ESTEC/Contract No. 18474/04/NL/LvH
Nano Satellite Beacons for Space Weather Monitoring
WP 410. Prospects and Recommendations

Nano Satellite Beacons for Space Weather Monitoring: Final Report

SWNS-RAL-RP-0001
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TABLE OF CONTENTS

1 Preface

1.1 Document change record

Issue	Date	Notes/remarks
Issue 1.0	28 Feb 2008	First issue
Issue 1.1	04 Aug2008	Updated in response to ESTEC comments: • Add reference to Cubesat community web site plus referencing text in sec 5.7 & Exec Summary. • Need for in-flight demonstration of MNT cross-referenced in the programmatics section & highlighted there as new recommendation 14. Also highlighted in ideas for future work (sec 16.2) and the Executive Summary. Also notes that this is a key element in assessing nanosat reliability. • definition of a nanosat added to Exec Summary • old figure 3 too complex – replaced by image of a passive device to focus on key message: that MEMS allows you make good quality miniaturised passive devices that can be combined with Ics. Text updated to match changed image. • Future nanosat mission objectives added as Annex J whilst a summary and cross-reference has been added in the main text. • Table 11 notes that M-cubed is now CrossScale • Section 6.3, point 3. Removed claim that solar imager is 'probably' best deployed in EUV. • Added a paragraph under future work to note the 2007 report that energetic electrons are seen 10-30 minutes before major SEP events – and that this is a potential new requirement that should be watched as the science develops. • Dropped footnote (old page 98) referring to a google search on 'constellation autonomy and formation flying'. • Dropped last paragraph in section 15.5 as it repeated earlier information. • Updated figures showing the nanosat configurations: added cross-refs to the instrument discussions in section 7, removed incorrect “new instrument” labels and cross-refs to the requirements (these are traced via the instruments) • Fix systematic error in cross-refs given in the systems analysis configuration sections • Extensive minor corrections (e.g. to spelling, punctuation and case) following proof-reading .

1.2 Purpose of Document

This document is the final report on the study of Nano Satellite Beacons for Space Weather Monitoring under ESTEC Contract No. 18474/04/NL/LvH. It presents a substantive review of the whole programme of work undertaken under this study.

1.3 Definitions, Acronyms and Abbreviations

2-D	Two dimensional
3-D	Three dimensional
AOCS	Attitude and orbit control system
APS	Active pixel sensor
ASAP	ARIANE Structure for Auxiliary Payloads
ASIC	Application-specific integrated circuit
ATB	Advanced Test Bed
BOL	Beginning of life
CCD	Charge coupled device
CCLRC	Council of the Central Laboratory of the Research Councils
CCSDS	Consultative Committee on Space Data Systems
CDF	Conceptual design facility
CHAMP	CHAllenging Minisatellite Payload
CME	Coronal Mass Ejection
CMOS	Complementary metal–oxide–semiconductor
COSMIC	Constellation Observing System for Meteorology, Ionosphere and Climate
COTS	Commercial off-the-shelf
CPU	Central processing unit
CRC	Corporate Research Centre
CSMR	Consolidated System Measurement Requirements
EADS	European Aeronautic Defence and Space company
E_b/N_0	energy per bit to noise power spectral density ratio
EIT	Extreme Ultraviolet imaging Telescope
EOL	End of life
ESA	European Space Agency
ESTEC	European Space Technology Centre
EUV	Extreme Ultraviolet
GEO	Geosynchronous orbit
GeV	Giga-electron volt
GNSS	Global Navigation Satellite System
GPS	Global Positioning System
GTO	Geosynchronous transfer orbit
IR	Infra-red
IRF	Swedish Institute of Space Physics
Isp	Specific impulse
keV	Kilo-electron volt
L1	Lagrangian libration point 1
LAP	Langmuir Probe
LASCO	Large Angle and Spectrometric Coronagraph experiment
LEO	Low Earth orbit
LEOP	Low Earth orbit operations
MOEMS	Micro-opto-electromechanical systems
MEMS	Micro-Electro-Mechanical Systems
MeV	Mega electron-volt

MNT	Micro- and nano-technology
n/a	Not applicable
NASA	National Aeronautics and Space Administration
PCB	Printed circuit board
PDA	Personal data assistant
PFM	Proto-Flight Model
RAL	Rutherford Appleton Laboratory
RF	Radio frequency
RPC	Rosetta Plasma Consortium
SOHO	Solar and Heliospheric Observatory
SOW	Statement of Work
SSD	Solid State Detector
STFC	Science and Technology Facilities Council
STM	Structural and thermal model
SWWT	Space Weather Working Team
TBD	To be defined
TEC	Total electron content
TIDI	TIMED Doppler Interferometer
TIMED	Thermosphere Ionosphere Mesosphere Energetics and Dynamics
URL	Uniform Resource Locator
UV	Ultra-violet
WP	Work Package

1.4 References

We list here the various documents used as source material for this report. These include both hardcopy and web sources. Documents may be referenced in the text and this is indicated by a sequential code of the form Xn, where n is an integer and X = A or R (for applicable and reference documents respectively). The series of integers are separate for applicable and reference documents.

1.4.1 Applicable documents

A1 Statement of Work, Nano-Satellite Beacons for Space Weather Monitoring,
Reference: TOS-EES/2004.153/AG

A2 ESTEC Contract No. 18474/04/NL/LvH
Nano Satellite Beacons for Space Weather Monitoring

A3 Proposal for Nano Satellite Beacons for Space Weather Monitoring, RAL/RRS/228/03

A4 Minutes from Negotiation Teleconference, 23/09/2004, SWNS-RAL-MN-0001

1.4.2 Study reports

R1 Space weather effects and requirements analysis for space weather monitoring by nanosats,
SWNS-RAL-TN-0001

R2 Refinement of Requirements for Nano Satellite Beacons for Space Weather Monitoring,
SWNS-RAL-TN-0003

R3 Micro and Nano Technology Review

R4 Review of MNT relevant to WP110 requirements

R5 Nano Satellite Beacons for Space Weather Monitoring: WP210 – Instrument requirements

R6 Nanosatellite Evolution & Definition of Spacecraft Solutions

R7 ESA Nanosatellite Beacons for Space Weather Monitoring Study

R8 Nano Satellite Beacons for Space Weather Monitoring: :Prospects and Recommendations
SWNS-RAL-TN-0004

R9 Notes from Solutions Workshop, ESTEC, 7-11 Feb 2005, SWNS-RAL-TN-0002

1.4.3 Other reference documents

R10 O. Norberg O et. al, Munin: A Student Nanosatellite for Space Weather Information, *Proceedings COSPAR Colloquium on Scientific Microsatellites*, 352-364, ed. Fei-Bin Hsiao, Pergamon, 1999
R11 Space Weather Feasibility Studies (RAL): http://www.wdc.rl.ac.uk/SWstudy/
R12 Space Weather Feasibility Study (Alcatel): http://www.estec.esa.nl/wmwww/WMA/spweather/esa_initiatives/spweatherstudies/public_doc.html
R13 European Space Weather Programme System Requirements Definition, ESWP-DER-SR-0001, available via http://www.wdc.rl.ac.uk/SWstudy/
R14 ESA Space Weather Programme - Alcatel contract, Space segment - Measurement and system requirements, WP 2200 and 2300 reports, available via http://www.estec.esa.nl/wmwww/WMA/spweather/esa_initiatives/spweatherstudies/public_doc.html
R15 Space Weather CDF Study final Report: http://www.estec.esa.nl/wmwww/wma/spweather/esa_initiatives/spweatherstudies/CDF_study/cdf.htm
R16 Space Weather Applications Pilot Project http://www.esa-spaceweather.net/spweather/esa_initiatives/pilotproject/pilotproject.html
R17 The SOHO Mission: Scientific and technical aspects of the instruments. ESA SP-1104.
R18 TIMED Doppler Interferometer (TIDI), http://tidi.engin.umich.edu/
R19 Welcome to WAVES, The Radio and Plasma Wave Investigation on the WIND Spacecraft http://lep694.gsfc.nasa.gov/waves/waves.html
R20 C.P. Escoubet, Russell, C. T.; Schmidt, R. The Cluster and PHOENIX missions, Kluwer, Dordrecht: 1997
R21 http://sec.noaa.gov/alerts/description.html#proton
R22 http://sec.noaa.gov/alerts/description.html#electron
R23 SPEE report, see http://www.ava.fmi.fi/spee/
R24 J.K. Hargreaves, The solar-terrestrial environment. Cambridge University Press, 1992.
R25 ESA web site for Wireless Data Communications Onboard Spacecraft, http://www.wireless.esa.int/
R26 CCSDS Spacecraft Onboard Interface Services, Wireless Birds of a Feather web site: http://public.ccsds.org/sites/cwe/sois-wir/
R27 Use of on-board autonomy for future space plasma studies, Proc. Cluster and Double Star symposium, 5th anniversary of Cluster in Space, ESTEC, Noordwijk, Netherlands, 19-23 Sep 2005
R28 Solar Wind Experiment (SWE) on the NASA WIND spacecraft: How a Faraday Cup works. http://web.mit.edu/space/www/wind/wind_instruments.html#Diagram
R29 Mind map, http://en.wikipedia.org/wiki/Mind_map
R30 Spacecraft Position, Velocity and Time, http://www.sstl.co.uk/index.php?loc=62
R31 http://www.fillfactory.com/htm/products/datasheet/star1000.pdf
R32 Grüger H., Gottfried-Gottfried, R. „CMOS Integrated Two Axes Magnetic Field Sensors – Miniaturized Low Cost Systems With Large Temperature Range“, Fraunhofer Institute for Microelectronic Circuits and Systems IMS, Dresden, Workshop "Preparation, Properties, and Applications of Thin Ferromagnetic Films" Vienna, June 2000
R33 National Institute of Standards and Technology NIST “Magnetic Field Sensors Roadmap”, Draft-Version, November 2003
R34 Voigt M. "Results of the 1st European Fuel Cell Study" VDI/VDE-IT Rio de Janeiro, 2004, www.fuelcellnet.com
R35 W. R. Crain, Jr, D. J. Mabry, and J. B. Blake, The Aerospace Space-Radiation “Dosimeter on A Chip”, Workshop on ionising particle measurements in space, ESTEC, Noordwijk, The Netherlands, February 2005.
R36 M. Ferrante, A. Ortenzi, L. Di Ciolo, M. Petrozzi, V. Del Re: “Debris Measure Subsystem of the nanosatellite IRECIN”, 9th International Symposium on Materials in a Space Environment - June 16-20 2003 ESTEC, The Netherlands.
R37 A. Mannucci et al, COSMIC and space weather http://www.cosmic.ucar.edu/gpsro2/presentations/Mannucci_GPSRO_2005.pdf
R38 M.J. Angling and B. Khattatov, Comparative study of two assimilative models of the ionosphere, Radio Science **41**(5), doi: 10.1029/2005RS003372, 2006.
R39 CHAMP - CHAllenging Minisatellite Payload http://www.gfz-potsdam.de/pb1/op/champ/index_CHAMP.html

R40 COSMIC - Constellation Observing System for Meteorology, Ionosphere and Climate http://www.cosmic.ucar.edu/

R41 Daniel Tran, Steve Shien, Rob Sherwood, Rebecca Castano, Benjaming Cichy, AshleyDavies, and Gregg Rabideau. The Autonomous Sciencecraft Experiment Onboard the EO-1 Spacecraft. Proceedings of Autonomous Agents and Multiagent Systems 2004, 2004.

R42 Jan Bergman, NanoSpace-1 web presentation, 2005. http://www.physics.irfu.se/Newpages/Satellites/nanospace.html

R43 SWARM, Space Weather Advanced Research Mission, The First Coherent Mission to Comprehensively Investigate the Global, Dynamic Magnetosphere, Submitted to ESA F2/F3 Flexi-Mission Call for Mission Proposals, 31 January 2000

R44 CubeSet Community Website, http://cubesat.calpoly.edu

R45 Posner, A. (2007), Up to 1-hour forecasting of radiation hazards from solar energetic ion events with relativistic electrons, *Space Weather*, *5*, S05001, doi:10.1029/2006SW000268.

1.5 Acknowledgements

I would like to thank all my colleagues who contributed to this study: Steven Eckersley from Astrium UK Ltd, Rickard Lundin from the Swedish Institute of Space Physics, Petrus Hyvönen from Orbitum AB, and Martin Kluge, Ulrich Prechtel and Kay Koppenhagen from EADS Deutschland GmbH. The work at Astrium UK was influenced by MSc thesis work of Kristina Larfars at Cranfield University. This work was suggested and monitored by Astrium and allowed a deeper investigation of the spacecraft issues reported in section 9.

2 Executive summary

This study has assessed the opportunities for using nanosats (i.e. spacecraft ≤ 10 kg) to monitor key space weather parameters. This started with a review of space weather measurement requirements and of the likely capabilities of nanosat technology to address those requirements (including using of MNT). The outcome was a set of space weather nanosat constellations, each of which addresses a focused set of measurement requirements. These constellations were then analysed to develop an outline concept for each case.

The review of capabilities looked at the current state of relevant work on MNT – looking at the development of important MNT devices such as RF components, accelerometers and power sources, at the packaging of MNT systems and of practical experience in flying an MNT spacecraft. The study also made an extensive review of current and future developments in nanosats, including the growing US and European interest in Cubesats.

The requirements review looked at requirements for space-based measurements to support space weather applications that help both space-based and ground-based services. This review consolidated the outputs of earlier ESA space weather studies and updated them to take account of recent developments in space weather services. The study identified where these measurements can be performed on nanosats and explored how to classify them.

A key conclusion was that it is not worthwhile to classify the measurements by their applicability to ground-based or space-based services. This is simply because the majority of space-based measurements have applicability to both domains. This reflects the chain of space weather from its source on the Sun to its impact on and around the Earth. The majority of the space-based measurements monitor the upstream space weather environment (e.g. solar and solar-wind measurements) which is critical to both space-based and ground-based services.

This negative conclusion is balanced by a positive conclusion – that it is possible to classify the space weather measurements into a small set of distinct spacecraft constellations: (a) two low-Earth orbit constellations aimed at ionospheric and solar observations, (b) a constellation in geosynchronous transfer orbit aimed at radiation belt and plasmasphere observations; (c) a Molniya constellation aimed at remote sensing of auroral activity, (d) a multi-orbit constellation for better measurements of the magnetospheric magnetic field; and (e) an L1 spacecraft for monitoring the solar wind and heliospheric particle fluxes.

This classification has driven the design elements of the study. These have developed a set of outline designs for each constellation and for the instruments that must be carried by each constellation. The instrument solutions are largely based on existing heritage with some extrapolation for developments in instrument miniaturisation. In one case a novel instrument concept is proposed – namely a low-resolution EUV solar imager for flare location.

The design work has outlined a nanosat concept for each constellation and explored how the nanosats might be launched, operated and de-orbited. One key issue here is the design of data links. This is always a critical issue for space weather missions since most missions have requirements for near-real-time downlink to ensure timely availability of data. Where feasible the data link designs make use of innovative ideas such as inter-spacecraft links and small ground station antennae.

Another key issue is the replacement strategy. Space weather services require continuity of data so it is important to replace spacecraft at regular intervals to reduce the risk of data failure. The use of constellations raises interesting issues about reliability – since the failure of one spacecraft will degrade constellation performance but not necessarily destroy it. Thus we had to consider how constellations could adapt to overcome failures, e.g. routing signals past failed spacecraft and adjusting spacecraft positions to optimise data sampling, This led us to consider the number of failures that each constellation could tolerate and then to model the likelihood of multiple failures (using a simple numerical model). This model allowed us to estimate replacement periods for each constellation and thus was central to developing the required replacement strategies (frequency of launches and need to produce multiple copies of spacecraft).

The final element in the study was to look at the prospects for using nanosats in space weather monitoring and make recommendations on how to bring those prospects to fruition. One set of recommendations is the need to develop instruments that are well suited to routine monitoring (rather than to measurements that support scientific research). We need to develop instruments that are smaller (in size, mass and power) in order to gain more opportunities to fly monitors; a key issue here is to reduce space weather sensor sizes, e.g. through simplified requirements or use of different measurement techniques. We also need sensors that are robust against extremes of space weather, in particular solar proton events. Finally we also need to reduce instrument costs. The long-term success of any programme for monitoring space weather will depend on the ability to build and replace adequate numbers of instruments.

Another set of recommendations is the need to develop ways to make best use of MNT/MEMS devices in space. This has several aspects including (a) developing methods to qualify MNT/MEMS devices for use in space, (b) developing methods to assess and mitigate radiation sensitivity of MNT/MEMS devices and (c) developing design environments and standards that are appropriate for multi-functional systems. The latter is particularly important as it will exploit synergies that can facilitate nanosat construction (e.g. reducing mass and cost) but it cuts across the traditional approach of decomposing design into separate systems. A key element in making best use of MNT/MEMS is to test and validate new devices in space; we therefore recommend establishing a programme for flight demonstration of technologies that will facilitate use of nanosats.

There is also a set of recommendations on communications links. This is a critical issue for space weather measurements because the majority of measurements have a requirement for real-time downlink. To address this problem it is important to develop European sources for low power RF systems (i.e. suitable for use on nanosats). It is also important to develop schemes for data compression; in the context of space weather data, the best way to do this is to develop schemes for on-board processing of raw data to un-calibrated physical parameters (e.g. calculation of moments for particle measurements). This exploits our knowledge of the underlying physics to make an intelligent compression of the data. It is likely to provide much better compression than mathematical schemes that have no knowledge of the data. In the longer-term we should encourage and exploit generic development of advanced satellite communication systems, e.g. on-demand links, as these will greatly facilitate data access from any spacecraft.

There are also recommendations on other important technical issues (to explore autonomous operation, to develop good methods for nanosat propulsion and for deeper study of nanosat constellation reliability) and on programmatic issues: (a) to explore and exploit the potential for synergy between operational space weather measurements and research programmes in solar-terrestrial physics, and (b) as noted above, to establish a programme for flight testing and validation of new technologies.

3 Introduction

This report presents the results of the space weather nanosat study carried out for ESA under Contract No. 18474/04/NL/LvH. The logic of the study is shown in Figure 1 below. The study started with a preparatory phase comprising workpackages (a) to establish requirements for space weather monitoring by nanosats, (b) two reviews of micro- and nano-technology (MNT) status – one by Swedish Institute of Space Physics (IRF) drawing heavily on their experience with nanosats such Munin [R8] and the other by EADS Deutschland (CRC) looking at the state-of-the-art on MNT devices and packaging – and especially at looking at prospects for devices based on Micro-Electro-Mechanical Systems (MEMS).

The original study plan was that the outputs of the preparatory phase would be used to develop a set of spacecraft and instrument solutions. These in turn would be subject to a mission analysis to derive a final set of proposals for nanosat missions that could monitor key space weather parameters. This conventional "waterfall" approach to mission studies did not work. An initial set of ideas for spacecraft and instruments solutions was first developed in an intensive co-location meeting attended by all team members, and by ESA. at which ideas for spacecraft and instrument solutions were developed. These solutions were consolidated by email discussion following the co-location meeting. This consolidation of the instrument solutions raised many ideas that necessitated refinement of the requirements, e.g.:

- which requirements could not be satisfied on a nanosat because of fundamental physical constraints on sensor size (e.g. optical system sizes are heavily driven by requirements on angular resolution)?
- where could instrument concepts be simplified to fit on a nanosat but still address a key part of an existing requirement?

Similar issues arose during the mission analysis. Thus there was a frequent need for feedback between requirements and design as indicated by the red lines in Figure 1. These feedback loops violate naïve ideas about the relationship between requirements and design, i.e. the waterfall model for system development, but are well-recognised as a normal and realistic technical approach. The study was therefore strongly driven to adopt a different approach to that given in the SoW, but the results are much more robust for that reason.

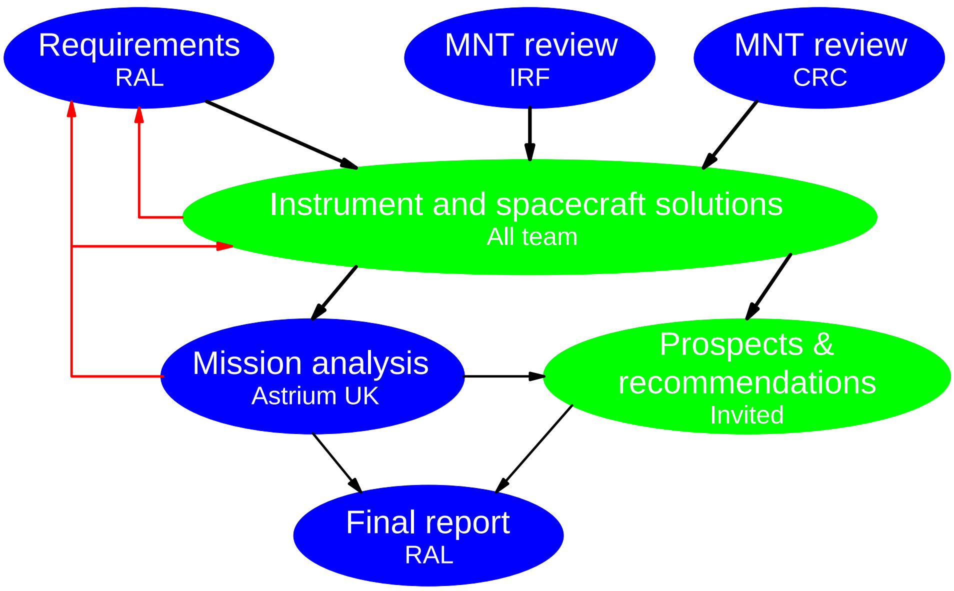


Figure 1. Logical flow of the space weather nanosat study. The red lines indicate the feedback loops that had to be developed during the course of the study.

The following sections describe the outputs of the study. Sections 3 to 5 present a range of background material. Section 3 presents the two MNT reviews from the preparatory phase: first that by IRF drawing on their practical experience with nanosats and then that by CRC looking at the state-of-the-art on MNT devices. Section 4 presents a wider review of current and planned nanosat projects with the aim of identifying key trends in this area of technology. This review was originally developed as part of the spacecraft solutions, but is included here as it logically forms part of the background material. Section 5 presents the requirements analysis. This starts with the existing requirements for space weather monitoring as developed in several earlier ESA studies. These are consolidated into a single requirements set and reviewed to identify any updates needed (e.g. by considering the ESA Pilot Project on space weather applications). The requirements set is then examined in detail to identify requirements that can be addressed using nanosats. This is the major area of refinement due to feedback from the design work and that refinement is discussed in some detail in this section. The refined requirements are then assessed to find ways to divide them into a small number of classes. We first considered classification by the three solution levels proposed in the SoW (see Table 12) but found that was not useful as most requirements fell into the first solution level. A much more useful approach was to classify the requirements in terms of the orbits and numbers of spacecraft required. This allowed us to identify five spacecraft constellations that can cover all the requirements. These constellations are listed in Table 13 and were used to order the rest of the study.

Section 6 presents the instrument solutions. It describes the twelve different instrument types needed to address the measurement requirements. It provides information on instrument characteristics (e.g. power, mass, dimensions) anticipated for contemporary and future instances of these instruments. These are followed by a discussion of various system level issues that emerged during the development of the instrument solutions: the evolution of electronics used in instruments, the potential for sharing electronics between instrument and the ways to operate instruments autonomously without need for regular uplink of telecommands.

Sections 7 to 13 then describe the overall system solutions. Section 7 provides background on how these were developed, whilst sections 8 to 13 present the detailed solutions. There is one section for each constellation, except that the low Earth orbit (LEO) constellation comes in two flavours and thus requires two sections. Section 7 outlines the methodology used to develop the system solutions and provides some general discussion of key issues including de-orbit at end of life, replacement strategy and communications. Sections 8 and 9 then cover the two flavours of the LEO constellation – first a flavour focused only on ionospheric measurements and second a flavour including solar observations. Sections 10 to 13 then cover constellations in Molniya, GTO, SWARM and L1 orbits.

Section 14 takes a very different tack. It discussed the prospects for using nanosat in space weather monitoring and provides a series of recommendations on actions needed to make this happen. This ideas in this section were developed at a workshop held during the study and attended by a number of external experts as well as the study team and ESA. They were then consolidated into a report and reviewed by participants. This section is the final version of that report.

The main part of the report finishes with Section 15 which gives a short discussion of the key results from the study and some suggestions for future work. It is followed by a series of annexes presenting ancillary details from the requirements analysis, the MNT review and the prospects and recommendations workshop. These also include two supplementary analyses – one establishing the number of spacecraft needed to monitor the radiation belt from GTO and one looking at the use of spacecraft magnetometer data to generate geomagnetic indices.

4 MNT REVIEW

4.1 Nanosat experience

IRF has actual flight experience with a nanosat that made space weather measurements. This was Munin [R10] a 6 kg spacecraft with dimensions 21 x 21 x 22 cm as shown in Figure 2 below. The match box in right foreground shows the scale.

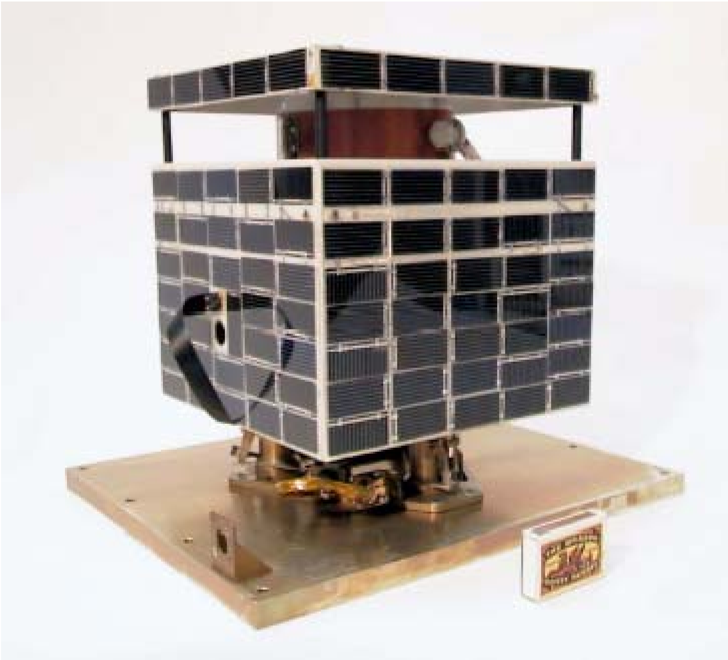


Figure 2. Munin nanosat in flight configuration with the TM antenna folded/bent on left.

Munin was launched into a polar orbit (700 x 1800 km) in November 2000 and operated for 2.5 months before an on-board problem prevented uplink of commands. Munin was an experiment to build a nanosat using existing technologies; it carried instruments to observe space weather effects such as energetic particles and the aurora. These are shown in Table 1 below.

Table 1. Munin instrument characteristics

	Meas. range	Field of view	Time res. (s)	Look dir.	Mass (g)	Power (mW)
MEDUSA, ions and electrons	2 eV - 15 keV	10° x 360°	0.25	0°-180°	588	1000
DINA, ions and neutrals	20 - 2400 keV	5° x 30°,	0.25	0°& 90°	900	500
HiSCC, visible imager	320 x 249 pixels	50°	30-60	0°	100	300

The spacecraft was powered by solar cells with 6W output and carried 4200 mAh capacity battery. The power consumption was 3 W nominal, rising to a peak of 9 W when transmitting to the ground station. The latter was based at Kiruna in Sweden and thus only used for the small parts of orbits when the spacecraft was visible. The radio link used frequencies in the UHF band at 400 to 450 MHz, thus allowing use of simple antennae on the spacecraft and the ground – and adaptation of an off-shelf transceiver (200g, 6W) for use on the spacecraft. This supported data rates up to 22 kbps.

The spacecraft was stabilised by inclusion of a permanent magnet which maintained orientation with respect to the geomagnetic field. This approach proved very successful and was entirely acceptable given the payload shown above. It would not, of course, be acceptable for a mission to make sensitive magnetic measurements. This would require a magnetically clean spacecraft or at least one whose fields were well-characterised.

Munin successfully made energetic particle measurements over its 2.5 month lifetime and returned much useful data to the ground. It showed that a simple space weather nanosat can be built today using existing technologies and the adoption of simple strategies (e.g. UHF radio link). But a long lifetime should not be expected due to limited availability of radiation-hardened components. One way to reduce the impact of radiation exposure, and a key lesson leaned from Munin, is the value of on-board autonomy. It would be very useful to eliminate the need for uplink of commands for routine operations. It was the failure of this aspect which brought Munin operations to a premature end.

4.2 MNT state-of-the-art

A wide range of Micro and Nano Technology (MNT) devices are now being developed by industry for use in future military and civilian technologies. Many of these may be adapted for use in space though it should be recognised that the key drivers for development of the basic MNT technology are often mass market applications such as cars and mobile phones. In this section we outline the state of the art in key areas that may be useful for space weather nanosats.

Within this outline we distinguish between (a) MNT, which is a generic term for the technology of miniaturisation, and (b) Micro-Electro-Mechanical Systems (MEMS), which is a specific technology for building miniature mechanical devices. MEMS is only a part (but a key part) of MNT.

4.2.1 RF devices

MEMS enables the miniaturisation of good-quality mechanical components, including filters, resonators, and switches, that are central to the performance of radio systems (by providing high quality factors in tuned circuits). This mechanical miniaturisation of passive devices complements the familiar miniaturisation of active devices via integrated circuits, and gives much better RF performance than can be achieved using standard semiconductor technologies to create passive devices such as resistors and capacitors. A combination of these two technologies has the potential to deliver highly miniaturised RF systems. For example, Figure 3 shows examples of MEMS devices that can be mounted on top of integrated circuits to deliver a fully integrated RF devices.

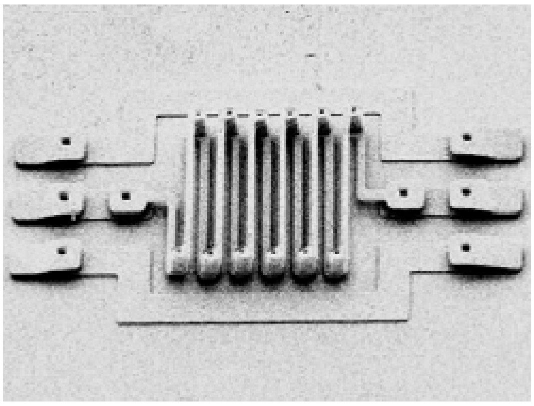


Figure 3. Examples of MEMS low loss inductors: left:Yoon, Chen, Allen & Laskar, Georgia Tech, right: "Above IC technology", MEMSCAPTransducers 2001

The demand for such systems is heavily driven by the commercial interest in radio applications such as mobile phones, wireless devices and GPS/Galileo receivers. That interest is currently focussed on frequencies below 6 GHz and thus MNT developments focus on solutions appropriate to those frequencies, e.g. use of mechanical resonators. Those solutions degrade at frequencies above 10 GHz, so a different approach will be needed, e.g. use of cavity or dielectric resonators.

An area of particular interest for nanosats, indeed for many satellite systems, is the development of MEMS-based phased-array antennae, i.e. electronically steerable high-gain antennae. In the long-term these could prove important because of the critical need for real-time data downlink to support space weather applications. A phased-array antenna could allow a nanosat to have a steerable downlink- thus improving the link budget available for a given power without the risks implicit when flying in a mechanically steered system in space.

A phased-array antenna comprises an array of many small antennae electrically connected so that the signal phase on each antenna can adjusted individually. The directional sensitivity of the overall system (i.e. the direction of the transmitting or receiving beam) is then set by the form of the array and the phase shift on each antennae. By varying those phase shifts the beam can be electronically steered. MEMS technology is enabling the development of phase-shifters with lower costs and power consumption than the expensive high-power devices used in current systems. Thus MEMS has the potential to enable much more extensive use of phased-array antennae than has previously been the case (phased-array usage is currently focussed on top-end applications such as military and major research radars). Nanosat applications will particularly benefit from the availability of low costs and low power solutions.

4.2.2 Attitude and orbit determination and control.

A wide range of MNT developments are relevant to this area, including both detectors to monitor spacecraft dynamics and devices to control those dynamics. The developments in detectors mainly focus on attitude determination as discussed below. But note that the development of miniaturised GNSS receivers is relevant to orbit determination for nanosats with some or all of their orbits inside the GPS and Galileo constellations.

1. **Optical detectors**. Spacecraft attitude has long been monitored using star mappers – devices that detect the direction of the Sun and bright stars with respect to spacecraft coordinates. Since the inertial directions of the Sun and stars are known independently, this measurement can be inverted to derive the spacecraft attitude with respect to inertial coordinates. The optical detectors are MNT-based – currently CCDs but with an ongoing trend to replace by use of Active Pixel Sensors based on CMOS technology, including radiation-hardened control and readout systems. CMOS APS has many advantages over CCDs including lower cost and power and simpler power requirements (single 3V supply). Commercial solutions for flight are already available in Europe; suppliers include Fill Factory and IMEC.

2. **Magnetometers**. These can also be used to determine spacecraft attitude (but also subject of significant commercial interest for use as industrial sensors and in magnetic recording devices). In the spacecraft application they measure the vector magnetic field in spacecraft coordinates. This is then compared with an estimate of the same vector in inertial coordinates (derived from geomagnetic field models) in order to drive the spacecraft attitude. Two different sensor types are used in such magnetometers: (a) fluxgate devices similar to those used in science-grade instruments, and (b) magneto-resistive devices whose electrical resistance changes with the magnetic field. There are several different types of magneto-resistive devices: anisotropic magneto-resistive sensors (AMR), giant magneto-resistive sensors (GMR) and tunnelling magneto-resistive sensors (TMR). MNT solutions for AMR are commercially available, e.g. Honeywell HMC1021 and Phillips KMZ10A, but the other technologies are still in the research domain. An MNT fluxgate has been demonstrated using small (4 x 7 mm) magnetic cores plus a CMOS ASIC. GMR and TMR devices have much potential for MNT magnetometers as they are expected to reduce costs while improving sensitivity. Table 2 below summarises the prospect of these different technologies.

Table 2. MNT magnetometer overview

Sensor Type	Sensitivity/ Field Range	Frequency Range	Min. Size/ Scalability	Costs	Advantages	Disadvantages
Fluxgate	1 pT/sqrt(Hz) @ 1 Hz	< 1kHz	4x7mm² S/N scaling down	Moderate	High sensitivity	Costs, Size, Energy Cons.
AMR	50-100pT	0-5GHz	< 1µm	Moderate	Lower 1/f noise	
GMR	20nT	0-5GHz	< 1µm	Cheap	Low cost in large quantities	
TMR	1nT	0-1GHz	< 1µm	Cheap	Large MR, Low cost in large quantities	High 1/f noise, Hysteretic

3. **Angular rate sensors**. This is an area of major commercial development driven by the needs of the automotive market. MNT development has focused on development of MEMS-based sensors. These are based on the principle of a tuning fork using MEMS-based mechanical oscillators (see Figure 4). During angular motion, the Coriolis force can transfer energy from a primary (driven) oscillator to a secondary oscillator used as a sensor. The sensor signal gives a measure of the rate of angular motion.

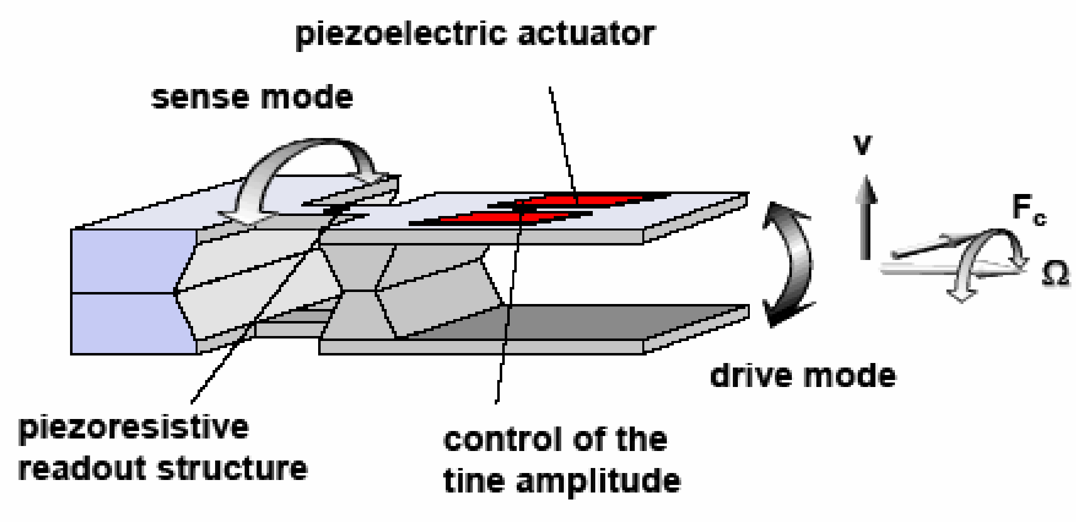


Figure 4.Working principle of tuning fork angular rate sensor

There is wide European interest in developing such sensors with at least 14 groups working in the area – mostly commercial R&D laboratories but also some public sector institutes. These developments cover a wide range of market solutions ranging from low end consumer products through automotive applications to high end products such as navigational devices. The high end navigational devices have much interest for aerospace applications including nanosats. A number of studies have been, or are being, undertaken with funding from ESA and national space agencies. Adaptation and qualification for the space environment is, of course, an important objective.

MNT devices also have much potential to provide new solutions for control of spacecraft dynamics.

1. **Momentum wheels.** These are widely used to stabilise and control spacecraft, especially where three-axis stabilisation is required. They are also known as reaction wheels. A momentum wheel comprises an electro-motor and a spinning inertial mass can be used to generate momentum that can slew a spacecraft (i.e. controlling the pointing) or stabilise it (i.e. removing uncontrolled spin or tumbling motions). Conventional solutions are rather bulky and expensive but MNT solutions are now appearing that will reduce size and cost – both attractive steps for use of such devices on nanosats. Commercial players include Micro Motors and Penny Motors. As an indication of MNT capability, we cite the Micro Motor solution: a motor of 1.9 mm diameter by 5.5 mm long can operate at up to 120000 rpm and give a maximum torque of 140 µNm.

2. **Propulsion systems**. All spacecraft need propulsion systems for deployment in, and maintenance of, the correct orbit. There is also a growing demand for a de-orbit capability. Thus the exploitation of nanosat applications will require the development of propulsion systems appropriate to the capabilities of nanosats (i.e. small, low mass, low thrust). The development of MNT propulsion systems is very much in the research domain. Solutions under study include MEMS technologies such as micro-jet engines and microthrusters. The former are miniaturised gas turbines built on a silicon chip using MEMS techniques; they have thrust-to-weight ratios similar to those of conventional macro-scale gas turbines. Microthrusters are highly miniaturised rocket engines built on silicon using MEMS techniques. Both solid and liquid fuel versions are in development. The solid fuel version can be built as an array containing several hundred individual thrusters, each of which can be fired independently. Thus one can combine individual firing to create continuous operation. See Figure 5 for an example.

Figure 5. Left: 3x5 microthruster array 'Rocket Chip'. Right: Firing a microthruster (C. Rossi, LAAS-CNRS)

4.2.3 Power systems.

Power is expected to be a critical issue for space weather nanosats because the area of solar cells will be limited by the small spacecraft dimensions and the need for sensor apertures looking out into space. Thus MNT-based developments in power generation could be important as discussed below.

1. **New solar cell technologies**. Since the power flux of sunlight is set by the distance from the Sun, technology development must focus on raising the power conversion efficiency of solar cells. One approach is to improve efficiency of conventional crystalline cells by developing multi-material solutions. This overcomes the limit of 31% on efficiency that bandgaps impose on any single semiconductor. A number of solutions are in development with potential efficiencies of 35 to 50%. The other approach is a major new development, namely organic solar cells. This technology is in its infancy but has the potential to transform solar cell technology by reducing costs and thus stimulating mass market applications (e.g. power for wearable electronics). The conversion efficiency of organic cells is low at present (up to 10%) but technology forecasts suggest that much higher efficiencies will be possible in future (30 to 60%), e.g. by exploiting new nano-materials such as quantum dots.

2. **Thermo-generators**. There is much interest in developing MNT devices that can exploit temperature gradients to generate electrical power. This could be attractive on a spacecraft as such gradients arise naturally due to the thermal balance between solar heating on the sunward side of the spacecraft and radiative cooling on all sides. Solutions include devices based on the thermoelectric (Seebeck) effect and devices where thermally-induced mechanical oscillations are converted into electrical energy. Infineon have demonstrated a device that can generate > 5.5 mW cm-2 from a temperature difference of less than 10 K (see Figure 6).

