Concept for a Small Satellite Mission SUNSAT-XE

Proposed by
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ABSTRACT

The paper describes the small satellite mission SUNSAT-XE. It is a planned cooperation between German and South African universities and research organisations in the field of satellite technology and space-born earth observation for the benefit of science, technology and students education.
Today, high resolution multispectral imaging in the optical/near infrared wavelength regime from space at affordable costs can be considered as one of the major goals in the field of satellite technology and engineering. A reasonable design goal is a spatial resolution in the range of 5 – 10 m which will allow agricultural applications and vegetation monitoring.
An important technological goal is to develop satellites of a small physical size and mass which can be operated in very low earth orbit. Orbit heights as low as 350 km will be considered, which will require electrical thrusters and autonomous on-board orbit altitude control to compensate atmospheric drag and allow phasing of multiple satellites in the same orbit. Within the framework of the partnership between Universität der Bundeswehr München, Neubiberg / Germany, University of Stellenbosch, Stellenbosch / South Africa and the Deutsche Zentrum für Luft- und Raumfahrt (DLR), Oberpfaffenhofen/Berlin / Germany, we plan to design, build, launch and operate a small Earth Observation Satellite by combining as well existing knowledge and experience of the partners as also the enthusiasm and talent of young students in the field of aerospace and electrical engineering.

1. INTRODUCTION

The overall objective is to develop satellites of a small physical size and mass that can provide multispectral imagery for agricultural applications and natural vegetation monitoring. Technically this will require lightweight imagers that can provide high-resolution images from very small satellites at very low autonomous controlled altitudes choosable during the mission. The engineering task will be a co-operative effort between research / academic institutions to simultaneously develop more than one complete satellite of this class for each institution. Two more important objectives will subsequently be demonstrated by this approach: (i) cost savings from synergy and sharing of subsystems, (ii) eventually having more similar satellites in orbit to reduce the revisit time to targeted remotely sensed areas, and (iii) sharing the data gained during the mission.

2. MISSION AND TECHNOLOGY

The main payload will be a multispectral imager that can perform remote sensing for agricultural applications and natural vegetation monitoring. Spatial resolutions in the range of 5 – 10 m is achievable and will compare well with that of SPOT 5 and Landsat 7. With

present detector technology, a swath width in the order of 50 - 100 km can be set as a reasonable design goal. With the targeted applications fields, the following spectral bands are recommendable: (i) blue, (ii) green, (iii) red, (iv) red edge, (v) near infrared and (vi) one panchromatic band overlapping the previous bands.
Other planed possible payloads are amateur radio and several space physics payloads not jet defined in detail.
Due to the high resolution of the imager, plus the fact that the multiple detectors will probably be parallel instead of co-aligned, and a desire to scan targets alongside the satellite nadir, a high accuracy three-axis attitude control system will be developed. The system will provide accurate position and rate control during imaging operations, and keep the solar panels sun pointing during the rest of the mission for maximum electrical power generation.

Technological goals are also set for data handling. The communications system will be designed with frequencies, bandwidths and communications protocols, which are compatible with existing remote sensing data collection centers. On board mass memory on the satellite will enable at least one full scene from anywhere on earth to be stored at a time.

To keep aperture and focal length of the imager (and as a consequence the physical size of the imager) within the dimensional constraints of the overall satellite, the satellite will be maintained in very low earth orbit. Orbit heights as low as 350 km will be considered, which will require electrical thrusters and onboard orbit altitude control to compensate atmospheric drag. The system will also facilitate phasing of multiple satellites in the same orbit in order to improve target revisit frequencies. One of the technological goals is to develop such an autonomous onboard orbit maintenance system to establish the desired satellite constellations.

