Concept Study Report for the New Millennium Program Space Technology 7 Opportunity

Disturbance Reduction System

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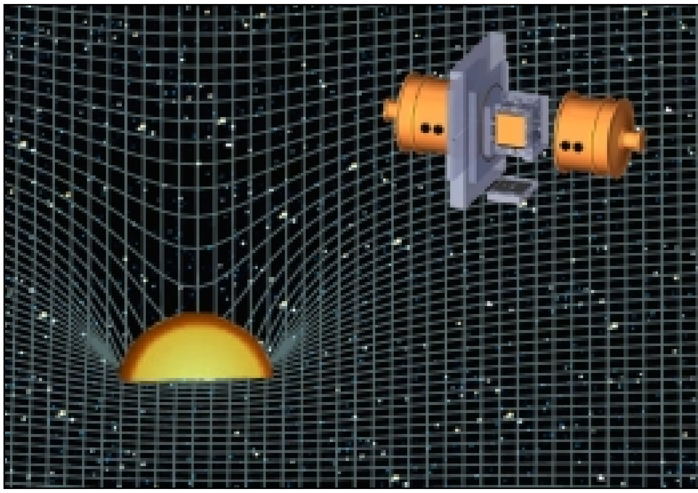
Jet Propulsion Laboratory, California Institute of Technology



Summary: The Disturbance Reduction System (DRS) will validate technologies for precision spacecraft control. The capability of following a trajectory determined by purely gravitational forces will enable missions for tests of general relativity, determination of planetary gravity fields, and space observatories for gravitational waves. The capability of controlling spacecraft position to a fraction of a wavelength of light will enable new types of separated-spacecraft interferometer missions. The DRS consists of an instrument package and a set of miniature ion thrusters. The instrument will measure the position of the spacecraft with respect to an internal free-floating test mass. The thrusters will be used to control the spacecraft position and attitude with high precision. The DRS will be attached to either the European Space Agency's SMART-2 spacecraft or the NASA StarLight spacecraft, both of which will be launched in mid-2006. While in space, far from the Earth and its perturbing environment, the DRS will characterize the limits on forces affecting freely floating test masses and test the performance of micro ion thrusters, with improvements of several orders of magnitude over previously demonstrated technology. DRS validation will take three months of operations in orbit, and be completed by end of FY 2007.

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Executive Summary

The Disturbance Reduction System (DRS) is designed to validate system-level technology required for two types of future missions: measurements of planetary gravity and of cosmic gravitational waves, and precision formation-flying interferometers. The validated technology will feed directly into the LISA gravitational-wave observatory, the MAXIM X-ray interferometer mission, and several other missions within the NASA roadmap. The DRS is based on the concept of a freely floating test mass contained within a spacecraft that shields the test mass from external forces. The test mass will ideally follow a trajectory determined only by the local gravitational field. The spacecraft position must be continuously adjusted to stay centered about the test mass, essentially flying in formation with the test mass.

Any departure of the test mass motion from a purely gravitational trajectory is characterized as acceleration noise, resulting from unwanted forces acting on the test mass. The DRS goal is to demonstrate a level of acceleration noise more than 4 orders of magnitude lower than previously demonstrated in space. To achieve this goal, the spacecraft shields the test mass from external forces. There are also forces imposed on the test mass by the spacecraft, including magnetic forces and gravitational forces due to the mass of the spacecraft components. These forces must also be reduced to meet the DRS acceleration goal. This is achieved by controlling the spacecraft position to follow the test mass within 10 nanometers, a distance that is only a few percent of the wavelength of light.

Spacecraft position control with nanometer precision is of interest beyond gravity missions. Separated-spacecraft interferometers require the control of the positions of mirrors with nanometer precision, so that light collected from separate spacecraft can be coherently combined. These interferometers will have the resolution of a telescope with size comparable to the separation between spacecraft, which can be hundreds of meters. In the visible and infrared, the control can be done in stages, with relatively coarse control of the spacecraft and fine control of the mirrors relative to the spacecraft. Validation of this approach is planned by the NASA StarLight project, which will demonstrate spacecraft position control at the centimeter level. The capability of controlling spacecraft positions to the nanometer level would simplify the optics for other visible and infrared interferometry missions, and this option is being studied for some future projects. For proposed separated-spacecraft X-ray interferometers, nanometer spacecraft positioning is mission-enabling.

The new component technologies for DRS are Gravitational Reference Sensors and Microthrusters. The Gravitational Reference Sensors include a test mass within a reference housing. The position of the test mass is measured with respect to the reference housing by changes in capacitance between electrodes on the reference housing, using the test mass as a virtual ground plane. The spacecraft position is controlled using Microthrusters, which are miniature ion engines.

The thruster performance will be validated by using the capacitance measurements of the test mass to find the acceleration of the spacecraft due to the thrusters, as the thrusters are controlled through their full range. The spacecraft position will be controlled by software that fires the thrusters in response to changes in the capacitive position signals, holding the test mass fixed relative to the spacecraft. This control system will be verified by suppression of variation in the capacitive position signals to the nanometer level.

To measure the residual acceleration of the test mass, its position will be measured with respect to a second test mass in the same payload, using a laser interferometer. The interferometer will have sufficient precision to verify that the relative acceleration of the proof masses is below the required acceleration noise level.

The DRS will consist of an instrument package and a set of thrusters, which will be attached to a suitable spacecraft. The DRS must be operated in an environment far from the Earth's gravity field and in a relatively low radiation environment. In a near-Earth orbit, the variations of the local gravity field would cause differential accelerations on the two test masses, and it would not be possible to determine the instrument performance level. In a high-radiation environment, such as the radiation belts that are crossed in a geosynchronous transfer orbit, the test masses would become charged from impacts from high energy particles. Ideally the DRS would be launched on an Earth-escape trajectory.

The European Space Agency has offered to include the DRS on their SMART-2 spacecraft, which is planned for launch in 2006. Alternatively, the NASA StarLight project, currently planned for launch in mid-2006, can accommodate the DRS system. Both of these projects are technology demonstration missions launching to Earth-escape orbit, ideal for the DRS tests. On the StarLight project, the DRS technology would enhance the StarLight technology goals by enabling tests of separated-spacecraft interferometry using nanometer spacecraft positioning.

Specific DRS tests are planned to validate performance of the microthruster and gravitational reference sensor technologies. The gravitational reference sensors can operate as accelerometers to allow measurement of thruster performance. The initial DRS test will be to measure the thrust of each microthruster as each is commanded to go through a preset range. The acceleration data will validate the thrust values and thrust noise levels. The thruster performance data will be sent to Earth for evaluation, and will be used to make final adjustments in control-law tables for the spacecraft position control. After uplink of the final control tables, the DRS will perform an initial experiment to keep the spacecraft centered on one test mass. During this initial test, the second test mass will be used as an accelerometer to check the levels of noise on the spacecraft. The capacitive measurements of the position of the freely floating test mass will provide validation of the spacecraft position control. Since the capacitive position measurements are used within the control loop, it is desirable to confirm the control with an independent measurement. The laser interferometer will provide independent confirmation of the spacecraft position control in one degree of freedom. Following this test, a test will be performed with both test masses freely floating. Bias forces may need to be applied to keep the two test masses from drifting apart while leaving both free at the frequencies of interest. The laser interferometer will measure changes in the distance between the test masses to validate the level of acceleration noise.

Following the tests to establish the lowest levels of acceleration noise, tests will be performed to introduce perturbations in the spacecraft environment, to validate models of how acceleration noise is related to environmental variables. Perturbations will be made by applying a periodic temperature variation; applying a varying magnetic field; applying perturbations to the spacecraft position control; and increasing the charge on the test mass. These tests will allow extrapolation of gravitational reference sensor performance to lower environmental noise.

With the successful completion of these tests, the DRS technologies will have been validated to levels that would allow them to be directly included in some missions, such as separated-spacecraft interferometers and Earth-gravity field measurements. For missions with the most stringent performance requirements, such as the LISA space gravitational-wave observatory, some refinements to the technology may be needed. The model-characterization test results will guide these design refinements.

For launch on either SMART-2 or StarLight, the DRS will be delivered for spacecraft integration in mid-2005. The DRS development schedule is based on Formulation Refinement starting in February 2002 and ending in April 2003. DRS System Requirements Review will be held in May 2002. A draft DRS system requirements document has been written and discussed with the StarLight and SMART-2 project teams. During Formulation Refinement, breadboard models of all DRS components will be developed. The breadboard models will be used to prove technology readiness at TRL 5 and test compatibility between the DRS subsystems. DRS Preliminary Design Review and Confirmation Assessment Review will be held near the end of Formulation Refinement in February 2003 and April 2003 respectively. During the first half of the Implementation Phase, high-fidelity brassboard models will be built to finalize parts selection and to test the mechanical design of the instrument assembly and thrusters. The Critical Design Review will be held December 2003, followed by the start of flight system development in April 2004. DRS subsystems will be completed in January 2005 and shipped to JPL for integration and test.

The DRS subsystems will be developed as protoflight models and tested to suitable environmental levels prior to shipment. During Integration and Test, the DRS subsystems will undergo final assembly and a vibration test will be performed at the system level. The DRS will be shipped to the spacecraft integration site.

The DRS system will be powered off while other technologies are tested on either the StarLight or SMART-2 project. Three months of DRS operations will take place several months after launch. Individual DRS tests would require two to three days of operation, and tests could be interspersed with other project tests.

The DRS lends itself to public outreach through its goal of testing the curvature of space-time, tying in to public fascination with concepts of curved space and black holes. The DRS Education and Public Outreach (E/PO) will tie these concepts to existing programs, including the New Millennium Program's Space Place web site, and to E/PO activities for related projects and programs, including the Navigator program, aiming at detection of planets about other stars; the Gravity-Probe B project aiming at testing properties of curved space; and activities at future user missions including LISA, TPF, and MAXIM as those project E/PO offices are developed.

The DRS will be managed by the Jet Propulsion Laboratory as a project within the Astronomy and Physics Directorate. The DRS Project Manager will be responsible for ensuring mission success and performance, cost, and schedule. The Gravitational Reference Sensor technology will be provided by Stanford University under contract to JPL. The Microthruster technology will be provided by Busek Co. Inc under contract to JPL. The software for control of the spacecraft position will be developed by Goddard Space Flight Center. The support structure and diagnostic interferometer will be developed at JPL.

The DRS project cost is $60.5 M, with $59.2 M from the New Millennium Program and $1.3 M in civil servant salaries and benefits from the Goddard Space Flight Center. The project cost includes 30% cost reserves during formulation refinement and implementation, and 15% cost reserves during operations.

1 TECHNICAL APPROACH

1.1 Objectives

1.1.1 Objectives Overview

The Disturbance Reduction System (DRS) is designed to demonstrate a) improved gravitational reference sensors and b) nanometer spacecraft position control. These two technology areas are closely coupled. Gravitational reference sensors are used in combination with "drag-free" spacecraft control in order to provide a spacecraft trajectory that is affected only by external gravitational forces, to the extent possible. Precision spacecraft control is needed to reduce noise forces from the spacecraft on the test mass. Precision spacecraft control is also of interest as a distinct technology area, for use with separated-spacecraft interferometry as well as for improved drag-free control.

A gravitational reference sensor consists of a freely floating test mass within a housing, along with a measurement system for determining the position of the test mass with respect to the housing. In order to validate the performance of a gravitational sensor, the motion of its test mass must be compared to a reference trajectory. For DRS this reference trajectory is provided by a second freely floating test mass in a second gravitational reference sensor in the DRS payload. The time variation of the separation of the two test masses will determine whether external forces have been reduced to the required level.

Nanometer spacecraft position control is achieved by precisely measuring the position of a spacecraft with respect to a reference (which could be another spacecraft) and controlling the spacecraft position to the desired value. For DRS, the spacecraft position measurement is provided with respect to a freely floating test mass within a gravitational reference sensor. The spacecraft position is adjusted by microthrusters, which are miniature ion engines. The precision spacecraft control is validated by independent measurements of the spacecraft position with respect to a second freely floating test mass.

Figure 1 illustrates the DRS mission concept. The test masses are intended to follow purely gravitational trajectories. In the absence of a shielding spacecraft, the test mass trajectories would be affected by solar radiation pressure in addition to other forces. The spacecraft shields the test masses from external forces. In the absence of a control system, solar radiation pressure would tend to push towards the test masses. To prevent this, the position of the spacecraft is measured with respect to the test mass. As the solar pressure forces the spacecraft towards the test mass, the position changes is sensed and used to control thrusters to adjust the position of the spacecraft back to the desired position. In the steady state the thrusters fire to exert the same force as the solar radiation pressure (plus other forces). The first use of this type of spacecraft position control was in Earth orbit, where the primary nongravitational force was atmospheric drag, leading to the terminology for 'drag-free' control. The DRS will orbit far from Earth, where solar radiation pressure will be the dominant nongravitational force.

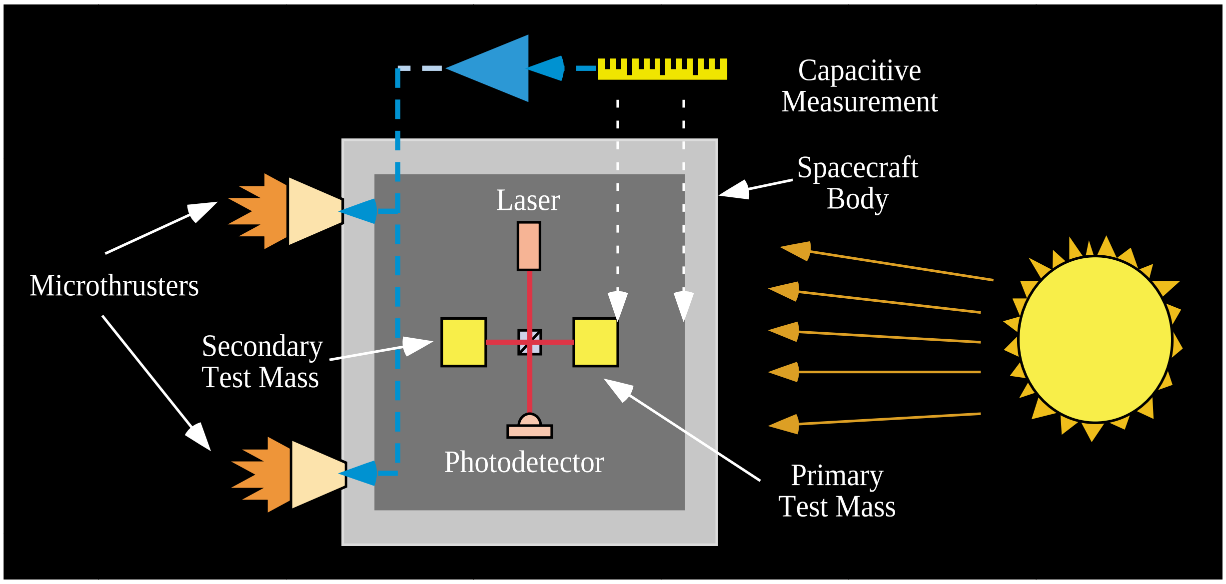

Figure 1 Schematic diagram of the Disturbance Reduction System

Table 1 DRS Level 0 Requirements

Technology area	Goal
Freely floating test mass	Demonstrate capability for spacecraft trajectory free of nongravitational forces
Spacecraft position control	Demonstrate capability for control of spacecraft position to fraction of optical wavelength

Table 2 DRS Level 1 Requirements

Technology	Goal	Requirement	
Freely floating test mass	Follow trajectory with low nongravitational forces, on time scales of seconds to hours	**Floor:**	Demonstrate test-mass acceleration noise $< 3 \times 10^{-13}$ m/s^2/√Hz for frequencies from 10^{-3} Hz to 10^{-2} Hz
		Baseline:	Demonstrate test-mass acceleration noise $< 3 \times 10^{-14}$ m/s^2/√Hz for frequencies from 10^{-3} Hz to 10^{-2} Hz
Freely floating test mass	Follow trajectory with low nongravitational forces, on time scales of seconds to hours	**Baseline:**	Characterize noise models affecting test mass acceleration noise to allow extrapolation to performance in quieter environments
Spacecraft position control	Control spacecraft position with respect to reference	**Floor:**	Demonstrate spacecraft position control to < 10 nm/√Hz for frequencies from 10^{-3} Hz to 10^{-2} Hz
		Baseline:	Demonstrate spacecraft position control to < 1 nm/√Hz for frequencies from 10^{-3} Hz to 10^{-2} Hz

The DRS project requirements are summarized in Tables 1 and 2. Validating the technology to meet these performance requirements will demonstrate an advancement in the performance of the two distinct but related spacecraft system technologies. This will benefit science projects that require improved sensitivity to gravitational forces and/or spacecraft position control (formation flying) at the nanometer level. Space tests of these technologies are essential for validation of the level of performance needed for future missions. These technologies are applicable to identified missions within NASA future project roadmaps of both the Office of Space Science the Office of Earth System Science. Related missions are also included in the Roadmap for Fundamental Physics in Space under the Microgravity Research Division. Thus the DRS is consistent with the overall goals of the New Millennium program to demonstrate new technologies that require demonstration in space and that address a large number of missions in multiple NASA theme areas. As a technology demonstration project, DRS will retire the unknowns (risks) for future mission at a relatively low cost.

Examples of science goals of projects requiring improved sensitivity to gravitational forces are measurement of the time-changing Earth gravitational field (enhancing) and the measurement of low-frequency gravitational waves (enabling). Examples of projects that could be enabled or enhanced by spacecraft position control at the nanometer level are separated-spacecraft interferometers in the visible or infrared portion of the electromagnetic spectrum (enhancing) or in the microwave or X-ray portion of the spectrum (enabling).

1.1.2 Technology Objective Context

1.1.2.1 Acceleration Noise

The performance of gravitational reference sensors is characterized by the level of acceleration noise. The acceleration noise levels desired for future gravitation missions (LISA, LIRE) are a factor of 10^6 lower than the levels already demonstrated in space by the Triad mission[1] and are a factor of 10^5 lower than will be demonstrated by near-term missions (GRACE, GP-B). Figure 2 shows the relevant acceleration noise levels for the various missions. The DRS is designed to demonstrate acceleration noise within a factor of ten of the most stringent mission requirements. This level of performance has been chosen as a compromise between the cost of performing the demonstration and the range over which a reasonable extrapolation in performance can be made. The DRS will include tests to characterize

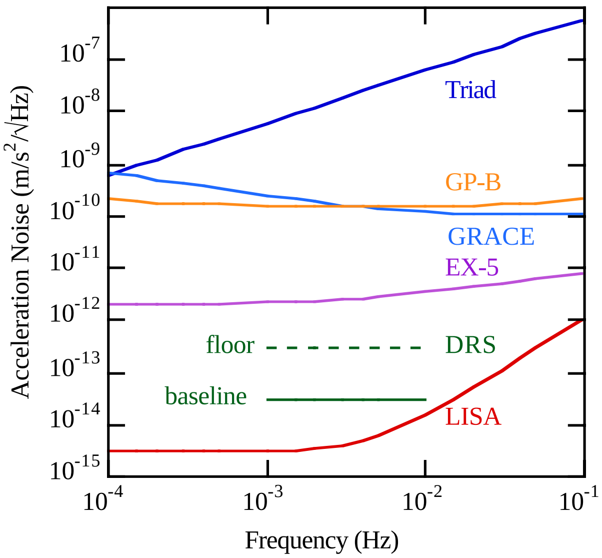

Figure 2 Acceleration noise levels for past and future missions

performance models to validate the required extrapolation.

