Title: Demonstration of Optical Stellar Interferometry with Near Earth Objects (NEO) using Laser Range Finder by a Nano–Satellite Constellation: A Cost – effective approach.

Primary POC: Prof. Hyochoong Bang
Organization: Korea Advanced Institute of Science and Technology – KAIST
POC Email: hcbang@ascl.kaist.ac.kr

Need: Optical Stellar Interferometry using satellite constellation has never been realized as of the current status of the Space Sciences goes, though few of the missions like Terrestrial Planet Finder and Darwin have been proposed. And using the technique of Laser Optical Interferometry using the Nano – Satellite constellation would be for the detection of manmade debris which are the potential hazards for the new missions to the LEO such as the obsolete Spacecraft, Space Debris, MMOD and NEO. The technique of Optical Interferometry has been proposed for the single satellite missions like STARLIGHT, which was later merged with the Terrestrial Planet Finder (TPF) Mission, to tap the potential of distributed Space Systems.

Mission Objectives:

1. Successful demonstration of Optical Stellar Interferometry using the Laser Optical Interferometry using a constellation of nano satellites
2. To successfully demonstrate the efficiency of a distributed system of satellites instead of one big satellite
3. Constant updating of the Space Debris and MMOD information database
4. Mapping of the trajectories of the NEOs
5. Usage of COTS items for the nano satellites

Concept of Operations: As stated earlier, the main objective of the proposed mission is to demonstrate an effective constellation of nano-satellites for the given scientific mission. In order to have a continuous coverage of the target objects, this mission uses a single orbit – two formation flight – two satellites each. One satellite in each of the formation flights acts as both laser emitter and receiver of the reflected light signal and the other satellite in each of the formation flights acts as only receiver of the reflected light signal and then sends the collected data to the main satellite in each of the formation flight. The only receiver satellite is in time synchronization with the main satellite so as to receive the signal with a finite time difference. All the Satellites are equipped with a CDGPS Sensor, which precisely updates the position of the satellites all the time to the ground station, as well as a camera, apart from the Laser Range Finder (primary payload), in order to monitor its proximity with the other satellites through inter – satellite communication. The launch of the mission would be a single launch and would go for the launch by PSLV, taking into account the success rate of the launches and also the launch through PSLV is at a highly competitive price than compared to other launch vehicles.

The main concept of the working of the complete mission is based on the Optical Stellar Interferometry as described above. Initially, the main satellites of the formation flights start the scanning of the desired region by the emission of the laser signal and once when an alien space object is first encountered by the one of the formation flights, it sends the information to the ground station. Then the ground station processes the data and based on its distance and the other information from the formation flight, even the second formation flight is also engaged in focusing on that alien object and depending on the data acquired, both of the formation flights would be following that alien object and would map the target object. Once the signal reflected by the space object is received by the satellites, there would be some phase difference (Φ) as well as the wave length difference ($\Delta\lambda$) from that of the emitted signal. This complete set of data information, given the constraints of weight, is sent to ground, where it is processed and further action would be taken on the amount of information the data provides. The satellites in the formation flights are separated by approximately 4 meters of distance that comes out to be around 0.3 μs. Depending on the distance of the object that's being observed, the distance between the each satellite of the formation flight can be increased. As the mission completely focuses on the use of the nano – satellites, there would be cut – off distance from the formation flight where in the information obtained would be less accurate and less reliable. From the information processed, that is, the information of the $\Delta\Phi$ as well as the $\Delta\lambda$ at different time instances, even the velocity of the target object is estimated. One of the other functions is that once the velocity of the target object is estimated, the maneuvers required for the estimation of the size of the target object would be executed such as the scanning of the target object, continuous pointing of the satellites towards the target object.

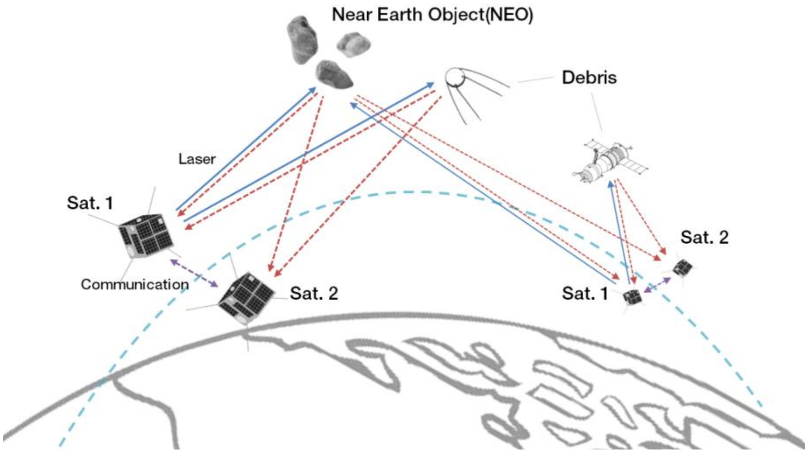

Figure 1: The artistic view of the working of the complete mission where the two formation flights are separated by 180^o.

