Elastic Backscattering Lidar for a MSTI Satellite

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Abstract

An elastic backscattering lidar is being built for the Ballistic Missile Defense Organization (BMDO) to fly on a MSTI (Miniature Seeker Technology Integration) satellite in low-earth orbit. The purpose of the this payload is to demonstrate the capability of a space-based lidar on a small satellite to track Theater Ballistic Missiles (TMDS) in both the boost and post-boost phases. This is a fast-track program to design, construct, and test the lidar payload in a two-year time span. System requirements which are based on calculated and measured plume lidar data will be presented for this approximately 100 kg payload. Details of the lidar subsystems, which include a Nd:YAG; diode-pumped laser; a 76-cm light collection telescope, an avalanche photodiode detector; and an R3081 microprocessor-based payload controller will be described. Finally, planned mission scenarios will be discussed.

1.0 Introduction

Lidar (light detection and ranging) is a technology which has been applied to the remote sensing of gases and particulates in the atmosphere for decades. Only recently, however, have major advances in efficient and lightweight lasers, microprocessors, and detectors made it feasible to build a lidar payload which could be flown on a small satellite. The Los Alamos lidar team is currently engaged in designing such a lightweight, space-borne lidar system payload for the Ballistic Missile Defense Organization's MSTI program. The objectives of the MSTI program are to develop space-based sensors for Theater Missile Defense surveillance from space and to generate target track files on-board the MSTI spacecraft using these sensors. MSTI will demonstrate sensor fusion, and explore the dual use of potential DoD sensors for environmental and ecological monitoring.

The MSTI program strategy is to minimize risk with a "build a little, test a little" development strategy. What this means is that there is increased complexity and capability in each subsequent mission - both in the payload and the spacecraft. For example, MSTI 1, which was launched in 1992, had a simple gas-jet stabilization system and an off-the-shelf IR camera. The entire MSTI 1 effort, from inception to launch, was accomplished in only nine months. MSTI 2, planned for launch in late 1993, will have two passive sensors - MWIR and SWIR cameras - and pitch momentum wheel stabilization. MSTI 3 and 4 will be three-axis stabilized, have three or four passive sensors, and perform on-board track file generation.

All the MSTI sensors in the first four missions are passive. The Los Alamos lidar payload will provide the first opportunity for BMDO to perform active tracking of the missile and its plume. This adds a substantial capability to the MSTI satellite as a TMD surveillance
demonstrator since the lidar provides additional knowledge of the target trajectory in the form of range-to-target measurements. Range information, used in conjunction with angular rate data from passive sensors, enables a more accurate determination of the target track file from a single satellite platform. This results in a smaller error basket when the track file is handed over to an interceptor.

The lidar being developed for MSTI is an elastic backscatter system based on several ground and airborne lidar systems which the Los Alamos lidar team has developed. The significant technological challenges of the space-borne lidar are driven by the needs to: reduce the mass of the laser and optics systems; space qualify the components and subsystems; operate at ranges of 400 to 700 km; and allow for autonomous operation of the payload. The MSTI lidar will be used to obtain volumetric images of rocket plumes by Mie scattering of the laser light from particulates and droplets contained within the plume. This information will then be used to derive the missile range and trajectory. The plume is much easier to detect and track than the missile hardbody during boost phase because it is much larger in extent and acts as a nearly isotropic scatterer compared to the missile hardbody. Once the missile trajectory during boost has been determined, it may be easier to detect and track the hardbody with the lidar during and after stage burnout.

This type of elastic backscatter lidar has several other potential applications. Using the lidar to scan a volume of the atmosphere allows for the generation of three dimensional images of droplet and particulate density within that volume. This can be used to identify clouds and plumes; it can be used to characterize visibility in a selected area; and to measure other weather related phenomena.

2.0 Mission Objectives

The primary objective of the MSTI LIDAR project is to demonstrate advanced technology for a space-borne, fixed-wavelength, elastic back scattering lidar as a means to detect and track a rocket plume, and to interpret volume images of targeted areas. Experience and information learned from this mission will demonstrate the technological capability for a space-borne lidar to track missiles and generate volumetric images of reflecting particulates and droplets within the atmosphere. The following tasks are steps which must be accomplished in order to meet the primary objective:

- Develop lidar system payload for flight experiments on a MSTI bus.
- Perform on-orbit boresight trim of laser and receiver optics.
- Demonstrate volume imaging capabilities from space.
- Validate requirements for missile tracking.
- Demonstrate closed-loop tracking of the missile plume.
- Provide real-time three-dimensional track file.

