ORWELL
Demonstrator

Summary of the Group Design Project
MSc in Astronautics and Space Engineering
1998/99
Cranfield University

College of Aeronautic Report 9918

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Dr. Stephen Hobbs,
College of Aeronautics, Cranfield University

Ray Turner,
Space Science Department, Rutherford Appleton Laboratory

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S.E. Hobbs, R.F. Turner

Abstract

ORWELL Demonstrator, the group design project for the MSc in Astronautics and Space Engineering 1998/99, is a demonstrator for an Earth observation (EO) system whose objective is to provide a commercial service complementing current and planned EO systems. Rapid response and low cost are the main mission drivers. The baseline developed uses a constellation of twelve satellites in four planes for the full system, and one of these four planes (with three satellites) for the demonstrator. The payload proposed is a lightweight low-power synthetic aperture radar (SAR). The SAR is technologically demanding but offers the possibility of all-weather 24-hour imaging which is critical for fast-response imaging. A standard minisatellite bus (the SpectrumAstro SA200) is proposed for use in the mission.

The report summarises the results of the project and includes executive summaries from all team members. Further information and summaries of the full reports are available from the College of Aeronautics, Cranfield University.

Acknowledgements

A project like this depends on input from a wide range of people, who all deserve acknowledgement. The work presented here is primarily that of the Astronautics and Space Engineering students for 1998/99 - which was a lively group to work with. Research students and staff (Tom Bowling, Cédric Seynat and others) have helped significantly, as have the many industry contacts around the world who responded to students' questions patiently, and often with enthusiasm; we gratefully acknowledge their input to the project.

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1. Introduction

ORWELL Demonstrator is the group design project for MSc students on the Astronautics and Space Engineering course for the year 1998/99. The project's aim is to develop an outline design for a demonstrator for a commercial Earth observation (EO) satellite system. The project name, chosen once the baseline had been defined, is an acronym for Observing Radar for (near) Whole Earth Low-cost Looking, inspired by George Orwell's description of Big Brother.

Earth observation is a part of the space business on the verge of commercial viability. Several EO programmes are well established (but not fully commercial), e.g. Landsat, SPOT, NOAA AVHRR, and the current trends in EO are towards increasing spatial resolution of optical sensors (approaching 1 m resolution), and much greater use of radar. Radar has the advantage of being able to image the Earth's surface in all weather conditions and during day and night. However, radars are more complex and expensive than optical imagers, and radar images can be less intuitive to use than conventional optical images. The usefulness of EO data is unquestioned; but whether any EO services (e.g. high resolution optical imagery) will ever be fully commercially viable is much less certain. Within the next five years though, several organisations plan to be operating commercial EO systems, so this question could soon be answered.

Mission Objectives

The aim of developing a commercial EO system for launch within about 5 years (i.e. on a relatively short timescale) led, after a survey of other EO missions, to the identification of rapid response high resolution imaging as a potential market which is currently poorly served. The objectives of the demonstrator were then identified as

- demonstrate the necessary technology for a commercial EO system capable of rapid response (< 6 hour) high resolution (< 10 m) imaging

- complement existing EO systems

- the demonstrator should provide a service useful in its own right (but need not be self-supporting commercially)

As discussed below, rapid response high resolution imaging led to synthetic aperture radar (SAR) for the payload. In parallel with this, commercial viability requires a low-cost approach wherever possible. The student team developed a credible outline mission design which by selecting a radar payload is in tune with the thinking of organisations such as NASA's Jet Propulsion Laboratory.

Report Structure

The remainder of the report summarises the work of the student team. The next section describes the organisation of the project, and chapter three covers the technical work by discussing the key decisions, and presenting the mission baseline developed. The final chapter reviews the project and its initial objectives in the light of the student team's findings. Executive summaries from all the student report are contained in Appendix B and give greater detail for the work carried out.

2. Project Organisation

The group project runs from October to March and accounts for 25% of the MSc course. Each student contributes about 500 hours. The project is directed by staff (Ray Turner, Rutherford Appleton Laboratory, Dr. Hobbs, Course Director) and supported by staff and research students. Educationally, the project is a key element of the MSc course in that it demonstrates in a relatively realistic environment much of the material taught on the course and gives students training in the sort of project work that, for many of them, will be their working experience on graduation. The project can also be very rewarding for the real progress that is made by the students on a realistic design task.

In broad terms, the period from October to December is used to determine the top level system design, and from January to March most of the detailed design work is carried out to refine the mission baseline. The first few weeks were spent considering possible applications for a commercial EO system. A wide range of options was considered (land, ocean, atmosphere applications, low and high spatial resolution), and existing systems were reviewed. After this phase it was possible to define a set of specific objectives for the student project based on a reasonable understanding of the project's technical and commercial context:

- The design should be for a demonstrator mission leading to a commercial Earth observation system.

- The mission should complement existing and planned EO systems.

The next phase was to develop an outline for the system baseline design, which was reviewed formally at the Interim Design Review in early December. Further refinement and detailed design for the baseline took place during the period January to March 1999. The project ended with a formal final presentation made before an invited audience of industry representatives in early March.

The student team was organised into several subgroups to tackle the project. The System Design group's task was to coordinate the team, while the subsystem groups (Payload, Data, Spacecraft, Orbits) carried out most of the detailed technical studies. Team working is important in a group project, and the student group used work packages (listed in Table 1) to structure their activity.

The output from the study is a set of design reports (one from each student) covering all the project work packages and a summary document (parts of these and slides from the final project presentation can be accessed on the College Web pages, http://www.cranfield.ac.uk/coa/coa_grad.htm).

Mission	Subgroup	Work Packages
ORWELL Demonstrator	1000 Instrument Design and User Requirements	1010 Instrument Selection and Characterisation
		1020 User Requirements
	2000 Data Transmission and Dissemination	2010 Communication
		2020 On-Board Data Handling
		2030 Data Dissemination
	3000 Spacecraft Design	3010 Spacecraft Configuration
		3020 AOCS
		3030 Power Systems
		3040 Launcher Investigations
		3050 Thermal Analysis
		3060 Orbital Debris
	4000 Orbit and Constellation Design	4010 Preliminary Orbit / Constellation Concept
		4020 Final Orbit and Constellation Configuration
		4030 Launch Vehicle Investigations
		4040 Orbit Perturbations and Radiation Environment
	5000 System Design	5010 Project Management
		5020 Financial Assessment
		5030 Law and Policy Considerations
		5040 Mission Reliability

Table 1. Project work package breakdown for the ORWELL Demonstrator.

3. System Design and Proposed Baseline

The detailed design work is described in the individual student reports. This chapter gives an overview of some of the main features of the design.

3.1 Key Decisions

Some of the key design decisions were made relatively early in the project based on a mix of qualitative and quantitative information. An important feature of the design project is that these decisions should all flow from the initial mission objectives. The design emphasis is on the system and not on any individual component, since it is the system's performance that determines mission success. Trade-offs are used widely in the design process to make choices more objective, to document the decisions, and also to deal with information which may be a mix of quantitative results and subjective judgements.

The mission objectives concerned complementing existing EO missions and commercial viability. From the initial study of existing and planned EO systems it became clear that high spatial resolution data were required for most commercial applications and that one area of significant weakness for current high resolution systems was their poor temporal response, i.e. satellites can generally only view a given area on Earth very infrequently (with repeat times of several days at the best). ORWELL thus aimed to provide high resolution data available at short notice for anywhere on Earth. A second implication of the objectives was the need to minimise system cost to strengthen the project's commercial viability.

3.1.1 Payload

The choice of payload was critical since it determined the rest of the system design. The drivers of low cost and rapid response tend to conflict since low cost favours visible imagers while rapid response favours radar (synthetic aperture radar, SAR) because of its ability to image unhampered by cloud or night.

Sensor Type	Strengths	Weaknesses
Visible	Relatively simple, low-cost, lightweight and low power High spatial resolution	Likely to have direct competitors Unable to view at night or through cloud
Thermal Infrared	Passive, therefore low power demand Day and night operation possible	Unable to view through cloud Coarse resolution Detectors require careful thermal design
Radar (microwave)	All weather day / night operation Few direct competitors Radar interferometry adds to value	Relatively complex system; usually expensive and high power Image interpretation less easy than for visible

Table 2. Payload alternatives main strengths and weaknesses.

The key factor listed in Table 2 leading to the selection of SAR as the payload was the ability to image in all weather conditions day and night. Without this ability, the system would be unable to provide rapid response imaging for large areas of the world for much of the time. A key assumption underlying this choice is that current research projects will be successful in reducing the system cost and mass of radars significantly - without success in this area it is unlikely that the system costs could be kept low enough to make the mission commercially viable.

3.1.2 Orbit and Constellation

Rapid response can only be achieved with high resolution satellites if there are several satellites operating simultaneously, since each individual satellite covers too small a proportion of the Earth's surface. A constellation of satellites is thus required. Simple orbit simulations indicated that with about 12 satellites the time to access an arbitrary point on Earth could be reduced to a few hours which seemed acceptable. To minimise the cost of launching and establishing the constellation the satellites were organised in a few common orbit planes rather than each having its own plane; the satellites for one plane could all be carried by one launcher.

Two drawbacks of the constellation proposed are (1) the lack of 100% Earth coverage, and (2) the lack of sun-synchronism. The proposed orbit allows almost 93% of the Earth's surface to be imaged, including all areas of high population density and so was judged to cover enough of the Earth for commercial viability. The lack of sun-synchronism is not a fundamental problem, but does mean that at some point all possible orientations of the spacecraft relative to the Sun will be encountered complicating thermal management and power raising.

3.1.3 General Design Issues

Several general design issues flow from the mission objectives, and were applied in all areas of the project. The requirement for low cost implies the use of off-the-shelf components wherever possible, and that within the project no significant research or development will be carried out. For the purposes of the project however, it was assumed that technology already under development on programmes due to be completed shortly would become available by the time the mission was being built, and so could be incorporated in the design.

3.2 Baseline Mission

The full mission design is described in detail in the students' reports. Table 3 summarises the mission baseline as finally defined. Several issues are not fully resolved, but most of the key features of the mission have been decided and quantified.

Issues not yet resolved include the place in the system where the data processing required to produce the SAR images will be performed, and what policy will be adopted regarding data archiving. A feature of this project is that a number of work packages relate to non-technical issues such as political considerations and marketing. Figure 1 is a general view of the spacecraft with the SAR antenna and solar array deployed (both are aligned along the direction of flight).

Overview	A satellite constellation to provide fast-response high resolution images of (almost) any point on Earth. Uses a synthetic aperture radar for all weather 24 hour imaging, and distributes data by Internet.
Orbit / Constellation	Demonstrator: Three satellites (uniformly spaced in the same orbit plane) in 65.58° inclination circular orbit, h = 564.88 km. Repeats every 48 hr, with full Earth coverage to ±67.82° latitude. Mean coverage gap = 22.50 hr (equator) to 2.67 hr (max. latitude).
	Full constellation: Four planes of 3 satellites (demonstrator forms one plane), based on Walker Delta constellation of 12 satellites, but with ascending nodes at 0.00°, 97.07°, 180.00°, 277.07° longitude to facilitate SAR interferometry. Orbit repeats every 48 hr; mean response time of 5.5 hr (equator) to 40 min (max. latitude). Interferometry possible at 7 - 53 hr delay.
	Launch: Use Zenit launcher with the long launch fairing.
	Lifetime: The design life for each spacecraft is 6 years in orbit.
Spacecraft (Fig. 1)	Spectrum Astro standard mini-spacecraft bus (SA-200HP). 300-500 kg launch mass, 0.8 - 3.0 kW power available
	Modifications to standard bus:
	use conventional solar array (no solar concentrators) aligned for minimum drag
	use triple stacking mechanism for launch
	derate AOCS if cost effective
	use broader beamwidth X-band antenna for data downlink
	additional batteries are carried to increase available SAR duty cycle
	Fuel: 50 kg (6 kg to establish constellation + 44 kg for orbit maintenance during 6 yr)
Payload	Synthetic aperture radar (SAR)
	C-band, multi-polarisation (VV, VH, HV, HH), prf = 1.43 kHz
	65 km wide footprint steerable within 450 km swath; incidence = 45° ± 10°
	Antenna: low mass (42 kg) inflatable, 1.06 x 10.55 m plus 10 cm diameter inflated structure round perimeter
	Electronics: 25 kg, 10 x 16 x 30 cm (based on JPL LightSAR)
	Mean power 200 W, sensitivity (σ^0) = -27 dB, 5 m spatial resolution
Data	Four ground stations (2 in N hemisphere, 2 in S hemisphere) with S-band link (2 kbps) for commands, and X-band (200 Mbps) for data downlink.
	Automated image delivery to customers by Internet.

Table 3. Summary of the baseline system design for the ORWELL Demonstrator.

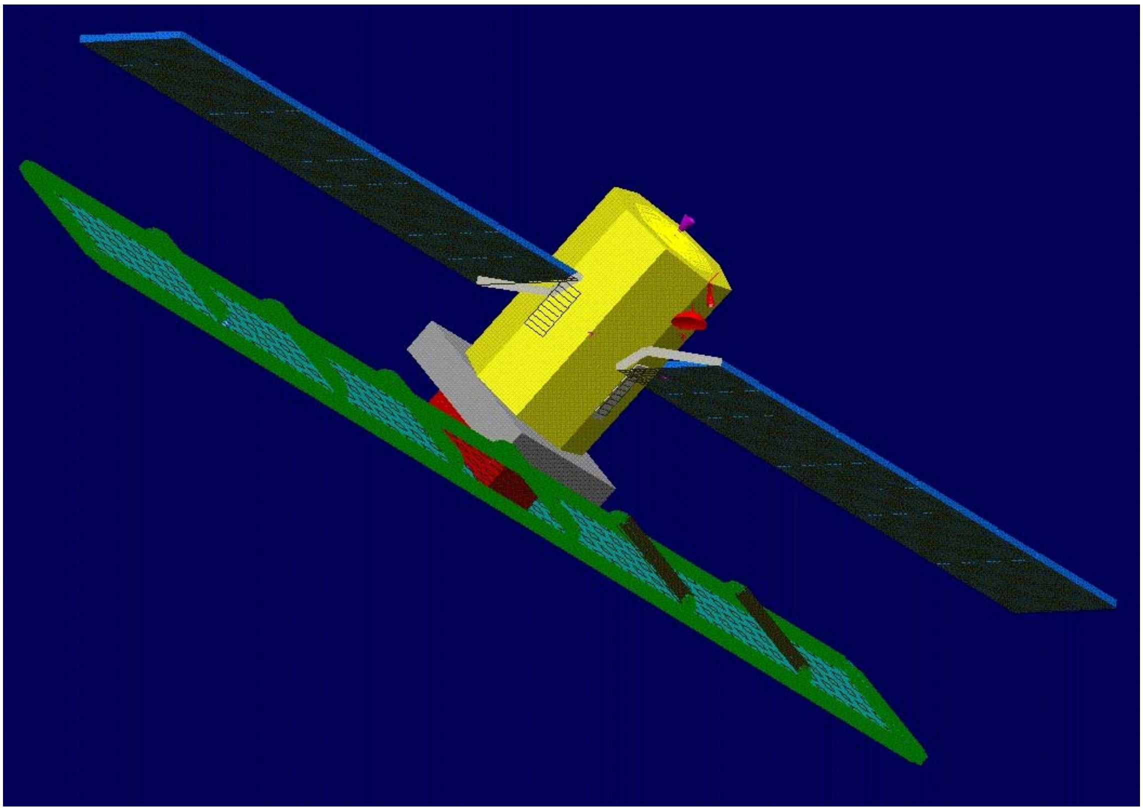

Figure 1. CAD model of the ORWELL spacecraft showing the satellite bus, the solar array (upper panels) and the SAR antenna (lower structure) deployed aligned along the flight direction. The SAR antenna measures 10.5 x 1.1 m.

4. Discussion and Conclusions

Issues raised by a project such as ORWELL go far beyond engineering because of its commercial nature and the type of service it aims to provide. These sections discuss the progress of the project so far, summarise some of the main issues raised, and discuss work remaining to take the project forward.