Key Performance Parameters: One of the main functions of this mission is collecting the reflected signal form the space object by the main satellite as well as the receiver satellite in the formation flight. One of the principal features of this mission is to construct a database of the manmade debris which are the potential hazards for the new missions to the LEO such as the obsolete Spacecraft, Space Debris within a few kilometers of the formation flights at the Sun Synchronous Orbit with an inclination of 97.8432^o and an altitude of 681.7km. The complete mission depends on the strength of the reflected signal form the space object. So, one of the key parameters would be to collect the reflected signal with the proper time synchronization between the satellites of the formation flight.

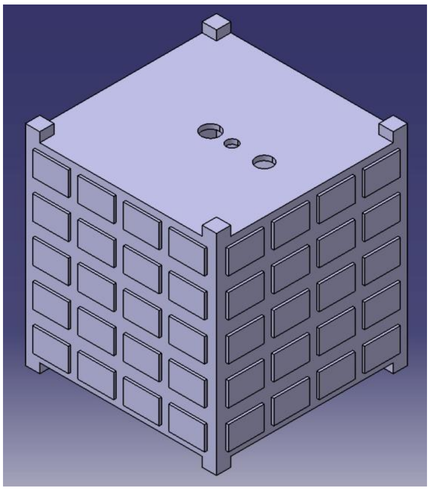

Figure 2: Main Satellite of the each formation flight (It acts of both transmitter and receiver, is a cube of 40x40x40 cm^3).

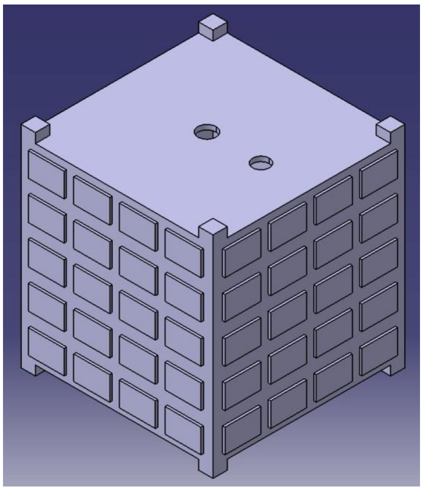

Figure 3: Secondary Satellite (Acts as a receiver and transmits the data to the main satellite, is a cube of 40x40x40 cm^3).

Figure 4: Candidate Payload (Main Satellite) – Er: Glass Laser Rangefinder (210 x 40 x 90 mm^3).

Space Segment Description: As described earlier, the mission uses a single orbit – two formation flight separated by 180° – two satellites in each formation flight. The orbit would be a Sun Synchronous Orbit with an inclination of 97.8432° and an altitude of 681.7km. The various specifications of the mission are described below, which includes Mass; Average Power etc. have been attached in the Appendix.

Orbit / Constellation Description:

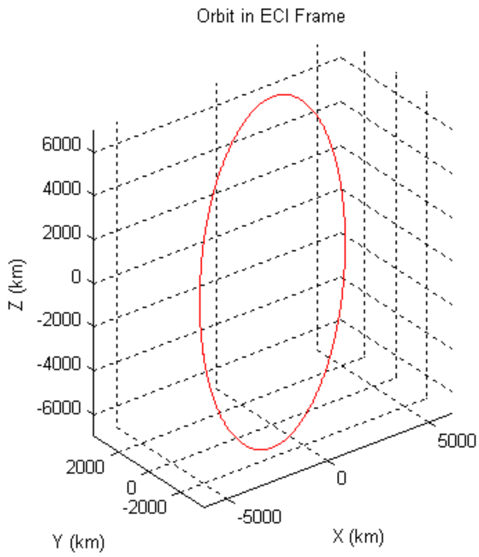

Figure 5: Candidate Mission Orbit.

