape and retain the same angles between the modules to ensure interface thermals are managed. The Cabin Module is a pressure vessel housing pilot ejection seats, electrical distribution system, and life support equipment. Equipment bays are located external to the pressure vessel and are designed to contain vehicle avionics. Re-sizing the Cabin Module to

accommodate a four man crew will provide the additional volume needed to increase the life support and other essential systems.

A large ablative heat shield at the base of the capsule protects the crew from re-entry heating; heat resistant shingles, composed of Rene' 41, protect the sides of the capsule. The Gemini ablative heatshield was made of a silicone elastomer; this will be re-evaluated for continued use on the Eclipse.

Each crewmember sits beneath a large, rectangular, hatch that is hinged on the outboard side and includes a small viewing window.

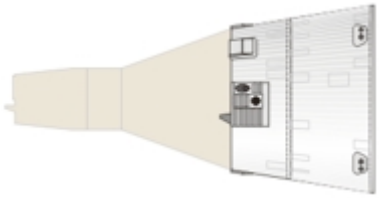

Figure 2.2.-5 Adapter Module Consists of the Retrograde and Service Modules

Adapter Module - The adapter section includes the retrograde section and the equipment section or service module. It houses most of the environmental supplies for the crew - oxygen, batteries, fuel cells, coolant and propellant, it supplies monometbylhydrazine and nitrogen tetroxide for all the maneuvering thrusters. When the two lower sections separate before re-entry, crew supplies are provided from smaller sources within the capsule itself.

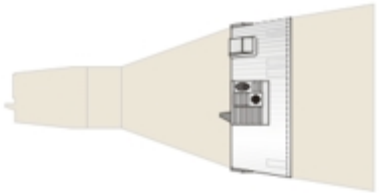

Figure 2.2-6 Retrograde Module

Retrograde Module - The Retrograde Rocket Motors consist of four 2,500 lbf thrust solid-propellant rocket motors symmetrically located about the longitudinal axis of the spacecraft. At retrograde, these motors are to impart an impulse to the Re-Entry Module. The resulting velocity decrement permits re-entry into the earth's atmosphere. The Retrograde rocket motors also provide the required spacecraft velocity and separation distance from the launch vehicle in the event of a mission abort.

The de-orbit motors are mounted on a cruciform structure within the section. Six maneuvering thrusters are located around the perimeter of this section. Four of these thrusters, located 90 degrees apart from one another and firing perpendicular to the spacecraft’s long axis, provides translation control in two axis. Two additional thrusters, firing forward, provided braking thrust.

The Orbit Attitude and Maneuvering System (OAMS) provides attitude and maneuver control of the spacecraft from the time of spacecraft separation from the launch vehicle until retrograde, at which point control of the spacecraft is transferred to the RCS on the Re- Entry Module. Like the RCS, the OAMS is a completely redundant hypergolic system.

Figure 2.2-7 Service Module

Service Module - Eight reaction control thrusters, located near the base of the service module provide roll, yaw, and pitch control. Two thrusters, firing aft and perpendicular to the main spacecraft axis, provide forward thrust. The Control System provides the Eclipse with the capability to maneuver in space, both translational and rotational about the three major axes. The Control System consists of the Horizon Sensor System, the Attitude Control and Maneuver Electronics, the Hand Controllers and the displays necessary to pilot the spacecraft.

While the Gemini design is a proven and very successful system, the Avionics System is an excellent candidate for taking advantage of 21st century technology. The weight and size reductions in avionics equipment may facilitate relocating many of these components to the outer shell of the Cabin module resulting in re-uses for future flights. Additionally the increase in Cabin size to provide crew seating for four astronauts may provide sufficient space to relocate most of the avionics and environmental systems. Items such as antenna arrays, thermal sinks, etc. would remain as needed.

The communication and tracking system: provides two-way voice communication, ground-to-spacecraft command link, spacecraft-to-ground telemetry transmission, radar tracking signals and recovery aids. Subsystems consist of telemetry, tracking, voice communications, digital command, antennas, and recovery aids.

Environmental control system components, oxygen tanks, fuel cells, battery containers, fuel tanks for the orbital maneuvering system, and other electrical equipment, are housed in this section. Equipment varies according to each mission's objectives. For instance, the longer duration missions carried fuel cells, while the shorter missions relies on batteries for electrical power. As a part of this effort the use and potential application of solar arrays will be evaluated.

The Environmental Control System: can be separated into five systems, or loops: the oxygen supply system, the cabin loop, the suit loop, the water management system and the coolant system.

- The oxygen supply system provides breathable atmosphere to the crew through the cabin loop and the suit loop at a regulated pressure. The crew is provided with redundant atmospheres by having a closed pressure suit circuit within the pressurized cabin.
- The water management system stores drinking water in tanks located in the Adapter Module and dispenses drinking water to a transparent tank in the Re-Entry Module. Waste water is collected and routed to the water evaporator or dumped overboard.
- The coolant system provides temperature control by active and passive means for the crew and spacecraft equipment. Heat is rejected to ground systems prior to launch. On orbit, heat is transferred by the coolant loop to the radiator where it is radiated to space. During re-entry, the heat is radiated to space by the Re-Entry Module surfaces and is also absorbed by the heat sink capacity of the Re-Entry Module structure.

The Service Module also provides the mating interface to the launch vehicle if a payload module is not included in the mission. The legacy Gemini spacecraft interfaced with the Titan which is a ten (10) foot diameter where the Eclipse will interface with the [redacted] which is a twelve (12) foot diameter interface.

Payload Module: An mission unique module the Payload Module (PM) extends from the aft end of the Eclipse Service Module to the forward flange on the [redacted] Second Stage interface. The conceptual design is a monocoque construction with external insulation to provide both aerodynamic and heat protection. Internal structure provides lateral stiffness for the Module as well as mounting locations for ISSEP payloads.

As defined in the mission plan the payload module may be separated from the Eclipse at a pre-determined time during the mission, and become a self-sufficient satellite or space experiment.

The PM will also be designed to allow the robotic arm on the International Space Station to extract payloads from the PM that are designed to be parked on the Space Station trunion mounts.

2.2.1 Spacecraft Development

The initial task is retrieval of the Gemini Mission XII flight drawings, component specifications and manufacturing process specifications that are currently in archival storage; SpaceOps has access to these drawings. Configuration Management (CM) of the drawings will be initiated immediately to control the baseline Eclipse configuration. Simultaneously the development plans, fabrication drawings and final test results of the Paraglide Landing System (PGLS) and the Parasail Landing System (PSLS) (refer para. 2.2.1.1.4) from archival storage will be conducted.

The interior arrangement of the Eclipse is shown in Figure 2.2.1-1; all-metal modules are of stressed skin and semi-monocoque construction fabricated using rings and circumferentially spaced stringers to carry axial and bending loads.

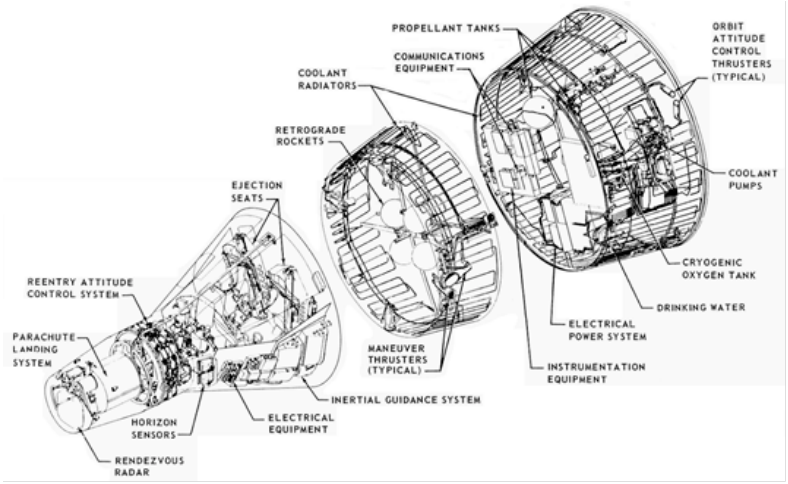

Figure 2.2.1-1 Interior Arrangement

The Eclipse spacecraft will consist of two basic configurations as follows:

- Two-person Crew Transportation System (CTS) comprised of the main capsule, retrograde module, service module, and the optional payload module, illustrated in Figure 2.2.1-2.

- Four-person CTS comprised of the main capsule, potentially a larger retrograde module and a potentially larger service module with an optional payload module, illustrated in Figure 2.2.1-3.

This approach will focus on the basic two man configuration development and demonstration so that the main features and success of the Gemini program can be exploited. The four man configuration will be developed and tested in parallel with the fabrication and integration phases of the two man configuration to meet the proposed launch schedule.

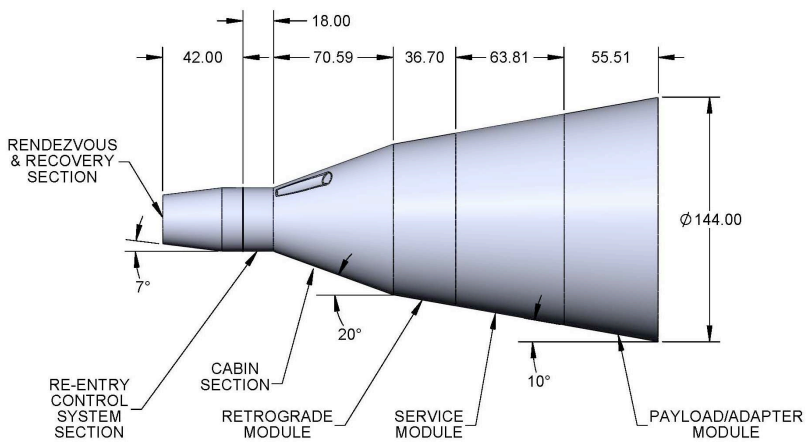

Figure 2.2.1-2 Eclipse A – Two Seat Configuration with Payload Module

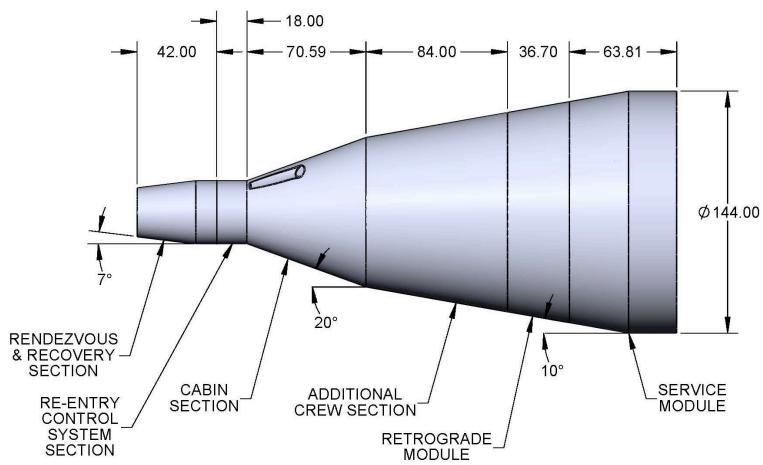

Figure 2.2.1-3 Eclipse B – Four Seat Configuration without Payload Module Shown

2.2.1.1 Structures

A design review of the final Gemini drawings and materials will be conducted to determine the feasibility of alternative material choices. SpaceOps already has 90% of the Gemini XII drawings converted to a 3-D structural CAD models which will be converted to 2D manufacturing, tooling and fabrication drawings. These drawings will be used to fabricate the structural components, electrical harnesses, and procure the avionics equipment. SpaceOps is working to put an SAA in place with Marshal Space Flight Center (MSFC) to utilize NASA clean room facilities, processes and procedures, and NASA personnel to aid in the Eclipse assembly, integration and test of the first three spacecraft.

2.2.1.1.1 Mass Properties

The utilization of modern technological advancement in avionics will require a through mass properties analysis to ensure the moment arms, moment of inertia, parallel axis and center of gravity are controlled. As the components become available the actual weights will be used to calculate the final mass properties. This information will be used by the flight control system for attitude and maneuvering calculations.