4.1 Discussion

No fundamental problems were identified with the mission as far as the study has progressed. The underlying logic of a commercial EO system aiming to provide high resolution images of anywhere in the globe at short notice is ambitious but coherent, and the rest of the mission design can be traced back clearly to these requirements.

Further work required to take the project forward falls into three groups: (a) re-evaluate critical mission features, (b) check key assumptions, and (c) complete the study by considering the topics not yet covered.

It is clear that some of the subjects considered are critical to the mission although the initial work suggests there are no fundamental problems. These subjects should be re-evaluated to confirm the initial findings, and possibly consider alternative strategies. Subjects requiring re-evaluation include the orbits (the current proposal is very good on resilience, i.e. the system's capability degrades gracefully if satellites fail, but does not cover polar regions well where sea-ice monitoring is a potential application), power raising and thermal management (both are influenced by the non sun-synchronous orbit), and data handling (the communication links are critical to the proposed service).

The main assumptions made so far concern the performance of the SAR and its antenna and the market for the ORWELL's service. Both these sets of assumptions need confirmation as soon as possible if the project is to continue further.

Topics not yet covered include where the SAR processor would be located (satellite or ground) and the data archiving policy. On-board processing would allow significant (lossy) data compression to be applied before transmission to ground, while ground processing may require data to be transmitted in full. The data archiving policy has also not been considered, even though some degree of archiving is implied by the plan to offer SAR interferometric products. The value of EO data also often lies in having a time series of images of an area, again pointing to the need for data archiving.

For a system like ORWELL to be successful many issues outside engineering need to be favourable - both commercial and political. Initial consideration has been given to several of these - the marketing strategy, and political issues surrounding technology transfer and high resolution imaging (national security, privacy). There are also moral issues concerning the system's use in time of emergency: who would pay for the use of ORWELL in response to natural disasters, what should happen if ORWELL could take images to help relief work but no-one was prepared or able to pay for them? The Internet is proposed as the primary means of distributing images; what policy should be adopted regarding intellectual property rights for the data? These give an indication of the wide range of issues raised by such a commercial EO system.

4.2 Conclusions

The ORWELL Demonstrator project has developed a coherent design for a potential commercial Earth observation mission based on rapid response radar imaging with a small constellation of satellites.

The main topics of further work required to take the project forward are:

- Re-evaluate several key mission features (e.g. orbits, power raising, lifetime, data handling).

- Confirm key assumptions as soon as possible (especially the radar performance and marketing)

- A few subjects not yet considered: data archiving and implementing the SAR processing

The main uncertainties probably lie in the area of commerce (and politics?) rather than engineering - a system like ORWELL appears to be technologically feasible, but the commercial viability is much less certain and depends on a wide range of factors.

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Satria Budi, N., ORWELL, A Low-Cost Earth Observation Mission; User Requirements and Commercialisation. Group Design Project report, College of Aeronautics, Cranfield University, April 1999.

All the individual student reports are available from the College of Aeronautics; contact Dr. Stephen Hobbs, Course Director, MSc in Astronautics and Space Engineering. This Summary Report is available on the Web pages of the MSc in Astronautics and Space Engineering.

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Appendix A. Project Work Packages

Expanded version of Work Package table:

ORWELL Demonstrator	1000 Instrument Design and User Requirements	1010 Instrument Selection and Characterisation 1011 SAR 1012 Visible 1013 Infrared 1014 Other 1015 Development of SAR 1020 User Requirements
	2000 Data Transmission and Dissemination	2010 Communication 2020 On-Board Data Handling 2030 Data Dissemination
	3000 Spacecraft Design	3010 Spacecraft Configuration 3011 Platform Survey 3012 Platform Trade-off 3013 Platform Modifications 3014 Inertia and CoM study 3015 Payload Integration 3020 AOCS 3030 Power Systems 3040 Launcher Investigations 3041 Launcher Survey 3042 Launcher Trade-off 3043 Stacking, Integration and Multiple Launch Capability 3050 Thermal Analysis 3060 Orbital Debris
	4000 Orbit and Constellation Design	4010 Preliminary Orbit / Constellation Concept 4020 Final Orbit and Constellation Configuration 4030 Launch Vehicle Investigations 4040 Orbit Perturbations and Radiation Environment
	5000 System Design	5010 Project Management 5020 Financial Assessment 5030 Law and Policy Considerations 5040 Mission Reliability

Table A1. Project work package breakdown for the ORWELL Demonstrator.

Appendix B. Executive Summaries from Individual Reports

This Appendix contains the executive summaries from each of the student reports. The summaries have been reorganised slightly to present related work packages together in a logical manner.

The summaries have been only lightly edited. The reports have been examined and any major errors identified have been corrected. However it is not possible to guarantee that no errors remain; users of these summaries and the full reports should bear this in mind.

A list of the summaries and their contents is given on the following pages.

Topic (Appendix reference)		**Report author(s)**
System Overview (B 1)		Jradeh, Lumley
Spacecraft Subsystem (B 2)	S/c bus selection (B 2.1	Edge
	Launch (B 2.2)	O'Donnell
	Payload (B 2.3)	O'Donnell
	Thermal Control (B 2.4)	Edge
	Power s/s (B 2.5)	Coulson
	ACS (B 2.6)	McGee
	Orbital debris, deorbit (B 2.7)	Edge
	Space environment (B 2.8)	Dyer
Orbits (B 3)		Hebden
SAR Design (B 4)		Ben Asker, Domps
Data Links (B 5)	Communication s/s (B 5.1)	Bitton
	OBDH (B 5.2)	Bitton
	TT&C, Data dissemination (B5.3)	Hand
Commercialisation (B 6)		Satria Budi

Table B1. List of subsystem areas and the corresponding project reports.

Appendix B.

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Appendix B.1.Systems overview

(Abdo Jradeh, Mark Lumley)

Appendix B.1.1.ORWELL mission objectives

It is proposed that a flotilla of low Earth orbiting spacecraft be established with the purpose of generating low cost Earth observations upon demand from the public or interested parties.
An investigation by the project team produces two main objectives from this,

- **To produce an Earth observation system with commercial viability**
- **To establish a new market for high temporal resolution images**

From a consideration of the requirements that these objective imposed, the main mission drivers were documented:

Primary Driver :	A Low Cost Approach
Secondary Drivers:	Maximise coverage of the Earth
	Minimise target acquisition and re-visit times
	Minimise data processing and delivery times
	Maximise resolution
	Maximise the mission lifetime
	Maintain the tight project schedule

These would drive the decisions and trade-offs to produce the baseline mission architecture.

It was quickly decided that the best way to achieve these objectives was to design a constellation of satellites that would complement rather than compete with existing Earth Observation systems. It would be capable of a wide range of applications, have fast global access times, short repeat times and would be able to operate and take pictures at any chosen moment.

There are many existing Earth Observation systems in operation; to design a mission to directly compete with these spacecraft would result in another competitor for an already small market. The Earth Observation market is very small, with world-wide revenue in 1995 of about 600 million ECU. Of this, only 60 million ECU was raw data sales. Demand for current systems is expected to rise by approximately 15% by the year 2000, leaving the market woefully small. However, there seems to be a latent mass market that could be tapped provided a new customer base could be found.

Appendix B.1.2.Technology demonstration

The technology demonstrator mission concept was developed due to the vast possible market for images. The user requirements were hard to identify, so it was proposed that emphasis be put on the instrument development. A trade-off was performed for different demonstrator mission configurations, the main contenders being a single large satellite (100kg) with 3 instruments on board and 3 mini satellites (100-500kg) each with a single satellite, orbiting in the same plane. The latter configuration won due to it being one plane of the proposed 4-plane final constellation. All necessary technology would be demonstrated and the products could test and create the market for the images.

The full constellation would consist of twelve SA-200-HP spacecraft each carrying a SAR. The orbit configuration would be four planes of three satellites in a Walker Delta Pattern, communicating with four ground stations distributed with two in each of the northern and southern hemispheres. The first objective of the mission would be to launch a technology Demonstrator mission, of three spacecraft, which would become the first plane of the constellation. This Demonstrator would test all the systems required to operate the full constellation, and begin to develop the customer base for the final mission, with each subsequent plane launched yearly after the Demonstrator.

Appendix B.1.3.ORWELL subsystems overview

Payload	
Instrument	Synthetic Aperture Radar (JPL)
PRF	1.43kHz
Operating Radar Wavelength	C-band (0.04m)
Noise Equivalent Ratio	-24dB
Power required	
Antenna	83.4 W
Electronics	Not specified
Electronic Beam Steering	From 35° to 55° providing swath scan of 450km
Footprint	Minimum 31.6 km × 31.6 km Maximum 64 km × 64 km
Resolution	5.28m
Pointing Accuracy Required	0.1° for 9.2 s
Data Rate	94.2Mbps
Time taken to shoot picture	9.2 s
Energy required to shoot picture	1840 J
Antenna	Multi-layer, thin membrane, microstrip array
Mass	24.5kg
Area	11.18m^2 (1.06m ×10.55m)
Inflation System	Nitrogen/water inflation, primary structure rigidising
Mass	10kg
Electronics	25kg
Full-polarised antenna	Ensures a better range of applications (mass requirements will certainly be greater but still <100kg), or at least cross-polarisation for Earth Solid applications.
Launch Package	
Launcher	Soyuz or Zenit Russian launcher
Fairing	Soyuz B or Zenit extended length fairing
Configuration	Stacked triple launch
Satellite Bus	
Bus	SA-200HP standard bus
Modification Specifications	
Lifetime	6 years
Power	8 ×12 Amp-hour Nickel Hydrogen batteries
ADCS System	
Mass	12.52 kg
Power Required	52.4 W
ADCS components:	
Four OSSS Bantam Series reaction wheels	Total Mass – 5.6 kg Power requirement – 4.5 W
Three ITHACO Torqrods;	Total Mass – 1.2 kg Power requirement – 2.1 W
Twelve MOOG 58 – 102 Thrusters	Total Mass – 0.18 kg Power requirement – 30 W per thruster per pulse
A CALCORP CALTRAC Star tracker	Total Mass – 3.4 kg Power requirement – 11-14 W (temp. dependent)
An ADCOLE Two-axis fine digital sun angle sensor	Total Mass – 1.06 kg (inc. electronics) Power requirements – 1.8 W
A MEDA TAM- 1 Magnetometer	Total Mass – 1.08 kg (inc. electronics) Power Requirements – 21 to36 Vdc
Total Propellant Required	57.28kg
Thermal SAR Control	No active components: vacuum deposited aluminium layer Active components: a high α/ε ratio coating, keeping the patches cool with insulation / reflective coatings.
TT&C	
Data Storage	31Gbytes
Communication antennae	"Quadro-phyla-helix" or shaped reflector with single horn each using a 200Mbps data rate
Duty Cycle Limit	20% of orbit
Structure & Mechanisms	
Solar Array Configuration	Faces parallel to line of flight, alpha axis drives
Shortened Bus	0.7m long (25cm shorter)

Table 1 Summary Specifications for the Payload and Spacecraft

Appendix B.1.3.1.Instrument Trade-off and Design

A qualitative trade off was performed to decide whether to fly 3 different instruments or not, and which instruments to fly. The demonstrator mission build towards the final constellation. Therefore, flying the same instruments to maximise the temporal resolution was the decision. The instruments included in the trade-off were narrowed down to optical and infrared cameras, and Synthetic Aperture Radar (SAR).

The chosen instrument was a Synthetic Aperture Radar. These are expensive instruments, but the new technology of inflatable structures can greatly reduce the cost of the instrument. The applications of this instrument include environmental monitoring of sea ice distribution, deforestation, monitoring of crop conditions. Interferometry can be used to create accurate three-dimensional terrain maps, calculate height data, or monitor fault lines for movement. Ships can be tracked at sea by their wakes, and oil spills can be easily detected and mapped.

It was decided that provided SAR technology could be advanced so that it could be accommodated on a mini satellite bus, and there were sufficient reductions in cost of this technology during the mission development, then SAR instruments would be flown by the satellites to take advantage of their unrestricted observing capabilities and the diversity of image products that they could produce.

The User Requirements and Instrument Design group developed detailed specifications for the SAR from initial constraints of 200W power, 100kg mass, 5m resolution and swath scan of 450km.

Appendix B.1.3.2.Initial Mass Budget

It was realised that saving propellant would be a key issue and that minimising the satellite mass would be essential for prolonging the lifetime. Therefore an initial mass budget, an estimate based on average spacecraft specifications was developed. The initial total mass was 350kg , refinement of the systems reduced this mass to 289kg.

Appendix B.1.3.3.Modifications To The Standard SA-200HP Bus

The standard bus decided upon was the SA-200HP but modifications to certain systems were needed to refine its performance for our mission. The power system needed higher capacitance batteries to extend the lifetime, the ADCS system could be downgraded to save mass, since the high pointing accuracy was unnecessary, the data handling capabilities needed upgrading due to the large amounts of data produced by the SAR, and the propellant tank needed resizing to accommodate the extra fuel required to maintain the accurate orbit station keeping needed for interferometry.

Appendix B.1.3.4.The Propellant Budget

The propellant budget proved to be the critical factor as it was here that the limit on the spacecraft lifetime was set. The initial budget required 106kg of propellant to achieve the 6 year lifetime, so methods to save porpellant had to be devised. Orbit maintenance was by far the most significant factor, the accuracy of station keeping needed to allow interferometry necessitating frequent re-boosting. Fuel was saved using the following measures:

- By changing the solar array configuration, so that they were aligned with the direction of travel, the ballistic coefficient was increased, and drag reduced.
- Increasing the catalyst temperature of the rocket motor could improve the specific impulse of thruster from a 225 average Isp to a 250 average Isp, thus making more efficient use of the fuel.
- Avoiding periods of high solar activity and the high levels of atmospheric density associated with them reduces drag further. This led to the recommendation of the year 2003 launch date.
- Modifying the ADCS
- Revising the De-orbiting Procedure

The final propellant mass required was reduced to 57kg. Necessitating the redesign of the standard fuel tank.

Elements	Kilograms of propellant used
Constellation Phasing	0.5
Orbit Maintenance	39
Attitude Control System	3.1
De-orbiting Procedure	9
Nominal Propellant (sum of the above)	51.6
Margin (minimum required ≅10%)	5.16
Residual (unavailable from tank ≅1%)	0.52
Total Propellant	57.28

Table 2 : Final Propellant Budget

Appendix B.1.4.Law and Policy Considerations

An investigation in to the political and legal issues that might affect the ORWELL mission produced the following recommendations.

- Predict problems associated with technology transfer and component integration to avoid unexpected delays to the mission schedule.
  - Produce a set of component integration procedures that conform to the constraints of the legislation.
- Ensure that all legal considerations have been addressed so that the project does not become delayed by red tape. Particularly,
  - Obtain export licenses for the American mission components.
  - Apply for government authorisation for the mission and hence register the ORWELL satellites with the United Nations Secretary General.
  - Obtain the necessary insurance cover.
  - Operate in accordance with the Principles Relating to Remote Sensing of the Earth from Space (1986).
- Pay constant attention to trends in national space policy and countries' long-range plans for space. Thus the risk of regulations being developed during project design which could limit the market for remote sensing pictures can be monitored.
- To gain faster mission approval, incorporate a policy of orbital debris prevention and a de-orbiting plan for the end of satellite life into the mission architecture.
- Do not lose the advantage of the lack of remote sensing legislation in this country by using the US government controlled TDRSS system for downlink of the satellite data.

Appendix B.1.5.Reliability Considerations

Detailed information was not available, so the assessment of reliability has had to be limited to a top level approach. To guide decisions about the satellite design a consideration of the basic failure modes of the satellite to see where the single point failures were, was made. It was assumed that the flight tested components (including solar arrays) of the standard SA-200HP bus were, as the specifications stated very reliable. Figures quoted for the bus are probabilities of >0.85 - >0.95 for mission effectiveness. It was therefore the payload and the modifications to the bus that were analysed.

Appendix B.1.6.Cost Analysis

The main areas of cost for the mission were collated and grouped. They were:

- Satellite Bus
- Satellite Subsystems
- Instrument
- Launch
- Insurance
- Operations
- Yearly Upkeep

Once the costs had been identified, a timeline relating the development of the ORWELL mission to the investment required at each phase.