Assuming realistic atmospheric densities which include the seasonal and local time dependence as well as the solar F-number and the geomagnetic activity index K_p , it can be shown that the drag force acting on an orbiting satellite exposing an areal cross section of ca. 1m x 1m lies in the range of 1 to 20 mN. This range compares well with a thrust range provided by ion thrusters such as the RIT 10 (EADS/Astrium) Xenon propulsion unit (0.2 to 20 mN). Such a thruster can be used to compensate the atmospheric friction and can keep a predetermined position to an accuracy of +/- 500 m in combination with a GPS on board orbit positioning system, orbit estimator and state–space controller (2,3). Integrated into several satellites, this system will be able to guarantee a spatial constellation for a long period of time. It could be demonstrated in simulation runs with the Small Satellite Simulator KSS of the UniBwM that such a system is stable against random and selective availability (SA) noise (3). The vector of the extreme low but variable thrust of the ion propulsion system will have the moution point in the center of mass of the satellite and will not create undesired cross-coupling effects from the orbit control system into the attitude control system typical for standard propulsion systems.
A further advantage is the very low propulsion mass needed for orbit keeping. It was estimated that aproximatly 17 kg Xenon will be needed to support a 3 years mission at 300 km. Considerably less Xenon gas mass is estimated for an operation at 350 km altitude and a one year mission. The operation of the RIT 10 unit requires a power of about. 100 W and needs additionally 30 W per mN thrust (3).

Except for the areas discussed above (where micro satellite technology improvements will be made), the next generation of satellites will as far as possible re-utilize the space proven SUNSAT 1 bus systems. This would include the onboard computer system, telecommand and telemetry system, power system and much of the attitude determination / control and telecommunications systems. Other satellite subsystems form institutions in Germany will also be re-utilized as appropriate for the next generation of micro satellites.

3. STUDENTS EDUCATION

An important goal of the micro satellite development programme is to combine it with student education. Students in electronic, aerospace and mechanical engineering as well as in computer science and physics will receive parts of the satellite development as their thesis responsibilities. They will also be involved in the practical aspects of satellite systems integration, testing, project management and data analysis.

4. SYSTEM CONFIGURATION

4.1 System

The system should compose of two satellites. One build under South African and the other build under German responsibility.
On one hand this will give both partners a big independence and on the other hand the revision time for agricultural sience and application will be 7 days. The global coverage for this 2 satellite constellation was simulated and is shown in the pictures
Orbit: 349 km
Inclination: 96.867°
SSO 10:30 with 7 days exact revisit time
Swath: 80 km

The payload is nadir looking but the satellites could be turned with 15° to each side
The possible access area is 280,586 km (the maximum outer angel of the optic will be 21,543°)
The phasing will be 39,61°.

4.2 Satellite

As a design goal, the ASAP5 (Ariane 5 Structure for Auxiliary Payload) specification for micro satellite payloads will be used, namely: total satellite mass less than 120 kg, outer dimensions less than 600 x 600 x 710 mm.
Its outer surface including also both sides of the deployable solar panels will be covered with solar cells (GaAs _=20%). In order to keep the overall structural mass to a minimum and to guarantee the necessary structural strength and stiffness, a CFRP central tube housing the optical system of the camera will be used as the load carrying structure with a base plate and an upper plate attached to the central tube. Presently, it is foreseen to fabricate the plates out of Al-honeycomb with Al-face sheets because of better heat transfer capabilities in comparison to foam plates with CFRP face sheets. The side panels will be fabricated from lightweight CFRP laminates in a sandwich construction with PMI foam core. The mass budget is shown in Table 1 (see below).

The power management subsystem consists of the primary power sources, a NiH_2 battery and GaAs solar generator, a Power Conditioning Unit (PCU), a Dc/Dc converter including current limiters and switches. Nominal bus voltage is 28 V. Power duty cycling will be designed that maximum power (approx. 170 W) has to be supplied only for occasions of simultaneous telemetry transmission and camera operation (approx. 10 minutes per average

orbit) and activation of the orbit control subsystem (ion thruster). When not in sunlight, the battery will supply the necessary power to the spacecraft. The size of the solar generator will be large enough to charge the batteries when in sunlight and in standby mode (see Table 2 below).
The satellite will be thermally insulated by Multi-Layer-Insulation (MLI) blankets. In order to cope with worst-case thermal conditions possibly existing in the transfer orbit, heaters will be installed at selected locations.

The Onboard Data Handling Subsystem (OBDH) combines the functionality of
a. the conventional onboard Data and Command Subsystem with
b. the logical function of the mass storage unit
c. the logical functions of the Attitude Control Subsystem
d. the logical functions of the Power Management Subsystem
e. the logical function of the TM/TC subsystem
f. the logical function of the Thermal Control Subsystem

For reliability reasons a dual-redundant microprocessor system shall be selected.