The secondary objectives of the space lidar are to demonstrate its operability day and night; its dual applications in defense and environment remote sensing; and the development of the technology base for future space-borne lidar systems.

3.0 Lidar System Requirements

The lidar system requirements are driven by the need to perform the missile tracking mission with a payload that satisfies the constraints imposed by a small satellite in low-earth orbit. The primary subsystems in the lidar payload are the laser illuminator; an optical telescope to collect back-scattered photons; a detector to measure the signal return; a passive sensor to detect the missile and generate pointing vectors for the laser illuminator; and electronics to autonomously control the payload and collect,
process, analyze, and store the experimental data. Since the MSTI bus is small (160 kg) and the launch vehicle is a Pegasus, the mass, power, and volume of the lidar must be minimized and the overall small satellite design must be carefully engineered to provide the best systems solution.

The bus system requirements are driven by the launch system constraints and the need to provide adequate power and pointing capability to the lidar payload. The lidar payload will be able to steer the laser illuminator within a 5 degree field of view (FOV) and the satellite must keep the lidar pointed at the target within that FOV during a tracking experiment. For the case of either fixed ground targets or moving missile targets, the satellite will have significant one-, two-, or three-axis steering maneuvers which must be accomplished within time, accuracy, and jitter constraints of the lidar payload. The power requirements for the bus include payload power profiles which might average less than 100 W OAP, and peak power loads in excess of 1 kW for tens of seconds. The thermal control requirements of the payload will also translate into post-experiment maneuvers of the spacecraft to ensure that the radiating surface is pointed in the correct direction.

The lidar backscatter signal from a target can be estimated by calculating the number of detected photons for one laser pulse using the following equation:

\[
\nu = \eta N \sigma_b \Delta \Omega \left( \frac{A_p}{A_{LS}} \right) \int_0^{x_0} \left( e^{-2\sigma_c x} dx \right)
\]

where

- \( \nu \) = detected signal photons
- \( \eta = (\text{telescope transmission}) \times (\text{detector quantum efficiency}) \)
- \( N = \text{number of output photons per pulse} \)
- \( \sigma_b = \text{differential cross section for backscattering} \)
- \( \Delta \Omega = \text{steradians subtended by the telescope} \)
- \( A_p/A_{LS} = (\text{plume area}) / (\text{laser spot size}) \)
- \( \sigma_c = \text{total extinction cross section} \)
- \( x = \text{distance through the plume} \)
- \( x_0 = \text{plume diameter} \)

It should be noted from the equation that the number of photons detected by the lidar depends fundamentally on the power-aperture product \((N \Delta \Omega)\). The payload designer can only vary the telescope diameter and laser pulse energy to effect the signal strength for a given set of mission requirements.

While a large power-aperture product is desired to increase signal strength, launch vehicle fairing size and small satellite bus power and mass capabilities place hard limits on the telescope and laser parameters. For a MSTI bus on a Pegasus launch vehicle, the maximum telescope diameter is approximately 76 cm. From a mass and power perspective, the most efficient, high-power laser available is the diode-pumped, Nd:YAG laser. As the result of substantial investment by BMDO and ARPA, current Nd:YAG lasers are capable of producing 1.0 to 1.5 joules per pulse at repetition rates of 50 to 100 Hz, with overall electrical efficiencies of 7 percent.

### 4.0 Mission Experiments

The experiments listed below are planned for the space-borne lidar to satisfy the mission objectives discussed previously. These experiments are listed in the order of increasing complexity, cost and development time. These experiments will drive the mission requirements, payload systems specifications and spacecraft requirements.

#### 4.1 On-orbit Boresight Trim

The subsystems of the payload will be calibrated individually and together as a payload system. The payload will then be integrated to the spacecraft which then will be
integrated with the launch vehicle. Functional testing will be performed along the way to ensure the payload operation. However the launch environment and subsequent temperature variation on-orbit will change the payload alignment between the laser and receiver optics line of sight. Therefore a boresight trim experiment will be performed after the satellite is launched into orbit and each time prior to performing any experiments. The purpose is to re-establish coaxial laser beam and receiver optics line-of-sight and demonstrate autonomous, on-orbit system alignment.