Appendix B.1.7.Revenue Analysis

To begin to provide an estimate of the possible revenue achievable by the constellation, a picture price was set at half that of ERS-1, of 350ECU. A revenue timeline was produced to find the revenue generated by each phase of the mission.

After the 12 mission years, from Hard Start to De-orbit of the fourth plane, a total of 955 million ECU would need to been spent to launch all twelve satellites. Total sales of 1.3 billion ECU would be taken. This results in a profit of 361 million ECU. With the subsequent planes launched yearly after the first, ORWELL becomes profit making during the seventh operational year. A total of 565 million ECU is needed to be invested over six years for ORWELL to become fully operational.

These investments are very high, but the mission is eventually profit making. A variation of the ORWELL constellation would be to launch the first and second planes only. Economically this may be a more favourable choice. The total revenue would be reduced to 770 million, and the profit reduced to 200 million, however, the amount of investment required is now only 190 million ECU, greatly reduced from the previous value for all four planes and the mission becomes profit making after the fourth operational year.

The success of the sales depends strongly on the size of the latent market. ESA suggest it may be as large as 2 billion ECU per year, which would be quite large enough to absorb the sales required to make ORWELL a profit making, viable Earth Observation satellite constellation system.

Appendix B.2.Spacecraft subsystems

Appendix B.2.1.Spacecraft bus selection

(Lucy Edge)

Appendix B.2.1.1.Standard Platform Trade-off

The Spacecraft Design (SCD) group was required to choose a standard satellite bus that could accommodate a SAR. An appropriate launch method was required to place three satellites into the same plane. Low cost was a strong mission driver. To launch at 2003 and take advantage of the quiet period between two solar maxima a rapid design and manufacture process was required. These two factors, along with the fact that there was no chosen instrument to design the bus around, led to the decision to use a standard satellite platform.

A survey of medium, mini and micro satellites (as defined by Surrey Satellite Technology Limited) was carried out. Micro-satellites were ruled out as they would not be able to carry any high quality imaging payload. Initial studies suggested that the medium satellites (500-1000kg) would be beyond the budget imposed by the Systems group. Overly specific mini-satellites were also removed from the list of contenders.

A trade-off was carried out of the seven remaining mini-satellites. This was done by assigning a score to each satellite in four areas: power capability, mass available for payload, pointing accuracy and cost of the cheapest launcher that could house three satellites in its fairing. A analysis was carried out whereby these parameters were weighted differently to see the effect. Figure 1 shows the results of three such parameter weightings that are listed below.

- no bias applied to scores
- power bias
- mass capability bias
- pointing accuracy bias
- launch cost bias
- power and launch cost bias
- power and launch cost bias with mass capability weighted slightly higher than launch cost

These weighting options are condoned in the report.

The proposed bus was the SA 200-HP from **SPECTRUM**ASTRO. It achieved the top score overall and was the optimal bus in four of the seven categories.

Appendix B.2.1.2.Adaptation of the SA 200-HP

Once an inflatable SAR was selected, its requirements could be compared with SA 200-HP. The platform's pointing and mass capabilities were superior to instrument needs. The attitude control subsystem was therefore redesigned to help keep within the cost budget.

A late specification from **SPECTRUM**ASTRO demonstrated that the bus dimensions had not included the propulsion tank. The chosen Soyuz launcher could not contain three satellites of the new dimensions and the triple launch interface. A shorter bus, as recommended by **SPECTRUM**ASTRO was modelled on CATIA in order to carry out a centre of mass analysis. With both full and empty fuel tanks it was found that the centre of mass remained reasonably central. The vertical shift in the z-axis from the full tank scenario to the empty tank case was 27cm.

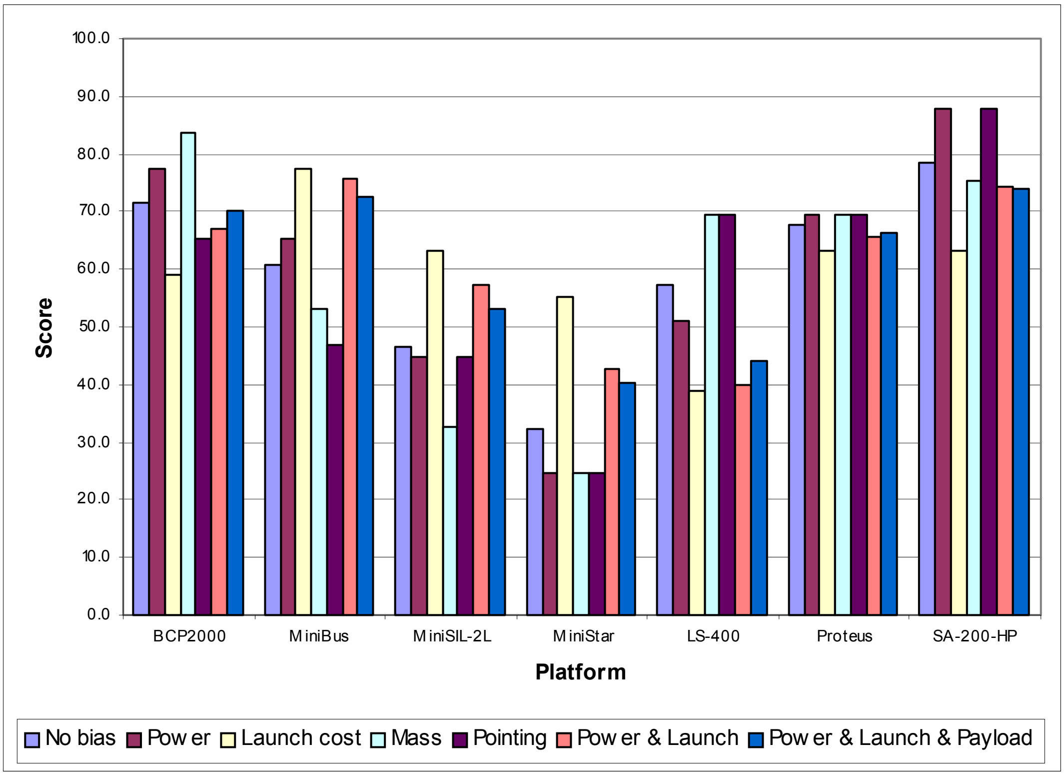


Figure 1 Results of the spacecraft bus trade-off

Appendix B.2.2.Launch

(Kathryn O'Donnell)

Appendix B.2.2.1.Launch package selection

The selection process used to define the launch package baseline ran concurrently to that determining the satellite platform assuming that each plane of the constellation would be achieved through an exclusive triple launch of three spacecraft. It eliminated options not capable of transporting the mass of three small spacecraft to the required LEO with a margin for multi-launch mechanism, and those not volumetrically capable of housing three of the small spacecraft considered in the final satellite bus trade-off. A systems budget imposed an upper price limit of $30M on the launch package and this was considered with the reliability of the launch options and their ability to launch directly into the required inclination orbit to further optimise the selection. Finally, interjection of the satellite trade-off result lead to the determination of the launch package baseline: a stacked triple launch from a Soyuz-B fairing.

Appendix B.2.2.2.Launch configuration

At a late stage in the study it was realised that erroneous dimensions had been used in the launch vehicle fairing calculations due to the omission of the fuel tank in data released to the group. As a result of this the initial launch baseline appeared no longer viable and several options to proceed were considered. These included:

alteration of the launch vehicle fairing
re-iteration of the launch vehicle selection process
re-configuration of the instrument to the spacecraft
re-configuration of the satellite platform subsystems.

The viability of each of these options is discussed in the report. It was determined that reconfiguration of the satellite subsystems could reduce the z-dimension considerably and as a result this was adopted as the standard satellite dimension. Even on inclusion of this alteration, the dimensions of the spacecraft remained incompatible for the launch package baseline.

In conclusion a preliminary investigation presented two options:
Re-iteration of the launch vehicle selection process to upgrade, on initial inspection to a Zenit (long) ground launch with standard stacking system at a cost penalty of $5M.
Assumption that further study into the re-configuration of the spacecraft will prove the initial launch package viable with a novel stacking system.

Appendix B.2.3.Payload

(Kathryn O'Donnell)

Appendix B.2.3.1.Payload Determination

In view of conventional SAR dimensions and masses an investigation into developing low-mass SAR technology was necessitated. This highlighted two potential options based on designs produced by the Advanced Radar Technology Program (ARTP) at the Jet Propulsion Laboratory:

- A low-mass hinged folding array composed of graphite composite panels folding out from a central support platform.
- An inflatable antenna composed of a rectangular primary inflatable support structure deploying from a rolled-up stowed geometry on a central support platform.

Both technologies incorporated apertures of multi-layer microstrip arrays with active patches. Seven performance characteristics were identified to perform a trade-off determining the optimal technology to be used on the SA-200-HP, these fell into two categories:

Those driven by the decision to use a small satellite platform:
- System mass
- Packaging efficiency – volume metric of the stowed configuration

- Those ensuring mission success and commercial viability;
  - Deployment reliability
  - Deployment controllability
  - Performance – aperture performance with each antenna structure
  - In-orbit stiffness
  - Reliability to orbit – how the antenna structure performs in the launch environment.

Successive sensitivity analyses were used to compare the two technologies. From Figure 2 it is clear that the optimal selection was that of the inflatable structure.

Conforming to the instrument requirements and small satellite platform constraints determined the SAR aperture size as:1.06m x 10.55m implying antenna dimensions of 1.36m x 10.85m.

Most of the elements of the ARTP antenna were integral to the structure, therefore in adapting this design it only remained to determine the optimal inflation system, method of controlling deployment and process of rigidization for the primary structure.

Two low-mass inflation concepts were considered; separate systems incorporating low-mass elements and the adaptation of the on-board propulsion subsystem for inflation. The decision to use interferometry in the mission placed stringent station-keeping propellant requirements on the propulsion subsystem and meant that the 4.41kg of Hydrazine required to inflate the antenna was not available. It was therefore decided to use a separate low-mass with liquid-based inflatant, optimally an N_2/H_2O combination. Technology transfer issues arising from ITAR meant only approximate mass values were available for this system, highlighting an area for further investigation. The complexity of such low-mass systems also raised reliability/redundancy issues.

The method of controlling deployment was determined as the use of constant return force springs embedded into the primary inflatable structure due to its simplicity and reliability. This placed stringent requirements on the method of rigidization to be employed as the springs still return a force once the antenna is deployed. The method of rigidization was therefore determined as the use of an epoxy resin which cures through dehydration in-orbit. This was selected as it requires no energy to cure and produces an extremely stiff and strong structure.

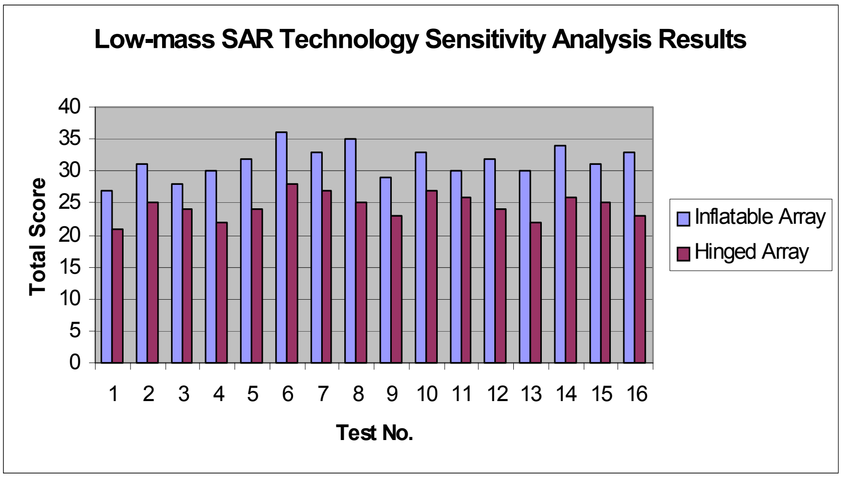


Figure 2 Results of the low mass SAR technology sensitivity analysis

Appendix B.2.3.2.Payload Configuration

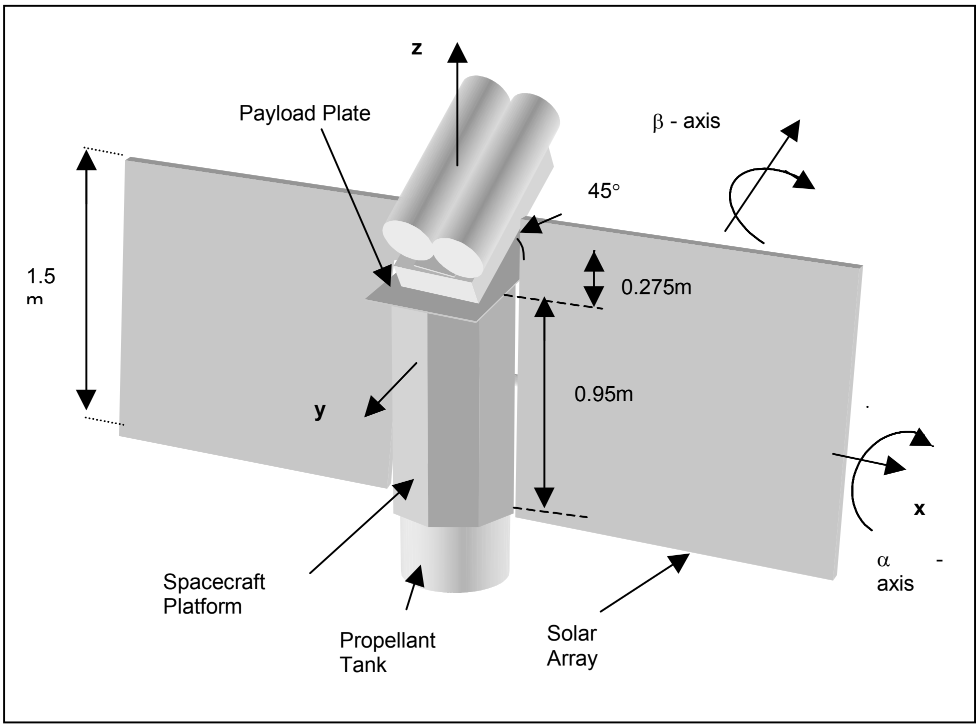


Figure 3 Spacecraft configuration

Having estimated the dimensions of the stowed geometry of the SAR, the main drivers in configuring the inflatable to the SA-200-HP were identified as:

- 45° orientation of the SAR to Nadir to enable electronic steering between 34° and 54.8°
- Consideration of the requirements of data antennae to avoid shadowing, interference etc.
- Possibility of physical interference of the solar arrays and the deployed antenna due to small volume of spacecraft
- Need to centre oriented instrument on the payload plate to maintain centre of gravity as close to the central vertical axis as possible, thus minimising requirements on the AOCS.

A CATIA model was developed to consider the optimal configuration. As may be seen from the simplified version presented in Figure 3, the primary consideration in this was that of physical interference of the deployed antenna and the motion of the solar arrays about the α -axis.

A solution to this was to introduce an extra volume elevating the SAR and central support platform above the motion of the solar arrays. This extra "payload box" was conceptual and could incorporate the composite orienting structure, the SAR inflation system and any additional features required on the bus, subject to thermal requirements and limitations of the AOCS.

Appendix B.2.4.Thermal Control Subsystem

(Lucy Edge)

Thermal control of the SAR and its inflation tank were considered in this report. As a standard bus was being used, the thermal control of all included subsystems was accounted for. The interface with the payload is highly insulated.

Worst case hot and cold scenarios were envisaged and the resulting SAR temperatures calculated. Technology transfer problems from the U.S. (due to International Trade and Arms Regulations) meant that many of the necessary details for thermal analysis were unavailable. Knowledge of the nature of the patches on the SAR was required as operating temperature ranges vary considerably between active and passive patches.

As the SAR is an inflatable structure it could not be insulated with MLI. Various thermal coatings that could be vacuum deposited were tested to investigate the temperatures experienced.