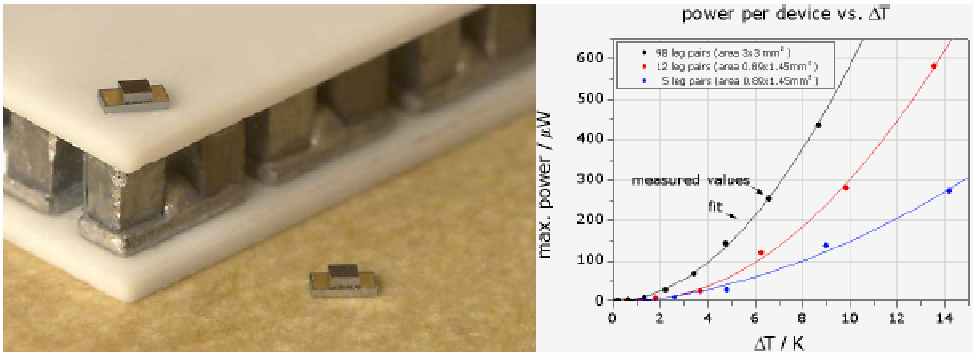


Figure 6. 5x5 mm² next gereration thermoelectric generator / peltier cooler developed by Infineon (source: www.micropelt.de)

3. **Micro combustors**. This involves the combination of propellants (fuel + oxidiser) to generate thermal energy and its conversion to electrical energy. Some demonstration systems have been built. This solution requires the development of systems to handle the propellants. Its lifetime is limited by the mass of propellants carried on the spacecraft.

4. **Nuclear batteries**. These are devices in which energy from radioactive decay is converted to electrical power. It has great potential because of the high density of energy associated with radioactive decay (see Table 3). Interest has focused on use of beta emitters such as Nickel-62/Tritium and Americium-241. These sources can generate electric energy via: (a) generation of electron-hole pairs in a semiconductor junction exposed to the beta particles, and (b) generation of mechanical oscillation in a MEMS structure through a cycle of electrical charging and discharging driven by the beta particles. This approach is different to the radioisotope thermoelectric generators used on deep space missions; those exploit the thermal energy arising from decay of Plutonium-238. The use of beta emitters is much safer – the beta particles can be shielded by 25 to 100 µm of plastic and even by human skin. In contrast Plutonium-238 generates high-energy gamma rays that are dangerous to humans. Thus the nuclear battery has potential as a mass market application.

Table 3. Comparison of energy sources.

Battery Type	Energy Density [kJ/g]	Lifetime	Cell Voltage
Li-Ion Battery	0.5	5 years (shelf)	3-5 V
Micro Fuel-Cell	20.16	1 day (w.o. refuel)	0.6-1 V
Solar Cell	5	> 10 years	~1 V
Micro Combustion	44	1 hour (w.o. refuel)	< 10 V
Nickel-63	18000	100 years (half-life)	< 1000 V

5. **Fuel cells**. This is an area of intense research as fuel cells are seen as an important mobile power source for the 21st century. There is strong market demand for power for mobile devices such as laptops, PDAs and mobile phones – and many players, in commercial R&D and in research institutes are working on advanced solutions. MNT is considered to be a key factor in decreasing the size of fuel cells and increasing their efficiency. Table 4 gives an overview of the different types of fuel cells now in development. The membrane fuel cells (PEMFC and DMFC) seem to best for use in autonomous devices with power requirements < 1kW – because of their operation at normal temperatures and avoidance of corrosive materials.

Table 4. Overview of fuel cell types

Type of Fuel Cell (FC)	Fuel/Oxidant	Temperature	Electrolyte	Electrical Efficiency of Cell	Power Range
Alkali FC	Pure H2	70 – 85°C	30% potash	60 – 70%	1 – 50 kW
Membrane FC (PEMFC)	H2/O2, air	0 – 100°C	Proton conducting membrane	50 – 70%	< 1W – 500kW
Direct Methanol FC (DMFC)	Methanol/O2, air	30 – 100°C	Proton conducting membrane	30 – 40%	< 1W – 500kW
Phosphoric acid (PAFC)	Natural gas, bio gas, H2/O2, air	160 – 220°C	Concentrated phosphoric acid	55%	<11MW
Melted Carbonate (MCFC)	Natural gas, bio gas, H2/O2, air	620 –660°C	Molten salt from alkali carbonate	65%	300 – 500kW
Sulfuric Acid (SOFC)	Natural gas, bio gas, H2/O2, air	800 – 1000°C	Yttrium doped zircon oxide	60 – 65%	2 – 100kW

However, all use of fuel cells is constrained by the need to remove waste heat and reaction products (e.g. CO_2 for DMFC systems) and by the need to refuel the cell at periodic intervals. These are serious limitations on space applications. However, if the market develops as expected so that small units are widely available at low cost, these could prove very effective as power sources for short-lived (a few months) nanosats where the need to refuel and remove waste products is not an issue.

4.2.4 Packaging and new materials

Thus MNT devices have great potential to support development of space weather nanosats. But the successful inclusion of these devices on spacecraft will require development of robust techniques for high-density integration, e.g. using standardised interfaces and 3D-stacking technologies to allow integration of sub-components to build "systems in a package" (SIP). See Figure 7 below for an example. Several different types of material are now in use or planned for use in packaging of MNT devices. These include ceramics (such as alumina, Al_2O_3, and low temperature co-fired ceramics, LTCC), polymers (such as Kapton™ and Liquid Crystalline Polymer) and carbon nano-tubes.

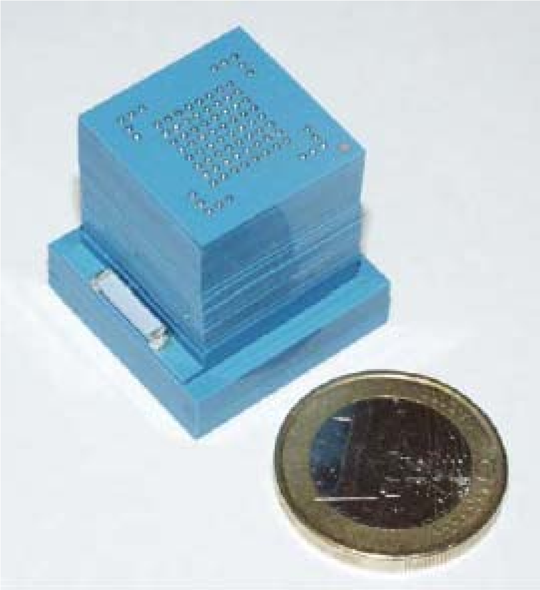


Figure 7. An example of MNT packaging. An EADS micropack containing atmospheric and inertial sensors, power supply, data handling unit and RF transceiver. A 1 Euro coin is shown for size comparison.

Carbon nano-tubes can also be applied more widely. They consider of a hexagonal rolled grid of graphite with a diameter of order of nanometres and lengths up to several millimetres. They have very useful properties – both mechanically (yield strength 6 times higher than steel, as hard as diamond but low density around 1.4 g cm^{-3}) and electrically (low electrical resistance so can support high current densities up to 10^9 A cm^{-2}) . It is also possible give them semiconducting properties and thus use them to build transistors. Several industrial players (e.g. Infineon, Nantero, BAE Systems) are developing electronic devices based on carbon nanotubes. The development of this area should be monitored to assess its potential for nanosats applications. The high strength of carbon nanotubes also gives them much potential as a structural material, including stronger and lighter structures for spacecraft.

4.2.5 Future and challenges for MNT

The likely evolution of the MNT devices discussed above has been summarised in a roadmap which gives their status at the time of the study (2005) and predicted status in 2010 and 2015. This is presented in Table 53 in Annex G.

However, in order to exploit this evolution and successfully apply MNT devices in spacecraft applications, a number of critical issues must be addressed. These include
a) system design and integration, where the small size presents a challenge for thermal management and electromagnetic compatibility
b) packaging as already discussed above
c) the development of efficient procedures to qualify and validate these new technologies.

5 Nanosat Review

As part of this study we have reviewed current work on small spacecraft including recent flight experience, missions in development and advanced concepts for future missions. We cover a range of spacecraft sizes focusing on the conventional nanosat range of 1 to 10 kg, but also looking at ideas for spacecraft below and above this mass range. Our overall aim is to gain insights into the types of mission and platform technology that are being considered, and the problems/issues that are associated in mission and spacecraft design. In particular, we look at missions that will demonstrate new technologies with the potential to revolutionise nanosat design. In the medium-term (5 years) these include extensive use of MEMS, multi-functional structures, formation flying and solar sailing), while in the longer-term (2025+) they include ideas such as a spacecraft on a chip and massive co-operative spacecraft swarms. We also emphasise nanosat missions that include an element of space weather monitoring.

5.1 Nanosat classification

The classification of the spacecraft by mass is important in determining applicability of MNT in various spacecraft system designs. Table 5 below shows a widely-used classification scheme. The top four categories are used by the US Air Force Research Laboratory (AFRL) and in Europe. The femtosatellite class is now emerging in order to cover the concept of a spacecraft on-a-chip.

Table 5. Spacecraft classification

Mass range	*Class*
100-1000 kg	minisat
10-100 kg	microsat
1-10 kg	nanosat
0.1- 1 kg	picosat
<0.1 kg	femtosat

5.2 Why use nanosats?

Over the fifty years since the start of the space age, there has been a trend towards larger and larger spacecraft in order to fly payloads that can meet rising user requirements. However, there is now a trend in the opposite direction, i.e. to distribute space mission capability over multiple coordinated spacecraft. There are several drivers for this new trend. The obvious one is to spread risk, e.g. mission risk may be reduced by off-loading scientific instruments from a single large spacecraft onto a fleet of micro- or nano-class vehicles. Loss of a single “component” may not jeopardize the whole mission; furthermore mission scenarios may be envisioned where some members of the fleet are purposely sacrificed or exposed to a higher risk for addressing particularly dangerous portions of a mission, such as orbits more susceptible to radiation damage. In some cases distributed missions may also provide improved technical performance. An example is formation-flying, which has the potential to allow mission components to be widely separated (e.g. optical components in a long train) without the need for massive structures to maintain that separation.

Over the past 20 years, micro and nanosat activities have increased due to greater availability of cheap launch opportunities with the introduction of Ariane Structure for Auxiliary Payloads (ASAP) and the Russian launchers. Technical advances have also aided this trend: (a) advances in electronics have reduced mass and increased capability; (b) the advent of microelectromechanical systems (MEMS) and the rapid evolution of micromachining technologies is raising much attention as a way to improve the capabilities of “next-generation” nanosats.

The future vision is one of swarms of micro and nanosats circling Earth, the Sun or other planets in the Solar System, performing critical and highly complex tasks, interacting with each other, with ground or space operators, and with a significant degree of autonomous control capability. Advanced micro and nanosat with distributed functionality are envisioned to take the place of more massive and expensive conventional spacecraft, with the additional advantage of increased survivability and flexibility. In this scenario, very small satellites (<30 kg), alone or in constellations, have gained a strong momentum and have been proposed for a number of missions in both the commercial and scientific domains,.

There is already considerable work around the world to reduce the mass of space weather type instruments such as magnetometers and particle analysers. Lightweight instruments are required for future science missions, which are overwhelmingly focused on multi-spacecraft solutions (e.g. NASA's magnetospheric constellation and the CrossScale proposal to the ESA Cosmic Vision programme). Within this study we go one step further by suggesting that solutions with significant or even entirely based on MNT are sought after, if possible. If this can be achieved nanosat will be ideally suited to act as space weather beacons.

5.3 Key trends from recent micro- and nano-sat projects

We reviewed some 36 micro and nanosat projects launched between 1998 and January 2005 (see Table 6 below for list). This revealed several key features that will have a major impact on the development and the qualification of new systems, especially those specifically intended for micro and nanosat applications. These are as follows

- There is a well-defined and continuous trend in mass reduction: the wet masses of earlier designs range around 50 kg or more, while newer designs are pushing toward even lower masses, into the 1 kg range or below.
- Current micro and nanosat designs have very significant power limitations: several of the earlier designs deliver less than 1 W/kg of spacecraft mass, newer designs aim at increasing this by factor 2 or 3, and although the situation may change in the future it seems that this order of magnitude will not be exceeded in the near or medium term.
- There is a trend is toward a decrease of micro and nanosat bus voltages, that will likely be less than the standard 28 V. The extensive application of microelectronics on board of the vehicles is leading to the development of 5 and 3.3 V standards for future bus. This will result in lower mass power conditioning equipment, which will benefit missions that are constrained by the maximum allowed mass.

Table 6. Recent projects considered in this analysis

3 Corner-Sat	DARPA Picosat 1A or 1B	Naxing 1	SaudiSat 1C
AAU Cubesat	DARPA Picosat 7 and 8 (aka MEPSI-2)	OPAL	SNAP-1
Amsat-Echo/ Amsat Oscar-E	DTUsat	PCSAT	StenSat
Artemis Picosat	Kolibri-2000	Quakesat	Tatiana, or Universitesky
Astrid 2	Latinsat-A, -B, -C and D (C and D also known as Aprizesat 1 and 2)	REFLECTOR	Tubsat N
ASUSAT-1	MASat-1	Sapphire	Tubsat N1
CANX 1	MEPSI-3 (2 picosats)	Saudicomsat 1	Unisat
Cubesat XI-IV	MUNIN	Saudicomsat 2	Unisat 2
CUTE 1	Nanosat1	Saudisat 1A &1B	Unisat 3

We also found that there is a trend to developing advanced technology for use on nanosats. The first generation of cubesats exemplifies this; they show an increasing trend towards using new/advanced technologies in the form of small experiments that would otherwise be cost-prohibitive to flight validate. This includes technologies such as MEMS, micro-reaction wheels and micro-electrodynamic tethers – see Table 7 below for more details.

Table 7. Advanced technologies flown on recent nanosats

Technology Area	Mission	Description
MEMS	Sapphire	MEMS IR detectors
	DARPA picosats 1A and 1B	MEMS RF Switches
	DARPA picosats MEPSI-3*	MEMS Pressure tranducers
	DTUsat*	MEMS Sun sensor
Picosat Launcher	Sapphire	First spacecraft to demonstrate this
Electrodynamic tether for de-orbit	DTUsat*	
Solar battery paddle	Cute 1*	For enhanced power
GPS-based position determination	Can-X1*	
CMOS horizon sensor and star-tracker	Can-X1*	low-cost
Active 3-axis magnetic stabilization	Can-X1*	
Small Cameras	AAU Cubesat	100-meter-resolution Earth imaging camera
	DTUsat*	small CCD camera to collect greyscale images of the earth
Small propulsion systems	DARPA picosats MEPSI-3*	cold gas propulsion system, with 0.1N thrust with 5 thrusters

* indicates a cubesat

5.4 Key trends from new micro- and nanosat projects

We reviewed some 34 small micro and nanosat projects that are in flight development in order to identify state-of-the-art technologies that can be applied to future space weather nanosats. These projects are show in Table 8 below (some have flown since the report was prepared). These missions address a wide range of objectives including (a) validation of new spacecraft systems (e.g. comms, attitude and orbit control, sun sensors, etc), (b) scientific observations (e.g. for astrophysics, space environment, earth observation and geophysics) and (c) educational activities for future space scientists and engineers. A ll full list of the known mission objectives is given in Table 54 in Annex J.

Table 8. New flight projects considered in this analysis

Aerospace Corp. Cubesat	Compass one	Katysat	Project Cubesat
AASUSAT-II	Cube-II PRISM	KUTESat Pathfinder	QuakeSat*
Arizona cubesats/RINCON 1	Cute1.7	Mea Huaka'i	SACRED (or Alcatelsat)
ATMOCUBE	Delfi-C3	MEROPE	SEEDS
Bluesat	Hausat 1	Munin*	Space Technology 5 (ST5)
Cal Poly Picosatellite Project (PolySat) 1 and 2	ICECube 1 & 2	Ncube 1	UWE-1
Can-X or BRITE	ION	NCube-2	XI-V
Can-X2	IRECIN	PACE	YAMSAT-1A

* we include two spacecraft (Munin and Quakesat) that were discussed in section 5.3, but are exceptionally of interest as state-of-the-art nanosats due to their space-weather-relevant instrumentation.

The results of our review are summarised in Table 9 below. This lists a number of generic technology areas, the missions that are using each technology and additional notes to expand on the particular technologies demonstrated on each mission.

Table 9. Advanced technologies in use on forthcoming nanosat projects

Technology Area	**Mission**	**Description**
MEMS	PACE*	MEMS temperature sensors and course sun-sensors
	YAMSAT-1A*	MEMS spectrometer to measure the sunlight scattering spectrum from the atmosphere
Spacecraft deployment mechanism,	ST5	Assumed to be the deployment mechanism from launcher and other spacecraft
Electrodynamic tether for de-orbit	Cute 1.7*	Tether satellite disposal system.
Power	Delfi-C3*	Test-bed for thin film solar cells
	Hausat 1*	experimental solar panel deployment mechanism and Li-ion battery cells
	XI-V*	demonstration of newly developed CIGS (Cu(In,Ga)Se2) solar cells in space
Communications	Delfi-C3*	wireless on-board communication
	ST5	X-Band Transponder
AOCS	AASUSAT-II*	Active AOCS stabilization to detumble and actively control the satellite utilizing coils and momentum wheels.
	CP2v	three-axis attitude determination and control
	CP1*	Low cost sun sensor and experimental magnetorquer
	Can-X2*	Nanosat-sized reaction/momentum wheel for momentum bias three-axis stabilized attitude control
	Can-X2*	Custom-designed attitude determination system using a suite of coarse and fine sun sensors and a three-axis magnetometer
	Cute 1.7*	Demonstrate various attitude control algorithms, such as three-axis stabilization, detumbling, and spin-up, with three magnetic torquers placed orthogonal to each other. AOCS is three-axis gyrosensor, a three-axis magnetometer, a sun sensor and an earth sensor.
	ION*	demonstrate the use of an active magnetic attitude system.
	PACE*	momentum wheel, magnetic coils and sensors such as a three axis gyro, three axis magnetometer and course sun sensors. Existing cubesats do not employ 3 axis stabilisation due to power, mass and computation constraints.
	SEEDS*	test a 3 axis geomagnetic sensor and 3 axis gyros to measure satellite orientation
	ST5	Miniature magnetometer, Miniature spinning sun sensor, Magnetometer, deployment boom Nutation Damper,

Technology Area	Mission	Description
Small Cameras	Compass one*	miniaturized CMOS camera
	ION*	CMOS imaging system for star tracking and earth photography
	Katysat*	optical (camera) sensors.
	KUTESat Pathfinder*	onboard camera
Small propulsion systems	Can-X2*	liquid-fuelled cold gas system, using sulphur hexafluoride (SF6) as a propellant.
	ION*	test a lowthrust, electric propulsion system
Miniature GPS	ATMOCUBE*	
	Can-X2*	dual-band GPS receiver
	ICECube 1 & 2*	investigate ionospheric scintillations using GPS signals
Science Instruments	AASUSAT-II*	Gamma ray detector
	Can-X2*	atmospheric imager and spectrometer
	ATMOCUBE*	Miniature spectral dosimeter
	ATMOCUBE*	Miniature magnetometer
	Can-X3*	seven centimetre optical telescope
	Cube-II (PRISM)*	telescope with 10m-class resolution at drastically low cost
	Cute 1.7*	Monitor low energy charged particles under 30keV, with a very small, low power, and high sensitive sensor based on Avalanche Photo Diodes (APDs)
	IRECIN	Debris Measurement System
	Katysat*	various sensors such as radiation, magnetic field,
	KUTESat Pathfinder*	measure the radiation in LEO
	MEROPE*	measure radiation in the Van Allen belts
	SACRED (or Alcatelsat)*	measure the total amount of high-energy radiation over mission and test four commercial integrated circuit components for their radiation hardness, functionality and annealing properties.
Ground Segment	IRECIN	real time ground segment architecture
	ST5	Autonomous Constellation Management / “Lights Out” Operations

5.5 Key trends in developing concepts for micro- and nano-sat projects

There is much interest in using small platforms for space weather measurements if instrument size can be driven down. This will be reinforced if more advanced nanosat concepts can be exploited, e.g. using MNT, future nanosats could utilise advanced packaging techniques with substantial modularity (i.e. ‘plug and play’ modules). Advanced integration techniques may also be valuable, e.g. developing instruments that contribute to, and perhaps, exploit the spacecraft structure. As well as MNT, other technology advancements will also play a part in driving future spacecraft concepts. These include autonomy, advanced data compression, quantum computing etc. Another important development is the use of ‘Hive’ or ‘Carrier’ spacecraft to facilitate the simultaneous launch and deployment of large sets of nanosats.

Table 10. Advanced nanosat concepts considered in this analysis

Aerospace Corporation Glass Satellites	Magnetospsheic Constellation (MagCon)	Palmsat	SWARM
APIES	Mcubed	PETSAT Project	TEST – Thunderstorm Effects in Space: Technology
CanX4 and CanX5	Munin-X	SCOPE	
EADS Micropacks	Mustang 0	SNAP2+/Proba 2.25	
Hausat 2	NanoSpace 1	Solar Kite	

We reviewed some 17 advanced nanosat concepts as shown in Table 10 above These missions address a wide range of objectives including (a) testing of new spacecraft technologies (e.g. structures, formation-flying and highly integrated systems) and (b) scientific observations fort space environment and planetary science. A full list of the known mission objectives is given in Table 55 in Annex J. We identified a number of platform concepts and technologies that could be useful or even critical for a future space weather service that includes use of nanosat beacons. The results of this review are shown in Table 11 below.

Table 11. Advanced technologies and concepts for future nanosats.

Technology Area	**Mission**	**Description**
MEMS	APIES	MEMS Ku-band antenna
	EADS Micropacks	Multi-application highly integrated MEMS system package
	Mustang 0	Demonstrate new technologies such as MST
	NanoSpace 1	extensive use of advanced multifunctional MNT (inc thrusters, thermal switches, GPS)
	Palmsat	A MEMS gyro (SiRRS01) is proposed to be used to achieve the specified pointing requirements
	SNAP2+/Proba 2.25	colloid thrusters
	Solar Kite	Some subsystems and instrumentation
Novel packaging or spacecraft design	Aerospace Corporation Glass Satellites	The tiny glass satellites have the potential to cut the high costs and lengthy production times. They can be mass-produced inexpensively and mass-customized
	EADS Micropacks	Multi-application highly integrated system package
	NanoSpace 1	large multifunctional silicon modules

Technology Area	Mission	Description
	PETSAT	This satellite is constituted by panels, in which each panel has different functionality like CPU, memory, communication, attitude control and other vital subsystems, and can be joined according to mission necessities like Plug-in systems. One of the advantages of this system is the folding capabilities of the panel connector and consequently reduction of the space required in the rocket during the launch
Novel packaging or spacecraft design, continued	Solar Kite	Solar Sail design
	TEST	Novel spacecraft design with new highly modular satellite bus structure and common electrical interface. TEST instrumentation amd satellite subsystems are packaged in modular cubes of 10cm increments (Cubesat3)
Hive Spacecraft	APIES	1 hive
	MagCon	3 hives
	Mcubed	TBD hives
	SCOPE	TBD hives
	SWARM	5 hives
Formation Flying	CanX4 and CanX5*	2 spacecraft demonstration
	Mcubed	3 nested tetrahedral
	Munin-X	The Objective is to develop spacecraft swarm technology for multi-spacecraft missions.
	NanoSpace 1	high precision formation flying demonstration
	SCOPE	The formation consists of one large mother satellite and four small daughter satellites. Three of the four daughter satellites surround the mother satellite 3-dimensionally maintaining the mutual distance that ranges between several km and several thousand km (variable). The fourth daughter satellite stays near the mother satellite with the distance between several km and 100km.
	SNAP2+/Proba 2.25	to investigate and demonstrate some of these formation flying techniques
Constellations	MagCon	30 spacecraft
	Solar Kite	~30 spacecraft
	SWARM	30 spacecraft

Technology Area	Mission	Description
Communications	CanX4 and CanX5*	inter-satellite communications link using radio and/or optical communication systems
AOCS	CanX4 and CanX5*	full complement of three nano-wheels, orthogonally mounted along with three magnetic torquers, providing three axis control.
	EADS Micropacks	MEMS accelerometers
	Hausat 2	Star Tracker
	Palmsat	MEMS gyro (SiRRS01) is proposed
Small propulsion systems	SNAP2+/Proba 2.25	Enhanced propulsion system from SNAP-1
	Palmsat	miniature propulsion and attitude control systems
	MagCon	Small cold gas system
	NanoSpace 1	Micropropulsion experiments, including proportional Cold Gas Micro Thrusters for full three-axis stabilization and high precision formation flying demonstration, and a Micro-kick motor
	CanX4 and CanX5*	Small iquid propulsion system
Miniature GPS	CanX4 and CanX5*	carrier-phase differential GPS
	Hausat 2	Space-borne GPS receiver
	Palmsat	GPS receivers
Science Instruments	Hausat 2	Electric Plasma Probe
	MagCon	Range of solar terrestrial physics instrumentation
	Mcubed (now CrossScale)	Range of solar terrestrial physics instrumentation
	NanoSpace 1	Wave Vector Receiver, pair of Langmuir probes, Magnetometer, Flux Gate Magnetometer, Antennas & Booms
	Palmsat	ionizing particle detectors, magneto-resistive magnetometers, thermal IR micro-bolometer imagers, near UV radiometers and multi-spectral imagers
	SCOPE	Range of solar terrestrial physics instrumentation
	Solar Kite	Range of solar terrestrial physics instrumentation
	SWARM	Range of solar terrestrial physics instrumentation

Technology Area	Mission	Description
	TEST	two 1m electric probes, a thermal plasma density langmuir probe, a VLF Receiver, two solid state detectors for electrons and ions (between 10keV and 1 MeV), a 3 axis magnetometer, a UV photometer, a transient photometer and a 630nm imager for airglow/lightning measurements

5.6 Very advanced concepts for the far-term 2025+

The following concepts have been suggested which could be ideas for the far term:

- credit card sat/CHIPSAT (being studied by Surrey Space Centre)
- smart 'dust' – enhanced/denser packaged micropacks (EADS)
- wafersat (Aerospace Corporation)
- self assembly (being studied by the Advanced Concepts Team at ESA)
- bio-inspired hive behaviour (been reviewed in a current ESA industrial contract by Surrey Space Centre, Astrium Ltd and Bath/Sussex universities, under the direction of the ESA Advanced Concepts Team)
- Extremely Large Swarm Array of Picosats (studied by the NIAC – NASA Institute Of Advanced Concepts)

These concepts continue some of the key trends noted in previous sections including: widespread use of MEMS, novel packaging or spacecraft design, use of hive spacecraft for deploying spacecraft and collecting ‘dead’ spacecraft and intelligent autonomous satellites.

5.7 Summary and latest developments

In this section we have reviewed a range of micro- and nanosat concepts ranging from recent missions, through the state-of-the-art to very advanced concepts looking twenty years ahead. There are a number of key trends that appear throughout this review. An obvious example is the trend to smaller spacecraft where we envision that spacecraft in the true nanosat range (1 to 10 kg) will be widely used for applications where the payload can be miniaturised to appropriate mass and dimensions. As we will see later in this report, payload miniaturisation is now the key step to exploiting nanosat technology for space weather applications.

A significant fraction of the concepts discussed here exploit the CubeSat standard platform (R44). This has gained significant interest in the US, especially in the university sector where the US National Science Foundation recently announced a new programme for regular cubesat launches; this has space weather measurements as a primary goal along with those for atmospheric science. The programme will also provide essential opportunities to train the next generation of experimental space scientists and aerospace engineers..

Interest has also spread to Europe where more than 25 universities are now working on CubeSat projects. The first dedicated European workshop on CubeSats was held at ESTEC in January 2008 to discuss plans for cubesats to be flown on the maiden flight of ESA’s Vega launcher.

6 Requirements

6.1 Overview

The logic of the requirements analysis is shown in Figure 8 below. The study did not have to perform a detailed collection of requirements as extensive work had already been done in previous ESA-funded studies, namely the two space weather programme assessments performed in 2000/2001 by teams led by RAL [R11] and by Alcatel [R12]. The main tasks here have been (a) to synthesis the requirements developed in previous studies, (b) to refine them in terms of their applicability to nanosats (which required iteration with later work on instrument solutions) and then (c) to look for patterns in the final set of requirements. Various patterns were considered and it found that a good approach was to simplify the presentation by grouping the measurement requirements into a small number of spacecraft constellations.

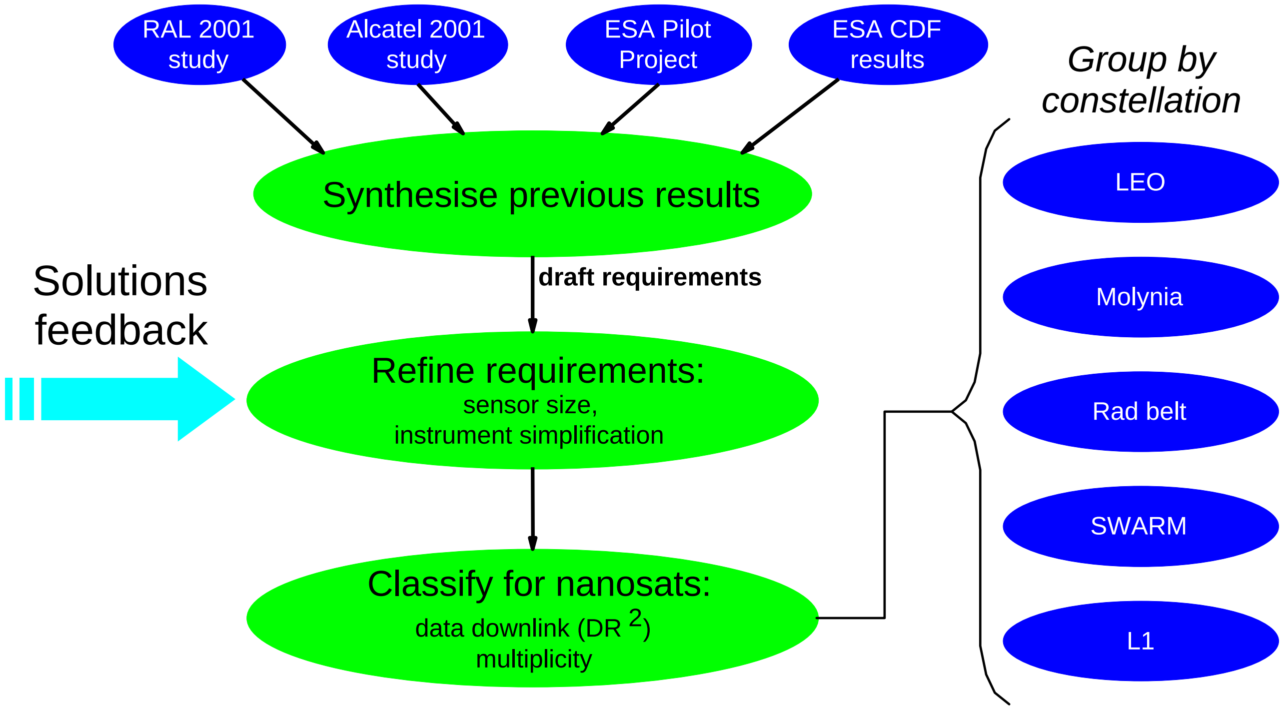


Figure 8.Overview of the requirements analysis

6.2 Synthesis

The synthesis of service requirements was very straightforward as they were part of the common user requirements agreed between the Alcatel and RAL studies. They were extended to include additional requirements identified during the course of the present study through discussions with ESA. They are all summarised in Table 51 below.

In contrast, the synthesis of measurement requirements was a major task because of the different methodologies used by the two teams. As shown in Figure 9 the RAL study produced a set of 74 consolidated system measurement requirements (CSMRs) of which 54 were space-based [R13]. The Alcatel study listed 22 key parameters to be measured in space and provided a description of some 25 possible instrument solutions, which gave further insight into the underlying measurement requirements, e.g. time resolution [R14]. The requirements data from the two studies were loaded into a database and subject to a preliminary analysis. This identified some 110 combinations of CSMRs, key parameters and instruments that were candidates for further assessment. The database was then used to generate a report on each combination. These reports were then subject to a detailed assessment, which gave us some 34 unique requirements.

To verify the completeness of these requirements we compared them with the requirements used in the ESA CDF space weather study [R15], which was carried out in parallel with the RAL and Alcatel studies, and with the requirements emerging from the present Space Weather Applications Pilot Programme [R16]. The CDF study did not yield any new requirements, which is to be expected as its requirements were drawn from the RAL and Alcatel studies. However, the analysis of the Pilot Programme did expose one new requirement – namely the need for monitoring of the solar farside to provide 14-day-ahead predictions of space weather phenomena such as atmospheric drag.

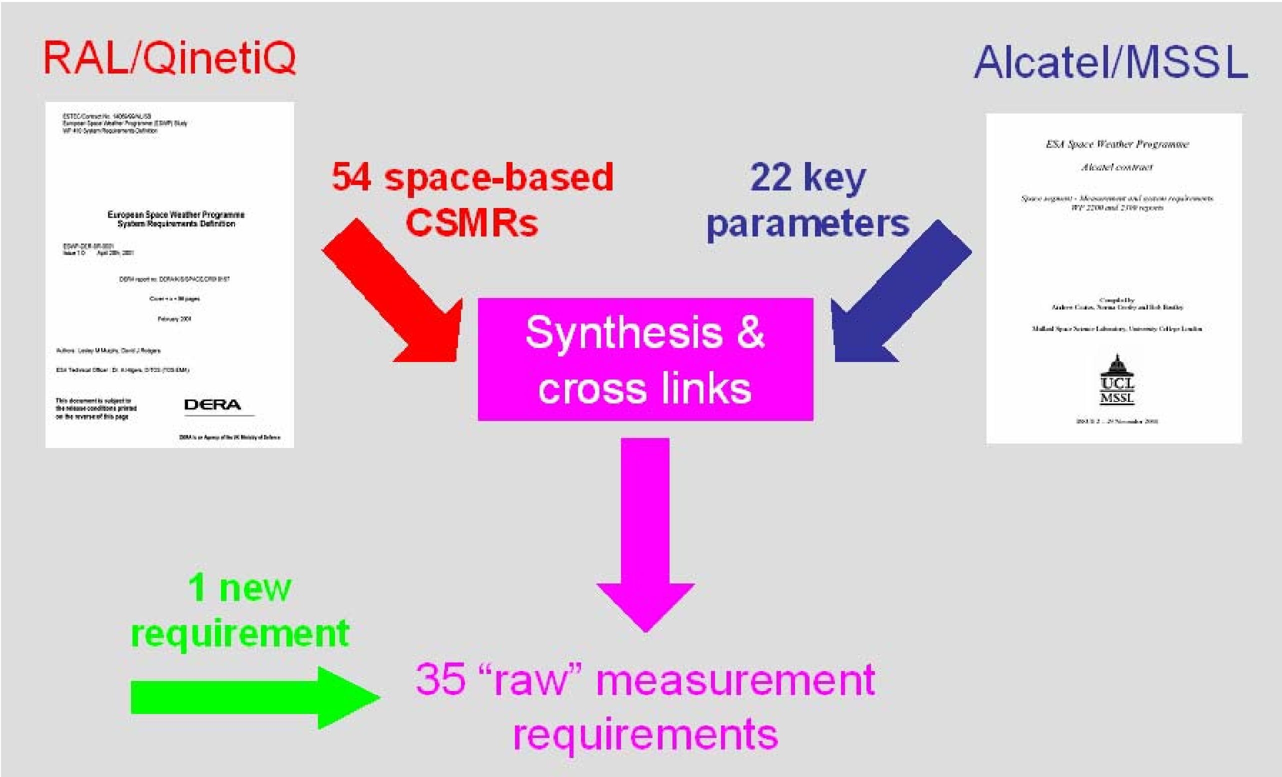


Figure 9. Synthesis of requirements

6.3 Refinement

Thus the initial requirements analysis yielded some 35 raw requirements. These were then subject to a process of refinement, much of which was driven by feedback from the instrument solutions. There are three main elements in the refinement process:

1. The first was to identify cases where the sensors would be too big to be accommodated on a nanosat. This is an important issue. In many cases it is hard to reduce sensor size because of physical limitations on the measurements. A classic example is optical measurements of faint phenomena (e.g. scattered sunlight from coronal mass ejections, airglow emissions in the upper atmosphere). In this case, light grasp matters and is determined by the aperture of sensor. The size of optical sensors is also constrained by other issues such as optical system angular resolution (where aperture sets a diffraction limit), the need to match that angular resolution to detector spatial resolution (focal length ≥ detector resolution ÷ angular resolution) and the need to reject straylight when observing faint phenomena (so complex baffle systems are needed). Another area where physical limits apply is measurements of waves in the electric field for which the antenna length must exceed the plasma Debye length (i.e. the range over which electric field variations are screened by the plasma), which may be of order 10m for magnetospheric plasmas. In all these cases measurements are currently made on science missions using relatively large

instruments, e.g. masses of 10-40 kg and dimensions of metres. Examples include the EIT and LASCO imagers on SOHO [R17], the TIDI Fabry-Perot instrument on TIMED [R18] and the wave instruments on WIND [R19] and Cluster [R20]. It seems likely that the physical limits discussed above will prevent sensor sizes being reduced to a level that can be accommodated on a nanosat. We have identified some 11 out of 35 raw requirements that fall in this category and have excluded them from further consideration in this study.

2. Another important area where sensor size matters is energetic particle detectors. The geometric factor of a detector sets the number of particles it will collect and thus the instrument's ability to count statistically significant numbers of particles. Sensor miniaturisation will reduce this ability and thus may compromise measurements. We have not rejected any requirements on this basis, but we have identified the critical flux level that must be measured for a number of key particle types (solar protons, radiation belt electrons, auroral electrons) – see Table 52. Sensor size can be reduced only where it does not compromise measurements at that level.

3. The other element in the refinement process is instrument simplification. We have identified that the requirements for solar imagery contain one aspect where relatively low resolution is acceptable (90 arc-seconds, compared with a requirement of 2 to 5 arc-seconds from the RAL and Alcatel studies). This aspect is the use of solar images (e.g. at X-ray or EUV wavelengths) to detect the location of solar flares. 90 arc-second would allow flare location within a 20 by 20 grid, which is entirely adequate for assessing the geo-effectiveness of the solar proton event associated with a flare (this is greatest for flares on the west side of the Sun). As a result we have established one new requirement for low-resolution solar imagery, which we believe could be satisfied by a nanosat-based instrument. Another example of refinement is the use of UV photometry, in parallel with imagery, to observe the auroral oval. This offers the prospect of a more miniaturised instrument and lower data rate which will aid implementation on a nanosat. Photometry does not provide as much information as imagery but is adequate to monitor overall auroral activity and, in particular, to identify major increases in activity.

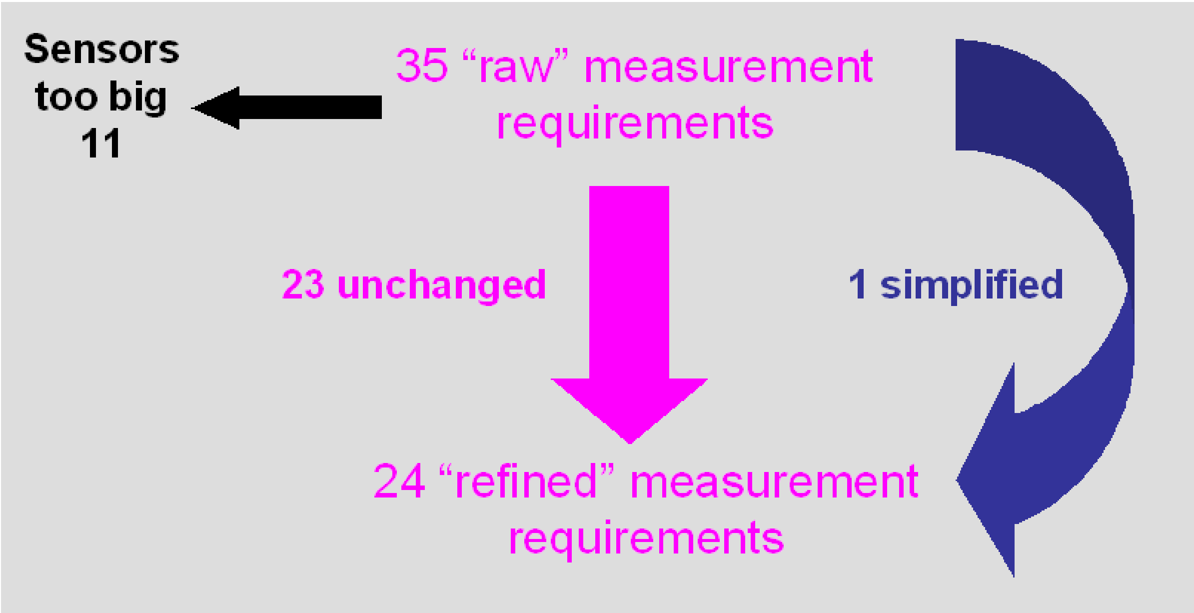


Figure 10. Refinement of measurement requirements

These steps are summarised in Figure 10 above and give a final set of 24 refined measurement requirements covering key areas such as (i) solar activity and the state of the solar wind, (ii) energetic particle fluxes and radiation dose in a variety of environments including the solar wind, radiation belts and auroral regions, (iii) the state of the auroral oval, and (iv) electron densities in the ionosphere and plasmasphere.