We have found that the single largest source of error in mass properties calculations is the uncertainty of the reference. The first step in calculating mass properties is to establish the location of the X, Y, and Z axes; it will be chosen at a location that can be accurately measured and identified. The dimensional data will be sufficiently accurate to permit mass properties tolerances to be met. Therefore serious attention will be given to the accuracy of the alignment of the modules when they are assembled. An alignment error is amplified as the vehicle gets longer for example a 0.001 inch lean introduced by an alignment error on a 12 inch diameter can result in a 0.007 inch CG error on a 15 foot long section. Equipment will be distributed to ensure a static imbalance is minimized or eliminated; if this cannot be accomplished cost effectively then a ballast weight will be considered to reduce the unbalance to zero. Mass properties will be calculated for each stage of launch, early orbit maneuvering, and re-entry operations.

2.2.1.1.2 Aerodynamics

Analysis and wind tunnel tests of the Eclipse in reentry will be used to confirm reentry aerodynamics analysis and virtual and mock-up simulations to confirm and validate the following:

Inertial-platform accelerometer measurements are used to calculate the force coefficients. These accelerations are transformed into the body axis system and then corrected for any rotational acceleration (due to the distance between the IMU and the spacecraft center of gravity before being put into coefficient form.

The flight aerodynamic-body force coefficients are calculated using these translational accelerations and the calculated dynamic pressure. The coefficients are then transformed into a rectangular coordinate frame relative to the ground. This frame is defined by the plane formed by the airstream velocity vector V_A and the geodetic Z-axis Z_{GD}, which is also in the orbital plane. The aerodynamic parameters relative to the orbital plane are those used to shape the trajectory.

In-plane parameters control longitudinal range, and parameters perpendicular to this plane control lateral (cross) range. Although the drag coefficient C is in the ground-relative frame, it is identical to the conventional drag coefficient, that is, the coefficient along the free-stream velocity vector. The resultant lift coefficient $C_{L,\,RG}$ is the total force coefficient perpendicular to the free-stream velocity vector. Coefficient ratios relative to the ground (and to the orbital plane) as well as resultant, or total, lift-to-drag ratio are also calculated and will be validated.

2.2.1.1.3 Internal Configuration and Human Factors

Although the Gemini design has an outer shell that provides approximately 80 ft^2 for the crew compartment the inner pressure shell and avionics storage between the shells reduces this volume to approximately 20 ft^2 per man as illustrated in Figure 2.2.1.1.3-1. This is sufficient for short duration flights but is not very accommodating to longer duration flights.

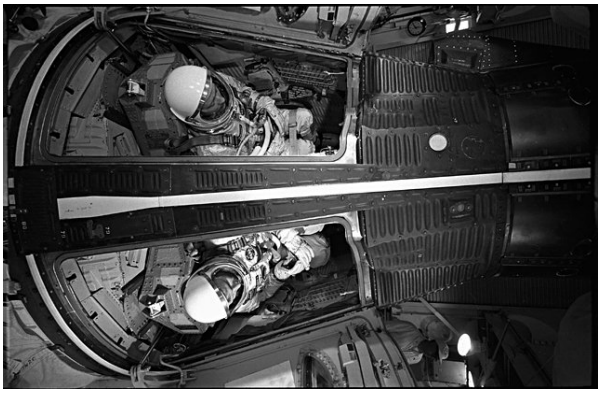

Figure 2.2.1.1.3-1 Crew Cabin Provides Sufficient But Limited Crew Mobility

During the design phase of the four seat (Eclipse B) configuration the SpaceOps team will utilize the guidelines defined in NASA-STD-3001 to define the human factors requirements while increasing the volume and to provide additional personnel mobility.

2.2.1.1.4 Re-Entry and Landing System

The Gemini Spacecraft utilized a water landing system that provided a safe rate of decent and a proper attitude for a water landing in the Atlantic Ocean as depicted in Figure 2.2.1.1.4-1.

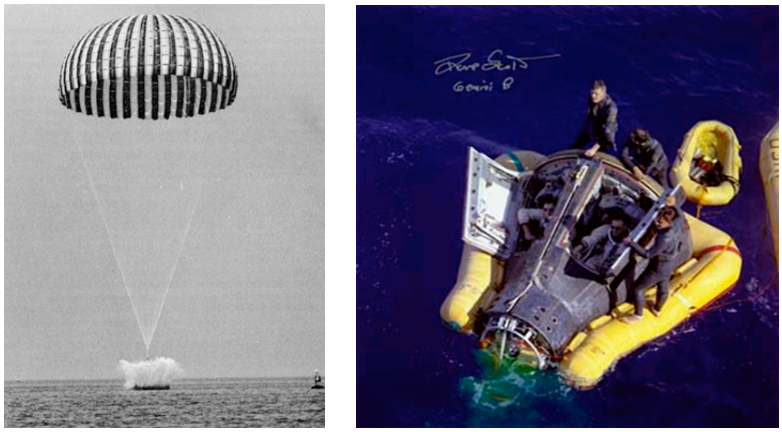

Figure 2.2.1.1.4-1 Gemini Water Recovery

Alternative systems that were developed during the Gemini program provided dry land landing systems; namely the Parasail Landing System and Paraglider Landing System. Additional trade studies and review of existing designs will be conducted to determine if these approaches are still viable and which system is the best method of safely returning the Eclipse Re-Entry Module to earth.

SpaceOps will attempt to obtain the original models to perform wind tunnel and land system landings to validate the data with the Eclipse vehicle.

Parasail Landing System (PSLS): In September, 1962, a land landing system development program was initiated with the following objectives: to compile a complete data description of system performance, to prepare detailed specifications for flight hardware, and to determine the operational procedures by which the land landing system could be employed in the Gemini Program. Twelve full-scale radio- controlled tests were conducted during the final phase of

testing using a Gemini boilerplate vehicle which was ballasted to the correct weight and center of gravity location and executed a nominal Gemini parachute deployment and attitude repositioning sequence. These full-scale flight tests integrated the developed components into a working system, and culminated in a completely successful demonstration of the PLLS at Gemini spacecraft design conditions.

The major system components of the PSLS shown in Figure 2.2.1.1.4-2 are: the parasail, retrograde solid propellant rocket motors fired just prior to touchdown to reduce decent to within the maximum allowable impact velocity of 10 ft. /sec, mechanical landing gear, turn control motors for steering the parasail, altitude sensing devices for ignition of the retrograde motors and visual reference requirements for the pilot.

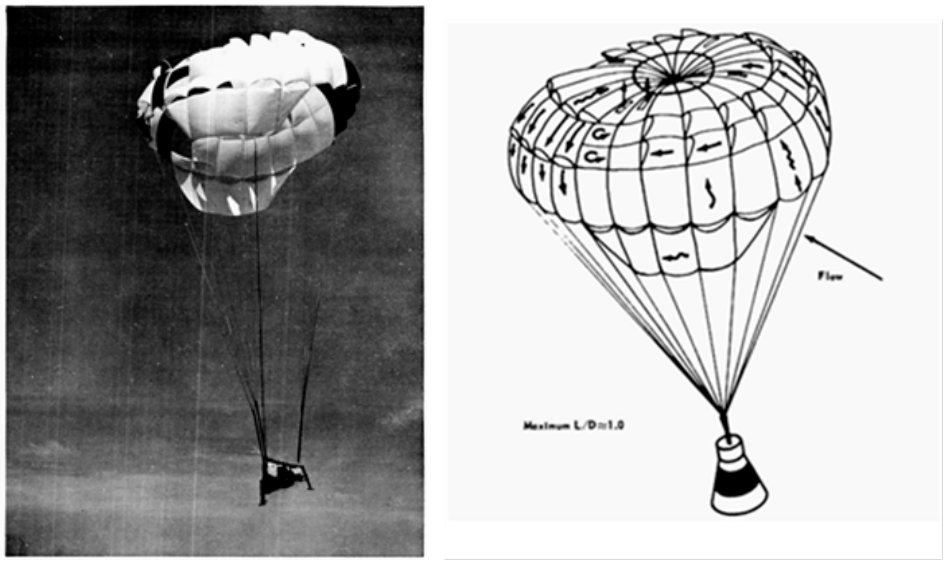

Figure 2.2.1.1.4-2 Parasail With Test Vehicle and flow indicators

The parasail is a high performance gliding parachute which has a forward velocity as well as a defined rate of sink. Unlike a conventional parachute, the parasail is capable of controlled gliding and turning. The PSLS incorporates means of directional control and, coupled with its forward velocity, allows limited range and maneuvering to avoid local obstacles. The parasail retains the light weight, low volume, and high reliability of standard parachute systems.

The final 70-ft parasail configuration met all the PLLS requirements and was stated to be ready for space flight qualification. At the completions of testing, the total weight of the parasail, deployment bag, pilot parachute and risers was 153 pounds.

Paraglider Landing System (PGLS): In May, 1961, development of a Manned Spacecraft PGLS was initiated with the purpose of establishing design parameters for a system to provide spacecraft maneuverability and controlled energy descent and landing by aerodynamic lift. The PGLS test program has demonstrated the system to be stable, responsive, and very maneuverable, displaying adequate control in all in-flight maneuvers and landings. The system was flown and landed on numerous occasions, and though landing data and several above-specification landings indicate that the system would require further landing tests to become operational, it can be said that concept is feasible for a landing system with which a trained astronaut could consistently make a safe land landing.

The PGLS shown in Figure 2.2.1.1.4-3, consists of the paraglider wing, mechanical landing gear, turn control motors and visual reference requirements for the pilot. The paraglider wing illustrated in Figure 2.2.1.1.4-4 is a Rogallo-type wing composed of an inflatable frame with an attached sail. The prototype wing had a sail area of approximately 600 sq. ft. and a working pressure of 15.7 +1.25/-0.00 psig.

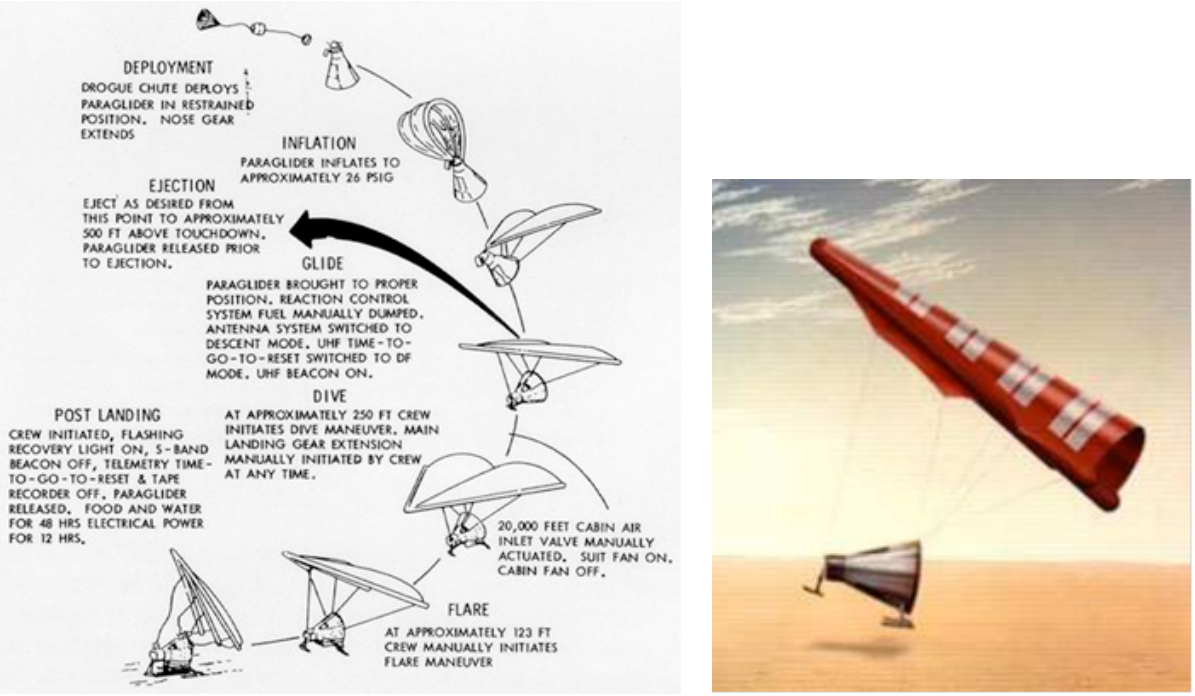

Figure 2.2.1.1.4-3 Paraglider Landing System with Test Vehicle

Following paraglider deployment, attitude control is accomplished by lengthening or shortening the control lines connecting the paraglider and the vehicle. These control lines are controlled by gas operated winches manually activated by the pilot through the Eclipse hand controller. The paraglider control system has pitch and roll capabilities, but no yaw control.