Thermal control for a passive SAR could be achieved by vacuum deposition of aluminium (α=0.2, ε=0.03) onto the surface. The control of active patches was not so easy. It was not possible to reduce the temperature range to within that required for operation of active patches by using a single material coating. Investigation into combinations of coatings resulted in a slight narrowing of the temperature range between hot and cold cases. It was proposed to keep the SAR body above the minimum operating temperature by using a high α/ε finish. The patches would then be insulated from the main membranes and coated with a high emissivity paint to prevent overheating. Thermal control of a SAR with active patches cannot be achieved without specific development to the instrument during manufacture. The hydrazine inflation system (+7°C to +20°C temperature constraint) will be stored inside the main bus which has a warm electronics box. With slight modification this will be suitable to prevent the fluid from freezing.

Appendix B.2.5.Power subsystem

(Neil Coulson)

Appendix B.2.5.1.Electrical power subsystem design

The electrical power subsystem on a satellite must be able to supply power to all the loads on the spacecraft through the mission life. The EPS interfaces to all the components on the spacecraft which use power. The power subsystem on the bus consists of the solar array, the charge control unit, battery and the power distribution unit. As a standard spacecraft bus was used as the basic platform, it already had the subsystems integrated on to it. The solar arrays were unsuitable for the mission so had to be replaced to give the required power production. The arrays used were Gallium Arsenide, the total area was 13.8m^2 and had a power conversion efficiency of 19%. The chosen low earth orbit means that the radiation incident on the arrays will create degradation in the power conversion efficiency of 2.75% a year.

As the mission is low earth orbiting the spacecraft will normally go through eclipse 15 times a day. The mission therefore requires a substantial amount of energy storage. The batteries used for the mission were Nickel Hydrogen, had an efficiency of 85% and could complete 10000 charge discharge cycles at 60% depth of discharge. The batteries would have to complete 30000 cycles over the mission life therefore a depth of discharge of around 20% was required. Thus the capacity of the batteries on the bus needed to be increased to fulfil the mission life. The final mission proposal had 6 extra batteries on the bus to meet the requirements for the depth of discharge.

The power distribution system uses a regulated bus. The centralised distribution system implies that the power converters are at each load interface.
Power regulation and control is achieved using a charge control unit. This unit keeps the bus voltage at a nominal 27.5V. In times of excess power production the energy is dissipated away from the bus using a direct energy transfer system.

Appendix B.2.5.2.Power requirements

Subsystems	Power (W)	
SAR	200	(per picture)
Download	100	
Data	55	
Thermal	210	(200W eclipse)
Electronics	20	
ADCS	60	
Power	30	

Table 2 Subsystems power requirements

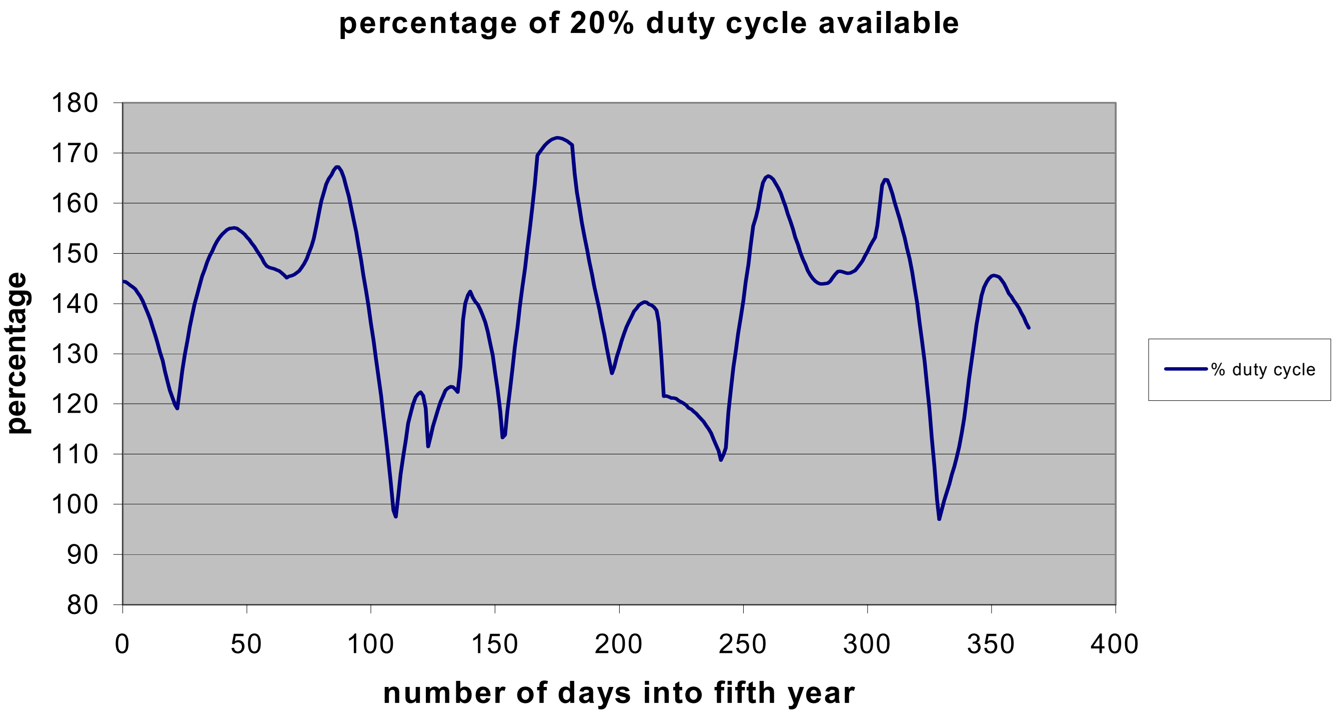
All the power loads of the spacecraft were supplied and it was then possible to model the power usage in orbit, with the aid of orbital data for power production at the arrays and the eclipse times. The degradation rate of the arrays was also supplied so the orbits could be modelled for the complete mission life.

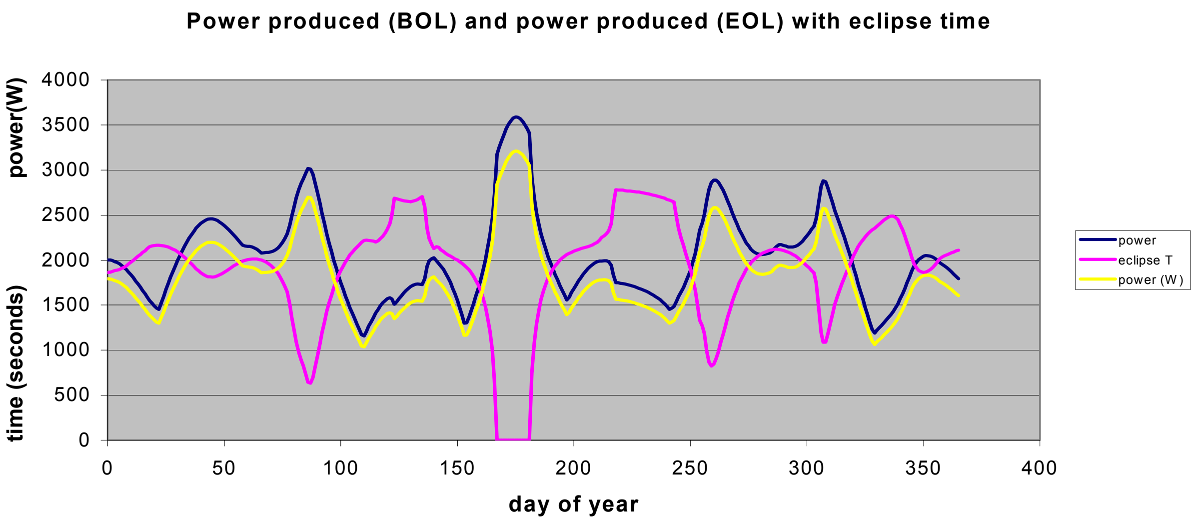
The data supplied enabled a complete analysis into the battery charge discharge cycles, battery reconditioning, rate of recharge, possible duty cycles and the usage and dissipation of surplus power.

The batteries require a period of reconditioning to sustain the capacity and therefore battery life. This could be achieved during the orbital period where there is no eclipse, the batteries can be completely discharged through 18 orbits prior to this period and then a small amount of power supplied to them over this period would allow complete recharge. The power requirement for recharge is a continuous 9W.

The mission could sustain a permanent 20% duty cycle, except for 2 points in the final year of mission life, as shown below. At these points it would be possible to use the batteries to supply the excess energy requirement. The spacecraft could also run a higher duty cycle for many periods in daylight, however a higher duty cycle could not be run during eclipse times as this created an unreasonable demand on the battery.

As can be seen in the periods there a many periods with a lot more available power this can be used to increase the duty cycle or this excess has to be dissipated to stop any unwanted power loads or heating. As can be seen below at BOL there is always a requirement to dissipate energy at a 20% duty cycle, this is due to the fact that the power requirements at the EOL determine the power production at BOL. If it was possible to run up to a 100% duty cycle (where power production allowed) there would still be a requirement for power dissipation at times of peak power production.





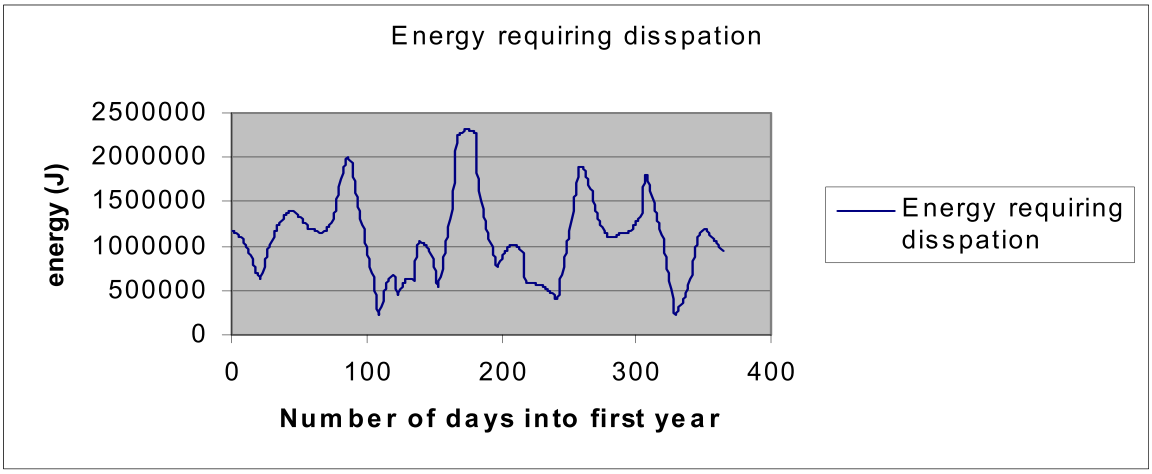


Figure 4 Power susbsystem design

Appendix B.2.6.Attitude control subsytem

(Adam McGee)

Appendix B.2.6.1.Selection of Control Type

The selection of control type is the initial decision, which fundamentally affects the ADCS system. There are three basic categories of spacecraft stabilisation, Gravity gradient, Spin, and Three-axis.

Accuracy was found to be the main driver in the selection process, and therefore to acquire the accuracy desired for the instrument, Three-axis stabilisation was deemed suitable. Three-axis stabilisation requires a better class of Star sensor and possibly gyroscopes, which proved to be some of the drivers when considering selection of the sensors.

Appendix B.2.6.2.Effects of the Disturbance Environment

The effect the satellite's environment has on the craft directly influences the sizing of the actuators. There are principally four environmental disturbance torques, the importance of these torques is directly related to the size of the craft, the mass, the mass distribution, and the altitude. The four main environmental disturbance torques, (worst-case) are as follows;

INPUTS:
Orbit Altitude = 550 km
Orbital Velocity = 7585 m/s
Atmospheric Density = $1.53X10^{-12}$ kg/m^2

OUTPUTS:
Gravity Gradient, $T_a = 7.3X10^{-5}$ Nm
Solar Radiation, $T_{sp} = 7.65X0^{-6}$ Nm
Magnetic Field, $T_m = 4.59X10^{-5}$ Nm
Aerodynamic Torque, $T_a = 2.86X10^{-5}$ Nm

The method to estimate the environmental disturbance is basic and would need updating to improve the accuracy of the results. Many important contributing factors were disregarded due to the simplistic model.

Appendix B.2.6.3.Sizing of the Actuators

Taking the environmental disturbances as the worst-case, allows us to size the actuators accordingly. Each possible actuator was considered and an appropriate sizing calculated. Final choice was made in the actuator trade-offs where drivers such as mass, power, etc. where used to choose the most suitable actuator.

Slew torque from array $T_A = 1.895X10^{-5}$ Nm
Calculated Reaction Wheel Size H = 0.7 Nms
Calculated Momentum Wheel Size = 60.2 Nms
Magnetic Dipole Size Required = 15 Am^2
Calculated Thruster force level = 1.167 N

Appendix B.2.6.4.Considerations for Selection of Sensors

Accuracy, mass, and power were found to be the main drivers when considering an appropriate sensor configuration. Cost could not be used, as no comprehensive list could be obtained for each component. Therefore, necessity of each component with respect to ORWELLS needs had to be considered, these considerations were then applied to the trade-off.

Appendix B.2.6.5.Final Actuator Trade-Off

As no comprehensive costing was obtained, mass and Power consumption became the main actuator selection drivers, redundancy and overall performance characteristics were considered where applicable. Final choice actuators are shown below:

Four OSSS Bantam series Reaction wheels	- Mass = 4X1.4 = 5.6 kg
	- Power = 4.5 W
Three ITHACO Torqrods	- Mass = 3X0.4 = 1.2 kg
	- Power = 2.1 W
Twelve MOOG 58 - 102 Thrusters	- Mass = 0.18 kg
	- Power = 30 W

Appendix B.2.6.6.Final Sensor Trade-Off

Accuracy, Mass, and Power were once again the main drivers affecting the selection process. At least two external measurements were required for full three-axis knowledge and choices were also influenced by redundancy, fault tolerance, and Field of view requirements. Final choice sensors are shown below;

CALCORP CALTRAC Star Tracker	- Mass = 3.4 kg
	- Power = 11 - 14 W
ADCOLE Fine Angle Sun-Sensor	- Mass = 1.06 kg
	- Power = 1.8 W
MEDA TAM - 1 Magnetometer	- Mass = 1.08 kg
	- Power = 21 - 36 Vdc

Appendix B.2.6.7.Final Configuration.

The final configuration weighed 12.52 kg, almost a third of the estimated 30 kg,(6% of the total satellite mass), the ADCS system was expected to weigh, and required a power supply of 52.4 W, falling just within the power budget of 60 W prescribed by power.

An idea of the cost for part of the system can be shown, (omitting the cost for the star tracker and the thruster system), using costs obtained from OSSS. The cost for the ADCS system is approx. $810,000 for the demonstrator and $2.916 million for the constellation.

Appendix B.2.6.8.Conclusion.

It is shown that using new technology, a light-weight, low-power and potentially low-cost ADCS system can be obtained for the ORWELL satellite that achieves the desired accuracy required by the instrument.

Appendix B.2.7.Orbital Debris and End of Life Disposal

(Lucy Edge)

The likelihood of a collision with one of the three demonstrator satellites was calculated as 0.36. This would not necessarily be a catastrophic collision but does highlight the need for spare orbiting satellites if the full constellation is to go ahead. The inflatable SAR has a pressure system to account for small holes in the membrane due to collisions. Loss of a satellite would lengthen the repeat time for imaging.

Three areas were highlighted where this mission must be designed to preclude the creation of further space debris.

- during launch sequence
- final orbit insertion and deployment of relevant hardware.
- EOL disposal of the satellite

Future preventative methods have been discussed for the first two issues.

EOL disposal methods were studied. Deorbit, using the onboard hydrazine propulsion system was proposed as the best method of EOL disposal. This would require 9kg of hydrazine to deorbit to 400km. Once this altitude is reached the solar arrays can be aligned to provide the maximum ram surface area thus increasing the drag and ensuring rapid deorbit from 400km resulting in reentry and burn-up in Earth's atmosphere.