The Attitude Control Subsystem (ACS) will provide a pointing accuracy and stability compatible with the camera requirements. Its actuators consist of a three-axis reaction control wheel system. Desaturation is provided by an electromagnetic torquer system. Attitude sensing is provided by a redundant set of star sensors, laser gyros and magnetometers.

The autonomous onboard orbit determination and control subsystem (ODCS) consists of a GPS navigation receiver, associated interface and control electronics and the Xenon ion thruster. The ion thruster is composed of the ion engine, a power and flow-conditioning unit (PCU/FCU); the rf-generator needed to ionize the neutral Xenon atoms, and associated high-pressure tanks including piping.

The TM/TC subsystem is fully redundant and uses S-band frequencies. The antennas will consist of a prime helix antenna directed into the nadir direction and a pair of low gain antennas providing omni directional coverage. Depending on the mission profile the transmitter will be activated only when being in the visibility range of a ground station. The contact will be used to dump the mass storage, to configure the satellite and to load the time-tagged command memory.

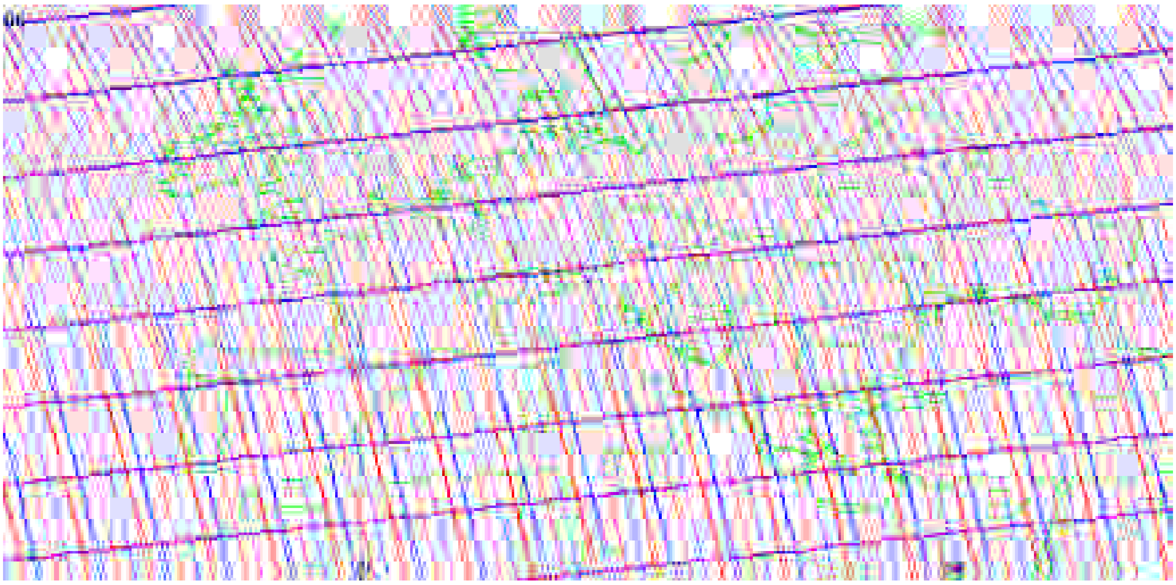
Table 1: Mass Budget.

		Mass [kg]	Commentes/Dimensions
S/C Bus			
Structure		10.6	Cube 0.8 x 0.8 x 0.8m + central tube for camera
Central Tube	2.5		Dia. = 400 x 5mm; h = 550mm; rho =1.536kg/m_
Base Plate and Outer Plates	5.6		0.8m x 0.8m x 25/15mm
Struts/screws/bolts/hinges/etc.	2.5		
Thermal Control		2	
Attitude Control System (ACS)		18.6	
Reaction Wheel 3 units á 4kg	12		Teldix TR30CFR; 35Am_/4.2W; l = 500mm, Dia. = 25mm
Magnetic Torquers 3 units á 1kg	3		Ithaco; 4Nms; 0.02Nm; Dia. = 205mm, h = 100mm
Star-Sensor 2 units á 0.6kg	1.2		110 x 95 x 65
Laser Gyroscope	0.6		Dia. = 94; h = 72
3-Axes Magnetometer	0.2		
GPS Receiver + Antenna (1.3+0.3)kg	1.6		(127 x 207 x 56)mm & (96 x 102 x 19)mm