4.2 Volume Imaging And Measurement Of Cloud Structure

This experiment will image cloud structure in three-dimensions. Information on cloud parameters, e.g. altitudes, thickness and number of layers, optical attenuation and reflectivity will be obtained for climate modeling and weather prediction. Cloud cover and cloud height information could be used to assist a tactical air force commander in selecting strategies and weapons appropriate to the atmospheric conditions at a target.

Goals: Measure cloud profiles using nadir pointing with cross-track scan. Post-process data on ground to reconstruct.

Assumptions: Preset time and location to turn payload on for 200 sec and take data. No tracking required; constant nadir pointing required.

Functional requirements:
• gate out ground return signal
• control scan mirror

Hardware requirements:
• visible CCD camera
• adjustable gate delay and number of samples for detector and digitizer circuit
• A/D sampling @ 10 Msamples/sec

Data requirements:
• 1064 nm image: 200 sec x 50 Hz x 1000 sample/pulse x 10 (or 12) bits/sample = 240 Mb ~ 30 Mbyte
• Visible CCD: 200 sec x 1 frame/sec x (256x256 pixel) / frame x 8 bit/pixel x 1/30 JPEG compression = 3 Mb ~ 0.38 Mbyte

4.3 Measurements Of Fixed Ground Target

This experiment will measure atmospheric conditions at a fixed ground site. Space-based mapping of measurable pollution movements over time will identify the sources and the relative seasonal contributions. Dispersal of airborne particulates from volcanic eruptions or forest fires may also be measured.

Assumptions: Preset time and location to turn payload on for 200 sec and take data. No tracking required; satellite must maintain pointing at ground site during experiment.

Goals:
• volume image 20 km x 20 km x 20 km along track or cross-track
• 30 m resolution
• post-process data

Functional requirements:
• autonomous pointing at ground site by spacecraft
• pointing laser and spacecraft - open loop
• concurrent visible image
• measure laser pointing (3-D reconstruct)

Hardware requirements:
• visible CCD camera
• adjustable gate delay and number of samples for detector and digitizer circuit
• A/D sampling @ 10 Msamples/sec

Data requirements:
• 1064 nm image: 200 sec x 50 Hz x 1000 sample/pulse x 10 (or 12) bits/sample = 240 Mb ~ 30 Mbyte
• Visible CCD: 200 sec x 1 frame/sec x (256x256 pixel/frame) x 8 bit/pixel x 1/30 JPEG compression = 3 Mb ~ 0.38 Mbyte
4.4 Detection / Measurement Of Missile Plume

This experiment will detect and measure the plume from missiles such as a Titan III or Aries rocket when the opportunities arise. The missile launch and satellite pass time will be coordinated. The missile’s typical flight trajectory will be also be up-linked to the spacecraft so that it can point the lidar towards the missile.

Assumptions: Preset time and location to turn payload on for 200 sec and take data. No active tracking is required. Pre-programmed spacecraft to slew toward target.

Goals:
- volume image of plume
- 15 m range resolution
- real-time range estimation
- post-process of images

Functional requirement: satellite must point lidar at missile.

Hardware requirements:
- 50 nsec between sample - A/D sampling at 20 Msample/sec

Data requirements:
- 1064 nm: 100 sec x 50 Hz x 1000 sample/pulse x 10 (or 12) bits /sample = 120 Mb = 15 Mbyte
- Visible CCD: 100 sec x 30 frames/sec x (512 x 512 pixel)/frame x 8 bit/pixel x (1/30) = 210 Mb = 26 Mbyte
- 2 steering mirrors: 100 sec x 100 Hz x 1 sample/pulse x 16 bits/sample x 2 channels = 320 kb ~ 40 kbyte

4.5 Closed-loop Spacecraft Tracking Of Missile; Lidar Measurement Of Missile Plume

This experiment will use a passive sensor to acquire the missile plume. Two-axes fine steering mirrors will then be used to accurately point the laser and optics towards the target. The lidar payload electronics will obtain range data in real time, which will be output to the spacecraft track-file generator board. This experiment will demonstrate the practicality, and determine the requirements of, a spaceborne lidar for early acquisition and closed-loop tracking of a missile plume.