There was not quite enough fuel for all the spacecraft's requirements. EOL disposal methods are predicted to become a legal requirement in the near future. It was decided that a deorbit capability was essential, if this mission was to be considered seriously. This could not be achieved at the expense of mission lifetime so minor resizing of the fuel tank was required.

Appendix B.2.8.Analysis of the Space Environment and its Impact on the ORWELL Mission

(Richard Dyer)

Appendix B.2.8.1.Analysis of the Space Environment

In order to achieve a low cost Earth observation mission, it was necessary minimise the impact of the space environment. By orbiting the spacecraft in a relatively benign environment, it was possible to reduce the environmental protection required by the spacecraft and increase the satellite lifetime. Minimising the environmental protection required by the spacecraft makes the design more flexible and less costly. Maximising the mission lifetime maximises revenue raised by the mission, offsetting design, production and launch costs.

The components of the space environment considered were the neutral environment, the particle radiation environment, the electromagnetic radiation environment and the plasma environment. The effects of each of these environmental components on spacecraft was studied. The variation of these environmental factors with altitude, solar activity and orbit inclination was also investigated. It was found that the only factors that are both harmful and altitude dependant were the neutral environment and the particle radiation environment. Being altitude dependant meant that these environments were avoidable.

It was found that the neutral environment causes drag and, hence, decay of orbital altitude. For a satellite with a ballistic coefficient of 75 kgm-2, having no station keeping thrusters and a required lifetime of 5 years, a 500 km orbit would be required. At this altitude, the effects of atomic oxygen are negligible.

It was found that particle radiation can cause significant and permanent damage to solar arrays and electronics. Orbiting below 850 km avoids the trapped particles in the Van Allen belts and, hence, minimises the impact of particle radiation. Around the South Atlantic Anomaly and the poles, solar flare particles and galactic cosmic rays can effect spacecraft. It was found that these areas can be avoided by selection of a suitable inclination. However, this was deemed inappropriate since the radiation received over these areas does not make a significant contribution to the cumulative radiation dose.

Appendix B.2.8.2.Impact of the Environment on the Spacecraft

It was found that the efficiency of the solar arrays would degrade by 2.5 % per year due to particle radiation. This figure was provided by Spectrum-Astro, the supplier of the spacecraft bus. The other impact of the space environment was found to be drag from the tenuous neutral atmosphere. In order to conduct SAR interferometry, it was necessary to orbit at 564.883 km with a tolerance of +/- 10 m. With the 50 kg of fuel available on the bus used solely for station keeping, it was calculated that the fuel would be expended within 3.5 years.

Appendix B.2.8.3.Maximising the Interferometry Lifetime

Due to volume constraints on the spacecraft bus, the addition of extra fuel tanks for station keeping was ruled out. In order to maximise the interferometry lifetime it was decided that the frequency between re-boosts should be reduced while still maintaining the orbit altitude to +/- 10 m. In order to reduce the re-boost frequency, the ballistic coefficient of the satellite was increased. This was achieved by ensuring the faces of the solar arrays were never parallel to the ram surface of the satellite. A computer programme was written to determine the power produced for different solar array orientations. The solar array orientation chosen produced an average orbit average power of 1834 W, compared with 1384 W produced by the other possible configuration. The configuration chosen also minimised the load on the attitude control system. Maximising the average orbit average powers increases the duty cycle of the SAR. Reducing the load on the ACS increases the fuel available for re-boost and, hence, the interferometry lifetime. Both these consequences lead to an increase in revenue raised by the mission.

The solar array orientation chosen provides the satellite with a ballistic coefficient of 170 kgm-2. This allowed an interferometry lifetime of 6 years.

Appendix B.2.8.4.Optimising the Orbit Average Power Profile

A study was carried out to see if it was possible to reduce the large fluctuations in orbit average power through the year. It was found that the minimums in the profile are unavoidable. The maximums, caused by zero eclipse, were avoided without effecting the average orbit average power by selecting a launch time. Since the final constellation will contain 4 planes, and it is not possible to avoid zero eclipse for more than 2 planes in a Walker - Delta constellation, it was concluded that recommending a launch time was not appropriate.

Appendix B.2.8.5.Re-boost Operations

The implications of the re-boost manoeuvre on satellite operations was considered. However, information about detailed design features of the spacecraft bus was not available. This meant no firm conclusions could be drawn. It was highlighted that further investigation on this subject needs to be done.

Appendix B.3.Orbits

(Richard Hebden)

The ORWELL constellation of Earth observing spacecraft is intended to be an economically viable, competitor in the remote sensing marketplace of the near future. In order to compete, the system needs to possess response times and Earth coverage times better than those of existing systems. The system comprises of two distinct phases: the simultaneous launch of a three-satellite demonstrator mission and the implementation of a twelve-satellite constellation, based on a Walker Delta Pattern.

Appendix B.3.1.Orbit Concept, Baselines and Trade-Off Methodology

The full, twelve-satellite constellation, is based on a Walker Delta Pattern of four planes of three spacecraft, each spacecraft equipped with synthetic aperture radar instruments with swath scans of width 450km. This configuration was developed because it provided the most rapid and efficient Earth coverage, for a relatively low number of spacecraft. Three main baselines were formulated, each providing different mission characteristics. A trade-off analysis was performed to decide upon the baseline which would be fully developed. The main issues in the trade-off were:

- Extent of coverage of the globe by the demonstrator and complete constellations.
- Minimisation of Earth coverage and repeat / response times.
- Incorporation of the facility to acquire both stereo and interferometric SAR data.
- Performance of the demonstrator as a standalone mission.
- Providing high levels of constellation redundancy, to achieve more graceful degradation characteristics.

From the preliminary baselines, one option was chosen for further development. The most important drivers, which influenced this choice and its development, were to strike a balance between achieving efficient coverage and providing a high degree of redundancy for the constellation.

Appendix B.3.2.The Demonstrator Mission Concept

The demonstrator mission baseline consisted of one plane of three spacecraft, with the spacecraft phased 120° apart in true anomaly, within the orbit plane. The orbit was initially selected to be at an altitude of 560km and an inclination of 60°. This inclination was selected because it provided coverage of approximately 90% of the populated surface of the globe. The altitude provided a starting point of 15 orbits per day; 14 orbits per day would result in the spacecraft being in an environment of increased radiation and solar radiation pressure, whereas 16 orbits per day would put the altitude in a region where residual atmospheric drag would be large.

The first stage in the process of designing a repeating ground track constellation (for the purposes of performing SAR interferometry) was provided by starting with an orbit which completed an integer number of revolutions in an integer number of days. An iteration was carried out by modifying both the altitude and the inclination of the orbit to account for the perturbing effects brought about by the oblateness of the Earth. The result of this iteration was a constellation structure, with a ground track repeat of $1^d23^h28^m$. The constellation was also developed with the issues of rapid coverage and graceful degradation in mind. The demonstrator constellation could provide potential complete Earth coverage, between latitudes of ±67.82°, in less than two days.

Appendix B.3.3.Implementation of the Full Constellation

The concept of the straightforward Walker Delta Pattern, with a uniform spread of the ascending nodes of each of the orbit planes, was modified slightly to achieve certain effects. For example, precise arrangement of the position of the ascending nodes provided a short period ground track repeat, between spacecraft in different planes. Increased flexibility for acquiring stereo SAR images was also incorporated into the constellation, as well as considerations relating to the graceful degradation aspects of the system. By having ascending nodes at longitudes of 0°, 97.07°, 180° and 277.07°, a short-period ground track repeat of 6^h24^m was built into the full constellation, as well as the provision for obtaining stereo images at any point on the globe. However, the deviation from the standard format of the Walker Delta Pattern was small. To deviate from this pattern to too great a degree, would have resulted in a very uneven distribution of coverage on the Earth's surface at any one time, and would not have allowed the rapid coverage time provided by the standard Walker Pattern. With the full constellation, potential complete Earth coverage between latitudes of ±67.82° could be achieved in approximately 18 hours.

Appendix B.3.4.Coverage Statistics

A point coverage simulation was performed, by creating a conceptual grid of 61 points on the surface of the Earth. The accesses between each SAR instrument and each grid point were then calculated using the simulation software (Satellite Toolkit). Figure 5and Figure 6 present the coverage gap statistics which were derived for both the demonstrator mission and the complete constellation. The mean coverage gap is essentially the average time between subsequent observations of an arbitrary point at a certain latitude.

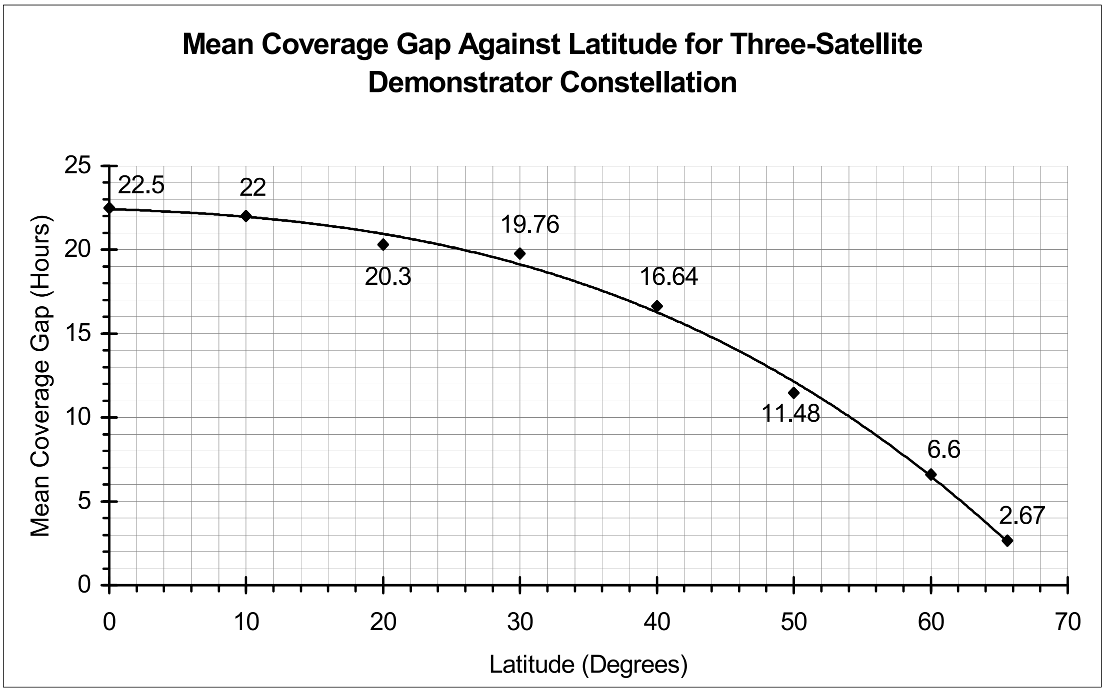


Figure 5 Mean coverage gap plotted as a function of latitude for the demonstrator mission

Appendix B.3.5.Station-Keeping and Navigation

A brief analysis of the navigational requirements for station-keeping has been produced. By augmenting both the GPS and GLONASS satellite navigation systems and overlaying these with a proposed system (for example EGNOS) it seems likely that sub-metre accuracy will be possible for the determination of the altitude of a spacecraft. It has also been asserted that there will be a future relaxation of the Selective Availability of these systems. The DGPS receiver integrated into the standard spacecraft bus is capable of making measurements to these levels of accuracy.

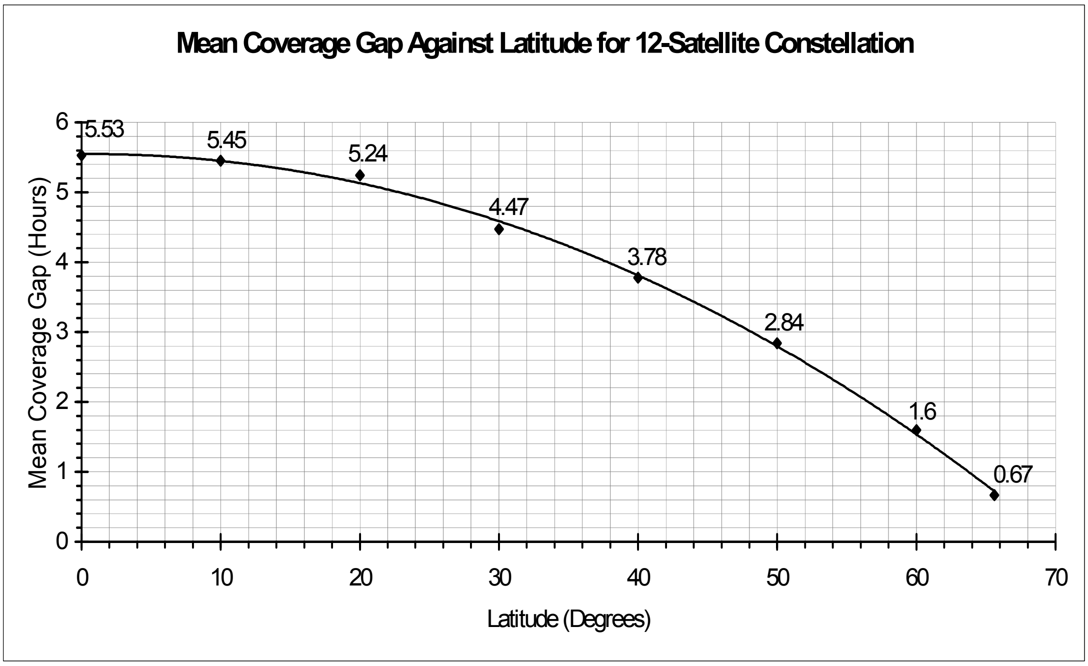


Figure 6 Mean coverage gap plotted as a function of latitude for the twelve-satellite constellation

Appendix B.3.6.Compensating for Launch Vehicle Injection Inaccuracies

The launch vehicle for the mission is the Soyuz (SL-4), capable of performing triple spacecraft injection. The typical injection accuracy for this vehicle is ±19s on the period and ±4.3 arcminutes on the inclination. The typical worse case injection scenario was examined, with a view to formulating an orbit attainment strategy, should inaccurate injection occur. Each spacecraft will perform a two-stage manoeuvre; the first stage will consist of a ΔV to raise or lower the orbit such that the altitude of 564.88km can be attained, the second stage will be a combined burn to effect any plane change and to circularise the orbit. The typical magnitude of ΔV for such a manoeuvre was evaluated to be 27.15ms^{-1}, corresponding to a consumption of 5kg of hydrazine propellant.

Through further development, the use of the FREGAT upper stage to the Soyuz launch vehicle was hypothesised. This would provide orbit trajectory correction prior to injection of the spacecraft, thus taking the onus off the spacecraft themselves in having to perform the manoeuvre.

Appendix B.3.7.Plane Phasing Manoeuvres

The procedure for phasing the spacecraft 120° apart within each plane, after orbit injection, was investigated. The orbits for two of the three spacecraft will be made slightly eccentric by performing a small ΔV; the orbits will then be re-circularised once the desired phase separation has been achieved. Based on a spacecraft commissioning time of 1.5 – 2 months, approximate **total** ΔVs required are in the region of 3ms^{-1}. These correspond to hydrazine consumptions of approximately 0.5kg.

Appendix B.3.8.Eclipse Conditions

Maximum, minimum and mean durations of eclipse were evaluated using the Satellite Toolkit software. Mean eclipse duration was found to be 30^{m}52^{s}.

Appendix B.4.SAR antenna design

(Frederic Domps, Mourad Ben Asker)

Appendix B.4.1.Introduction

Following a decline in imaging radar research in the 1970s and 1980s, the 1990s have witnessed a resurgence of activity as researchers apply active and passive microwave capabilities to Earth observation. Visible sensors, as powerful as they become, are inherently limited by cloud cover, and other weather conditions or persistent phenomena which may be expected to accompany natural disaster. ORWELL's all-weather, day-night remote sensing capability and visibility anywhere on Earth is expected to result in numerous scientifically valuable and commercially lucrative applications.

Existing SAR systems have been severely constrained by their very large volume, mass, and power requirements and obviously their cost. However, the ORWELL baseline design incorporates new technologies in instrument design that could result in significant size, mass, power, and cost savings compared to existing international SAR systems.