6.4 Classification

These requirements were then analysed to look for patterns. We first tried to classify them in terms of their applicability to each of three solution levels specified by ESA and shown in Table 12 below. Unfortunately this did not prove to be a very useful classification scheme. The vast majority of our requirements (22 out of 24) fall in level 1; this is because they are either measurements of direct interest for spacecraft operations (e.g. the energetic particle environment) or measurements of generic precursors of space weather (i.e. solar activity and solar wind state). Moving to level 2 the remaining two requirements are included by the addition of measurements needed for space weather modelling of the geomagnetic field (requirement 13.1) and the micro-particle environment (25.1). Our 24 requirements do not contain any items specific to ground-based space weather applications (those are done on the ground) or to the explicit needs of science (excluded by the derivation of our requirements analysis from the assessment studies [R11,R12]).

Table 12 The three solution levels

Solution level 1: Low level solution:
the minimum measurements required for input to services geared at mitigating space weather effects on spacecraft operations
Solution level 2: Medium level solution:
incorporates all elements of the low level solution plus additional measurements of value for modelling aspects of the geospace environment and data of importance for services geared towards mitigating ground-based space weather effects (as opposed to focusing on spacecraft effects alone)
Solution level 3: High level solution:
incorporates all elements of the low and medium level solution plus other space weather measurements of interest to the scientific community e.g. imaging data

A more successful classification scheme was to assess their appropriateness to nanosats solutions. This appropriateness was judged against two objective criteria:

1. The level of challenge in returning data, taken as the product of the data rate and the square of range to Earth. This is a useful criterion because the downlink capacity from a nanosat is expected to be constrained by available power (a few watts). The criterion value can be determined directly from the requirements, since these include the data rate and the locations at which measurements must be made.

2. The number of spacecraft needed for the ideal solution. This criterion is thought to be appropriate because a nanosat approach may facilitate mass production and deployment of spacecraft - and this favour large numbers. The criterion value can be determined from the requirements for resolution in time and space.

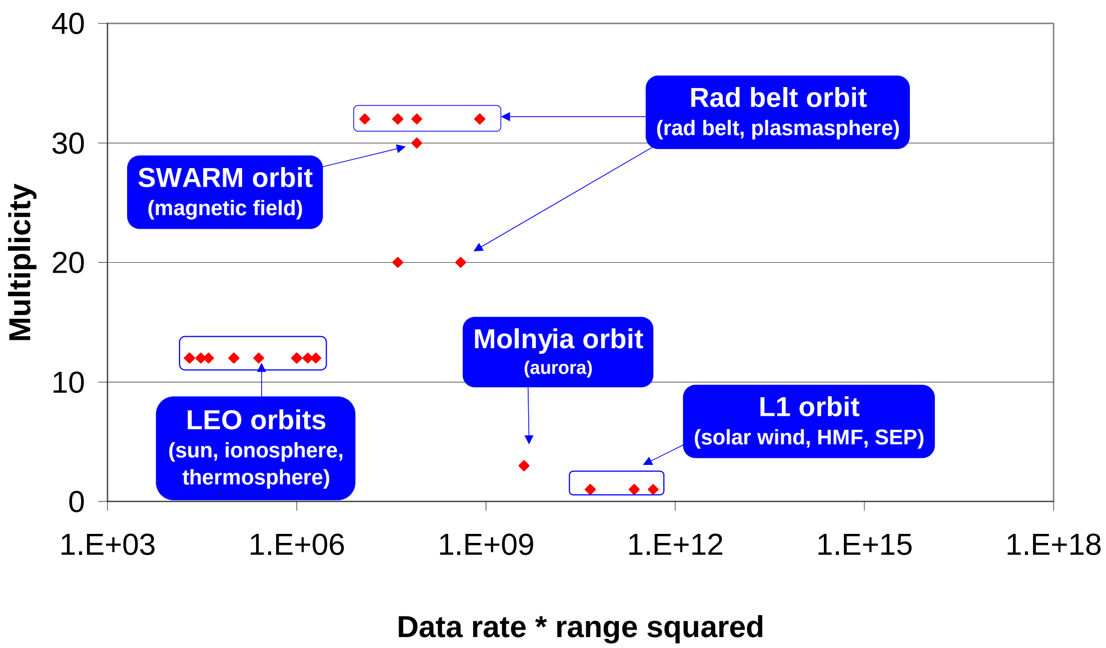


Figure 11. Classification of requirements by DR² (units=kbps km²) and multiplicity.

The results of this classification are shown in Figure 11 above. Each requirement is plotted against the two criteria and is marked by a small diamond shape. They break down into five groups each of which requires a distinct constellation of spacecraft to make measurements in the appropriate regions. The identification of these constellations is an important result that has been carried forward into subsequent work, especially mission analysis. The constellations are shown in Table 13 below.

Table 13. The space weather constellations (M = multiplicity)

Location	M	Description
L1	1	In-situ observations of solar wind, magnetic field and energetic particles at L1. Only one sampling point is required as L1 provides access to solar wind.
Low earth orbit	12	Suitable for solar, ionospheric and thermospheric observations. Use two orbits separated by 90 degrees – one in dawn-dusk plane, the other in noon-midnight. Observe sun only in dawn-dusk. The two orbits sample ionosphere and thermosphere at four local times. Use 6 spacecraft per orbit to obtain a time separation/sampling of 15 minutes.
Molniya	3	High inclination elliptical orbit (1470 x 38900km, 63.4°), suitable for remote-sensing observations of the polar ionosphere and thermosphere. The orbit period of 12h facilitates ground station coverage. This orbit is relatively stable against luni-solar perturbations. 3 s/c ensure that one spacecraft is always near apogee to make observations.
Rad belt	32	Four GTO-like orbits separated by 6 hours in local time - for in-situ observations of the radiation belts over a range of L values; multiplicity gives a resolution of 6 hours in MLT and 1 in L. Also make in-situ and remote sensing observations of the plasmasphere when inside 4 Re.
Swarm	30	A set of five highly elliptical orbits with apogee in the range 15 to 20 Re and perigee just above the atmosphere, as in proposal by Schwartz et al [8], but here used only for global study of geomagnetic field. There would be six spacecraft spaced around each orbit to give resolution over a range of geocentric distances.

6.4.1 LEO constellations

This class is selected by $DR^2 < 1x10^7$ kbps km^2 and is shown in Table 14 below. It contains the LEO applications, which are mainly ionospheric measurements with sufficient number of satellites to ensure global coverage with cadence less than the typical orbit period of 90 minutes. But it also now contains some solar applications in LEO.

Table 14. LEO constellation

Requirement number	Sub-requirement number	Measurement sub-type	Location	DR2	Multiplicity
1	1	EUV images of Sun	Solar-LEO	2.00E+04	12
25	1	Microparticle measurements	Ionospheric-LEO	3.00E+04	12
3	1	Solar X-ray flux monitor	Solar-LEO	4.00E+04	12
19	1	Dosimetry	Ionospheric-LEO	1.00E+05	12
20	1	Total electron content of iono/plasmasphere	Ionospheric-LEO	1.00E+05	12
4	2	Solar UV flux	Solar-LEO	2.50E+05	12
4	1	Solar EUV full disc flux	Solar-LEO	1.00E+06	12
22	1	Neutral density in thermosphere	Ionospheric-LEO	1.00E+06	12
21	1	Plasma velocity in ionosphere	Ionospheric-LEO	1.00E+06	12
20	2	Electron density of iono/plasmasphere	Ionospheric-LEO	1.00E+06	12
14	1	In-situ magnetospheric E field	Ionospheric-LEO	1.50E+06	12
11	2	Auroral particle precipitation	Ionospheric-LEO	2.00E+06	12

6.4.2 Molniya constellation

This group is selected by DR^2 between $3x10^7$ and $3x10^9$ kbps km^2 and multiplicity < 12 and is shown in Table 15 below. It focuses on auroral activity monitoring by UV observations from a Molniya orbit

Table 15. Molniya constellation.

Requirement number	Sub-requirement number	Measurement sub-type	Location	DR2	Multiplicity
11	1	Auroral UV imaging	Molniya	4.00E+09	3

6.4.3 Radiation belt constellation

This group is selected by multiplicity > 15 and is shown in Table 16 below. It reflects the requirement for extensive measurements of key parameters in this orbit:

- Energetic particle fluxes in the radiation belts – a key issue for spacecraft protection from radiation and charging effects, especially in the outer belt.
- Electron densities in the plasmasphere – an important issue for GNSS signals
- The magnetospheric magnetic field in order to improve magnetospheric magnetic field modelling, which is a major requirement for many space weather applications.

Table 16. Radiation belt constellation

Requirement number	Sub-requirement number	Measurement sub-type	Location	DR2	Multiplicity
20	1	Total electron content of iono/plasmasphere	Rad belt	4.00E+07	20
20	2	Electron density of iono/plasmasphere	Rad belt	4.00E+08	20
13	1	Magnetospheric magnetic field	Rad belt	8.00E+07	32
19	1	Dosimetry	Rad belt	4.00E+07	32
18	1	> 10 MeV protons in rad belt	Rad belt	4.00E+07	32
17	1	High energy electrons in rad belt	Rad belt	4.00E+07	32
16	1	10-100 keV electrons in magnetosphere/rad belt	Rad belt	8.00E+08	32
15	1	1-10 keV electrons in magnetosphere	Rad belt	8.00E+08	32
25	1	Microparticle measurements	Rad belt	1.20E+07	32
13	1	Magnetospheric magnetic field	Swarm orbit	8.00E+07	30

6.4.4 L1 orbit

This is the requirement for space weather monitoring at the Lagrangian L1 point, 1.5 million kilometres sunward of the Earth. It excludes high performance imagery (because it is very difficult to do on a nanosat) and focuses on monitoring of the solar wind, heliospheric magnetic field and heliospheric population of energetic particles.

Table 17. L1 orbit

Requirement number	Sub-requirement number	Measurement sub-type	Location	DR2	Multiplicity
10	3	2-20 MeV electrons from heliosphere	Upstream	4.50E+10	1
10	2	2-100 MeV ions from heliosphere	Upstream	4.50E+10	1
10	1	>100 MeV ions from heliosphere	Upstream	4.50E+10	1
8	2	Solar wind bulk density	Upstream	2.25E+11	1
8	1	Solar wind bulk velocity	Upstream	2.25E+11	1
9	1	Heliospheric magnetic field	Upstream	4.50E+11	1

7 Instrument solutions

7.1 Introduction

In this section we present the final set of instrument solutions developed in the course of the study. As previously discussed this is the result of a long series of iterations in which the synthesised requirements (section 6.2) were tested against what is feasible in terms of instrument design and, in particular, the constraints imposed by operation on a nanosat. We found that the latter is often challenging in terms of instrument sensor size and its ability to extract an adequate signal from the observational environment. The laws of physics often mandate a sensor that is too big to fit on a nanosat. Thus a key part of the iteration of instrument solutions has been to look deeply at the requirements so that we understand where a descoped requirement can deliver useful space weather measurements while making it feasible to fly an instrument on a nanosat.

The details of the resulting instrument solutions are presented below in section 5.2 – together with information on the key factors that guided the iteration to those solutions. This is done separately for each type of measurement: solar imagery, solar photometry, solar wind plasma, magnetic fields, thermal energy particles, high energy particles, etc. This separation is important since these measurement types are largely independent and the iteration to the solution is different for each type. Some types needed little iteration whilst others required extension iteration. Following these details a summary table (Table 18) of instrument solutions is presented in section 7.2.12.

As required by the Statement of Work [A1] we have looked at a range of near-term, medium-term and long-term solutions. The near-term solution is applicable to the year 2005 when the solutions were developed., i.e. it represents instrument technology that was flight proven in 2005. The medium- and long-term solutions are targeted towards the year 2010 and 2020. Thus medium-term solutions reflect technologies that existed in the lab in 2005 and that could be flight-proven by 2010. Long-term solutions include speculative concepts where long-term technology development is needed to achieve the necessary performance and to space-qualify the resulting instrument.

While developing the detail of instrument solutions, we also looked for system level issues that apply across the whole range of solutions. These are presented in section 7.3.

7.2 Detailed solutions

7.2.1 Solar imagery

Images of solar activity are a key space weather measurement and have been the subject of much discussion during this study. They have been a key focus for refinement of requirements discussed in section 6.3. That refinement excluded the requirements for high performance imagery; these are very difficult to satisfy on a nanosat because of the large (1 metre) optical system needed to provide adequate resolution and straylight suppression. This includes requirements: 1.4 (stereo observations of CMEs), 1.5 (helioseismology), 1.6 (Lyman-alpha scattering), 2.1 (coronagraph) and 7.1 (magnetograms). It also includes those aspects of requirements 1.1, 1.2 and 1.3 that are relevant to detection of the precursors of CME/flare activity.

The refinement identified one requirement (an updated version of 1.1) that would be possible on a nanosat - namely coarse resolution EUV imagery with sufficient resolution to estimate flare location and magnitude. It would be acceptable to measure that location to 90 arcseconds, giving 20 pixels across solar disc. Thus we were able to develop a conceptual design for a flare location imager with aperture and focal length to be reduced to scale much more appropriate for nanosat solutions. This concept would also greatly reduce the data rate; a 10 minute cadence would require only 20 bps (compared with 6000 bps needed for 5 arcsecond resolution). This could be further reduced (below 1 bps) if on-board processing were used to reduce the data to a flare status flag and location code (i.e. cell in a 20 x 20 matrix). The strawman instrument characteristics are:

- 10 x 2 x 2 (optics)
- Mass 0,1 kg
- Power 1W

7.2.2 Solar photometry

Measurements of the total flux from the solar disc are useful as ways to detect solar flares and to monitor the solar inputs that heat and ionise the upper atmosphere. Thus there are requirements to measure solar flux at several wavelengths: X-ray (3.1), EUV (4.1) and UV (4.2).

We assume that the X-ray and EUV fluxes can be measured on a nanosats with a small instrument similar that proposed for EUV images (section 7.2.1) and that these can be combined into a single instrument as has been done on GOES-R.

The UV flux may be measured by adaptation of the UV photometers described below for monitoring of UV emissions from the auroral oval – see section 7.2.6. The main difference will be the stronger solar emission, which may require some reduction of the incoming flux to match the sensitivity of the instrument. We assume similar instrumental characteristics.

7.2.3 Solar wind monitoring

Here we consider instruments to monitor the density and bulk velocity of the solar wind (requirements 8.1 and 8.2). This requires measurement of the velocity space distribution of charged particles in the solar wind, in particular the distribution for protons. The density and velocity can then be derived from the moments of the velocity space distribution. It is important to have good 3D measurements of this distribution because the solar wind bulk velocity is large compared to the mean proton thermal velocity in the solar wind.

The measurement can be achieved in the near-term (ICS1) using heritage from well-proven 2-D ion energy and mass analyzer flown on e.g. Freja (1992-1996), and Nozomi (1998) The analyzer system uses spherical electrostatic energy analyzers and a cylindrical magnetic moment analyzer. The solid angle FoV with aperture deflection is (5° x 360°). Mounted on a spinning s/c a 3d ion distribution function is obtained for every half spin period. The mass imaging system enables the ion mass composition to be obtained instantaneously. The key instrument characteristics are:

- Energy range (eV/q) 10– 35 000
- Energy resolution, ΔE/E 0.08
- FoV 5° x 360° (16 angular sectors)
- Mass range (m/q) 1 – 106
- Conversion factor 3·10-5 (cm2 sr keV/keV/cts)

In the medium-term (2010-2015) we anticipate miniaturisation of existing 3-D ion energy and mass analyzers such as those flown on Freja (1992-1996), Mars Express (2003->), Rosetta (2004->) and Venus Express (2005->). The mass resolving capability makes this design highly suited for measuring the drift velocity of the solar wind plasma. The solid angle FoV with aperture deflection is 2π (90° x 360°).

7.2.4 Magnetometer

There are two requirements for magnetometer measurements – one for the heliospheric magnetic field at L1 (9.1) and one for magnetospheric field measurements (13.1). The measurement can be achieved in the near-term using heritage from Rosetta and work in progress for Bepi-Colombo. This instrument provides 3-axis measurements of the ambient magnetic field with high accuracy and high time resolution. Magnetometers have gradually decreased in mass and size, the sensor itself today reduced to some tens of grams. The major onboard item requiring mass and additional constraints is the boom-mounting and onboard magnetic cleanliness. The magnetometer on board ESA-Rosetta, RPC-MAG, may serve as a good example of the state-of-the art in light-weight magnetometry. RPC-MAG consists of a system of two ultra light - about 28 g each - triaxial fluxgate magnetometer sensors (see Figure 12), mounted on the tip of a 1.5 m long spacecraft boom and an electronics located in the RPC electronics box inside the spacecraft. The double-ring core MACOR-cube fluxgate sensors have dimensions of about 25x25x25 mm3 each.

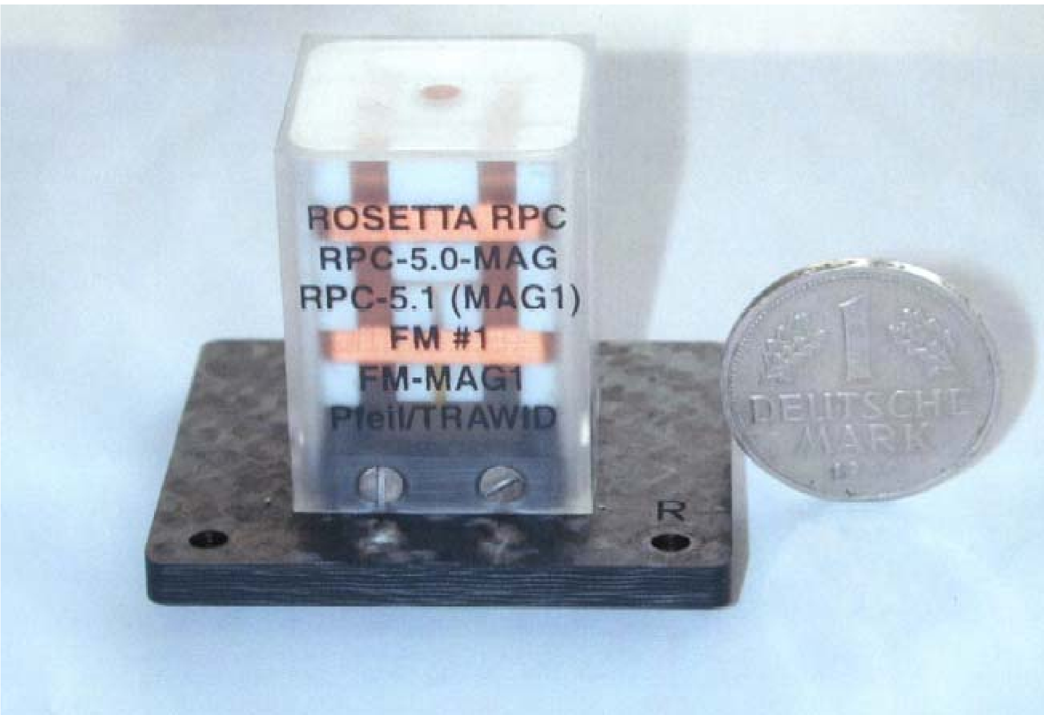


Figure 12. The MAG sensor on Rosetta. (1 Deutsche Mark coin shown for scale).

7.2.5 High energy particle measurements

There are several requirements for measurements of high energy particles: (a) high energy ions in the solar wind at energies of 2 to 100 MeV (10.1) and above (10.2); (b) high energy electrons in the radiation belt (17.1). The latter also extends to medium energy electrons in the 10 to 100 keV range (16.1).

These measurements can generally be achieved using heritage from solid state detectors flown on many missions. The boundary between use of solid state detectors and thermal plasma instruments (as discussed in section 7.2.6) is typically around 20 to 30 keV rather than the 10 keV given in the requirements. We do not consider this significant in addressing requirement 16.1, rather that the energy boundary between that requirement and requirement 15.1 (1-10 keV electrons) should be moved to 20 keV. This would reflect the realities of instrument capabilities and, perhaps more importantly, it would also reflect the fact that those capabilities are closed linked to the impact of energetic particles on spacecraft systems. In both cases what matters is the ability of energetic particles to deposit energy inside solid materials.

SSD particle detection principles are well established: the energy range is defined by surface thickness (lowest energy) and the thickness of the semi-conducting part of the SSD (highest energy). Thus high energy detectors tend to be larger and have more mass. High energy detectors also tend to have larger fields of view so that they can deliver a reasonable signal from lower fluxes. Note that it is desirable to shield sensors from high-energy particles entering the instrument from outside the field-of-view, but this will add to instrument mass.

Electron detectors typically have a thin barrier (e.g. an aluminium layer on the surface of the detector) to prevent ions reaching the detector. It is more difficult to exclude electrons from ion detectors: broom magnets can be used at low energies (<500 keV) but at higher energies ion detectors measure both electrons and ions. So the high energy ion flux is obtained by subtracting the measured electron flux from total (ion + electron) flux measured by the ion sensor. Thus ion measurements require the presence of separate electron and ion detectors. At very high energies (>20 MeV) ion fluxes dominate over electron fluxes, thus high energy ion measurements usually neglect the electron contribution - and , indeed, we do not have any requirements for electron measurements.

There is limited scope to miniaturise sensors based on solid state detectors. This would decrease the geometric factor and the sensitivity. However, if the purpose is primarily to detect high ion fluxes (as discussed in section 6.3 and Table 52), a low sensitivity is acceptable.

Thus we can summarise the instruments as follows:

- Requirement 16.1 (electrons, 20 to 100 keV) can be addressed by a small barrier-free SSD giving a modest field-of-view (say 5 x 25 degrees). Multiple detectors can be used to measure a full two- or three-dimensional distribution.
- Requirements 10.3/17.1 (high energy electrons, say 500 keV to 20 MeV) can be addressed by a barrier-free SSD and with a large field-of-view (say 60 degrees).
- Requirement 10.2 (ions, 2 to 100 MeV) can be addressed by a SSD with a barrier that prohibits energetic protons with energies less than 2 MeV from reaching the SSD. This will have a large field of view, say 60 degrees. The aluminium barrier will be transparent to high energy electrons. Thus there must be a parallel electron detector so that the ion flux can be obtained by subtraction of the electron flux. The ion flux must also be corrected for energy loss in the aluminium layer.
- Requirement 10.1 (>100 MeV ions) can be addressed by an omni-directional SSD with a stopping layer enclosing the sensor and thick enough to stop all ions below 100 MeV. The lack of an aperture (omni-directional) facilities sensor placement.

A complete sensor system for the high energy requirements (10.1, 10.2, 10.3, 17.1) can be prepared as a single instrument, including the analogue electronics for pulse-height, energy, analysis. The total mass is estimated to be 1.5 kg.

7.2.6 Aurora and magnetospheric electrons monitoring

In this section we consider instruments to monitor the aurora by UV remote sensing and by in-situ measurement of precipitating electrons. Since aurora electrons have energies in the range 1 to 10 keV, the in-situ instrument can also be applied to measurements of the thermal electrons population in the magnetosphere.

The basic requirement for remote sensing (11.1) was for a UV imager to observe the auroral oval and thereby determine its size and brightness. This instrument (UV1) can be achieved using heritage from imagers on Viking (launched in 1986), Freja (launched 1992) and Interball (launched 1996). The advantage with these imagers is that they can electronically “freeze” the image on a spinning spacecraft. The mass of such an imager is rather high (≈4 kg), driven by the existing UV-measurement technology.

During discussions on instrument solutions, it became clear that the underlying measurement requirement could reasonably be simplified – leading to an instrument more suitable for a nanosats (smaller, less mass, lower data rate). It is feasible to monitor the state of the auroral oval using only a photometer (but one operating at UV wavelengths so that (a) scattered sunlight is not significant and (b) we focus on auroral emissions that arise directly from electron precipitation). This can continuously monitor the total auroral output from the Earth´s polar region, thus providing a “global auroral index”. This instrument (UVP) can be achieved using heritage from the PIA UV-photometers on Astrid-2 – see Figure 13 below. The key instrument characteristics were:

- Focal Length 247 mm

- Resolution per pixel >8 km
- Geometric factor per pixel 4·10-4 cm2 sr
- Sample period 256 samples per second
- Spectral passband 110 – 160 nm
- Mass (total PIA-1/2 assembly) 440 g
- Power (total for PIA-1/2 assembly) 0.5 W

The UVP on Astrid-2 had a narrow field of view (2°x2°), thus providing auroral images along the track of a spinning s/c. The UV-photometer proposed here is intended to have a broader FoV (≈20°x20°), covering the entire polar region of the Earth when near apogee (Molniya orbit). The mass and power requirements for a "simple" UVP with only one photometer is quite low. The mass given above represents PIA-1/2/3, the full constellation of three photometers on Astrid-2.

Figure 13. The PIA photometers with PIA-3 at the top, and the PIA-1/2 assembly below.

There are two requirements for in-situ measurements of 1 to 10 keV electrons. One is for auroral precipitation (11.2) and one is thermal electrons in the magnetosphere (15.1). This instrument (IES) can be achieved using heritage from the MEDUSA design flown on Astrid-2 (1998) and Munin (2000) [R10]. A scaled-down MEDUSA, measuring electrons only (the current requirement), is in operation on Mars Express (ASPERA-3) and Venus Express (ASPERA-4). The key instrument characteristics for electrons are:

- Mass 1.3 kg (O.6 kg sensor and O.9 kg DPU)
- Power 2.9 W
- Energy range 4eV —22 keV
- Geometric factor incl efficiency 1.9 x 10^{-4} cm^2sr
- Energy resolution, ΔE/E 15%
- Number of sectors 16
- Number of energy steps 1 + one sweep reset step
- Number of sweeps per second 16

These requirements for in-situ measurements of 1 to 10 keV electrons can also be addressed using heritage from the PEACE instrument flown on Cluster and Double Star. A new version of PEACE is developed but with a much smaller mass, volume and power consumption.

7.2.7 Electric field monitoring

There are two requirements for electric field monitoring. One (14.1) is an explicit requirement to measure the electric fields associated with magnetospheric convection, while the other is implicit via a requirement (21.1) to measure the plasma velocity in the upper atmosphere. These two quantities (electric field **E** and plasma velocity **v**) are equivalent measures of magnetospheric convection and are related by the simple vector equation $\mathbf{E} = -\mathbf{v} \times \mathbf{B}$, where **B** is the magnetic field. Thus these two requirements are identical and a single instrument type is sufficient (but multiple instances are required to have adequate global coverage). This instrument can be achieved using heritage from instruments on Freja, Astrid-2 and Cluster and from developments for Bepi-Colombo. The instrument would have electric field probes mounted at the end of booms as shown in . When mounted on a spinning spacecraft this will provide 2-axis measurements of the ambient (convection) electric field. The key characteristics including state-of-the-art in miniaturization of the probes and boom deployment: .

- Maximum field strength ± 500 mV/m
- Bit resolution 0.0015 mV/m
- Power 1.3 – 1.7 W
- Volume 288 cm3
- Mass 1.5 kg

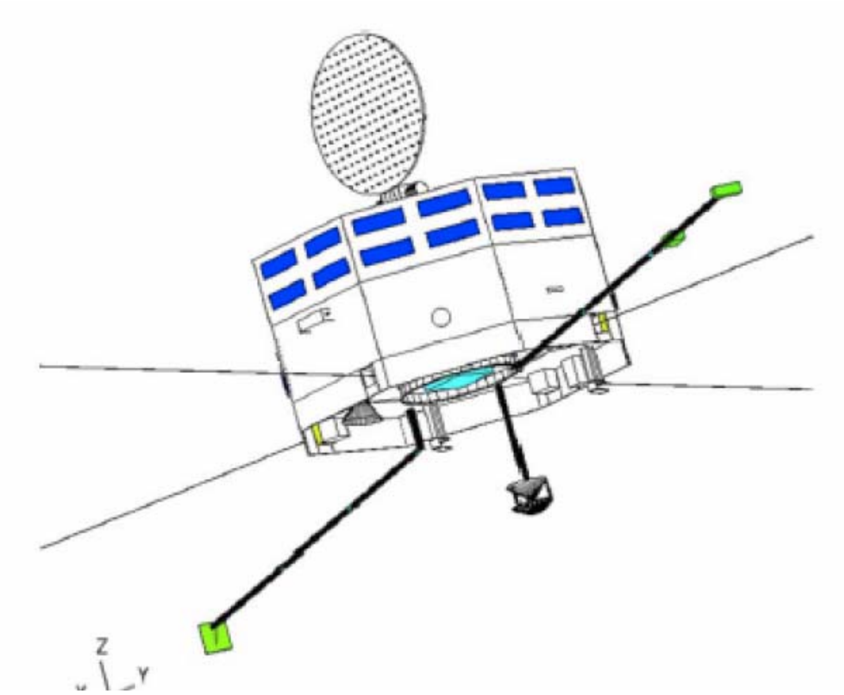


Figure 14. 2.7 E-field probes (2 x 15 m) mounted on Bepi Colombo Magnetospheric Orbiter

7.2.8 GNSS ionospheric monitoring

The state of ionosphere has profound effects on many radio-based applications and thus there are space weather requirements for monitoring the state of the ionosphere (20.1, 20.2). Spacecraft GNSS receivers in low-Earth orbit may be used to measure total electron content (i.e. column density between the receiver and the GNSS spacecraft) in the same way as ground-based GNSS systems. However, space-based measurements have a different geometry to ground-based measurements and therefore they complement rather than replicate ground-based measurements. A simple example is that, for near vertical paths, space receivers can only measure that part of the TEC that lies above the spacecraft (this may help to distinguish between the ionospheric and plasmaspheric contributions to TEC). However, the key application for space receivers is limb sounding, i.e. measurements of TEC along a series of paths as the receiver observes a GNSS spacecraft rising or setting at the limb of the Earth. Those TEC measurements can then be analysed to derive vertical profiles (e.g. by ingestion into assimilative models of the ionosphere [R37, R38]). Recent examples include the German CHAMP mission [R39] and the Taiwan-US COSMIC mission [R40].

The GNSS limb sounding technique is particularly powerful for measuring vertical profiles in the topside ionosphere, i.e. above the height of maximum electron density, which is not readily accessible to sounding by ground-based radars such as ionosondes. However, topside profiles are thought to be fairly straightforward – exponential decline with increasing altitude. In contrast ionosondes are very cost-effective at monitoring in the bottomside ionosphere (i.e. below maximum electron density). This region exhibits significant vertical structure due to the presence of a rich set of chemical processes that determine electron densities.

Space-qualified GNSS systems are readily available today with mass, size and power requirements suitable for use on a nanosats (though the high-quality systems used on CHAMP and COSMIC are too large for nanosat use). We anticipate that the quality of small GNSS systems will improve in coming years - in part because there is strong market demand for small GNSS systems for terrestrial applications. It will be important to see if these systems can be qualified for space use. We anticipate key characteristics:

- Power 0.1 W
- Dimensions 7 x 3 x 3 cm
- Mass 25 g

7.2.9 Accelerometer

The density of the neutral upper atmosphere (thermosphere) varies significantly in response to space weather events. Thus there is a requirement (22.1) to monitor this density. This is done using sensitive accelerometers that can detect the spacecraft deceleration (typically 10^{-8} to 10^{-5} ms^{-2}) due to atmospheric drag. The thermospheric density can then be derived given knowledge of the spacecraft mass and drag characteristics (e.g. cross-section in direction of motion). Existing space accelerometers (e.g. as used on CHAMP) have relatively high mass (several kg) and are therefore not very suitable for nanosats. However, accelerometers are a key target for miniaturisation using MEMS techniques, because of automotive market demands for miniature accelerometers working at 10^{-2} to 10^{2} ms^{-2}. Thus this requirement could be satisfied if that technology could be developed to give sufficient sensitivity and to be space-qualified. This is clearly a long-term prospect and thus we consider it only as available in the 2020 timeframe.

7.2.10 Dosimeter

There is also a requirement (19.1) to measure radiation dose, i.e. the effect of energetic particle radiation rather than the radiation flux itself. This is very well-suited to nanosat applications as state-of-art dosimetry is moving towards a chip size instruments, e.g. the “dosimetry-on-the-chip” being developed by Aerospace Corporation in the US [R35]. This will soon be tested in flight so we assume that instrument characteristics equivalent to those of the Aerospace development:

- Power 0.3 W
- Dimensions 4 x 4 x 1 cm
- Mass 0.3g
- Energy range 50 keV to 10 MeV
- Dose rates 1 μrad s^{-1}. to 10 mrad s^{-1}

7.2.11 Microparticles

The scope of this study includes the microparticle environment, so there is a requirement (25.1) to monitor micropartice impacts. Two methods are feasible for a nanosat: (a) a piezoelectric plate on the spacecraft surface, and (b) use of a Langmuir probe (LAP). In both cases an impacting particle will deposit energy in a target area, producing a pulse that is roughly proportional to the particle energy.

a. There is a long heritage of using piezoelectric sensors to study microparticles in space, going back at least twenty years to Giotto. An example designed for flight on a nanosat is the ISIS sensor [R36] which was developed to fly on the Italian IRECIN nanaosat. This used a sensor with dimensions 2 x 5 cm and mass 8g. This awaits flight tests so we consider this only as a medium-term option.
b. Using the LAP technique the entire satellite surface is the target, thus providing a large geometric factor. It is well-known that the impact of a microparticle creates a cloud of charged particles and a rapid change of satellite potential. This can be seen a characteristic current- or voltage-pulse that can be

detected by a Langmuir probe, with total mass (probe + electronics) around 380 g (Astrid-2 design). However, the practical exploitation of this technique for micro-particle monitoring still has to be demonstrated. Thus we give priority to use of piezoelectric sensors and flag the Langmuir probe technique as a line of research.

7.2.12 Summary

The table on the following pages summarises the instrument solutions. The fields in the table are as follows:

- Req — The measurement requirement addressed by the solution (as shown in section 6). Note that there are several cases where two consecutive requirements can be addressed by a single solution – and thus the rows for those requirements are merged to show that single solution. There are also some cases where there are multiple solutions to a requirement; in these cases the rows for those requirements are repeated by with different solutions in each row.
- Sec — The section above where the instrument solution is discussed in detail (including key factors in the iteration to that solution)
- Description — A short description of the measurement requirement
- Instrument — Instrument code
- Heritage — Flight heritage from which the instrument solution could be derived
- Dim — Estimated instrument dimensions in centimetres. Three figures are given for rectangular dimensions; two figures preceded by Ø are given for cylindrical dimensions (in this case the first dimension is the diameter). A single figure is given.
- m — Estimated instrument mass in kilograms. Values are given for 2005, 2010 and 2020.
- P — Estimated instrument power in watts. Values are given for 2005, 2010 and 2020.
- Constellation — The constellations (as discussed in section 6.4) to which the solution can be applied.

Table 18. Instrument solutions

					2005			2010		2020		
Req	**Sec**	**Description**	**Instrument**	**Heritage**	**m (kg)**	**dim (cm)**	**P (W)**	**m (kg)**	**P (W)**	**m(kg)**	**P (W)**	**Constellation**
1.1	7.2.1	EUV images of Sun	SSI	none	n/a	10 x 2 x 2 (optics)	n/a	n/a	n/a	0.1	1	LEO
3.1	7.2.2	Solar X-ray flux monitor	SSF	none	n/a	10 x 6 x 2 (optics)	n/a	n/a	n/a	0.3	2	LEO
4.1	7.2.2	Solar EUV full disc flux										
4.2	7.2.2	Solar UV flux	UV2	Astrid-2 (MIO)	0.4	Ø2 x 20	0.5	0.3	0.4	0.1	0.1	LEO
8.1	7.2.3	Solar wind bulk velocity	ICS1	MEX / VEX	1.8	Ø10 x 20	3.5	1.5	3	1.4	2.8	L1
8.2	7.2.3	Solar wind bulk density										
8.1	7.2.3	Solar wind bulk velocity	ICS3 (a)	MEX/VEX (60 mm)	n/a	Ø8 x 20	n/a	0.9	2.5	0.8	2.3	L1
8.2	7.2.3	Solar wind bulk density										
9.1	7.2.4	Heliospheric magnetic field	B-field+boom	Bepi-Colombo	1	2 x 2 x 2	2	1	1.5	0.95	1.4	L1
10.1	7.2.5	>100 MeV ions from heliosphere	ENI *	Astrid-1, adapted	1.5	Ø10 x 3	2	1	1.5	0.8	1.4	L1
10.2	7.2.5	2-100 MeV ions from heliosphere										
10.3	7.2.5	2-20 MeV electrons from heliosphere	ENE *	Astrid-1, adapted	0.5	Ø10 x 3	0.5	0.5	0.5	0.4	0.4	L1
11.1	7.2.6	Auroral UV imaging	UV1	Viking	4	Ø12 x 20	4	3	4	3	3.5	Molnyia
11.1	7.2.6	Auroral UV photometer	UVP	Astrid-2	0.44	Ø2 x 20	0.5	0.3	0.5	0.1	0.1	Molnyia
11.2	7.2.6	Auroral particle precipitation	IES	Munin/Astrid 2	0.6	Ø12 x 8	0.9	0.6	0.9	0.5	0.8	LEO
11.2	7.2.6	Auroral particle precipitation	PEACE	Cluster	1.9	15x7x5?	2.5	0.1	0.5	0.02	0.5?	LEO
13.1	7.2.4	Magnetospheric magnetic field	B-field+boom	Bepi-Colombo	1	2 x 2 x 2	2	1	1.5	0.95	1.4	Swarm/rd belt
13.1	7.2.4	Magnetospheric magnetic field	MAG	Nanospace 1	n/a	0.4x0.8x0.4	n/a	n/a	n/a	0.004	<<1?	Swarm/rad belt
14.1	7.2.7	In-situ magnetospheric E field	E-field+booms	Bepi-Colombo	2	probe Ø 3.0	2	1.5	1.5	1.4	1.4	LEO

					2005			2010		2020		
Req	**Sec**	**Description**	**Instrument**	**Heritage**	**m (kg)**	**dim (cm)**	**P (W)**	**m (kg)**	**P (W)**	**m(kg)**	**P (W)**	**Constellation**
15.1	7.2.6	1-10 keV electrons in magnetosphere	IES	Munin/Astrid 2	0.6	Ø12 x 8	0.9	0.6	0.9	0.5	0.8	Rad belt
15.1	7.2.6	1-10 keV electrons in magnetosphere	PEACE	Cluster	1.9	15x7x5?	2.5	0.1	0.5	0.02	0.5?	Rad belt
16.1	7.2.5	10-100 keV electrons in magnetosphere/rad belt	ENE *	Astrid-1, adapted	0.5	Ø10 x 3	0.5	0.5	0.5	0.4	0.4	Rad belt
17.1	7.2.5	High energy electrons in rad belt										
19.1	7.2.10	Dosimetry	DOC	Aerospace	0.045	4 x 4 x 1	0.3	0.045	0.3	0.045	0.2	LEO, rad belt
20.1	7.2.8	Total electron content of iono/plasmasphere	GPS/CMC	CMC Superstar	0.025	4.6x7.1x1.3	0.5	n/a	n/a	n/a	n/a	LEO, rad belt
20.1	7.2.8	Total electron content of iono/plasmasphere	GPS/Trimble	Trimble	n/a	2.5x2.5x0.7	n/a	0.006	0.1	0.006	0.1	LEO, rad belt
20.2	7.2.8	Electron density of iono/plasmasphere	GPS/CMC	CMC Superstar	0.025	4.6x7.1x1.3	0.5	n/a	n/a	n/a	n/a	LEO, rad belt
20.2	7.2.8	Electron density of iono/plasmasphere	GPS/Trimble	Trimble	n/a	2.5x2.5x0.7	n/a	0.006	0.1	0.006	0.1	LEO, rad belt
21.1	7.2.7	Plasma velocity in ionosphere	E-field+booms	Bepi-Colombo	2	probe Ø 3.0	2	1.5	1.5	1.4	1.4	LEO
22.1	7.2.9	Neutral density in thermosphere	ACC	none	n/a	3 x 3 x 1?	n/a	n/a	n/a	0.02	0.06	LEO
25.1	7.2.11	Microparticle measurements	MPS	IRECIN	n/a	sensor 2 x 5	n/a	0.008 (sensor)	<<1?	0.008 (sensor)	<<1?	LEO

7.3 System issues

7.3.1 Electronics evolution

We anticipate that current trends in the miniaturization of electronics will continue and will be strongly driven by market needs for higher processing capability for a given size and power consumption. This will impact instrument electronics as new technologies become space-qualified - giving a choice of a lower total power consumption or a more capable on-board processing. Greater on-board processing power is a great advantage for space weather measurements; it can be exploited in a variety of ways:

1. Greater on-board autonomy for instrument operations – tailoring instrument modes to the ambient conditions. This is particularly important for those constellations (e.g. radiation belt) where the orbit takes spacecraft through regions with very different environments. But it could also be important in adjusting instruments to operate through major space weather events that can generate interference in sensors, e.g. solar proton events. See section 7.3.3 below.
2. Loss-less compression of the raw measurement data thereby facilitating downlink of full information to the ground. This is generally more important for science measurements, where full data downlink is necessary in order to fully understand the data and thereby apply them to cutting-edge science problems. For space weather full downlink is not routinely needed but can be important for instrument calibration and performance verification and for investigation of performance problems.
3. On-board science processing of data to generate uncalibrated physical products for downlink. This approach can greatly reduce the amount of data that needs to be downlinked in routine operations. A well-established example is downlink of uncalibrated moments from particle distributions; these enable the calculation of bulk parameters such as density, velocity and temperature whilst reducing data downlink volume by several orders of magnitude. This technique has been widely used on science missions and is used operationally for downlink of real-time solar wind data from the ACE spacecraft.