The DRS performance is expected to be limited not by instrument performance but by perturbations in its environment due to spacecraft properties. Achieving the spacecraft environment necessary to reach the expected instrument performance level might significantly raise the technology demonstration cost. Also, to perform a demonstration for low cost, the DRS is based on two test masses within a single spacecraft. Since the two test masses will tend to drift apart, some control forces will have to be applied to one of the test masses to keep it near the primary test mass. Application of these forces will perturb the controlled test mass and limit the performance. This could be avoided by having the two test masses on separate spacecraft, which would clearly significantly raise the cost.

In order to have a gravitational reference sensor demonstration that will satisfy the needs of the most demanding future missions, the DRS will not only demonstrate a level of performance within a factor of ten of the ultimate goal, but will also perform a series of tests to validate performance models so that the DRS performance can be extrapolated to the expected performance in the expected environment. In these tests the acceleration noise on the test masses will be deliberately increased by applying known increases in system temperature, control forces, spacecraft motion, etc. to verify that the resulting acceleration noise as a function of the perturbation is understood. In this manner the expected performance under lower levels of perturbation can be inferred from the DRS test data.

1.1.2.2 Spacecraft Position Control

For separated-spacecraft interferometry missions, the position of optics on different spacecraft must be controlled to a fraction of a wavelength, in order that the light collected from different spacecraft can be combined coherently. Spacecraft position control is also important for drag-free gravity missions, since interactions between the spacecraft and reference test masses inside them are affected by motion of the spacecraft relative to the test masses. This requirement can be met by control of the spacecraft position combined with control of the position of the optics on the spacecraft using optical positioning components.

For separated-spacecraft interferometry missions observing in the visible and infrared, the optics position control can be done in stages with a relatively coarse position control of the spacecraft and fine control of the optics using optics mounted on movable stages (voice coils or piezo-electric crystals). Validation of this approach is planned by the NASA StarLight project, which will demonstrate spacecraft position control at the centimeter level. The capability of controlling spacecraft positions to the nanometer level would simplify the optics for visible and infrared interferometry missions, and this option is being studied for some future projects. For separated-spacecraft X-ray interferometer projects, for which grazing optics are used, nanometer spacecraft positioning is mission-enabling.

The position control for interferometry mission is based on the relative position of optics on separate spacecraft, with interspacecraft metrology used to make the measurements. In some cases the interspacecraft metrology is more accurate than the needed optics control since in those cases post-processing of the data can use the metrology measurements to achieve improved results.

Spacecraft position control is also needed for future gravity missions such as LISA and EX-5. For these missions the spacecraft position is controlled not with respect to another spacecraft but with respect to freely floating test masses within them The source of the position control requirement is the combination of forces on the test mass due to spacecraft equipment, including electric, magnetic, and gravitational interactions. Spacecraft position changes thus cause changes in the force on the test mass, which cause noise in the gravity measurements. Spacecraft position control is used to reduce this force noise. The Triad spacecraft position was controlled to ± 0.9 mm of the freely floating test mass within it. The GP-B spacecraft position will be controlled to 100 nm with respect to a freely floating test mass (used as a gyroscope) by throttling helium gas from a liquid-helium reservoir.[2]

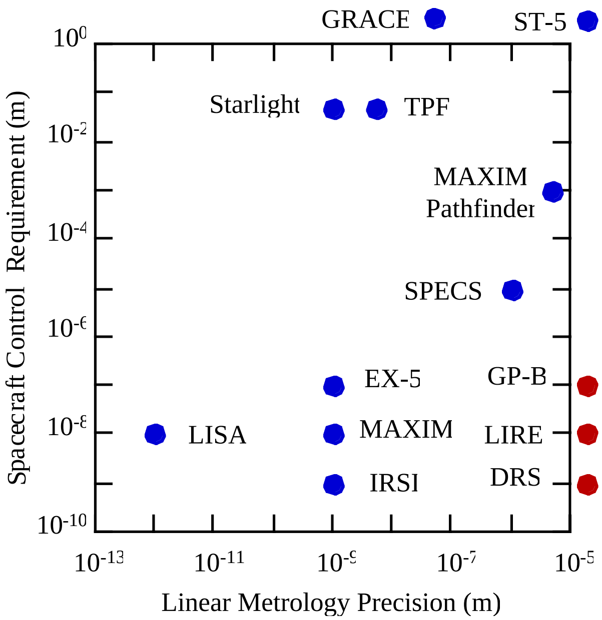

Figure 3 Position control and interspacecraft measurement requirements for formation flying and drag-free missions. Single-spacecraft drag-free missions are shown with no linear metrology requirement (in red). GRACE has no position control requirement. ST5 requirements are off scale.

Figure 3 shows the range of interspacecraft metrology measurements versus spacecraft position control envisioned for a number of future mission. The range between the spacecraft position control that will be demonstrated by the StarLight mission and that would be required for LISA or MAXIM is a factor of 10^7. Thus there is a large range for potential technology demonstration.

The DRS will demonstrate the spacecraft position control performance needed to satisfy all foreseen roadmap missions. The DRS will use new microthruster technology rather than the gas system based on the cryogenic liquid-helium system used by GP-B.

1.1.3 DRS Technologies

The DRS includes two specific technologies. The first DRS technology is a gravitational reference sensor, which provides a test mass designed to follow a trajectory in space based only on gravity (as nearly as possible). The test mass must be isolated from external forces. The sensor must also provide a means of measuring the position of the test mass with respect to the housing. The spacecraft must be kept centered on the test mass to shield the reference mass from external forces, such as atmospheric drag and solar radiation pressure. (Such operation is often called "drag-free" operation since the first application of the technology was to reduce the effect of atmospheric drag on the trajectory of an Earth-orbiting spacecraft.)

The second DRS technology is a set of microthrusters used to control the spacecraft position to a few nanometers. The microthursters are miniature ion engines with continuous output controlled by the voltage with which ions are accelerated and ejected. The DRS will use thrusters based on a colloidal propellant for which the liquid propellant is fed by pressure through a small needle. At the tip of the needle a high electric field causes droplets to be formed. The droplets are spontaneously ionized, pulled from the needle, and ejected at high velocity. The thrusters are capable of extremely precise thrust control, and have a high specific impulse so the propellant mass is small.

1.1.4 Technology Interrelationship

The two DRS technologies are both essential for the two interrelated DRS system-level performance goals. The gravitational sensor test mass will be used as the reference for spacecraft position control while the microthrusters will provide the needed control of the spacecraft position. The gravitational sensors can also be operated as high-precision accelerometers to validate the performance characteristics of the thrusters. Nanometer spacecraft position control by the microthrusters is needed to keep noise forces on the freely floating test mass small enough to meet the acceleration noise performance goals.

1.1.5 Project Cost and Schedule

The DRS will consist of an instrument package and a set of thrusters, which will be attached to a suitable spacecraft. The DRS will be attached to either the ESA SMART-2 spacecraft or the NASA StarLight Collector spacecraft, both of which are scheduled for launch in mid-2006. Attaching the DRS to a host spacecraft allows for a large fraction of NMP funding to be invested in new technologies. DRS development would start in February 2002 with a 15 month Formulation Refinement phase during which breadboard units will be built and tested. A two-year Implementation phase will start in May 2003 with a one-year brassboard development and one-year flight system development. DRS subsystems will be delivered to JPL in February 2005 for Integration and Test, and shipped to the spacecraft contractor site for integration to the spacecraft in June 2005. After launch, the DRS system will be powered off while other technologies are tested on either the StarLight or SMART-2 project. Three months of DRS operations will take place several months after launch. Individual DRS tests will require 12-48 hours of operation, and tests may be interspersed with other project tests. The DRS project will end in September 2007.

The estimated DRS project cost is \$60.5 M, with \$59.2 M from the New Millennium Program and \$1.3 M in civil servant salaries and benefits from the Goddard Space Flight Center. The project cost includes 30% cost reserves during formulation refinement and implementation. Approximately \$25 M will be spent on new technology development (excluding reserves).

1.2 Infusion Target

The successful demonstration by DRS of its suite of technologies will, in some cases, allow them to be directly incorporated into future projects. In other cases, the results will be further enhanced in performance or lifetime in laboratory testing before use on future projects. The primary target missions for DRS technologies are described briefly below.

The Laser Interferometer Space Antenna (LISA) is a planned space observatory for gravitational waves. The LISA mission is included in the 2000 Office of Space Science Strategic Plan for Implementation starting in the 2006-2008 time frame. LISA is planned as a cooperative mission between NASA and ESA. LISA observations are based on measurement of distance changes between widely separate test masses. LISA has stringent requirements on acceleration noise on the freely floating test masses. The DRS will directly demonstrate acceleration noise performance within a factor of ten of the LISA requirements and provide model validation measurements to allow extrapolation to the LISA requirements.

EX-5 is a planned follow-on mission to the NASA/DLR Gravity Recovery and Climate Experiment (GRACE) mission. EX-5 is included in the Earth Science Program strategic plan for implementation in the 2006-2008 time frame. The EX-5 concept is based on two drag-free spacecraft in low Earth orbit. Measurements of the distance between test masses contained within the spacecraft provide measure of changes in the Earth gravity field. In order to improve on the GRACE mission by more than a factor of ten, EX-5 requires a drag-free mission implementation. The acceleration noise requirements are less stringent than those for LISA. DRS would directly demonstrate capabilities required for EX-5.

Measurements to a freely floating test mass would allow high-precision tests of gravity theories such as general relativity. Some of the best tests of gravity to date are based on measurement between an interplanetary spacecraft and Earth. These determinations are partly limited by acceleration noise on the spacecraft. A freely floating test mass would enable an improvement in tests of gravity theories by a factor of $\sim10^3$ beyond current knowledge. The Laser Interplanetary Ranging Experiment (LIRE) is a concept for a solar system test of gravity that is included in the Roadmap for Fundamental Physics in Space from the NASA Microgravity program. Specific acceleration noise requirements are not available for this project, which is not being actively studied. A similar mission concept called the General Relativity And Coronal Explorer, studied as a mission concept under the 1997 New Mission Concepts in Astrophysics program, did include an assessment of acceleration noise. These requirements were very similar to the current LISA requirements, which would be addressed by DRS.

The Terrestrial Planet Finder (TPF) is a mission for direct detection of light from planets orbiting other stars. The nominal TPF architecture consists of five spacecraft with four collecting infrared light an re-directing it to a fifth spacecraft where the light is combined with appropriate matching of delays to null out the light from the central star to allow detection of light from the associate planets. The nominal TPF architecture includes control of the spacecraft position with centimeter accuracy and fine adjustment of the positions of optics using delay lines. The position control technology for this will be validated by the StarLight project. Several alternative architectures haven been studied, including some which call for the spacecraft positions to be controlled with accuracy of ~100 nm, as is baselined for the current ESA baseline concept for the Darwin mission. The DRS would demonstrate the nanometer-level position control that would enable these architectures, which would increase the range of options available for TPF.

The Micro-Arc-second X-ray Imaging Mission (MAXIM) is a separated-spacecraft interferometer included in the Roadmap for the Structure and Evolution of the Universe program as a far-horizon mission. MAXIM includes several spacecraft with a set of grazing-angle collection optics and a second spacecraft containing detectors at the image plane. The imaging detectors must be positioned with respect to the collecting optics with nanometer accuracy, and movable delay lines are difficult to build with grazing angle optics. Therefore the baseline concept for MAXIM includes nanometer-level control of spacecraft positioning. The DRS will validate the required position control technology.

Stellar Imager is an ultraviolet separated-spacecraft imaging interferometer concept designed to image the surfaces other stars for comparison with the Sun. Stellar Imager is included in the Roadmap for the Sun-Earth Connection program as a far-horizon mission. Like TPF, Stellar Imager (SI) requires collection of light by several spacecraft for recombination on a detector spacecraft and has several possible architectures. Currently the Stellar Imager baseline concept calls for nanometer spacecraft control. The DRS will validate the required position control technology for this architecture.

1.3 Flight Validation Rationale

The DRS technologies require flight validation because environmental noise makes ground testing impossible at the levels required. At low frequencies the gravitational reference sensors are sensitive to vibrations and slow changes in surrounding mass distribution (with sources such as vehicle motion and weather changes), which cannot be reduced by shielding. The best acceleration performance achieved on Earth to date is a factor of ~10^4 larger than the DRS goals and in only two degrees of freedom. Sensitive torsion balances have the potential to demonstrate performance within a factor of 100 of the DRS goal in one degree of freedom. Ground tests can provide significant validation of some components of gravitational reference sensors performance models, but will always be limited by environmental noise, and will never allow the investigation of possible coupling among all degrees of freedom. Only a space demonstration can validate the levels of performance desired.

The microthruster performance can similarly be characterized in laboratory tests in large vacuum chambers. However some key parameters of the microthruster system cannot be tested even in the largest vacuum chambers. The stability of the thruster output is very difficult to measure in the laboratory since the required stability is much smaller than the Earth' gravity. So far the thrust noise is inferred indirectly through measurement of the current and velocity of the propellant. Several apparatuses are under construction to allow direct measurement of the thrust force, but these are likely to remain limited in performance. The DRS will allow very accurate measurements of thruster performance. The DRS will also include the thrusters in a three-dimensional control law, as system demonstration that cannot be achieved on Earth.

Besides the difficulty in measurement of force noise, some aspects of the thruster depend on distribution of charged particles over many kilometers. The microthrusters emit positively charged propellant at high velocity. In order to avoid leaving the spacecraft negatively charged, which would tend to attract the propellant particles and reduce thruster performance, the microthruster system includes a cathode to emit electrons to keep the spacecraft electrically neutral. The positively charged propellant ions and the electrons are expected to recombine at distances of tens of kilometers from the spacecraft. In laboratory tests the walls of the vacuum chambers are relatively close to the thrusters and interfere with the recombination process.

1.4 Project Design

1.4.1 DRS Project Overview

The DRS hardware consists of an instrument package and a set of thrusters. The DRS hardware will be attached to a host spacecraft. The host spacecraft will provide power to the DRS and radio communications with Earth. The major subsystems of the DRS are the gravitational reference sensors, the microthrusters, the laser interferometer, and the instrument enclosure. Attaching the DRS to a host spacecraft allows for a large fraction of NMP funding to be invested in new technologies.

The two Gravitational Reference Sensors each include a freely floating test mass within a reference housing. The position and attitude of the test mass with respect to the reference housing is determined by measuring the capacitance between the test mass and electrode surfaces on the reference housing. The test masses are used to provide a reference for spacecraft position and attitude. The position and attitude of the spacecraft are determined by measuring the position and attitude of the test masses with respect to the spacecraft. The DRS computes the thrust required to keep the spacecraft centered on the two test masses and executes the thrusts using the microthruster subsystem. The laser interferometer is used to measure changes in distance between the two test masses to determine the level of acceleration noise on them.

At preprogrammed times the host spacecraft will go into a DRS mode. In DRS mode the spacecraft will assume a known orientation with respect to the sun. Under normal conditions, the DRS will keep the spacecraft position and attitude fixed with respect to the freely floating test masses. The spacecraft will be programmed to not fire its attitude control thrusters as long as the attitude remains within a few degrees of the nominal orientation. If the attitude changes by more than a few degrees, the spacecraft will assume an error in the DRS and power the DRS down and assume a safe attitude using its attitude control thrusters.

DRS operations will consist of several different experiments of 12-48 hours each. The different experiments will be used to characterize different aspects of the DRS performance. Because performance goals involve measurements of quantities at frequencies down to 10^{-3} Hz, each experiment must be many thousands of seconds long. At the end of each experiment the DRS performance data will be transferred to the host computer for relay to Earth. Nominally each DRS experiment can be done independently at arbitrary times, allowing flexibility in scheduling by the host spacecraft. The total set of DRS experiments can be accomplished in 90 days of operations.

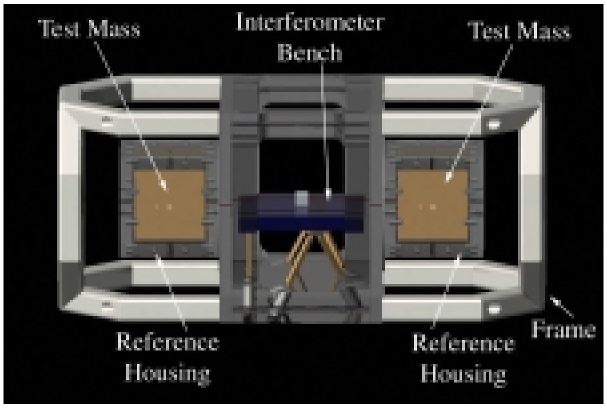

Figure 4 Artist's view of the DRS instrument assembly. Each test mass is a 4-cm cube. The reference housings have electrodes deposited on the inner walls for capacitive position and attitude measurement. Vacuum enclosures are not shown.

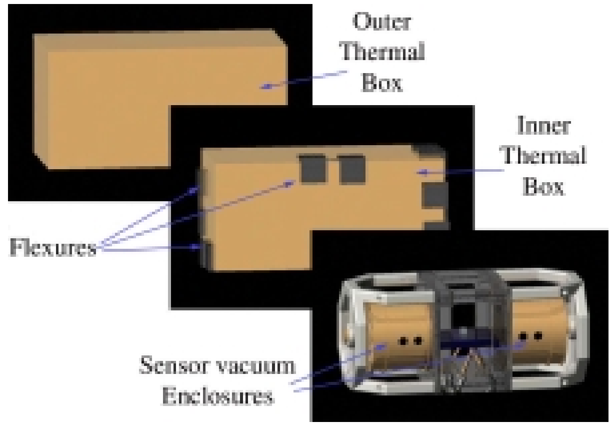

Figure 5 DRS thermal isolation system. The DRS instrument assembly is mounted within an inner thermal box, which in turn is mounted within an outer thermal box. This makes a three-stage thermal filter which isolates the gravitational sensors from external temperature fluctuations.

1.4.2 DRS Instrument Assembly

The core of the DRS consists of two Gravitational Reference Sensors (GRS). The two sensors will be attached to a reference block, which will provide precision position and alignment of the two sensors with respect to each other. A laser interferometer will also be attached to the reference block. The laser interferometer will measure changes in the distance between the two test masses to determine the level of acceleration noise. Figure 4 shows the core instrument assembly.