On landing, we propose to use the Rogallo wing concept due to the large amounts of development and test data available. New materials with an improved aerodynamic configuration will provide an optimum landing method. Clearly, a Rogallo wing will have to be developed from the original Gemini design and tested for flight worthiness and safety assurance.

On the Gemini program, the Rogallo paraglider was chosen as the preferred landing system and tested at half scale. It was deemed the most likely concept to meet the program goal of a pilot-controlled pinpoint landing. If trade studies again indicate that this is the best approach the SpaceOps will continue testing this approach with a full scale mock-up drop test from a minim of 40k feet and a second drop with a first article Eclipse spacecraft.

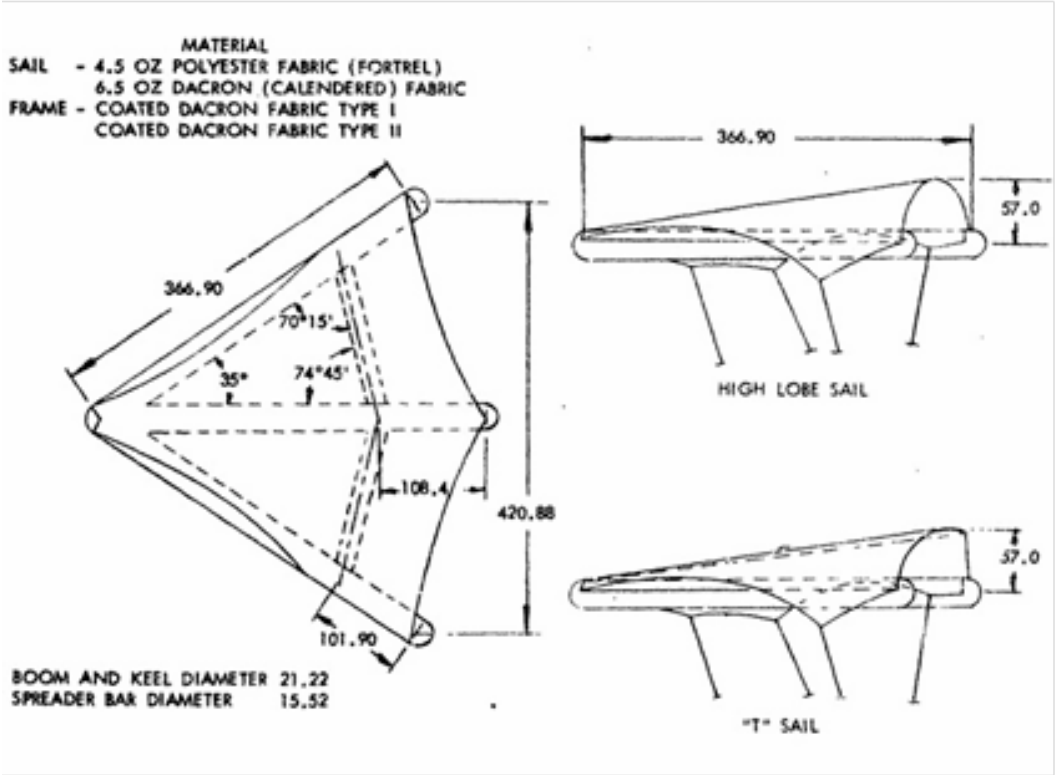

Figure 2.2.1.1.4-4 Paraglider Prototype Wings

2.2.1.2 Propulsion

The propulsion system consists of the Reaction Control System (RCS), the Retrograde Rocket Motors and the Orbit Attitude and Maneuvering System (OAMS).

The RCS provides attitude control of the Re-Entry Module from retrograde to deployment of the Paraglider, subsequent to jettisoning of the OAMS. The RCS consists of completely redundant systems, each with eight 25 pound-force (lbf) thruster assemblies operating on storable hypergolic propellants.

The Retrograde Module rocket motors consist of four 2,500 lbf thrust solid-propellant rocket motors symmetrically located about the longitudinal axis of the spacecraft refer Figure 2.2.1.2-1. At retrograde, these motors impart an impulse to the Re-Entry Module. The resulting velocity decrement permits re-entry into the earth's atmosphere. The retrograde rocket motors also provide the required spacecraft velocity and separation distance from the launch vehicle in the event of a mission abort.

The OAMS provides attitude and maneuver control of the spacecraft from the time of spacecraft separation from the launch vehicle until retrograde, at which point control of the spacecraft is transferred to the RCS System on the Re-Entry Module.

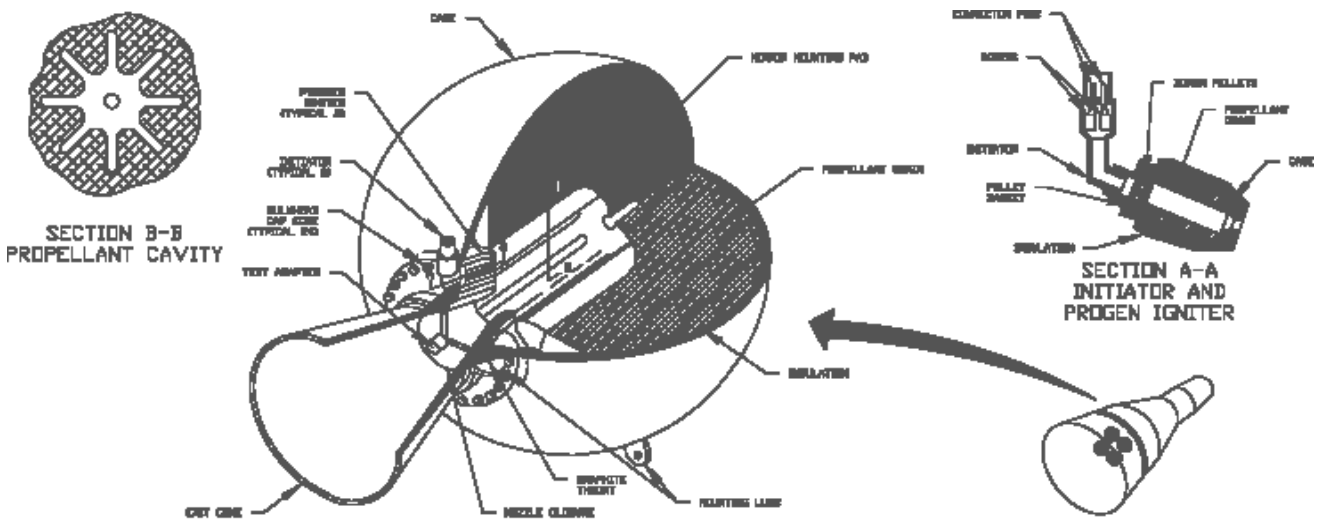

Figure 2.2.1.2-1 Eclipse Retrograde Rockets

2.2.1.3 Electrical

Three types of silver-oxide zinc batteries were used throughout the Gemini Spacecraft. The flight crew can control interconnection of the batteries, thus providing redundancy and maximum power utilization from all onboard sources. However, battery technology has improved dramatically since Gemini, with Nickel-Hydrogen, Nickel Metal Hydride, and Lithium batteries also commonly used in space applications. For typical Eclipse missions, electrical power would be provided by a combination of battery and fuel cell technology, with the fuel cells located in the Service Module. While the fuel cells would provide the majority of on orbit power, batteries would be required during the re-entry phase of the missions, to provide lighting, communications, GNC, environmental control, pyrotechnics, and any other power requirements. The total time that the spacecraft would operate solely on battery power is limited to the period between the jettison of the Service Module and the time the spacecraft is powered down at the landing site, for a nominal mission. To accommodate an off nominal scenario, the batteries would be sized to provide the re-entry time, plus an amount that would be determined thru trade studies during the initial project phase. These trade studies would also analyze battery technology, weight, capacity, and other factors, to select a preferred battery technology.

In addition to battery technology, a number of other electrical improvements may be made to the original Gemini design. Current bus technologies provide a significant weight savings over the original point to point wiring methods. Use of technologies such as Mil-Std-1553B and Ethernet are commonly used in modern spacecraft, with equipment that utilize these technologies commonly available. Use of such interfaces, where feasible, will provide a rapid integration cycle utilizing off the shelf technology. This will contribute to a robust electrical wiring subsystem that will provide a significant cost saving in realization, integration, and maintenance, while providing access to state of the art instrumentation and avionics. The bus approach will also provide integration of capabilities such as real time video which commonly use powered Gigabit Ethernet technologies. In addition to bus wiring, there will be a number of instances where point to point wiring is required. These instances will include, but not be limited to, Power wiring, safety circuits, pyrotechnic circuits, antennas, and other systems where continuous and or deterministic circuits are either required or preferred to multiplexed circuits.

2.2.1.4 Avionics

The heritage avionics system used on the Gemini Spacecraft, while proven and very successful is an excellent candidate for taking advantage of 21st century systems used by the latest launch vehicles. The Eclipse avionics system will be redundant and incorporate the latest safety features to ensure the safe return of the crew and spacecraft.

2.2.1.4.1 Guidance, Navigation, & Control

The development of the concept of a glass cockpit has provided a number of manufacturers who currently build Guidance Navigation and Control (GNC) systems. The primary consideration of the manufacturer would be the size and weight of the main electronics and the configuration and ability to customize the main displays and input systems. A number of companies have emerged that provide the ability to customize the layout, develop touch screen capabilities, and provide informational displays that were unthinkable 20 years ago. Current technologies also provide enhanced reliability and redundancy that would virtually eliminate the risk of an in-flight failure, since each cockpit seat would have fully independent GNC systems, and the ability to provide a third backup system with minimal weight impacts. A number of companies, with experience in GNC, provide complete systems that are easily integrated with Eclipse via modern communication busses. The availability of such complete systems does not imply that no integration work would be required. Rather the basic hardware system would be selected via trade study, and the software would be modified to include the specific Eclipse interfaces, inputs, and controls. A large amount of integration would be required, but selection of the appropriate GNC hardware would provide the necessary integration tools to customize the GNC software with the minimum amount of new software development.

Communications: The communications subsystem will require the use of all common voice and data systems, including but not limited to S-band, Ku and Ka Bands as well as TDRSS. During the launch phase of the mission, a large amount of data is transmitted to ground stations from both the [redacted] supplied [redacted] as well as the Eclipse spacecraft. As the spacecraft travels down-range, the direct downlink capabilities will transition to TDRSS for constant communications capability. Trade studies will be performed during phase one of the program, which will identify an optimum communications provider that will support the program. Selection will identify all communication types required by Eclipse, while providing redundancy and minimizing mass. While the entire program is planned for commercial payloads, the communications systems will require encryption to prevent unauthorized interception of ground communications, as well as to prevent any possible interference with onboard systems by unauthorized ground stations. If the spacecraft will be used for classified missions, an appropriate, approved encryption system will be required.

2.2.1.5 Crew Systems

The primary escape system for the Gemini crew was the ejection seat system, consisting of ejection seats, seat rocket/catapult, hatch actuating system, personnel parachute system, and survival equipment. The escape system was designed to safely allow the astronauts to escape from the spacecraft while on the pad, during launch or after re-entry.

2.2.1.5.1 Human Factors

Human factors design issues include volumetric allowances, architecture and layouts, closed life support systems, health maintenance systems, and automation. An understanding of behavioral

responses to design elements is a precondition for critical design decisions; the NASA-STD-3001 will be specified in the design of the four seat Eclipse configuration.

An Eclipse mockup will be used as a simulation tool to provide the capability for static part-task evaluation of the ingress and egress mission phases. Mockups provide the opportunity to study the design problem in three dimensions through both observations and demonstrations. Key uses of mockups include, but are not limited to:

- Evaluation of alternative equipment configurations.
- Determination of workspace difficulties from simulating operational tasks.
- Identification of accessibility problems from simulating maintenance operations.
- Planning locations for routing wiring, plumbing, etc.
- Determination of geometry or volume problems affecting ingress or egress.
- Evaluation of procedures.
- Determination of optimal placement of crew controls from clearance, reach, and visibility envelopes.