Power		15.2	
PCU	2		290mm x 140mm x 165mm (EADS Dornier)
Solar Arrays	3		five surfaces (about 3.2m_, GaAs; EOL = 20%)
Batteries 12x30Wh	10.2		NiH2; 360Wh; Dia. = 90; h = 200; each => 200x180x540
OBDH		2.0	2 Siemens C167 processors; 0.25x0.20x0.10 mm
Mass Storage		1.5	
TM/TC S-band (redundant)		5.0	
Antenna		1.5	Helix + patch antennas
S/C Payload			
Camera		13.0	**Mass + Electronics**
Sensor/Optic/near sensor electronics	11		Length: 0.7m; Dia.: 0.4m;
DPU + Solid State Buffer	2		0.25 x 0.20 x 0.10 m
Ion Thruster RIT 10		17.7	
Thruster	2.2		Dia. = 100mm; h = 100mm
PCU/FCU/HF-Gen.	12		230 x 280 x 180 / 115x110x75 / 190x130x115 mm
Harness & Piping	3.5		for thruster and tanks
Xenon Tanks 2 units; á 3.5 kg		7.0	6.55l each, Pmax = 310bar, Dia. = 260mm
Propellant (Xenon)		9.0	310bar, 273K
Harness, Connectors, etc.		5.0	
Balance Mass		1.5	
Interface Satellite/Launcher		4.0	
Margin		8.0	
Total Mass		**119.6**	

Table 2: 2: Power Budget.

	Power	Duration	Energy	Commentes
	[W]	[min]	[Wh]	T_orbit=5431sec| h=300km=90.5min
Thermal Control	5	45.8	3.8	
Attitude Control System				
Reaction Wheel 3 units	7	90.5	10.6	
Magnetic Torquers 3 units	4.2	8.5	0.6	
Star-Sensor 2 units	2	90.5	3.0	100 % (each 1W)
Laser Gyroscope	3.9	90.5	5.9	100 %
3-Axes Magnetometer	0.1	90.5	0.2	100 %
GPS Receiver	4.5	90.5	6.8	100 %
PDU	2	90.5	3	100 %
OBDH	4	90.5	6	100 %
Mass Storage	0.5	90.5	0.5	100 %
Receiver	2.4	90.5	3.6	2 units, hot redundancy
Transmitter (2W)	6	3.5	0.5	3.5 min/orbit
Camera	30	5	2.5	5 Scenarios/day
Ion Thruster RIT 10			105.5	(from KSS)
			152.8	**TOTAL ENERGY DEMAND [Wh]**
Solar Arrays	**257**	**45.8**	**196.2**	**TOTAL ENERGY INPUT (worst case)**
1353W/m_*0.2*0.95m_				
Battery-Charging			**13.4**	**[Wh/Orbit]**