To achieve the required low level of acceleration noise on the GRS test masses, each test mass must be in a very good vacuum and be in a very stable thermal environment. To meet the vacuum requirements, for a pressure of $< 10^{-7}$ Pa, each GRS will be contained within an individual vacuum enclosure. The enclosure will be pumped out and sealed prior to launch. Experience with accelerometers for GRACE indicates no pumping is needed on orbit. If necessary for DRS, a small getter pump can be installed for pumping on orbit. To achieve the required thermal stability, for temperature fluctuations less than 10 µK, the core instrument assembly will be placed inside a double-walled thermal enclosure. A similar system has been used in laboratory experiments to achieve the required temperature stability[3,4] and analysis shows that this design will work for DRS.[5]

Figure 5 shows the pieces of the overall instrument assembly. The reference block is attached to a frame for attachment to the inner thermal box. The attachment is made using flexures to avoid introducing stress in the core instrument due to thermal strain. The flexures make a connection with low thermal conductivity. The thermal boxes are coated with gold to reduce radiative thermal coupling. The inner box serves as a thermal heat reservoir midway between the core instrument and the outer environment. This design makes for a two-stage thermal filter. The temperature of the core instrument is not actively controlled and does not need to be any particular value. A temperature anywhere in the range of 0-20° C is acceptable as long as the temperature is stable. External temperature fluctuations on the order of 1 K are filtered to less than 10 µK using the two-stage thermal filter.

The gravitational reference sensors and the interferometer must be precisely positioned and aligned. Because the test masses will be electrically suspended within their reference housings, small adjustments in position and alignment can be made on orbit. The reference block will provide the basic alignment. The faces of the reference block will be made parallel to within a fraction of a milliradian. Locating pins and stops will be machined with accuracy of ~10 µm (0.5 milli-inch). The interferometer bench and gravitational sensors will have corresponding locating pins and holes. The mating flange of the GRS vacuum enclosure will be precision machined and have locating pins for attachment to the reference block. The GRS reference housing will have similar pins. This will allow the electrode surfaces of the two GRS assemblies to be parallel within a milliradian and located with an accuracy of 10 µm. The interferometer bench will have similar pins to allow precision attachment to the reference block. During integration and test, dummy GRS flanges will be used with flat mirrors to allow final alignment of the interferometer. On orbit, the interferometer readout will allow the orientation of each test mass to be read. The orientation of each test mass will be adjusted to be perpendicular to the interferometer beam within ~1 microradian by applying small correction voltages to the GRS electrodes.

1.4.3 DRS Electrical Interfaces

Figure 6 shows the functional connections among the various DRS subsystems. The GRS electronics includes a computer that controls all DRS subsystems. The GRS electronics will receive unregulated power from the host spacecraft and distributes it to the other subsystems. The GRS computer will receive its commands from and send data to the host computer through a standard interface. The GRS computer will send commands to the microthrusters using RS-422 serial ports. The GRS computer will control the laser and photodiode amplifiers for the interferometer through digital-to-analog lines. The GRS computer will read out the position and attitudes of the test masses with respect to their reference housings. The attitude of the test masses will be controlled to keep the laser beam from the interferometer perpendicular to the test masses. This control will be performed by changing voltages on the electrodes on the reference housing. The amount of voltage will be determined by a digital control loop program running on the GRS computer. A similar control loop program will be used to compute the required thruster firing to keep the spacecraft position and attitude fixed with respect to the test masses. Data will be stored in volatile memory on the GRS CPU board before being transferred to the host spacecraft computer for transmission to ground.

1.4.4 Project Life Cycle

The DRS development schedule is based on Formulation Refinement starting in February 2002 and ending in April 2003. DRS Mission Design Review and System Requirements Review and will be held in April 2002. A draft DRS system requirements document has been written and discussed with the StarLight and SMART-2 project management, During Formulation Refinement, breadboard models of all DRS components would be developed. The breadboard models would be used to prove technology readiness at TRL 5 and test compatibility between the DRS subsystems. DRS preliminary design review and confirmation assessment review will be held near the end of Formulation Refinement in April 2003. During the first half of the Implementation phase, high-fidelity brassboard models will be built to finalize part selection and to test the mechanical design of the instrument assembly and thrusters. Critical design review will be held near the end of the brassboard development, followed by the start of flight system development in April 2004. DRS subsystems will be completed in February 2005 and shipped to JPL for integration and test. The DRS subsystems will be developed as protoflight models and will have environmental testing

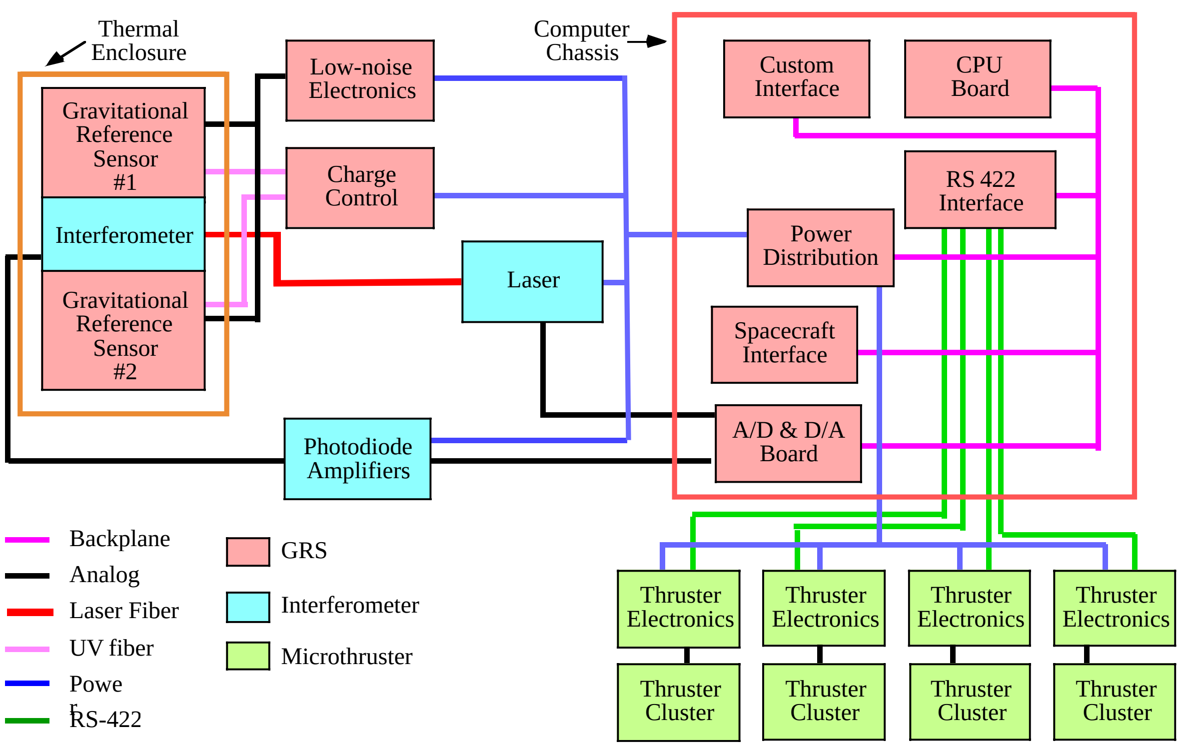

Figure 6 DRS electrical interface diagram

done prior to shipment. During integration and test the DRS subsystems will undergo final assembly and a environmental testing will be performed at the system level. The DRS will be shipped to the spacecraft integration site for integration onto the spacecraft. The DRS system will be powered off while other technologies are tested on either the StarLight or SMART-2 projects. Three months of DRS operations will take place several months after launch. Individual DRS tests would require 12-48 hours days of operation, and DRS tests may be interspersed with other project tests.

1.4.4.1 DRS Resource Requirements

The current best estimate of the DRS mass, power, and volume are given in Table 3. The DRS instrument assembly, includes the two GRS assemblies in individual vacuum cans, the reference block, the laser interferometer, and the thermal enclosure. The GRS electronics and charge control system resource requirements are based on comparable units that have been developed for GP-B. The thruster requirements are based on a prototype thruster. The balance mass will be configured to balance the force on the GRS test masses from the rest of the spacecraft equipment. A 30% contingency is included given that most of the system represents new flight hardware, even for the subsystems based on GPO-B hardware.

The DRS resource requirements are within the allocations of 80 kg and 170 W from the StarLight

Table 3 DRS mass, power, and volume

Subassembly	Mass, kg	Power, W
Gravitational Sensor #1	6.3	0.0
Gravitational Sensor #2	6.3	0.0
GRS Analog Electronics	3.0	23.0
GRS Digital Electronics	2.5	25.4
GRS Charge Control System	3.5	11.4
Reference Block	2.5	0.0
Frame	2.1	0.0
Interfereometer Block	0.6	0.0
Thermal Enclosure	3.8	0.0
Laser	0.2	9.0
Laser Controller	0.3	1.0
Interferometer Electronics	0.4	0.8
Thruster Cluster #1	3.4	12.4
Thruster Cluster #2	3.4	12.4
Thruster Cluster #3	3.4	12.4
Thruster Cluster #4	3.4	12.4
Cabling and Misc. Structure	3.0	0.0
Balance Mass	5.0	0.0
Subtotal	*53.2*	*120.0*
Contingency (30%)	16.0	36.0
Total	**69.1**	**156.1**

project. The SMART-2 project is in an early stage of development and has not yet made allocations for DRS.

1.5 Advanced Technology Description

1.5.1 Gravitational Reference Sensors

1.5.1.1 GRS Summary

The GRS subsystem consists of a test mass that floats freely in its housing. The separation between the test mass and its housing in all six degrees of freedom is monitored by non-contacting capacitor plates fixed to the housing. The sensing noise is required to be low enough to satisfy the spacecraft position noise requirement in Table 2. Any back-action or other GRS-induced force noise must be below the test mass force noise requirement in Table 2.

The GRS is a project-defining technology. The technology provider is Stanford University, which has based the GRS design on its experience with the GP-B payload system.

1.5.1.2 GRS Heritage

The first gravitational reference sensor, DISCOS, was developed at Stanford University and was a single drag-free sensor flown on the US Navy's Triad mission in 1973.[2,6] The test mass was an 22-mm diameter sphere made of a platinum-gold alloy with a gap between the test mass and the housing of 9 mm. Capacitive measurements were used to determine the position of the test mass with respect to its reference housing. Figure 7 shows a schematic diagram of the DISCOS sensor. The Triad mission was in operation from September 1972 through September 1973. The residual acceleration on the test mass was inferred from the spacecraft trajectory, which was determined by radio tracking measurements. The measured performance, shown in Figure 2, was limited by the noise in the orbit determination.

The French institute ONERA has flown a number of sensors starting with the CACTUS sensors which also used a spherical test mass. Later systems have been optimized as six degrees of freedom measurement accelerometers with a faceted parallelepiped test mass.[7] A 1995 flight on the space shuttle showed a performance of $10^{-8}\ \mathrm{m{\cdot}s^{-2}/\sqrt{Hz}}$. A redesigned sensor was launched on the German CHAMP mission in July 2000 but has experienced some problems so is not well characterized at this time. The GRACE mission planned for launch in February 2002 has two spacecraft each with an updated accelerometer designed by ONERA accelerometer. GRACE is expected to have a residual acceleration noise of about $10^{-10}\ \mathrm{m{\cdot}s^{-2}/\sqrt{Hz}}$ at a frequency above 5 mHz, as indicated in Figure 2. The

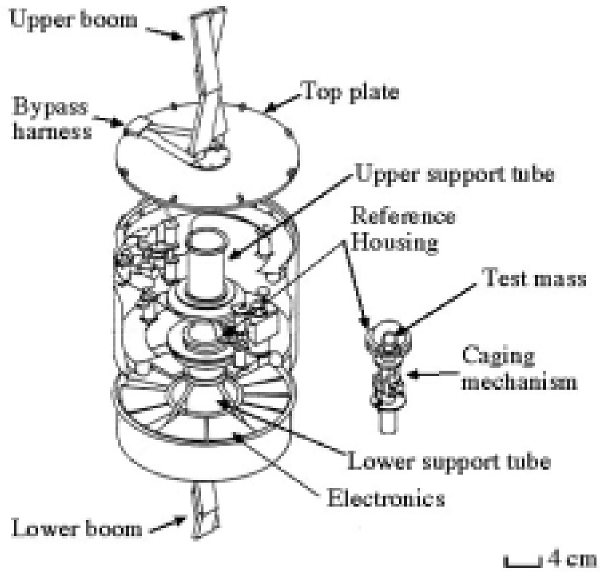

Figure 7 Schematic view of DISCOS sensor

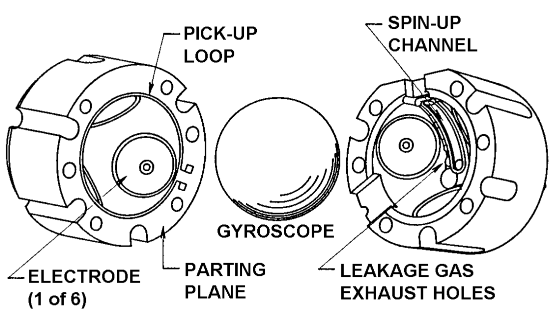

Figure 8 Schematic view of the GP-B drag-free sensor

GRACE acceleration performance will be verified by comparing the results of the accelerometers on the different spacecraft.

GP-B, also developed at Stanford, uses a spherical test mass that doubles as a gyroscope. The test mass is contained within a housing with electrodes on the inner walls for measurement of the position of the test mass with respect to the housing. The gyroscope design is shown in Figure 8. The sensor is a 20-mm radius niobium coated quartz sphere with a 30-μm gap to the housing.[8] The GP-B payload includes four independent gyroscopes. The spacecraft will be flown in a drag-free manner with respect to one of the gyroscope masses. The acceleration noise can be inferred by comparison of capacitive readout of the test-mass positions from different gyroscopes. The GP-B requirements are driven by the gyroscope performance, so GP-B is not optimized to test the ultimate limits of acceleration noise. The acceleration noise performance will be limited by digitization noise in the capacitive position readout. The expected acceleration noise for the GP-B drag-free sensor is $2 \times 10^{-12}\ \mathrm{m \cdot s^{-2}/\sqrt{Hz}}$ at 5 mHz.[9]

1.5.1.3 GRS Design

The principal subsystems composing the DRS gravitational reference sensors include (1) a test mass and housing with caging mechanism, (2) sensing and forcing systems with associated electronics, and (3) charge management system. The gravitational sensor will be designed to meet the DRS requirements within the system mass, power, geometry, telemetry and computational constraints. The GRS will include a cubical test mass with capacitive sensing and electrostatic forcing using innovative technologies in the areas of sensing, forcing, charge management and manufacturing, many of which are derived from our experience with the Triad and GP-B missions. The main issues to be considered are the disturbance accelerations, the system complexity caused by the required forcing in some dimensions, and the difficulty and constraints due to the need to test the forcing system on the ground.

The largest disturbances to the inertial trajectory of a craft in space (radiation pressure, residual gas drag, and particulate impacts) are cancelled by the basic concept of a drag reduction system. The final performance of the system will be limited by a number of smaller disturbances. These disturbances fall into three categories: 1) variations in the gravitational potential at the test-mass location, 2) momentum transfer to the test mass by residual gas and cosmic radiation particles, and 3) variations of the electromagnetic fields at the test-mass location. The main gravitational fluctuations are due to the thermal distortion of the spacecraft and to the relative displacement of the test-mass with respect to the spacecraft. Improving the thermal shielding, using materials with low temperature coefficient, and maximizing the symmetry of the mass distribution of the spacecraft will reduce the thermal distortion effects. Reducing the gravity gradient and displacement of the test mass minimizes the gravity noise caused by spacecraft displacement. For reasonable space-experimental pressures, $10^{-7} – 10^{-6}$ Pa at 250 – 300 K, the forces caused by residual gas impacts are dominant compared to forces produced by cosmic radiation, though well below the requirement level. A number of electromagnetic effects cause test-mass disturbances, and each can be minimized to a considerable extent. Radiation pressure differences across the gravitational sensor housing are reduced by thermal isolation and making heat leaks as symmetrical as possible. Discharging the test-mass, reducing its displacement, and maximizing the test-mass-to-housing gap minimizes electric forces on the charged test mass.

Table 4 DRS acceleration noise budget in units of 10^{-15} m·s^{-2}/√Hz

Acceleration Noise Source	Acceleration Noise
Capacitive sensing error coupled through residual stiffness	20
Thruster spectral noise	2
External housing forces	0.2
Lorentz force on charged test mass	0.1
Magnetic force on test mass from fluctuating interplanetary field	-
Thermal radiation pressure on test mass	0.06
Radiometer effect	-
Gravity noise due to spacecraft displacement	3
Residual gas impacts on test mass	1.5
Electronic losses in the electrodes	-
Other substantial effects	5
Total Acceleration Noise Budget (for one gravitational sensor)	21

Fluctuating magnetic fields cause magnetic forces and, for a charged test mass, Lorentz forces.[10,11,12] These forces are reduced by choosing a test-mass material with magnetic susceptibility of less than 10^{-6} and by discharging of the test mass.

As a point of reference, the required performance for the DRS is a residual acceleration of less than 3×10^{-14} m·s^{-2}/√Hz from 10^{-3} Hz to 10^{-2} Hz, with a position sensing noise of less than 10^{-9} m/√Hz. Stanford University has developed a noise tree that relates design variables and disturbance parameters through various design choices to a final total noise figure. This allows the designer to make a direct comparison of the effects of various design choices on the performance of the sensor. Table 4 shows a comparison of the expected performance of the DRS sensor as derived from this noise tree.

1.5.1.3.1 Test Mass and Housing

The DRS drag-free test mass will be a cube of low magnetic susceptibility Au-Pt that is 4 cm on a side and has a mass of approximately 1.25 kg. A series of tests related to the precise machining and polishing of the Au-Pt test mass have been performed at Stanford, which will indicate of how the test mass can be machined with current manufacturing techniques. Various materials will be considered for the construction of the housing, which must offer both low geometric distortion and high thermal conductivity. Current materials under investigation include beryllia, ULE glass, and SiC.

Manufacture of the test mass and housing is currently being studied. The requirements for parallelism and perpendicularity of critical surfaces is on the order of an arcsecond, with surface polish to $\lambda/20$. To establish that the machining requirements can be met, a sample test mass of steel has been machined. The resulting test mass, shown in Figure 9, meets the configuration requirements.