Appendix B.4.2.Why Flying a Synthetic Aperture Radar (SAR)?

At any instant of time, about half of the Earth's surface will be covered by clouds. Visible sensors will be limited by the presence of significant amounts of cloud cover. It has been proved that the Landsat satellite, whose revisit time is 16 days, will obtain a cloud-free scene of a certain location in the UK only once per year. Thus, because of their day-night, all weather capability, microwave systems represent the best approach to collecting data for a given region at a specific time. Unlike those from optical systems, signals returned by radars are sensitive to the physical structure and moisture content of the surface being sensed and may offer new data for research and applications.

Moreover, recent web survey demonstrates the world-wide interest in commercial radar data (September 5, 1997; http://biobio.vexcel.com.com/radar/lightsar2.html). This survey released by Vexcel Corporation of Blouder, Colorado had for purposes to help to determine the optimal sensor for LightSAR (NASA small SAR satellite). This international web survey shows that:

87% are interested in learning more about radar products.
73% have become increasingly interested in radar data over the past five years.
86% consider that remote sensing products are important to their work.
56% consider that remote sensing data needs are not being satisfactorily met by existing data sources.
76% believe radar data is very different work with than optical data
76% perceive that radar data is becoming more popular.
78% (30% regularly; 48% occasionally) use or buy data of regions where cloud cover is prevalent.

There is no doubt that SAR systems remain less familiar and are less frequently used than optical sensing systems, but they are demonstrating their worth in some applications.

Appendix B.4.3.General considerations for the design of a SAR

The amount of energy scattered in different directions is dependent on the magnitude of the surface roughness relative to the wavelength used. For spaceborne radars a particular interest is made for the scattered energy back towards the sensor. The scattering properties of the ground as a radar target are usually expressed in terms of average differential scattering cross section σ^0. This term is a function both of the ground properties (surface penetration, roughness, and electrical properties) and of the radar parameters (wavelength, look angles).

The frequency of the incident wave plays a major role in the interaction with the surface. It is a key factor in the penetration depth, scattering from a rough surface and scattering from discrete scatterers. For most materials the penetration depth varies linearly with the wavelength λ. An L-band signal will penetrate about ten times deeper than a K_u-band signal thus providing access to a significant volumetric layer near the surface. However some surface parameters such as moisture strongly affect the penetration depth. The polarisation is also a key factor and should be chosen in accordance with the main application.

The design of a radar system is quite complex, so design compromises must be made and such systems should be tailored to specific needs. Limitations are placed on the radar systems by available space, available power, stability of the bus, available data storage, and other factors coming from the transmitters/receivers whose performance can be limited with the frequency band used. Thus limitations on performance of the system are specified by the environments in which it operates, and design compromises are based on the environments. The most severe restriction on radar systems is limited space available for antennas.

Appendix B.4.3.1.Symbols and constants

A: antenna area (m^2)
W: antenna width (m)
L: antenna length (m)
θ: pointing angle from nadir (deg)
θ_b: beam width (deg)
h: satellite height (m)
S: swath width (m)
σ°: backscatter coefficient
R: Slant range (m)
λ: radar wavelength (m)
T: ground temperature (K)
Pt: required power (W)
Xr: range resolution (m)
Xa: azimuth resolution (m)
η and α: some loss coefficients associated to the design of the antenna, usually taken as $\eta / \alpha = 4$.
P_{rf}=Pulse Repetition Frequency (Hz)
τ: pulse length (s)

Appendix B.4.3.2.Radar equations

The radar antenna illuminates a surface strip to one side of the nadir track. As the spacecraft moves in its orbit a continuous strip of swath width S is mapped along the flight time: $S = \frac{h\theta_b}{\cos^2\theta} = \frac{\lambda h}{W\cos^2\theta}$.

The two resolutions are:

$$X_r = \frac{c\tau}{2\sin\theta} \qquad X_a \geq \frac{L}{2}.$$

One of the factors which determines the quality of the imagery acquired with a radar sensor is the signal-to-noise ratio (SNR). A simple way of characterising an imaging radar sensor is to determine the surface backscatter cross section, which gives a signal-to-noise ratio equal to one. This is called the noise equivalent:

$$\sigma_n{}^o = \frac{4\pi\eta Vr^3\lambda ckT\sin\theta}{\alpha P_{rf}P_t\delta r^2 A^2\cos\theta}$$

This equivalent noise is measured in dB.There are a number of approaches, which would improve the SNR (i.e., decrease σ^0). One such approach is to use of a dispersed pulse with power compression. The average transmitted power of a given radar may be raised by increasing the pulse length within the given transmitter constraints. However, the increased pulse length (reduced receiver bandwidth) has the undesirable effect of reducing the range resolution capability. Knowing this, it is desirable to raise the transmitted power by increasing the pulse lengths and simultaneously keeping a constant bandwidth.

The SAR ambiguity relationship originates from two constraints on the SAR design:
- the Nyquist criterion in the along-track direction:

$$P_{rf} \geq \frac{V}{L/2}$$

- no overlaps of echoes in the across-track direction:

$$P_{rf} \leq \frac{cW}{4r\lambda \tan\theta}$$

Combining these two inequalities the SAR ambiguity relationship is given by: $A = LW \geq \dfrac{8Vr\lambda \tan\theta}{c}$

This relation defines a minimum area of the antenna to achieve its requirements relative to the resolution.

Appendix B.4.4.ORWELL design

SAR remote sensing is one sophisticated observational technique receiving rapidly increased attention. A major disadvantage to rapid exploitation of SARs has been the cost associated with launching the massive and complex instrumentation. Past and current SAR missions have been larger and expensive. Radarsat cost around $500 million; ERS-1 about $750 million and Envisat about $1.2 billion. Existing SAR systems have been also severely constrained by their large volume, mass, and power requirements. Because it is apparent that SAR will assume a position of great importance in Earth sciences, that success in implementing a more affordable SAR could have profound implications for understanding our planet.

Appendix B.4.4.1.Design process

Because only small swath widths are achievable (~60km) by keeping the angle of elevation of the SAR antenna constant that electronic beam steering seems necessary. So, a ScanSAR spacecraft has been preferred to meet the requirements imposed by the scope of the GDP. This alternative ensures the opportunity to fly an Interferometric SAR (InSAR) with a global swath width of 450km at an altitude of 550km. A program, which runs on Matlab, has been performed to handle with all the parameters, formula and constraints.

The power required is computed according to the radar equation, and the power carried by all the signal is compared to the power available, that is to say 100W (200W times the efficiency of the electronics, which is assumed to be around 50%). Hence, several conditions have been implemented to fulfil all the requirements. These requirements are relative to the power, to the global swath width, and to the angle of elevation. The matrix ‘Results’ is set according to these constraints and also by avoiding the parameters corresponding to an area greater than the maximum area wanted by the user. To finish with, the program looks for the row of ‘Results’ that gives the minimum antenna area because small area means ease of deployment in space and low-cost.

Appendix B.4.4.2.Frequency choice

As a result of the data obtained and the science requirements for different application, it appears that:
- L-band provide large field of applications but require large antenna (Area>30m2, Resolution>7m)
- X-band ensures small antenna with good resolution but poorer application area (Area~10m2, Resolution~5.5m)
- C-band achieves to a compromise between applications and antenna sizes.

Appendix B.4.4.3.Results

Two values of the L/W ratio have been chosen from the data obtained. With a ratio of 8, good resolution is obtained with average area, but with a value of 10, the instrument would be less power consuming and provides larger footprint (worse resolution: 5.28m compare to 4.75m). Adding the capability for the full-polarised system increases the power requirement, cost, and complexity of both radar and data systems. But whatever its constraints a quad polarisation radar is preferred to a single polarisation system.

Ratio	Global SW	Power required	Area	Width	Length	Resolution	PRF	Steering angle
L/W	M	W	m^2	m	M	M	Hz	deg
8	450935,87	92,24	11,27	1,19	9,5	4,75	1596,19	35 to 55
10	450179,99	83,42	11,13	1,06	10,55	5,28	1436,54	35 to 54,8

Table 3 SAR antenna characteristics

Appendix B.4.4.4.Description of the electronics and the antenna

An inflatable structure suits to the GDP requirements. Even it is a new technology it has proved its reliability. The size of antenna would be 1.5m by 10m, these dimensions take in account the inflatable cylinder that round the planar skin and assume that the thickness is less than 2cm. The structure is inflated with nitrogen gas and its rigidization with UV radiation avoids its bending. Indeed, a non-flatted antenna would raise errors in the data processing, as it will add Doppler shift terms. The projected mass density of such structure is 1.5kg/m2 (including Rigidizable Frame, Inflation System, Storage Container, and SAR), so assuming that a weigh of 22.5kg will be achieved.

The panel is a phased array based on the techniques of inflatable structures, and composed of patch radiators emitting the wave. The antenna substract, contains the feed lines for the patch radiators, and has been described after the inflatable structure proposed by the JPL and IDC Dover. This panel contains the Transmitters Receivers modules, as well as the phase shifters used for electronic beam steering. The required pointing angle for this panel is 0.1 deg El and 0.1 deg Az. These values have been chosen after the NASA study and in order to have the same mapping error in both directions of 1.7 km.

The electronics size should be roughly similar to LigthSAR: 10cm x 16cm x 30cm for a mass of 25kg. We did not investigate much the electronics linked to the SAR signal processing. We just assume it is composed of usual radar devices : Timing and control to generate the pulse, the RF electronics composed of the exciter, transmitters and receivers modules, and then digital and electronics data routing delivering a signal corresponding to 8 bits per pixel. The weight and size of the electronics has been derived after a NASA study, it should weight 25 kg an fill a space of 10*16*30cm.

Appendix B.4.4.5.Comparison with current and planned SAR systems

Compared to current large spacecraft ORWELL baseline emphasises the fact that a constellation of small SAR spacecraft reveals better. Its power and mass requirements are far from ERS1-2 or EnviSat ones, which operate in the L-band and but provide worse resolution.
Hence, ORWELL fulfils perfectly the GDP baselines, that is to say flying a small, light-weigh and low-cost remote sensing system. NASA has also followed this strategy. The LightSAR would operate at L-band frequencies (large domain of applications) but will be high power consuming (1.24kW opposed to less than 200W) and will need larger antenna (2.90m x 10.8m compare to 1.06 x 10.55m for ORWELL). Note that ORWELL achieves to good resolutions with less power due to a higher compression ratio (10 times higher than for LigthSAR: pulse width 15.3µs, PRF of 1600Hz with a power requirement of 1.24kW).

The parameters we compared are the ones on which we had major constrains and defining the capabilities of the SAR. The elements of comparison are two studies made by Alenia Spazio and NASA in 1998. The resolution of 5.28 m we achieved is lower than the 3 m of NASA and 5 m of

Alenia. Our antenna area is around 11 m^2, which is much bigger than the Alenia Antenna, much smaller than the NASA. The American instrument is to run in the L band and the Italian in the X band. We plan to use really less power than all the other instruments with 200 W when the NASA and Alenia plan to use 600 w.

Referring to the previous equation, the more power, the better resolution, so it is no surprising that we get a smaller resolution than the Italian and the American. The SAR ambiguity equation also shows that the bigger the wavelength, the larger the antenna you need to achieve, and this is why the American instrument requires such a big antenna, and the Italian instrument can run with a smaller antenna, and that we are close to the Italian. The choice of the wavelength is based on the applications chosen for the mission: L band for the American who plan to do both commercial and scientific applications, X band for the Italian who only want to use their instrument for scientific applications, and we made the choice of the C band because it offers a good compromise between the number of applications and the antenna size.

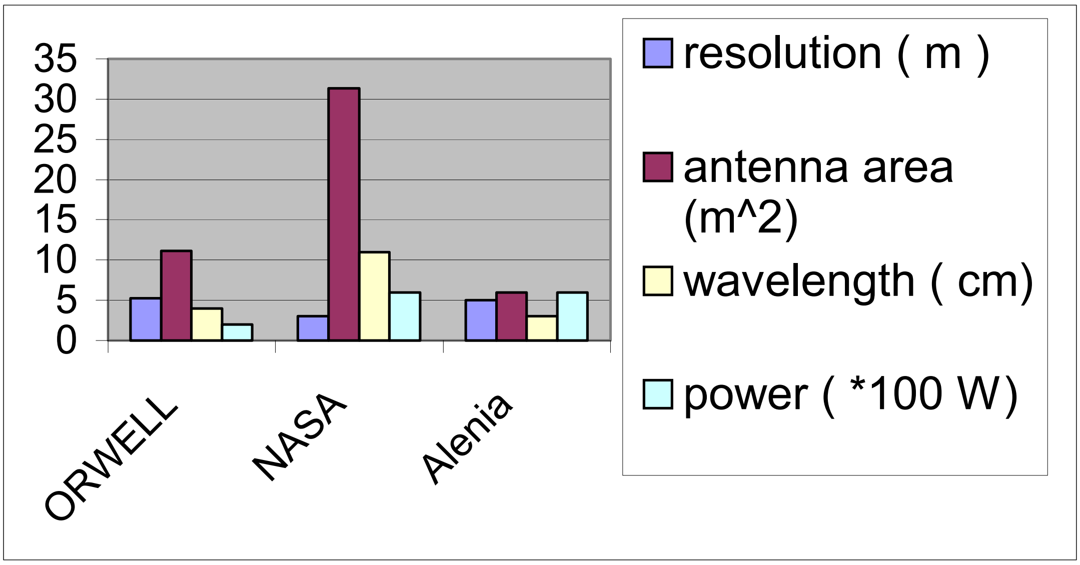


Table 4 Comparison or the ORWELL SAR with the NASA and Alenia designs

Appendix B.4.5.Conclusion

The ORWELL SAR adapted antenna has been designed to cover a large field of applications with large footprint (<63km). It operates at a single-frequency (C-band, 4cm wavelength) and provides 5.28m resolution pictures. Power consumption, antenna area and instrument total mass are far from the limits imposed by the objective constraints, thus with 11m^2 area and less than 50kg mass, and less than 170W, ORWELL can compete with current and planned SAR systems. Gathering all the data, at least a 20% duty cycle seems achievable.

The ORWELL concept lies on the current SAR specifications, so further investigations should follow. Hence, better electronics performances can certainly be achievable (efficiency>50%), an accurate glance at picture distortion should also be done (As the spacecraft is travelling, the beam must be kept normal to the track. If not, it should introduce Doppler error. To overcome that steering the beam along the yaw axis can be feasible.). As a final point, added capabilities for ORWELL instrument should also be performed: multifrequency (both C- and X-band) and Spotlight mode (azimuth scanning would achieve to resolution of less than 3m) would provide new fields of applications and would ensures ORWELL to be competitive in the Earth Observation market.

Appendix B.5.Data links

Appendix B.5.1.Communication Subsystem

(Samuel Bitton)

Two main communication systems have been investigated which had to meet the specific characteristics of the two preferred data dissemination architectures, i.e. the Store and Forward and the use of TDRSS.

Appendix B.5.1.1.Store and Forward

For this system, each satellite of the ORWELL constellation communicates with the control centre via 4 receiving ground stations. The uplink is done in S-band at a low data rate of less than 2Kbps while the downlink is done in X-band at 200Mbps. The ground elevation angle is 5° so that the access time achieved allows a maximum of 22% instrument duty cycle. The link budget for the downlink is provided in Table 5

.