The packaging of instrument electronics is also an important issue for the future. The electronics of current instruments are mainly built by packing standard components on printed circuit boards. This gives a relatively low cost, a known technology and high flexibility for modifications and tuning. But new solutions are emerging that can decrease mass, power and sometimes noise. One example is the system-on-a- chip, where a module is created from a set of components mounted in on a substrate such as silicon, thin-film or ceramic. Raw chips can be obtained from the manufacturer, and on-chip interconnect between these is customized. This packaging does not just reduce the physical size of a system; the interconnections will be shorter, enabling better connections, which can reduce power supply noise and improve signal integrity. However, this smaller packaging can make heat dissipation worse. With high level of integration the power dissipation in a small area becomes high, and risks heating parts of the module to critical levels. An example of this higher level of integration is the NanOspace-1 project developed by the Ångström Space Technology Centre [R42].

7.3.2 Shared electronics

The sharing of electronics between multiple sensors is a recognised method for decreasing payload mass. One power supply and data handling system can be used for a set of sensors, instead of each sensor having its own processing. However, the interface becomes more complex, and coordination is needed between the instrument groups (e.g. to manage interfaces and to carry out integration and verification).

The advent of more densely packed electronics, will affect this strategy in several ways, as shown in Figure 15 and discussed below:

1. The mass of electronic components will go down. PCB size will be drastically reduced as thus reduce housing mass as this follows the physical dimensions. However, the mass of harnesses is

much less likely to change. These provide data and power links between sensors (in many cases mounted on the surface of the spacecraft) and central processing units.
2. The initial design cost of a unit will increase as production will require access to specialised facilities for production of densely packaged electronics (and to design tools compatible with those facilities). Furthermore modification of the design may then only be possible by a complete (and expensive) rerun through the production facility. This trend can already be seen in the development of ASICs for space applications.
3. Given component miniaturisation, it may be worthwhile to minimise interconnects, to have as few power and data lines to a sensor as possible (e.g. local conversion to the needed voltages), compress data locally, and use fewer lines for communications.

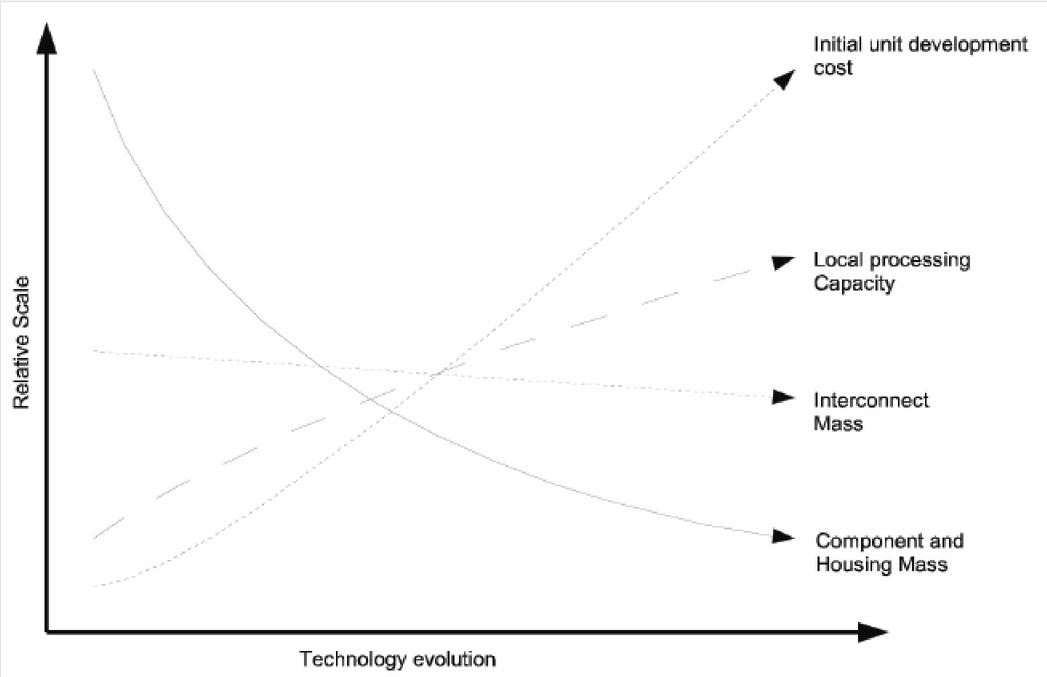


Figure 15. Estimated electronic packaging trends (only for clarification, source R. Lundin)

7.3.3 Autonomy

With the continuing development of electronics and computer system capabilities the possibilities of on-board autonomy increases. There are different aspects of on-board autonomy [R41]:

Onboard planning and scheduling. The on-board system has a set of goals that it aims to reach, and a model of its resources such as data memory, power availability, processing time etc. From this the system continuously generates a mission operations plan, fitting the on-board resources to the selected goals (in a space weather context this might be a prioritised set of relevant measurements and associated time resolutions). This is a major research area with much interest in use of AI techniques.

Onboard science processing. The onboard system analyses the instrument measurements and tries to detect significant events. This analysis will require various models to be stored onboard the spacecraft, including
- A model of the spacecraft orbit
- Model of the expected nominal readings
- Model of the instrument and algorithm for extracting general parameters from the measurements

Such on-board processing of data can be a step towards a more intelligent spacecraft, where decision rules can be implemented to decide what are the most useful products to deliver at any particular time and what combination of measurements is needed to generate those products. This measurement autonomy can be implemented on different system levels in the spacecraft, from a low level where the instrument itself processes its measurements and extracts important parameters, possibly adjusting its future operational modes, to a higher level where a network of different instruments are participating, in an intelligent interactive way.

Constellation issues. Although each satellite can achieve a good autonomy in terms of its own planning, instrument operations and data extraction, the goals of a space weather constellation reaches beyond the single satellite view, and the set of satellites should be viewed as a resource, continuously changing its geographical distribution. This could enable the handling and operational adjustments of large-scale events, such as an increased geomagnetic activity, magnetic field connection with injection of energetic particles, etc. Each spacecraft must then be aware of the entire constellation and take action that maximises the overall return from the constellation. The existence of events not evident from only single satellite measurements must be taken into consideration. Events might only be visible on a satellite-to-satellite comparing level. Solutions to this might be a central processing node, located on ground or in space, or local processing, where all satellites have knowledge of the others orbital parameters, and can communicate with a selected subset of satellites to compare measurements.

In all these cases it is necessary to do a trade-off between the value added by autonomous operation and the cost of implementing that autonomy and any extra risks involved. The trade-off should also consider the impact of autonomy on link capacity and any operational cost reductions that could arise from autonomous operation.

8 Background to the system solutions

In this section we set the scene for the later presentation of the system solutions that can implement the six space weather constellations identified in section 6. Here we outline the methodology used to develop the system solutions; we also present detailed analyses of a number of issues that are generically applicable to the system solutions (e.g. de-orbit, replacement strategy, ..).

The system solutions are described in detail in later chapters of the report (sections 9 to 14). We prefer to keep each constellation separate wherever possible. Thus we do not assume that data can be transmitted between constellations (e.g. data links from the GTO group to the LEO group would mean unfeasibly large antennas on the LEO nanosatellites). Nonetheless, there are certain cases where resources could be shared, e.g. the ground antenna network for GTO could also be used for the SWARM mission.

8.1 Methodology

The key system elements considered are shown in Table 19.

Table 19. System elements and methodology

Refinement of potential space segment orbit locations and architectures - tailored to the system solution.
Delta-V, launch, transfer and deployment strategy – see generic discussion of drag and de-orbit in section 8.2
Replacement strategy - extensively discussion in section 8.4.
Attitude and orbit control
Power generation, storage and management
System hardness and renewal
Any problems arising from use of MNT, e.g. sensitivity to the space environment, (e.g. thermal, radiation, debris, shock etc), power, radiation hardness, reliability etc.)
Thermal constraints and control. Note that it was agreed at the February 2005 workshop to just consider the qualitative thermal issues.
Communications requirements and strategy (e.g. continuous broadcast, uncommanded concept, ground station requirements, relays and ISLs). See generic discussion in section 8.5.
Ground segment, especially the need for prompt and reliable data processing and dissemination
System level parametric costing of deployment, constellation maintenance and operational costs with respect to the three levels of solution.

We assume that autonomous operation will be preferred and that the system should be operational for a minimum period of 10 years with the inclusion of a nanosat replacement strategy.

As nano-satellites can be built without the specific inclusion of nano-technologies, we make a clear separation between nano-satellites built from the foreseeable evolution of traditional technologies, and evolution driven by exploitation of MNTs. In several cases the design process is iterative. For instance, the number of satellites launched into a given constellation can have a knock-on effect on ground station coverage, link distance, constellation robustness etc. This in turn affects the limits on spacecraft size, for instance.

8.2 Drag and de-orbit

The area-to-mass ratio of a spacecraft (A/M) is the key factor that determines the atmospheric drag on a spacecraft and hence the time it takes to de-orbit due to drag. The larger this ratio is, the faster a spacecraft orbit will decay. Thus small spacecraft, identical in shape and density to large satellites, will de-orbit more quickly because they have larger area to mass ratios. This trend is mitigated to some extent by a tendency for nanosats to have higher densities than large satellites due to greater packing of hardware - see some examples in Table 20.

Table 20. Data for a range of example spacecraft

Spacecraft	Type	Mass	Volume	Density	side length	X-sec area	A/m	m/(A*Cd)
		(kg)	(m^3)	(kg/m^3)	(m)	(m^2)	(m^2/kg)	kg/(m^2)[1]
INMARSAT 4[2]	Large spacecraft	5940	46.69	127.2	3.6	12.97	0.002183	208.2
Myriades	Minisat	120	0.288	416.7	0.66	0.4361	0.003634	125.1
Munin	Nanosat	6	0.01	600	0.215	0.04642	0.007736	58.76
Cubesat	Picosat	1	0.001	1000	0.1	0.01	0.01	45.45
EADS micropack	Femtosat	0.025	0.000008	3125	0.02	0.0004	0.016	28.41

[1] Ballistic Coefficient

[2] http://www.tbs-satellite.com/tse/online/prog_inmarsat_4_spec.html

We can therefore compare nanosat decay periods for real density examples with those with an assumed uniform density (80 kg m^{-3}). This is shown in Figure 16. We can clearly see that in nanosat satellite mass range, the spacecraft density is significantly greater than 80 kg m^{-3} although smaller nanosatellites still decay faster than larger nanosatellites due to their larger A/m values, despite the increased density due to more integrated packaging.

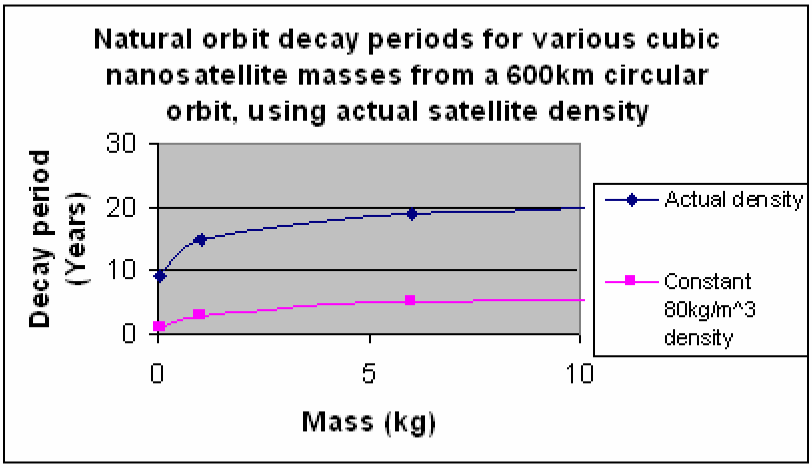


Figure 16. Decay of a cubic spacecraft

De-orbit requirements are expected to be an ESA protocol for all future missions regardless of size (colliding with a 1kg nano- satellite will still result in the end of the mission). The current ESA recommendation is that any satellite below a perigee of 2000km is required to de-orbit within 25 years after the end of standard mission operations. This affects certain LEO missions with high enough altitudes and missions with elliptical orbits (e.g. GTO and Molniya), such that re-entry does not occur passively after 25 years.

An obvious problem by using this classification is generally distinguishing between whether propulsion is required to de-orbit for particular LEO missions, or whether the spacecraft will decay naturally within the 25 year period. This problem is caused by the fact that a requirement to de-orbit depends on the individual spacecraft cross-sectional area to the velocity vector, and mass. This is often termed as the ballistic coefficient or the area/mass ratio. A spacecraft with a high ballistic coefficient value (or a low area/mass ratio) will take longer to de-orbit than a spacecraft with a lower value. To complicate matters further, deployables such as solar arrays can increase the area/mass ratio when they are normal to the velocity vector.

For tumbling satellites at EOL, it is difficult to estimate which plane the solar array is in, and usually average area/mass ratios are taken

For simplicity, a crude assumption has been made during this study in order to estimate nanosatellite de-orbit data. An example spacecraft is taken, without solar arrays using a mean MSIS-86 atmospheric density to calculate the highest orbit that would de-orbit naturally. Using the Munin microsatellite bus as an example, with mass 6kg, average cross-sectional area 0.215m x 0.215m, and ballistic coefficient 2.2, the area/mass ratio is 0.00774 m2/kg. This leads to the highest altitude for non de-orbit as 617km as shown in Figure 17. Therefore this spacecraft in an orbit above 617km in altitude would require a de-orbit. Of course, on further inspection this satellite at higher altitudes than 617km may have higher average area/mass ratios than that assumed and still de-orbit in less than 25 years. Therefore we expect spacecraft in circular orbits above these altitudes to require a propulsive deltaV to reduce the perigee sufficiently to induce de-orbit. The delta-V requirements may be quite large, e.g. up to 140 m/s to deorbit a 10 kg nanosat in circular orbit at 1000 km..

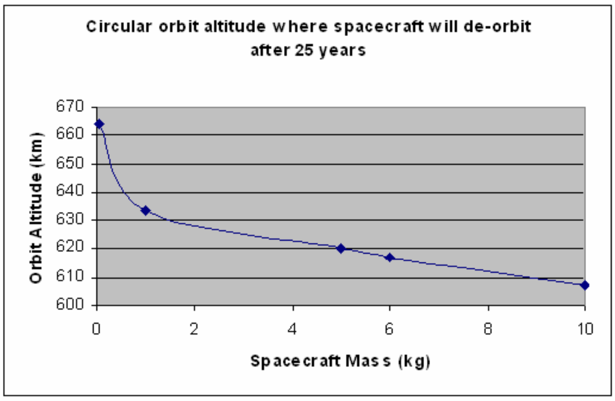


Figure 17. Decay of circular orbit

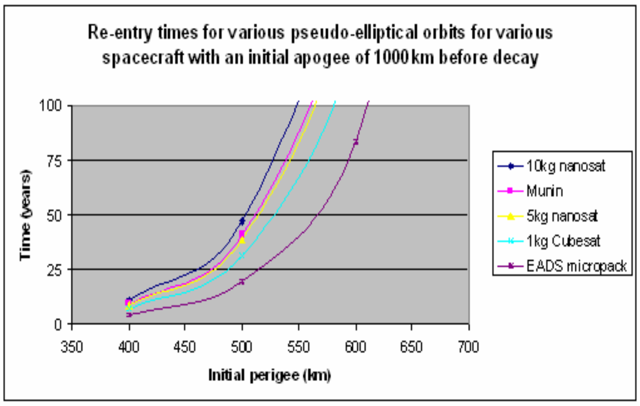


Figure 18. Decay of elliptical orbit

For spacecraft in highly elliptical orbits (e.g. Molniya and GTO), the analysis is more straightforward. The natural de-orbit is very low - as exhibited by Ariane4 stages which are still in orbit and by Figure 18. Therefore a propulsive de-orbit is required to reduce the perigee to 100km to ensure immediate de-orbit. The required delta-V is shown below.

Table 21. De-orbit requirements for highly elliptical orbits

Orbit	*Perigee*	*De-orbit deltaV's*
GTO	200km	10.6m/s
GTO	300km	21m/s
GTO	400km	31.4m/s
GTO	500km	41.6m/s
GTO	600km	51.7m/s
Molniya	1000km	85.6m/s

8.3 Nanosatellite fundamental physical limitations

Nanosatellites are defined as satellites with a wet mass of 10kg or less. This ceiling mass results in several other related constraints. We have already shown that spacecraft volume generally reduces spacecraft mass, therefore the mass limitation also results in a volume limitation, which in turn results in size limits for the following spacecraft subsystems:

Instruments	The small available volume limits the ability to include some instruments, e.g. large optical instruments. A reasonable estimate of instrument mass is around 20-30% unless there is good reason for it to be higher.

Propulsion	The limit on mass and volume reduces the hardware mass for the propulsion system and also the propellant mass. Obviously the propellant mass must not represent an unreasonable amount to be accommodated by a 10kg spacecraft.
Communications	Limited nanosatellite size also means a limit to the size of an antenna for either downlink or crosslink purposes. This effectively sets a limit on the amount of transmittable data. However in most cases the data rates are quite low due to the type of data that is measured.
Power	The limited size of nanosatellites limits the amount of power that can be available from body-mounted solar arrays. This can be alleviated by using deployed unfurlable arrays, but this adds cost and complexity and means they are unlikely to be used for nanosatellites. A simple deployable panel could also be used

Using the previous parametric information on nanosatellites, we can now calculate rough maximum power availability limitations for spacecraft depending on their relative size. We assume a triple junction GaAs solar array, efficiency of 28% and solar constant = 1400 W/m2.

Table 22. Nanosatellite available power as a function of wet mass

Spacecraft type	Mass	Volume	side length	X-sec area	Max available power W)
	(kg)	(m^3)	(m)	(m^3)	
Nanosat (Theoretical)	10	0.017	0.257	0.0661	25.9
Nanosat (Munin)	6	0.01	0.215	0.0464	18.2
Nanosat (Theoretical)	5	0.0082	0.202	0.0407	15.9
Picosat (Cubesat)	1	0.001	0.1	0.01	3.92
Femtosat (EADS micropack)	0.025	0.000008	0.02	0.0004	0.157

The power available for various subsystems can be a showstopper for potential nanosat types. For instance, a cubesat using a transmitter with output power 0.5W and input power ~4W would not have enough power available in the case of real-time data link, as all available power would have to be used for communications all the time, which of course is not possible as some power must be used for other susbsystems. Therefore a maximum antenna power of say 0.1W output and 1W input would have to be used, at least in a real-time cross-link. This could be higher if the power required is only temporary, e.g. in an elliptical orbit with variable range, where a battery could be used in conjunction.

Using Table 22 we can now say what the expected maximum power availability will be for various satellite masses, although this is just a guide. External hardware, e.g. antennas or instruments, will reduce the power availability. This table is very useful as it tells us that the ceiling maximum power for the largest nanosat class (10kg) is around 26W. Therefore we can see that mission design using nanosatellites is significantly more constrained than using more 'traditional' spacecraft.

8.4 Replacement Strategy

Here we discuss the nanosat replacement strategy. This strategy is fundamental to maintaining constellation robustness against failure and is in turn strongly dependent on system hardness. We found that replacement is a key issue given the requirement for continuous real-time operations over a period of 10 years. It is important because of two factors: (a) the perception that nanosats may be less reliable than standard spacecraft, and (b) the assessment of the impact of failure on any nanosat constellation has to be tailored to the requirements on, and design of, that constellation.

In this section we model the impact of failure on two representative cases: (a) small constellations where single spacecraft failure can significantly degrade overall system performance and (b) large constellations where overall system performance is critically dependent on how failures accumulate (i.e. the constellation can work round failures on single isolated spacecraft but not failures on adjacent spacecraft.. We analysed all these cases using a simple Monte-Carlo approach implemented in the IDL computer language (analytical approaches may also be possible but will require a deeper knowledge of number theory than was available to the study team). This software models random failures in a ring of spacecraft assuming that individual spacecraft have a constant failure rate with time, i.e. the reliability R = e-λt, where λ is the failure rate and t is the time. In this study we assumed two values for λ: giving a 20% individual failure rate after 2 years and the other 20% after 5 years. (We recommend that ESA consider a technology demonstration mission to firm up this arbitrary number, as it drives the replacement strategies.)

We note that there are other models for the random failure rate, e.g. the so-called ‘bathtub curve’, where some items fail very quickly due to flaws, whilst the rest will prove to be relatively reliable until they reach a critical age, and then most will fail within a short space of time. This is an important issue for future work.

We also note that the orbit will influence the failure rate of a nanosat. One would expect two identical nanosats placed in different orbits, i.e. benign LEO and harsh GTO, will fail at different rates. However, GTO nanosat could be designed to fail at the same rate as a LEO nanosat, although this would require either more shielding or rad-hard devices which could be more expensive (though the detail of this is beyond the scope of the study) The purpose of the following analysis is to look at reliability requirements for service operation, and orbit specific reliability is not further discussed within this section. However this is accounted for in the nanosat costing exercise

8.4.1 Small constellations and single nanosatellite failure

Here we take the Molniya constellation as representative. It requires only 2 nanosats to fulfil the science requirements, but the failure of any one of the two would result in mission failure. We therefore assess how this can be mitigated by use of replacements: both in-orbit spares and the launch of a replacement constellation. (We exclude piecemeal launch of replacements because of the launch costs.) We consider the replacement period required to maintain constellation reliability above 80% depending the reliability and calculate how it depends on the spacecraft reliability rate and the availability of an in-orbit spare (if we launch 3 nanosats there is one spare). Table 23 shows the results of these calculations together with their implications for the number of launches and nanosats needed over 10 years.

Table 23. Molynia constellation replacement strategy

Reliability	Nanosats per launch	Replacement period (years)	Replacement cycles in 10 year	Total nanosats needed in 10 years
80% over 2 years	2	0.966	11	22
80% over 2 years	3	3.05	4	12
80% over 5 years	2	2.415	5	10
80% over 5 years	3	7.62	2	6

This analysis suggests that it is more cost efficient to have a spare nanosat in orbit and it dramatically reduces the number of launches and spacecraft needed over ten years. This is a classic example of parallel redundancy. However it is important to understand that this result is strongly dependent on the elementary failure rate model. This predicts a short time to failure even with a relatively reliable nanosat with failure probability of 20% after 5 years. This is in disagreement with the Iridium mission where only one spare is used per orbit plane of 11 spacecraft. If the elementary failure rate model were used for Iridium a failure per orbit plane is predicted only just under every 0.5 years, which is clearly untrue in reality where the failures would probably occur near the end of the lifetime as with the bathtub model. Again this reinforces the need for future work to include the bathtub model.

8.4.2 Large constellations and multiple adjacent failures

Here we take the LEO ionospheric constellation as representative. It requires a ring of satellites to make measurements and transfer data via cross-links. The latter is more critical against failure. The constellation can cope with a single failure by two-way routing of data. But a second failure would break the ring into two separate segments, such that the ground station would receive no science data from the segment outside ground contact. This would be unacceptable and amount to mission failure.

This can be further mitigated in two ways: (a) the cross-links should be designed to communicate with the 'next nearest neighbour' so that it requires 2 or more adjacent satellites failures to break the ring, and (b) two nanosats should be always in view of a ground station at any time so that there is redundancy on this link.

There appears to be no analysis of this case in the published literature, so a detailed analysis was undertaken to explore the likelihood of 2 or more, 3 or more, and 4 or more adjacent satellites will fail in various constellation rings. The results for 2 or more adjacent failures (i.e. the key case for the LEO ionospheric constellation) are summarized in Figure 19 and Table 24.

Table 24. LEO ionospheric constellation replacement strategy

Reliability	Nanosats in ring	Replacement period (years)	Replacement cycles in 10 year	Total nanosats needed in 10 years
80% over 2 years	10	1.548	7	70
80% over 2 years	20	1	10	200
80% over 2 years	40	1.86	6	240
80% over 5 years	10	3.87	3	30
80% over 5 years	20	2.62	4	80
80% over 5 years	40	4.77	3	120

These results are interesting. As we add more spacecraft to the ring, the replacement period falls and the number of nanosats required increases. But at some critical point, the replacement period rises again; this rise occurs when the spacecraft become sufficiently close that there are three spacecraft in view of the ground station at any time, i.e. there is an increase in parallel redundancy such that it takes three (rather than two) adjacent failures to cause mission failure.

As with the single spacecraft case, the replacement period may be increased by including in-flight spares in each launch of a constellation. For large constellation there is also the possibility to launch supplementary constellations that strengthen the existing constellation by replacing failed spacecraft and providing new in-orbit spares. This is an exciting concept but challenging to analyse as the strengthened constellation would contain a mix of nanosats of different ages and thus different reliability. It also greatly complicates the decision about when to de-orbit existing operational spacecraft. It is an interesting topic for future work. But for the purposes of the present study, we adhere to the conservative approach of replacing the whole constellation at regular intervals.

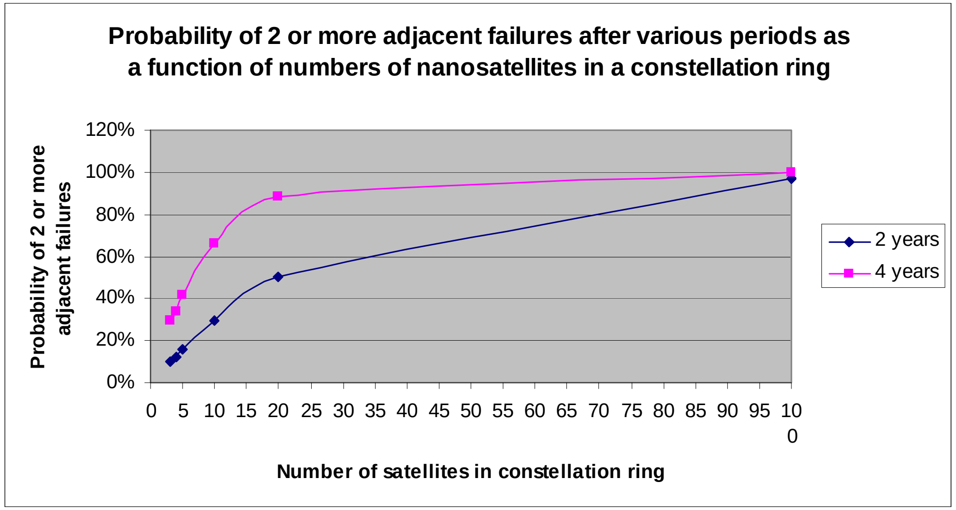
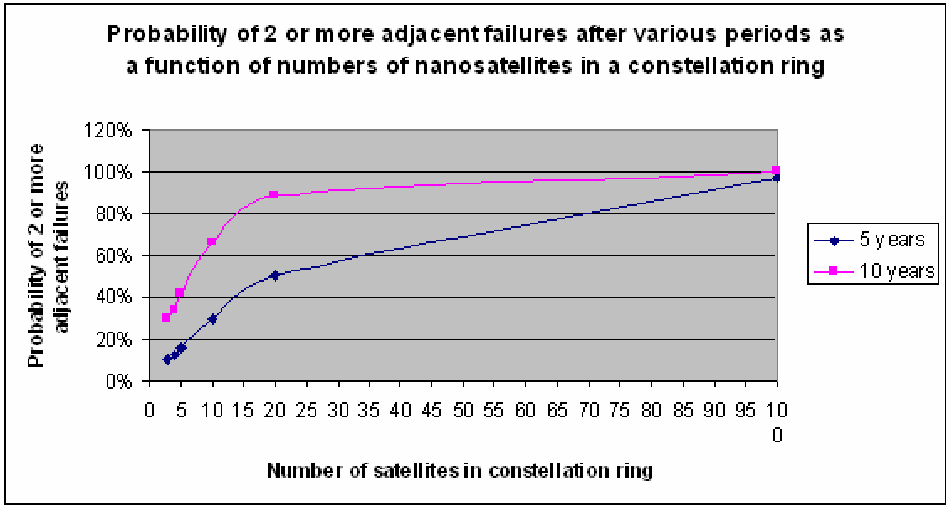
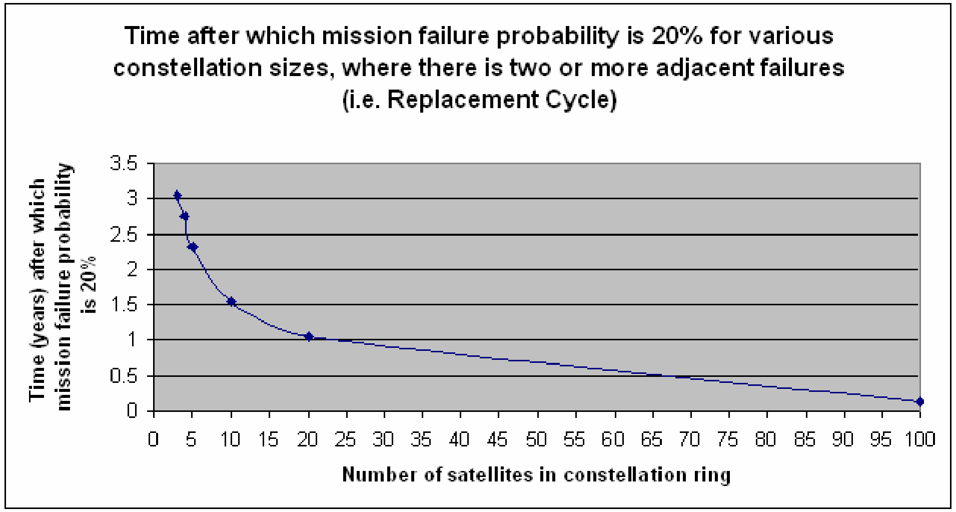
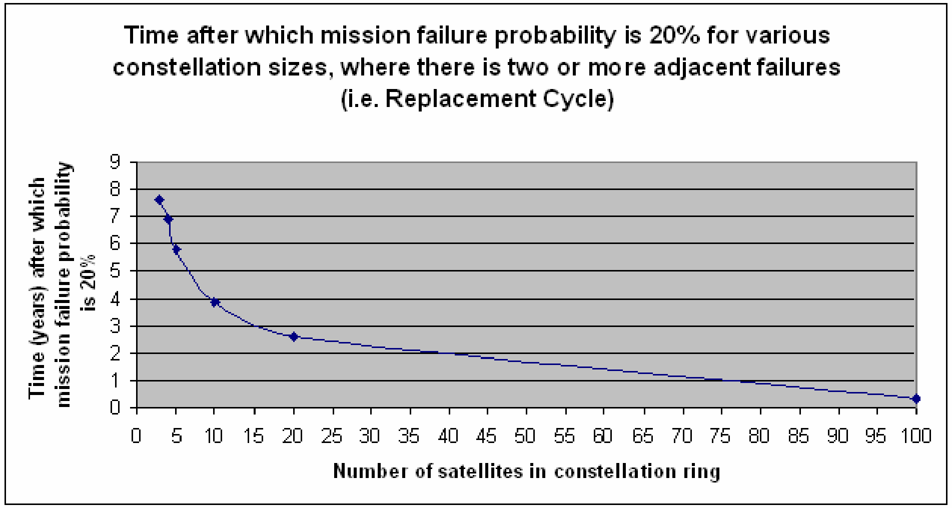
Nanosat failure probability is 20%after 2 years	Nanosat failure probability is 20%after 5 years
	
	

Figure 19. Analysis of adjacent failures in a constellation ring

8.5 Communications Requirements and strategy

Most of the space weather constellations have a requirement for their data to be delivered in real time. Thus on-board data storage is not an option and spacecraft generating data must maintain continuous downlink with the ground segment – either through direct contact to a ground station or via cross links to another spacecraft that has that direct contact.

Several of the constellations, especially the two LEO constellations, will make extensive use of cross-links in order to reduce ground station requirements. However, this approach may constrain the geographic location of the ground station to polar regions – since a very high-latitude ground station (poleward of 75 degrees latitude) can maintain continuous visibility of a constellation in polar orbit. In such cases, we have proposed to use a ground station on Svalbard, which is the best-placed European region at very high latitudes. However, an interesting complement to this would be to use a ground station at very latitudes in Antarctica (e.g. in the McMurdo region). The McMurdo site was excluded from this study as being a non-European option (and thus not appropriate for a core mission function). However, it is clear that there could be value in developing a European facility in Antartica. An Antarctic ground station would also provide an additional downlink route to overcome limits on constellation redundancy

An important issue for downlink to ground stations is the availability of space-qualified S-band transmitters with low power requirements suited to use on nanosats. At present, there is a dearth of European sources for such transmitters but one is available from UTIAS in Canada. They have developed low power/mass S-Band System for the CanX nanosatellites, with a 0.5W power output and 4W power input. Use of this transmitter is widely assumed in the system solutions.

9 Mission 1 - Ionospheric LEO constellation

This constellation will make in-situ measurements of the space environment in LEO. It includes two orbital planes (Midday/Midnight and Dusk/Dawn) with a minimum of six satellites in each plane. The scientific instrumentation requires real-time down link, therefore the constellation must have a cross-link network that can communicate to a ground station. The ground station most suitable is the Norwegian high-latitude station at Svalbard, which can give continuous contact for a constellation ring at sufficiently high altitude (1000 km). To provide adequate downlink redundancy the minimum number of satellites in each ring was raised to twenty. Since the two planes have different orbit orientations, they need to have different solar cell configuration. These requirements give the two final satellites configuration types. The satellites will be three-axis stabilised using reaction wheels and thrusters. The Up/Down link will transmit on S-band and the cross-link on X-band. The power supply unit consists of Li-Ion batteries and high efficiency Triple Junction solar cells. Each of these mission and spacecraft types is discussed in the following sub-sections.

9.1 Review of requirements & instrument solutions

The requirements and instrument solutions for this mission are shown in Table 14 and Table 18 respectively. However, analysis of these inputs showed that some cannot use the spacecraft solution outlined above and thus have been dropped as part of the trade-off:

1. The accelerometer measurements of thermospheric density (requirement 22.1) are in-situ measurements and thus need to be made at altitudes where drag is significant (say 300 km and below). This causes a problem because at this low altitude, the constellation ring is not always in view of Svalbard. We considered other solutions, e.g. use of an additional ground station in the southern polar region such as McMurdo. However, none of these was satisfactory, so we dropped requirement 22.1 as it would require a very complex data downlink. We note that this requirement might be better addressed through analysis of data obtained by space situational awareness systems and their tracking of space debris.
2. The E-field measurement requires a spin-stabilised spacecraft. This is incompatible with the choice of 3-axis stabilised spacecraft to facilitate cross-link communications. Therefore requirements 14.1 and 21.1 have been dropped. This also reduces the instrument mass and power budget to a level that better fits a nanosat solution.

Following this trade-off we conclude that the core measurement requirements (auroral precipitation, radiation dose, total electron content and microparticle fluxes) can be addressed by a standard (<10kg) nanosatellite in near term. We considered use of a picosat/cubesat in the mid/far term, but the low power availability on this solution (0.1W) cannot realistically support the requirement for cross-link communications. Thus a standard 10 kg nanosat is the recommended solution on all timescales.

9.2 Choice of orbit

This is the most complex mission in the study because of challenging requirements for real time data links and nanosatellite replenishment. The most critical issue is the choice of orbit altitude. This required a trade-off between several factors including: the number of spacecraft in the ring, the cross-link budget and de-orbit delta-V. To reduce the spacecraft costs, the trade-off led to choice of a 1000km altitude orbit with 20 spacecraft in total. In this case the crosslink requires a 19 degree beamwidth X-band antenna of diameter 0.14m with output power 0.5W. Only 6 of the spacecraft are assumed to generate data thus meeting the measurements requirements while limiting the overall data rate to 18.78kbps (= 3.13 kbps per spacecraft). The minimum beamwidth requirement for crosslink is 9 degrees - to enable additional view of the next neighbour should the immediate neighbour fail. As the beamwidth of the X-Band antenna is ~19 deg, the AOCS pointing tolerance is ±5degress, which should be easy to meet, even on a nanosat.

9.3 Launch strategies

We considered a range of launchers including Dnepr, Falcon1, FalconV, Rockot and Vega and assessed them in terms of cost, primary/secondary launch, the need for emergency launches and sourcing policies. We quickly concluded that a primary launch is more appropriate as the launch mass of 20 nanosats will fully exploit the capabilities of a cheap launcher while avoiding the need for in-orbit propulsion to reach the operational orbit. After consultation with ESA, we have also assumed that sourcing policies impose no constraint on choice of launchers.

We assumed that there is a separate direct launch into each orbit plane – again avoiding the need for in-orbit propulsion. Thus the nanosats can be carried by a dry 'hive' spacecraft whose mass should be a small fraction of the total payload mass. Even assuming a pessimistic fraction of 15%, we estimate a total launch mass of 235.3kg. This is well within the capability of the Falcon1 launcher which can carry 350kg to 1000km.

We estimate that the cost of a single Falcon 1 launch to 1000km Sun-Synchronous orbit is 9.9M€ including hive spacecraft. In comparison, the equivalent costs for Rockot and Dnepr are 16M€ and 17M€ respectively.

9.4 Transfer, deployment and DeltaV analysis

This mission involves a direct injection into the operational orbit and does not have any requirements for precise selection of altitude. In this case the main requirements for delta-V on board the spacecraft are:

1. Orbit phasing, i.e spreading the spacecraft equally around the orbit. To obtain this spread within one month, we require drift start and stop manoeuvres each with deltaV up to 5.8 m/s. Therefore the overall deltaV is 11.6 m/s. Some of the drift start deltaV (typically about 0.3 m/s) can be provided by the deployment mechanism from the hive spacecraft. To be conservative we ignore these potential deltaV gains.
2. Drag compensation and inclination correction. This is negligible. The drag is very small for a 10 kg spacecraft at 1000 km and the inclination correction is zero for noon-midnight and dawn-dusk orbits.
3. De-orbit at end-of-life. This mission will not de-orbit naturally with the 25 year requirement of ESA policy so a de-orbit mechanism is required. De-orbit of a 10 kg nanosat at 1000km requires delta-V of 141 m/s; this will drop perigee to 460 km at which point it will de-orbit naturally within 25 years. But the big question is when to trigger this manoeuvre. This might be set to occur when a certain failure probability threshold (say 10%) is approached.

We conclude that the overall deltaV requirement is about 153 m/s and is dominated by the requirement to de-orbit at end of life.

9.5 Replacement strategy

Noon-midnight constellation. This has 20 nanosatellites that must be replaced at regular intervals as discussed in section 8.4 and shown in Table 25. This strategy assumes cross-link robustness as shown in Figure 20.

Table 25. LEO noon-midnight constellation replacement strategy

Reliability	Replacement period (years)	Replacement cycles in 10 year	Total nanosats needed in 10 years
80% over 2 years	1	10	200
80% over 5 years	2.62	4	80

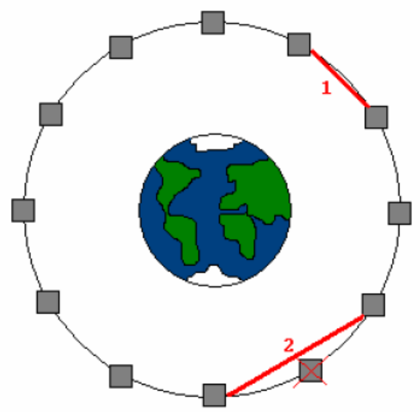


Figure 20: Principle of cross-link robustness in LEO

Dawn-dusk constellation. Here the LEO-ionospheric nanosats will be interweaved with optical-spacecraft (see next section). The replacement strategy will be discussed in section 10.5.

9.6 Systems Analysis

Table 26. Systems analysis for LEO-ionospheric constellation

Item	Solution
Propellant/propulsion	Hydrazine systems with an Isp of 214s; propellant mass required for a 10kg nanosat in a 1000 km orbit is 0.69 kg. This certainly appears feasible.
Attitude and orbit control	A 3 axis system will be used.
Power generation and management	The maximum power generated at beginning of life is estimated to be almost 26W. An estimate of the power required by the nanosat is 13W.
Power storage, eclipses	Need battery capacity of 7.6WHr, using a 13W requirement and a max 35min eclipse.
Thermal constraints and control	LEO is not considered to be a problem orbit. Cubesats function normally in this orbit.
Environment	The LEO environment at 1000km is relatively benign, though radiation must be shielded for sensitive components
Configuration	A simple cubic spacecraft is assumed with a 257 mm side length. See Figure 21. The instrument dimensions are derived from section 7.
Communications Requirements and strategy	Requires downlink data at all times and thus cross-links through a very high latitude ground station (Svalbard) where there is continuous coverage by the constellation ring. For continuous operation, need a dedicated antenna at Svalbard. A 2m S-band steerable antenna is sufficient. We assume availability of a low power S-band transmitter for the nanosat (see section 8.5).