Assumptions: A missile will be launched specifically for this experiment. At a preset time and location, the payload will be operated for approximately 200 sec and take data. The spacecraft will slew to follow the target.

Goals:
- real-time range to plume measurement
- laser pointing position
- 15 m range resolution
- real-time track file generation
- post process data

Functional requirements:
- spacecraft must maintain pointing lidar at target within 5° FOV
- payload passive sensor for closed-loop tracking
- passive sensor with centroid algorithm to generate laser pointing vector
- control laser pointing

Hardware requirements:
- passive sensor (MWIR camera for plume and warm body tracking)
- passive sensor centroid algorithm to give target vector @ 30 Hz
- track-file generator board on spacecraft

Data requirements:
- 1064 nm images and target range data, visible and IR CCD images
- passive sensor: 100 sec x 100 frame/sec x (128 x 128 pixel)/frame x 8 b/pixel x (1/30) JPEG = 44 Mb ~ 6 Mbyte

5.0 MSTI LIDAR Baseline Design

The baseline design for MSTI LIDAR is an imaging elastic backscatter lidar operating at 1.064 μm. It has an energy-aperture product of 0.55 J-m² (1.5 J per shot and an effective aperture of 0.37 m²). The lidar payload consists
of the following subsystems: laser module; optics; detector and associated electronics; payload control electronics; and MWIR camera. The laser and optics subsystems are being procured with 1.064 μm as the nominal wavelength.

Figure 1, MSTI lidar functional block diagram, shows the subsystems and interfaces within the lidar payload, and the interfaces between the lidar payload and MSTI bus. This functional block diagram includes subsystems (ERADS, GPS, RLG) which give the payload the capability of autonomously performing the target tracking and track-file generation mission.

The heat stored within the laser thermal control system during the experiment will be rejected into space through the spacecraft radiators when the experiment is completed. The control of this thermal loop will be sent through the payload controller to the spacecraft controller. Heaters, which are not shown in the block diagram, may be needed to maintain the payload within the survival temperatures during nonoperating periods. The payload and its subsystems are designed to survive the following environments:

- Non-operating environment
  50 to +60 °C ambient temperature
- 0 to 1 atm. ambient pressure
- 100 G shock, 1 to 10 kHz
- 1 G rms vibration, 100 to 2000 Hz
- quasi-static acceleration: ± 15 G y-axis,
  ± 10 G z-axis, ± 8 G x-axis
- Operating environment
  10 to +40 °C ambient temperature
- 0 to 1 atm. ambient pressure
- Radiation Tolerance: 1 krad over one year (with 50 mil Al shielding)

### 5.1 LASER SUBSYSTEM

The baseline laser is a diode-array pumped, Q-switched, Nd:YAG laser operating at 1.064 μm. The laser subsystem includes the laser, power supply, thermal control system and laser electronics. The entire laser subsystem can not exceed 35 kg, and will fit into a volume of 19” (48.2 cm) long x 14” (35.5 cm) wide x 9” (22.8 cm) high.

The laser subsystem is designed to be mounted on the spacecraft deck (figure 2). An optical interface will be provided to link the beam exit with the optics subsystem. To interface with the payload control electronics and to receive power from the spacecraft bus, the laser subsystem will have three cables with flight qualified connectors: (1) for power transport from the spacecraft; (2) RS422 interface for control command and housekeeping signal transport; and (3) high speed interface with less than 5 ns jitter for transporting timing signals. The housekeeping signal includes: laser status; measured real-time pulse energy; critical voltages and currents; and critical temperatures.