Parameter	Symbol	Telemetry and Data			Units
		Parabolic	Helix	Horn	
Carrier Frequency	*Fc*	7.5			GHz
WaveLength	*Lamda*	0.04000			m
Transmitter Power	*P*	80			Watts
Transmitter Power	*P*	19			dBW
Transmitter-to-Antenna line loss (Worst Case)	*Ll*	-3			dB
Elevation angle on Earth	*Ep*	5			deg
Altitude of Satellite	*H*	565			km
Propagation and Polarization Loss (Sea Level)	*La*	-0.874			dB
System Noise Temperature	*Ts*	552			K
Data Rate	*Rd*	200			Mbps
Bit Error Rate	*BER*	1.E-05			-
Required Eb/No for BPSK and with BER given +Reed Solomom	*Req Eb/No*	4.7			dB
Ground Station Antenna Diameter	*Dr*	12			m
Ground Station Antenna Gain	*Gr*	56			dB
Ground Station Antenna Beamwidth	*TetaR*	0.21			deg
Propagation Path Length	*S*	2243			km
Max antenna pointing offset angle	*Lteta*	66			deg
Satellite Antenna Beamwidth	*Theta*	132			deg
Transmit Antenna Length	*Lta*	-	0.004	0.004	m
Transmit Antenna Diameter	*Dt*	0.021	0.015	0.022	m
Peak Transmit Antenna Gain toward G/S	*Gpt*	1.86	2.18	1.80	dB
Transmit Antenna Pointing Loss (Worst Case)	*Lpt*	-3.00			dB
Transmit Antenna Gain (Gpt+Lpt)	*Gt*	-1.14	-0.82	-1.20	dB
Space Loss	*Ls*	-177			dB
Effective Isotropic Radiated Power	*EIRP*	14.9	15	14.8	dBW
Received energy-per-bit to noise-density ratio	*Eb/No*	10.7	11.0	10.7	dB
Carrier-toNoise density ratio	*C/No*	93.7	94.0	93.7	dB
Implementation Loss (Estimate)	*Il*	-2			dB
Margin	*Marg*	4.0	4.3	4.0	dB

Table 5 Downlink budget

The difficulty in the design of the downlink communication system is that a low gain antenna is required to achieve the wide half power beamwidth. Different antenna alternatives have been investigated: the standard parabolic reflector, the horn, the helix, the multi-feed horn reflector, the steerable antenna, the quadra fila helix and the shaped reflector. The recommended antenna for the system is the quadra fila helix. It is a fixed antenna without any deployable structure, which allows a wide beamwidth of up to 120°. The wide beamwidth is achieved by modifying the radiated field pattern that would be produced by a standard helix antenna. Such quadra fila helix antenna has the disadvantage of being power limited and would need further detailed investigation to check if it meets the link power requirements.

This communication system is simple and low cost (several hundred thousand dollars) compare to other options such as the steerable antennas (up to $1M). However extra development for the downlink antenna is required which could increase the cost. The major disadvantage of this system is that the instrument duty cycle is restricted to about 22%.

Appendix B.5.1.2.TDRSS Case

The design is based on using the facilities provided by the Advanced TDRSS which is a system based on the current TDRSS but should provide extended facility to meet small customer satellite limited communication resources such as DC power, space and weight. The choice is to use the available 600Mbps, Ka-band in single access, for the return link and the 10Kbps data rate, multiple access in S-band for the forward link.

As far as the onboard hardware is concerned, a Ka-Band phased array antenna (currently under development at GSFC) is recommended for the return link. The communication system based around this antenna has an estimated cost of about $1M, including the transmitter and additional components.

Communication with the ground segment via TDRSS allows higher instrument duty cycle, up to 100%, provides more flexibility in the access with the Ground Segment, and could potentially lower the cost of access if unconstrained communication was often achieved. However, it has been demonstrated that the communication system required is more complex as it necessitates steerable antenna and tracking system of TDRSS spacecrafts. Also, the complete communication system provided by the satellite bus needs to be modified or replaced which would require extra cost.

As the instrument duty cycle is not always limited by the power and can go up to 100%, then, transmitting data via TDRSS is the only way to cope with such duty cycle. This would be the ideal solution for the demonstrator mission as the number of spacecrafts is only 3. Even if the 3 satellites were using TDRSS very often it would be unlikely to overload it. However, for the full constellation where 12 satellites will be in orbit, communicating data via TDRSS might become a problem when long accesses due to high instrument duty cycle are required and therefore, TDRSS might saturate.

Appendix B.5.1.3.Communication Subsystem Conclusion

The satellite bus provides an X-band 320Mbps download communication system. The antenna has a 20° half power beamwidth with a directional gain antenna. The actual footprint achievable is unknown. Therefore for the Store and Forward system, it is recommended to use the S-band uplink communication system provided but modify or replace the downlink system by the quadra fila helix antenna in order to allow the long access with the ground station. If TDRSS was to be the final data dissemination solution for the mission, a complete new communication system would have to be fitted increasing significantly the cost. In either system, a pair of two omni-directional antennas is recommended for a communication back up system.

Finally, a more detailed cost analysis is still required in order to decide if whether the extra cost due to the complex communication system required for TDRSS could be cut down by the (potential) improvement of the revenues gained by the high instrument duty cycle achieved.

Appendix B.5.2.Onboard Data Handling Subsystem

(Samuel Bitton)

As in every spacecraft, the OBDH subsystem is there to orchestrate all the subsystems by distributing commands, monitoring the health of the spacecraft, carrying out the interface with the communication system by formatting data prior downlink and retrieving uplink data, and finally (and most importantly) dealing with the payload data.

For the ORWELL mission, the OBDH critical task is to collect all the data provided by the SAR instrument. Therefore, the main objective was to provide a system that could maximise the instrument duty cycle, which in turn would minimise the repeat time of the constellation. The main work was carried out on the design of the Instrument Data Management system as high data throughput had to be achieved.

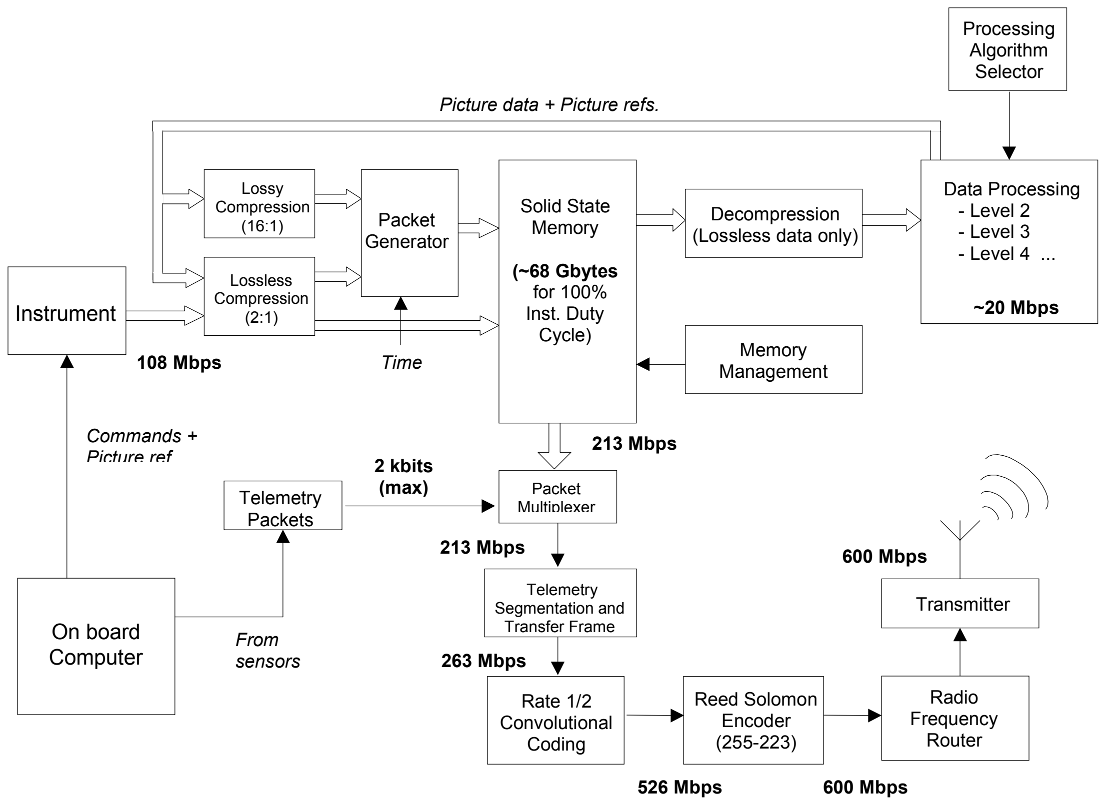


Figure 7 Instrument Data management Unit block diagram

This unit is able to cope with the high data rate of 108Mbps delivered by the SAR. Compression is used in order to maximise the onboard storage and therefore maximise the instrument duty cycle. The minimum compression algorithm has a compression ratio of 2:1 and is lossless. The Rice Algorithm is the recommended for the compression algorithm for its rapidity and its advantage of not requiring extra RAM. The maximum compression should allow a 16:1 ratio with a distortion (or loss of data) of less the 1%.

The onboard processing should use Artificial Intelligent algorithms in order to allow complex processing algorithm to be carried out onboard without human interaction, and therefore reduce the ground system complexity and also accelerates the delivery to customer.

The onboard storage facility provided by the satellite bus is used. A maximum of 60GBytes is required for 100% instrument duty cycle. This memory based on the RAID technology is heavy (5kg), expensive ($300,000) and power demanding (55Watts when writing or reading). Future technology could be considered such as the ultra high density memory developed by Pf. Williams from Keele University (3.4Tbytes on the size of a credit card). The Telemetry and Telecommand subsystems is based on the CCSDS Standard Protocols which provides many advantages, such as a fast data delivery to customer.

The Onboard Data Handling system available from the SA-200HP satellite bus is organised around the R6000 processor. The MIL Std-1553 data Bus allows easy interfacing with additional units that have to be connected to the OBDH system, such as the Instrument Data Management system.

Appendix B.5.3.Telecommand and data dissemination

(Robert Hand)

Appendix B.5.3.1.Primary Objective

The primary drivers of this design project are commercial viability and cost. The dissemination architecture that is required will have to handle the move towards a fast return of data to the customer, if the system is to be commercially viable. With the current explosion in Internet related business and the speed with which information can be transmitted around the globe, it is apparent that the future broadband systems (such as Teledesic and Skybridge), will provide an ideal, low cost system for the transmission of telemetry and telecommand. The concept of low cost should also extend to the customer; that is to say, the cost of data produced by the EOS should be as low as possible and compliment the existing systems. This can be achieved by rapid repeat times and an initial low cost for the mission; but if the data relay system is, in comparison expensive to set up and run, then the overall effect will be little different from SPOT or Landsat.

Appendix B.5.3.2.Ground Systems

There are two options available for data relay at present, the Tracking and Data Relay Satellite System (TDRSS), or Ground Stations. The choice is further confused by the option of ground stations that are already in place or building specialised systems, as well as the various options available through TDRSS. The main interface between instrument and user is traditionally the Control Centre and Archiving facility. Requests for data are processed and then transferred via the Telecommand network to the satellite for execution. For high-speed transfer of data to the user, a fully transparent interface is required, so that instructions are processed quickly, the data is retrieved and returned to the customer promptly. The Archiving and Control Centre will be a large section of the initial cost of the mission and the significant portion of the cost in the long term. Well designed archiving, processing and distribution centres, capable of handling many terabytes of information daily, will become the norm for Remote Sensing.

- Orbital Constraints: Orbital altitude plays a very large part in the number of ground station and their placement. A high altitude orbit, such as Geosynchronous, needs only one station because of its stationary nature and its wide field of view. On the other hand, LEO spacecraft often require dozens of stations to service them. At the altitude chosen for the mission (564Km), it is possible to reduce the number of stations by careful placement of new, or a choice of existing stations. The other main restrictions on placement are Geographical, Political, Data Rate and Cost.

- Geographical Constraints: The land mass distribution on Earth is not even; i.e. the high latitudes in the Northern Hemisphere contain far more land than the high latitudes in the Southern Hemisphere. Longitudinally the landmass distribution is also uneven.

A brief synopsis of the typical stations is given here.

- S-Band: This is a well-used technology capable of moderate data rates (~50 – 60 Mbps). There is a multitude of S-Band stations around the world, which provide comprehensive coverage for tracking, telemetry and command, as well as limited Data handling capabilities.

- X-Band: The technology behind X-Band is maturing now and is capable of high data rates (~100-300Mbps). Unfortunately, far fewer Ground Stations, which comply with CCSDS, have the X-Band capability. This is likely to change as more telecom systems switch to higher frequencies to cope with the use of data transfer and video conferencing.

- K-Band: New Technology capable of very High Data rates (~600MBps). There are very few stations available with this capability, although, with the continued increase in required capacity in the lower frequencies, the likelihood is that far more K-Band stations will be built in the future.

Appendix B.5.3.3.Transmission of Telecommand

The primary task of Telecommand is to achieve a very high probability of successful transmission of commands to the satellite. Currently S-band is the main telecommand transmission frequency. This is likely to remain so since, generally, the type of data sent via telecommand is simple on/off commands to onboard systems, the initiation of telemetry download and confirmation of telemetry transmission. Even the need to upload complete files or software can be achieved at this frequency.

Appendix B.5.3.4.Tracking and Data Relay Satellite System (TDRSS)

The system is split into the ground segment and the space segment. The space segment consists of a constellation of six satellites in GEO, three of which are in use and three as backup. The three operational satellites are at 41, 174 and 275 degrees west of longitude. The ground segment consists of two terminals, the data for telecommand is transferred to these terminals from the customer and then onwards to the customer satellite. TDRSS covers 85% of most orbits used by its customers. The system provides forward transfer of data at 300kbps up to 50Mbps, for command of spacecraft and onboard instruments, and return service for telemetry up to 300Mbps with the prospect of 600Mbps in the future.

Appendix B.5.3.5.Broad Band Systems

The principle behind the broadband systems is to provide an "Internet in the Sky". Broadband systems allow multiple users to simultaneously exchange data at high rates. There are several systems currently under development, all backed by big names within the satellite and communications industry. These will provide links capable of rivalling the data throughput of fibre optic landlines and complimenting the current Internet service.

Appendix B.5.3.6.Control Centres

Control of spacecraft is, at present, a human occupation with the assistance of computer data processing. This is expensive since it requires full time coverage around the clock, consisting of four or five teams of between one and 60 people depending upon the complexity of the system.

Appendix B.5.3.7.Data Centre Functions

The primary requirements for a Data Centre are to receive and archive sensor data derived from the mission so that it is available for future retrieval. It must keep a check on the data quality so that it meets the standards set out for each level of data product. The cataloguing of data from the mission and generation of summaries of data, inputs and directories of catalogues, is vital so that users can browse the archive and select the relevant data, fully informed of its potential content. Current formats for dissemination are, in physical electronic media, 8mm digital videotape, Digital Audio Tape (DAT), CD-ROM, in virtual electronic media, web based requests or email, and in physical hardcopy format, photo quality prints.

Appendix B.5.3.8.Spacecraft Operations

This area covers the flight section immediately after launch. Here the spacecraft is checked-out in orbit so that any minor bugs are ironed out before the craft goes operational. Goddard Space Flight Centre (GSFC) have, under development, a system which can be used in both the test and integration section of the ground segment and when the craft is on orbit and operational. GSFC have developed the system in such a way as to allow its transfer to the flight segment operation with the Test & Integration engineers. This provides a large measure of commonality between sections and reduces cost through common hard and software. Spacecraft operations then commission the craft and proceed to operate it within the specified limits for correct function.

Appendix B.5.3.9.Archiving

The world is currently 10^{14} bits for complete coverage using a moderate resolution SAR. This amount of data would be provided over the cycle of the satellite ground track. This is currently undertaken at ground stations and requires that large amounts of data are stored in several locations around the world, and smaller local data is stored at other locations. Any storage system for a constellation of EOS Satellites will have to have a high storage capacity and be scaleable with the increased data that are likely to require archiving. If a unique set of Ground stations is constructed and other satellites can be added then further expansion and cross support will be necessary.

Appendix B.5.3.10.Product Level Description

Five levels of product, as applied to data returned from the remote sensing spacecraft.

Level	Description
0.0	Raw telemetry data on high density digital tape (HDDT)
0.5	Annotated, time-ordered, demultiplexed raw data on computer compatible media
1.0	Earth-located, time-corrected, single instrument data in engineering or physical units
1.5	Single instrument data merged with detailed orbital information, fully corrected for instrument and atmospheric effects
2.0	Single instrument data from a single pass and converted to geophysical units.