The configuration for the reference housing shown in Figures 10 and 11 has a 2-mm gap, several stops to prevent contact between the test mass and the electrodes, a 3-mm diameter opening for passage of the interferometer beam, two optical fibers for charge

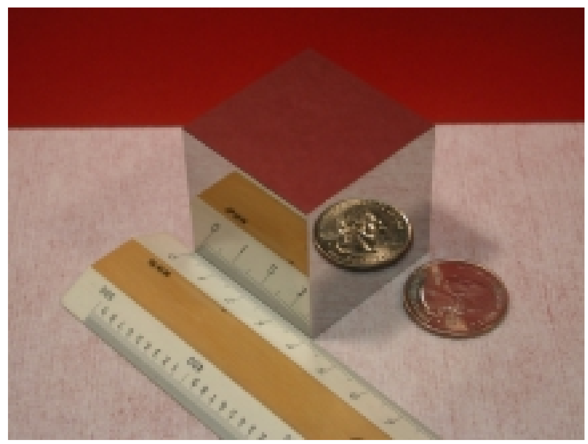

Figure 9 Steel cube machined to GRS test mass requirements

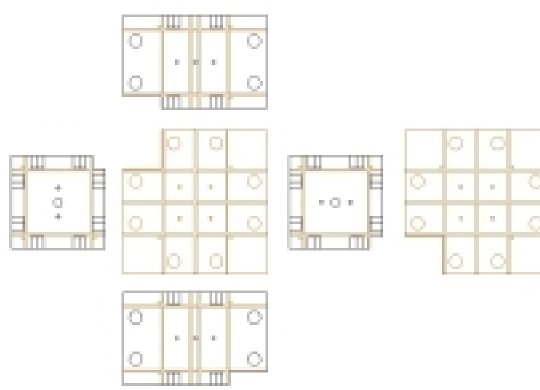

Figure 10 GRS housing components

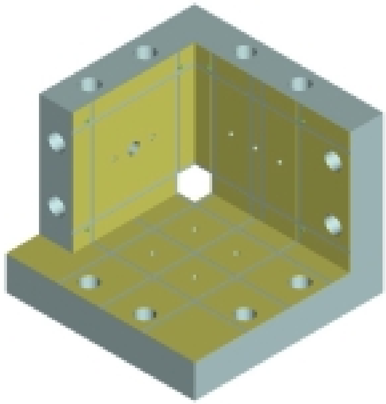

Figure 11 GRS housing assembly

control, and a caging mechanism. The electrode layout allows for control of all three rotational degrees of freedom and the two transverse degrees of freedom without forcing the test mass in the sensitive direction. All remaining housing surfaces are grounded to avoid accumulation of electrostatic charge. The caging mechanism for the DRS will hold the test mass in the center of the housing and allow it to be traversed along the axis of the caging pistons. It will also be used for ground testing and calibration of the position sensing system. Two rods at opposite corners of the cube will extend into the housing, holding the test mass at its center with a three-point contact on the test mass. Piezoelectric nano-steppers will be able move the rods, thus moving the test mass across the housing, which will allow testing of the capacitive sensing electronics. For uncaging, the nano-steppers will retract the rods to the edges of the housing.

1.5.1.3.2 Electrostatic Readout and Forcing

The capacitive electrostatic technology is well established for position readout and forcing of gravitational sensors. As an example of the available technology, a short description of the flight system for the GP-B test mass is given below. Note that this system incorporates both position readout and forcing capability, and that its primary design drivers are minimization of torque and high reliability. Due to its digital design and forcing capability, the GP-B suspension system requires a dedicated processor.[14]

The position of the rotor is measured with capacitance bridges, using a 40 mV-PP 34 kHz sinusoidal sense signal superimposed onto the drive electrodes. The high-precision, low-noise design of the bridge results in an operational noise floor of 0.1 nm/√Hz, a resolution that allows the control system to meet the rotor centering requirements for the mission. The choice of magnitude and frequency of the sensing signal was limited by specific GP-B systems considerations.

The overall design is partitioned into two physical enclosures in order to separate the forward analog low-noise portion of the system from the aft digital noisy section. Thermal stability of the analog electronics is of a prime concern in the space environment. This system has passed all engineering tests, and the flight units are undergoing systems and integration testing.

1.5.1.3.3 Charge Management System

Charging of the test mass by cosmic radiation is a major source of disturbance for gravitational sensors. A number of techniques have been used to solve the problem: a) very thin wires are run from the test mass to ground (ONERA), b) bringing the test mass into contact with the grounded housing at intervals (CACTUS), and c) a non-contact system for charge measurement and control using UV photo-emission (GP–B).[9] The requirements of DRS make the non-contact method the only choice and the baseline candidate design. Table 5 compares the requirements for the DRS charge control to the achieved performance of the flight GP-B system.

For GP-B the charge measurement is achieved by a force modulation technique wherein the test mass position is forced at a frequency high compared to the bandwidth of the drag-free control system. The motion in the position is proportional to the potential of the test mass. This technique is insensitive to test mass centering and achieves an accuracy of better than 5×10^{-12} C for an integration time of 100 s. UV photo-emission is used to generate the electrons used for charge control. Rotor and biasing electrode are illuminated with UV light, and the electrons generated by photo-emission from both these surfaces are added to or removed from the rotor using a dedicated biasing electrode. The direction of the charge flow is controlled by biasing of the charge control electrode to ± 3 V with respect to the test mass surface. Flight hardware testing has confirmed the functionality of the system.

Table 5 Charge control requirements for DRS and GP-B

	DRS	**GP-B**
Charging rate from ionizing radiation	10^{-17} to 10^{-18} C/s	~ 5×10^{-17} C/s
Maximum allowable charge	< 10^{-12} C	< 10^{-11} C
Charge measurement accuracy	< 5×10^{-13} C	< 5×10^{-12} C
Gravitational sensor compatibility	3×10^{-14} m·s^{-2}/√Hz	10^{-12} m·s^{-2}/√Hz

Table 6 GRS Path to Technology Readiness Level 5

Technology	Current TRL	TRL after Formulation Refinement	Method
Error Analysis	3.0	5.0	Analysis via noise tree
Test-mass Manufacturing	3.5	5.0	Manufacture of sample test mass
Housing Manufacturing	4.0	5.0	Manufacture of housing prototype
Caging Mechanism	3.0	5.0	Assembly and test of breadboard
Capacitive Sensing System	4.5	5.5	Assembly and test of breadboard
Electrostatic Forcing System	4.5	5.5	Assembly and test of breadboard
Charge Management System	6.0	6.0	None

For DRS the charge measurement system performance will be adjusted to meet the requirements. The low charging rate for DRS (as well as for the other missions of interest) will allow the reduction of the duty cycle for both charge measurement and electron generation.

1.5.1.4 GRS Technology Readiness Level and Path to TRL 5

Systems demonstrating the concept of both spherical and faceted gravitational sensors have been flown on a number of occasions, with the most representative ones being DISCOS and the ONERA accelerometer on the CHAMP spacecraft. The next generation of sensors, on the GP-B and GRACE missions, are scheduled to fly in the next two years, further validating the concept and advancing the technology. The current and projected TRLs of all DRS gravitational sensor subsystems are shown in Table 6. All subsystems are at a TRL of 3 or better (particularly those with GP-B heritage). The DRS schedule includes raising the TRL of all of the major GRS subsystems to TRL 5 or higher by the end of the Formulation Refinement Phase, through construction and testing of a breadboard sensor and the associated electronics.

The justification for the current Technical Readiness Levels of the specific subsystems are as follows:

- Error Analysis – A noise tree has been created for DRS that evaluates effect of disturbance parameters on total instrument noise. This model has been applied to the baseline DRS design described here to establish a TRL 3 for the overall system.
- Test-mass Manufacturing – The DISCOS test mass was constructed from the same Au/Pt alloy planned for DRS. Samples of this material are currently being tested to establish that they can be polished to an optical finish. Prototype test masses of steel have been manufactured and measured to establish the required precision.
- Housing Manufacturing – GP-B has demonstrated that fused quartz housings can be manufactured with the dimensional tolerances necessary for drag-free sensors. A preliminary design has been completed for the DRS sensor housing.
- Caging Mechanism – GP-B has shown that a mechanical system can be used to cage a drag-free test mass for launch without damaging its surface. A baseline concept for the caging mechanism has been developed.
- Capacitive Sensing System – A variant of the GP-B capacitive sensing system will be used. The GP-B system has seen extensive bench testing of both prototype and flight units.
- Electrostatic Forcing System – A variant of the GP-B electrostatic forcing system will be used. The GP-B system has seen extensive bench testing of both prototype and flight units.
- Charge Management System – The GP-B charge management system described above will be used. It requires only minor modification. This system has seen extensive bench testing of both prototype and flight units.

The path to TRL level 5 is shown in Table 6. The activities to be undertaken during Formulation Refinement include

- Error Analysis – Additional analyses will be done to bring the noise tree to maturity. These include a) a detailed study of the effects of cross-coupling between the sensitive axis and the other degrees of freedom,. b) completion of the effects of measurement error in the noise tree, and c) additional detail on existing errors.
- Test-mass Manufacture – The activities related to test-mass manufacture include a) manufacture and measurement of a prototype test mass, b) machinability study of the Au/Pt alloy,. c) measurement of the magnetic susceptibility of the Au/Pt alloy, and d) Manufacture of one or two Au/Pt test masses during the Formulation Refinement phase.
- Housing Manufacture – A mechanical design of the housing will be completed, and a prototype will be manufactured. The most important open design and manufacturing issues include a) isolation of electrodes from ground plane, b) electrical connection to

electrodes, c) incorporation of current sensing element into housing design, d) manufacturing tolerances e) verification of manufacturing tolerances, f) electrode sputtering procedures, and g) thermal design.

- Caging Mechanism – The mechanical design of the caging mechanism will be completed and the manufacture of the prototype has been started in the Concept Definition Study Phase. The important attributes of the caging mechanism are a) ability to withstand launch loads without damage to test mass, b) robustness for uncaging, and c) use of caging mechanism for verification of capacitive sensing. d) breaking of symmetry in ground plane due to caging mechanism.
- Electrostatic Forcing, Capacitive Sensing, and Charge Measurement and Control – Analysis of the necessary upgrades to existing GP-B electrostatic forcing, capacitive sensing, and charge measurement and control will be done. The manufacturing and testing of the prototype system will be started in the Concept Definition Study Phase and completed in the Formulation Refinement Phase. The open issues in these areas include a) Test mass and Electrode Edge Effects. b) Cross-coupling between degrees of freedom in electrostatic forcing and capacitive sensing. c) Improved electron emission due to ultraviolet light. d) Capacitive sensing and electrostatic forcing frequencies. e) Charge measurement strategy

1.5.1.5 GRS Validation Experiments and Success Criteria

The GRS has requirements on both position measurement and acceleration noise that are incorporated into the DRS Level 1 requirements in Table 2. In order to validate the position readout performance, the GRS readout of the position of the test mass will be compared to the laser interferometer readout of the position of the test mass. In order to be successful, the GRS position measurement must agree with the laser interferometer measurement to within 1 nm. To validate the acceleration noise performance, the laser interferometer will measure changes in the distance between the two test masses. To be successful, the measured distance changes must reflect a level of acceleration noise consistent with the Level 1 requirements. A more detailed description of the series of tests planned is given in Section 1.8.2.

1.5.1.6 GRS System Characteristics

The GRS mass, power, and volume are listed in Table 7. Contingency of 30% is included in the DRS system requirements in Table 3.

Table 7 GRS Resource Requirements

Item	Mass, kg	Power, W
Sensor Assembly #1		
Test Mass	1.3	0.0
Reference Housing	1.0	0.0
Vacuum Housing	2.5	0.0
Vacuum Lid	1.0	0.0
Caging Mechanism	0.5	0.0
Subtotal	6.3	0.0
Sensor Assembly #2		
Test Mass	1.3	0.0
Reference Housing	1.0	0.0
Vacuum Housing	2.5	0.0
Vacuum Lid	1.0	0.0
Caging Mechanism	0.5	0.0
Subtotal	6.3	0.0
GRS Analog Electronics		
LNE Signal Conditioning	0.5	4.5
Electrostatic Suspension	0.5	18.5
Power Conditioning	1.0	0.0
Analog Electronics Housing	0.5	0.0
Cabling	0.5	0.0
Subtotal	3.0	23.0
GRS Digital Electronics		
CPU Board	1.0	14.0
Digital/1553 Interface	0.5	5.4
A/D Board	0.5	6.0
Digital Electronics Housing	0.5	0.0
Subtotal	2.5	25.4
Charge Control System		
UV Lamps	2.4	11.4
UV Switches	0.6	0.0
Charge Control Housing	0.5	0.0
Subtotal	3.5	11.4
Total	**21.5**	**59.8**

1.5.1.7 GRS Risk Mitigation

Table 8 lists a number of technology risks associated with the development of the GRS and means for mitigating these risks. In general, risk will be reduced by early construction and testing of breadboard models. Early testing will identify whether possible problems do occur. Generally there are several possible design choices for each risk area, and early breadboard development will allow selection of a successful strategy. For example, the surface properties of the test mass and the nearby electrodes can cause acceleration noise on the test mass due to relative motion of the surfaces. This noise is in general dependent on surface material, purity, annealing history, etc. Early testing of

Table 8 GRS Technology Risks and Mitigation Strategies

Risk	Likelihood	Severity	Mitigation
Excess noise from patch fields	Medium	Medium	- Test surface properties for several methods - Electro-deposit surfaces of various materials and thickness - Vary surface polishing, annealing - Increase gap between surfaces
Electrostatic forcing system and control algorithm introduces additional test mass acceleration.	Medium	Medium	- Investigate several control law possibilities - Simulation of control algorithm - Use control algorithm with test mass simulator
Caging mechanism design is incompatible with other requirements	Low	Medium	- Fabricate prototype caging mechanism - Investigate alternative mechanisms
Unanticipated problems with housing manufacturing	Low	Medium	- Manufacture model - Fabricate test parts - Fabricate and test current sensing element
Charge measurement or management system is inadequate to control charge to specified level.	Low	Medium	- Use charge measurement and management system based on GP-B design. - Measure photo-current of discharge system
Omission or errors in noise analysis	Low	Low	- Complete existing noise tree. - Compare noise tree with other noise analyses.

surface properties will allow the selection of a suitable material. In the case that the best material chosen has more noise than is desirable, the effect can be mitigated by increasing the gap between the test mass and the electrodes. The acceleration noise is proportional to the cube of the distance between the two surfaces. Increasing the gap reduces the position readout accuracy approximately linearly with the distance between surfaces, which must be weighed against the reduction of acceleration noise.

Use of two test masses on a single spacecraft to test acceleration noise requires that the position of one of the masses be controlled to remain within the spacecraft. For DRS this control system will be designed to adjust the test mass position at low frequencies while leaving the test mass free to move at high frequency. Preliminary analysis of the control system indicates that there are at least some suitable solutions. During Formulation Refinement several possible control laws will be investigated and simulated to find one that meets DRS requirements.

All GRS risk items will be peer-reviewed in an initial Technology Assessment Review to be held early in Formulation Refinement. A second assessment review will be held prior to the preliminary design review, after breadboard models have been developed and tested.

1.5.1.8 GRS Descope Options

The GRS system has been scoped to the minimum necessary to meet the DRS Level 1 baseline performance requirements. A minimum of two units are required for assessment of acceleration noise, and the support electronics and charge control system are single-string with functions shared between the two sensors. One GRS-specific descope would be to eliminate the charge-control system entirely. Without the charge-control system, excess charge accumulated on the test mass from cosmic ray bombardment would have to be reduced by deliberate contact of the test mass to the housing. This would limit the duration of performance tests and, depending on the charging rate, remove the ability to measure acceleration noise at low frequency. The other significant descope possible would be to eliminate the brassboard stage, allowing more time to correct problems at the breadboard stage before proceeding directly to protoflight units.

1.5.2 Micro Newton Thrusters

1.5.2.1 Microthruster Summary

The DRS requires microthrusters capable of smoothly varying thrust from 1 to 20 μN with 0.01 μN resolution and temporal stability are required for drag-free operation of the DRS spacecraft. The maximum needed is 20 μN which is sufficient to counter the solar radiation pressure on the spacecraft. The thrust will be controlled with 0.01-μN resolution in order to control the spacecraft position with respect to the reference (test mass) within 10 nm. The microthrusters must smoothly and continuously counter all external disturbances, primarily solar pressure, with control authority over all 6 degrees of freedom of the spacecraft motion.

The DRS microthrusters consist of miniature ion engines. The propellant is a colloidal fluid, which is fed through a needle by a pressurizing system. At the tip of the needle, a high electrical field is applied, which causes droplets to form and to be ejected from the tip of the needle. The droplets are spontaneously ionized and accelerated by high voltage. Precise changes in thrust can be achieved by changes to the accelerating voltage. In order to prevent the spacecraft from becoming negatively charged by the continual ejection of positively-charged droplets, the microthrusters include a cathode to emit electrons to keep the spacecraft neutral.

The microthrusters are project-defining for the DRS since they are required for spacecraft position control. After validation by DRS the thrusters will be useful without modification for precision spacecraft control on many missions. Performance requirements for the microthrusters, given in Table 9, are derived from the DRS Level 1 spacecraft position control requirements.

1.5.2.2 Microthruster Heritage

The colloidal microthrusters produce a stream of small droplets that are electrically charged and accelerated by an applied electric field to generate variable thrust in the desired range. On the DRS spacecraft, eight colloid thrusters arranged in four clusters of two thrusters will generate the thrust to control the spacecraft in six degrees of freedom.

Busek has been developing a colloid microthruster since the early 1990s and has recently delivered a complete prototype thruster to JPL with a thrust range of 1 μN to 20 μN. A similar system will be delivered to NASA Glenn Research Center under an SBIR program. The delivered system has all the subsystems required by the DRS including the thruster itself, the field emission neutralizer based on carbon nanotubes, propellant storage and feed system, and a power processing unit. This system, the first of its kind, has demonstrated the crucial performance parameters, and has pioneered the use of field emission cathode/neutralizer in electric propulsion. The system, estimated to be at TRL 4, is the baseline from which the flight DRS unit will evolve. The photographs in Figures 12-16 show the colloid thruster, a typical single-needle electrospray, the neutralizer, and the system as a whole during final stages of assembly into its housing.[15]

1.5.2.3 Microthruster Design

The colloid thruster[16-21] is one of only two or three known technologies capable of achieving the DRS requirements. It functions by emitting and accelerating charged droplets, as described below.

Consider a small-diameter tube filled with low-volatility fluid, as sketched in Figure 17. In general, the smaller the tube diameter the better, but practical considerations limit the tube inner diameter to some tens of microns. The fluid must be relatively nonvolatile to minimize its evaporation when exposed to the vacuum of space. It also should have electrical conductivity (K) that is a small fraction of 1 Si/m or greater (typical sea water has K ≈ 5 Si/m).