The Cabin Module and seats must be designed to accommodate 40-year-old 95th percentile American male and 5th percentile Japanese female crew members as defined by NASA-STD-3001, with a 3% spine stretch due to zero gravity. The three dimensions that must drive the seat design are the sitting height, the popliteal height, and the buttock to popliteal length.

The human factors evaluations include human testing in all phases of ground and in-flight tests. The in-flight tests will be performed as part of the crew training and selection process. Further, specific design issues addressed in the evaluations will include:

- Preferred hatch opening, closures and locations.
- Suitability and function of conceptual layout (seats, stowage, displays, handholds, etc.).
- Ease of ingress for de-conditioned and injured persons.
- Assessment of optimal seat locations for crew control and ill or injured crew.
- Assessment of general crew module volume.

2.2.1.5.2 Emergency Egress

The primary escape system for the Gemini crew was the ejection seat system, consisting of ejection seats, seat rocket/catapult, hatch actuating system, personnel parachute system, and survival equipment. The escape system was designed to safely allow the astronauts to escape from the spacecraft while on the pad, during launch or after re-entry. Advancements in ejection seat technology in the last 50 years has yielded more efficient, reliable and cost effective options.

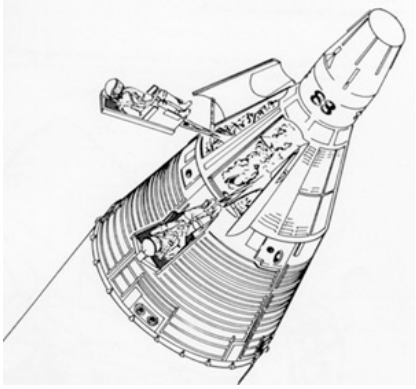

Figure 2.2.1.5.2-1 Upgraded Ejection Seats will be Used on the Eclipse

The Eclipse emergency egress method for both lower atmosphere launch and the landing mode is proposed to be modern ejection seats such as, the Martin-Baker Mk. 12H. The Mk.12 seat is a cost effective escape system that provides significant seat performance over the speed range zero to 625 knots and from ground level to the service ceiling of the Harrier aircraft (~50,000 ft). The seat has the capability of sensing airspeed and adjusting its mode of operation accordingly. The Mk.12 seat has been qualified for the UK Royal Air Force Harrier GR7 (AV8B). The Crew Escape System will be linked to the Falcon 9 flight termination system to ensure the mission abort sequence, if required, is properly initiated.

The testing conducted during the Gemini program resulted in the development of a "ballute" (short for balloon-parachute) system that ensures rapid chute opening at low altitudes such as on a launch pad. This system will be evaluated for application.

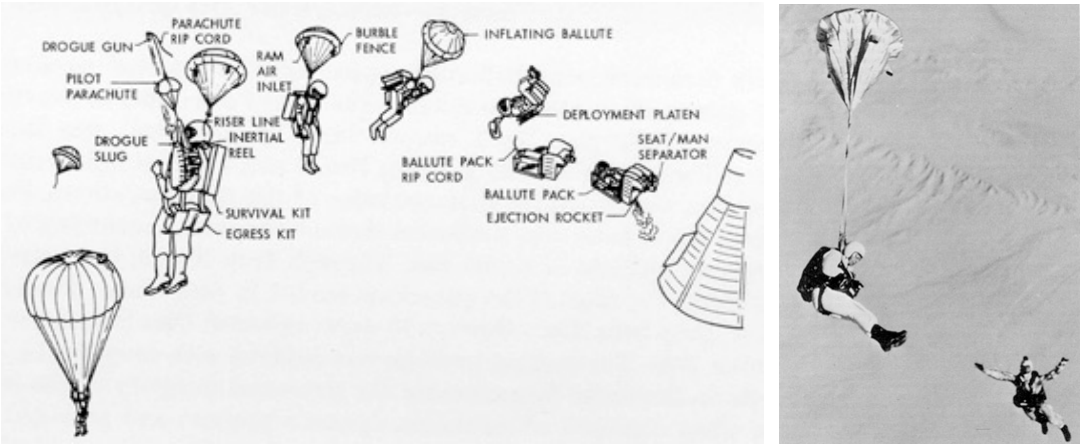

Figure 2.2.1.5.2-2 Proven Ballute Design

In addition to egress required during launch, the Gemini's ejection seats served as the crew escape method in the lower atmosphere and for the original landing mode, which involved deployment of a huge inflated Rogallo wing with a piloted landing on skids or wheels.

2.2.1.5.3 Life Support Systems

The elements necessary for maintaining human life and health in the Eclipse depend greatly upon specific mission requirements and will, at a minimum, provide breathable atmosphere, suitable pressure and temperature, and likely food and water. The Life Support Systems (LSS) may or may not collect, dispose, or reprocess wastes such as carbon dioxide, water vapor, urine, and feces since the mission will be of limited duration and space suits will be required. The major subsystems required to accomplish the general functions are as follows:

- *Breathing and pressurization gas storage system* – The air supply system provides breathable atmosphere to the crew through the cabin loop and the suit loop at a regulated pressure. The crew is provided with redundant atmospheres by having a closed pressure suit circuit within the pressurized cabin.
- *Temperature and humidity control system* – The coolant system provides temperature and humidity control by active and passive means for the crew and spacecraft equipment. Heat is rejected to ground systems prior to launch. On orbit, heat is transferred by the coolant loop to the radiator where it is radiated to space. During re-entry, heat is radiated

to space by the Re-Entry Module surfaces and is also absorbed by the heat sink capacity of the Re-Entry Module structure.

- *Carbon dioxide control system* – CO2 levels must be continuously monitored and controlled.
- *Trace contaminant control system*
- *Water management system* – The water management system stores drinking water in tanks located in the Adapter Module and dispenses drinking water to a transparent tank in the Re-Entry Module.
- *Waste management system* – Waste water is collected and routed to the water evaporator or dumped overboard

Fire Suppression System: The Eclipse fire suppression system will be an upgrade based on the surprising results of an experimental investigation. The experiment was to determine the effectiveness of various suppressant agents in spacecraft fires under normal-gravity conditions; it was conducted at the Center for Commercial Applications of Combustion in Space at the Colorado School of Mines. Ultra-fine water mist driven by air or by nitrogen is compared to carbon dioxide and nitrogen to evaluate their performance to suppress and extinguish fires involving energized wires, electrical components on circuit boards, and cloth. Extinction time and suppressant amount are used as a measure of agent effectiveness. On a mass-basis, ultra-fine mist driven by air is found to be more effective than carbon dioxide and nitrogen, with the exception of highly energized burning wires, where ultra-fine mist is used to cool the hot wire, as well as to extinguish the small remaining fire. In this case, ultra-fine mist driven by nitrogen performs similarly to the gaseous agents. The low momentum and pseudo-gas properties of ultra-fine mist with lower-than-10-μm droplets are also more effective than the high-momentum and larger droplet sizes generated by high-pressure water-mist sprays.

On a mass-basis comparison, ultra-fine mist is found to be more effective than CO_2 and N_2, with the exception of the burning-wire scenario with a high preheating power. In this case, ultra-fine mist is used to not only extinguish the fire, but also to cool the hot copper wire. This fact, in conjunction with the "small-fire syndrome" characteristic of water-mist suppressed fires, results in longer extinction times for ultra-fine mist, as compared to gaseous agents. Although at first the longer extinction times and additional mass of agent may appear as a disadvantage of water mist against CO_2or N_2, the cooling effect of the water mist has the advantage of preventing the rupture of the wire, as it occurs when CO_2 and N_2 are applied at low mass-flow rates. This cooling effect may be essential to avoid disrupting the operation of critical equipment onboard the spacecraft while the fire is being suppressed. If shorter extinction times are desired, ultra-fine mist driven by N_2can be used to combine the unique suppression properties of water mist in conjunction with the oxygen-depletion attributes of N_2. It is observed that the use of this mixed-agent suppressant results in a superior combination with excellent performance.

Space Suits and Personal Life Support Systems: As a transportation vehicle with limited mission duration, astronauts will be required to have fully operational space suits on the Eclipse missions in order to minimize on-board LSS reliance.

Consideration will be given to include a Personal Life Support System (PLSS) which will allow for extra-vehicular activity with maximum freedom. The functions of the PLSS to be considered include:

- regulating suit pressure,
- providing breathable oxygen,

- removing carbon dioxide, humidity, odors, and contaminants from breathing oxygen,
- cooling and recirculation of oxygen through the pressure garment, and water through a Liquid Cooling and Ventilation Garment or Liquid Cooling Garment,
- two-way voice communication,
- display and/or telemetry of suit health parameters,
- telemetry of an indicator of the wearer's immediate health (e.g. heart rate)

SpaceOps will procure the space suit for the third test flight from ILC Dover. Since the Apollo program, ILC has been the designer and producer of all the space suits for NASA. ILC continues to support NASA's ISS work and future space exploration initiatives. ILC Dover is continually pursing a number of advanced technologies relating to space suits and flexible systems. Technologies such as dust mitigation, radiation protection, self-healing, shape morphing, anti-microbial materials, power assist space suits and robotic covers, and structural health monitoring are being developed and demonstrated.

ILC has developed a number of shape morphing technologies for use in space suits, this technology reduces the number of suit sizes required and enhances fit and performance for space suits. Several space suit components have been built and extensively tested. Manual cord adjustments, nastic cells, and pneumatic tendons have all been investigated with excellent results.

Figure 2.2.1.5.3-1 ZEI Commercial Crew Suit (I-X4) and the Commercial Crew Suit (I-C2)

2.2.1.6 Mock-ups and First Articles

A preliminary mock-up is planned to be fabricated during the development phase of the base period. Mock-ups will hasten the developers' task of arranging communication, navigation, life support, power, reaction control, and emergency equipment. SpaceOps will rely on its subcontractors to perform this arrangement to maximize internal space and providing coordinated integration while taking full advantage of the modern developments in display and human factors technology.

2.2.2 Falcon 9 Integration

2.2.2.1 Integration

Integration begins at the MSFC clean room assembly in Huntsville, AL. SpaceOps intends to finalize an SAA with MSFC to utilize the facilities and experienced personnel during the first three (3) Eclipse assembly and integration phases; then to transition to SpaceOps personnel and facilities. As delineated in table 2.2.2.1-1, the integration process begins with the assembly and testing of each Eclipse module, then the assembly of the spacecraft, testing of each subsystem, system testing, and simulated flight and altitude testing prior to shipping the Eclipse to the [redacted] facilities at Cape Canaveral.

SpaceOps is proposing a vertical integration of the Eclipse with the [redacted] launch vehicle in the same manner as the Gemini was integrated (refer Figure 2.2.2.-1), however, if needed a horizontal integration and the associated tooling will be provided.

Table 2.2.2.1-1 Proposed Eclipse Assembly to Launch Integration Schedule

Proposed Integration and Test Schedule			
	E1-2U	E2-4U	E3-4M
Equipment Installation	13-Jul-13	30-Sep-13	23-Dec-13
Reentry Adapter Mating	4-Aug-13	3-Oct-13	14-Jan-14
System Assurance Tests	14-Aug-13	5-Oct-13	24-Jan-14
Environmental Control System Validation	14-Aug-13	5-Oct-13	24-Jan-14
Simulated Flight Test	1-Sep-13	15-Oct-13	1-Feb-14
Altitude Chamber test	15-Sep-13	7-Nov-13	15-Feb-14
Ship to SpaceX Facilities (Cape Canaveral)	1-Oct-13	26-Nov-13	1-Mar-14
Complex 19/Electronics Interface Integrated Validation and G&C	6-Oct-13	1-Dec-13	6-Mar-14
Mechanical Mate	20-Oct-13	15-Dec-13	20-Mar-14
Joint Combined System Test	3-Nov-13	27-Dec-13	3-Apr-14
Full Command Mission Training/Final Systems Test	12-Nov-13	4-Jan-14	12-Apr-14
Wet Mock-upSimulated Launch/Simultaneous Launch Demo	17-Nov-13	17-Jan-14	17-Apr-14
Final Simulated Flight Test	27-Nov-13	27-Jan-14	27-Apr-14
Launch	1-Nov-13	1-Feb-14	1-May-14

The [redacted] Payload Users Guide details the mating connector, mass properties, separation accuracy and mission accuracies. It describes the integration facilities, launch operations and testing, safety requirements, etc. All dimensional interface connections along with squib selection and placement will be coordinated with [redacted] in detail during the development of both transition sections for the Eclipse vehicles. Preliminary layouts indicate that the mating of the Eclipse and the [redacted] will require some analysis for dynamic response and thermal interaction; however, the [redacted] vehicle has been designed for the mating and the commonality between [redacted] vehicle and the Eclipse transition section is evident.