Table 6 Hierarchy of Product Processing Levels[1]

Appendix B.5.3.11.Results

Number of pictures / 14 days / satellite	38904	Max picture that can be downloaded	165
Number of pictures / 14 days	116711	Average Duty Cycle	24%
Price per picture ($)	300	Max Storage required (Gbytes)	12.8
Years Incomes Demo mission (M US$)	910.3	TDRSS Access Duty Cycle	45%
Cost of Access (M US$)	31.9	Instrument Data Rate (Mbps)	108
Cost of Staff (M US$)	2.2	Download Data Rate (Mbps)	600
Total Income (M US$)	876.3	Instrument Duty Cycle %	50%

Table 7 Breakdown of Access and Costs for TDRSS with and Estimate of Income

The table shows the total expected income from the demonstration mission per year, using TDRSS. Ray Turner provided the estimate of personnel cost as a rough order of magnitude from RAL. The cost for access is based on the total access time to TDRSS multiplied by the estimated access duty cycle and given unconstrained single access. The initial estimate of picture cost was $300 so that the demonstration mission significantly undercut current systems.

Appendix B.5.3.12.Conclusion

The keys to competing with them are timeliness of data and a resolution and repeat time that easily out performs them. The fast repeat time has been achieved by the orbit group and to compliment that we provide an architecture whereby data can be transferred to the ground quickly, efficiently and with little or no error.

The analysis of the STK simulations for access to the ground stations, and to TDRSS, have shown that both systems allow fast return of data and are economically viable. However, for the constellation, in terms of pricing and expansion, TDRSS comes out on top, if the duty cycle of the instrument can be kept in the region of 25% or above. With unconstrained access the cost is half of that expected for the ground stations and the system will have a higher overall though put of data.

The CCSDS protocols chosen are the Path protocol for transmission of data from spacecraft to ground, this has the advantage of allowing the transmission of several virtual channels over a single link by multiplexing the output from several memory buffers. A bitstream service is also available for raw data for SAR interferometry.

The overall grade of service required for the data to be high quality is grade one, that is the data is delivered complete, sequence preserved with a high probability of no induced errors from the transmission media. If errors are detected a further Automatic Request for Retransmission will be sent either to the ground station or to the satellite.

TDRSS takes care of the tracking and the Telecommand with instructions from the Control Centre. The type of access required to TDRSS will be single access Ka-band that will allow 600Mbps-download rate. The Control Centre will act as a central archiving facility suing the proposed high-density storage medium from Keele University. This will reduce the area required for storage and allow fast retrieval of data to a secondary storage medium for delivery to the customer. Effectively the Constellation will become a Wide Area Network in orbit using the TDRS system network to

[1] D. R. Sloggett. Satellite Data: Processing, Archiving and Dissemination, Vol.2: functions, operational principles and design. Ellis Horwood Library of Space Science and Space Technology. J. Wiley & Sons.

communicate. Each Satellite will be a Local Area Network with its own Databus and distributed systems for Tracking, Telemetry and Control connected, via a single phased array antenna and TDRSS, to a single ground station at White Sands.

All requests for data will be processed by a central facility combining the operation of Payload operations and Mission control centre. The Spacecraft control centre will be distributed between the central facility and White Sands. This will require a dedicated link in the form of either a leased satellite/phone link or the future broadband system. The latter will require high level security to prevent unauthorised access to mission vital areas. The protocols used for transmission of data are the Path and Bit-Stream protocols as defined by CCSDS in their recommendation CCSDS 701.0-B-2 Advanced Orbiting Systems, Networks and Data Links architectural specification. The quality of service will be grade 1 as defined in the above recommendation. These recommendations then allow the use of the Still-Image communication service, recommendation CCSDS 704.0-B-1, to specify the still-image mode 2, (to the ISO standard CD10918-1 (JPEG lossless coding)) for image quality after onboard processing up to level 2 if appropriate.

Appendix B.6.Strategic Analysis for Future Commercialisation of the ORWELL Earth-Observation Satellite Mission

(Novan Satria Budi)

This section summarises the strategic analysis for future commercialisation of ORWELL Earth-Observation Satellite Mission, including mission constraints, keys to success, strategy under uncertainty, user requirements, global marketing, applications, commercial viability, competitiveness and pricing decisions prediction for this mission.

Appendix B.6.1.Strategy in Action for Earth-Observation Satellite

The fundamental purpose of ORWELL's strategic analysis is to ensure the survival, growth, and profitability of the mission over the long run in a changing and potentially competitive environment.

Appendix B.6.1.1.Mission Constraints

A strategy is needed to enter the future global market of Earth-Observation, because there are some constraints, namely:

- Very High Risk Business
- Limited Lifetime
- Hard Global Competition
- Uncertainty (Technology, Political and Economic Conditions)

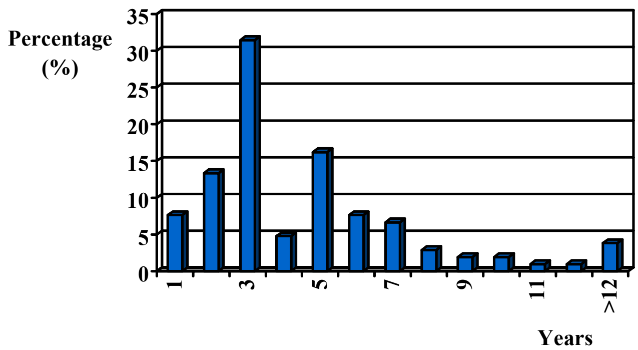


Figure 8 Lifetime of Earth-Observation Missions (Launch1984-2003)

As can be seen from figure 1 that average lifetime of Earth-Observation satellites is between 3 and 6 years. It has to be taken into account before designing satellite mission.

Appendix B.6.1.2.Keys to Success

In order to improve competitiveness of Earth-Observation satellite mission, some actions below must be considered, namely:

- Future Market Analysis
- Integrated Management
- Cost Reduction and Efficiency
- Time-to-Market Reduction
- Technical Excellence for Value-added Products
- Government Support and International Co-operation

Appendix B.6.1.3.Strategy Under Uncertainty

Entering an emerging market such as Earth-Observation can be categorised into the third level of uncertainty, which is in a range of future posible outcomes.

There are several analytic tools that can be used in this situation, namely:

- Latent-demand research
- Technology forecasting
- Scenario planning

Appendix B.6.2.User Requirements and Preconditions

In future global market, the only thing that users concern will be whether the products fulfil their requirements or not. Consequently, for ORWELL Earth-Observation satellite mission to be commercially attractive to potential users, it must meet certain requirements, namely:

- **Availability**
  - Timeliness
  - Reliable Service
  - Continuous Availability
  - Unlimited Availability
- **Quality of Service**
  - Competitive Prices
  - Customer-oriented Service
  - Flexible Purchase Conditions
  - Exclusivity

Appendix B.6.3.Global Market and Applications

Appendix B.6.3.1.Global Market of Earth-Observation

One of the keys to success of Earth-Observation mission is future market analysis. It must be realised that this market is very huge, in order to be optimistic to go ahead. A world-wide market in terms of the revenues by region in 1996 can be seen in Table 8.

Region	Revenue
North America	21%
Latin America	4%
Africa	6%
Europe	30%
Middle East	7%
Asia Pacific	32%

Table 8 1996 Revenues by Region

Appendix B.6.3.2.Potential Applications of Earth-Observation

The Mapping and Geographic Information System are the biggest markets of Earth-Observation, and the other applications still growing up. Nevertheless, commercial geological and mining applications do not require continuous observation and real-time data processing. Hence, the petroleum industry is not considered to be a long-term customer of Earth-Observation data.

It is important to know some potential applications of Earth-Observation in the future global market. Figure 9 shows a good analysis of some potential applications in terms of their spatial resolution and repetitivity (global-cover repeat-time).

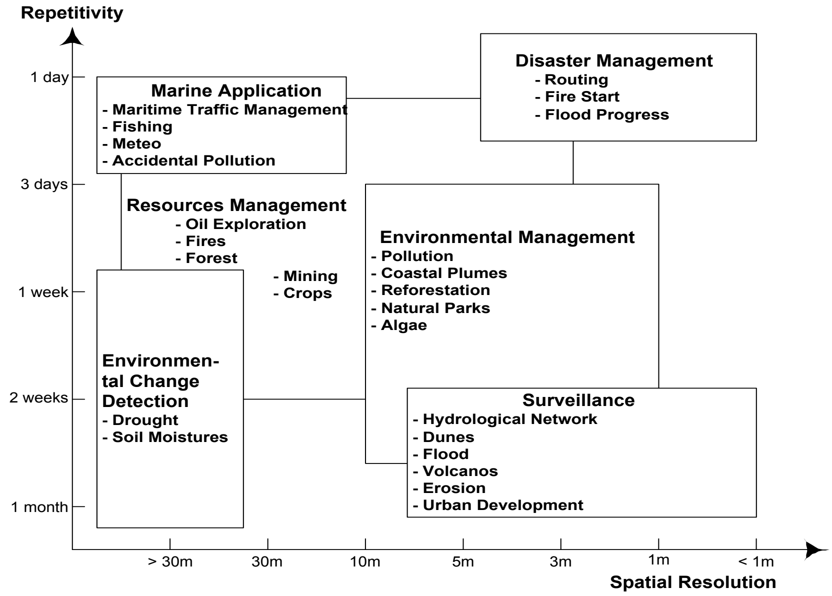


Figure 9 Current potential EO applications

Appendix B.6.3.3.Applications of ORWELL

It has been determined that this mission will have 5 meters spatial resolution and 2 days global-cover repeat-time. With this information, it can be seen in Figure 9 that ORWELL has most of the potential applications, except for Disaster Management.

Appendix B.6.4.5. Commercial Viability of ORWELL Earth-Observation Satellite Mission

Appendix B.6.4.1.Projected Cumulative Costs & Revenues

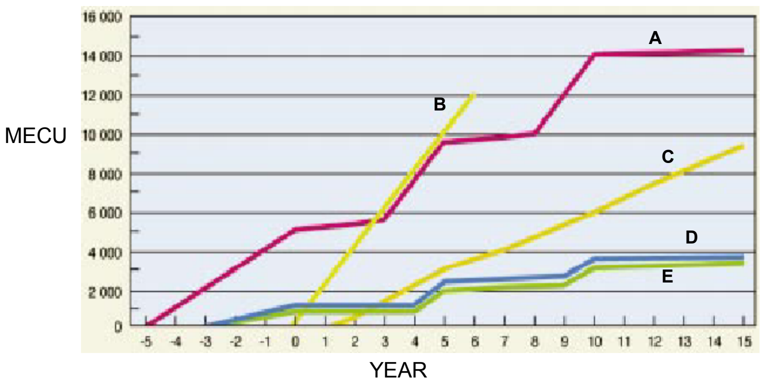


Figure 10 Projected Cumulative Costs & Revenues for Earth-Observation Mission (A = Current Cost; B = Revenue Best Case; C = Revenue Worst Case, D = Future Cost Availability 95,; E = Future Cost Availability 50%; Year 0 is the time for launching the first mission)

From Figure 10 some ideas can be obtained, namely:

- Current revenue to cost ratio is not good enough
- Current manufacturing time is longer than future manufacturing time
- Future technologies and methods will decrease cost
- Future revenue to cost ratio will be good, even if we take the revenue worst case
- Availability of 95% (with in-orbit spares) should be taken as the baseline, because the cost difference with availability 50% is not significant
- Revenue best case means getting revenue just after launch and earning money as much as possible.

The ORWELL Earth-Observation mission was designed to start operation around year 2003. With the ideas obtained from Figure 10, it can be said that as a future Earth-Observation mission, ORWELL will be commercially viable in terms of the revenue to cost ratio.

Appendix B.6.4.2.Projected Market

There will be a large area of business in Earth-Observation, and the biggest area is Space Derived Services where ORWELL Earth-Observation mission is right there. This fact gives us more confidence to deal with some economic constraints in this high-risk business.

Appendix B.6.4.3.Number of Earth-Observation Satellites In-Service

The number of Earth-Observation satellites in-service tends to increase. It will give some interesting consequences for the future Earth-Observation missions, namely:

- Very dynamic future market
- Hard future competition
- Lower future development costs
- Lower future product prices
- Value-added future products.

Appendix B.6.5.Competitiveness of ORWELL

The growth of competition on a global basis makes competitiveness become a critical factor affecting success. For this reason, the competitiveness of ORWELL Earth-Observation satellite mission must be investigated correctly.

Appendix B.6.5.1.Comparison with Competitors

ORWELL Earth-Observation satellite mission was designed to be launched after year 2003. Hence, only the future missions were chosen in table 2 to be compared to ORWELL mission. As can be seen in Table 9 that ORWELL mission is not the best in terms of spatial resolution and swath width, but it would be the best one in terms of repetitivity and its lifetime is very good.

Appendix B.6.5.2.Innovations for Competitiveness

ORWELL needs a coupling process called innovation, which brings together technology and user needs. It is a creative process, which needs a good understanding and experience of the technology and market concerned. Without a major effort in innovation in industrial practices, organisation, and research, this mission will lose its competitiveness in comparison to the continued increase in efficiency and initiatives of other space industries.

Appendix B.6.6.Pricing Decisions Prediction for ORWELL

The objective of pricing decisions is to value how a product is worth to the user, as well as to cover all costs and to provide a margin for profit in the process. The prediction of ORWELL's products can be seen in table 3 that follow some features of current products. Obviously, it is not a fixed result to be used in real operation. Market situations must be investigated continuously to ensure the right prices of ORWELL's products. Prediction in table 3 was made by focusing on Radar Satellites (RADARSAT and ERS) and competitive future market assumptions. Prices in Table 10 were made by a roughly assumption that they would be around 75% of current prices.

Mission	Lifetime	Spatial Resolution	Swath Width	Repetitivity
ORWELL	6 – 10 years	5 m	60 km	2 days
SPOT-5	5 years (2002)	5 m	120 km	26 days
LandSat-7	6 years (1999)	15 – 30 m	185 km	16 days
ENVISAT-1	5 years (2000)	30 m	100 km	35 days
RadarSat-2	7 years (2001)	25 m	100 km	24 days
QuickBird-1	5 years (1999)	1 – 4 m	20 km	148 days
EOS AM-1	7 years (1998)	15 m	60 km	49 days
Resource21	5 years (1999)	10 m	200 km	4 days
IRS-2A	5 years (2000)	5 m	148 km	22 days
Almaz-1B	7 years (1998)	2.5 – 10 m	170 km	-
METOP-1	5 years (2002)	-	-	5 days
GDE	6 years (1999)	1 m	-	-
MSG-1	7 years (2000)	-	-	-

Table 9 Comparison with competitors

Image Specification		Image Price
Digital Products		
Path Image		$1,400
Path Image Coarse		$1,400
Signal Data (RAW)		$1,200
Single Look Complex		$1,400
Single Look Detected		$1,400
Multi Look Detected		$1,600
Precision Image Geo-referenced (PRI)		$1,400
Full Resolution Image		$700
Fast Delivery Image (FDC)		$700
Annotated Signal Data (RAW)		$1,200
Single Look Complex Full Scene (SLCF)		$1,400
Print Products		
20 x 20 cm^2	1:500,000	$25
Enlargements	1:125,000	$150
	1:100,000	$200
Interferometry Data		
Signal Data	Per scene	$1,200
	Per set of 3	$2,500
	Extra scene	$700
Single Look Complex	Per scene	$1,400
	Per set of 3	$5,000
	Extra scene	$800
	Per set of 3 quads	$1,400
	Quad	$300
Film Products		
(20 x 20 cm^2) Film orders must be accompanied by a digital order	1:500,000	$25
Satellite Programming Services		
Emergency	29 – 48 hours	$750
Urgent	2 – 6 days	$600
Express	7 – 13 days	$300
Priority	14 days +, date-sensitive	$150
Basic	14 days +, best-effort	No Charge
Meteorological	None, any one, or all three scenes acquired over each site	$550
Data Processing Services		
Near-Real Time	Within hours	$1,150
Rush	Within 48 hours	$750
Regular	Within 14 days	No Charge
Delivery		
Courier	National: overnight International: 2 – 7 days	Paid by client
Electronic	On a case-by-case basis	First 3 files: No Charge Each additional file: $200

Table 10 Pricing Decisions Prediction of ORWELL's products