When sufficient voltage is applied between the extractor and the microtube (emitter), the liquid surface deforms into a cone, as sketched in Figure 17. Taylor[22] found that this cone has a fixed angle of 49.3°, regardless of the type of fluid, its exact properties, or emitter geometry. Equilibrium on the surface of the cone is maintained by the balance of the liquid surface tension and electrostatic pressure. Near the tip, the electric field intensifies to a value that cannot be counteracted by the surface tension, and the cone tip transits into a small-diameter jet, which breaks up into charged droplets, which are accelerated to produce thrust. A photograph of a typical Taylor cone from one single-needle thruster and the issuing jet is shown in Figure 16.

Table 9 Microthruster Performance Requirements

Parameter	Requirement	Comment
Thrust Range	1–20 μN	Smoothly variable between end point values
Thrust control resolution	0.01 μN	
Thrust noise	0.1 $\mu N/\sqrt{Hz}$ from 10^{-3} Hz to 10^{-2} Hz	Stable over given period
Specific impulse	$\geq$ 500 sec	May vary depending on thrust
Total operating time	$\geq$ 2000 hours	At maximum thrust

Figure 12 The 20-μN colloid thruster, which has 57 needles (nano droplet emitters), extractor grid, and an accelerator grid. The double grid is necessary to achieve the required thrust stability.

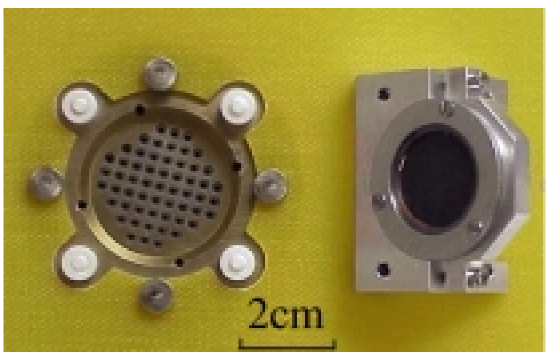

Figure 13 The 20-μN colloid thruster head and carbon nanotube field emission neutralizer

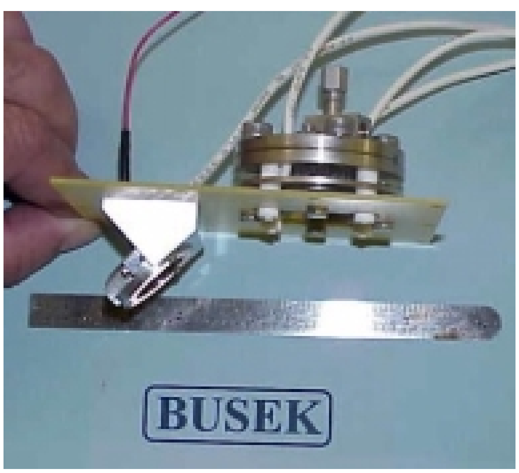

Figure 14 The 20-μN colloid thruster head and carbon nanotube field emission neutralizer mounted on a common dielectric base. The neutralizer gate and its housing are at common potential biased positively relative to the emitter.

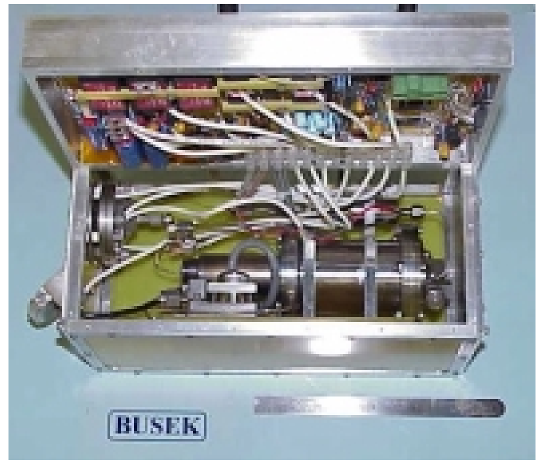

Figure 15 The 20-μN colloid thruster system during integration with the power-processing unit

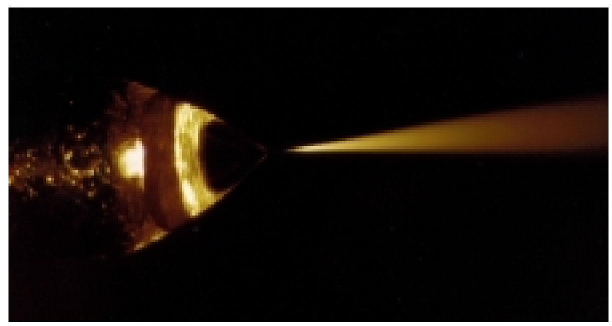

Figure 16 Typical electrospray jet in vacuum. The electrospray needle, liquid meniscus, and emitted spray are observed in this picture. The outside diameter of the tube is 0.3 mm.

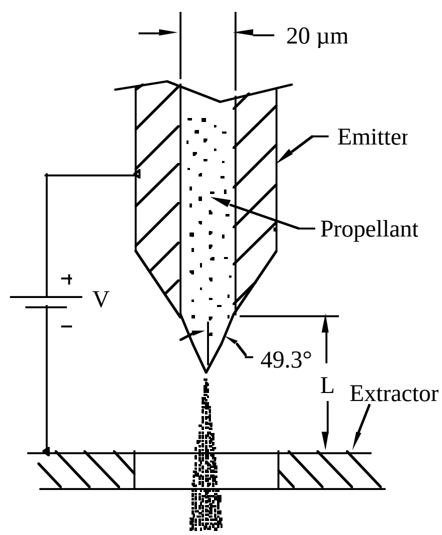

Figure 17 Schematic of the basic elements of a colloid thruster. A Taylor cone forms at the tip of the emitter upon application of sufficient voltage between the emitter and the extractor.

A typical single emitter needle thruster produces a maximum thrust of 0.5 µN. To achieve larger thrust, multiple needles are needed. To achieve a thrust range of 1–20 µN requires variation of the acceleration voltage and variation of the flow rate. The DRS microthruster is based on the first laboratory colloid thruster system recently delivered to JPL. It was designed to meet the requirements listed in Table 9. Figure 18 shows its overall schematic. The system consists of the thruster, neutralizer, propellant feed system and the power-processing unit (PPU). The PPU contains all the DC-DC converters to power the system and the autonomous controls for the carbon nanotube field emission neutralizer.

The feed system schematic is shown in Figure 19. It functions as follows. Upon command, the zeolite heater is supplied with a maximum of 4 W of power from the PPU. The zeolite, preloaded with CO_2, desorbs the gas, creating a CO_2 pressure that is exponentially dependant on the zeolite temperature. The CO_2 then pushes on the propellant-storing bellows increasing the propellant pressure. Upon opening the magnetic latching valve, the propellant is expelled at a rate linearly proportional to its pressure. The flow rate is lowered by allowing the zeolite to cool, which adsorbs the CO_2, lowering both the CO_2 and the propellant pressure. Cooling of the zeolite chamber is accomplished by radiation and conduction (heat sinking) into surrounding structure.

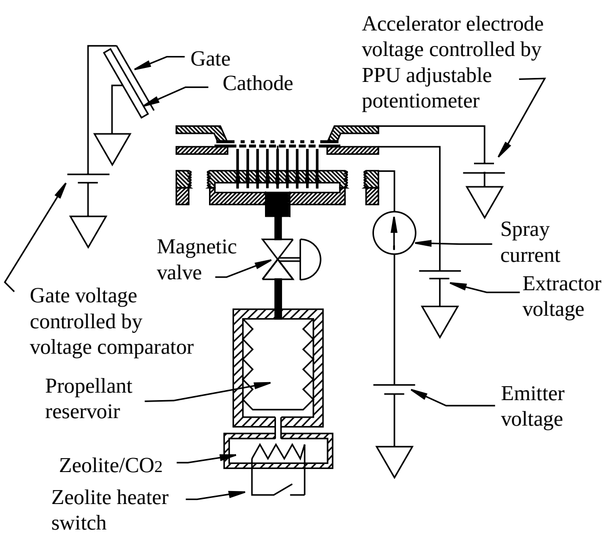

Figure 18 Schematic of the colloid thruster system.

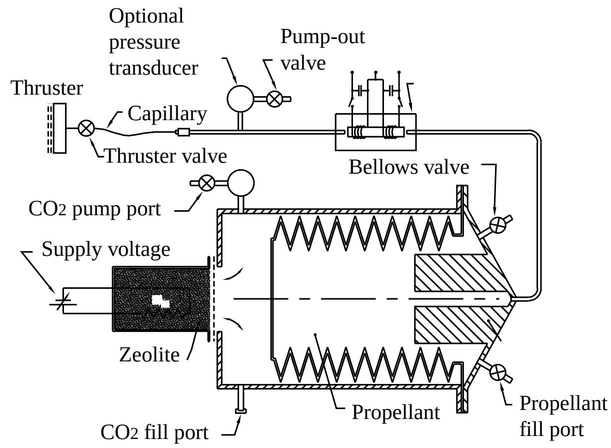

Figure 19 Schematic of the colloid thruster propellant storage and feed system

1.5.2.4 Microthruster Technology Readiness Level and Path to TRL 5

The existing prototype microthruster has been built to meet the DRS project requirements, but has not yet been tested fully as a system. Each component of the prototype thruster system has been subjected to tests in a laboratory environment that show the subsystems performed as expected. The individual TRL ratings of each subsystem are shown in Table 10, and a justification for these levels follows:

Emitter/Accelerator Assembly – The needles and accelerator electrodes have already been tested extensively at Busek. To date, over 100 hours of operation in vacuum have been accumulated on a single-tip emitter without incident. Direct measurements of thrust from a single-tip emitter have been made at Busek and agree with models of thruster operation. For the DRS project, the thruster must consist of an array of needles to meet the required thrust range. Busek has already tested a multiple-needle emitter design and integrated it into the prototype thruster system. These tests establish the current TRL of 4. More long-duration tests of the complete prototype thruster will be conducted at both Busek and JPL to reach TRL 5.

Propellant Feed System – The bellows and zeolite propellant feed system has already been tested in a laboratory environment at Busek and integrated into the prototype colloid thruster, establishing a TRL of 4. The feed system will be tested together with the rest of the thruster in long duration tests at Busek and JPL to establish TRL 5.

Table 10 Microthruster Path to Technical Readiness Level 5

Technology	Current TRL	TRL after Formulation Refinement	Method
Emitter/Accelerator Assembly	4.0	5.0	Assembly and test of breadboard
Propellant Feed System	4.0	5.0	Assembly and test of breadboard
Cathode Neutralizer	4.0	5.0	Assembly and test of breadboard
Power Conditioner	3.5	5.0	Assembly and test of breadboard

Cathode Neutralizer – The carbon nanotube field emission cathode neutralizer has already been tested in a vacuum chamber at Busek to TRL 4. During the formulation refinement phase, JPL will test a cathode in an ultra-high-vacuum (UHV) environment. These tests will continue through beginning of the brassboard development phase, when the cathode will be tested with the thruster in a UHV environment to establish TRL 5.

Thruster Electronics – A breadboard version of the power-processor unit using commercial hardware has already been integrated with the prototype thruster, consistent with a TRL of 4. A digital control interface has not yet been designed. During formulation refinement an updated breadboard model of the power-processor unit and a breadboard model of the digital control interface electronics will be assembled and tested at Busek to establish TRL 5.

1.5.2.5 Microthruster Validation Experiments and Success Criteria

The initial validation check-out and calibration of the individual thrusters will be done with one of the gravitational reference sensors operating in an accelerometer mode. Each of the microthrusters will be fired to sweep through its entire operating range while the accelerometer measures the resulting thrust. The thrust stability will be demonstrated by allowing the thrusters (one at a time) to operate at least 1000 seconds (typical data-taking period) while recording the accelerator output for subsequent analysis. The thruster calibration will be repeated several times during the DRS operations to see if any performance changes occur with time. The thruster success criterion is to show thrust of the desired range with the relevant noise performance. The lifetime will be investigated by performing the calibrations at several intervals during the nominal 90-day DRS operations period, but these additional tests are not part of the success criterion.

1.5.2.6 Microthruster System Characteristics

The microthruster mass, power, and volume requirements are listed in Table 11. The thrusters will be mounted in four clusters of two thrusters each at suitable points around the spacecraft.

1.5.2.7 Microthruster Risk Mitigation

A prototype thruster that should meet all of the DRS requirements has already been delivered to JPL. During the design refinement phase, the prototype thruster will be tested at Busek and JPL, which will mitigate potential technology problems and provide important information for any design modifications in the brassboard and flight model development phases.

Direct thrust measurements of a single-tip emitter have already been made at Busek using a 0.01 μN resolution torsion balance. These measurements confirm previous thrust models based on the electrode voltage and current values, and show that the colloid microthruster provides the requested thrust stability. In addition, a prototype thruster with multiple needles has already demonstrated the required power, specific impulse, and range of operation based on these models. More testing of the complete prototype will be conducted at both Busek and JPL to ensure that mission requirements, including operational lifetime, are met. The thrust delivered by the breadboard, brassboard, and flight units will be measured directly with a torsion balance.

Problems may arise in the following areas that have not yet been thoroughly tested: propellant feed system, emitter erosion, and cathode neutralizer operation. These are summarized in Table 12 and discussed briefly below.

Accurate Propellant Feed System Control – Accurate control of the propellant feed rate over the entire mission is necessary to meet the thrust range and resolution requirements. The nominal design uses a pressurized bellows that contains the propellant and controls the flow rate. The bellows movement, in turn, is controlled by the release of CO_2 from heated zeolite. A closed-loop feedback control circuit will monitor the

Table 11 Microthruster Resource Requirements

Item	Mass, kg	Power, W
Thruster+Neutralizer (1 of 2)		
Needle Array Asssembly	53	0.0
Neutralizer	30	0.0
Array structure	20	0.0
Harness and Connectors	20	0.0
Latching Valve	50	0.0
High Voltage Isolator	50	0.0
Enclosure	25	0.0
Misc. Hardware (10% of above)	25	0.0
Subtotal	*273*	*0.0*
Power Processor (1 of 2)		
Beam Converter	100	1.0
Extractor Converter	75	0.5
Accelerator Converter	75	0.5
Neutralizer Gate Converter	75	0.5
Zeolite Heater Converter	150	2.5
Valve Driver	50	0.2
Microcrocontroller	100	1.0
Enclosure and Connectors	50	0.0
Subtotal	*675*	*6.2*
Feed System (1 of 2)		
Propellant Feedsystem	159	0.0
Tank	50	0.0
Bellows	50	0.0
Zeolite canister	20	0.0
Zeolite	13	0.0
CO2 Gas	1	0.0
Filter	10	0.0
Fill/Drain Ports	40	0.0
Valve	150	0.0
Pressure Transducer	100	0.0
Misc. Hardware (10% of above)	59	0.0
Subtotal	*651*	*0.0*
Cluster Housing	250	0.0
Total (Cluster of 2 thrusters)	**3449**	**12.4**

beam current and control the zeolite temperature. This feedback control is yet to be tested in laboratory, but manual control experiments have worked as expected. The effectiveness of the control loop and lifetime of the system after repeated cycles and varied throttle levels will be investigated during design development tests at Busek and the life test at JPL. An alternative design, using a precision regulation valve, will also be investigated. If for any reason neither of these pressure-driven feed systems deliver the expected performance, an alternative strategy will be developed based on the use of the electrostatic pressure at the emitter tips to drive the propellant flow. This method is similar to the propellant feed mechanism used in field emission electric propulsion (FEEP) thrusters.

Erosion of Emitter Tips and Electrodes – Degradation of the emitter tips and/or electrodes could lead to thruster failure or a degradation of performance during the required operational lifetime (>2000 hrs). To mitigate this risk, two separate long-duration tests of the prototype thruster will be conducted at Busek and at JPL, each accumulating more than 2000 hrs of nearly constant operation. During the tests, images of the emitters and electrodes will be examined along with careful electron microscope inspections before and after the test to determine wear. The thruster performance and ion beam profiles will also be measured during the test. Performance degradation could be caused by electrochemical corrosion on the tips of the emitters or by sputtering of the grids and emitter tips from the highly energetic ion beam and secondary emissions. If necessary, electrochemical erosion rates can be reduced by changing the needle material and/or the electrolyte and propellant combination. Sputtering damage could be mitigated or eliminated by changing the design of the grid electrodes.

Cathode Neutralizer Operation – A new type of field emission cathode (carbon nanotube) will be used to neutralize the colloid ion beam. This device has already been integrated and tested with a thruster at Busek and was shown to perform well. Additional tests are being conducted at JPL to characterize the emission over a longer duration. The neutralizer will also be tested as part of the prototype thruster to ensure that thruster operation does not degrade performance over the required lifetime. If necessary, the extraction gate voltage and/or active cathode area could be changed to meet the requirements.

Table 12 Microthruster Technology Risks and Mitigation Strategies

Risk	Likelihood	Severity	Mitigation
Propellant Feed System Control	Low	Medium	- Test existing prototype - Investigate electrostatic propellant control
Erosion of Emitter and Electrodes	Medium	Low	- Test existing prototype - Investigate alternative materials
Cathode Neutralizer Operation	Low	Low	- Test existing prototype - Modify extraction gate or cathode area

1.5.3 Laser Interferometer

1.5.3.1 Interferometer Overview

The DRS includes a laser interferometer for validation of the gravitational reference sensors. While the laser interferometer is not a technology requiring spaceflight validation, its performance is key to achieving the DRS goals.

The interferometer design chosen is a simple homodyne system with an unmodulated laser beam. This design has been chosen to give adequate performance within the ST7 project cost constraints. A heterodyne interferometer with better performance was considered during the concept study but would have raised the project cost by $4M-$5M.

1.5.3.2 Interferometer Design

The homodyne interferometer is based on a simple Michelson interferometer, and compares the distance between the two GRS test masses with the length of a stable reference arm. The reference arm is formed by two mirrors on the optical bench. The stability of the reference arm comes through using a material of low thermal expansion for the optical bench combined with an environment of high thermal stability. The thermal stability requirements are driven by the need to keep thermal radiation force variations on the test masses sufficiently small; the interferometer thermal stability requirements are less stringent.