The power conditioning electronics is designed such that: (1) the total power input to the laser subsystem will not exceed 1.0 kW; (2) at turn-on, the initial current will not exceed three times the average operating current; (3) thereafter, the supply current variations will not exceed 25% of the average operating current. The laser subsystem and the cables will be designed and demonstrated to survive the electromagnetic emission tests defined in MIL-STD-461C. Other specifications for the laser include:

- Output Energy: > 0.8 J/pulse, 1.5 J desired
- Power: 940 W peak, ~ 94 W average
- Laser linewidth: < ±0.05 nm
- Pulse Width: ≤ 30 nsec
- Repetition Rate: ≥ 50 pulses/sec, 100 desired
- Laser Beam Quality: < 3 times diffraction limit
- Pointing Jitter: < 0.1 mrad
- Beam Divergence: < 0.6 mrad (e^-2 point)
• Beam Diameter < 1.1 cm (e\(^{-2}\) point)
• Overall Wallplug Efficiency > 7%
• Operating Life > 3 \times 10^8 shots
• Maximum Duty Cycle < 3.3% (200 s ON, 5800 s OFF)

5.2 OPTICS SUBSYSTEM

The lidar optics subsystem consists of: light gathering receiving telescope; relay optics to the detector; relay mirrors for laser output beam (laser beam expander and boresight trim); and 2-D scanning mirrors to control the laser and telescope line-of-sight. The mass of the optics subsystem will not exceed 31.8 kg. It will fit within a volume of 30” (76 cm) diameter x 35.71” (90.7 cm) high. As with the laser, the optics subsystem will operate at 1.064 µm nominal wavelength.

At least 50% of the light gathered at the 30” diameter receiving telescope will reach the detector. The instantaneous field of view (IFOV) will be 0.7 mrad and the total field of view (TFOV) will be not less than 5 degrees in two orthogonal directions. The resolution of the telescope-detector lens combination will enable all the photons in a 0.5 mrad FOV to be captured by a 2 mm x 2 mm detector without vignetting.

The nominal magnification for the laser beam expander will be 5, with the beam input size (at the e\(^{-2}\) point) being approximately 1±0.1 cm. The performance will not be more than 1.1 times the diffraction limit. Mirrors are placed after the beam expander for on-orbit boresight trim. The boresight alignment between the laser beam expander and receiving telescope will be within 0.1 mrad in the nominal static alignment case.

The scanning mirrors total field of view will be greater than 6 degrees circular. The settling time will be less than 5 msec. The scan mirror drive may be incremental, but with the TFOV quantized into 100 steps. The scan mirror position encoder must be absolute and provide 16 bits of angular resolution.

5.3 DETECTOR SUBSYSTEM

The detector design is an avalanche photodiode detector (APD) with a transimpedance amplifier which has been developed at LANL for air-borne lidar systems. The desired image size on the detector is 2 mm x 2 mm. The actual size of the detector is 5 mm by 5 mm to allow for misalignment and beam jitter. The detector design requirements are to use available technology to detect 30 to 200 photons at 1.064 µm in 1 to 25 bins of 50 nsec (0.1 to 0.6 photons/nsec) against a potential solar background of 320 photons per 50 nsec bin (6.4 photons/nsec). The satellite version of the detector is a C30627E 1.06 µm-enhanced silicon unit.

5.4 PAYLOAD CONTROL ELECTRONICS SUBSYSTEM

The payload electronics will be built around a high-speed, intelligent 32-bit RISC processor which is being used on another LANL program which will fly on the September 1994 STEP3 mission. The R3081-based system was developed with Honeywell as part of their small satellite flight control and management systems and is capable of 20 MIPs / 5 MFLOPS throughput at 25 MHz. The electronics unit uses a global bus to connect to multiple slices. Each slice provides mechanical stability and heat transfer for two printed circuit boards. The slices are assembled and bolted together for mechanical stability. This electro-mechanical design of the payload electronics allows for parallel development and testing of subsystems.

The functions of the payload electronics include:

• test and search for correct line-of-sight alignment of the laser beam optics, beam expander and receiver optics after launch and prior to carrying out experiments
• power up and safe shutdown of laser
• receive uplink time to start experiment and proposed target location
• read scan mirror position
• control scan mirror
• execute open-loop algorithm
• for the plume track experiment:
  • locate missile plume in passive sensors’ FOV
• execute passive sensor centroid algorithm
• execute scan mirror control algorithm
• select photodetector high-voltage bias
• select digitizer operating mode
• time-tag events and trigger data acquisition
• record outgoing laser-pulse energy
• measure range to target and transfer information to spacecraft in real time
• transfer data (lidar, CCD and MWIR, scan mirror, etc.) from temporary storage to spacecraft memory after experiment
• regulate and distribute payload power
• control temperature of payload electronics
• turn on/off laser heat rejection system