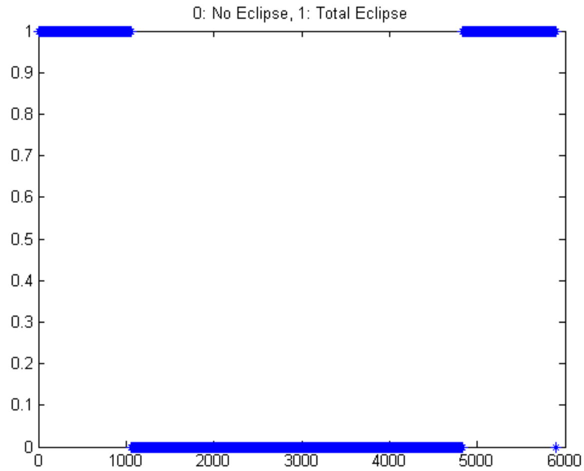

Figure 6: Simulations of the Eclipse.

Table 1: Orbital Elements

	KOMPSAT-1 (Target)	Proposed Mission Satellite
Semi – major Axis (km)	7052.7	7052.7
Eccentricity	0.0004353	0.0004353
Inclination (deg)	97.8432	97.8432
Right Ascension	209.6872	210.0372
Argument of Perigee	326.522	326.522
Altitude (km)	681.7	681.7

Figures 5 and 6 depict the desired mission orbit and the eclipses for the mission and also the simulations of the eclipse of the desired orbit. One of the main reasons to go for a Low Earth Orbit is that it is the Low Earth Orbits is populated with more Space Debris than the Other Orbits. The reason for going for the 180^o separated formation flight is for the potential targets that have different altitude than our system, which enables a constant coverage of the desired object. As it is evident that the proposed satellite system possesses the same orbit altitude as KOMPSAT(Korea Multipurpose Satellite) – 1, an obsolete satellite, which seems to be a promising candidate target object for our mission. For instance, considering the maintenance of the relative position with the KOMPSAT – 1, we have a constant coverage of KOMPSAT – 1 with only one satellite group. If the target is at either low or high altitudes, then the orbital speed would be different and in that case the relative position would be different, so, we need at least two satellite groups for a constant coverage.

Attitude Control: Attitude control for the spacecraft will be three-axis stabilization with a star tracker and reaction wheels. Sun sensor and magnetometer shall be augmented with star tracker for reliable attitude determination. The reaction wheel and rate gyros will be used for attitude stabilization as well as attitude pointing during the mission to ensure the returning signal is received by the receiving optics within the pointing accuracy of the spacecraft.

Ground Segment: KAIST Aerospace Engineering Department has satellite ground station equipment which needs a small amount of maintenance work for initial operation. For operation of a pair of satellites simultaneously, the ground station will have alternative contact with each satellite during contact. As the mission command can be executed in the form of stored command, the maximum load for ground station is expected to be a reasonable level.

Formation Flying (Orbit Maintenance): For formation flying of two satellites in a pair, onboard thruster systems shall be used. As the satellites could be launched as close as possible to the target objects – KOMPSAT1 with updated orbital elements information, the initial orbit acquisition work could be minimized. By using CDGPS sensor, thrusters, and inter-satellite link to exchange own orbit information with the other satellite, the relative formation dynamics could be maintained within the accuracy of CDGS sensor. To minimize drift effect, the formation flying would be focused on in-plane formation with periodic maintenance on semi-major axis and eccentricity.

Preliminary Bus System Budget (Single Satellite):

System	Margin	Comments
Link(TT&C)	17.198dB(Downlink) 25.777dB (Uplink)	Details provided in Appendix
Link(Inter-satellite)	26.670dB	Details provided in Appendix
Mass	1.784Kg	Details provided in Appendix
Power	15.725W	Details provided in Appendix
Cost	$ 1753800 (USD)	Details provided in Appendix

Implementation Plan: The MIC program will be primarily managed by KAIST Aerospace Engineering faculty members and graduate students of Aerospace Engineering Department. Prof. Hyochong Bang, an expert in the Satellite Systems, will be the Principal Investigator and would be responsible for the overall system management as well as the systems engineering work. Prof. Jae Hung Han, an expert in the Structural Design, will be responsible for Spacecraft Bus Systems design especially the Structural Systems. Prof. Martin Tajmar, an expert in the Space Propulsion, will be in charge of the propulsion subsystem. About 20 graduate students including the graduate students of *Space Exploration Engineering* program will be involved with the program. The graduate student team will be arranged into 6 or 7 groups to carry out dedicated Systems Engineering work.

The MIC team will have technical support from the nearby government research institute, KARI (Korea Aerospace Research Institute) and a local company, SaTReci, a pioneer in the Earth

Observation Systems. SaTRec (Satellite Technology Research Center), on – campus center, dedicated to the micro – satellite development and launch are also expected to provide technical support for the MIC program.