Figure 20 is a schematic diagram of the interferometer. Light from the laser is brought to the optical bench via a fiber. The laser beam is split so that half goes to each of two simple Michelson interferometers which measure the change in distance between each test mass and the optical bench. For each test mass, light from the laser is divided at a beam splitter with half reflected off the test mass and half reflected off a reference mirror. The beams recombine at the beam splitter and are imaged on a quadrant photodiode. A change in position of the test mass causes the intensity of the beam on the photodector to change. The change in intensity is recorded by an analog-to-digital converter in the GRS computer electronics. This double-Michelson design allows separate readouts for the two test masses. Quadrant detectors are used to measure the angle between the

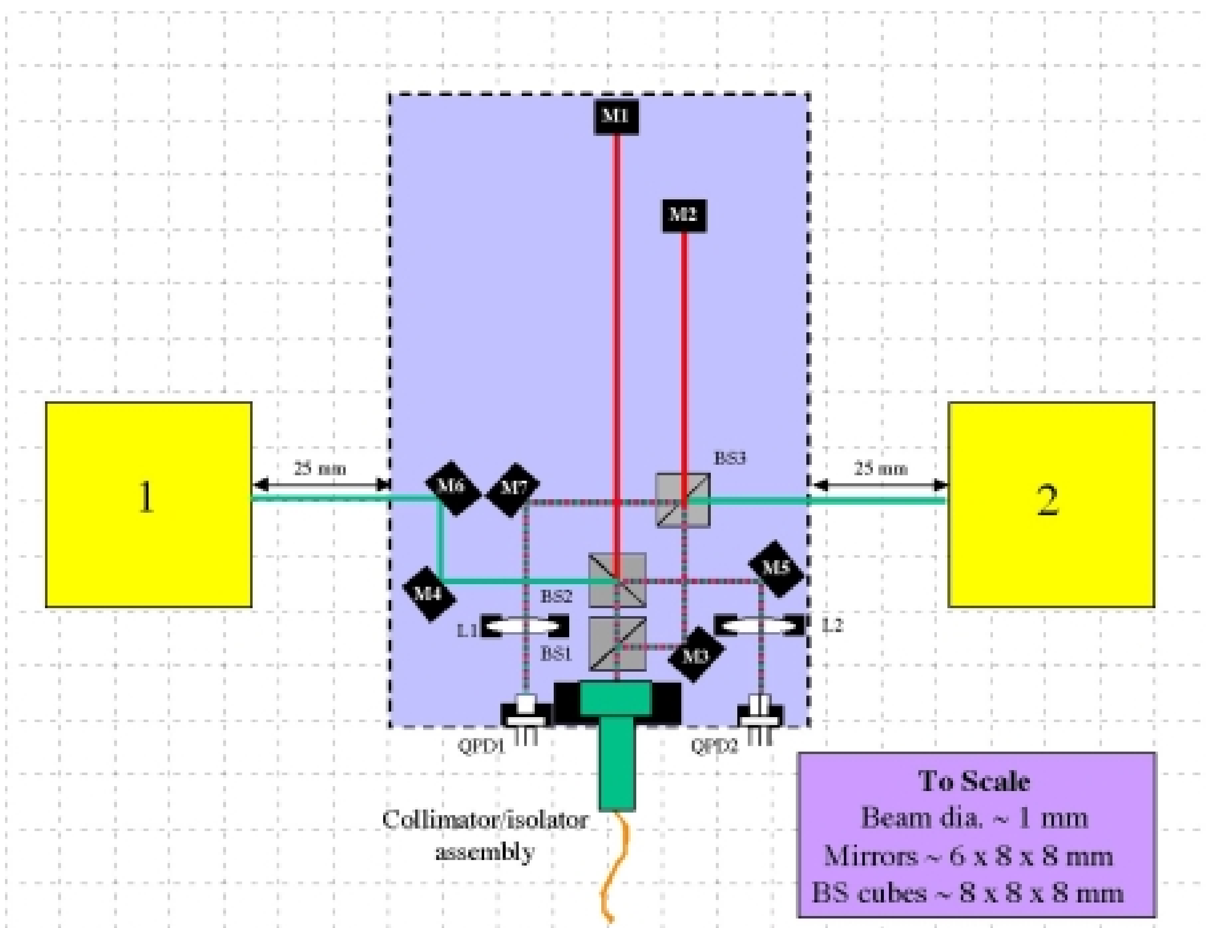

Figure 20 Scale diagram of the DRS homodyne interferometer

normal to the test mass surface and the laser beam direction. The pointing information is used to adjust the orientation of the test mass by applying voltages to the orientation control electrodes.

1.5.3.3 Interferometer Performance

The unstabilized homodyne interferometer is simpler to construct than is a frequency-stabilized heterodyne interferometer, since no electro-optical modulation is used and no phase measurement electronics are needed. This simplicity comes at the price of being more sensitive to laser power and frequency fluctuations and requiring the distance between the test masses to be near a nominal value within a fraction of the wavelength of the laser light. With the homodyne system, the expected distance measurement sensitivity is 50 pm/√Hz at a frequency of 10 mHz, and 500 pm/√Hz at a frequency of 1 mHz.

The acceleration performance requirements are set mainly by the LISA mission. Figure 21 shows the characteristic LISA sensitivity curve along with various acceleration noise levels. The DRS goal is to validate acceleration noise performance within a factor of ten of the LISA performance goal over the frequency range 1 mHz-10 mHz. In addition, the DRS acceleration noise performance will be characterized to determine the relationship between environmental parameters and acceleration noise. This characterization will provide confidence that an extrapolation of the DRS system to a quieter spacecraft environment will meet the LISA requirements. The homodyne interferometer can validate the level of acceleration noise to the DRS baseline requirement.

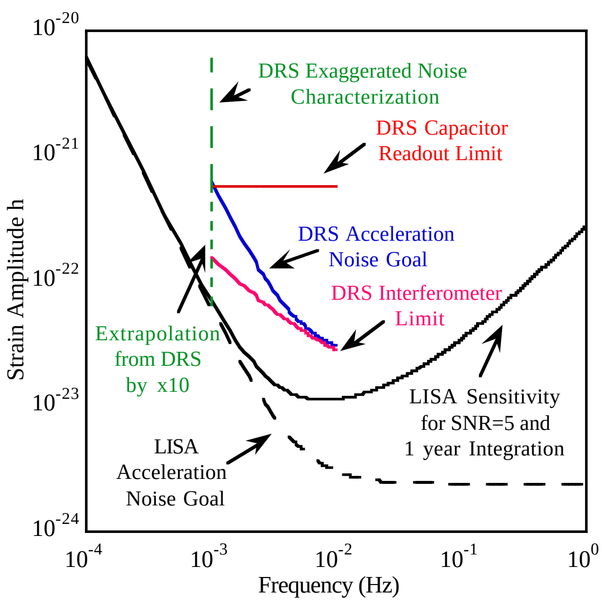

Figure 21 LISA sensitivity with various levels of acceleration noise

Also shown in Figure 21 is the acceleration performance validation level achievable without any interferometer. With no interferometer, the distance between test masses would be measured by the GRS electrostatic sensing system. This sensing will be used for spacecraft position and attitude control. As an acceleration noise validation, the electrostatic sensing is near the DRS acceleration noise goal at 1 mHz but far above the goal at 10 mHz.

1.6 Spacecraft

The DRS project does not include construction of a spacecraft. Instead the DRS will be a payload attached to either the ESA SMART-2 spacecraft or the NASA StarLight spacecraft. The spacecraft will provide power and interface to the ground. The DRS consists of the instrument package, instrument electronics, and microthrusters. The instrument package and instrument electronics can be distributed around the spacecraft to best suit the needs of the spacecraft. The four microthruster clusters need to be mounted at four points around the perimeter of the spacecraft.

The ESA SMART-2 mission has just begun two parallel industry studies for definition of the spacecraft. Specific configuration drawings are not yet available. ESA plans to down-select to one contractor in March 2003. An example of how the microthrusters could be mounted on a simple spacecraft is shown in Figure 22.

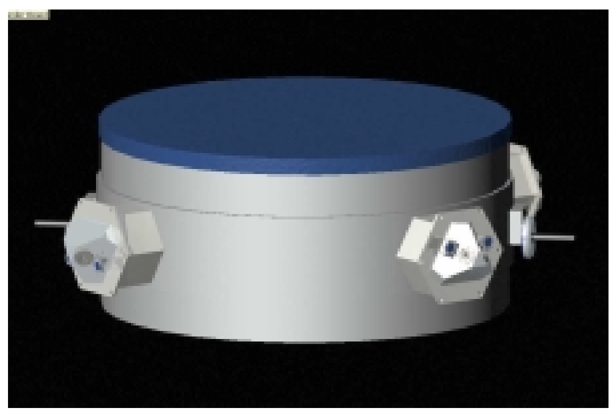

Figure 22 Four clusters of microthrusters attached to simple spacecraft

The StarLight project consists of two spacecraft, the Combiner and the Collector. Both spacecraft will be built by Ball Aerospace. The Collector spacecraft has a smaller instrument package and therefore has volume available for mounting the DRS package. Figure 23 shows one configuration of mounting the DRS package on the Collector spacecraft.

During DRS operations the spacecraft will assume an attitude with appropriate illumination of solar panels and pointing of antennas to Earth. The spacecraft attitude control system would have a predefined dead band of ~1°. The DRS will operate to maintain spacecraft attitude and position with respect to the two test masses in the sensor package, combined with information from the spacecraft star tracker. The spacecraft should not have any parts moving or have and thruster firing during DRS operations. If the attitude dead band is exceeded, the spacecraft attitude control system will fire thrusters as needed to resume nominal positions and power down the DRS system.

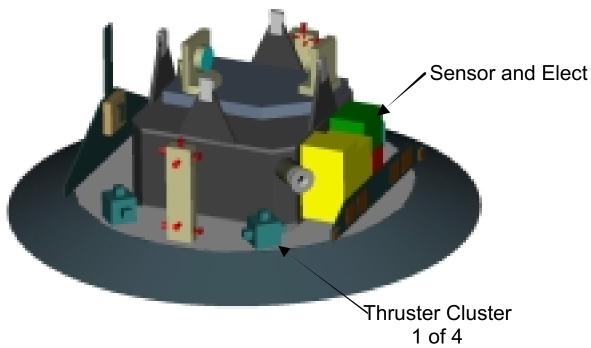

Figure 23 Configuration drawing of DRS sensor package, electronics, and thrusters on StarLight's Collector spacecraft

Data acquired during DRS tests will be stored in the GRS computer system and sent to the spacecraft over a standard interface for relay to the ground. The DRS data volume is modest, on the order of 1000 bits per second for several hours per experiment.

Aside from accommodating the DRS mass, power, volume, and data handling, the primary interface issue is the mass distribution. The DRS test masses require that the gradient in the gravity due to mass on the spacecraft be less than 10^{-6} N/m, which corresponds to a mass of 100 kg at a distance of 10 cm from the test mass. A gravity gradient of this magnitude is acceptable as long as it is known early during construction of the GRS system, since small balance masses near the test mass can be installed to counter the pull from the larger masses.

1.7 Access-to-Space Approach

The DRS will be launched in conjunction with either the ESA SMART-2 spacecraft or the NASA StarLight spacecraft. The SMART-2 launch is the baseline option, since the SMART-2 mission is primarily focused on testing drag-free technology. Flying the DRS on the SMART-2 mission will allow real-time comparison of the DRS technology to European analogue technology, and can help determine sources of problems. ESA has invited NASA to fly DRS technology on SMART-2 (Appendix 6). A Letter of Agreement (LOA) signed between NASA and ESA allows exchange of interface information (Appendix 6). A more detailed LOA will have to be developed and signed by the end of Formulation Refinement to allow shipment of DRS hardware and exchange of performance data.

Launch of the DRS on the NASA StarLight project is a backup option. The StarLight project has sufficient mass, power and volume to accommodate the DRS. Since the DRS would be a secondary payload on the StarLight project, DRS operations would nominally take place at the end of the primary StarLight mission six months after launch. Flying the DRS on StarLight would have some advantages in technology synergy, since StarLight is a separated-spacecraft interferometer as are several future DRS target missions. Inclusion of the DRS microthrusters on StarLight would allow demonstration of interferometer performance using precision spacecraft position control. The microthrusters could also allow longer integration times with smoother spacecraft control, as well as possible continued observations during reconfiguration of the spacecraft constellation.

Final decision of which platform to launch the DRS on will come during Formulation Refinement and at the end of the SMART-2 Industrial Phase-A study. During the spring of 2002, design discussion will be held between the DRS and SMART-2 teams to determine interfaces and allocate resources. Unless problems arise in terms of capability, the DRS will launch on SMART-2, with the decision to be documented in the final ESA-NASA LOA.

1.8 Technology Validation and Infusion Approach

1.8.1 Technology Plan and Review

During the Formulation Refinement Phase, the two DRS technologies will be developed as discussed in sections 1.5.1.4 and 1.5.2.4. Since both technologies can be tested in only limited ways, the development will include specific tests of individual subsystems, functional system-level tests to show that the subsystems work together, and modeling to estimate the performance in space. In addition, a numerical system simulation will be developed to model the system performance of the DRS. At the beginning of Formulation Refinement, a Technology Working Group (TWG) will be formed to function as a peer review panel. The DRS team will develop a Technology Development Plan describing the planned ground testing and modeling effort for review by the TWG. Near the end of Formulation Refinement, a Technology Validation Report will be prepared based on the results of the program. This will be peer-reviewed by the TWG prior to the project Confirmation Review.

1.8.2 On-Orbit Validation

On orbit, a series of test sequences will be performed. A summary of each type of test is given below. During each test sequence, measurement will be acquired from the gravitational sensors and the interferometer, along with diagnostic data from these devices and from the microthrusters. The test data generally consists of a series of measurements of distance, orientation, and acceleration. The acceleration data will be used to evaluate the performance of the microthrusters, with performance given explicitly by the acceleration level and noise corresponding to the commanded thrust. The measurements of the distance between test masses by the interferometer will be differenced to determine the level of acceleration noise on the test masses. This will directly determine the best acceleration noise achieved. In addition, test mass acceleration will be measured during application of specific periodic disturbances, such as temperature changes, to validate the models that predict accelerations in response to disturbances.

The DRS Level 1 requirements imply specific performance levels for the key subsystems. Table 13 shows the required range and resolution of these subsystems. The GRS range is determined by the gap between test mass and enclosure. The resolution is a factor of three below the overall requirement, to accommodate combined noise from several degrees of freedom. For drag-free control, the response scale factor of the GRS and the microthrusters must be known to a precision of approximately 10%.

Table 13 Range and resolution of advanced technologies for DRS

Sensor/Actuator	Range	Resolution
GRS sensor	1 mm	1 nm/√Hz
GRS actuator	1 mm	10^{-14} m/√Hz
Interferometer (sensor)	100 nm	10^{-14} m/√Hz
Microthruster (actuator)	20 µN	0.01 µN/√Hz

The test sequences will verify functionality, scale factor calibration, range, and resolution of each subsystem, operating individually or in comparison with other subsystems.

Each test sequence is assumed to take place over a three-day period between telecommunications sessions. The nominal test sequence is planned over 90 days of operation, with a 30 day schedule reserve. The schedule reserve is considered necessary to account for possible problems in operations of the individual subsystems, and for possible need to update models to account for unexpected interactions among the various subsystems. The schedule reserve will also allow for maneuvering from the postlaunch orbit into the desired operational orbit, in case a boost from GTO is necessary. The nominal test sequence is outlined below.

Sequence 1: Initial systems check

At the start of DRS operations, the spacecraft will establish the desired attitude, with the solar panels pointing towards the sun and one low-gain antenna pointed towards Earth. The GRS and interferometer systems will be powered up, with the test masses remaining mechanically caged. The interferometer system will read out the operation of the laser and the positions of the caged test masses to validate the interferometer functionality. The positions of the caged test masses will be read out by the GRS sensors to verify functionality of the readout electronics. The GRS readout and the interferometer readouts will be compared to verify resolution, though at a level considerably poorer than specified in Table 9.

Sequence 2: Establish accelerometer mode

One of the test masses will remain caged, and the other one will be uncaged. The GRS sensors will determine the position and orientation of the uncaged test mass, and the GRS electronics will hold it centered in its housing by application of voltages to the electrodes on the reference housing. In this mode, the GRS will act as an accelerometer. The control voltages will measure the acceleration of the spacecraft in six degrees of freedom.

Sequence 3: Microthruster calibration

The microthrusters will be powered on and run through a sequence where each thruster is programmed to fire at a given thrust for a few minutes, varying over the range of the thrusters. The GRS in accelerometer mode will record the induced spacecraft acceleration. The interferometer will measure the separation between the caged and uncaged test masses, as a one-degree-of-freedom consistency check of the GRS accelerometer mode calibration.

Sequence 4: Force modeling definition

Using the calibrations of induced acceleration versus commanded thrust, the coefficient matrix relating GRS sensor signal to required thrust for drag-free control will be updated.

Sequence 5: Initial drag-free operation

The parameters for drag-free operation will be uplinked at the start of the sequence. The spacecraft will transition to drag-free mode, in which the microthrusters will fire in order to keep the spacecraft centered about the freely-floating test mass. The second test mass will remain mechanically caged. The interferometer will measure the position of the freely-floating test mass against the caged test mass, to validate the drag-free sensor position readout. Since one of the masses is caged, the readout will include the noise associated with spacecraft motion; that is, the resolution requirement will not be verified at this step.

Sequence 6: Spacecraft position control evaluation

At the conclusion of the initial drag-free operation cycle, the test mass of the second drag-free sensor will be mechanically uncaged. Control voltages will be applied to keep the test mass centered in its housing while the spacecraft remains in drag-free operation about the first test mass. The spacecraft will be following the first test mass, and the second test mass will be following the spacecraft. The accelerometer readout should be consistent with the Table 2 position noise requirement.

Sequence 7: GRS Resolution cross-check

Sequence 6 will be repeated, with the roles of the first and second test masses reversed. The GRS sensor signals will be compared with those of Sequence 6 to check for consistency and to determine the unbalanced forces caused by asymmetry in the spacecraft.

Sequence 8: Test mass acceleration noise evaluation

The two test masses will float as freely as possible, with the spacecraft operating to keep centered around one of the test masses. The distance between the two test masses will be measured by the interferometer. The noise in the interferometer readout should be consistent with the Table2 force noise requirement.

Sequence 9: Dual-drag-free evaluation

The data from the drag-free sequence will be downloaded and evaluated to determine the relative trajectories of the two test masses. Any errors or unmodeled effects will be evaluated so that corrections can be uploaded. The spacecraft will remain in the dual-drag-free state if no problems have occurred; otherwise the spacecraft will resort to accelerometer mode for both sensors. Successful completion of this sequence will meet the DRS level 1 requirements.

Sequence 10: Microthruster calibration 2

The microthruster-level control sequence will be repeated with both sensors in accelerometer mode. This will allow for investigations into any time-variation in the performance of the thrusters after several weeks of continuous operation.

Sequence 11: Drag-free cross-check 2

One test mass will be used for drag-free operation, with the other used in accelerometer mode to measure the response of the thrusts needed for control. The results will be used to redetermine any control parameters for dual-drag-free operation.

Sequence 12: Dual-drag-free operation 2

A longer period of dual-drag-free operation will be initiated to evaluate the inertial sensor performance over a longer time span. This will allow characterization of the system performance over longer times scales.