5.5 Payload Thermal Control

The MSTI satellite orbit will be a high-inclination, 450 km circular orbit. The satellite payload is nominally nadir pointed and the solar array is sun-oriented. A high inclination orbit produces two limiting thermal orientations. The nominal orbit condition is along the dusk-dawn terminator. Due to the earth’s oblateness and revolution about the sun, this orbit changes to a noon-midnight orbit after a few months. The satellite orbit will progress through these limiting conditions several times during the course of the flight experiment.

Passive control is the overall objective of the MSTI payload thermal design. Heat loads from the sun and the earth are minimized by colocating radiators; adjusting satellite attitude and orientation periodically; using shutters; and using low α/ε radiator surface materials. Because of the large power transients during operations, the payload electronics is isolated from the MSTI bus. Supplemental power is provided by heaters to the laser illuminator, electronics assemblies, batteries, and optical telescope and structure as required.

Passive thermal control of the laser is sought by eliminating the use of pumped cooling of the laser illuminator module. The laser module generates heat at a rate approximately ten times that of the rest of the satellite (1000 W versus 100 W) when the laser is operating. An operational duty cycle of less than five per cent is adequate to meet mission needs. The laser module is thermally isolated from the MSTI lidar bus and optical telescope. A phase change thermal storage material absorbs the heat generated by the laser module during lidar operation. The phase change material rejects its heat load to a radiator during the remainder of the orbit. The satellite orientation and sun conditions during that time may require additional satellite orientation changes to reject heat and maintain sun illumination on the solar arrays.

The MSTI bus and the optical telescope are thermally coupled. Multilayer insulation protects the side surfaces of the optical telescope from solar illumination. Heaters maintain the optical telescope’s temperature to prevent thermally-induced alignment distortions.

5.6 Acquisition, Tracking, and Pointing Requirements

There are several important technical issues associated with placing a LIDAR payload on a MSTI-class satellite. The dynamics of satellite motion are significantly affected by a relatively massive payload which requires 3-axis slewing of the satellite. There is an explicit assumption that gimbaling the LIDAR telescope, or even just a front end aperture, is not
reasonable on a small satellite because of the size of the optics. The LIDAR pulse repetition rate and target measurement requirements are going to set the satellite position and attitude knowledge requirements.

The satellite position, velocity, and time estimates are updated at a 1 Hz rate. There is no information at this time as to the power spectral density (PSD) of the errors. If a celestial sextant solution is used, the position error is probably of the order of 50 meters (SEP). For our proposed earth reference attitude determination system (ERADS) sensor, the celestial sextant solution may approach the 25 meter value that can be provided by a GPS receiver. References (1) and (2) discuss the use of ERADS as a celestial sextant.

The error in pointing knowledge of the lidar beam is comprised of several sources. The position knowledge error has been identified. The lidar payload is assumed to be tightly coupled to the spacecraft. Since the lidar repetition rate may be as high as 100 Hz, the fine steering position measurement should have a bandwidth of approximately 500 Hz. This will generate a two-axis angle measurement of where the beam was pointed at the time of firing. There will be a certain amount of mechanical drift and misalignment of the lidar primary axis with respect to the satellite attitude determination system. This will include relatively slow changes due to thermal cycling, residual stresses, and uncompensated torques on the spacecraft frame.

If the lidar payload must perform the inertial measurement unit (IMU) functions, then the attitude determination system of the satellite will operate at two rates. The ERADS will provide satellite attitude information at approximately 1 Hz. The ring laser gyroscopes (this could also be either mechanical rate gyroscopes or an interferometric fiber optic gyroscope) will be used to provide attitude updates at the 500 Hz bandwidth. During rapid satellite slewing maneuvers, the ERADS could not update at 1 Hz and the RLGs would be integrated for a longer time. However, even if this time increases to 30 seconds, the drift requirements on the gyroscopes will still be in the affordable range.