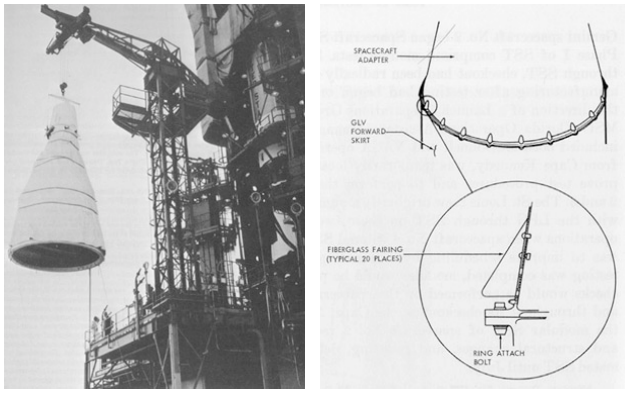

Figure 2.2.2.1-1 Verticle Integration Procedures have Been Proven with the Gemini

2.2.3 Launch Facilities

[redacted]

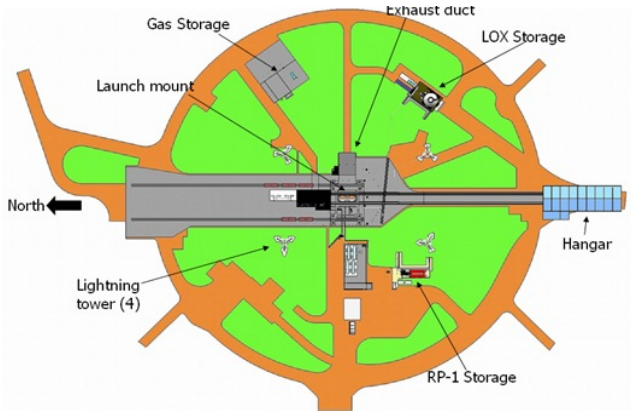

Figure 2.2.3-1 Launch Complex 40 at Cape Canaveral Air Force Station (CCAFS), FL

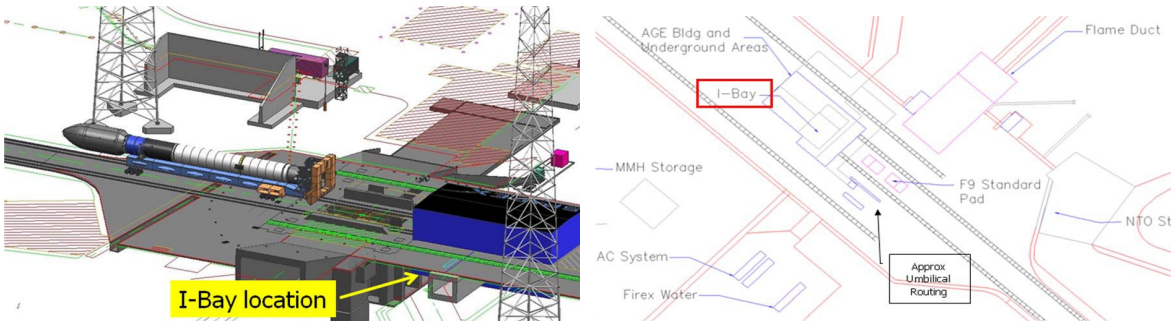

Figure 2.2.3-2 Space Launch Complex 40 Instrumentation Bay Location

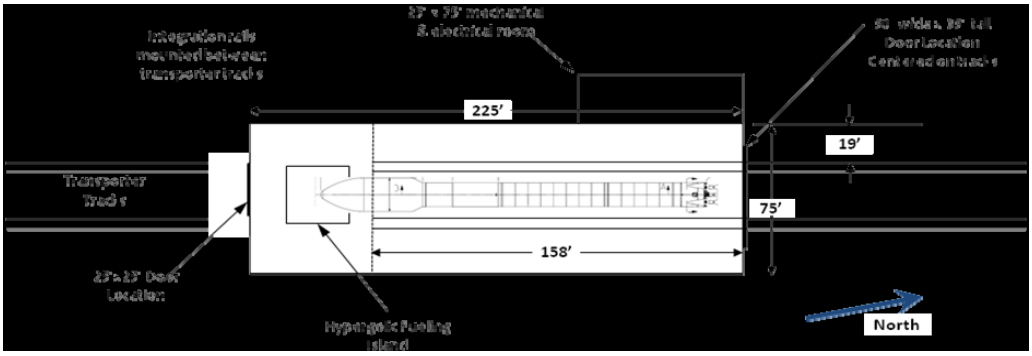

Figure 2.2.3-3 Space Launch Complex 40 Hangar Layout

[redacted].

Figure 2.2.3-4 [redacted] launch site, Kwajalein Atoll

2.2.4 Flight Plan

Three Eclipse test flights are being proposed as delineated in Table 2.2.4-1;

The first flight is to demonstrate the re-use of the Gemini spacecraft design with modifications to the material selection, potential relocation of avionics hardware, demonstration of the paraglider reentry canopy, and demonstration of the controlled de-orbit of the Service Module. It will be an unmanned two seat (Eclipse A) configuration that will orbit for one (1) day at 150 miles and reenter for a dry landing at Edwards AFB or Bonneville Salt Flats, illustrated in Figure 2.2.4-1.

The second flight is to demonstrate and gather data for a four (4) crew Cabin Module, demonstrate orbit changes to catch the ISS, perform an ISS control handover and fly-by maneuver. It will be configured to include an instrumented seat pallet for each crew position, and a four seat Cabin. It will be an unmanned four set (Eclipse B) spacecraft that will orbit for one (1) day at 150 miles to catch the ISS, perform an orbit adjust maneuver to 250 mile orbit, hand-over flight controls to the ISS in order to demonstrate a docking maneuver (but no docking will be performed), orbit at 250 miles for two (2) days and return for a dry landing at Edwards AFB or Bonneville Salt Flats, illustrated in Figure 2.2.4-2.

The third flight is similar to the second flight it is to demonstrate a four seat configuration, and perform an ISS docking maneuver. It is a four manned flight that will follow the same flight plan as the second flight plan except it will remain at an altitude of 250 miles for days before returning for a dry landing at Edwards AFB or Bonneville Salt Flats, illustrated in Figure 2.2.4-3.

Space Ops is in the process of establishing a SAA with Goddard Space Center to provide the necessary satellite ground tracking required for each of the three proposed flights.