Sequence 13: Drag-free/drift operation

In this mode, both sensors will operate with no control voltages applied to either test mass. The test masses will freely drift. The spacecraft will be commanded to remain centered about one of the test masses. Due to asymmetries in the spacecraft, it is expected that the second test mass will come close to the housing after a few hundred seconds. When that occurs, the sensor electronics will apply voltages to re-center the test mass and then removed to allow the second test mass to again float freely. This mode will allow investigation of the limits to performance in the absence of control voltages needed for two sensors on one spacecraft, which might indicate improved performance for missions that need only one sensor on a spacecraft. This mode will also allow investigation of the interferometry system over a range up to the level in Table 9.

Sequence 14: Drag-free/drift evaluation

The data from the drag-free/drift mode will be analyzed to determine the system performance in the absence of the small control voltages normally used in dual-drag-free mode. Any needed changes for this mode to work successfully will be uploaded at the next sequence.

Sequence 15: Drag-free/drift operation 2

Sequence 13 will be repeated, with modified control parameters as necessary.

Sequence 16: Thermal disturbance evaluation 1

This is the first of a series of tests to measure levels of disturbance to the inertial sensors based on different types of environmental factors. In this test, a heater near one GRS will be run with a sinusoidal heat variation (via a controlled voltage) with a 100-second period. The spacecraft and GRS units will be operated in dual-drag-free mode. The difference in test mass trajectories, measured by the interferometer, will be measured over the three-day sequence to allow evaluation of the effect of thermal disturbances.

Sequence 17: Thermal disturbance evaluation 2

A thermal disturbance will be applied with 1000-second period to determine the effect on GRS performance.

Sequence 18: Magnetic disturbance evaluation 1

A sinusoidal magnetic field will be applied with period 100 seconds to evaluate the effect of varying spacecraft magnetic fields on GRS performance.

Sequence 19; Magnetic disturbance evaluation 2

A sinusoidal magnetic field will be applied with a 1000-second period to evaluate the effect of varying spacecraft magnetic fields on GRS performance.

Sequence 20: Charge disturbance evaluation 1

The charge on one of the test masses will be allowed to grow to a larger level than normal. The discharge system will keep the charge near the desired level, to evaluate the effect of a test mass charge on the GRS performance.

Sequence 21: Charge disturbance evaluation 2

The charge on one of the test masses will be allowed to grow to a larger level than allowed in sequence 21. The discharge system will keep the charge near the desired level, to evaluate the effect on the GRS performance.

Sequence 22: Gravitational disturbance evaluation 1

The spacecraft control system will be modified to produce a periodic error in the position of the spacecraft with a sinusoidal motion in the direction of the line connecting the two test masses and with period 100 seconds.

Sequence 23: Gravitational disturbance evaluation 2

The spacecraft control system will be modified to produce a periodic error in the position of the spacecraft with a sinusoidal motion in the direction of the line connecting the two test masses and with period 1000 seconds.

Sequence 24: Gravitational disturbance evaluation 3

The spacecraft control system will be modified to produce a periodic error in the position of the spacecraft with a sinusoidal motion transverse to the direction of the line connecting the two test masses and with period 100 seconds.

Sequence 25: Gravitational disturbance evaluation 4

The spacecraft control system will be modified to produce a periodic error in the position of the spacecraft with a sinusoidal motion transverse to the direction of the line connecting the two test masses and with period 1000 seconds.

Sequence 26: Microthruster calibration 3

The two GRS units will be placed in accelerometer mode, and the thrusters again exercised throughout their dynamic range to evaluate any changes in their performance after 3 months of continuous operation.

Sequence 27: Dual drag-free operation 4

The GRS units will be placed in dual-drag-free mode to evaluate performance over a 6-day period.

1.8.3 Data Archival and Dissemination

The DRS data consists of a time series of measurements of distance, orientation, thrust, and other diagnostics. The data rate is low, one to ten times per second, so the total data volume is rather small. The data will be formatted as simple ASCII text tables. The data will be distributed to the DRS team via e-mail attachment. After calibration, the data will be made available to the community through a web site.

Data analysis will be performed by examining the time series of the GRS and interferometer to look for consistency and appearance of any spurious signatures. Smooth time periods of data will be Fourier transformed to determine the power spectral densities of the acceleration noise. The level of acceleration noise is the end result of the data analysis. For periods of data where no noise terms are introduced, the acceleration noise will indicate whether the requirements for future missions have been demonstrated. For data with noise forces added, the relationship between forcing function and resulting acceleration will be determined. These transfer function will allow extrapolation in performance to missions with quieter spacecraft.

After the end of the mission all data will be archived in the Planetary Data System. The DRS team members will produce a Technology Validation Report and will also present the results at relevant scientific and engineering conferences and published papers in professional journals.

1.8.4 Relevance of Validation for Infusion

The DRS is designed to measure specific performance parameters of technology subsystems that are characteristic of the flight units and of direct interest to future missions. The DRS Technology Working Group will include members from each candidate future flight mission, so that they can interact with the DRS team during development and after validation. The DRS project will result in test flights of technology subsystems that can be directly incorporated into most of the candidate future missions, as well as validation of operation of the precision spacecraft control.

1.9 Project Development Plan

1.9.1 Overview

During Formulation Refinement, breadboard models of all DRS components will be developed. The breadboard models will be used to prove technology readiness at TRL 5 and test compatibility between the DRS subsystems. DRS preliminary design review and confirmation assessment review will be held near the end of Formulation Refinement in February 2003. During the first half of the Implementation phase, high-fidelity brassboard models will be built to finalize part selection and to test the mechanical design of the instrument assembly and thrusters. Subsystems with particular concerns will have full-fledged engineering models built for vibration and environmental testing. Electronics will generally not be developed to the EM level. Critical Design Review will be held near the end of the brassboard developments, followed by the start of flight system development in May 2004

DRS subsystems will be completed in February 2005 and shipped to JPL for integration and test. The DRS subsystems will be developed as proto-flight models and tested to suitable levels for thermal, vacuum, and vibration prior to shipment. During Integration and Test the DRS subsystems will undergo final assembly and a vibration test will be performed at the system level. The DRS will be shipped to the spacecraft integration site for integration onto the spacecraft.

The DRS subsystem development is described briefly below. Each subsystem is responsible for functional and environmental testing of hardware and software prior to delivery to JPL for integration and test. Each subsystem will also provide electronics and software support for integration and test and operations.

1.9.2 Gravitational Reference Sensors

The Gravitational Reference Sensor subsystem is responsible for the two GRS units, the Digital Control Electronics, Low-Noise Electronics, and the Charge Control subsystem. GRS Control Software and Command and Data Handling Software are also part of the GRS subsystem, with the latter responsible for interface to the spacecraft computer, interface and control of the thrusters, and readout of the Interferometer. A Dynamic Model will be developed (In Matlab computer language) to emulate the performance of the GRS subsystem for testing in a Dynamic Testbed. The GRS subsystem will be built by Stanford University.

The GRS units include the test mass, the reference housing, the caging mechanism, and the vacuum enclosure. The reference housing will be based on techniques adapted from GP-B. The test mass will be made of a gold-platinum alloy with high density and low magnetic susceptibility. This material was used in the DISCOS sensor built by Stanford and flown in the Triad mission. For DRS the test mass has stringent electrical loss requirements and must reflect the laser beam for the Interferometer. Test samples of the AuPt alloy are available. During Formulation Refinement tests on the samples will be made to measure the electrical losses, machinability, interaction with a caging mechanism, and laser beam reflection properties. A breadboard GRS unit will be developed to check design features and the performance of the capacitive sensor readout. A breadboard computer will be built with software to check interfaces to the GRS unit. During Implementation and Engineering Model GRS unit will be developed for vibration and thermal/vacuum testing. Two flight units will be developed and tested for function, vibration, and thermal/vacuum performance before shipment to Integration and Test.

The Low-Noise Electronics will measure the position and orientation of the test masses using capacitance bridges. The Low-Noise Electronics will be based on the GP-B electronics. A breadboard unit will be built for testing with the GRS breadboard and brassboard units. A flight unit will be built and tested function, vibration, and thermal/vacuum performance before shipment to Integration and Test.

The Digital Control Electronics include a computer and interface boards. The interfaces include a custom analog interface to the Low-Noise Electronics developed by GP-B, RS422 interfaces to the Laser and Microthrusters, a 1553 interface to the spacecraft computer, and analog-to-digital converters for readout of the Interferometer detectors. During Formulation Refinement an assessment of available space-qualified computers will be made and one selected for implementation. A breadboard unit will be built for testing purposes and software development. During Implementation a flight unit will be developed and tested for function, vibration, and thermal/vacuum performance before shipment to Integration and Test.

The Charge Control subsystem consists of an ultraviolet light source (mercury lamp) with switches and fiber to feed light at intervals to the GRS unit. The Charge Control is based on the subsystem developed for GP-B. During Formulation Refinement, a breadboard unit will be built for testing. During Implementation, a flight units will be developed and tested for function, vibration, electronic interference, and thermal/vacuum performance before shipment to Integration and Test.

1.9.3 Microthruster

The Microthruster subsystem is responsible for effecting control of the spacecraft position in response to the Dynamic Control software in response to outputs from the Gravitational Reference Sensors. The Microthruster subsystem include the Thruster Head, the Propellant Feed, the Cathode, and the Power Processing Unit. The Thruster Head consists of the array of needles and the set of electrodes which eject and accelerate the propellant. The Propellant Feed includes a propellant reservoir and a pressuring system for feeding the propellant to the Thruster Head. The Cathode emits electrons to match the positive charge expelled by the Thruster Head to leave the spacecraft uncharged. The Power Processing Unit is the control electronics which interfaces the computer, determines the appropriate pressure and voltage levels, and generates the electrical drive signals. The Microthruster subsystem will be built by Busek Co., Inc.

During Formulation Refinement, the prototype Thruster Head, Propellant Feed, and Cathode will be thoroughly tested. A breadboard Power Processing Unit will developed for testing the prototypes. A mathematical (Matlab) model will be developed for inclusion in the Dynamic Control test bed. During Implementation, an engineering model Thruster Head, Cathode, and Propellant Feed will be built and tested for function, vibration, and thermal/vacuum performance. The breadboard Power Processing Unit will be shipped to Stanford for compatibility testing with the GRS computer. Four flight Thruster Clusters will be built, with 2 Thruster Heads and Cathodes per cluster. Eight flight Power Processing Units will be built. All flight units will be tested for function, vibration, electrical interference, and thermal/vacuum performance before shipment to Integration and Test.

1.9.4 Interferometer

The Interferometer is responsible for measurement of the distance changes between the two GRS test masses. The Interferometer includes the Optical Bench, the Detector Amplifiers, and the Laser. The Interferometer will be developed by JPL's Observational Systems Division

A breadboard Interferometer and breadboard Detector Amplifiers will be built during Formulation Refinement. Sample optical benches and beam splitters are already available. The breadboard tests will validate the performance of the design and confirm selection of electronic components. During Implementation a Engineering Model optical bench will be constructed and attached to the EM Reference Block. The assembly will subjected to vibration tests to verify performance and alignment requirements. The flight Interferometer and Detector Amplifiers will be built and tested for function, vibration, and thermal/vacuum before delivery to Integration and Test.

A single-mode Nd:YAG laser with wavelength 1 µm will be used for the Interferometer. The baseline plan is to procure a commercial space-qualified from the Lightwave company developed for the TES instrument. Alternative laser sources will be investigated, including commercial lasers from the Bosch company or a possible shared laser development with the StarLight and SIM projects.

1.9.5 Dynamic Control Software

The Dynamic Control Software (DCS) subsystem is responsible for determining the thrust level for each microthruster for control of the spacecraft position based on the capacitive sensor measurements from the gravitational reference sensors and information from the spacecraft star tracker. The DCS serves to "close the loop" of the overall DRS control system necessary to keep the spacecraft centered on the two GRS test masses. Assuming the nonlinear effects are not significant in the dynamics of the spacecraft-proof-mass interactions, the DCS will consist of a Linear Quadratic Gaussian controller which includes a Kalman Filter estimator. The Dynamic Control Software will be developed by the Goddard Space Flight Center's Guidance, Navigation and Control Center.

A dynamic model for the DRS, with 18 degrees of freedom, will be developed using the industry standard MATLAB/*Simulink* dynamic system simulation software. This dynamic model will incorporate a detailed dynamical model of the GRS from Stanford, a dynamic model of the colloidal thruster from Busek, rigid-body dynamics of the spacecraft based upon data from the SMART-2 or StarLight project, and the controller/estimator dynamics.

The mathematical for of the controller/estimator algorithms will be defined, tested, evaluated, refined and de-bugged using the end-to-end DRS dynamic model. The detailed mathematical definition of DCS algorithms (including the precise specification of the input/output data for each algorithm) will be captured in an DCS flight software requirements. These algorithms will subsequently be coded into flight software, in the C language, using GSFC's flight proven development practices and techniques. Closed loop testing of the DCS flight software will be performed in a software development test bed environment.

In the envisioned DRS flight software architecture the DCS will be executed as a single task on the GRS computer while all external interfaces to the ACS software will be managed through appropriate C&DH function calls. An Interface Control Document (ICD) will be written to define all the software interfaces between the GSFC DCS flight software and the Stanford C&DH flight software. The DCS flight software development can therefore proceed independent of the development of the C&DH flight software at Stanford. Integrated testing of the complete DRS flight software package (the GRS electrostatic sensing and control flight software, the DCS flight software and the C&DH software) will be performed at Stanford in an integrated test environment. As described in Section 1.9.7 (Integration and Test) both open-loop and closed-loop testing of the DCS flight software will be performed using a GRS hardware simulator, a thruster hardware simulator and breadboard GRS computer hardware in an integrated software environment.

1.9.6 Structure

The Structure subsystem is responsible fore development of the Reference Block, the Thermal Enclosure, an instrument Structural/Thermal Model, and Cables between subsystems. The Structure subsystem work will be performed by JPL's Mechanical Systems Engineering Division.

The Reference Block is a precision structure to which the two gravitational reference sensors and the interferometer is attached. It will be mechanically rigid and made of a thermally stable material. Nominally fused silica will be used, though quartz and other materials will be considered. A breadboard reference block will be made out of aluminum to test mechanical interfaces with the sensors and interferometer. Since the Reference Block is the central alignment system, an engineering model will be developed early in the Implementation phase. Test fixtures will be attached to the EM and vibration tests performed to check the alignment stability. The precision requirements are ~10 µm for location and ~0.5 mrad for alignment. As the spacer between the freely floating masses, the reference block does not have any nanometer-level requirements. The flight unit will be built and tested prior to delivery to Integration and Test.

The Thermal Enclosure will be used to isolate the gravitational reference sensors and interferometer from external temperature variations. In Formulation Refinement a thermal model will be developed to confirm performance predictions based on similar models with different configuration. A breadboard model will be built to test assembly and cabling interfaces. Since the Thermal Enclosure is required to minimize thermal conductivity between the boxes and the instrument, low-conductivity titanium struts forming an isostatic set will support the instrument assembly from the inner enclosure and support the inner enclosure from the outer enclosure. During Implementation an Engineering Model will be built and subject to vibration testing, followed by construction and test of the flight model.

For check of interfaces with the spacecraft, structural-thermal models will be built during Formulation Refinement for each assembly deliverable to the spacecraft, except the cabling. These models will have the proper mounting features, mass, center-of-gravity, external geometry, heaters and external surface finishes. They are suitable for spacecraft structural and thermal characterization, but not for structural or thermal characterization of the assemblies themselves.

Cables between the interferometer and the GRS computer, and between the microthruster and the computer are the responsibility of the Structure subsystem. During Formulation Refinement simple commercial-quality cables will be used. During Implementation, a mockup of the DRS layout on the spacecraft will be built, and flight cables developed and tested.

1.9.7 Integration and Test

The DRS Integration and Test (I & T) activity will include integration of all DRS subsystems, functional and environmental testing of the entire system, shipment to the spacecraft integration site, and integration onto the spacecraft. The I & T activity assumes that each subsystem has been integrated at the subsystem level. I & T will be performed by JPL's Observational Systems Division.

Prior to shipment of subsystems to JPL for final I & T, the gravitational sensor subsystem and microthruster subsystem will have completed integration and test at their subsystem level. The microthrusters will have been assembled and tested, bot functionally and environmentally, prior to shipment to JPL. The GRS subsystem I&T activity will include the gravitational sensors, the electronics, and software from Stanford and from the Dynamic Control Software effort at Goddard Space Flight Center. The sensor and electronics hardware will be subjected to environmental testing before shipment to JPL.

1.9.7.1 Software Integration and Test

Software I & T at Stanford will include incorporation of the flight dynamic control software from GSFC. The dynamic control software will be delivered to Stanford as C source code for compilation on the breadboard and/or flight GRS computer. Software testing at Stanford will include testing of software and hardware interfaces. Figure 24 depicts an envisioned DRS test configuration. In this test configuration the GRS computer will execute the full software system, including the command and data handling software, the GRS control software, and the dynamic control software. Software tests will be performed using simulators for both the gravitational sensors and the microthrusters. The simulators will consist of laboratory computer with electrical interfaces like the associated flight hardware.

The GRS simulator will generate simulated position readout signals corresponding to expected ranges of motion. The GRS computer will read the simulated position measurements, pass the measurements to the DCS software, take the resulting thruster commands and transmit the commands to the thruster simulator. The GRS computer will also generate correction signals for the GRS simulator. The

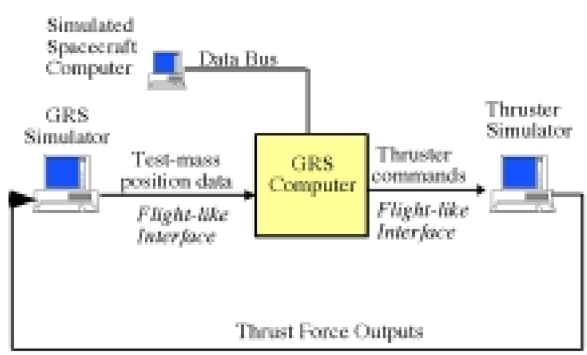

Figure 24 Software test configuration

thruster simulator will accept thrust level commands from the GRS computer, compute the appropriate control pressure and voltages for each thruster, and compute a simulated thrust. The thruster simulator will send monitor telemetry to the GRS computer.

The goal of the integrated flight software testing is to verify the proper overall DRS flight software operating environment, hardware-to-software interface compatibility, DRS command/telemetry functionality, DRS modes and mode transitions, operational compatibility between flight software elements, task timing and execution, time synchronization /maintenance, data latency, end-to-end functional data flows and CPU and I/O stress tests to verify processor and throughput margins.

The test configuration will be used to perform selected open-loop performance tests. In these tests a prescribed set of GRS test mass position/attitude signals would be generated and passed to the DRS breadboard Payload Processor for the computation of commanded thrust levels. These commanded thrust levels will in turn be passed to a breadboard thruster electronics, which will compute the necessary thruster-internal voltage/pressure commands. These commands will generate thruster control voltages which will be monitored. A simulated thrust output will be sent to the GRS simulator for closed-loop tests.