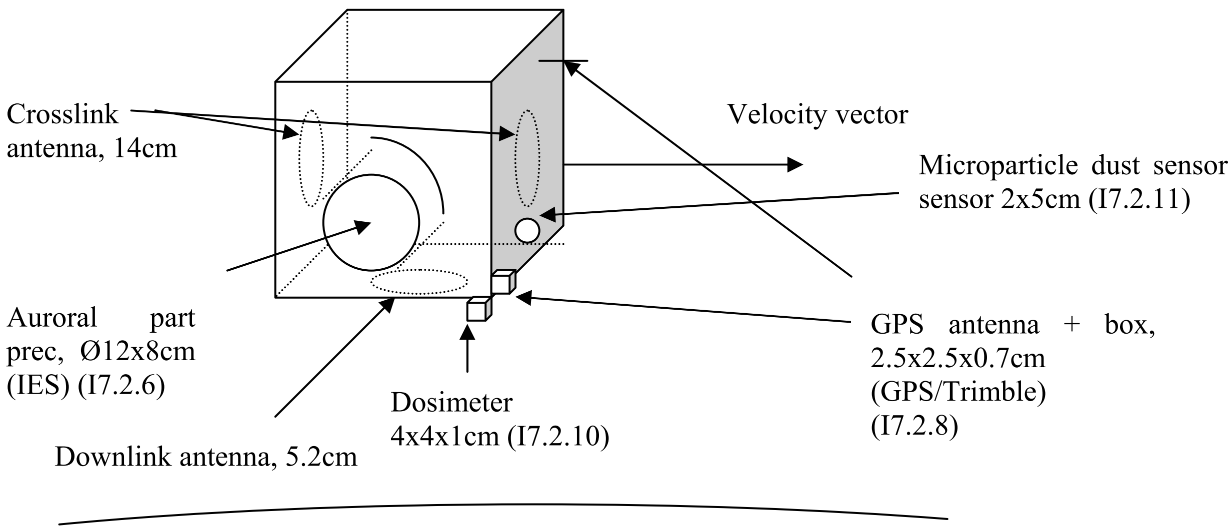


Figure 21. Sketch of LEO ionospheric spacecraft. Codes I7.2.z indicate that the instrument is discussed in section 7.2.z.

9.7 Use of MNT

Here we summarise how MNT can be utilised in the Ionospheric LEO mission.

Table 27. MST mapping to Ionospheric LEO mission spacecraft subsystems

Spacecraft Subsystem	Impact and Relevance of MEMS	Issues for using MEMS in nanosats in this mission?	Detail
Space systems engineering	System level knowledge of implementing MEMS into a larger system (e.g. packaging and interfaces). Multifunctionality and smart structures	Yes	Mass Manufacture. Introduction of MEMS culture into space industry. Product Assurance.
Space environment	Radiation dose effects in MEMS, Atomic Oxygen, charging, thermal, shock, vibration	Yes	Med-Low radiation dose
Propulsion system	Could be MEMS thrusters or MEMS components part of a larger propulsion system	Possibly	MEMS thrusters for AOCS?
Attitude control system	e.g. MEMS gyros, accelerometers and associated electronics	Yes	Electronics mass penalty for Gyros. Accuracy of gyro/rate sensor - drift. See Palmsat example
Power system	MEMS batteries, turbines, generators, seebeck effect, nuclear MEMS. Power supply	No	No MEMS power systems
Thermal control system	MEMS louvres, MEMS temperature sensors	Yes	MEMS thermal sensors envisaged
Command and data system	Integration of computers	Not at first, but eventually	
Communications system	MEMS antennae, RF switches, filters etc	Maybe	Possible MEMS antennae
Structural system	Smart/Multifunctional structures, MEMS packaging, PolyMEMS actuators (PMA)	Maybe	Possible embedded instruments/ subsystems
Payloads (sensors)	MEMS pressure, humidity, atmospheric sensors, magnetometers, GPS, particle analysers, electromagnetic fields, mass spectrometers,	Yes	Magnetometers and in-situ sensors
Reliability	Reliability of MEMS devices	Yes	Affects overall nanosat reliability

9.8 Summary

Table 28. Ionospheric LEO constellation mission summary

Mission Objective	Operational Space Weather programme • To provide near-real time monitoring of the Earth ionosphere and particles	
Payload	The instruments: (Total budget 0.2kg, 1W by 2010) • Auroral Particle Sensor • Dosimeter • GPS Antenna • Microparticle Dust Sensor	
Launcher	• 20 Nanosatellites in one hive spacecraft using Falcon1 to the noon-midnight orbit plane • 21 Nanosatellites (12 for this LEO Ionospheric constellation and 9 for LEO Optical Constellation) in one hive spacecraft using Falcon1 to the dawn-dusk orbit plane • Performance: ~350 kg to 1000 km orbit	
Spacecraft		**Baseline**
	Design Lifetime	5 years with 80% reliability (min 10 year programme)
	Attitude control	3-axis stabilised
	Total mass	10 kg each nanosat 200kg total nanosat mass (LEO-Iono constellation only) 210kg total nanosat mass (inc. LEO Optical Constellation) 22kg for hive (LEO-Iono constellation only) 23kg for hive (inc. LEO Optical Constellation)
	Spacecraft Dimensions	Cubic (257 mm diameter, 257 mm height per S/C)
	Pointing Requirements	~1 degree
	Solar array	Triple Jun GaAs, max 0.066 m^2
	Power	Estimated 13W Consumption
		Li-Ion battery 7.6Wh for noon-midnight orbit plane Dawn-Dusk 13mins max, 0 mins min Noon-Midnight 35mins max, 34.9 mins min
		~max 25.9W BOL normal to sun
	Propellant	0.69kg (Hydrazine assumed)
	Downlink Antennas (S-Band)	Hemispherical antenna 0.052m, 0.5W output power Required Eb/No assumed 10
		18.8 kbps (3.13kbps per satellite – only 6 spacecraft required for science) 20.05 kbps (inc. 3 from LEO Optical Constellation)
	Crosslink Antennas (X-Band)	2 Medium Gain transceivers (fore and aft), 0.14m, 0.5W output power, 19.1deg beamwidth Required Eb/No assumed 2.77
		18.8 kbps (3.13kbps per satellite) (LEO-Iono constellation only) 20.05 kbps (inc. LEO Optical Constellation)
Mission	**Orbit**	1000km circular Sun-Synchronous (noon-midnight & dawn-dusk)
	Programme period	10 years
	No. of Constellation Cycles	4 (noon-midnight) 5 (dawn-dusk)
	DV	152.6 m/s Total 11.6m/s Orbit Phasing of constellations 141 m/s EOL de-orbit
Operations	**Ground stations**	Svalbard (and possibly McMurdo)
		2m Antenna Okay (11dB margin)
		LEOP using ESA LEOP ground-stations?
	Timeliness	5 minutes (i.e. Real time)
Programmatics	**Phase A start** **Phase B start**	2007 2008
	Launch date	2012
	Model philosophy	STM, ATB & PFM?
Risk	**Maturity of technology**	In use or soon to be demonstrated
	Expected reliability of individual nanosatellite	0.8 after 5 years
	Expected reliability of constellation	0.8 after 2.5 years (noon-midnight) 0.8 after 2.41 years (dawn-dusk)

10 Mission 2 - Optical LEO constellation

This constellation will make optical measurements of the Sun measurements from LEO (an approach used by many solar missions). Three separate nanosat concepts are required as there are three different optical instruments to be accommodated. Given the sizes of these instruments, we envisage a separate nanosat for each instrument, with little scope for size reduction (unless some form of intelligent interferometry or free flying optical membrane can be imagined). However for continuous monitoring through eclipses and for redundancy purposes, 3 instances of each concept are required in orbit, equally spaced in mean anomaly. Therefore a total of 9 LEO optical nanosats (3 of each instrument type) are required.

This solutions requires a real-time down link, therefore the constellation needs to have a cross-link network that can communicate to a ground station. We envisage that these nanosats would be placed in the same orbit as the ionospheric LEO dawn-dusk solution and that the two constellations would be interwoven to provide a common cross-link capability with a high latitude ground station at Svalbard. This would allow us to reduce the number of ionospheric nanosats to 12 – and thus a total of 21 nanosats in the dawn-dusk plane. The data rate from the optical instruments in very low, due to on-board processing, and can neglected in comparison with the 19 kps produced by the ionospheric solution.

The satellites will be three-axis stabilised using reaction wheels and thrusters. The Up/Down link will transmit on S-band and the cross-link on X-band. The power supply unit consists of Li-Ion batteries and high efficiency Dual Triple Junction solar cells. The final orbit chosen was at an altitude of 1000km.

10.1 Review of requirements & instrument solutions

The requirements and instrument solutions for this mission are shown in Table 14 and Table 18 respectively. No further refinement of these inputs was needed.

Examination of size requirements indicates that each of three instruments (EUV imager, X-ray/EUV flux monitor, UV flux monitor) needs a separate nanosat in the 10 kg range. The instrument masses (<= 0.5 kg) and power budgets (~ 1 W) can easily be accommodated on such platforms. We note that only the UV flux monitor is mature enough for early flight; the EUV imager and X-ray/EUV flux instruments are concepts that could fly only in the longer-term (~2020).

10.2 Choice of orbit

This mission shares the complexity of the LEO ionospheric constellation. As before a critical issue is the choice of orbit altitude, which required a trade-off between factors including the number of spacecraft in the ring, the cross-link budget and de-orbit delta-V. To reduce the spacecraft costs, the trade-off led to choice of a 1000km altitude orbit with 21 spacecraft in total (9 optical and 12 ionospheric). In this case the crosslink requires a 19 degree beamwidth X-band antenna of diameter 0.14m with output power 0.5W. Only some of the spacecraft are assumed to generate data (6 ionospheric + 3 optical) thus meeting the measurements requirements while limiting the overall data rate to 22.59 kbps (= 6 x 3.13 kbps ionospheric + 3 x 1.27 kbps optical). The minimum beamwidth requirement for crosslink is 8.57 degrees - to enable additional view of the next neighbour should the immediate neighbour fail. As the beamwidth of the X-band antenna is ~19 deg, the AOCS pointing tolerance is ±5 degrees, which should be easy to meet, even on a nanosat.

10.3 Launch strategies

We considered a range of launchers including Dnepr, Falcon1, FalconV, Rockot and Vega and assessed them in terms of cost, primary/secondary launch, the need for emergency launches and sourcing policies. We quickly concluded that a primary launch is more appropriate as the launch mass of 21 nanosats will fully exploit the capabilities of a cheap launcher while avoiding the need for in-orbit propulsion to reach the operational orbit. After consultation with ESA, we have also assumed that sourcing policies impose no constraint on choice of launchers.

We assumed that there is a separate direct launch into the operational orbit – again avoiding the need for in-orbit propulsion. Thus the nanosats can be carried by a dry 'hive' spacecraft whose mass should be a small fraction of the total payload mass. Assuming a fraction of 10%, we estimate a total launch mass of 233kg. This is well within the capability of the Falcon1 launcher which can carry 350kg to 1000km.

We estimate that the cost of a single Falcon 1 launch to 1000km Sun-Synchronous orbit is 9.9M€ including hive spacecraft. In comparison, the equivalent costs for Rockot and Dnepr are 16M€ and 17M€ respectively.

10.4 Transfer, deployment and DeltaV analysis

This mission involves a direct injection into the operational orbit and does not have any requirements for precise selection of altitude. In this case the main requirements for delta-V on board the spacecraft are:

1. Orbit phasing, i.e spreading the spacecraft equally around the orbit. To obtain this spread within one month, we require drift start and stop manoeuvres each with deltaV up to 5.8 m/s. Therefore the overall deltaV is 11.6 m/s. Some of the drift start deltaV (typically about 0.3 m/s) can be provided by the deployment mechanism from the hive spacecraft. To be conservative we ignore these potential deltaV gains.
2. Drag compensation and inclination correction. This is negligible. The drag is very small for a 10 kg spacecraft at 1000 km and the inclination correction is zero for noon-midnight and dawn-dusk orbits.
3. De-orbit at end-of-life. This mission will not de-orbit naturally with the 25 year requirement of ESA policy so a de-orbit mechanism is required. De-orbit of a 10 kg nanosat at 1000km requires delta-V of 141 m/s; this will drop perigee to 460 km at which point it will de-orbit naturally within 25 years. But the big question is when to trigger this manoeuvre. This might be set to occur when a certain failure probability threshold (say 10%) is approached.

We conclude that the overall deltaV requirement is about 153 m/s and is dominated by the requirement to de-orbit at end of life.

10.5 Replacement strategy

This constellation comprises 12 LEO-ionospheric nanosats interweaved with 9 optical-spacecraft. This leads to the replacement strategy shown in Table 29 below. Note that each constellation launch must then contain a mix of 12 ionospheric nanosats and 9 optical. We assume complete replacement of that constellation on each occasion.

Table 29. Ionospheric-optical constellation replacement strategy

Reliability	Replacement period (years)	Replacement cycles in 10 year	Total nanosats needed in 10 years
80% over 2 years	0.973	11	231 (132 ionospheric, 99 optical)
80% over 5 years	2.41	5	105 (60 ionospheric, 45 optical)

10.6 Systems Analysis

Table 30. Systems analysis for ionospheric-optical constellation

Item	**Solution**
Propellant/propulsion	Hydrazine systems with an Isp of 214s; propellant mass required for a 10kg nanosat in a 1000 km orbit is 0.69 kg. This certainly appears feasible.
Attitude and orbit control	A 3 axis system will be used.
Power generation and management	The maximum power generated at beginning of life is estimated to be almost 26W. An estimate of the power required by the nanosat is 13W.
Power storage, eclipses	Need battery capacity of 2.82 Wh, using a 13W requirement and a max 13 min eclipse.
Thermal constraints and control	LEO is not considered to be a problem orbit.
Environment	The LEO environment at 1000km is relatively benign, though radiation must be shielded for sensitive components
Configuration	A simple cubic spacecraft is assumed with a 257 mm side length. See Figure 22Figure 22. Sketch of a LEO optical spacecraft.. The instrument dimensions are derived from section 7.
Communications Requirements and strategy	Requires downlink data at all times and thus cross-links through a very high latitude ground station (Svalbard) where there is continuous coverage by the constellation ring. For continuous operation, need a dedicated antenna at Svalbard. A 2m S-band steerable antenna is sufficient. We assume availability of a low power S-band transmitter for the nanosat (see section 8.5).

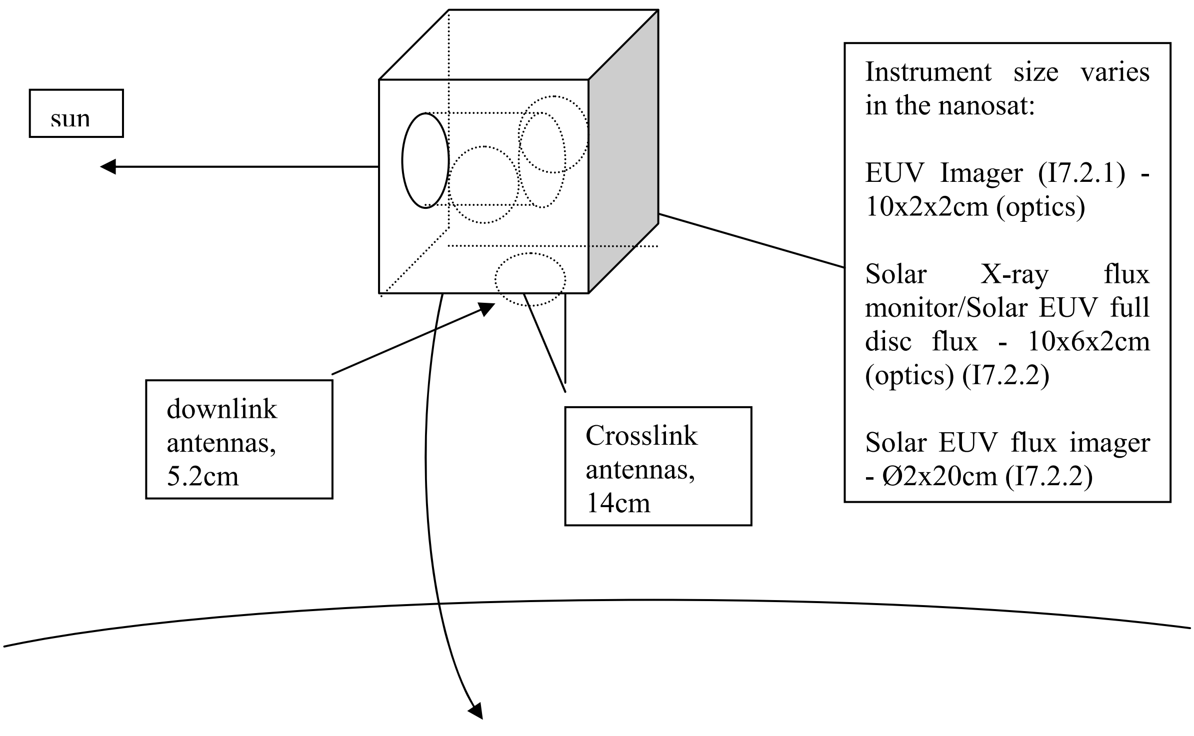


Figure 22. Sketch of a LEO optical spacecraft. Codes I7.2.z indicate that the instrument is discussed in section 7.2.z.

10.7 Use of MNT

Here we summarise how MNT can be utilised in the ionospheric-optical constellation mission

Table 31. MST mapping to ionospheric-optical spacecraft subsystems

Spacecraft Subsystem	**Impact and Relevance of MEMS as a Nanosatellite technology**	**Issues for using MEMS in nanosats in this mission?**	**Detail**
Space systems engineering	System level knowledge of implementing MEMS into a larger system (e.g. packaging and interfaces). Multifunctionality and smart structures	Yes	Mass Manufacture. Introduction of MEMS culture into space industry. Product Assurance.
Space environment	Radiation dose effects in MEMS, Atomic Oxygen, charging, thermal, shock, vibration	Yes	Med-Low radiation dose
Propulsion system	Could be MEMS thrusters or MEMS components part of a larger propulsion system	Possibly	MEMS thrusters for AOCS?
Attitude control system	e.g. MEMS gyros, accelerometers and associated electronics	Yes	Electronics mass penalty for Gyros. Accuracy of gyro/rate sensor - drift. See Palmsat example
Power system	MEMS batteries, turbines, generators, seebeck effect, nuclear MEMS. Power supply	No	No MEMS power systems
Thermal control system	MEMS louvres, MEMS temperature sensors	Yes	MEMS thermal sensors envisaged
Command and data system	Integration of computers	Maybe	In long term
Communications system	MEMS antennae, RF switches, filters etc	Maybe	Possible MEMS antennae
Structural system	Smart/Multifunctional structures, MEMS packaging, PolyMEMS actuators (PMA)	Maybe	Possible embedded instruments/ subsystems
Payloads (sensors)	MOEMS	Yes	Possible use of MOEMS in the instrumentation If it is possible to reduce sensor size without compromising the physics
Reliability	Reliability of MEMS devices	Yes	Affects overall nanosat reliability

10.8 Summary

Table 32. Ionospheric-optical constellation mission summary

Mission Objective	Operational Space Weather programme • To provide near-real time monitoring of the Sun	
Payload	The instruments (each nanosatellite carries only one instrument): • EUV Imager (LEO-Solar Nanosat 1) – 0.1kg, 1W, 0.02kbps (ready only in 2020) • Solar X-ray flux monitor/Solar EUV full disk flux monitor (LEO-Solar Nanosat 2) – 0.3kg, 2W, 1kbps (ready only in 2020) • Solar EUV flux (LEO-Solar Nanosat 3) – 0.3kg, 0.4W, 0.25kbps (2010)	
Launcher	• 21 Nanosatellites (12 for this LEO Ionospheric constellation and 9 for LEO Optical Constellation) in one hive spacecraft using Falcon1 to the dawn-dusk orbit plane • Performance: ~350 kg to 1000 km orbit	
Spacecraft		**Baseline**
	Design Lifetime	5 years with 80% reliability (min 10 year programme)
	Attitude control	3-axis stabilised
	Total mass	10 kg each nanosat 210kg total nanosat mass 23kg for hive
	Spacecraft main body dimensions	Cubic (257 mm diameter, 257 mm height per S/C)
	Pointing Requirements	~0.1 degrees
	Solar array	Triple Jun GaAs, max 0.066 m^2
	Power	Estimated 13W Consumption
		Li-Ion battery 2.82 Wh Dawn-Dusk 13mins max, 0 mins min
		~max 25.9W BOL normal to sun
	Propellant	0.69kg (Hydrazine assumed)
	Downlink Antennas (S-Band)	Hemispherical antenna 0.052m, 0.5W output power Required Eb/No assumed 10
		20.05 kbps (inc. 3 from LEO Optical Constellation)
	Crosslink Antennas (X-Band)	2 Medium Gain transceivers (fore and aft), 0.14m, 0.5W output power, 19.1deg beamwidth Required Eb/No assumed 2.77
		20.05 kbps (inc. 3 from LEO Optical Constellation)
Mission	**Orbit**	1000km circular Sun-Synchronous (noon-midnight and dawn-dusk)
	Programme period	10 years
	No. of Constellation Cycles	5
	DV	152.6 m/s Total 11.6m/s Orbit Phasing of constellations 141 m/s EOL de-orbit
Operations	**Ground stations**	Svalbard (and possibly McMurdo)
		2m Antenna Okay (11dB margin)
		LEOP using ESA LEOP ground-stations?
	Timeliness	5 minutes (i.e. Real time)
Programmatics	**Phase A start** **Phase B start**	2007 2008
	Launch date	2012
Risk	**Maturity of technology**	In use or soon to be demonstrated
	Expected reliability of individual nanosatellite	0.8 after 5 years
	Expected reliability of constellation	0.8 after 2.41 years

11 Mission 3 - Molniya constellation

This constellation will make measurements of the auroral oval from a Molniya orbit. It includes a minimum of three nanosats in the same Molniya orbit, equally spaced in mean anomaly. Only 2 spacecraft are required to ensure visibility of the oval but the third provides valuable redundancy. Real-time down link is required, but only from one spacecraft, so a single high-latitude ground station is sufficient. The satellites will be 3-axis stabilised. The Up/Down link will transmit on S-band, and a small 2m ground antenna is just sufficient. The power supply unit consists of Li-Ion batteries and high efficiency Dual Triple Junction solar cells. The satellite is assumed to have a mean power consumption of 13W during normal operation. Experience from DSRI's proposed Roemer microsatellite could be used for a future mission to Molniya orbit.

11.1 Review of requirements & instrument solutions

The requirements and instrument solutions for this mission are shown in Table 15 and Table 18 respectively. No further refinement of these inputs was needed.

Examination of size requirements indicates that in the near- and medium-term the instrument (UV imager) needs a separate nanosat in the 10 kg range. The instrument masses (<= 0.5 kg) and power budgets (~ 0.5 W) can easily be accommodated on such platforms. In the long-term we expect that miniaturisation may allow this instrument to be accommodated on smaller nanosat or cubesat/picosat, but it may be difficult for a small nanosat to accommodate a down-link antenna that is big enough to allow use of small and cheap 2m ground antennae. Thus use of 10kg nanosat is considered in this study.

11.2 Choice of orbit

The use of the Molniya orbit is specified in the requirements (see Table 15). However, the space weather use of this orbit does not require apogee fixed over northern Russia - unlike the long-standing Russian use of this orbit for communications satellites. This distinction is important: it facilitates use of secondary launches (e.g. to GTO and LEO) by simplifying the propulsion requirements for transfer to Molniya orbit.

The challenging requirements that arise from this are the real time data links and nanosat replenishment strategy. We need a minimum of two spacecraft to ensure that one always has view of the auroral ovals and the ground station, which must be at high latitude. The required downlink rate is 10kbps per spacecraft, which can be achieved using a small antenna. However, as shown below, we need more spacecraft to ensure adequate measurement redundancy and to provide cost efficiency: 1 redundant spacecraft per 2 active spacecraft. We also note that the Molniya orbit is subject to eclipses up to 57.7 minutes long and to relatively high radiation doses.

11.3 Launch strategies

We considered four main launch options as follows:

- Direct to Molniya using either just the launcher or the launcher plus an additional propulsion module. Main launcher would be Dnepr or Rockot.
- Primary launch to GTO, then boost to Molniya using a carrier spacecraft. This would be expensive, but would not rely on another primary spacecraft/launcher. Main launcher would be Falcon V, Soyuz, PSLV or Ariane 5.
- Secondary launch to GTO, then boost to Molniya using a carrier spacecraft. This is potentially cheaper than a single primary launch, but must rely on a primary spacecraft. Main launcher would be Ariane 5, e.g. using the ASAP system.
- Launch (either single or multiple) to LEO then boost to Molniya using either an additional propulsion module or a carrier spacecraft. For low inclination LEO main launcher would be Falcon, Falcon 5, Vega, GSLV or PSLV; for high inclination LEO the main launcher would be Dnepr or Rockot.

These options were then assessed in terms of cost, primary/secondary launch, the need for emergency launches and sourcing policies. This led us to the following options:

- First choice is secondary launch via GTO using an Ariane 5 and the ASAP 5 "Duosat" system. The estimated cost is ~11M€, including the hive spacecraft. This can carry 71 kg per hive for the nanosats. However, there is some uncertainty whether the available volume is adequate, even using the largest ASAP slot (900mm height). Earlier studies for Earthshine show that the tank will take up substantial volume and may not leave enough room for the nanosats. This is an important issue for any follow-up study.
- Second choice is a Falcon1 launch via low -inclination LEO. The estimated cost is ~11.9M€, including the hive spacecraft. This can carry 85 kg per hive for the nanosats. It is a non-European vehicle, but after consultation with ESA, we have assumed that sourcing policies impose no constraint on choice of launchers.
- Final choice is a Rockot launch via high-inclination LEO using either a Star module or the hive spacecraft for extra propulsion. The estimated cost is 18M€ including the hive spacecraft/Star module. But it can carry significantly more payload than we require (594 kg per hive for the nanosats).

Note that all options have more capacity than that needed to deliver three 10 kg nanosats to a Molniya orbit. We also considered the option of sharing on Russian launches direct to Molniya orbit, but this was difficult to evaluate because of the limited availability of cost data that can scaled to nanosat launches.

11.4 Transfer, deployment and DeltaV analysis

This first choice launch option involves a single launch into an intermediate GTO orbit after which the hive spacecraft immediately changes the inclination and slightly boosts the apogee/perigee into the operational orbit for the nanosats. The rapid transition (a few hours) from GTO to Molniya orbit simplifies requirements on the hive spacecraft, i.e. it can be relatively simple with just propulsion and structure, thus saving cost. We assume that the hive spacecraft is only responsible for injection into the operational orbit and that subsequent manoeuvres will be performed using a propulsive capability on-board each nanosat. The main delta-V requirements for this capability are:

1. Orbit phasing, i.e spreading the spacecraft equally around the orbit. To obtain this spread within one month, we require drift start and stop manoeuvres each with deltaV up to 8.5 m/s. Therefore the overall deltaV is 17 m/s. Some of the drift start deltaV (typically about 0.3m/s) can be provided by the deployment mechanism from the hive spacecraft. To be conservative we ignore these potential deltaV gains.
2. Drag compensation and inclination correction. This is negligible.
3. De-orbit at end-of-life. This mission will not de-orbit naturally with the 25 year requirement of ESA policy so a de-orbit mechanism is required. De-orbit of a 10 kg nanosat in Molniya orbit (perigee 1000km, apogee 39360 km) requires delta-V of 85.6 m/s; this will drop perigee to 100 km at which point it will de-orbit promptly. But the big question is when to trigger this manoeuvre. This might be set to occur when a certain failure probability threshold (say 10%) is approached.

We conclude that the overall deltaV requirement is about 103 m/s and is dominated by the requirement to de-orbit at end of life.

11.5 Replacement strategy

Table 33. Molniya constellation replacement strategy

Reliability	Replacement period (years)	Replacement cycles in 10 year	Total nanosats needed in 10 years
80% over 2 years	3.05	4	12
80% over 5 years	7.7	2	6

11.6 Systems Analysis

Table 34. Systems analysis for Molniya constellation

Item	**Solution**
Propellant/propulsion	Hydrazine systems with an Isp of 214s; propellant mass required for a 10kg nanosat in a 1000 x 39360 km Molniya orbit is 0.48 kg. This certainly appears feasible.
Attitude and orbit control	A 3 axis system will be used.
Power generation and management	The maximum power generated at beginning of life is estimated to be almost 26W. An estimate of the power required by the nanosat is 13W.
Power storage, eclipses	Need battery capacity of 12.5 Wh, using a 13W requirement and a max 57.7 min eclipse.
Thermal constraints and control	Molniya is not considered to be a problem orbit. [it was agreed at the Feb 2005 workshop to just consider the qualitative thermal issues]
Environment	The Molniya environment is relatively harsh, so radiation must still be shielded for sensitive components accordingly.
Configuration	A simple 3-axis stabilized cylindrical spacecraft is assumed with a 257 mm side diameter and height as shown in Figure 23. Figure 22. Sketch of a LEO optical spacecraft. The instrument dimensions are derived from section 7.
Communications requirements and strategy	Requires continuous data downlink from the active spacecraft. Ground station must be at high latitude for visibility (e.g. Kiruna, Svalbard). Also we take a small 2m ground station as the goal. Restrict spacecraft antenna size to <20cm (46.6deg beamwidth at S-Band) to accommodate auroral imager on Earth-pointing face. This implies output power of 1.61 W and thus input power of 20W. This can be accommodated if battery storage supplements the 13 W array power; the battery can be re-charged when the spacecraft is not the active one.

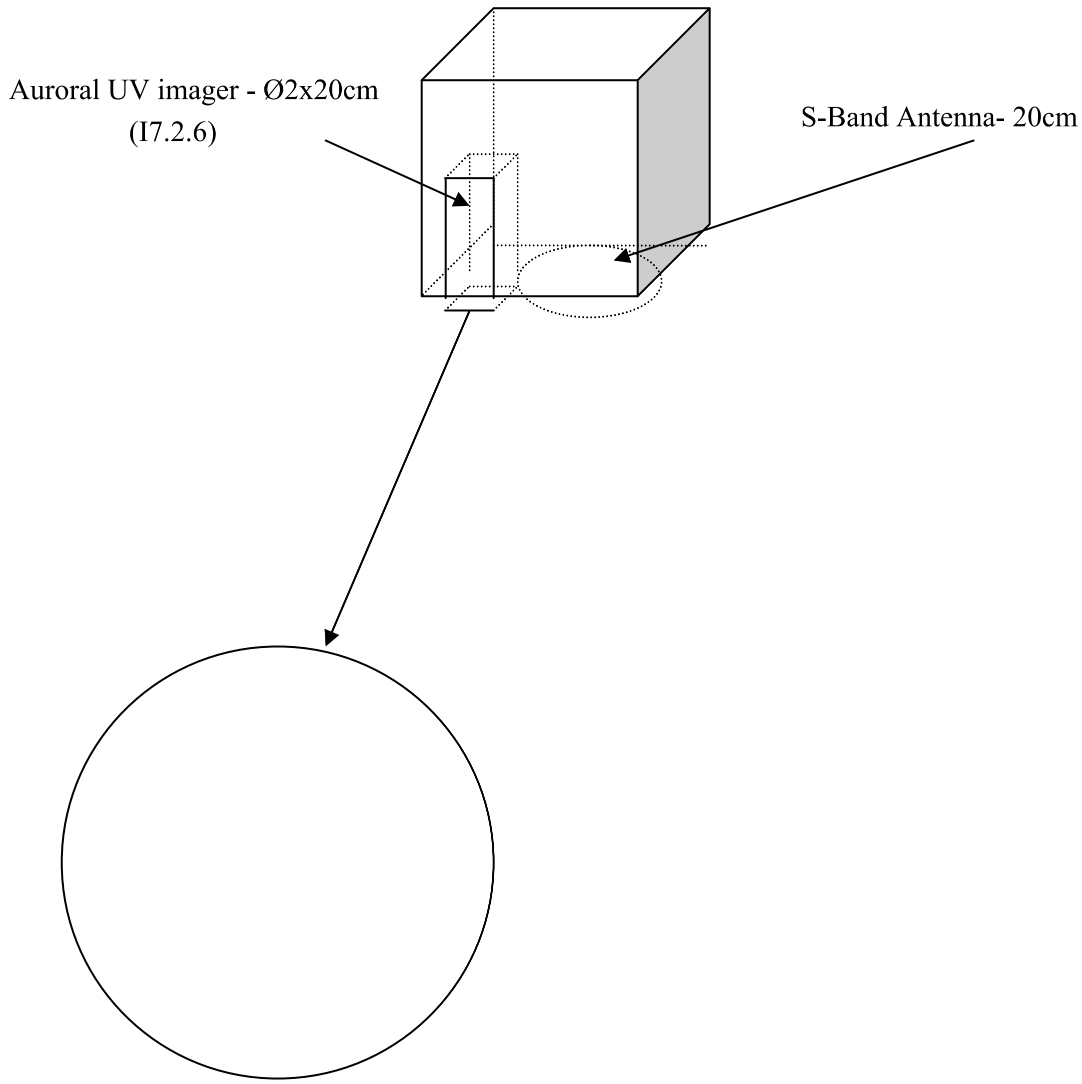


Figure 23. Sketch of Molniya spacecraft. Codes I7.2.z indicate that the instrument is discussed in section 7.2.z.

11.7 Use of MNT

Here we summarise how MNT can be utilised in the Molniya mission

Table 35. MST mapping to Spacecraft Subsystems

Spacecraft Subsystem	Impact and Relevance of MEMS as a Nanosatellite technology	Issues for using MEMS in nanosats in this mission?	Detail
Space systems engineering	System level knowledge of implementing MEMS into a larger system (e.g. packaging and interfaces). Multifunctionality and smart structures	Yes	Mass Manufacture. Introduction of MEMS culture into space industry. Product Assurance.
Space environment	Radiation dose effects in MEMS, Atomic Oxygen, charging, thermal, shock, vibration	Yes	Relatively high radiation dose
Propulsion system	Could be MEMS thrusters or MEMS components part of a larger propulsion system	Possibly	MEMS thrusters for AOCS?
Attitude control system	e.g. MEMS gyros, accelerometers and associated electronics	Yes	Electronics mass penalty for Gyros. Accuracy of gyro/rate sensor - drift. See Palmsat example
Power system	MEMS batteries, turbines, generators, seebeck effect, nuclear MEMS. Power supply	No	No MEMS power systems
Thermal control system	MEMS louvres, MEMS temperature sensors	Yes	MEMS thermal sensors envisaged
Command and data system	Integration of computers	Maybe	In longer term
Communications system	MEMS antennae, RF switches, filters etc	Maybe	Possible MEMS antennae
Structural system	Smart/Multifunctional structures, MEMS packaging, PolyMEMS actuators (PMA)	Maybe	Possible embedded instruments/ subsystems
Payloads (sensors)	MOEMS	Yes	Possible use of MOEMS in the instrumentation If it is possible to reduce sensor size without compromising the physics
Reliability	Reliability of MEMS devices	Yes	Affects overall nanosat reliability

11.8 Summary

Table 36. Molniya Constellation Mission Summary

Mission Objective	Operational Space Weather programme • To provide near-real time monitoring of the Auroral Ovals from a Molniya orbit	
Payload	The instruments: (Solution 2 Total budget 0.3kg, 0.5W by 2010) • Auroral Imager	
Launcher	• 3 Nanosatellites in one hive on ASAP 5 "Duosat" to GTO, followed by plane change and orbit correction/change to Molniya • Performance: ~71kg of useful nanosat mass to Molniya orbit	
Spacecraft		**Baseline**
	Design Lifetime	5 years with 80% reliability (min 10 year programme)
	Attitude control	3-Axis stabilised
	Total mass	10 kg each nanosat 22.5kg for empty hive, 150kg total of empty hive+propellant+nanosats
	Spacecraft main body dimensions	Cubic (257 mm diameter, 257 mm height per S/C)
	Pointing Requirements	~1 degree
	Solar array	Triple Jun GaAs, max 0.066 m^2
	Power	Estimated 13W Consumption
		Li-Ion battery 12.5 Wh depending on orbit plane Eclipse 0.96 hours/57.7min max
		~max 25.9W BOL normal to sun
	Propellant	0.48kg (Hydrazine assumed)
	Downlink Antennas (S-Band)	Semi-Omni antenna 0.16m, 0.5W output power nominal 3dB margin at 37000km max link distance Required Eb/No assumed 2.77
		10 kbps per satellite
Mission	**Orbit**	Molniya
	Programme period	10 years
	No. of Constellation Cycles	2
	DV	102.6m/s Total 17m/s Orbit Phasing of constellations 85.6m/s EOL de-orbit
Operations	**Ground stations**	1 high latitude (Svalbard or Kiruna)
		2m Antenna Okay (3dB margin, more if only data from fraction of spin allowed)
		LEOP using ESA LEOP ground-stations?
	Timeliness	5 minutes (i.e. Real time)
Programmatics	**Phase A start** **Phase B start**	2007 2008
	Launch date	2012
Risk	**Maturity of technology**	In use or soon to be demonstrated
	Expected reliability of individual nanosatellite	0.8 after 5 years
	Expected reliability of constellation	0.8 after 7.7 years

12 Mission 4 – GTO constellation

This constellation will make in-situ measurements of the space environment in geosynchronous transfer orbit (GTO). The constellation comprises four orbits separated by 90 degrees in local time and with a minimum of eight satellites in each orbit. The instruments require real-time down link, so the constellation must have a crosslink or down-link network that can always communicate to a ground station. A network of at least 7 small equatorial ground stations was selected as being simpler and cheaper than the challenging crosslink equivalent. During the development of the concept the minimum number of satellites was increased to 16 per orbit (64 in total) - to have sufficient redundancy to cover each orbit to the resolution specified in the requirements. The satellites will be spin-stabilised at 5 rpm. The Up/Down link will transmit on S-band. The power supply unit consists of Li-Ion batteries and high efficiency Dual Triple Junction solar cells. The nanosat is assumed to have a mean power consumption of 13W during normal operations.

12.1 Review of requirements & instrument solutions

The requirements and instrument solutions for this mission are shown in Table 14 and Table 18 respectively. These can be divided into two groups according to the solution levels shown in Table 12. Only a subset of the requirements and instrument solutions (particle fluxes, dosimetry and ionospheric densities) are needed to support services that mitigate space weather effects on spacecraft operations (solution level 1). But the complete measurement set (including magnetic fields, microparticles) extends this to include support for modelling aspects of the geospace environment (solution level 2).

Examination of the requirements indicates that both solutions need a separate nanosat in the 10 kg range. The solution 1 payload mass is 1.2 kg in 2005 falling to 0.5 kg in 2020 while the power budget is 2.2 W in 2005 falling to 1.2 W in 2020. This can easily be accommodated on a 10 kg nanosat. The solution 2 requirements are more demanding: payload mass of 2.2 kg falling to 0.5 kg in 2020, power of 4.2 W in 2005 falling to 1.4 W in 2020. Nonetheless they can be accommodated on a 10 kg nanosat, especially in the longer-term. We also note that it may be difficult for a small nanosat to accommodate a down-link antenna that is big enough to allow use of small and cheap 2m ground antennae. Thus use of 10kg nanosat is preferred in this study (but we note that a smaller 5 kg nanosat might be considered in the longer-term.

12.2 Choice of orbit

We need a minimum of eight spacecraft in each of four GTO orbits separated by 90° in longitude. This ensures adequate sampling of the radiation belts by local time and by distance from the Earth (as represented by the McIwain L parameter). The required downlink rate is 5.33 kbps per spacecraft in real-time. If this were to be handled by same plane crosslinks between adjacent spacecraft, we would have to increase the number of nanosats per plane to 36 to keep the separation within a maximum link distance if 9600 km. The use of inter-plane cross-links is even more complex as it would require steered antennae (omnidirectional antennae are probably not viable because of the large spacecraft separations). Thus the preferred solution is continuous direct downlink from each spacecraft. This requires a minimum constellation of 7 small ground stations spaced equally around equator. Nonetheless we still need more than eight spacecraft - to ensure adequate measurement redundancy and to provide cost efficiency as discussed below. To achieve this we need 1 redundant spacecraft per active spacecraft. Finally we note that the GTO orbit is subject to eclipses up to 2 hours long and to relatively high radiation doses.

12.3 Launch strategies

We considered four main launch options as follows:

- Direct launch into GTO as primary payload followed by dispersion in local time using a carrier spacecraft. This gives full control of the launch conditions and the direct costs of the launch are relatively low. However, it requires a sophisticated carrier spacecraft - essentially a fully functional spacecraft that must survive for up to 1 year in a harsh orbit environment. This spacecraft must provide a delta-V of 874 m/s to set-up and correct a drift orbit to the required local time.

- Direct launch into GTO, but as a secondary payload, again followed by dispersion in local time using a carrier spacecraft. This differs from the previous option in that secondary launch costs are much less but the launch conditions are dictated by the primary payload.

- Multiple direct launches into GTO as a secondary payload with no dispersion in local time. This approach exploits the low cost of secondary launches but requires that we find a primary launch that will place our spacecraft close to each of the four required local times. This is potentially the cheapest option as the carrier spacecraft can essentially be a cheap empty shell. The disadvantage is the strong reliance on opportunities provided by primary launches.