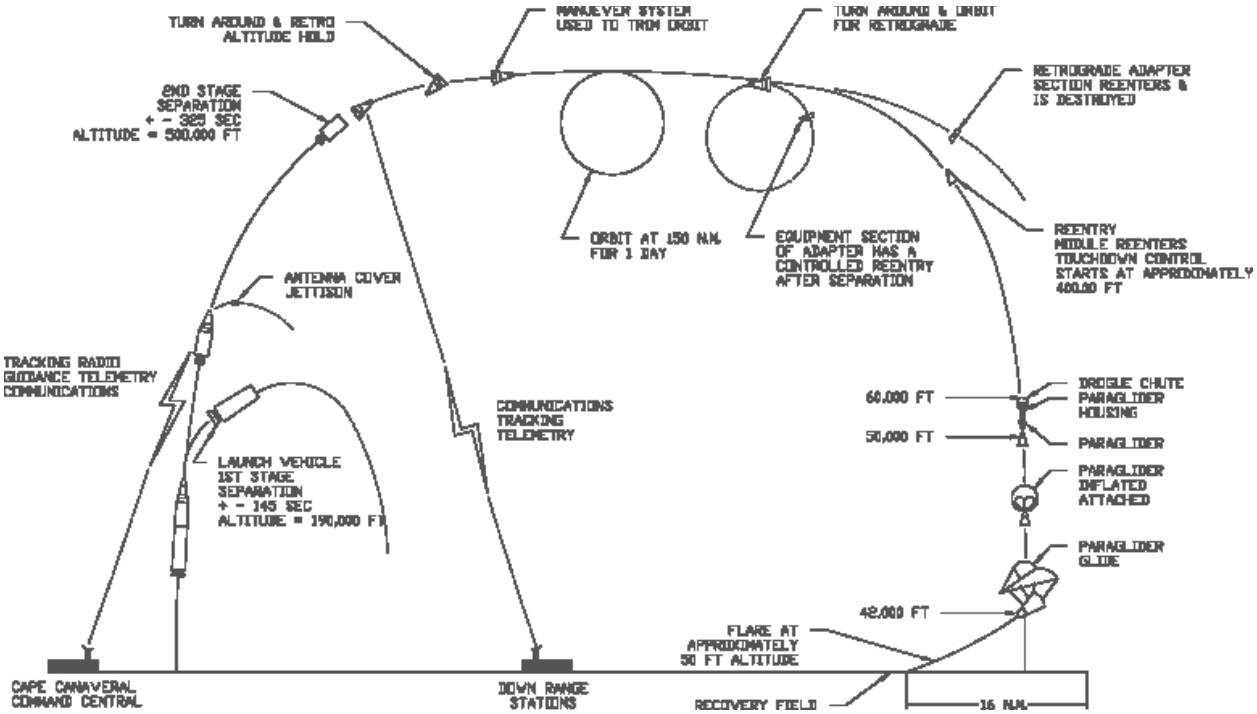

Figure 2.2.4-1 Proposed Flight Plan for the First Demonstration Flight

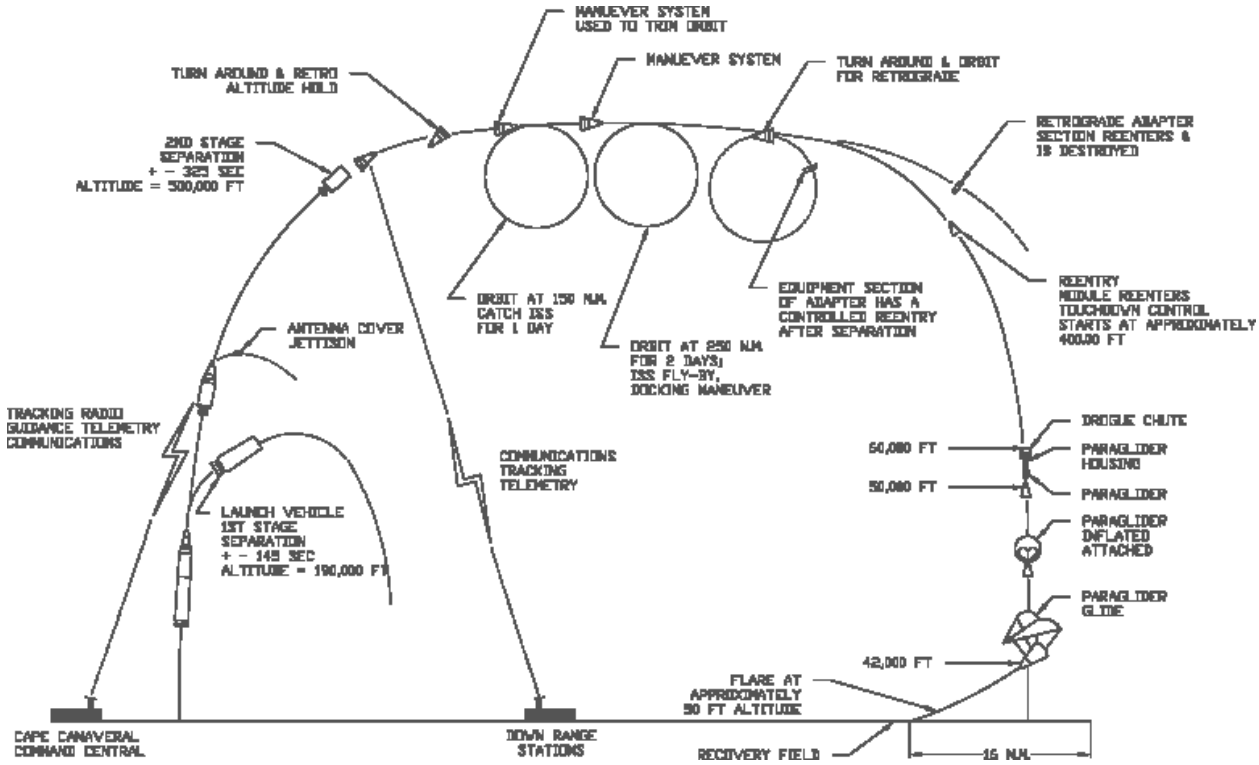

Figure 2.2.4-2 Proposed Flight Plan for the Second Demonstration Flight

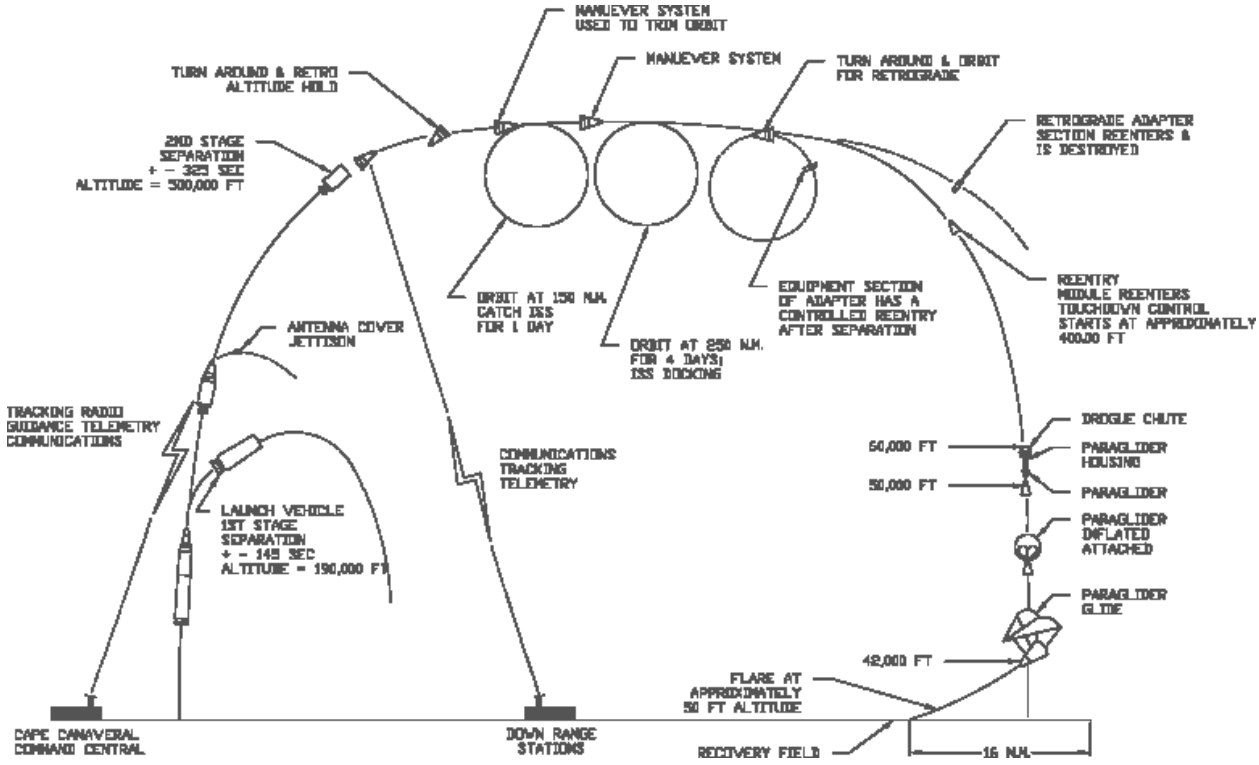

Figure 2.2.4-3 Proposed Flight Plan for the Third Demonstration Flight

Table 2.2.4-1 Proposed Mission Plan for the Eclipse Flight Tests

MISSION PLAN	I	II	III
ECLIPSE SPACE VEHICLE			
Spacecraft No.	E1-2U	E2-4U	E2-4M
Launch Weight (lb.)	6882	8250	8370
Launch Vehicle	[redacted]	[redacted]	[redacted]
Flight Crew			
Command Pilot	Unmanned	Unmanned	TBD
Pilot	----------	----------	TBD
Mission Commander	----------	----------	TBD
Principle Investigator	----------	----------	TBD
Back-up crew			
Command Pilot	----------	----------	TBD
Pilot	----------	----------	TBD
Mission Commander	----------	----------	TBD
Principle Investigator	----------	----------	TBD
Launch Date	2-Nov-13	4-Feb-14	6-May-14
Time (e.s.t.)	TBD	TBD	TBD
Launch Azimuth	83.6	87.4	98.8
ORBITAL PARAMETERS			
At insertion			
Apogee (n.m.)	140.00	150.00	150.00
Perigee (n.m.)	86.90	87.00	87.00
Period (min.)	87.92	89.00	89.00
Inclination Angle	28.97	29.00	29.00
At retrofire			
Revolution	16	48	80
Apogee (n.m.)	155.00	250.00	250.00
Perigee (n.m.)	143.00	230.00	230.00
Period (min.)	90.00	91.00	91.00
Inclination Angle	29.00	51.60	51.60
Highest Apogee (n.m.)	170	400	400
Lowest Perigee (n.m.)	86	86	86
G.E.T. to OAMS preretro (hr:min:sec)	----------	----------	----------
RETROFIRE & REENTRY			
Orbits to retrofire	16.4	48.4	80.8
G.E.T. to retrofire (hr:min:sec)	TBD	TBD	TBD
G.E.T. to touchdown (hr:min:sec)	retro + 20 min	retro + 25 min	retro + 25 min
Landing Coordinates			
Latitude	34:54:29N	34:54:29N	34:54:29N
Longitude	117:53:07W	117:53:07W	117:53:07W
Landing Accuracy (n.m. from planned landing point)	TBD	TBD	TBD
RECOVERY			
location	Edward AFB	Edward AFB	Edward AFB
Status	Pimary	Pimary	Pimary
Time (e.s.t.)	TBD	TBD	TBD

2.3 Safety and Mission Assurance

Crew safety is enhanced by requiring redundant systems throughout the overall configuration. Continuous review and verification of the application of the redundancy requirement will be conducted to eliminate single-point failure modes. To ensure crew safety, a comprehensive fail operational/fail safe (FO/FS) approach will be adopted for critical avionics systems on the Eclipse spacecraft. This requirement means that the avionics system must remain fully capable of performing the operational mission after any single failure and fully capable of returning safely to a runway landing after any two failures. The FO/FS requirement and the incapability of degraded backup systems to achieve a safe return to earth dictates the use of multiple avionics "strings," each independent from a reliability standpoint but each with equivalent capability. The Eclipse program will follow the longstanding NASA rule that a mission must be aborted unless at least two means of returning safely to Earth are available.

The Eclipse spacecraft will be fully human-rated and qualified for manned space flight. The avionics will include rugged flight computers, inertial navigation, GPS receivers, S-band transmitters for telemetry and video, a star tracking system and rendezvous radar. The vehicle will be linked to ground stations using NASA's TDRSS.

A trade study will be conducted to assess the power requirements for any given mission. Batteries, fuel cells and solar panels are all viable candidates for providing mission power. The Propulsion System, Environmental Control System and Thermal Protection System will provide functionality equivalent to that of the heritage subsystems while taking advantage of the latest advances in technology.

Safety: SpaceOps will comply with AFSPCMAN 91-710 Range User's Manual requirements in the design and operation of their flight and ground systems. These requirements encompass mechanical design, electrical design, fluid and pressurant systems, lifting and handling systems, ordnance and RF systems, ground support equipment, and other design and operational features.
Hazardous Systems and Operations: Most ranges consider hazardous systems or operations to include ordnance operations, pressurized systems that operate below a 4-to-1 safety factor, lifting operations, operations or systems that include toxic or hazardous materials, high power RF systems, laser systems, as well as a variety of other systems and operations. The details of the system design and its operation will determine whether the system or its operation is considered hazardous. Typically, additional precautions are required for operating systems that are considered hazardous, these will be determined during the safety approval process with SpaceX and the launch range. All hazardous operations will require procedures that are approved by both SpaceX and the launch range prior to execution. Ordnance operations, in particular, require coordination to provide reduced RF environments, cleared areas, safety support, and other requirements.
Waivers: For systems or operations that do not meet safety requirements but are believed to be acceptable for ground operations and launch, a waiver is typically produced for approval by the launch range safety authority. Waivers are a last resort solution and require considerable coordination. They should not be considered a standard practice. SpaceX will assist in determining whether an issue should be elevated to require a waiver as the integration process evolves.

2.3.1 Test Operations

Acceptance testing of any critical system like the proposed Eclipse spacecraft is conducted throughout the manufacturing of all of its elements to ensure the vehicle conforms to all

requirements as a basis for acceptance. Typically, acceptance testing is performed in conjunction with the manufacturing and assembly process. The rationale for conducting all acceptance testing at the factory is that early detection of an anomaly allows for the issue to be addressed in a timely fashion and also, that the factory will be better equipped to remove and replace faulty hardware.

Testing is also conducted at the launch site. Testing conducted at the launch site is to ensure performance of each subsystem has not degraded after shipment from the factory and to verify the vehicle is ready for launch, by conducting system level integrated testing. This will be followed by verification of functional interfaces between the [redacted], the SSM and the Eclipse Spacecraft after the vehicle has been stacked. No new or first-time testing will be performed on the SSM or Eclipse Spacecraft at the launch site.

Specific test and checkout requirements, as well as applicable tolerances, will be defined and documented. These test requirements must be performed on the spacecraft at the factory prior to final acceptance and at the launch site prior to flight. The test and checkout requirements shall clearly define what is to be tested.

Test and Checkout Procedures shall be prepared that define the detailed step-by-step sequence of events in a specific test. A test matrix will be developed to correlate test data with a specific test requirement as part of the test verification process.

2.3.1.1 Ground Testing

The structural design of space systems is dictated by the rigors of the liftoff and ascent environments during launch as well as the extreme thermal conditions and operational requirements of spacecraft equipment and payloads on orbit. At liftoff and for the next several seconds, the intense sound generated by the propulsion system exerts significant acoustic pressure on the entire vehicle. This pressure induces vibration, externally and internally, in the space vehicle structures. In addition, the vehicle experiences intense vibrations generated by engine ignitions, steady-state operation, and engine shutdowns as well as sudden transients or "shocks" generated by solid rocket motor jettison, separation of stages and fairings, and on-orbit deployments of payloads. Space vehicles will also experience wide fluctuations in temperature from the time they leave the launch pad to the time they settle into orbit. Both individually and in combination, the mechanical environments of pressure, vibration, shock, and thermal gradients impose design requirements on many structural components. Ensuring the survivability of the delicate hardware poses challenges that can be met only by extensive preflight tests encompassing acoustic, shock, vibration, and thermal environments. The [redacted] Payload Users Guide has well defined environmental parameters that will be used to establish the design and test criteria.

2.3.1.1.1 Vibration & Acoustics

Environmental testing is performed at varying magnitudes and durations to verify the design of space systems and to screen flight hardware for quality of workmanship. The first step in this process is the definition of the maximum expected environments during launch and on-orbit operation. These environments are then flowed down from the space vehicle level to the various subsystems and components for use as design requirements and, later, as test requirements. Defining and testing to the maximum predicted environments ensures that space systems are designed to withstand the rigors of flight.

Acoustic Noise: The acoustic test levels for the Eclipse space vehicle or subsystem are derived from measurement of data on the Gemini structures on past flights and ground tests. SpaceOps is using this data to predict the test levels in the early stages of the program and in advance of the acoustic test. This provides the program early awareness of the structural acoustic requirements for component design so that any deficiency can be addressed prior to the actual tests. If sufficient data are not available in the database, analytical tools such as statistical energy analysis for frequencies above 100 hertz and finite-element and boundary-element methods for frequencies below 100 hertz are sometimes used to derive test levels. The predicted acoustic environment is adjusted using statistical methods to derive a maximum predicted flight environment. Margin is added to ensure that the hardware is sufficiently robust and to account for analytical uncertainties in the derivation of the environment and design of the hardware. A typical qualification margin is 6 decibels, or four times the energy of the maximum predicted environment. The test lasts at least 1 minute to establish a duration margin of four times the exposure in flight. Additional test time may be accumulated. Hardware that is susceptible to the acoustic-pressure loading are items with large surfaces and low mass density such as composite material solar arrays and antenna reflectors.