Another source of unmeasured jitter in the platform which couples into the lidar optical path. Since there will be sources of high frequency and low frequency vibration on the platform, control of this coupling into the optical path is critical. There is minimal data published on measured jitter and vibration levels on small satellites. There is some data from larger spacecraft (1000 to 2500 kg) that indicates baseline jitter values in the 50 to 100 μrad range at frequencies in excess of 100 Hz.

One of the design requirements on the lidar telescopes is to use passive vibration damping techniques in the structural design. The range error to the target is the combination of the width of the digitizer bin and the timing jitter between the laser pulse and the digitizer clock.

The last source of pointing error considered here is the divergence of the laser beam. Since the target is flood-loaded, the full angle divergence of the beam is the error source. The pointing error determines the probability of the laser beam hitting the target. Table 1 summarizes estimated values for these errors. In the absence of other information, these error sources are assumed to result in pointing errors with zero mean, a Gaussian distribution, and the error values are 3 sigma values. The errors are also assumed to be white noise. While that will not be true when the real hardware is built, there is no a priori reason to treat those errors as colored noise at this time. These assumed error terms reflect a technically aggressive approach to knowledge measurements on a small satellite. No assumptions are made yet as to control capability.
5.7 Satellite Requirements

The MSTI satellite must provide power; command and data handling; and pointing control for the lidar payload. The following bullets summarize the satellite requirements for the lidar payload:

- System mass of ~ 100 kg with dimensions of 81 cm dia x 117 cm dia x 112 cm long
- Peak power of 1100 Watts at less than 10% duty factor
- Gigabyte per day of data storage and telemetry
- 350 - 450 km and 45 - 75° inclination
- Three-axis attitude control stabilized to 0.1° (3σ) with additional jitter requirements
- Attitude and position knowledge requirements determined by mission
- 200 μrad, 100 m - fixed target
- 100 μrad, 25 m - missile target
- Potentially high slew rates > 1° per second
- Real-time experiment control for coordinated satellite and ground tests
- No translational maneuvering capability

The most stressing requirement for a small satellite performing this mission is the need to perform rapid slewing of the spacecraft about two or three axes while maintaining adequate jitter control and not losing stability. Figure 3 shows the engagement geometry used in the slewing maneuver calculations. The MSTI satellite is in a 450 km, circular orbit. Two target missiles are launched at 60° elevation angles. One is launched generally parallel to the satellite track and one is launched perpendicular to the satellite track. The apogee of the target missile is approximately 300 km.

Figure 4 shows the range to the along-track target from the satellite as a function of time for three relative launch times for each of the two missile trajectory cases. Relative to when the satellite is overhead of the missile launch location; the target launch is assumed to be either 2.5 minutes early, 2.5 minutes late, or launched when the satellite is overhead. Figure 5 shows the same data for a cross-track target.

Figures 6 and 7 show the azimuthal and elevation angles to the two targets. Azimuth and elevation angles are defined in fixed spacecraft coordinates where azimuth refers to the angle around the nadir direction and elevation is the angle relative to that axis. Figure 8 and 9 show the azimuthal and elevation angle rates to the targets and figures 10 and 11 show the azimuthal and elevation angular accelerations to the targets. Since the lidar is an axi-symmetric, non-imaging optical system there is no preferred orientation about the center of the 5° FOV in the payload coordinate system. The requirement is that the target be maintained somewhere within the FOV. As a result, the apparently high angular accelerations implied by the azimuthal plots do not necessarily correspond to the spacecraft torque requirements. The slew control maneuver control algorithms will be designed to minimize torque requirements while satisfying pointing error constraints (i.e., the target must be within the 5° FOV).

6.0 Mission Operations

Figure 12 shows an outline of a typical mission operation for a target tracking experiment. At the start of the sequence, the lidar payload is powered on and the payload controller performs state-of-health checks. The spacecraft command and data handler passes the experiment commands to the payload controller. The IR camera is cooled to operating temperature and powered on. The laser module is powered on and goes through a warm up sequence. The satellite is slewed to point at the target launch point and a final attitude calibration is made.

The passive sensor detects the missile launch
and starts to track the target. Pointing vectors for the steering mirrors are generated and used to control the laser aim point. The range to the target is measured by the signal return at the APD. Range-to-target measurements are extracted from the data and passed to the track-file generator board on the satellite. APD data, steering mirror position, and IR raw data are stored for later processing.