- Launch (either single or multiple) to LEO then boost to GTO using a carrier spacecraft. For a single launch we require dispersion in local time using a carrier spacecraft. In this respect the LEO scenario has the advantage of providing rapid precession in local time [R15]; even in the worst case, the carrier spacecraft has to survive only for 5 weeks. For multiple launches we can use separate launches to each local time and thus very cheap carriers (effectively just something that carries the nanosats during launch and deploys them upon orbit acquisition.).

In all cases we require a carrier spacecraft (hive). In the case of multiple launches (one per local time target) we require 16 nanosats per hive, so therefore each hive must carry 160kg of nanosats. But if we have a single launch, that must populate all four local time targets. So we require 4 separate hives sharing 640kg of nanosats. The cost of each hive cost is estimated as (a) 3M€ with no propellant and instant deployment after injection, (b) 5M€ with propellant and instant deployment, and (c) 10M€ with propellant and a significant time in orbit before deployment. It is also assumed that the empty hive mass is 15% of the overall launch mass.

These options were then assessed in terms of cost, primary/secondary launch and sourcing policies. We found that the LEO options were more desirable than the GTO options. The availability of low-cost launchers and the faster dispersion in local time makes it much easier to get the spacecraft in the right orbits. All GTO options require two of the following three items: an expensive primary launch, an expensive high-performance carrier spacecraft or good luck in finding the right quartet of secondary launches. This led us to the following options:

- First choice is four **Falcon1** launchers to LEO followed by a carrier spacecraft boost to GTO. This a good, cost effective choice here as it can carry 200kg per hive for 16 nanosats and also has adequate fairing volume for such a mission. The desired orbit can be achieved straight away which significantly reduces hive complexity, and there is also a risk spreading advantage with 4 separate launches. The major downside with Falcon1 is that it is non-European. Estimated cost is 47.6M€, inc. Hive spacecraft
- Second choice is two **Dnepr** launches to LEO. This is capable of carrying a large 302kg per hive in two launches for 16 nanosats, and is attractive in terms of its cost. The desired orbit can be achieved in twenty or so days in the worst case, and there is also a risk spreading advantage with 2 separate launches. The major downside with Dnepr is that it is non-European. Estimated cost is 56M€.
- Single **Vega** to LEO, This is capable of carrying 175kg per hive for 16 nanosats, and is attractive in terms of its cost and the fact that it is European. Estimated cost is 60M€, inc. hive spacecraft.

Thus the launch strategy is dependent on sourcing policies for launch vehicles. Discussions with ESA during the course of the present study indicated that the launcher does not fall within the study requirement to consider European-only hardware. Thus we conclude that the use of a non-European launcher is a realistic option for the GTO mission and can reduce costs.

12.4 Transfer, deployment and DeltaV analysis

This first choice launch option involves four separate launches into an intermediate LEO orbit after which the hive spacecraft immediately boosts the apogee into the operational orbit for the nanosats. This rapid transition simplifies requirements on the hive spacecraft, i.e. it can be relatively simple with just propulsion and structure, thus saving cost. We assume that the hive spacecraft is only responsible for injection into the operational orbit and that subsequent manoeuvres will be performed using a propulsive capability on-board each nanosat. The main delta-V requirements for this capability are:

1. Orbit phasing, i.e spreading the spacecraft equally around the orbit. To obtain this spread within one month, we require drift start and stop manoeuvres each with deltaV up to 7.9 m/s . Therefore the overall deltaV is 15.8 m/s. Some of the drift start deltaV (typically about 0.3 m/s) can be provided by the deployment mechanism from the hive spacecraft. To be conservative we ignore these potential deltaV gains.
2. Drag compensation and inclination correction. This is negligible over the short mission duration.
3. De-orbit at end-of-life. This mission will not de-orbit naturally with the 25 year requirement of ESA policy so a de-orbit mechanism is required. De-orbit of a 10 kg nanosat with perigees between 500 and 600 km will require delta-Vs between 41.7 to 51.7 m/s. But the big question is when to trigger this manoeuvre. This might be set to occur when a certain failure probability threshold (say 10%) is approached.

We conclude that the overall deltaV requirement for nanosats in GTO orbit with perigees of 500 to 600 km is between 57.4 and 67.5 m/s and is dominated by the requirement to de-orbit at end of life.

12.5 Replacement strategy

Table 37. GTO constellation replacement strategy

Reliability	Replacement period (years)	Replacement cycles in 10 year	Total nanosats needed in 10 years
80% over 2 years	1.18	9	576
80% over 5 years	2.94	4	256

12.6 Systems Analysis

Table 38. Systems analysis for GTO constellation

Item	Solution
Propellant/propulsion	Hydrazine systems with an Isp of 214s; propellant mass required for a 10kg nanosat in a GTO orbit with 600 km perigee is 0.32 kg. This certainly appears feasible.
Attitude and orbit control	A 3 axis system will be used.
Power generation and management	The maximum power generated at beginning of life is estimated to be almost 26W. An estimate of the power required by the nanosat is 13W.
Power storage, eclipses	Need battery capacity of 28.3 Wh, using a 13W requirement and a max 2.18 hour eclipse.
Thermal constraints and control	GTO is not considered to be a problem orbit as shown in previous ESA CDF studies [R15]. [it was agreed at the Feb 2005 workshop to just consider the qualitative thermal issues]
Environment	The GTO environment is very harsh, so radiation must be shielded for sensitive components accordingly. Experience from STRV c/d should be used.

Configuration	A spin stabilized cylindrical spacecraft is assumed with a 363 mm diameter and 182 mm height and with two equal and opposite booms in the middle of the cylinder wall as shown in Figure 24. The instrument dimensions are derived from section 7.
Communications Requirements and strategy	Requires continuous data downlink from the active spacecraft. Data rates are not demanding. Need minimum of seven equally spaced ground stations at equatorial latitudes (more may be needed to take account of distribution of equatorial land mass). Aim at dedicated network of a small 2m ground stations with routine operation being unstaffed. This should minimise costs. 18cm spacecraft antenna working at S-Band and transmitting only when Earth-pointing. Needs output power of 1 W and thus input power of 10W. Limited need for uplink especially in the longer term.

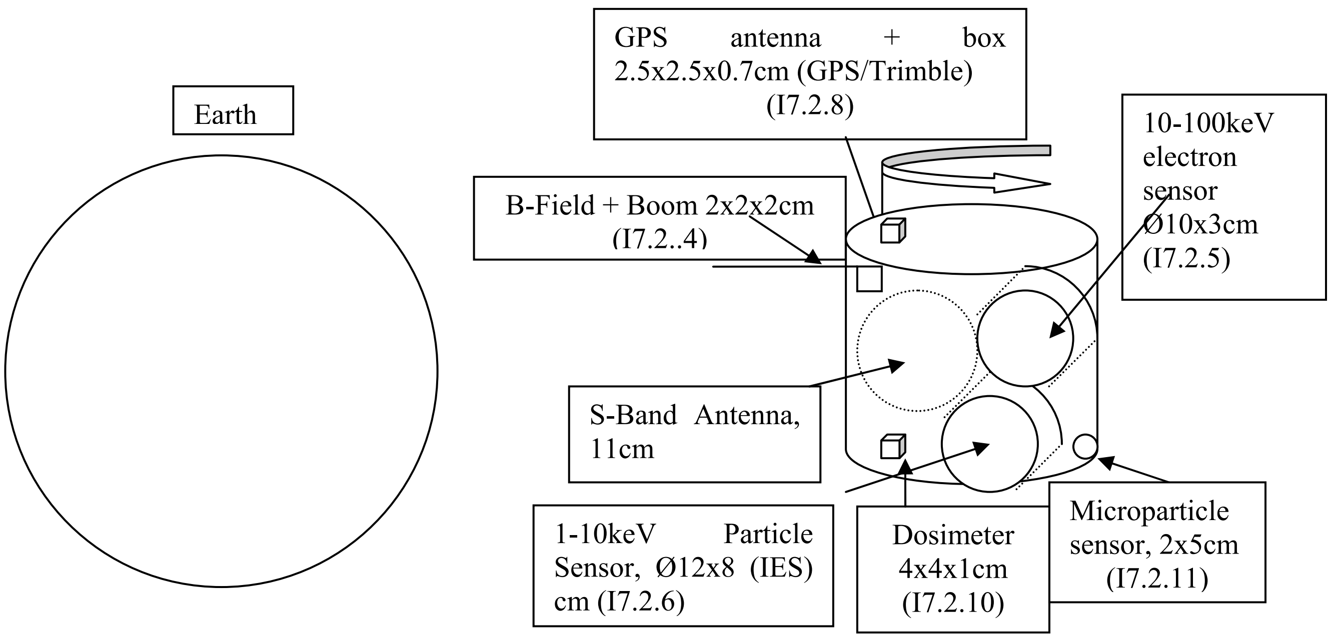


Figure 24. Sketch of GTO spacecraft. Codes I7.2.z indicate that the instrument is discussed in section 7.2.z.

12.7 Use of MNT

Here we summarise how MNT can be utilised in the GTO mission

Table 39. MST mapping to Spacecraft Subsystems

Spacecraft Subsystem	Impact and Relevance of MEMS as a Nanosatellite technology	Issues for using MEMS in nanosats in this mission?	Detail
Space systems engineering	System level knowledge of implementing MEMS into a larger system (e.g. packaging and interfaces). Multifunctionality and smart structures	Yes	Mass Manufacture. Introduction of MEMS culture into space industry. Product Assurance.
Space environment	Radiation dose effects in MEMS, Atomic Oxygen, charging, thermal, shock, vibration	Yes	Very radiation dose
Propulsion system	Could be MEMS thrusters or MEMS components part of a larger propulsion system	Possibly	MEMS thrusters for AOCS?
Attitude control system	e.g. MEMS gyros, accelerometers and associated electronics	Yes	Electronics mass penalty for Gyros. Accuracy of gyro/rate sensor - drift. See Palmsat example
Power system	MEMS batteries, turbines, generators, seebeck effect, nuclear MEMS. Power supply	No	No MEMS power systems
Thermal control system	MEMS louvres, MEMS temperature sensors	Yes	MEMS thermal sensors envisaged
Command and data system	Integration of computers	Maybe	In longer term
Communications system	MEMS antennae, RF switches, filters etc	Maybe	Possible MEMS antennae
Structural system	Smart/Multifunctional structures, MEMS packaging, PolyMEMS actuators (PMA)	Maybe	Possible embedded instruments/ subsystems
Payloads (sensors)	MEMS pressure, humidity, atmospheric sensors, magnetometers, GPS, particle analysers, electromagnetic fields, mass spectrometers,	Yes	If it is possible to reduce sensor size without compromising the physics
Reliability	Reliability of MEMS devices	Yes	Affects overall nanosat reliability

12.8 Summary

Table 40. GTO Constellation Mission Summary

Mission Objective	Operational Space Weather programme • To provide near-real time monitoring of the Earth rad belt environment at GTO	
Payload	The instruments: (Solution 2 Total budget 1.7kg, 3W by 2010) • Magnetometer • 1-10keV electron detector • 10-100keV high energy electron detector • Dosimeter • GPS Antenna • Microparticle Dust Sensor	
Launcher	• 16 Nanosatellites on one stack using FalconI to each of the four orbit apses (i.e. 64 nanosats in total) • Performance: Total launch Mass 620kg to a 500km circular orbit. ~200 kg of useful nanosat mass to GTO orbit	
Spacecraft		**Baseline**
	Design Lifetime	5 years with 80% reliability (min 10 year programme)
	Attitude control	Spin-stabilised at 5 rpm
	Total mass	10 kg per nanosat: 93kg for empty hive, 200kg nanosat mass capability, 327kg hive bipropellant
	Spacecraft y dimensions	Cylindrical (257 mm diameter, 257 mm height per S/C)
	Pointing Requirements	None
	Solar array	Triple Jun GaAs, max 0.066 m^2
	Power	Estimated 13W Consumption
		Li-Ion battery 28.3 Wh depending on orbit plane Eclipse 2.18hours max
		~max 25.9W BOL normal to sun
	Propellant	0.32kg (Hydrazine assumed)
	Downlink Antennas (S-Band)	Semi-Omni antenna 0.11m, 0.5W output power Required Eb/No assumed 2.77
		5.33 kbps per satellite
Mission	**Orbit**	GTO
	Programme period	10 years
	No. of Constellation Cycles	4
	DV	152.6m/s Total 15.8m/s Orbit Phasing of constellations 67.5m/s EOL de-orbit
Operations	**Ground stations**	Min 7 equatorial equally spaced in longitude
		2m Antenna Okay (3dB margin, more if only data from fraction of spin allowed)
		LEOP using ESA LEOP ground-stations?
	Timeliness	5 minutes (i.e. Real time)
Programmatics	**Phase A start** **Phase B start**	2007 2008
	Launch date	2012
Risk	**Maturity of technology**	In use or soon to be demonstrated
	Expected reliability of individual nanosatellite	0.8 after 5 years
	Expected reliability of constellation	0.8 after 2.94 years

13 Mission 5 – SWARM-type constellation

This constellation will make in-situ measurements of the magnetospheric magnetic field using a constellation of 30 spacecraft spread over six orbital planes (i.e. a minimum of 5 satellites in each plane). The orbit planes cover the full 24 hours of local times, a range of apogees, and also a polar orbit. This constellation is termed SWARM to reflect its heritage from the SWARM magnetospheric proposal developed by Schwartz et al [R43]. This is different to the SWARM mission now being built as part of the ESA Earth Observation programme; the latter is a much smaller constellation and will fly in low Earth orbit.

This constellation does not require a real-time down link as the aim is to collect data for improved modelling of the magnetospheric field (which is a key requirement for improving radiation belt and other space weather models). Data downlink can be done using a network of equatorial ground stations, similar to that used for the GTO constellation, supplemented by a high-latitude station (e.g. Svalbard) to downlink of data from the polar orbiting spacecraft. During the development of this mission concept the number of satellite per orbit plane was increased to 10 (i.e. 60 in total) to provide adequate redundancy. The satellites will be spin-stabilised at 12 rpm. The Up/Down link will transmit on S-band. The power supply unit consists of Li-Ion batteries and high efficiency Dual Triple Junction solar cells. Each satellite has a mean power consumption of 13W and a mass of 10 kg.

13.1 Review of requirements & instrument solutions

The requirements and instrument solutions for this mission are shown in Table 14 and Table 18 respectively. However, analysis of these inputs showed that the requirements can be satisfied by nanosat in the 10 kg range. In the near- and mid-term (2005/10) the estimated payload mass is 1 kg and the power budget is 1.5 to 2W. These are readily accommodated on a 10 kg nanosat. In the long term (2020) we look to MEMS magnetometer technology becoming capable of measuring nanotesla magnetic fields. This would enable a major reduction in payload resource requirements- a payload mass of a few grams and power consumption of 0.1 W. Thus in long-term we should consider use smaller spacecraft in the picosat class. However, thus must traded against the use of a down-link antenna and power output of sufficient size to allow use of small and cheap 2m ground antennae. Therefore a larger 10kg-type (or at least dimensionally) nanosat is preferred as the baseline.

13.2 Choice of orbit

This mission requires 5 equatorial orbit planes as shown in Figure 25 below. These five planes have their major axes placed at intervals of 6 hours in local time (thus 2 are co-aligned). The last two cases have orbit sizes of 2.5 x 12 Re and 2.5 x 30 Re. The other three cases have size of 2.5 x 20 Re. In addition, the mission requires a polar orbit (inclination of 90°) with size 2.5 x 15 Re. The sampling of the equatorial region at four longitudes provides simultaneous local time coverage the variability of the magnetospheric field, while the sampling of one longitude with two differently-sized orbits provides simultaneous radial coverage of that variability. The polar orbit provides sampling of variability at high latitudes. The common perigee height of 2.5 Re allows use of GPS to derive spacecraft position and velocity around perigee and thus the determination of orbit parameters.

Since this mission will provide data for improving space weather models (solution level 2 of Table 12), real-time downlink is not required. Instead we aim to downlink of all data within 1 day of their recording. Given the data rate of 0.2kbps per spacecraft continuous around the orbit, downlink can be done using a network of at least seven small ground stations spaced equally around equator – similar to that proposed for the GTO constellation.

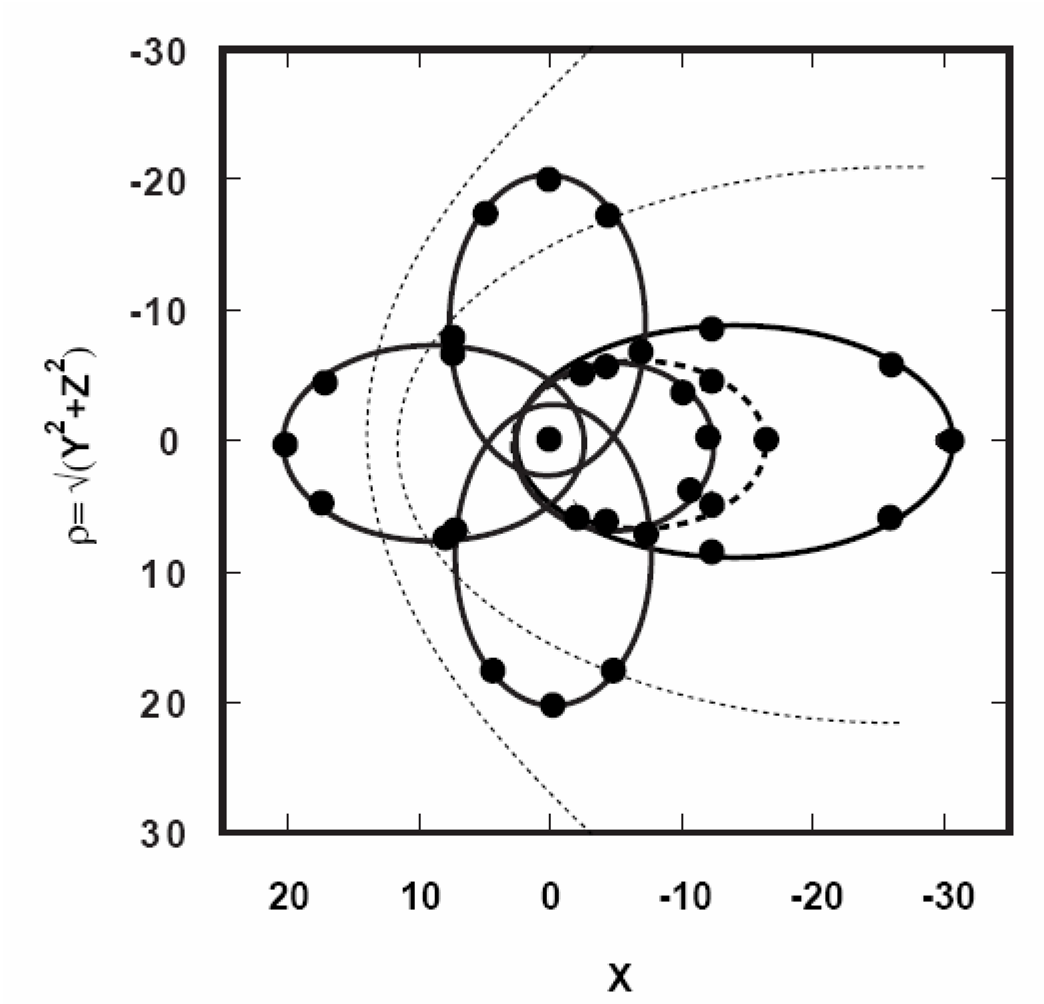


Figure 25: The SWARM orbit: five in the equatorial plane (solid ellipses) and one at high inclination (dotted ellipse).

To ensure adequate measurement redundancy we need more than five spacecraft per orbit plane; we recommend to provide 1 redundant spacecraft per active spacecraft. Finally we note that the SWARM orbits are subject to eclipses up to 3.7 hours long and to relatively high radiation doses (as they pass through the Earth's outer radiation belts and, under some conditions, the inner belts). Thus the spacecraft require adequate thermal blankets and heating to maintain vital electronics for the duration of the mission. Electronic components, particularly spacecraft-critical ones, will require adequate protection and radiation-hard specifications to complete the mission. No radiation modelling was performed in this study, but results from the STRV series of UK micro-satellites demonstrate that spacecraft such as SWARM can easily survive in this environment. But the design must be tolerant to radiation-induced faults such as latch-up.

13.3 Launch strategies

The launch of this constellation is similar to that of the GTO constellation, i.e. we have to place the spacecraft in several highly elliptical orbits distributed in local time. Thus we focus on a similar solution: single or multiple launches into an intermediate low Earth orbit followed by apogee-raising manoeuvres. A deltaV of 3157 m/s is required for boosting the equatorial orbits; the polar orbit boost requires 3614 m/s as it includes a plane change.

We therefore consider just one option: launch (either single or multiple) to LEO then boost to GTO using a carrier spacecraft. For a single launch we require dispersion in local time using a carrier spacecraft. This again exploits the rapid precession in local time available in LEO [R15]; even in the worst case, the carrier spacecraft has to survive only for 5 weeks. For multiple launches we can use separate launches to each local time and thus very cheap carriers (effectively just something that carries the nanosats during launch and deploys them upon orbit acquisition.).

In this last case we require 10 nanosats per carrier spacecraft (hive), so therefore each hive must carry 100kg of nanosats. But if we have a single launch, that must populate all six orbit planes. So we require 6 separate hives sharing 600kg of nanosats. The cost of each hive cost is estimated as (a) 3M€ with no propellant and instant deployment after injection, (b) 5M€ with propellant and instant deployment, and (c) 10M€ with propellant and a significant time in orbit before deployment. It is also assumed that the empty hive mass is 15% of the overall launch mass.

These options were then assessed in terms of cost, time delay to reach operational orbit and sourcing policies. This led us to the following options:

- First choice is a single **Dnepr** launch of 6 hives to LEO, followed by a boost of each hive to one of the SWARM orbits. This can carry 110kg per hive of 10 nanosats and also has adequate fairing volume. The major downside is that it is non-European. Estimated cost is 68M€.
- Second choice is six **Falcon1** launchers to LEO, each followed by a boost of its hive to one of the SWARM orbits. This can carry 136kg per hive of 10 nanosats (105kg for the polar hive) and also has adequate fairing volume. There is a significant risk spreading with 6 separate launches. The major downside with Falcon1 is that it is non-European. Estimated cost is 71.4M€.
- Third is a single **Vega** launch of 5 hives to an equatorial LEO orbit plus a single Falcon1 launch to the polar LEO orbit. The Vega is capable of carrying 95kg per hive of 10 nanosats; this is slightly under the 100kg target but close enough to be retained at this stage. The big advantage of Vega is that it is European. Estimated cost is 81.9 M€.

Thus the launch strategy is dependent on sourcing policies for launch vehicles. Discussions with ESA during the course of the present study indicated that the launcher does not fall within the study requirement to consider European-only hardware. Thus we conclude that the use of a non-European launcher is a realistic option for the Swarm mission and can reduce costs.

13.4 Transfer, deployment and DeltaV analysis

This mission involves an injection into an intermediate LEO orbit, separate precession in local time for the orbit of each hive (precession periods up to 5 weeks) and finally manoeuvres of the hive spacecraft to boost apogee and perigee into the operational orbit. In this case the main requirements for delta-V on board the spacecraft are:

1. Orbit phasing, i.e spreading the spacecraft equally around the orbit. To obtain this spread within one month on the largest orbits (30 Re apogee), we require drift start and stop manoeuvres each with deltaV up to 21.8 m/s . Therefore the overall deltaV is 43.6 m/s. Some of the drift start deltaV (typically about 0.3 m/s) can be provided by the deployment mechanism from the hive spacecraft. To be conservative we ignore these potential deltaV gains.
2. Drag compensation and inclination correction. This is negligible. The drag is very small for a 10 kg spacecraft at 1000 km and the Inclination correction is zero for noon-midnight and dawn-dusk orbits.
3. De-orbit at end-of-life. Not required as the perigee height is higher than 2000km.

We conclude that the overall deltaV requirement on-board the nanosats for the SWARM orbit is about 44 m/s and is dominated by the requirement for orbit phasing.

13.5 Replacement strategy

Table 41. SWARM constellation replacement strategy

Reliability	Replacement period (years)	Replacement cycles in 10 year	Total nanosats needed in 10 years
80% over 2 years	1.6	7	420
80% over 5 years	3.99	3	180

13.6 Systems Analysis

Table 42. Systems analysis for SWARM constellation

Item	Solution
Propellant/propulsion	Hydrazine systems with an Isp of 214s; propellant mass required for a 10kg nanosat in a SWARM orbit is 0.21 kg. This certainly appears feasible.
Attitude and orbit control	A 12rpm spin-stabilised system will be used.
Power generation and management	The maximum power generated at beginning of life is estimated to be almost 26W. An estimate of the power required by the nanosat is 13W.
Power storage, eclipses	Need battery capacity of 47.7 WHr, using a 13W requirement and a max 3.67 hour eclipse.
Thermal constraints and control	SWARM is not considered to be a problem orbit, e.g. see the SWARM proposal to the ESA F2 call in 2001 [R43]. [it was agreed at the Feb 2005 workshop to just consider the qualitative thermal issues]. High-apogee orbits can suffer long eclipses, so adequate thermal blankets and heating are needed to maintain vital electronics for the duration of the mission.
Environment	The SWARM environment is very harsh as it passes through the Earth's outer radiation belts (and inner belts under some conditions). Critical electronic components require adequate protection and radiation-hard specifications. The design must be tolerant to faults such as latch-up stemming from radiation.
Configuration	A spin stabilized cylindrical spacecraft is assumed with a 363 mm diameter and 182 mm height and with two equal and opposite booms in the middle of the cylinder wall. See Figure 27. The instrument dimensions are derived from section 7. This configuration is different (smaller in mass and size) to that in the SWARM F2 concept [R43] as the current measurement requirements are less demanding.
Communications Requirements and strategy	Data can be stored on-board and downlinked during late ground station contact (within 24 hours of data acquisition). Need network of seven ground stations at equatorial latitudes, plus one at high latitude. Aim at dedicated network of a small 2m ground stations with routine operation being unstaffed. This should minimise costs. 18cm spacecraft antenna working at S-Band and transmitting only when Earth-pointing. Needs output power of 1 W and thus input power of 10W. Can support downlink up to 200 bps. Limited need for uplink especially in the longer term.

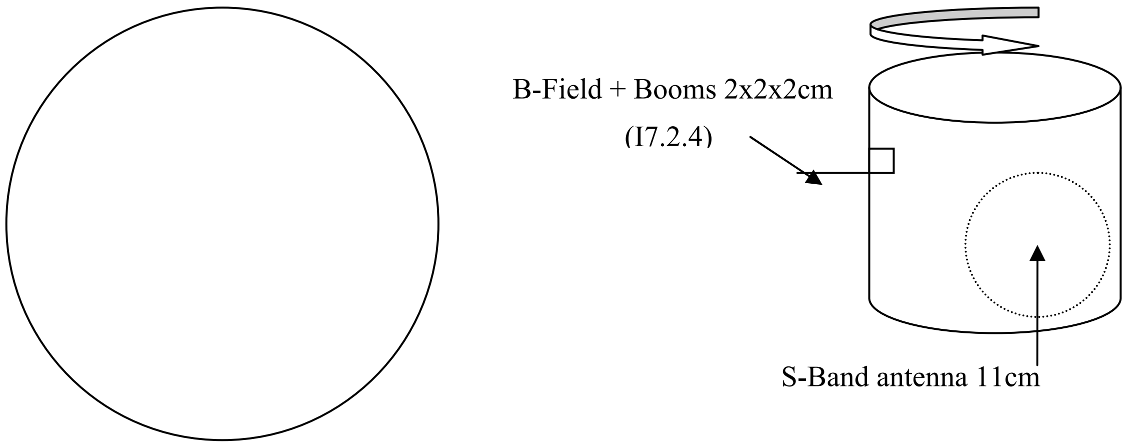


Figure 27. Sketch of SWARM spacecraft. Codes I7.2.z indicate that the instrument is discussed in section 7.2.z.

13.7 Use of MNT

Here we summarise how MNT can be utilised in the SWARM constellation mission

Table 43. MST mapping to Spacecraft Subsystems

Spacecraft Subsystem	Impact and Relevance of MEMS as a Nanosatellite technology	Issues for using MEMS in nanosats in this mission?	Detail
Space systems engineering	System level knowledge of implementing MEMS into a larger system (e.g. packaging and interfaces). Multifunctionality and smart structures	Yes	Mass Manufacture. Introduction of MEMS culture into space industry. Product Assurance.
Space environment	Radiation dose effects in MEMS, Atomic Oxygen, charging, thermal, shock, vibration	Yes	Very high radiation dose
Propulsion system	Could be MEMS thrusters or MEMS components part of a larger propulsion system	Possibly	MEMS thrusters for AOCS?
Attitude control system	e.g. MEMS gyros, accelerometers and associated electronics	Yes	Electronics mass penalty for Gyros. Accuracy of gyro/rate sensor - drift. See Palmsat example
Power system	MEMS batteries, turbines, generators, seebeck effect, nuclear MEMS. Power supply	No	No MEMS power systems
Thermal control system	MEMS louvres, MEMS temperature sensors	Yes	MEMS thermal sensors envisaged
Command and data system	Integration of computers	Maybe	In longer term
Communications system	MEMS antennae, RF switches, filters etc	Maybe	Possible MEMS antennae
Structural system	Smart/Multifunctional structures, MEMS packaging, PolyMEMS actuators (PMA)	Maybe	Possible embedded instruments/ subsystems
Payloads (sensors)	MEMS pressure, humidity, atmospheric sensors, magnetometers, GPS, particle analysers, electromagnetic fields, mass spectrometers,	Yes	If it is possible to reduce sensor size without compromising the physics
Reliability	Reliability of MEMS devices	Yes	Affects overall nanosat reliability

13.8 Summary

Table 44. SWARM Constellation Mission Summary

Mission Objective	Operational Space Weather programme • To provide non-real time monitoring of the Magnetospheric Magnetic field environment in SWARM orbits	
Payload	The instruments: (Solution 2 Total budget 1kg, 1.5W by 2010) • Magnetometer	
Launcher	• 1 single Dnepr launch with 6 hive spacecraft to each orbit plane, carrying 10 Nanosatellites per hive • Performance: 575kg to 300x500km orbit at 50.5deg inc. ~110 kg of useful nanosat mass to worst SWARM orbit (30Re apogee)	
Spacecraft		**Baseline**
	Design Lifetime	5 years with 80% reliability (min 10 year programme)
	Attitude control	Spin-stabilised at 12rpm
	Total mass	10 kg each nanosat 575kg Total Hive mass (110kg Nanosat capability) 86kg for empty hive 379kg Hive propellant
	Spacecraft main body dimensions	Cylindrical (257 mm diameter, 257 mm height per S/C)
	Pointing Requirements	None
	Solar array	Triple Jun GaAs, max 0.066 m^2
	Power	Estimated 13W Consumption
		Li-Ion battery 47.7 Wh depending on orbit plane Eclipse 3.67hours max
		~max 25.9W BOL normal to sun
	Propellant	0.21kg (Hydrazine assumed)
	Downlink Antennas (S-Band)	Semi-Omni antenna 0.11m, 0.5W output power Required Eb/No assumed 2.77
		0.2 kbps per satellite
Mission	**Orbit**	SWARM orbits
	Programme period	10 years
	No. of Constellation Cycles	3
	DV	43.6m/s Total 43.6m/s Orbit Phasing of constellations
Operations	**Ground stations**	Min 7 equatorial equally spaced in longitude One at Svalbard
		2m Antenna Okay (3dB margin, more if only data from fraction of spin allowed)
		LEOP using ESA LEOP ground-stations?
	Timeliness	24hours (i.e. Non-Real time)
Programmatics	**Phase A start** **Phase B start**	2007 2008
	Launch date	2012
Risk	**Maturity of technology**	In use or soon to be demonstrated
	Expected reliability of individual nanosatellite	0.8 after 5 years
	Expected reliability of constellation	0.8 after 3.99 years

14 Mission 6 – L1 mission

This mission will make in-situ measurements of the solar wind density and velocity, vector magnetic field and high energy (>MeV) particle fluxes) near the L1 Earth-Sun Lagrangian point. This point is approximately 1.5 million kilometres sunward of the Earth. The solar wind and its embedded magnetic field cross this region 20 to 60 minutes before arrival at Earth (the travel time varies with solar wind speed). Measurements at L1 can therefore give a small but useful advance warning of the arrival at Earth of coronal mass ejections and other heliospheric disturbances.

The majority of space weather measurements at L1 require a real-time down link. A network of at least 3 large 15m class ground stations is required to provide this. Only one nanosatellite is required. The satellites will be spin-stabilised between 5-12rpm. The Up/Down link will transmit on S-band. The power supply unit consists of Li-Ion batteries and high efficiency Dual Triple Junction solar cells. The satellite is assumed to have a mean power consumption of 13W during normal operation and have a mass of 10 kg.

14.1 Review of requirements & instrument solutions

The requirements and instrument solutions for this mission are shown in Table 14 and Table 18 respectively. Analysis of these inputs showed that the requirements can be satisfied by nanosat in the 10 kg range. The estimated payload mass for the full set of requirements is 4.8 kg in 2005, falling to 3.4 kg in 2010 and 2.95 in 2020. Similarly the power budget is 8 W in 2005, falling to 6 W in 2010 and 5.5 W in 2020. In the medium to long-term these are readily accommodated on a 10 kg nanosat.

14.2 Choice of orbit

This mission requires that a single nanosat be injected into a halo orbit around the L1 point. The requirement for real-time data (0.34kbps per spacecraft) can then satisfied if we use a network of large low latitude ground stations (at least 3 sites in the network).

The single spacecraft is sufficient to meet our reliability requirements at any one time. We do not need more spacecraft for measurement redundancy, so long the spacecraft is replaced at regular interval (such that the spacecraft failure rate is low enough to meet our reliability requirement). The L1 point is not subject to eclipses but can receive significant radiation doses in the form of solar energetic particle events.

14.3 Launch strategies

This mission requires that the spacecraft be launched onto a transfer orbit from Earth to the vicinity of the L1 point. On arrival at that location, a manoeuvre is needed to insert the spacecraft into a halo orbit around L1. We assume that the insertion is carried out by a carrier spacecraft (hive). Thus this is essentially a normal spacecraft with power, etc, which must survive until the nanosat is inserted into the halo orbit (3 to 12 months depending on the launch option). Since we require only a single nanosat at L1, the mass requirements on the hive are modest.

We considered four main options launch for launch into the transfer orbit:

- Direct to L1 using either just the launcher or the launcher plus a provided propulsion module (e.g. ST1 by Dnepr or Star 37 for Rockot). This is expensive, but does not rely on a primary spacecraft. The carrier must survive for up to 3 months.
- Launch to GTO as primary payload, then boost to L1 using a carrier spacecraft. This is expensive, but does not rely on a primary spacecraft. The carrier must survive for up to 3 months.
- Launch to GTO as secondary payload, then boost to L1 using a carrier spacecraft. The initial launch is then cheaper, but launch conditions are determined by the primary spacecraft. Thus this option requires a more capable carrier spacecraft, which must survive for up to 12 months or have extra deltaV (up to ~300m/s) to reduce the transfer time.

- Launch to LEO then boost to L1 using either a provided propulsion module or a carrier spacecraft. If a propulsion module is provided (e.g. by Dnepr or a Star for Rockot) then no hive spacecraft is required. Transfers from LEO require much higher deltaV's to L1 (over 3km/s). Launches from equatorial orbits may have an additional deltaV penalty (of several hundred m/s) to launch into an L1 halo orbit. If a hive spacecraft is required it must survive for up to 3 months

These options were then assessed in terms of cost, time delay to reach operational orbit and sourcing policies. This led us to the following options:

- The first choice is launch as secondary payload (Auxiliary passenger or ASAP) on an **Ariane 5** launch to GTO, followed by a carrier spacecraft boost to L1. The majority of Ariane 5 GTO launches have apogee in the sunward direction which should facilitate apogee boost onto a transfer orbit to L1. However, as noted above, this cannot be guaranteed. Thus we have to add propulsion (deltaV up to 300 m/s) to reduce the transfer time back to 3 months. This option has adequate fairing volume for the mission and can launch a total mass of 120 kg comprising 18 kg hive and 34kg fuel. This leaves 68kg per hive for nanosats, which gives plenty of margin. Estimated cost is ~12M€ (2M€ for ASAP and 10M€ for the hive spacecraft).
- The second choice is launch to equatorial LEO as primary payload on a **Falcon1,** followed by a carrier spacecraft boost to L1. This can launch a total mass of 620 kg, comprising 93 kg hive and 389kg fuel. This leaves 138kg per hive for nanosats, which gives plenty of margin. Estimated cost is ~16.9M€, inc. hive spacecraft. Estimated cost is ~16.9M€ (6.9M€ for the Falcon and 10M€ for the hive spacecraft).

14.4 Transfer, deployment and DeltaV analysis

This mission involves launch onto a transfer orbit to L1 and then injection into a halo orbit around L1. In this case the main requirements for delta-V on board the spacecraft are:

1. Injection into the halo orbit: a deltaV of 5 m/s is also required from the nanosat
2. Orbit phasing, not applicable.
3. Drag compensation and inclination correction, not applicable.
4. De-orbit at end-of-life, not applicable.
5. Orbit maintenance. A deltaV of 2 m/s per year (for 2 years) is required for station keeping, to maintain the halo orbit

We conclude that the overall deltaV requirement is about 9 m/s and is split almost evenly between initial acquisition and subsequent maintenance of the operational halo orbit around L1.

14.5 Replacement strategy

Table 45. L1 mission replacement strategy

Reliability	Replacement period (years)	Replacement cycles in 10 year	Total nanosats needed in 10 years
80% over 2 years	2	5	5
80% over 5 years	5	2	2

14.6 Systems Analysis

Table 46. Systems analysis for L1 mission

Item	Solution
Propellant/propulsion	Hydrazine systems with an Isp of 214s; propellant mass required for a 10kg nanosat in an L1 halo orbit is 0.04 kg. This is a very low mass and a very small propulsion system may be applicable assuming the thrust is high enough for the mission.
Attitude and orbit control	A 12 rpm spin-stabilised system, using sun-sensors.
Power generation and management	The maximum power generated at beginning of life is estimated to be almost 26W.
Power storage, eclipses	Battery use not studied as this orbit has no eclipses.
Thermal constraints and control	L1 is not considered to be a problem orbit as it is thermally stable, e.g. see the ESA CDF study [R15]. [it was agreed at the Feb 2005 workshop to just consider the qualitative thermal issues].
Environment	The L1 environment is usually benign, but is exposed to solar energetic particle events, so sensitive components need to be hardened or shielded for such events. Transfer orbits to L1 spend the majority of their time outside of the harmful radiation belts.
Configuration	A spin stabilized cylindrical spacecraft is assumed with a 363 mm diameter and 182 mm height and with two equal and opposite booms in the middle of the cylinder wall. See Figure 28. The instrument dimensions are derived from section 7.
Communications Requirements and strategy	Data must be downlinked in real-time. Need network of at least three ground stations at low latitudes to maintain continuous view of L1. Low data rates mean that standard 15m ground stations can be used despite the long link distance (1.5 million km). 18cm spacecraft antenna working at S-Band and transmitting only when Earth-pointing. Gives 53° beamwidth, so no pointing mechanism needed. Needs output power of 1 W and thus input power of 10W. Can support downlink up to 340 bps. Limited need for uplink especially in the longer term.