For Assembly, Integration and Test (AIT) of the spacecraft, facilities at KARI or SaTRec could be rented at affordable cost. All previous experiences by KARI, SaTRec, and SaTReci which have geographically favorable location for collaboration with KAIST will contribute towards the success of the MIC. The total development time will be about two years and some margin to reserve for unexpected delay.

Time Line:

Sl. No.	Event	Date
1.	Proposal of the Basic Idea	20th December, 2010
2.	Conceptual Project Presentation	14th – 16th March, 2011
3.	Preliminary Development Report – 1	September – October, 2011
4.	Preliminary Development Report – 2	January – 2012
5.	Initialization of Laboratory work and Fabrication	February, 2012
6.	Critical Design Report and Review	February – March, 2012
7.	Development and acquiring of individual Sub – System	September – October, 2012
8.	Assembly, Integration	November – December, 2012
9.	Testing	December, 2012 – January, 2013
10.	Submission of the Satellites for the Launch	February – March, 2013
11.	Launch	March – April, 2013

The main risks that are associated with the present mission are:

1. Time synchronization between the satellites in the formation flight
2. Failure of the satellite to receive the reflected signal from the space object
3. Failure of communication between the satellites of the formation flight
4. Tight schedule for system development and test
5. Meeting performance specifications in attitude control as well as navigation accuracy to implement the mission idea within the constraints in nano-satellites system.

References:

1. Terrestrial Planet Finder Mission Concept.
2. Larson, W.J., and Wertz, J.R., *Space Mission Analysis and Design*, Microcosm Press, 3rd Edition, 2005.
3. Dubovitsky Serge, et. al., Optical Metrology for STARLIGHT separated Spacecraft Stellar Interferometry Mission, JPL, California Institute of Technology.
4. Er: Glass Laser Rangefinder ELEM 10k, www.jenoptik.com/lasersensors

Appendix

Link Budget (TT&C):

		Units	Command	Telemetry and Data
Input	Frequency	GHz	0.145	0.45
Input	Transmitter Power	Watts	20	1
	Transmitter Power	dBW	13.01029996	0
Usual Value	Transmitter Line Loss	dB	-3	-3
Input	Transmit Antenna Beamwidth	deg	30	78
	Peak Transmit Antenna Gain	dBi	14.75757491	6.458107946
	Transmit Antenna Diameter	m	4.827586207	0.598290598
Input	Transmit Antenna Pointing Offset	deg	15	27
	Transmit Antenna Pointing Loss	dB	-3	-1.437869822
	Transmit Antenna Gain (net)	dBi	11.75757491	5.020238124
	Equiv. Isotropic Radiated Power	dBW	21.76787486	2.020238124
Input	Propagation Path Length	km	1440	1440
	Space Loss	dB	-138.8446099	-148.6815001
Usual Value	Propagation & Polarization Loss	dB	-0.3	-0.3
Input	Receive Antenna Diameter	m	0.1	1
	Peak Receive Antenna Gain (net)	dBi	-18.95901306	10.87787717
	Receive Antenna Beamwidth	deg	1448.275862	46.66666667
Input	Receive Antenna Pointing Error	deg	90	2
	Receive Antenna Pointing Loss	dB	-0.046340816	-0.022040816
	Receive Antenna Gain	dBi	-19.00535388	10.85583635
Table	System Noise Temperature	K	614	135
Input	Data Rate	bps	1200	1200
	E_b/N_0	dB	33.49807411	40.3773834
	Carrier-to-Noise Density Ratio		64.28988657	71.16919586
Input	Bit Error Rate		0.0000001	0.00001
Fig	Required E_b/N_0		11.3	9.6
Usual Value	Implementation Loss	dB	-5	-5
	Margin		17.19807411	25.7773834

Link Budget (Inter – Satellite Communication):

		Units	
Input	Frequency	GHz	2
Input	Transmitter Power	Watts	0.1
	Transmitter Power	dBW	-10
Usual Value	Transmitter Line Loss	dB	-3
Input	Transmit Antenna Beamwidth	deg	78
	Peak Transmit Antenna Gain	dBi	6.458107946
	Transmit Antenna Diameter	m	0.134615385