. The coverage analysis shows a revision time of exact 7 days

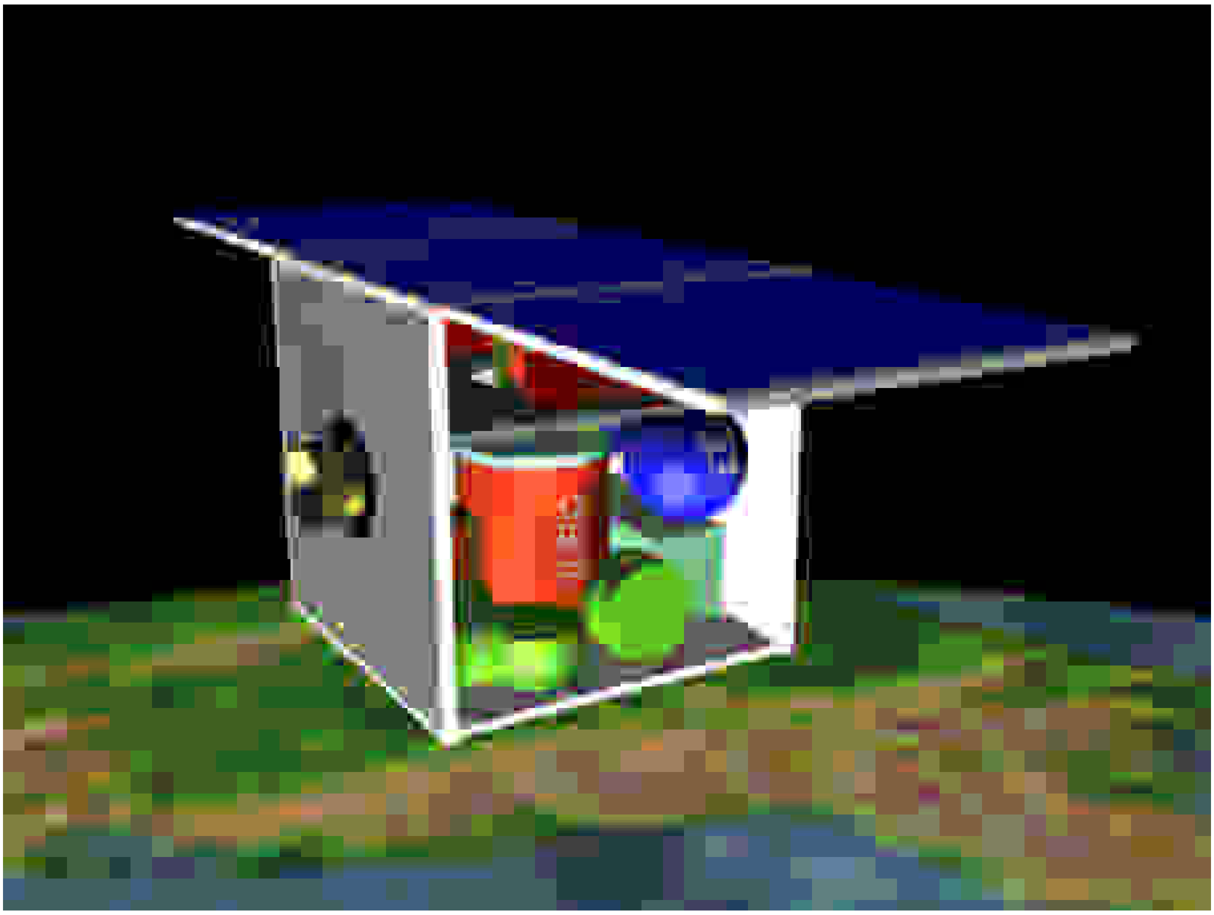


Figure 4: SUNSAT - XE in orbit showing Xe-thruster, satellite subsystems and camera payload

4.3 Launcher

Despite the fact that it is intended to design the satellite compatible to an Ariane5 (ASAP 5) launch interface other launchers will be considered as well. The preference will eventually be given to a launch system injecting the satellite into an orbit as close as possible to the desired low circular orbit. Due to the fact the existence of the orbit control system any other orbit could be chosen for a piggyback launch. For example the calculated fuel consumption to change an orbit from ca.650km to 340km is half a kilo.

4.4 Model Philosophy

It is planned to build a structural/thermal test model (STM) and an electrical model (EM), which will serve also as protoflight model (PFM).

5. PROPOSED WORK SHARING AND RESPONSIBILITIES

Based on the experiences the partners will combine their know-how and efforts on a no exchange of funds basis.

6. PROGRAMMATIC AND MANAGEMENT ASPECTS

Based on the proposed scheme of shared responsibilities the SUNSAT-Xe project will be led by a project management and design office (PMDO) headed by a project manger (PM). It is proposed to locate the office at the University Stellenbosch. The project manager will be assisted by one experienced project manager/engineer from each participating institution who will communicate with the PM via electronic media. The project engineers shall be supported also in their management work by students who will gain throughout the course of this program considerable experience in the management of space projects. The project will structured in four phases (A-D) and shall be controlled by a steering committee consisting of two senior experts representing the funding authorities of both countries involved. At least three reviews shall be conducted during the project.

7. LITERATURE AND REFERENCES

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