Closed-loop DRS testing will be performed in this environment using the same dynamic model used by GSFC to develop the dynamic control software. Thrust level outputs would be fed to the spacecraft/GRS dynamic model and the modeled test mass motions then included in the GRS simulator to generate new test mass position/attitude data for transmission to the GRS computer.

Pass/fail criteria will be established for these tests by specifying an allowable level of mismatch. Trouble shooting and debug activities would be supported through the use of multiple data ports/monitors not shown in Figure 24. In this manner the end-to-end performance of the DRS hardware/software system can be verified. Preliminary sensor-to-actuator polarity testing can also be conducted in this test environment.

1.9.7.2 Hardware Integration and Test

Hardware I & T will take place in two stages: DRS I & T, and spacecraft I & T. The DRS I & T flow is indicated in Figure 25.

The gravitational reference sensors will be delivered to JPL as integrated units. Upon delivery, the gravitational reference sensors and the interferometer will be assembled onto the reference block and mounted within the double-walled thermal enclosure. The instrument assembly will be attached to a mounting plate, to which the microthrusters and all electronics will also be mounted. The complete payload assembly will then be functionally tested. The functional testing of the instrument assembly is fairly limited since the gravitational sensor test masses will be mechanically caged during I & T. Functional testing will include:

- Operation of the sensor electronics to establish that the control interface and position sensing are operating properly
- Operation of the laser and interferometer to establish that they operate to measure the distance between the two caged test masses
- Operation of the interface between the GRS computer and the microthrusters

Following functional testing, the entire DRS assembly will be subject to environmental testing to ensure that the assembly is mechanically and thermally sound. After a second functional check, the DRS will be shipped to the spacecraft integration site, still mounted to the vibration test fixture.

At the spacecraft integration site, DRS personnel will be involved in acceptance testing of the DRS. DRS personnel will assist in dismounting the DRS from the test fixture and attachment to the spacecraft. Due to the limited tests possible in a 1-g environment, unpacking and post-shipment validation is expected to take only a few days, yet 3 months have been allocated. Similarly, launch support is expected to take no more than 6 weeks, but a full 3 months have been allocated. There is a 6-month or greater period between these activities where no JPL I&T work is planned.

1.9.8 Ground Systems and Data Processing

(The description here assumes that the DRS is launched on the ESA SMART-w2 spacecraft. If the DRS is instead launched on the StarLight spacecraft, DRS operations will be combined with StarLight operations.)

The DRS ground data system will consist of a small instrument operations center of one or two workstations at JPL. The DRS operations center will be the point of contact between the technology providers, who will aid in monitoring system performance, and the spacecraft operations center (in Europe).

The DRS will be launched with a number of specific test sequences planned and programmed in. DRS commanding will nominally be limited to determination of the start and stop times for each test sequence. The request for initiation of test sequences be sent by the DRS operations center to the spacecraft operations center. In addition to sequence initiation, it may be necessary to have new control tables prepared by the DRS technology team. The DRS operations center will be format and forward the updates to the spacecraft operations center for inclusion into command files for the spacecraft. The spacecraft operations center will generate command uploads for transmission

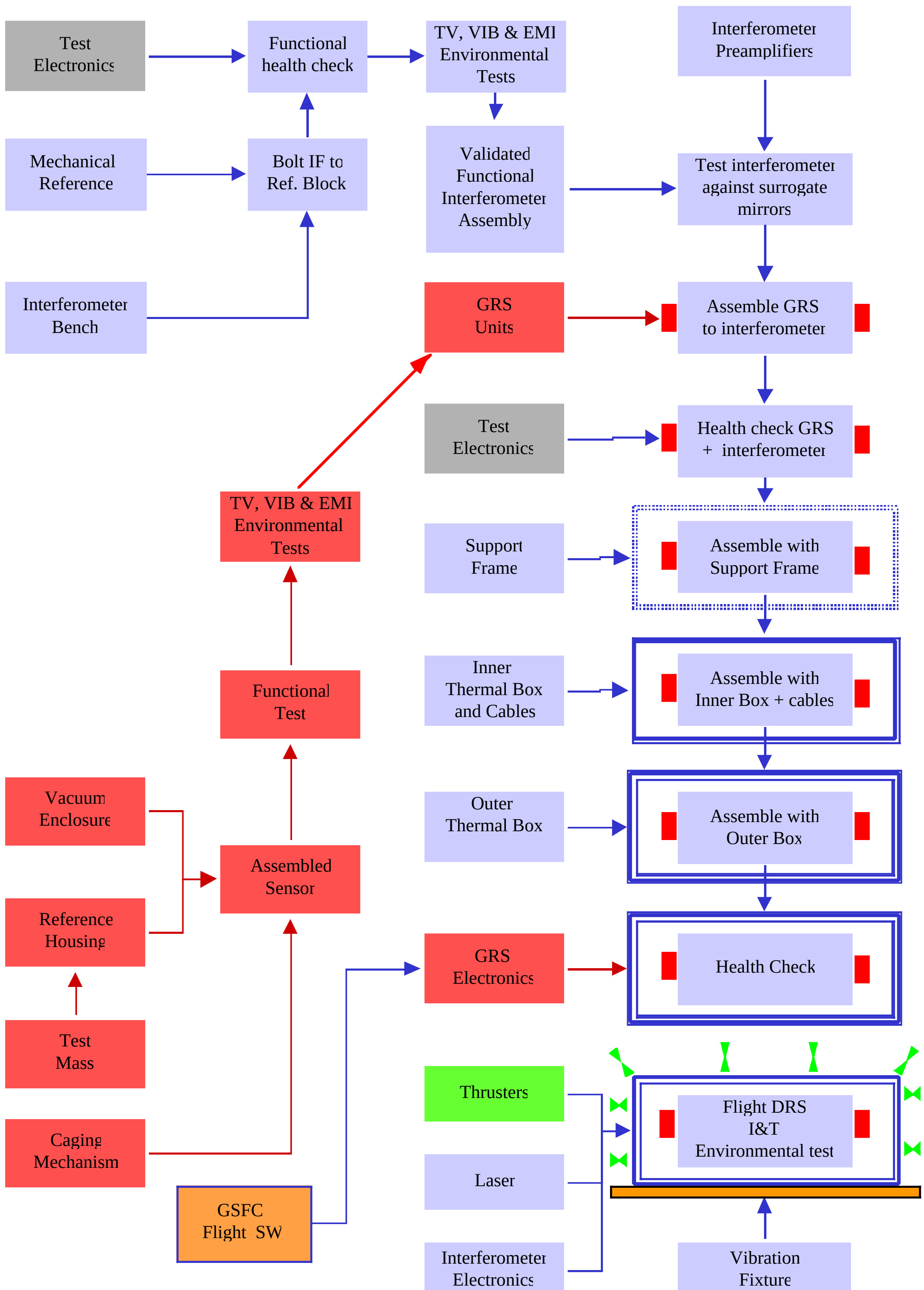

Figure 25 DRS integration flow diagram

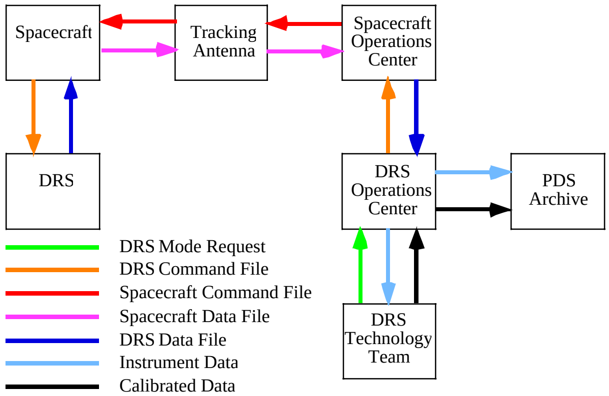

Figure 26 DRS end-to-end data flow diagram

by the ESA tracking stations. Data from the DRS will be included in spacecraft telemetry which will be received by the tracking station and forwarded to the spacecraft operations center. The spacecraft operations center will separate the DRS data and send to the DRS operations center. The DRS operation center will unpack the DRS data and forward to the technology team. The DRS operations center will archive the raw DRS data. The technology team will calibrate and process the DRS data. Calibrated data will be delivered to the DRS Operations Center for archival. Figure 26 shows the DRS end-to-end data flow.

The DRS data rate is fairly low. The primary data are the x, y, and z positions of each test mass with respect to its housing and the eight photodetector readouts from the interferometer. These will be sampled once per second. Other instrument diagnostic data will include thrust command levels and voltages from the microthruster, laser power and temperature, and spacecraft star tracker data used in the dynamic control, and gravitational reference sensor orientation, temperature, and voltage monitor data. The low daily and total data volume will allow distribution via e-mail so special data storage or distribution is needed. The data types and volume are summarized in Table 14.

Table 14 DRS data rates

Data	bits	#	bit/s	Note
GRS 1 position	16	3	48	x, y, z
GRS 2 position	16	3	48	x, y, z
Interferometer	16	8	128	photodetector volt
GRS 1 angles	16	3	48	orientation angles
GRS 2 angles	16	3	48	orientation angles
GRS 1 voltage	16	8	128	applied voltage
GRS 2 voltage	16	8	128	applied voltage
Charge control	3	12	48	power, temp, switch
Laser status	12	3	36	power, PZT, temp
Thrust value	16	8	128	command value
Thrust voltage	12	16	288	accel, grid volts
Neutralizer	12	4	48	electron current
Star tracker	16	2	32	quaternion
Temperatures	12	20	240	misc.
Total			1396	

1.9.9 Mission Operations

On orbit a series of test sequences will be performed. A summary of each type of test is given in Section 1.8.2. Data acquired during DRS tests will be stored in the GRS computer system and sent to the spacecraft over a standard interface for relay to ground.

The small operations team will be responsible for routine operations. They will schedule contacts with the DSN, format and send command loads, and monitor the health of the instrument. No special communication is required. The spacecraft will use the standard tracking services of the DSN. The ground data system will be protected by a firewall from external access. Information associated with the mission will be pushed through the firewall for access by the technology team. There are no real time ground support requirements for normal operations. No special equipment or skills other than those developed during instrument development and test are required for mission operations.

1.9.10 Product Assurance and Safety

A comprehensive DRS Mission Assurance and Safety extending throughout the Project lifecycle will be developed based on the processes and procedures that exist at JPL and at the other project team members: Goddard Space Flight Center (GSFC), Stanford University and Busek Co., Inc. All flight equipment developed in conjunction with the proposal will follow the requirements and procedures outlined in this section. The concept of built-in mission/project assurance will be emphasized with stress on continuous Risk Management. Past project lessons learned will be incorporated to assure that those portions of the DRS that have inheritance from or derivatives of an Earth orbiter design will meet the ANSI/ASQC 9000 series document, NASA NPG 7120.5B, and NIAT recommendations. JPL will have management and oversight responsibility for the integrated Safety and Mission Assurance (SMA) Program. JPL will develop overall DRS Project SMA plans in collaboration with the team members who will have complimentary plans.

1.9.10.1 Verification

The verification test program will provide for the realistic simulation of all significant environments and for functional demonstration, in end-to-end system tests, of all required operating modes. The verification program will include a written Test Plan that defines in detail the verification tests necessary for demonstrating acceptability of the flight hardware in the system acceptance test configuration. Completion of the prescribed tests, using approved test procedures and documented in a final report, will prove design qualification, verify workmanship and material integrity, and acceptability for flight. The verification test program will also include software independent verification and validation (IV&V) to insure that all requirements are validated during testing.

1.9.10.2 Environmental Requirements

The DRS environmental program process will be established which defines and imposes the polices and requirements that control the environmental design, analysis and testing of the flight hardware. The requirements will address test/analysis philosophy and objectives, types of tests and analyses required, re-qualification, subsystem and system level tests and documentation. As part of the final acceptance test program, environmental testing will be accomplished. The tests will be in agreement with mission requirements and will include instrument level random vibration, thermal vacuum and EMC as a minimum. A functional acceptance end-to-end test program for critical performance parameters will be included, as applicable, in each environmental test. An Environmental Requirements Document will be prepared to define the environmental program based on sponsor and mission environmental requirements.

1.9.10.3 Reliability Assurance

The DRS Reliability Assurance program includes policies, requirements and activities that will be implemented during design, fabrication, test and delivery of the flight hardware. The DRS design analyses activity will include temperature/voltage margin testing, Failure Mode Effects Criticality Analysis (FMECA) of the DRS-to-spacecraft interfaces, parts stress analyses on new or modified hardware, and other analyses necessary to support reliability and safety requirements. Probabilistic Risk Assessments will be performed when there is clear value added. Starting with the first power-up functional test of the hardware in the instrument or spacecraft configuration, formal Problem/Failure Reporting will be implemented to assure that all problems or failures that occur are addressed. The flight hardware will be designed and analyzed to preclude propagation of failures across interfaces. As part of the Reliability Assurance program to ensure minimizing electronic parts infant mortality, the flight hardware will be subject to 200 hours of operating time in the subsystem configuration followed by 300 hours of operation in the system configuration prior to launch. Total operating time prior to launch will be 500 hours with a goal of 1000 hours.

1.9.10.4 Quality Assurance

The DRS Quality Assurance program includes policies, requirements and activities that will be implemented during design and fabrication, test, and delivery of the flight hardware and support equipment. The quality assurance program stresses quality tasks and their integration with design, fabrication, and test phases. The Quality Assurance team will carry out workmanship inspection at the subsystem level and configuration inspection at the system level. The inspection activity will continue through post environmental test and shipping. Quality Assurance will verify compliance with interface control requirements for the flight hardware and that inspectable characteristics of the flight hardware are in conformance with applicable documentation.

1.9.10.5 Electronic Parts Reliability

A DRS Electronic parts program will be implemented that includes the selection, qualification, acquisition and correct application of electronic parts. Program requirements for the selection, procurement, screening and application review will be established and implemented. The electronic parts program will be developed based on requirements consisting of grad B+ quality parts. Lists of all electronic parts will be developed and maintained to allow for assessments in the event generic parts problems are identified. JPL will review and approve the team's parts lists. Any parts problem identified through the GIDEP alert system that effect the project will be identified to all team members for disposition.

1.9.10.6 Materials and Process

A DRS Materials and Processes control program will be implemented to assure that all materials and process requirements are met. Included in this program are requirements imposed on the selection of materials, material applications and fasteners allowed for flight applications. Design Item Lists will be developed and maintained and Materials Process specialist approval will be required for all safety critical items. Materials and Process engineering activities that will assist materials/process selection, testing, and evaluation will be identified. The program will also address non-destructive evaluation of materials and processes. JPL will review and approve the lists for the team.

1.9.10.7 System Safety

A DRS system safety plan will be prepared and implemented that will define the safety requirements and guidelines for fabrication, inspection, testing, transporting, prelaunch and operation of the flight hardware as well as all associated ground and operational support equipment. The safety plan will include precautions required to avoid hazardous situations or injury to personnel interfacing with or operating flight hardware during ground and flight operations. The plan sets forth the overall safety responsibilities and requirements and establishes an organized and direct system for evaluation and implementation of the safety program. Potential hazards will be identified, eliminated and/or adequately controlled through a hazards-analysis procedure.

1.9.10.8 Problem Failure Reporting and Management

A closed loop problem failure and management program will be implemented using the JPL Problem/Failure Reporting System or the team members' compatible system. Assessment of the adequacy of each P/FR closure will be conducted by cognizant engineers and Mission Assurance personnel. A continuous Risk Management assessment will be performed throughout the project life cycle.

1.9.10.9 Software Product Assurance

The SMA team's software product assurance program will be conducted in accordance with the requirements identified in the software management section of the Experiment Implementation Plan. Primary emphasis will be to ensure the development of reliable software by auditing the software requirements for clarity, technical adequacy, and traceability. A software IV&V program has been included, and will ensure that all requirements are validated during testing.

1.9.10.10 Contamination Control

The DRS Contamination Control program will ensure that contaminants and instrument contamination flow paths and deposition targets are well understood. Parts, materials, and processes assessments will define the likely contamination sources. Material and process selection, clean room environments, handling procedures, and appropriate purge capabilities will be identified to ensure equipment cleanliness to maintain performance levels at all sites, when necessary.

1.9.10.11 Risk Mitigation Activities

New technology hardware will be flight qualified through tailored tests and analyses. Reviews of the hardware pedigree (Inheritance Reviews) will be performed to determine the design and test history relative to the mission requirements. The reviews will emphasize reliability and qualification status.

The SMA team will support, as required, the independent Red Team reviews.

1.9.10.12 Mission Operations Assurance

The final risk mitigation function of the DRS Mission Assurance team is Operations Assurance. The approach is based on a teaming arrangement with the Operations PEM and team. Operations Assurance planning consists of over a dozen major assurance functions which insures that assurance planning covers the critical operations phase and establishes a unique proactive mind set for operational problem solutions and problem avoidance.

1.9.11 Project Descope Options

The DRS design is based on the recommendations of the Pre-Phase A study team. The DRS design has changed during Phase A to reduce cost while maintaining the desired performance. Changes include changing the interferometer design from heterodyne homodyne, the deletion of a separate payload control computer, and reducing the number of microthrusters from twelve to eight. The reduction in the number of thrusters has associated increase in risk, since with eight thrusters all must operate properly for the performance of the experiment. (With twelve thrusters a failure of up to three thrusters could be tolerated.) Further descopes are possible but most involve a reduction in performance.

The first descope would be to reduce the number of thrusters further, from either to seven or six. This reduction would not reduce performance and would save approximately $500 k. It would complicate the accommodation of the thrusters on the spacecraft, since the smaller thrusters has more constraints on how they are placed on the spacecraft.

Another descope would be to eliminate the brassboard development. The breadboard development is considered mandatory, to demonstrate the required technology readiness level and for development of the final design. The brassboards are primarily for verifying the design and testing for suitability for space operation. They are also intended for testing for compatibility with the spacecraft. In case of cost and schedule problems, the brassboards would be eliminated, with the breadboards (possibly updated) used for spacecraft tests.

The two-stage thermal enclosure could descoped to a single stage. This would save ~$100 k and simplify integration and test, but would increase temperature-related noise on the GRS systems.

Deleting the interferometer would results in a cost saving of ~$2 M, including not only the cost of the interferometer but also simplifying the structure and integration and test. This would seriously degrade performance, meeting the baseline requirement lowest frequencies of interest but not meeting the performance floor at the highest frequencies.

A final descope option would be to remove the GRS charge control subsystem. This would save ~$500 k, but require that the test mass be contacted with the housing wall to excess charge accumulated on the test mass. Because of contact potential differences, the remaining charge on the test mass would be much higher than when using the charge control system, possible as much as 100 or 1000 times higher average charge. Additional electrical shielding of the test mass might allow this configuration to reach the performance floor, but probably not reach the baseline performance.