Input	Transmit Antenna Pointing Offset	deg	15
	Transmit Antenna Pointing Loss	dB	-0.443786982
	Transmit Antenna Gain (net)	dBi	6.014320964
	Equiv. Isotropic Radiated Power	dBW	-6.985679036
Input	Propagation Path Length	km	5
	Space Loss	dB	-112.45
Usual Value	Propagation & Polarization Loss	dB	-0.3
Input	Receive Antenna Diameter	m	0.1
	Peak Receive Antenna Gain (net)	dBi	3.834226808
	Receive Antenna Beamwidth	deg	105
Input	Receive Antenna Pointing Error	deg	90
	Receive Antenna Pointing Loss	dB	-8.816326531
	Receive Antenna Gain	dBi	-4.982099722
Table	System Noise Temperature	K	135
Input	Data Rate	bps	1200
	E_b/N_0	dB	42.97074457
	Carrier-to-Noise Density Ratio		73.76255703
Input	Bit Error Rate		0.00001
Fig	Required E_b/N_0		11.3
Usual Value	Implementation Loss	dB	-5
	Margin		26.67074457

Power Budget:

	Power (W)		**Power (W)**	**Note**
Communication	2.75	CDGPS Receiver	0.85	Phoenix receiver (DLR)
		Downlink Transmitter	1.7	UHF
		Uplink Receiver	0.2	VHF command receiver (ISIS)
		Inter-Satellite Comm. Module	0.5	S-band transmitter (ISIS)
OBC	1.5		1.5	
ADCS	3.325	Magnetometer	0.675	HMR2300r (Honeywell)
		Sun Sensor	0.25	1 unit
		Rate Gyro	0.4	QRS11 (Systron Donner)
		Star Tracker	2	Miniature Star Tracker (MST)
		Reaction Wheel	4.8	controllable upto 22kg
Payload	2.4	Laser Rangefinder	2.4	Stand-by, ELEM 10k (Jenoptik) (Max. current consumption 3A)
		Camera	0.2	Idle
		Total Power Consumption (W)	15.475	
Thruster	6		6	12 units
Payload	72	Laser Rangefinder	72	Operating, ELEM 10k (Jenoptik) (Max. current

		Camera	0.634	consumption 3A) Image Acquisition
		Day Power Generation	31.2	
		Charging Power (Wh) / Orbit	17.2975	
		Power Consumption (Wh) @ Eclipse	8.51125	
		Battery Size (Wh)	87.8625	Assumption: the mission lifetime is 1 years (DoD=10%)

Mass Budget:

	Mass (kg)		Mass (kg)	Note
Communication	0.35	CDGPS Receiver Downlink Transmitter Uplink Receiver Inter-Satellite Comm. Module	0.08 0.085 0.06 0.125	Phoenix receiver (DLR) UHF VHF command receiver (ISIS) S-band transmitter (ISIS)
OBC	4		4	Referred from HokieSat
Structure	4.5	Frame Solar Panel	1.5 3	 40cm x 40cm Solar panel
Power	0.75	Battery	0.75	Assumption: the mission lifetime is 1 years (DoD=10%)
ADCS	1.8545	Magnetometer Sun Sensor Rate Gyro Star Tracker Reaction Wheel	0.0645 0.5 0.06 0.5 0.73	HMR2300r (Honeywell) 1 unit QRS11 (Systron Donner) Miniature Star Tracker (MST) controllable up to 22kg
Thruster	0.6		0.6	12 units
Payload	0.94	Laser Rangefinder Camera	0.94 0.222	ELEM 10k (Jenoptik)
		Total Mass (kg)	13.2165	

Cost Budget:

	Cost (USD) (x 1000)		Cost (USD) (x 1000)	Note
Communication	48.15	CDGPS Receiver Downlink Transmitter Uplink Receiver Inter-Satellite Comm. Module	25 6.60 5.3 11.25	 UHF VHF command receiver (ISIS) S-band transmitter (ISIS)
OBC	7	Hardware	7	
Structure	38.1	Frame Solar Panel	11.5 26.6	 40cm x 40cm Solar panel

Power	10.75	Battery	10.75	Assumption : the mission lifetime is 1 years
ADCS	157.55	Magnetometer Sun Sensor Rate Gyro Star Tracker Reaction Wheel	1 20 15 90 31.55	HMR2300r (Honeywell) 1 unit QRS11 (Systron Donner) Miniature Star Tracker (MST) controllable up to 22kg
Thruster	40		40	12 units
Payload	50	Laser Rangefinder Camera	35 15	ELEM 10k (Jenoptik)
Constellation	1406.2		351.55 x 4	
AIT Cost (including labor)	2000		500 x 4	
Labor	840	Students ($0.75 * 20 * 24) Experts ($5 * 4 * 24)	360 480	
		Total Cost (USD)	4246.2	