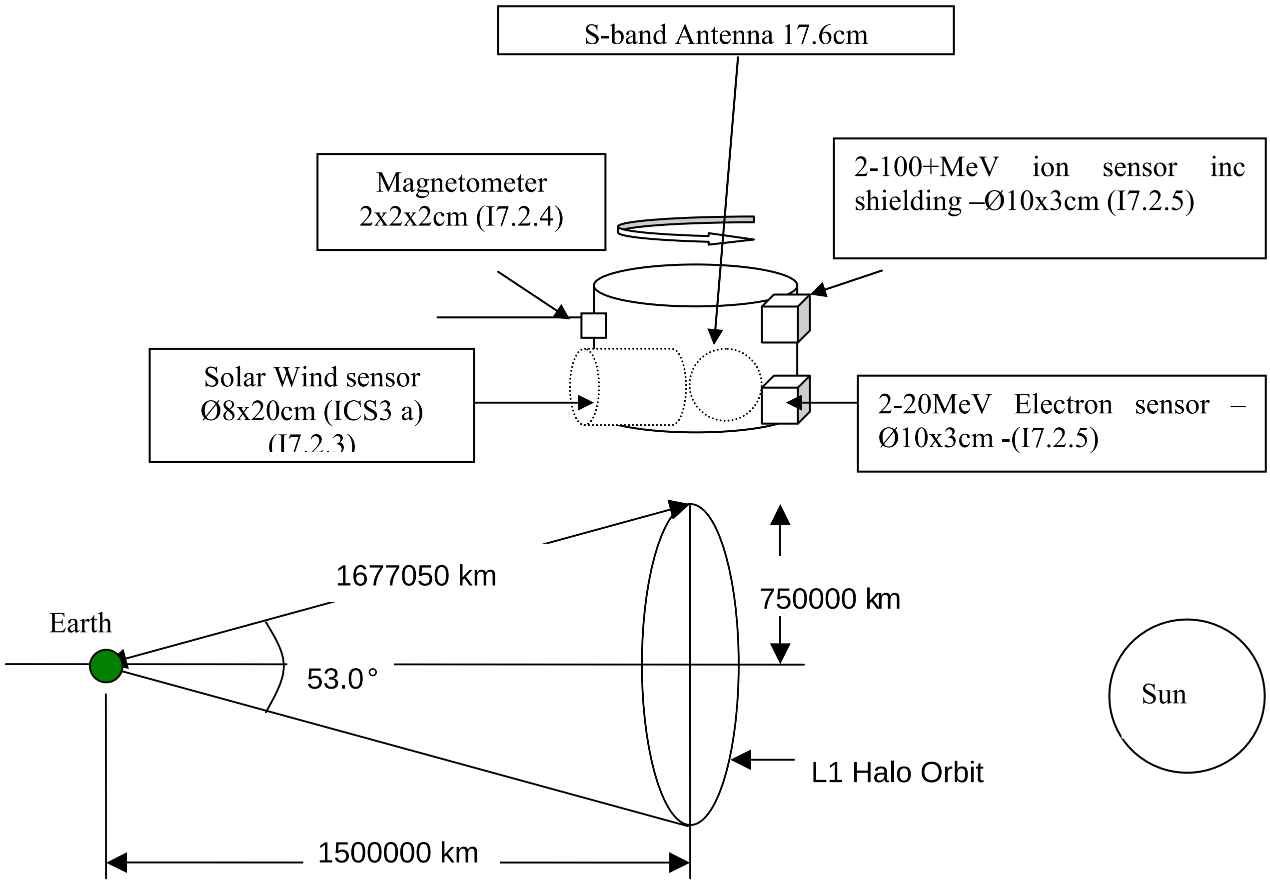


Figure 28. Sketch of L1 mission and spacecraft. Codes I7.2.z indicate that the instrument is discussed in section 7.2.z.

14.7 Use of MNT

Here we summarise how MNT can be utilised in the L1 mission

Table 47. MST mapping to Spacecraft Subsystems

Spacecraft Subsystem	**Impact and Relevance of MEMS as a Nanosatellite technology**	**Issues for using MEMS in nanosats in this mission?**	**Detail**
Space systems engineering	System level knowledge of implementing MEMS into a larger system (e.g. packaging and interfaces). Multifunctionality and smart structures	Yes	Mass Manufacture. Introduction of MEMS culture into space industry. Product Assurance.
Space environment	Radiation dose effects in MEMS, Atomic Oxygen, charging, thermal, shock, vibration	Yes	radiation dose from SPE
Propulsion system	Could be MEMS thrusters or MEMS components part of a larger propulsion system	Possibly	MEMS thrusters for AOCS?
Attitude control system	e.g. MEMS gyros, accelerometers and associated electronics	Yes	Electronics mass penalty for Gyros. Accuracy of gyro/rate sensor - drift. See Palmsat example
Power system	MEMS batteries, turbines, generators, seebeck effect, nuclear MEMS. Power supply	No	No MEMS power systems
Thermal control system	MEMS louvres, MEMS temperature sensors	Yes	MEMS thermal sensors envisaged
Command and data system	Integration of computers	Maybe	In long term
Communications system	MEMS antennae, RF switches, filters etc	Maybe	Possible MEMS antennae
Structural system	Smart/Multifunctional structures, MEMS packaging, PolyMEMS actuators (PMA)	Maybe	Possible embedded instruments/ subsystems
Payloads (sensors)	MEMS pressure, humidity, atmospheric sensors, magnetometers, GPS, particle analysers, electromagnetic fields, mass spectrometers,	Yes	If it is possible to reduce sensor size without compromising the physics
Reliability	Reliability of MEMS devices	Yes	Affects overall nanosat reliability

14.8 Summary

Table 48. L1 Mission Summary

Mission Objective	Operational Space Weather programme • To provide near-real time monitoring of the environment at L1	
Payload	The instruments: (Solution 2 Total budget 3.4kg, 6W by 2010) • Particle sensor (for Solar wind bulk velocity and density) • Magnetometer (for Heliospheric magnetic field) • Particle sensor (for 2-100MeV and >100 MeV ions from heliosphere) • Particle sensor (for 2-20MeV electrons from heliosphere)	
Launcher	• 1 Nanosatellite in a hive spacecraft/propulsion module as an ASAP 5 “Auxiliary passenger” to GTO • Performance: 120kg to GTO. ~68 kg of useful nanosat mass to the L1 orbit	
Spacecraft		**Baseline**
	Design Lifetime	5 years with 80% reliability (min 10 year programme)
	Attitude control	Spin-stabilised at 5-12rpm with sun-sensors
	Total mass	10 kg each nanosat 18kg for empty hive, 34kg hive bipropellant Total Hive mass 120kg
	Spacecraft main body dimensions	Cylindrical (257 mm diameter, 257 mm height per S/C)
	Pointing Requirements	None
	Solar array	Triple Jun GaAs, max 0.066 m^2
	Power	Estimated 13W Consumption
		Li-Ion battery TBD Wh No Eclipses
		~max 25.9W BOL normal to sun
	Propellant	0.04kg (Hydrazine assumed)
	Downlink Antennas (S-Band)	Semi-Omni antenna 0.176m, 1W output power
		0.34 kbps per satellite
Mission	**Orbit**	L1
	Programme period	10 years
	No. of Constellation Cycles	2
	DV	9m/s Total 5m/s Orbit Acquisition 4m/s Station-keeping
Operations	**Ground stations**	Goal: Min 3 low inclination 15m antennas relatively equally spaced in longitude Maspalomas, Perth, Kourou
		15m Antenna (6.67dB margin, more if only data from fraction of spin allowed)
		LEOP using ESA LEOP ground-stations?
	Timeliness	2 minutes (i.e. Real time)
Programmatics	**Phase A start** **Phase B start**	2007 2008
	Launch date	2012
Risk	**Maturity of technology**	In use or soon to be demonstrated
	Expected reliability of individual nanosatellite	0.8 after 5 years
	Expected reliability of constellation	0.8 after 5 years

15 Prospects and Recommendations

15.1 Methodology

A key output of this study is to review the prospects for implementing the solutions discussed above and making recommendations on key issues to be addressed in order to achieve those prospects. These prospects and recommendations were developed through a small workshop held at ESTEC on 6/7 October 2005. This involved members of the study team, the ESA staff directing the study and a number of invited European experts with a range of expertise covering space weather, advanced space plasma instrumentation and nanosats. The workshop started with a series of presentations to set the context by presenting results from the early phases of the study. This was followed by an extensive brainstorming session covering relevant topics including nanosats reliability, communications, spacecraft architectures, launch and de-orbit issues, power systems, payload and autonomy. An overarching theme was the generic problem of using MNT in space. This was reviewed at end of the workshop in order to pick up on relevant discussion under the various specialist topics above.

The workshop discussion was summarised as a mindmap, i.e. a graphical representation of a set of ideas [R29]. This allows the author to show links, especially hierarchical links, between ideas. It is a powerful but concise way to summarise meeting outputs. The mindmap was circulated to meeting participants and updated in response to comments received. The final version is shown in section 24 of this report.

The mindmap was analysed to identify key issues to be discussed and to group them into sections as follows:

- Use of MNT. This is an extensive section because of the importance of MNT to future nanosat development. A key issue is the use of both COTS and bespoke devices and the implications of the two approaches for testing. Another issue is multi-functionality and its implications for the design environment. This section also addresses issues such as radiation tolerance and thermal design.
- Other new technologies. This discusses the use of wireless communications and the use of autonomy.
- Payload. This focuses mainly on the issue of instrument miniaturisation including limitations on sensor size and the need to refine measurement requirements where this supports miniaturisation.
- Communications. This is a major section because of the importance of real-time communications for space weather applications. It discusses how space communications capability now lags behind the enormous growth in terrestrial capability arising from the commercial growth of the internet. Another important issue here is the need to compress data and the potential to do so by on-board processing to physical parameters.
- Launch and orbits. This focuses on options for nanosat deployment, orbit maintenance and de-orbit – and on the continuing importance of good orbit determination.
- Replacement strategy. This discusses what reliability means in the context of the constellations that are ubiquitous in space weather applications. This is important issue where there is a dependency between adjacent spacecraft, e.g. for communications or measurement resolution, and thus a need to develop failure models that incorporate that dependency.
- Programmatics. This discusses the organisational and political issues that can drive a space weather programme.

These issues are discussed in detail in the following sections (15.2 to 15.8). Key recommendations are highlighted by shaded boxes within the text. They are also consolidated into a table in section 15.9.

15.2 Use of MNT

The use of MNT/MEMS-based devices on spacecraft is a significant new development. It offers the potential to fly smaller and lighter devices thus increasing the functionality that can be placed in orbit for a given mass budget. Given the criticality of mass budgets, MNT/MEMS technology has significant potential to improve what can be done on spacecraft of all sizes, but especially so for nanosats. The table below provides a summary of some key MNT devices types as discussed at the workshop. For more details see the Micro and Nano Technology Review in section 4.2 above.

Table 49. Table of MNT device types

Device type	Status
Microthrusters	Space applications under development, expected to be ready for flight by 2010.
Accelerometers	Strong mass-market drive for automotive applications.
Gyros	Mass-market interest for navigational systems
Attitude control	Magnetometers and angular rate sensors in development
Star and sun sensors	Active pixel sensors - devices available in Europe
Communications systems	Strong mass-market drive for MNT-based RF devices to support mobile communications, including resonators, switches, phase-shifters, antennae.
Power	Wide range of solutions under development: advanced solar cells, thermo-generators, nuclear batteries, fuel cells.
Mass memory	The original MNT device
Micropacks	Packaging techniques are a major focus for development work

However, the effective application of MNT to space demands a careful and realistic assessment of what it can do in any particular case. Key issues to consider are:

1. The low mass and volume of MNT/MEMS devices facilitates use of multiple devices on any spacecraft, even a nanosats. This aids redundancy. Where MEMS sensors are used they can improve measurement accuracy and reliability by statistical averaging and rejection of outliers.

2. The industrial development of MNT/MEMS is largely driven by mass-market applications such as mobiles and cars. We can reasonably anticipate that the market drive will result in the availability of COTS devices at low/modest cost. However their application to space will require assessment and testing. The reliability of MNT/MEMS devices in space will vary markedly depending on the design of each device and the susceptibilities of its components to space environment hazards such as radiation and charging. For example CMOS-based devices often have significant susceptibility to radiation and thus special care is required to make such devices fit for use in space. In contrast MEMS devices built from silicon have expected to be robust to radiation damage (though their susceptibility to charging is an open issue). Thus it is clear that COTS devices built on MNT/MEMS technology will need qualification for use in space. There is a need a methodology for such qualification. It should include:
   - Theoretical assessment – based on existing knowledge of how the space environment affects the materials and designs used in the devices. This is an essential first step that should guide the formulation of practical tests. The power of theoretical analysis will grow as more knowledge is acquired and thus there is also a need to capture that new knowledge.
   - On-ground simulation – based on exposure of sample devices to a simulated space environment. This is an important step towards flight testing. It has the advantage of being well-controlled and avoiding the extra costs involved in flight tests. But it is limited as ground tests cannot replicate all aspects of the space environment.
   - Flight testing – based on exposure of sample devices to the actual environment. This provides the ultimate validation and is crucial for aspects that cannot be tested in space. A demonstrator

programme is needed to validate devices for the range of different environments that exist in space, e.g. low Earth orbit (including polar effects), inner and outer magnetosphere, interplanetary space, and with a emphasis on testing in harsh environments to cover the worst case. Flight testing also plays an important role in providing confirmation that is easily presented to decision-makers.
- Standards for documentation of testing to ensure that results are clear and so is the basis on which they were obtained. Independent testing with open dissemination of results will help to build knowledge and confidence.

3. The applicability of MEMS is not limited to COTS devices. MEMS technology can be used to construct bespoke devices for specialist applications. However, these will necessarily be produced as short runs. Thus the unit costs will be higher but that is appropriate to specialist applications. More importantly the short runs will require a different approach to reliability as it will not be possible to select for quality by testing large batches. Bespoke devices will require a different approach to reliability; perhaps putting more stress on theoretical analysis at an early stage and more rigorous production procedures.

Recommendation: Develop a methodology for qualification of MNT/MEMS devices for space. This shall allow tailoring to each case – in particular distinguishing use of COTS and bespoke devices.

4. Sensitivity to radiation is a critical issue for MNT/MEMS. The test programmes described above will assess that sensitivity. But we also need to consider how to respond when that assessment shows sensitivity. The key decision is whether it is possible to improve radiation hardness or whether it would be better to add shielding. For example, temporary interference due to charge deposition can be mitigated by adding parallel redundancy and procedures to reset affected elements. For many cases shielding may be the only answer, but will be subject to a trade-off between mass and risk. The small size of MNT/MEMS will facilitate use of selective shielding. It is important to ensure the availability of suitable design tools for shielding – in particular the availability of Geant4 libraries to support shield design for highly miniaturised systems and the integration of shielding with packaging techniques for MNT/MEMS.

Recommendation: Develop methods to assess and mitigate radiation sensitivity of MNT/MEMS devices.

5. Good thermal design is always vital for space devices. There is a need to assess the implications of different MNT/MEMS packaging techniques for thermal management so that designers and thermal engineers can work together for good solutions.

6. There is a clear technical distinction between MEMS, as an advanced technology for miniaturisation, and what we may call "high-end miniaturisation", i.e. the continuing evolution of older technologies to smaller and smaller sizes. But this distinction is unimportant for space weather applications. All that matters is the ability of the technology to address the requirements and to work reliably in space. For example the Munin nanosat [R10] contained no MEMS devices and demonstrated that a space weather nanosat can be built using "high-end miniaturisation". Thus the design of any nanosat for a space weather application should exploit both MEMS and high-end miniaturisation - and do a trade-off between the two approaches to find the best solution at that time. It is expected that use of MEMS will increase in the future as the availability and performance of MEMS devices improves, but that use of high-end miniaturisation will continue for many years.

7. A key issue for nanosats is use of multi-functional components, i.e. that particular components may address multiple requirements and thereby reduce the mass budget needed to satisfy those requirements. A simple example is that structural integrity may be provided by integration of instrument and spacecraft and not by either alone. Another example is the use of shared electronics for on-board data processing, i.e. minimising the electronics dedicated to any single component. To exploit such synergies it is essential to establish a system engineering methodology that support integrated ways of working:
   - A design environment that can assess the overall impact of multi-functionality. This will possibly include a greater role for engineering simulation as a way to explore the consequences of multi-functionality before any build phase. A key output from such simulations will be to identify critical

issues that require laboratory or flight testing via physical prototypes. The design environment should certainly be capable of supporting distributed working so that geographically dispersed experts can work together.

- A lower reliance on isolation of components via well-defined and document interfaces. Interface control should still applied where appropriate (e.g. at interfaces external to the spacecraft), but it must be recognised that multi-functionality cannot be fully exploited if there is a strict hierarchical decomposition of systems down to the component level.
- Development of protocols to facilitate the sharing of on-board resources, e.g. shared use of data-processing resources.
- A greater focus on all-up testing and to do so as early as possible, e.g. through the early availability of realistic models.
- Possibly the adoption of prototyping methods in which the overall design is iterated through a series of spacecraft models. This will be facilitated if MNT/MEMS developments can reduce the build cost of individual models and thus it becomes economic to build a series of models.

The scope of multi-functionality must include the payload, so the issues discussed above apply to payload instruments as much as to spacecraft sub-systems.

Recommendation: Develop design environments and standards that are appropriate for multi-functional systems.

15.3 Other new technology

15.3.1 Wireless

The current trend towards wireless connections between devices may be may applicable to data flows on spacecraft. Wireless has several potential advantages over wired connections:

1. It saves some harness mass. But this would need to be traded off against the extra mass of the wireless systems.
2. it will likely facilitate integration since the links would then becomes software configurable items rather than physical structures. The ability to configure such links in orbit would provide some robustness against integration errors and also has potential for redundancy.
3. It may be better for electromagnetic compatibility because the signal could largely be focussed into the wireless frequency band.
4. It would facilitate the use of multiple sensors and inter-sensor links because the addition of links would have little or no mass penalty.

Key constraints on wireless links include:

- The need to qualify wireless technology for space use, e.g. Bluetooth technology, which is currently used in many terrestrial applications (e.g. wireless earpieces) may be difficult to migrate to space because it is based on CMOS devices.
- Availability of power for wireless devices. Harness mass will still be required to distribute power to devices if the spacecraft has a traditional centralised power system. But, in the longer-term, one can forsee the development of distributed power sources for spacecraft use.

We note that there is strong general interest in the development and exploitation of wireless technologies in space. A formal ESTEC-Industry Wireless Onboard Spacecraft Working Group was founded in April 2003 [R25] and there is an inter-space agency activity to coordinate space wireless protocols through CCSDS [R26].

15.3.2 Autonomy

Autonomy is an important issue for reducing the costs of space weather monitoring because it offers the potential to greatly reduce, or even eliminate, the need for command uplink to the spacecraft. Uplink can be a significant cost driver through the need for human effort to prepare and validate command content as well as the planning and operation of the uplink itself. Uplink is usually more demanding of human effort than downlink, e.g. it may require a human presence at the ground station to monitor the correctness of the active transmission.

Many space weather measurements, particularly of energetic particles, require adjustment of instruments in response to spatial changes in the particle environment. A prime example is the radiation belt monitor using an elliptical GTO-like orbit as discussed in measurement requirements 15.1 to 17.1 of this study. Others include the changing particle environment in low earth polar orbits, where energetic electrons (requirement 11.2) are largely confined to crossing of the auroral oval.

There are at least two potential approaches to on-board autonomy for space weather measurements:
Predictive autonomy. This involves the on-board planning of observations using knowledge of spacecraft position to predict the crossing of different plasma regions and thus select the appropriate observing modes. This is essentially the transfer on-board of the planning processes currently performed by ground-based operations centres.
Responsive autonomy. This involves the on-board assessment of current plasma conditions and their interpretation to set appropriate observing modes. This is the automation of the manual control used at ground-based science operations centres where and when real-time spacecraft access was available. This approach has the advantage that it could respond to temporal, as well as spatial, change.

If such approaches can be implemented, their transfer on-board could eliminate the need for routine command uplink and thus significantly reduce the costs of space weather monitoring. It is desirable to retain some uplink capacity, e.g. for updates of on-board software and calibration data. However, this is an occasional activity that can be scheduled well in advance and will not incur major costs.

One critical element in predictive autonomy is on-board knowledge of the spacecraft position. Ideally this might be achieved via on-board orbit determination, e.g. using a spacecraft GNSS receiver if and where the spacecraft is inside the GPS or Galileo constellations. Such receivers are commercially available with low mass suitable for nanosats. But if on-board orbit determination were not available, the fall-back position would be to determine Keplerian elements on the ground via traditional spacecraft tracking and uplink those elements for use by on-board software. This would require only a very modest and occasional uplink (perhaps 100 bytes once per day). Another critical element is accurate on-board knowledge of the current time in a system suitable for orbit calculations, i.e. a calibrated scale such as Coordinated Universal Time and not just clock ticks as generally used on contemporary spacecraft. This will be facilitated by current development of chip-size atomic clocks.

Many requirements for space weather monitoring require measurements over a constellation of spacecraft and not just at a single point, e.g. the requirements for radiation belt monitoring specify measurements at four local times and a range of L-values at each local time. Thus we need to consider whether responsive autonomy should operate at constellation level or just at the level of individual spacecraft. This is important because constellation level autonomy will require inter-spacecraft links and drives important elements of spacecraft design. Constellation-level autonomy will be important for future missions where inter-spacecraft coordination is essential, e.g. formation-flying and differential science observations [R27]. However, for space weather constellations, such coordination is an aid to performance but not essential, e.g. it could be used to adjust the spread of the constellation in response to loss of a spacecraft.

Recommendation: Review the potential of autonomy to simplify SW nanosat operations.

15.4 Payload

The majority of space weather measurements are adaptations of measurements developed to support research on solar-terrestrial physics (STP), e.g. magnetic fields, particle fluxes, solar imagery, etc. But the small size of nanosats is a challenge to accommodation of many conventional STP instruments. To make best use of nanosats it is necessary to address several issues:

- To explore the feasibility of sensor miniaturisation. This is the critical step in instrument miniaturisation and is independent of miniaturisation of the electronics used to control and process sensor signals, which is subject to usual trend to smaller systems. The minimum size sensors is set primarily of their ability to collect and resolve the signal of interest, e.g. to collect enough particles or photons to generate statistically reliable results, to provide adequate spatial or spectral resolution).
- To explore whether the measurement requirements can be simplified such that smaller instruments can provide useful data. For example, a simplified requirement for solar imaging is to locate flares with fairly coarse angular resolution (~100 arcsecs). This is adequate to support nowcasting of solar proton events, but probably not the forecasting of CME launches (where current research suggests we may need high-resolution (≤ 5 arcsecs) imaging to assess the state of the magnetic structures whose later reconfiguration may lead to a CME launch). This kind of simplification – a focusing on selected requirements – can enable a significant reduction in instrument size and mass. Another example of simplification is to descope the payload to eliminate instruments that are difficult to accommodate on a nanosats. This is a straightforward trade-off. If the nanosat approach is to be exploited successfully it is important to focus on what they can do well - and drop what cannot be done well.
- To look for different instrument concepts that allow nanosats solutions where contemporary STP instruments cannot do so. In particular, it may be worthwhile to revisit older instrument concepts from the early days of spaceflight. A possible example is the Faraday cup; this can measure the current carried by a beam of particles and has been applied in space to measure bulk fluxes of solar wind ions; a recent example is on the NASA WIND spacecraft [R28].
- To assess and improve the robustness of instrument designs to space weather effects such as solar proton events. It is important that instruments for space weather monitoring are able to operate during major space weather events and thereby provide users with information on the progress of events.
- To assess how instruments can contribute to, and benefit from, multi-functionality, e.g. through contributions to the overall structure of the spacecraft and through use of shared electronics. The design of instruments must be coordinated with the overall spacecraft design as discussed in section 3. It is not sufficient to specify an instrument-spacecraft interface (e.g. dimensions, mass, power, data); the design process must support a deeper level of integration.

Recommendation: Stimulate work to reduce SW sensor sizes, e.g. through simplified requirements or use of different measurement techniques

Recommendation: Develop SW sensors that are robust against extremes of space weather, in particular solar proton events.

15.5 Communications

A crucial feature of space weather monitoring is the need for real-time or near real-time communications, so that data can be made available in time for use by nowcasting and forecasting applications. Given the time scale of magnetospheric phenomena, this timeliness is typically around 15 minutes. Thus space weather monitoring is most effective when it is straightforward to provide and maintain continuous ground station contact, e.g. solar wind monitoring at L1 requires only a few ground stations as demonstrated by the real-time solar wind experiment on NASA's ACE spacecraft.

Unfortunately, many of the space weather requirements in this study are in orbits where continuous contact is more difficult, e.g. low Earth and radiation belt orbits. These orbits require either:

- Advanced communications systems such as cross-links, relay spacecraft and space application of internet techniques, to provide real-time contact.
- A relaxation of requirements to focus on data collection for modelling where the timeliness requirement is much less important and downlink can wait some hours for a ground station contact.

Recommendation: In the short-term focus space weather monitoring, in LEO and radiation belt orbits, on collection of data for modelling but not for nowcasting and forecast.

We note that the need for advanced space communications is applicable to many applications, not just space weather. Current space communications capabilities are very limited (in both bandwidth and availability) when compared to everyday capabilities on the ground, where 5 Mbits/s is the norm for domestic internet. There is a generic requirement to improve space communications that is not limited to space weather. The space weather community needs to build links with other space users who would benefit from improved communications and lobby for general improvement of the infrastructure for space communications. It is unlikely that real-time space weather monitoring in LEO or radiation belt orbits will viable until it can exploit a generic infrastructure at modest marginal cost. Certainly space weather cannot support that infrastructure alone.

It's worth noting that there is a history of applications being enabled by advances in communications infrastructure. A clear example from space science is the way that remote data access has grown since the mid-1990s with the advent of the web technology and high-speed networks. There were many attempts to introduce similar systems in the 1980s, but these were severely hampered by limited bandwidth and the lack of interface standards (as now provided by the web). In hindsight it is clear that these early efforts had to wait to be enabled by the advances of the 1990s.

Recommendation: In the long-term encourage and exploit generic development of advanced satellite communication systems

Given the critical importance of real-time communications for links for space weather monitoring, and the technical challenge this poses, it is important to find the minimum data rate needed to provide a service. Data compression is an obvious issue to consider, but for space weather applications it will be more effective to reduce data flow by moving more data processing on board the spacecraft and downlinking only results that are needed for space weather services. This is very different to scientific applications where one usually downlinks raw measurements and performs data processing on the ground. This is important for science since it is usually necessary to revisit the data processing to take account of improved understanding of the instrument and, most importantly, to verify datasets critical to scientific papers. But space weather applications do not need this. They have little or no need to revisit past data. What is needed is reliable near-real time availability of data for forecasting, nowcasting and prompt post-event analysis. Thus on-board generation of physical parameters is a very attractive solution of space weather monitoring, especially the reduction of energetic particle measurements to a set of parameters describing their distribution (e.g. moments for thermal distributions, exponents for power-law distributions). Such schemes may leave the final calibration to be applied on the ground, so that updates may be applied without the need for uplink. On-board processing schemes must, of course, return information on the quality of such processing, e.g. goodness of fit when parameterising a particle distribution. Thus the ground segment can respond to quality factors, provide appropriate advice to end users and where necessary replace poor quality data by nulls values or modelled data (e.g. extrapolation from previous good data).

Recommendation: Develop schemes for on-board processing to un-calibrated physical parameters as a way to compress data.

A critical issue for nanosats is the choice of communications frequency and the availability of transceivers suitable for flight on nanosats. S-band would be a natural choice for many nanosat applications but its application within the scope of a European project is limited by the lack of suitable transceivers. European nanosat projects would be enhanced by greater availability of low power solutions (≤10W) which would fit better within nanosat power budgets, while still providing adequate output power for downlink (~1W). For low Earth orbit applications it is also worth considering UHF/VHF frequencies, as used on Munin [R10]; this allows use of simple antennae on the spacecraft and the ground.

Recommendation: Develop low power RF systems suitable for nanosats.

Availability of ground stations is also an issue for nanosat solutions in low Earth orbit. The approach adopted in this study has been to focus on polar orbits at heights of 800 to 1000 km, so that a single high-latitude ground station will see every spacecraft while over the relevant polar region. Thus the spacecraft can downlink once per orbit and, if inter-spacecraft links are used, a ring of spacecraft can be kept in continuous ground station contact. In a European context the likely location for a high latitude ground station is Svalbard in the high Arctic, where ESA already has some facilities. However, we note that redundancy would be improved if there were a similar European facility in the Antarctic; this would probably need to be poleward of 75° South, e.g. in the Ross Sea sector.

15.6 Launch and orbits

A key issue for the launch of nanosat constellations is how to deploy the constellation where a single rocket is used to launch a number of spacecraft which are then dispersed to form the constellation. There are two competing approaches:

- Own propulsion: the launch vehicle deploys the nanosats close together and each then uses its own propulsion to reach the final orbit
- Hive: the launch vehicle deploys a hive spacecraft that carries the nanosats. The hive then uses its propulsion to reach the final orbit of each nanosat in turn. It deploys one nanosat and then moves to the next deployment location.

Another key issue is the maintenance of the nanosats constellations after deployment – in particular, the maintenance of the inter-spacecraft separations. These must be set so as to satisfy the spatial and temporal resolution in the measurement requirements (see section 18). Thus separation maintenance is critical to the overall performance of the constellation. This will be a particular challenge for constellations that are subject to significant drag around perigee (e.g. LEO, radiation belt). The perigee drag environment is highly variable (due to space weather effects) so there can be marked differences in the drag-experienced by each spacecraft within a constellation. This can cause dispersion in their orbital periods and loss of stability in the spacecraft separations. That stability requires close synchronisation of orbital periods, e.g. to maintain the separation of the radiation belt constellation to 10% over a year the periods must be synchronised to better than one second. Thus the maintenance of constellations with low perigees will require that each nanosat has some propulsion in order to trim its orbital period in response to drag changes.

Orbit determination is an important issue for space weather nanosat operations – in particular for ground station contacts, orbit maintenance and instrument control. It may be performed by classical ground-based tracking or via on-board determination such as space-GPS. Good orbit data are an important support for on-board autonomy as discussed in section 15.3.2. Orbit determination is a key issue for the high-level mission design of a space weather nanosat constellation (covering both space and ground segments) and not simply a ground segment issue.

For nanosats in low orbits there is a requirement to ensure deorbit on internationally-agreed timescales. In some cases this may be provided by the orbit evolution but in many cases active measures may be required, e.g. to provide propulsion for deorbit. In the latter case, the control of deorbit must be carefully considered at the design stage, e.g. should design provide for autonomous deorbit after a planned lifetime or via a watchdog that triggers after some period of inactivity. These autonomous approaches bring a risk of accidental deorbit so it will be important to trade-off that risk against the desire for guaranteed deorbit.

Recommendation: Encourage development of on-board propulsion for orbit maintenance and de-orbit.

15.7 Reliability and the replacement strategy

Reliability is a key issue for use of nanosats for space weather monitoring. It is assumed that such nanosats will be subject to periodic replacement to ensure continuity of an operational service. However, the current generation of experimental nanosats tend to have short lifetimes, around a few months. This is probably too short for a viable replacement programme, where we should perhaps be looking for a lifetime of two years or so. This value must ultimately be subject of a trade-off between the cost of improved reliability and cost of more frequent launches. Nonetheless it is clear that effort to improve reliability will be valuable.

It is important to understand what reliability means in the content of space weather monitoring by constellations of spacecraft. In many cases the loss of any individual spacecraft will simply degrade the spatial resolution of measurements but, in other cases, it may be a critical item (e.g. a communications relay) whose failure can lead to constellation-level failure. It is important to develop models for constellation-level failure. These will certainly include numerical models, e.g. the present study includes a Monte-Carlo simulation of the case that a constellation fails when N adjacent spacecraft fail. Analytical models may also be considered, but experience from this study indicates that this will require deep knowledge of number theory (e.g. partition functions) to provide an enumeration of failure configurations within a multi-spacecraft system. It will also be important that these models include different approaches to the failure of sub-systems, e.g. constant rate with time (as used in the present study) or bathtub curve (failures most likely at start or after a nominal lifetime). Failure modelling should be accompanied by the development and use of test beds to verify the output from models.

Recommendation: Develop reliability models for satellite constellations

It will be valuable to explore ways of improving nanosat reliability, e.g. by providing or increasing parallel redundancy on each spacecraft. MNT/MEMS devices have great potential here by facilitating accommodation of multiple devices on a nanosats. Other important issues are the availability of radiation-hardened components, selective use of shielding to protect susceptible components and the possibility to design for graceful degradation in case of failure.

Finally we also note that space weather measurements, unlike science, will not benefit from increased performance of instruments.

15.8 Programmatics

A critical issue for programme of nanosat-based space weather monitoring, and indeed for any space weather programme, is to establish a firm organisational basis for the programme. Experience shows that this is a big issue. Most existing space weather services are based on a mix of scientific programmes and application activities. This reflects the origin of these services in scientific programmes for space research and the continuing synergy whereby measurements for space weather are also required for many research activities. There has also been a long tradition (dating back to the 1950s) of links between space science programmes and applications activities, especially for radio propagation. But these links have become much less important in recent years:

- A tendency of some space research programmes to focus on studies of specific and often esoteric phenomena to the exclusion of the broader space and planetary environment issues common in solar system science, including the science behind space weather. This is a cultural rather than scientific issue. We need to promote a cultural change to recognise the importance to humanity of our space and planetary environment and that this requires scientific study alongside the terrestrial environments studied by the geosciences. The re-development of the old synergy between space science and space weather is one of the many benefits that would arise from such change.

- A closely related issue is to recognise that the core science behind space weather is plasma physics, and especially the physics of collisionless plasmas. Space weather is essentially an application of plasma physics and would benefit from the establishment of better links between space weather, solar-terrestrial physics and the traditional plasma physics community (both laboratory and theory). This is a natural synergy.
- A reduction in public sector support for measurements that underpin applications work - in the belief that private sector then will pay for those measurements that are underpin useful applications. There is no concrete evidence that this works. The evidence that does exist shows that: (a) the private sector will pay for services that add value to measurements, e.g. specialist interpretation to assess the measurements and provide results customised to user needs, but (b) this process does not generate enough money to pay for the underlying measurements. The private sector expects these measurements to be publicly funded, e.g. by obtaining access to measurements already funded by research programmes, but scientific programmes are increasingly reluctant to do this without some cost recovery. The underlying problem is a conflict of economic models between scientific programmes and the vendors of space weather applications. There is no process to resolve the conflict and this leads to stalemate. An effort is needed to resolve this conflict, e.g. by a proper identification of the costs of providing scientific data access to the applications community. This is another area where work is needed to restore the old synergy between space science and space weather.

A successful space weather programme must seek support via a good mix of science and applications interest. The applications role is vital since that reflects the needs of the end users, but there is an important role for science as a source of knowledge transfer to space weather applications and through the technical value of synergistic measurement programmes. What is critical today is to start to build the political support for an inter-disciplinary space weather programme that combines science and applications. This needs to be done both within ESA and beyond (e.g. within the organisation of relevant EU programmes). It must be underpinned by an effort to identify good examples of space weather applications, e.g. Swarm, the ESA Earth Observation mission to study the internal magnetic field of the Earth, is critically dependent on a good understanding of space weather to distinguish internal magnetic field signatures from those caused by space weather.

Recommendation: Raise awareness of the synergy between space weather and STP science in respect of the monitoring of the space environment.

The section 15.2 discussion on use of MNT in nanosats noted the need for flight testing. Exposure of sample devices to the actual environment provides the ultimate validation and is crucial for aspects that cannot be tested in space. This is cross-referenced in the present section as the proposal for a demonstrator programme is also a programmatic issue. There needs to be a programme that can validate devices for the range of different environments that exist in space and that should emphasise testing in harsh environments to cover the worst case.

Recommendation: Establish a flight demonstration programme to allow new technologies to tested and validated in space.

Another programmatic issue is the need to develop cost models for nanosat development, construction and operation. This should include both top-down and bottom-up methods.

15.9 Summary of recommendations

The table below provides a numbered summary of the thirteen recommendations developed through the Prospects and Recommendations workpackage. These highlight key issues that should be addressed if we wish to exploit the potential of nanosats to monitor space weather conditions. The table is sub-divided according to the sub-sections above (see there for background on each recommendation).

The bulk of the recommendations (9 out of 13) address three main areas (a) the use of Micro- and Nano-technology (clearly fundamental for nanosat applications); (b) payload miniaturisation and robustness against extreme space weather (fundamental to use of nanosats for space weather monitoring) and (c) communications (a critical issue for space weather monitoring). The other four recommendations are highlight other important technical issues (autonomous operation, nanosat propulsion and nanosat constellation reliability) and one programmatic issue – namely the potential for synergy between operational space weather measurements and research programmes in solar-terrestrial physics.

Table 50. Recommendations on space weather nanosat development

N	Recommendation
Use of MNT (14.2)	
1	Develop a methodology for qualification of MNT/MEMS devices for space. This shall allow tailoring to each case – in particular distinguishing use of COTS and bespoke devices.
2	Develop methods to assess and mitigate radiation sensitivity of MNT/MEMS devices.
3	Develop design environments and standards that are appropriate for multi-functional systems.
Other new technology (14.3)	
4	Review the potential of autonomy to simplify SW nanosat operations.
Payload (14.4)	
5	Stimulate work to reduce SW sensor sizes, e.g. through simplified requirements or use of different measurement techniques
6	Develop SW sensors that are robust against extremes of space weather, in particular solar proton events.
Communications (14.5)	
7	In the short-term focus space weather monitoring, in LEO and radiation belt orbits, on collection of data for modelling but not for nowcasting and forecast.
8	In the long-term encourage and exploit generic development of advanced satellite communication systems
9	Develop schemes for on-board processing to un-calibrated physical parameters as a way to compress data
10	Develop low power RF systems suitable for nanosats.
Launch and orbit maintenance (14.6)	
11	Encourage development of on-board propulsion for orbit maintenance and de-orbit.
Reliability (14.7)	
12	Develop reliability models for satellite constellations
Programmatics (14.8)	
13	Raise awareness of the synergy between space weather and STP science in respect of the monitoring of the space environment.
14	Establish a flight demonstration programme to allow new technologies to tested and validated in space.

16 Key conclusions and future work

16.1 Conclusions

The study has reviewed the current state of relevant work on MNT – looking at the development of important MNT devices such as RF components, accelerometers and power sources, at the packaging of MNT systems and of practical experience in flying an MNT spacecraft. The study also made an extensive review of current and future developments in nanosats. These reviews both provided important background for later parts of the study. This study has also reviewed the requirements for space-based measurements to support space weather applications that help both space-based and ground-based services. This review consolidated the outputs of earlier ESA space weather studies and updated them to take account of recent developments in space weather services. The study identified where these measurements can be performed on nanosats and explored how to classify them.

A key conclusion was that it is not worthwhile to classify the measurements by their applicability to ground-based or space-based services. This is simply because the majority of space-based measurements have applicability to both domains. This reflects the chain of space weather from its source on the Sun to its impact on and around the Earth. The majority of the space-based measurements monitor the upstream space weather environment (e.g. solar and solar-wind measurements) which is critical to both space-based and ground-based services.

This negative conclusion is balanced by a positive conclusion – that it is possible to classify the space weather measurements into a small set of distinct spacecraft constellations: (a) two low-Earth orbit constellations aimed at ionospheric and solar observations, (b) a constellation in geosynchronous transfer orbit aimed at radiation belt and plasmasphere observations; (c) a Molniya constellation aimed at remote sensing of auroral activity, (d) a multi-orbit constellation for better measurements of the magnetospheric magnetic field; and (e) an L1 spacecraft for monitoring the solar wind and heliospheric particle fluxes.

This classification has driven the design elements of the study. These have developed a set of outline designs for each constellation and for the instruments that must be carried by each constellation. The instrument solutions are largely based on existing heritage with some extrapolation for developments in instrument miniaturisation. In one case a novel instrument concept is proposed – namely a low-resolution EUV solar imager for flare location.

The design work has outlined a nanosat concept for each constellation and explored how the nanosats might be launched, operated and de-orbited. One key issue here is the design of data links. This is always a critical issue for space weather missions since most missions have requirements for near-real-time downlink to ensure timely availability of data. Where feasible the data link designs make use of innovative ideas such as inter-spacecraft links and small ground station antennae.

Another key issue is the replacement strategy. Space weather services require continuity of data so it is important to replace spacecraft at regular intervals to reduce the risk of data failure. The use of constellations raises interesting issues about reliability – since the failure of one spacecraft will degrade constellation performance but not necessarily destroy it. Thus we had to consider how constellations could adapt to overcome failures, e.g. routing signals past failed spacecraft and adjusting spacecraft positions to optimise data sampling, This led us to consider the number of failures that each constellation could tolerate and then to model the likelihood of multiple failures (using a simple numerical model). This model allowed us to estimate replacement periods for each constellation and thus was central to developing the required replacement strategies (frequency of launches and need to produce multiple copies of spacecraft).

The final element in this study was to look at the prospects for using nanosats in space weather monitoring and make recommendations on how to bring those prospects to fruition. These are discussed in section 14 and are not repeated here as they can be treated as a standalone result.

16.2 Ideas for future work

There are two main areas here for future work. One is to take forward the concept of space weather nanosat constellations. These five constellations now need deeper study as separate mission concepts. Thus it is important to establish a prioritisation among them. Here are a few thoughts:

1. L1 monitoring. One can make a strong case for giving this a high priority. Real-time L1 monitoring is fundamental to all space weather services since it provides a 30 to 60 minutes ahead estimate of the energy input into the magnetosphere. The global space weather community has long experience of these data and continues to develop new services that will require these data. Indeed the global space weather effort would be seriously damaged if L1 monitoring were not available. Yet, this service depends on NASA's aging ACE spacecraft, backed up to some extent by instruments on the even older SOHO and WIND spacecraft. But note that ACE has an on-board system dedicated to supporting real-time data access using low-bit rate downlink (which is cheaper to operate). The other two spacecraft lack this and must provide data by running their normal scientific data downlink in real-time. Thus there is a crying need for a replacement monitor at L1. One possibility is for NASA to fly the cancelled TRIANA spacecraft, but that now seems unlikely. In the medium-term the proposed KuaFu A mission could provide the opportunity to put new European instruments at L1, but that is part of a larger scientific mission and thus awaits decisions on the funding of the mission and the instruments. The present study indicates that a dedicated space weather monitor at L1, based on nanosat technology, is feasible. It would be a great opportunity for Europe to position itself in a key role in global space weather activities – while showing that it can be done well at modest cost. For this reason we recommend that this option be given a high priority for further study.