Vibration: As the launch vehicle lifts off from the stand and throughout powered flight, the vibration caused by the operating engines excites the vehicle and spacecraft structure. Additional vibration is caused by the fluctuating acoustic pressure experienced during liftoff, transonic flight, and the maximum-dynamic-pressure phase of flight.

Vibration testing helps demonstrate that hardware can withstand these conditions. Random vibration tests are conducted on an electrodynamic vibration machine or "shaker," which consists of a mounting table for the test item rigidly attached to a drive-coil armature.

Most spacecraft vibration tests use response-limiting major-appendage accelerations to reduce input at discrete frequencies so as not to cause unrealistic failures. For test structures that exhibit distinct, lightly damped resonances on a shaker, force limiting is used in conjunction with input vibration to control the shaker. In the force-limiting approach, transducers that measure the input force are mounted between the test item and the shaker.

Shock: Stage, fairing, and vehicle separations are often accomplished by means of pyrotechnic devices such as explosive bolts, separation nuts, bolt cutters, expanding-tube separation systems, clamp bands, ordnance thrusters, and pressurized bellows. When activated, these devices produce powerful shocks that can damage equipment and structures. The characteristics of these shocks depend on the particular separation mechanism, but the energy spectrum is usually concentrated at or above 500 hertz and is measured in a frequency range of 100 to 10,000 hertz. A typical shock response spectrum plot is used to gauge the damage potential of a given separation event.

Separations or deployments generate brief impulsive loads even if no pyrotechnic devices are used. Non-explosive initiators may produce significant shock levels simply through the release of structural strain. Experience has shown that shock can induce a hard or intermittent failure or exacerbate a latent defect. Commonly encountered hardware failures include relay transfer, cracking of parts, dislodging of contaminants, and cracking of solder at circuit-board interfaces. Shock analysis includes consideration of the source amplitudes, durations, transmission paths, path materials, and path discontinuities. Development tests employ an accurate replica of the flight structure with all significant constituents simulated. Deployed hardware is forced to physically separate at least a small amount to provide realistic shock transmission paths. When practical, a shock-producing event is repeated several times to permit meaningful statistical

evaluation of the resulting data. Qualification margins at the unit level are typically 6 decibels on amplitude and twice the number of flight activations. At the system level, it is generally impractical to impose an amplitude qualification margin; however, a margin of two or three activations is imposed.

2.3.1.1.2 Thermal/Vacuum

Launch vehicles and spacecraft must endure a wide range of temperatures associated with liftoff and ascent through the atmosphere, direct impingement of solar radiation, and travel through the extreme temperatures of space. The thermal environment is generally considered the most stressful operating environment for hardware in terms of fatigue, and it has a direct bearing on unit reliability. For example, the use of materials with differing coefficients of thermal expansion has resulted in unsuccessful deployments of mechanical assemblies and payloads. Outgassing increases significantly with temperature, and the resulting contaminants will more readily adhere and chemically bond to colder surfaces. Electronic parts are especially sensitive to the thermal conditions and are subject to problems such as cracks, delamination, bond defects, discoloration, performance drift, coating damage, and solder-joint failure.

Thermal testing is used to screen out components with physical flaws and demonstrate that a device can activate and operate in extreme and changing temperatures. The four most common thermal tests are thermal cycling, thermal vacuum testing, thermal balance testing, and burn-in testing. Thermal cycling subjects the test article to a number of cycles at hot and cold temperatures in an ambient-air or gaseous-nitrogen environment; convection enables relatively rapid cycling between hot and cold levels. Thermal vacuum testing does the same thing, but in a vacuum chamber; cycles are slower, but the method provides the most realistic simulation of flight conditions. In thermal balance testing, also conducted in vacuum, dedicated test phases that simulate flight conditions are used to obtain steady-state temperature data that are then compared to model predictions. This allows verification of the thermal control subsystem and gathering of data for correlation with thermal analytic models. Burn-in tests are typically part of thermal cycle tests; additional test time is allotted, and the item is made to operate while the temperature is cycled or held at an elevated level.

All COTS items will require exposure to vacuum environments. In particular, major off-the-shelf items such as the desired Mk. 12H ejection seat, although functionally acceptable, will require careful examination for outgassing elements and then be exposed to a vacuum to ensure proper operation both before and after on-orbit operations.

2.3.1.1.3 Electrical

The completed Eclipse spacecraft will require integrated ground testing. While each subsystem will consist of equipment that is space rated, and individually tested for Power, EMI, ESD, and HERF, if required. The integrated system will require testing to the same requirements. System power will require testing to verify all subsystems are operating as expected, with no unusual power consumption. The power system will also require testing to verify that no component of the operational system is creating an unexpected electrical noise situation.

EMI testing will monitor the spacecraft, externally, to detect any undesirable noise emissions. In addition, the spacecraft will be tested to verify that all systems will not be affected by externally induced electromagnetic interference.

ESD testing will verify the ability of the spacecraft to withstand potential ESD discharges, both internally and externally. While most external ESD scenarios will be tested by HERF, interior

testing will be critical. However, the potential for internal ESD damage may be mitigated by the appropriate control of internal vehicle humidity levels.

HERF testing will be required for vehicle safety. During launch phase, there is always the chance of a lightning strike upon the spacecraft, and verification of the integrated system components ability to withstand high energy events is required. In addition to lightning, HERF testing will verify resistance to ESD and potential high energy electromagnetic events. While there is no indication of prior manned interference, the possibility exists that some unfriendly act may attempt to cause damage to a spacecraft utilizing directed high energy electromagnetic radiation. Capabilities exist for test requirements in both commercial test facilities and NASA based facilities.

2.3.1.1.4 Guidance, Navigation and Control System (GNC)

Ground testing of the GNC system will require the use of ground based simulation and test. Hardware in the loop platforms exist to test all GNC systems for correct stimulus/response criteria. While the GNC subsystem will be the most complex single system on board the spacecraft, test procedures will be automated to allow for exhaustive test of the system, both during system customization/development and during individual spacecraft test during assembly and checkout. The ability to stimulate the GNC system with any possible situation will allow autonomous control validation, as well as provide training scenarios for selected crew members. Implementation of this test capability will be a priority at the start of the integration phase of the Commercial Crew Integrated Capability program.

2.3.1.2 Range Testing

Range testing will be performed on all newly developed subsystems or modifications of the Gemini subsystems.

Ejection Seats: Martin-Baker developed the Mk.12 seat to provide a cost effective escape system with an improved low speed and adverse attitude recovery capability. The Mk.12 seat introduces a significant improvement in seat performance over the speed range zero to 625 knots and from ground level to the service ceiling of the aircraft or the limit of the pilot's protective equipment. The seat has the capability of sensing airspeed and adjusting its mode of operation accordingly. These characteristics will be determined analytically and verified through rocket assisted sled tests which will be performed to ensure the ejection seats perform reliably while mounted in the Eclipse under all anticipated launch, reentry, and recovery angles. The tests will include induced angles in all three axes to ensure safe crew egress.

Landing Demonstration and Drop Test**:** The final selection of the recovery system, either the paraglider or the parasail will be determined early in the Eclipse development. In order to establish the testing remaining to be conducted; conduct the landing tests and to validate confirm the data collected during the Gemini program. The deployment of the paraglider will also be validated on the rocket sled testing planned for the program.

2.3.1.3 Flight Test

Flight testing of the Eclipse B configuration will include an instrumented seat pallet much like the one used during the Gemini program to gather data from all four positions in the crew Cabin Module.

2.3.2 Training

SpaceOps is in the process of completing an SAA with Marshall Space Flight Center which will include tasking for Johnson Space Center for Mission Operations Planning, Mission Operations Support and Training.

2.3.2.1 Mission Operations

The Johnson Space Center (JSC) offers capabilities in mission planning, crew training, flight product generation, and real-time operational support in the Mission Control Center.
SpaceOps and JSC will work together to develop the plans, procedures, checklists and training to complete a successful mission.
Plan:

- Spaceflight Timelines
- Orbital Flight and Trajectories
- Spacecraft Monitoring System Parameters
- Spacecraft Design for Operations Considerations

Train:

- Flight Crews for Spacewalks
- Flight Crews for Spacecraft Systems
- Spaceflight Simulation Systems

Fly:

- Real-Time Spaceflight Anomaly Risk Analysis
- Spacecraft Control Systems for Mission Control
- Worldwide Spacecraft Communications Network
- Flight Procedures and Check List
-

Facilities:

- Neutral Buoyancy Laboratory
- Spacecraft Motion-Based Simulator
- Spacecraft Fixed-Based Simulator
- Single Systems Trainer Simulators
- Space Station Training Facility
- Space Vehicle Mockup Facility
- Mission Control Center Facilities

2.3.2.2 Flight Crew

The training is geared to the special conditions and environments astronauts will be confronted with during launch, in space, and during landing. All phases of the flight must be taken into account during training to ensure safety to, and functionality of the astronauts, as well as to ensure a successful completion of the mission.
Candidate Selection: To be selected as a pilot astronaut candidate an applicant must meet a number of basic qualification requirements. A bachelor's degree in engineering, biological science, physical science or mathematics is required. A graduate degree is desired, although not essential. The applicant must have had at least l,000 hours flying time in jet aircraft. Experience as a test pilot is desirable, but not required. All applicants -- pilots and missions specialists -- must be citizens of the United States.

Physically, an applicant must pass a strict physical examination and have a distant visual acuity no greater than 20/50 uncorrected, correctable to 20/20. Blood pressure, while sitting, must be no greater than 140 over 90. An applicant also must also be between 64" to 76" tall.
Mission Specialist Astronauts: Mission specialist astronauts, working closely with the commander and pilot, are responsible for coordinating on board operations involving crew activity planning, use and monitoring of the consumables, and conducting experiment and payload activities. They are required to have a detailed knowledge of systems and the "operational characteristics, mission requirements and objectives and supporting systems for each of the events to be conducted on the assigned missions." Mission specialists perform on-board experiments, spacewalks and payload handling functions involving the RMS arm.
The basic physical qualifications for selection as a mission specialist astronaut are the same as those for pilots, except that uncorrected visual acuity can be as high as 20/100, correctable to 20/20. A candidate's height can range from 60" to 76".
Academically, applicants must have a bachelor's degree in engineering, biological science, physical science or mathematics plus at least 3 years of related and progressively responsible professional experience.
Payload Specialist Astronaut: Payload specialists for major non-NASA payloads or experiments are selected by the sponsoring organization. Payload specialists do not have to be U.S. citizens. However, they must meet strict NASA health and physical fitness standards.
In addition to intensive training for a specific mission assignment at a company plant, a university or government agency, the payload specialist also must take a comprehensive flight training course to become familiar with Shuttle systems, payload support equipment, crew operations, housekeeping techniques and emergency procedures. This training is conducted at JSC and other locations, as required. Payload specialist training may begin as much as 2 years before a flight.
Initial training for new candidates consists of a series of short courses in aircraft safety, including instruction in ejection, parachute and survival to prepare them in the event their aircraft is disabled and they have to eject or make an emergency landing. Learning to function in a weightless environment is simulated in aircraft and in the "neutral buoyancy" water tank at JSC. Other major operations training facilities at JSC include the Computer-Aided Instructional Trainer (CAIT) in Building 4, which fills the gap between textbook lessons and more complex trainers and simulators; the Crew Software Trainer (CST) used to demonstrate spacecraft software capabilities.
Crew training becomes more structured and is directed by a training management team. These specialized teams are responsible for directing the remaining advanced training needed for a specific flight. This includes what is described as "stand-alone training and flight-specific integrated and joint integrated training." It involves carefully developed scripts and scenarios for the mission. This intensive training is designed to permit the crew to operate as a closely integrated team, performing normal flight operations according to a flight timeline. The astronauts selected for the manned flight will observe the Eclipse assembly and integration; participate in the system functional testing, altitude testing, etc. to ensure they have a comprehensive understanding of the Eclipse.