At the completion of the tracking experiment, the laser module is powered down and any necessary satellite attitude adjustments are made in order to reject heat from the phase change material. Payload experiment data is transferred to the satellite telemetry system for down-linking at the next available ground station contact.

7.0 Summary

The design of a lidar payload for small satellite applications has been presented. The system engineering and mission integration issues of such a payload as part of a small satellite program, with the expected launch vehicle and cost constraints on such a mission, are challenging. However, advances in several areas of technology make this a realizable goal.

8.0 References


### Table 1: Error Budget

<table>
<thead>
<tr>
<th>error source</th>
<th>magnitude</th>
<th>pointing error</th>
<th>comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>satellite position</td>
<td>25 m SEP</td>
<td>62.5 μrad</td>
<td>at 400 km range</td>
</tr>
<tr>
<td>satellite attitude</td>
<td>0.005°</td>
<td>100 μrad</td>
<td>ERADS plus rate gyroscopes</td>
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<tr>
<td>fine steering mirror angular position</td>
<td>0.005°</td>
<td>100 μrad</td>
<td>measured from position encoders</td>
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<tr>
<td>mechanical drift</td>
<td>0.005°</td>
<td>100 μrad</td>
<td>z-axis drift causes negligible error in optical train</td>
</tr>
<tr>
<td>jitter</td>
<td>0.005° at 100 Hz</td>
<td>100 μrad</td>
<td>PSD is assumed to be “white”</td>
</tr>
<tr>
<td>250 μrad laser</td>
<td>RSS = 210 μrad</td>
<td>beam divergence / pointing error = 1.19</td>
<td></td>
</tr>
<tr>
<td>0.5 mrad laser</td>
<td>RSS = 210 μrad</td>
<td>beam divergence / pointing error = 2.38</td>
<td></td>
</tr>
<tr>
<td>target range</td>
<td>50 nsec</td>
<td>15 m</td>
<td>+/- 7.5 m (assume timing jitter &lt; 3 nsec)</td>
</tr>
</tbody>
</table>
autonomous navigation

GPS

IR DSP

passive track

ERADS

100 Hz altitude knowledge

Spacecraft C&DH

1553B or FDDI

R3081

RLG

SSDR

payload control & data storage

laser

detector

fine steering

telescope

EARTH
Figure 2 Lidar Configuration on MSTI Bus

- $\pm 3^\circ$ field-of-regard
- target detection & closed-loop track
- bore-sighted with receiving optic
- avalanche photo-diode detector
- phase-change material

MSTI BUS
• Figure 3 Engagement Geometry For MSTI Satellite and Target

• Figure 4 Range to Along-Track Target
• Figure 5 Range to Cross-Track Target

![Range to Cross-Track Target](image)

- 2.5 min late
- 450 km, 55 deg satellite
- 2.5 min early
- 0 min early

• Figure 6 Angles to Along-Track Target

![Azimuth & Elevation to Along-Track Target](image)

- 0 - az
- 0 - el
- late - az
- late - el
- early - az
- early - el
- 450 km, 55 deg satellite
• Figure 7 Angles to Cross-Track Target

Figure 7 Angles to Cross-Track Target

Azimuth & Elevation to Cross-Track Target

450 km, 55 deg satellite
ear - az
late - az
late - el

angle (deg)

0 20 40 60 80 100 120 140 160 180 200
time (sec)

• Figure 8 Angular Rates to Along-Track Target

Figure 8 Angular Rates to Along-Track Target

Angle Rate to Along-Track Target

450 km, 55 deg satellite
ear - az
late - az
late - el

degrees per second

0 200 400 500 600 700 800 900
time (sec)

pg. 17
• Figure 9 Angular Rates to Cross-Track Targets

- Figure 10 Angular Accelerations to Along-Track Targets
Figure 11 Angular Rates to Cross-Track Targets

Angular Accelerations to Cross-Track Target

- 450 km, 55 deg satellite
- early - az
- early - el

Degrees per second^2

Time (sec)
passive launch detection

slew satellite at launch point
dump heat

warm-up laser

closed loop tracking

prepare passive tracker

dump heat

autonomous navigation

open-loop inertial track of "expected trajectory"
closed-loop track with passive sensor