2. GTO constellation. This offers the possibility to make more comprehensive measurements, in both space and time, of the radiation belts, especially the highly variable outer (electron) belt. The expensive part of any mission to study the radiation belts at multiple locations (many local times and L values) is deployment to the various different orbits. This requires either multiple launches or extensive on-board propulsion. However, this cost is driven by the idea that we must put the spacecraft in pre-specified set of orbits. It is that requirement to control the orbits that drives deployment costs. An alternative that might reduce costs is to adopt an opportunistic strategy, i.e. fly as secondary payload on a number of GTO launches and accept the local time distribution that results. To assess this option we need to assess the local time distribution that is likely to arise, e.g. take subsets of past launches and derive the resulting local time distribution.

3. SWARM constellation. This constellation is unusual in that it seeks to improve a space weather model rather than provide data for immediate use in services. But the model to be improved, the magnetospheric magnetic field, is central to many other space weather applications. Thus it is an appropriate topic for a dedicated space weather mission. It could also have useful spin-out into other ESA programmes, e.g. the Earth Observation programme is interested in the internal magnetic field of the Earth. But to study that it is important to have a good model of the magnetospheric field, so its effects can be subtracted from observations of the total field.

4. Molniya constellation. This seeks to provide remote sensing of the auroral region, which is a major focus for the energy flows that drive space weather. The present measurement requirements are fairly simple and perhaps of lower priority than the previous constellations. In this case future work should focus on exploring whether there are other useful activities that can be carried out by this constellation, e.g. remote sensing of the polar environment or perhaps providing communication links in the polar region. The latter is the traditional use of Molniya orbit and a useful one given the increasing economic importance of polar regions and the limitations on use of GEO-based satcom at high latitudes.

5. LEO constellation. This is perhaps the most challenging constellation. The low altitude makes it hard to satisfy the real-time data requirements without either a huge ground station network or a sophisticated system of inter-spacecraft links combined with use of high-latitude ground stations. In other respects it is an attractive option since it can be launched quite easily and can use simple instruments. But the space weather context means that the communications requirement is critical – and thus creates a major technical challenge. This is a mission that would greatly benefit from novel space communication architectures, e.g. on-demand links to relay spacecraft data akin to the on-demand access provided by mobile telephone systems. Thus we give this constellation a low priority at the present time, but note that advances in space communications could change that ranking.

The other main area for future work is take forward the recommendations developed in section 14. There are four key issues where work is required:

1. One issue is the need to develop instruments that are well suited to routine monitoring (rather than to measurements that support scientific research). We need to develop instruments that are smaller (in size, mass and power) in order to gain more opportunities to fly monitors; a key issue here is to reduce space weather sensor sizes, e.g. through simplified requirements or use of different measurement techniques. We also need sensors that are robust against extremes of space weather, in particular solar proton events. Finally we also need to reduce instrument costs. The long-term success of any programme for monitoring space weather will depend on the ability to build and replace adequate numbers of instruments.

2. Another issue is to develop ways to make best use of MNT/MEMS devices in space. This has several aspects including (a) developing methods to qualify MNT/MEMS devices for use in space, (b) developing methods to assess and mitigate radiation sensitivity of MNT/MEMS devices and (c) developing design environments and standards that are appropriate for multi-functional systems. The latter is particularly important as it will exploit synergies that can facilitate nanosat construction (e.g. reducing mass and cost) but it cuts across the traditional approach of decomposing design into separate systems.

3. A key element in making best use of MNT/MEMS devices will be to facilitate their flight testing. We therefore recommend to establish a programme for in-flight demonstration of this and any other technologies needed for nanosats. We note that the growing European interest in CubeSats [R44] provides a convenient way to design and build the demonstrator spacecraft. The demonstration programme might encourage flight testing of new European technologies through a mixture of human networking activities, technical support and, most importantly, opportunities for launch and operation of test spacecraft. The demonstration programme will also provide data on the reliability of nanosats and their sub-systems; as noted above, such data will be a valuable resource in assessing the reliability of nanosats and of space-based MEMS/MNT technologies. It is important to gather that data to enable the optimisation of replacement strategies for operational nanosat systems.

4. The fourth and final issue is to look at communications links. As discussed previously, this is a critical issue for space weather measurements because the majority of measurements have a requirement for real-time downlink. To address this problem it is important to develop European sources for low power RF systems (i.e. suitable for use on nanosats). It is also important to develop schemes for data compression; in the context of space weather data, the best way to do this is to develop schemes for on-board processing of raw data to un-calibrated physical parameters (e.g. calculation of moments for particle measurements). This exploits our knowledge of the underlying physics to make an intelligent compression of the data. It is likely to provide much better compression than mathematical schemes that have no knowledge of the data. In the longer-term we should encourage and exploit generic development of advanced satellite communication systems, e.g. on-demand links, as these will greatly facilitate data access from any spacecraft.

These issues, especially the second and fourth, are suitable for discussion in dedicated workshops. Such workshops might serve to pull together the appropriate European players. The first issue is a very broad field and probably requires a broader programme of workshops and more practical activities. Development of miniaturised instruments could be stimulated by competitive activities, e.g. an annual competition to identify the best European developments.

Finally we note that new space weather measurement requirements will inevitably emerge in the light of new scientific understanding. A potential example is the 2007 report [R45] that relativistic electron enhancements are seen 10 to 30 minutes before major solar energetic particle events. This clearly has potential to provide short-term warnings of energetic particle events and would imply a new requirement for measurements of heliospheric electrons at energies of 200 keV to 1 MeV. This issue should be monitored to see if further studies confirm the report.

17 Annex A – Service requirements

Table 51. Synthesised service requirements

N	Requirement	Main characteristics
1	The space and ground segments for a space weather monitor shall provide continuous data availability during and after extreme events. [UR 23 from RAL/Alcatel studies]	Robust sensors and operations
2	The space and ground segments for a space weather monitor shall provide continued data availability in the event of premature failure or end-of-life of key space weather systems. [UR 24from RAL/Alcatel studies]	Redundancy in orbit and rolling replacement at end of design life
3	The space and ground segments for a space weather monitor shall provide efficient distribution of data to users and continuous availability [UR 25from RAL/Alcatel studies]	Prompt availability of data, design to ensure data gaps are below maximum acceptable gaps.
4	The space and ground segments for a space weather monitor shall support good calibration of that monitor. [new requirement in this study]	Dedicated calibration activities both pre-launch and in-flight; regular monitoring of instrument performance; configuration control
5	The space and ground segments for a space weather monitor shall provide a data quality flag for each data record [new requirement in this study]	Assessment of data quality through on-board and on-ground procedures
6	The development, deployment, commissioning and operation of space weather monitors shall be subject to appropriate and effective quality assurance procedures [new requirement in this study]	Document and review activities; avoid bureaucratisation of QA process by focusing on value to programme
7	The space and ground segments for a space weather monitor shall, to the greatest extent possible, be independent of presently operational or planned scientific missions. [new requirement in this study]	Avoid unacceptable trade-off with science objectives

18 Annex B – Measurement requirements

This section specifies the detailed requirements derived during this study as follows:

- Requirement reference comprising the high level requirement number and sub-requirement number separated by a period symbol.
- A concise description of the space weather parameter needed
- Cadence of measurements in minutes
- Instrument data rate in kilobits per second
- Timeliness – time interval between data acquisition and provision of product to user.

Req	Measurement	Cadence (mins)	Data rate (kbps)	Timeliness (mins)
1.1	EUV images of Sun	10	0.02	30
3.1	Solar X-ray flux monitor	5	0.04	5
4.1	Solar EUV full disc flux	1440	1	1440
4.2	Solar UV flux	60	0.25	60
8.1	Solar wind bulk velocity	1	0.1	30
8.2	Solar wind bulk density	1	0.1	30
9.1	Heliospheric magnetic field	1	0.2	2
10.1	>100 MeV ions from heliosphere	5	0.02	5
10.2	2-100 MeV ions from heliosphere	5	0.02	5
10.3	2-20 MeV electrons from heliosphere	5	0.02	1440
11.1	Auroral UV imaging	60	10	5
11.2	Auroral particle precipitation	60	2	5
13.1	Magnetospheric magnetic field	1	0.2	1440
14.1	In-situ magnetospheric E field	180	1.5	5
15.1	1-10 keV electrons in magnetosphere	1	2	90
16.1	10-100 keV electrons in magnetosphere/rad belt	1	2	60
17.1	High energy electrons in rad belt	1	0.1	5
18.1	> 10 MeV protons in rad belt	1	0.1	5
19.1	Dosimetry	5	0.1	5
20.1	Total electron content of iono/plasmasphere	5	0.1	5
20.2	Electron density of iono/plasmasphere	1	1	5
21.1	Plasma velocity in ionosphere	0.1	1	5
22.1	Neutral density in thermosphere	30	1	60
25.1	Microparticle measurements	1440	0.03	1440

19 Annex C – Special requirements on particle flux measurements

Many requirements to measure energetic particles are related to their ability to cause radiation damage and charge deposition. The space weather risk is significant only when there are high particle fluxes leading to a high impact rate. Thus we can constrain such requirements to detect only high fluxes that lead to risk. The table below shows the measurement requirements for energetic particles and indicates:

- Whether the requirement can be restricted to high fluxes
- An estimate of the threshold for high fluxes
- A rationale for that estimate

Table 52. Measurement requirements for energetic particles.

Requirement number	Sub-requirement number	Measurement sub-type	Location	Highflux only	Threshold	Source/ rationale for threshold
8	1	Solar wind bulk velocity	Upstream	N	n/a	
8	2	Solar wind bulk density	Upstream	N	n/a	
10	1	>100 MeV ions from heliosphere	Upstream	Y	10 particles $cm^{-2}s^{-1}sr^{-1}$	SEC solar proton limit [R21]
10	2	2-100 MeV ions from heliosphere	Upstream	Y	10 particles $cm^{-2}s^{-1}sr^{-1}$	SEC solar proton limit [R21]
10	3	2-20 MeV electrons from heliosphere	Upstream	Y	10^3 particles $cm^{-2}s^{-1}sr^{-1}$	as SEC electron limit [R22]
11	2	Auroral particle precipitation	Ionospheric -LEO	Y	10^6 particles $cm^{-2}s^{-1}sr^{-1}$ keV^{-1}	SPEE report gives typical charging flux as 10^6 to 10^8 particles $cm^{-2}s^{-1}sr^{-1}keV^{-1}$ [R23]
15	1	1-10 keV electrons in magnetosphere	Rad belt	Y	10^6 particles $cm^{-2}s^{-1}sr^{-1}$ keV^{-1}	as SPEE charging issue [R23]
16	1	10-100 keV electrons in magnetosphere/ radiation belt	Rad belt	Y	10^6 particles $cm^{-2}s^{-1}sr^{-1}$ keV^{-1}	as SPEE charging issue [R23]
17	1	High energy electrons in radiation belt	Rad belt	Y	10^3 particles $cm^{-2}s^{-1}sr^{-1}$	SEC solar electron limit (= daily fluence of 8.6×10^7 electrons $cm^{-2}sr^{-1}$) [R22]
18	1	> 10 MeV protons in rad belt	Rad belt	Y	10 particles $cm^{-2}s^{-1}sr^{-1}$	SEC solar proton limit [R21]

20 Annex D – Dropped requirements

Req	Description	Rationale
1.2	H-alpha images	Too big - focus on small EUV imager for flare /detection and location
1.3	Soft X-ray imager	Too big - focus on small EUV imager for flare /detection and location
1.4	Stereo images of Sun-Earth space	Too big - large optical instrument
1.5	Helioseismology	Too big - large optical instrument
1.6	Lyman-alpha monitoring	Too big - large optical instrument
2.1	Coronagraph	Too big - large optical instrument
6.1	Radio bursts	Too big - needs long antenna
7.1	Solar magnetograms	Too big - large optical instrument
11.3	Auroral visible imaging	Focus on UV imager - better auroral detection. Visible subject to sunlight scattered in atmosphere.
12.1	AKR	Too big - needs long antenna
23.1	Neutral wind	Too big - needs big Fabry-Perot device

21 Annex E. Multiplicity of radiation belt measurements

The number of spacecraft required to monitor the radiation belts has been analysed in some detail as part of this study so that: (a) we can explore the possibility of deploying large numbers of nanosats for radiation belt monitoring and research, and (b) we can understand how we could adjust the ideal situation to match funding opportunities.

The number of spacecraft used for this task is determined by the required resolution of monitoring in terms of magnetic local time and McIlwain L value. Our aim is to quantify this relationship. We assume that the monitoring will by done from a geosynchronous transfer orbit (GTO) as in the previous studies [R11, R12, R13]. For purpose of modelling we take GTO as 600 by 35700 lm altitude. The resolution in local time is then set by using a number of different GTO orbits separated in right ascension, as discussed in previous studies. If we have M such orbits spread equally around all 360 degrees of right ascension, the local time resolution is 24/M hours. We propose to use M=4 to get a resolution of 6 hours. Other values will improve or degrade resolution as M is increased or decreased.[2]

The resolution in L value is complex. We have modelled this by assuming that (a) we have N spacecraft spread evenly in time around each GTO orbit and (b) the orbit is close to the equatorial plane. We also assume a dipole geomagnetic field (a good assumption for L>4), so that for our equatorial orbit we can take the L value to be equal to the geocentric distance in Earth radii. We then set up a simple Kelperian model of the spacecraft orbits around the Earth and determine the time variation of L value for each spacecraft over a single orbital period (10.6 hours). The result for N=8 is shown in Figure 29 below. (Note: the small kinks in some orbit curves are artefacts arising from use of a simple two-stage solution to Kepler's equation.)

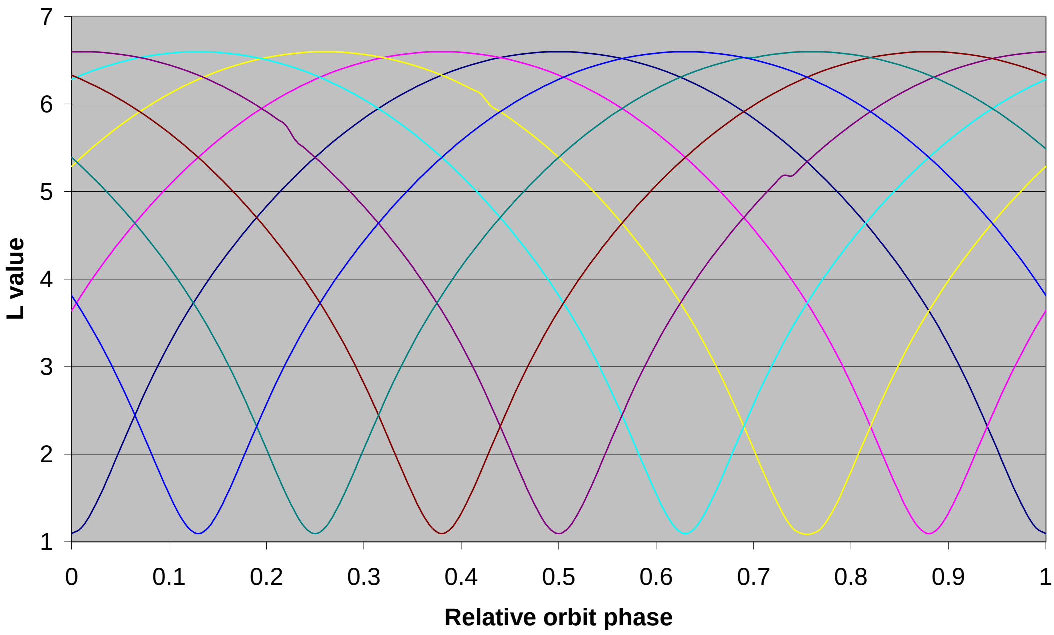


Figure 29. L values for 8 radiation belt monitors spread evenly around GTO orbit

This figure shows eight overlapping orbits with each spacecraft sampling different L-values at different times. The next step is to convert this plot into a form which shows the resulting resolution in L-value. At each time step, we rank the spacecraft in order of increasing L value. We then plot the time variation of L

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[2] This assumes that it is possible to launch into GTO orbits with their lines of apsides distributed in right ascension. This requires further work, beyond the scope of present study, but we note that the normal launch configuration is to enter an orbit with apogee towards the Sun. Thus a range of right ascension could be achieved by launches at different seasons.

value for each rank, i.e. the identity of the spacecraft changes when the ranking changes. The result is shown in Figure 30 below. Note how the curves in Figure 30 can join up to demonstrate their derivation from Figure 29.

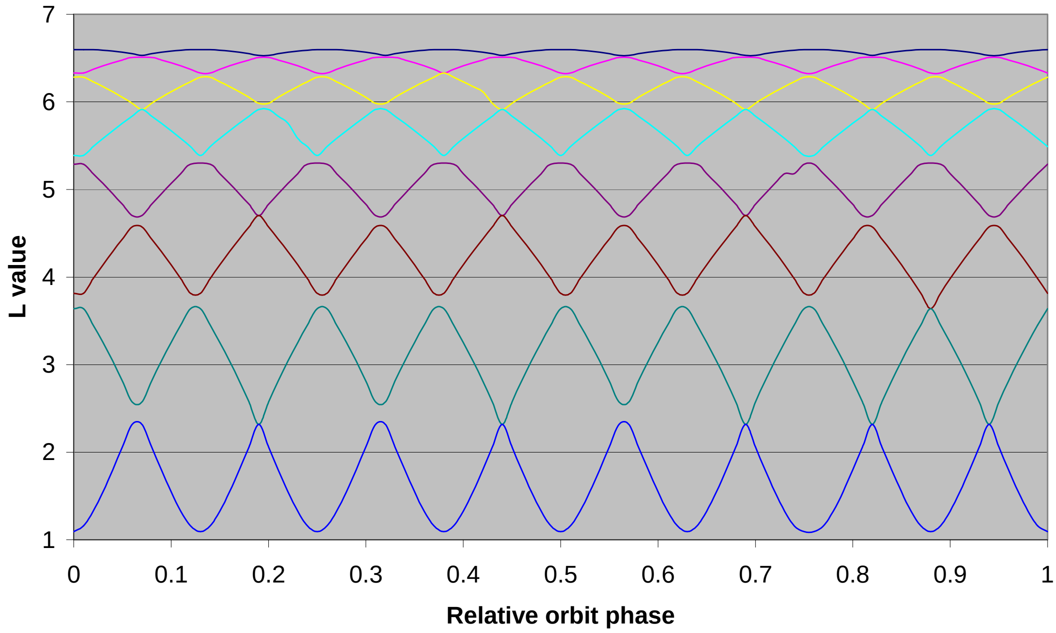

Figure 30. Radiation belt monitors in GTO ranked by L value

The great advantage of Figure 30 is that the ranking gives us a clear sequence of measurements across the full range of L values. Each ranked position spans a well-defined sub-range, e.g. L=2.6 to 3.6 for the second ranked position. The temporal changes of spacecraft identity for each sub-range is a simple operational matter and will not concern us further here. The example shown (N=8) has been chosen as our preferred solution as it gives a resolution of about 1 in L value. Other solutions will simply degrade or improve this resolution as the number of spacecraft is decreased or increased.

Note that our N=8 solution has two outer positions that do not contribute significantly to L value resolution. This is a simple consequence of Kelperian orbits; the spacecraft near apogee will be close-spaced and contribute little to the resolution. The resolution is set by the spacecraft distribution away from apogee. Thus the number of spacecraft needed is greater (say 30%) than the number of L value sub-ranges to be resolved.

Thus we must accept that each spacecraft will have a period around apogee when its measurements are of limited value. During the design phase, it may be worth considering whether to stop data-taking and downlink during this period in order to reduce power consumption and data volume downlinked.

22 Annex F – Use of spacecraft magnetometer data for index generation

Geomagnetic indices play a critical role in many space weather applications by providing a quantitative estimate of the state of the magnetosphere. The indices used for this purpose are those established over the past 60-70 years, e.g. the mid-latitude indices Ap/Kp and aa, the equatorial index Dst and the auroral electrojet index AE. These are all derived from analysis of ground-based magnetometer data drawn from networks at the appropriate latitudes. The coverage of those networks is not ideal but rather has evolved historically in response to the availability of land on which to place magnetometers and the scientific capability and interest of different countries to operate magnetometers. These technical factors are then vulnerable to economic and political considerations.

Despite these deficiencies the ground-based indices lie at the heart of much space weather modelling. The reasons are straightforward:

1. The ground-based indices are readily available. Their statistical properties and limitations are well understood. There is now a reasonable understanding of their relationship to magnetospheric and ionospheric current systems.
2. There is a huge body of research knowledge that characterises space weather in terms of these indices.

Ongoing work within the Space Weather Applications Pilot Project (ESA, private communication) has raised the question of whether it would be better for space weather applications to use geomagnetic indices derived from spacecraft magnetometers. This offers the advantage of being able to design coverage without the topographic and political constraints inherent in ground-based systems and thus has the potential to obtain more consistent data. However, the requirements for such measurements are poorly understood. A key issue here is that space-based magnetometers will usually pass inside the current circuits of magnetospheric and ionospheric current systems and thus obtain a very different view that obtained by ground-based magnetometers, which necessarily sit outside those current circuits.

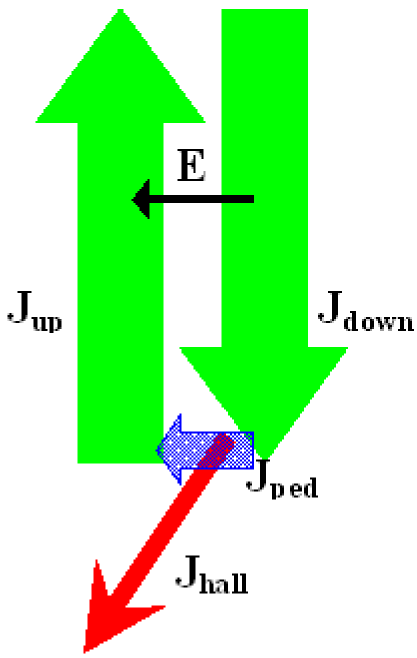

Figure 31. Auroral zone current systems

This is illustrated in Figure 31 above. This shows the typical configuration of currents in the auroral zone, which is a major site of space weather activity. There are two field-aligned (Birkeland) current systems slightly separated in latitude (J_{up} and J_{down}). These currents link the auroral ionosphere to the magnetosphere. The upward current corresponds to downward electron flow and thus is the site of intense electron precipitation into the atmosphere. The two current sheets are linked by a horizontal meridional current flowing in the conductive region of the ionosphere (100-150 km), where the electrons can move freely into response to electric fields but ion motion is inhibited by ion-neutral collisions. This linking current usually flows parallel to the electric field imposed by the magnetosphere and is thus a Pedersen current (J_{ped}). But given the presence of the geomagnetic field, there is also a Hall current flowing perpendicular to the electric (J_{hall}). The relative orientation of these currents depends on the magnetic field orientation (down in northern hemisphere, up in the southern hemisphere) and the local time (e.g. in the evening sector J_{up} is poleward of J_{down} in the evening sector and J_{hall} flows east; in the morning sector these are all reversed).

How are these current systems viewed by ground-based and space-based magnetometers? The main set of field-aligned and meridional currents (J_{up}, J_{down}, and J_{ped}) form a solenoidal current system whose magnetic field will be confined largely inside the current circuit. As a result an LEO spacecraft passing through the field-aligned currents will see a east-west magnetic perturbation (e.g. see Figure 5.55 of [R24]) but a ground-based magnetometer will not see any significant part of this field. All that the ground-based magnetometer sees is the field from the Hall current (the auroral electrojet), which appears as a north-south magnetic perturbation, traditionally termed a “magnetic bay” (e.g. see figure 8.27 of [R24]).

The key conclusion is that ground-based and space-based magnetometers can have very different responses to magnetospheric and ionospheric current systems. Thus the use of space-based magnetometer data as a substitute for ground-based data is not at all straightforward. Significant work is needed to explore how space-based magnetometer data could address the space weather measurement requirements currently covered by ground-based data. There are several possible approaches:

- To establish relationships between ground-based and space-based geomagnetic data such that existing models inputs could be generated from space-based data.
- To re-characterise space weather models in terms of space-based geomagnetic data. This is a major undertaking but would offer the opportunity to develop new geomagnetic indices firmly based on our modern understanding of magnetospheric physics.

In summary, a space-based magnetometer network has the potential to provide a more consistent set of geomagnetic data but significant work is needed to understand how space-based data could address the requirements on inputs for space weather models. Space-based data is not a simple substitute for ground-based data.

23 Annex G. Timescales for MNT device availability

Table 53 below summarises the MNT devices discussed in section 4.2, their status at the time of the study (2005) and predicted status in 2010 and 2015.

Table 53. MNT Device Roadmap

Communication Systems	**2005**	**2010**	**2015**
Passive RF Components (< 6 GHz)	First integrated passives available (MEMSCAP, Philips)	Integrated passives including tuneable varactors	Highly Integrated passive chip including tuneable varactors, inductors, and high Q resonators (Philips et al.)
MEMS high Q Resonators (oscillators/filters < 6 GHz) MEMS high Q (oscillators/filters > 30 GHz	Ceramic, SAW FBAR/SMR (e.g. Agilent,Infineon) Research on silicon based solutions (Samples from Discera) Research (e.g. car-radar)	very promising MEMS developments, major size reduction, first components (Discera, Philips?) Unknown	Integrated filter solutions Integrated radio solutions Available (car radar?)
RF-Switches < 8 GHz RF-Switches > 20 GHz	Available (Magfusion, Teravicta) restricted access, research (Radant, Thales?)	serial production Samples (Radant ??, Thales?, BAE?, EADS?)	Mass production In commercial use
3D-stacked radio modules e.g. micropack < 6 GHz solutions > 30 GHz	Solutions with high integration density research supported by governments	Stacks possible unknown	In use available (car radar)
MEMS based mm-wave Phased array antennas	Ongoing research	Samples available ? (Thales, Alcatel, EADS,)	Use of small PA-systems
AODCS	**2005**	**2010**	**2015**
Active Pixel Sensors	Radhard Devices **available**in Europe	*Further improvements expected (e.g. power, pixels), but no reliable forecast available*	
Magnetometers	MNT Fluxgate and AMR **available**(FHG-IMS, Phillips, Honeywell)	New Materials, Improvements in GMR, TMR sensors	
MEMS Angular Rate Sensors	Performance not jet sufficient, Current devices not radiation hardened, Research in progress	Radhard MEMS ARS available [1-10°/h]	Radhard MEMS ARS with improved performance [0.1-1°/h]
Momentum Wheels	MNT Motors **available**(e.g. Faulhaber, MicroMo)	*no forecast available*	
Propulsion Systems	Various concepts **available**at research institutes, some with flight heritage	*no forecast available*	

Power Generation	2005	2010	2015
Solar Cells	InGaAs 32% efficiency on Ge substrate (ESA, RWE Solar)	Ultra-thin lightweight and flexible polymer DSC cells (efficiency > 10%)	Ultra-thin lightweight and flexible quantum dot solar cells (efficiency 30% or more)
Thermo Generators	Microgenerators based on Seebeck-effect (Micro-peltiers) are**available**, other concepts (e.g. bimetal) have low efficiency	*no forecast available*	
Micro combustors	Low efficiency	*May not be commercialized due to low efficiency, no reliable forecast available*	
Beta Radiation Converters	Very promising, under research	R&D	Nuclear Battery available for nano/picosats ?
Fuel Cells	Very promising, strongly supported by German Government, First solutions available (e.g. SFC 50W)	Portable Fuel Cells (Methanol-DMFC) for mobile phones and laptops	Fuel Cell adaptation for nanosats ? (Issues: propellants, waste gases)
Packaging and Materials	**2005**	**2010**	**2015**
Ceramic Packaging	LTCC MCM, standardized package interfaces (e.g. Match-x), 3D System in a Package		LTCC Pico-satellites
Polymer Packaging	Polyimide Substrates, 3D System in Package	Liquid crystal polymer (LCP) high density integration	
Carbon Nanotubes	CNT Transistors in research (Infineon), CNT Composite Materials in research	CNT Composite Materials	CNT Electronics, NRAM (nano-RAM)

24 Annex H. Prospects and Recommendations Mindmap

Figure 32 on the following page shows the mindmap that summarises the outputs of the Prospects and Recommendations workshop. It breaks the workshop ideas into a number of high-level ideas such as payload, communications, orbit, reliability, etc - and thence a further break down into lower level ideas. A key example is use of new technologies, which breaks down into wireless communications, autonomy and, most importantly, the generic use of MNT.

The mind map also prioritises these ideas with respect to the development of the prospects and recommendations report. To do this the branches of the map are marked with a priority value from 1 to 3. 1 is the highest priority indicating issues that must be discussed in some detail and where recommendations should be made. 2 is a middle priority indicating issues where some discussion is needed but recommendations are not expected. 3 is the lowest priority - issues that should not be forgotten but can be summarised with little discussion.

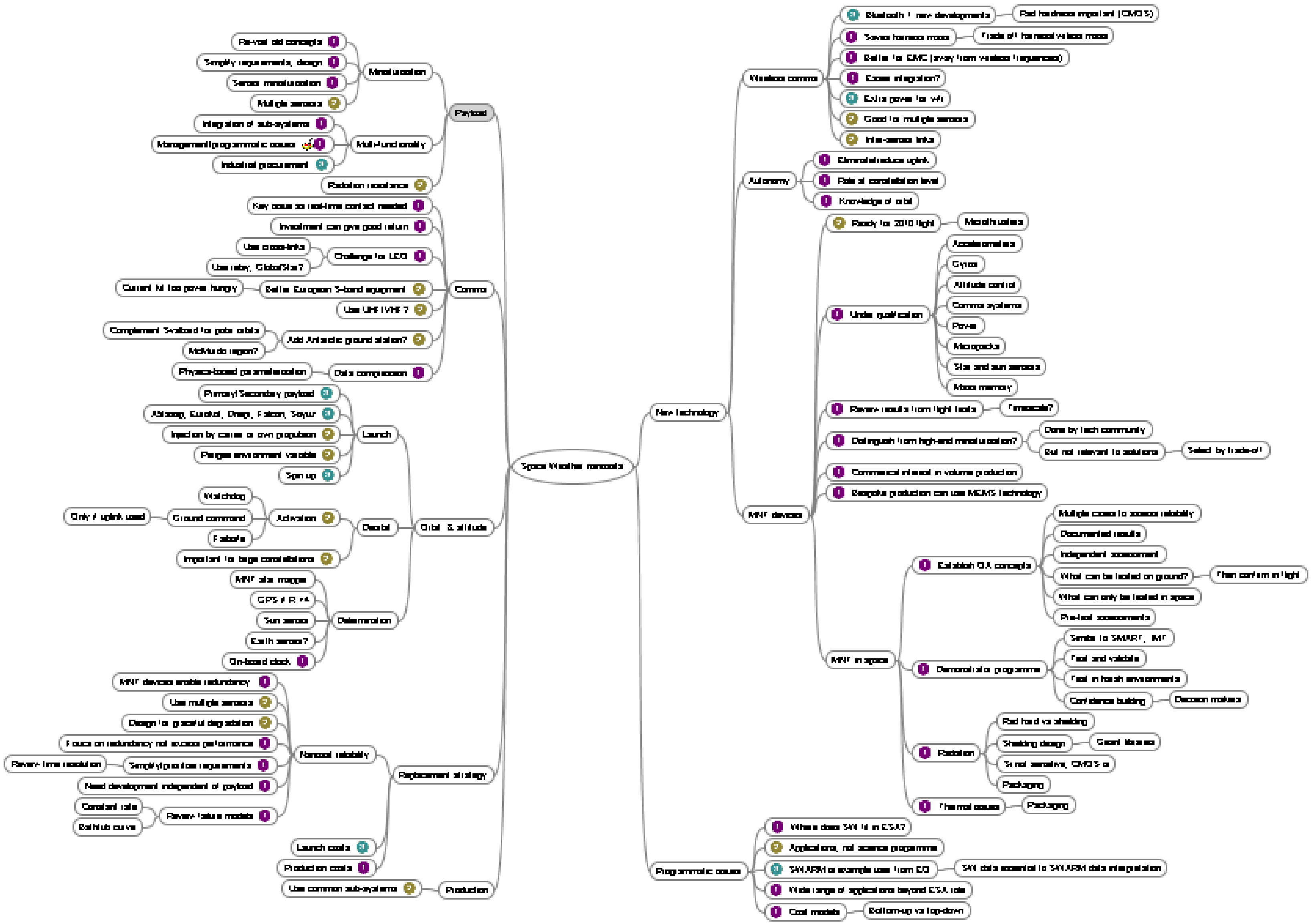

Figure 32. Mind map from SW nanosat prospects and recommendations workshop, 6-7 October 2005

25 Annex J. Nanosat mission objectives

Section 5 presented a short review of existing concepts for future nanosat missions. This annex provides a short summary of the objectives of those future missions. We present three tables: Table 54 covers state-of-the-art concepts, whilst Table 55 covers future concepts out to 2015 and Table 56 covers very long-term concepts.

Table 54. State-of-the-art Micro and Nanosatellites under 30kg

"Aerospace Corp. Cubesat"	No Info
AASUSAT-II	Establish one-way communication with the satellite, and then establish two-way communication. Two science experiments, an ADCS system and a gamma ray detector, ADCS system is to detumble the satellite, but it can also actively control of the satellites attitude in space, utilizing coils and momentum wheels. The gamma ray detector is made by DSRI and it will detect gamma ray bursts from outer space.
Arizona cubesats/RINCON 1	sophisticated, low-power beacon board that was produced by Rincon Research Corporation and provides a redundant means of relaying sensor data in analogue form. This data can be compared with the digital forms, sent by the primary transmitter.
ATMOCUBE	The goal is to build a precise map of the Earth magnetic field and of the flux of radiation impinging on the instrument, which is related to Space Weather effects.
Bluesat	enhanced undergraduate experience and Progressive improvement of capabilities. GPS, Imager, Lexan Experiment (test the UV resistance and durability of a piece of specially coated Lexan)
Cal Poly Picosatellite Project (PolySat) 1 and 2	CP1 - Intended to be a 30-60 day mission. Will operate simplex on 70cms flight testing a sunsensor, magnetorquer and bus system CP2 - Intended to be a 90 day mission. The space station will provide station housekeeping data. This includes temperature, voltage, and current levels. The payload is a surge energy source. Data will be collected on payload temperature, energy, and power levels
Can-X2	atmospheric pollution studies and are capable of determining ground pollutant concentrations with one-kilometre resolution. CanX-2 is also equipped with a SFL-designed nano-propulsion system
Can-X or BRITE	make photometric measurements of bright stars to help calibrate the cosmic distance scale and provide information on long term variability in these stars.
Compass one	carry a miniaturized CMOS camera module on board as primary payload, which is capable of taking pictures of the earth
Cube-II PRISM	obtain images with 10m-class resolution at drastically low cost

Cute1.7	Facilitate future microsatellite development by demonstrating a new design methodology. To realize it, there are three aspects to consider: 1)reliable use of high performance and low cost commercial devices in space, 2)science mission, and 3)satellite disposal after the end of mission
Delfi-C3	Test-bed for thin film solar cells and wireless on-board communication
Hausat 1	Will gather and downlink satellite status data including GPS position information. Also experimental solar panel deployment mechanism and Li-ion battery cells
ICECube 1 & 2	investigate ionospheric scintillations using GPS signals
ION	Test an experimental lowthrust, electric propulsion system which was designed and built in a joint effort with Alameda Applied Sciences Corp (AASC). The second primary mission is to utilize a Photomultiplier Tube (PMT) to observe airglow phenomenon in the Earth’s upper atmosphere. There are also several other o
IRECIN	Includes two areas of interest - a Debris Measurement System composed by two couples of piezoelectric sensor units detecting particle impacts and an electronic board to store relevant data information. Also a seemingly real time ground segment architecture
Katysat	KatySat will have two communications systems: a higher bandwidth (greater than 9600 bps) S-band transceiver (http protocol), and a lower bandwidth UHF/VHF full duplex transceiver (AX.25 protocol) for amateur radio communications. KatySat will also carry various sensors such as radiation, magnetic field, and optical (camera) sensors. Large data files can be stored aboard an MMC flash memory card.
KUTESat Pathfinder	The primary mission of this satellite is to measure the radiation in LEO and take photographs with an onboard camera.
Mea Huaka'i	active antenna will be flown to determine its feasibility for use in space
MEROPE	measure radiation in the Van Allen belts
Munin	Auroral research - A testbed for miniature, autonomous, monitoring satellites, in addition to being a proof of concept mission precursor to constellation missions.
Ncube 1	Student Education
NCube-2	Student Education
PACE	Demonstrate 3-axis stabilisation and MEMS
Project Cubesat	Provide the Swiss Federal Institute of Technology Lausanne and its partners with a independent low-cost satellite system capable to operate small scientific or technological payloads in low to medium earth orbits
QuakeSat	Collect from space the extremely low frequency (ELF) radio signals that are earthquake precursors. QuakeSat had to be large enough to include a one-foot-long magnetometer that extended on a telescoping boom.
SACRED (or Alcatelsat?)	measure the total amount of high-energy radiation over a two-year span and will test four commercial integrated circuit components for their radiation hardness, functionality and annealing properties.

SEEDS	communication with the amateur ground stations, the sensing of the satellite housekeeping data, and the analysis of its orbit and attitude. test a 3 axis geomagnetic sensor and 3 axis gyros to measure satellite orientation
Space Technology 5 (ST5)	ST5 will test multiple technologies, including space weather instrumentation that may enable the success of future nanosatellite missions.
UWE-1	Test adaptations of Internet protocols [such as: TCP (Transmission Control Protocol), UDP (User Datagram Protocol), STCP (Stream Control Transmission Protocol)] to the space environment, characterized by significant signal propagation delays due to the large distances and much higher noise levels compared to terrestrial links
XI-V	demonstration of newly developed CIGS (Cu(In,Ga)Se2) solar cells in space. GaAs cells are also tested on XI-V, which will be used on ISSL's next nano-satellite "PRISM".
YAMSAT-1A	obtain space qualification for spacecraft components constructed in Taiwan. Fly a MEMS spectrometer to measure the sunlight scattering spectrum from the atmosphere

Table 55. Future nanosat concepts for the near/mid term (around 2010/2015)

Mission	**Mission Objectives**
Aerospace Corporation Glass Satellites	The tiny glass satellites have the potential to cut the high costs and lengthy production times associated with most present-day satellites and could spawn a new generation of missions. “They can be mass-produced inexpensively and mass-customized
APIES	characterize a statistically significant sample of the Asteroid Belt, measuring mass & density and imaging over 100 asteroids during its 6 year mission
CanX4 and CanX5	this pair of nanosatellites will demonstrate actual formation control and inspection manoeuvres. The focus is not on specific applications of formation flying or servicing, but on demonstrating accurate relative position and attitude determination and control, as well as inter-satellite communications on a nanosatellite platform.
EADS Micropacks	Multi-application highly integrated system package
Hausat 2	Includes a Electric Plasma Probe, Animal Tracking System, Star Tracker and Space-borne GPSR
Magnetospsheic Constellation (MagCon)	The main objective of the MC mission is to determine how the magnetosphere stores, processes, and releases energy derived from the solar wind interaction, accelerating particles that supply the radiation belts.
Mcubed (CrossScale)	examine the links between different spatial scales in and around the terrestrial magnetosphere: M3 or CrossScale. Inner tetrahedron, i.e. spacecraft with electron scale separations, do not need a full complement of ion instrumentation, and is expected to be a nanosat. The other tetrahedra (to examine ion scales and to examine MHD scales are expected to be microsats.
Munin-X	The Objective is to develop spacecraft swarm technology for multi-spacecraft missions.
Mustang 0	fly Mustang 0 in space and demonstrate new technologies such as MST
NanoSpace 1	qualify this new breed of spacecraft together with individual functional microsystem modules or subsystem
Palmsat	a picosatellite which has a number of applications when launched as a swarm or alongside a mothercraft.

Mission	**Mission Objectives**
PETSAT Project	how to deploy small satellites into a larger one on orbit, and developing nano-scale Earth observation system suited for nano-satellites.
SCOPE	The main purpose of this mission is to investigate the dynamic behaviours of plasmas in the Terrestrial magnetosphere that range over various time and spatial scales. The formation consists of one large mother satellite and four small daughter satellites. Three of the four daughter satellites surround the mother satellite 3-dimensionally maintaining the mutual distance that ranges between several km and several thousand km (variable). The fourth daughter satellite stays near the mother satellite with the distance between several km and 100km.
SNAP2+/Proba 2.25	to investigate and demonstrate some of these formation flying techniques.
Solar Kite	multi-spectral lunar surface imager (Kingston) carry out magnetospheric (in particular magnetotail) measurements (Surrey)
SWARM	direct understanding of the three-dimensional dynamic response of the Earth's magnetosphere as it is buffeted by variations, both dramatic and subtle, in the solar wind.
TEST – Thunderstorm Effects in Space: Technology	implement a new highly modular satellite bus structure and common electrical interface.