2.3.2.1 Recovery

A dry land recovery significantly reduces the cost and complexity of an Eclipse recovery. During the developmental paraglider testing and demonstrations phase the recovery team will practice the techniques of recovering a returned Eclipse, moving and transporting the vehicle to a

low-boy truck. SpaceOps will utilize an experienced government approved professional trucking company to develop pallets, tie down points, straps, and cross-country driving routes, define maximum driving velocities, and packing requirements.

2.4 Performance Milestones

Table 2.4-1 shows the performance milestones scheduled for the execution of the Eclipse Program. These dates are consistent with the requirements of the CCiCap Program.

Table 2.4-1 Eclipse Program Performance Milestones

Performance Milestones					
	Contract award	1 Aug. 2012			
Eclipse 1	PDR 3 Months after contract award	1 Nov. 2012	Eclipse 3	PDR 9 Months after contract award	1 May. 2013
	CDR 6 months after contract award	1 Feb. 2013		CDR 12 months after contract award	1 Aug. 2013
	IRR 9 months after contract award	1 May. 2013		IRR 15 months after contract award	1 Oct. 2013
	Ejection seat testing completed	1 Jun. 2013		TRR 18 months after contract award	1 Jan. 2014
	Reentry paraglider testing complete	15 Jun. 2013		Complete System Functional Testing	1 Mar. 2014
	TRR 12 months after contract award	1 Aug. 2013		Completed astronaut training	1 Mar. 2014
	Complete System Functional Testing	1 Oct. 2013		Successful Launch Operations	1 May.2014
	Successful Launch Operations	1 Nov. 2013		Successful on-orbit operations	2 May. 2014
	Successful on-orbit operations	1 Nov. 2013		Successful ISS docking	3 May. 2014
	Successful Reentry	2 Nov. 2013		Successful recovery operations	6 May. 2014
	Successful recovery operations	2 Nov. 2013			
Eclipse 2	PDR 6 Months after contract award	1 Feb. 2013			
	CDR 9 months after contract award	1 May. 2013			
	Instrumented Seat pallets complete	15 Jul. 2013			
	IRR 12 months after contract award	1 Aug. 2013			
	TRR 15 months after contract award	1 Oct. 2013			
	Complete System Functional Testing	26 Nov.2013			
	Successful Launch Operations	1 Feb. 2014			
	Successful on-orbit operations	2 Feb. 2014			
	Successful ISS fly-by	3 Feb. 2014			
	Successful recovery operations	4 Feb. 2014			

3 Business Information

3.1 Business Viability

3.1.1 Space Operations, Inc.

Space Operations, Inc. (SpaceOps) is a locally (Huntsville, Alabama) owned and operated company that has been developed to provide a cost effective method to acquire products and services from a specialized group of companies whose combined capabilities and talents can provide reliable, and technologically sophisticated spacecraft hardware focused on fulfilling the needs of carrying flight crews and vital cargo into space. The Corporation will manage the acquisition, integration, operation and refurbishment of spacecraft vehicles for use by the U.S. Government, major prime contractors, satellite manufacturers, and other organizations or individuals with a requirement or a desire for space flight to Low Earth Orbit (LEO).

SpaceOps Corporate Headquarters is located in the Huntsville, Alabama area [redacted]. Operating in the North Alabama region provides direct access to our team of subcontractors and Marshall Space Flight Center for rapid development and sustainment of this proposed spaceflight program.

By setting up a management organization, SpaceOps will be able to acquire major components that are manufactured by specialized companies and provide service to end users with no cost stacking and at very low rates.

3.1.2 The SpaceOps Team

The SpaceOps Team is composed of four primary members/companies each having particular expertise in one or more of the essential aspects of development and/or mission operations support. Additional members will likely be added to the team to augment overall capability and to resolve development requirements as they are identified. At this point, SpaceOps is working to establish a Space Act Agreement with MSFC. The development of the Eclipse vehicle and mission operations will require that SpaceOps obtains assistance with MSFC, JSC, KSC, and GSFC; however, it is our understanding that we may be able to obtain these services through the single SAA with MSFC. In the event that this is not possible, then additional SAA's will be obtained for each of the needed NASA centers. Depending on our ability to obtain the needed services from the NASA centers, we may find it necessary to add to our team to fill essential gaps; however, this team provides experience from space programs all the way back to the days of Gemini and Apollo.

[redacted] will provide the primary propulsion unit (i.e., the [redacted] booster) and the necessary launch facility integration capability. Clearly, their efforts over the past decade have resulted in the development of a two-stage boost capability that provides a natural solution for this program.

[redacted] will provide an engineering and manufacturing capability to develop, integrate, and produce the Eclipse spacecraft. Engineering and testing capabilities will be provided in the initial phases of this program to ensure adequate planning and execution of all SSMA elements and milestones. Manufacturing expertise will provide mock-ups and assist in the fabrication of the test articles.

[redacted] will provide engineering and manufacturing expertise to develop the Reaction Control System (RCS) and to assist [redacted] in test article fabrication and the overall system integration. DI will also provide a significant amount of test design and support for the RCS. [redacted] will provide the avionics development along with the necessary software to handle on-orbit GNC and communications.

3.2 Organizational Structure

SpaceOps is organized as shown in the following diagram. As indicated, the company is currently composed of essential personnel and a team of companies willing to invest in the project along the lines of their focused capabilities. The positions indicated in the lighter shades are not yet required for the progression of the program but, will be filled once a need arises.

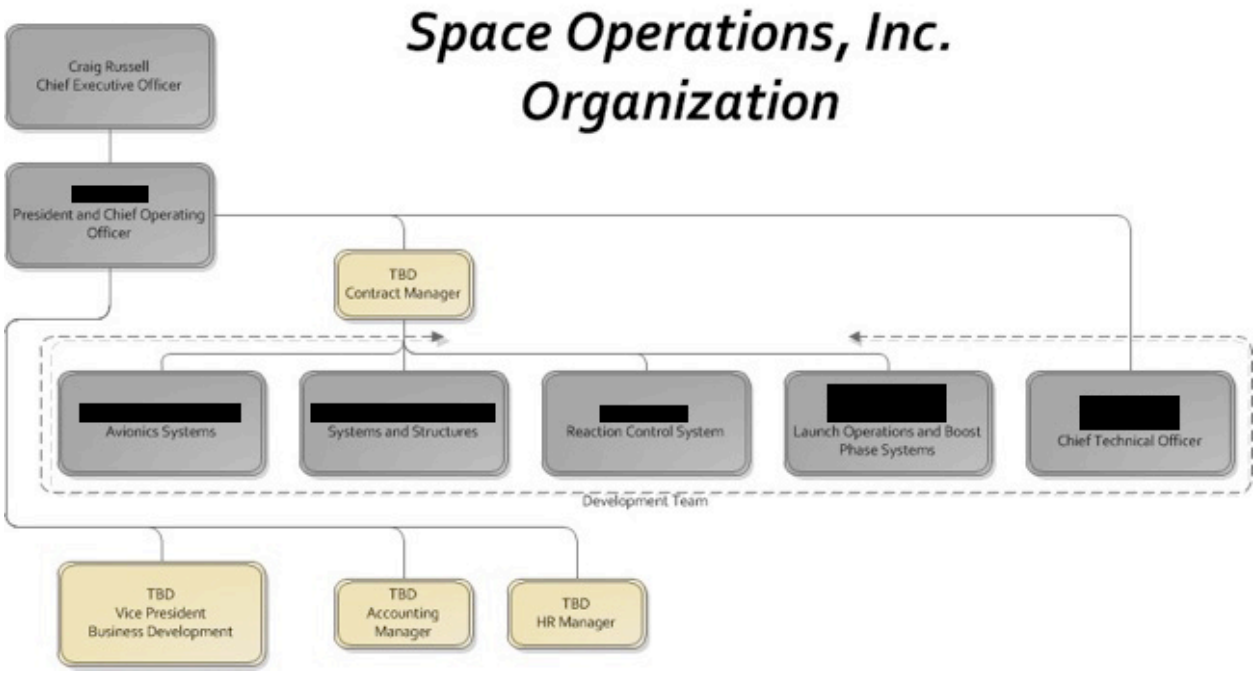

Figure 3.0-1 Space Operations, Inc. Organization

3.3 Finance

Financing this significant of a program is clearly beyond the scope of most businesses; however, many smaller businesses invest in themselves at level relative to their ability to afford and based on the potential for a return on investment. Therefore, we have taken a team approach to spread the requirements of investment into the specific business sectors and looked to these companies to contribute time and talent in accordance with their fiscal business plans. Based on the construction of the SpaceOps team, the Eclipse development will require significant enhancement in order to meet time lines that are interesting to potential investors.

3.3.1 Target Market

SpaceOps will place special emphasis on marketing the U.S. Government and its agencies, in providing spacecraft services for transporting astronauts to and from the International Space Station, replenishment of space station supplies, satellite deployment and servicing, and debris

de-orbiting. Marketing efforts will also be placed on supporting manufacturers of satellites and organizations and individuals having a requirement or a desire to travel in space.

A major objective of SpaceOps is to propel the company into a prominent market position with steady growth over a relatively short period of time. The current industry demand for spacecraft access combined with the growing interest in the next-generation infrastructure, places SpaceOps in the right industry at the right time.

The relevant time frame is to begin a 30 month development, integration, and test program with initial launches in early to mid-2014. Then provide launch services until the next-generation of spacecraft with their larger crew capabilities can come on-line. Continued service can be provided for lower cost and special mission options.

The company will secure contracts one year in advance of launch date for crew seating and cargo reservations. This time period will be utilized for mission planning, crew training for assigned tasks and preparing the spacecraft for the launch mode.

The plan to position the company is well underway. The company has:

- Identified the major users having in-orbit space travel requirements.
- Organized the users into groups and identified their common needs.
- Conducted meetings with partnering contractors to discuss major milestones and schedule completion dates.
- Conducted meetings with former astronauts on crew requirements and potential customer spacecraft users.

3.4 Resources

The key Government resources needed for this program are listed in the following table.

Table 3.4-1 Key Government Resources

Resource	Location	Attachment
Hypersonic Wind Tunnel	Marshall Space Flight Center	Pending SAA
Large Scale Vibration Table	Marshall Space Flight Center	Pending SAA
Large Scale EMI Chamber	Marshall Space Flight Center	Pending SAA
Thermal/Vacuum Chamber	Marshall Space Flight Center	Pending SAA
Hardware Safety Assessment	Marshall Space Flight Center	Pending SAA
TDRSS Link Assistance	Goddard Space Flight Center	Pending SAA
Communications Assessment	Goddard Space Flight Center	Pending SAA
Flight Dynamics Assessment	Jet Propulsion Lab/Ames	Pending SAA
Flight Testing	Jet Propulsion Lab/Ames	Pending SAA
Sled Testing – Ejection Seats	Holloman High Speed Test Track	No Contracted
Sled Testing – Parasail Deployment	Holloman High Speed Test Track	No Contracted
Human Factors Assessment	Johnson Space Flight Center	Pending SAA
Astronaut Training Program Assessment	Johnson Space Flight Center	Pending SAA
Astronaut Training	Johnson Space Flight Center	Pending SAA

Resource	Location	Attachment
Launch Operations Procedure Development	Kennedy Space Flight Center	Pending SAA
Launch Operations Assistance	Kennedy Space Flight Center	Pending SAA

3.5 Compliance

Space Operations, Inc. and all team members are wholly owned by U.S. interests and do not employ foreign nationals as a result of the classified nature of their work.

[redacted] is a long standing supplier of aviation products and upgrades, engineering solutions, manufactured products, and technical services to the U.S. military since 1999.

[redacted] is a long standing supplier of missile products, engineering solutions, manufactured products, and technical services to the U.S. military and NASA for over two decades.

[redacted] is a [redacted] small business with a history of providing avionics products and engineering services to the U.S. military and commercial corporations for space flight systems.

All of these companies do and will comply with the following U.S. Laws, Regulations, and Policies:

- Iran, North Korea, and Syria Nonproliferation Act (P.L. 106-178, amended by P.L. 107-228, P.L. 109-112, P.L. 110-329)
- U.S. National Space Policy, June 2010
- Commercial Space Act of 1998 (P.L. 105-303, 51 U.S.C. 50101 et seq.)
- Commercial Space Launch Act (51 U.S.C. Chapter 509)

Appendix 1 Draft Space Act Agreement

Appendix 2 Eclipse Program Milestones