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A Modular, Reconfigurable Approach For A Commercial Small Spacecraft Programme

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ABSTRACT

This thesis presents the work performed in producing a system-level design for a modular, multipurpose small satellite platform. A multipurpose platform may be applied to a wide range of missions, and, to be commercially viable, the envelope of missions for which it is suitable should be as large as possible. The research therefore addresses the particular requirements that are specific to different mission types, and produces characteristic requirement sets for each. General design requirements are also derived, such as those for enabling modularity and allowing compatibility with different launch vehicles.

The commercial requirements arising from the different market and customer sectors are also examined. Industry analysis allows identification of general market trends, and predictions are made regarding the likely size and characteristics of the market in which the proposed platform would compete. It is anticipated there could be a worldwide demand for more than twenty small satellites each year, for which a flexible small spacecraft platform could potentially compete.

After derivation of the necessary requirements has been performed, a system-level design of the spacecraft platform is undertaken. The resulting design is based on a multi-module, reconfigurable concept, which can be adapted to fit the different launch envelopes of Pegasus-XL, Taurus, ASAP-5 and larger launchers, and also to accommodate a wide range of payloads. The subsystems are offered in different capability variants, which may be interchanged in response to different mission requirements. The platform equipment and structure forms a "standard parts list", from which the appropriate configuration can be built up. Schedule reductions are obtained due to the modular design allowing more of the integration and testing of the platform to be performed in parallel.

The proposed programme for development of the platform uses up-front investment to conduct much of the detailed design of the platform in advance of any actual project. This allows the design effort to be shared across many subsequent projects, and the design phase of each new project to be minimised. The key benefits of the proposed platform and programme are adaptability, ability to rapidly reconfigure to mission requirements, suitability for future upgrading, and reduction of the project schedule.

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This thesis is dedicated to Pat Bradley, 1919-2003.
With love.

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ACRONYMS AND ABBREVIATIONS

A/D	Analogue/Digital
ADCS	Attitude Determination and Control Subsystem
AFRL	Air Force Research Laboratory
AIT	Assembly, Integration and Test
Al	Aluminium
AMSAT	Radio Amateur Satellite Corporation
AO	Announcement of Opportunity
AOCS	Attitude and Orbit Control Subsystem
AOS	Acquisition Of Signal
ASAP	Ariane Structure for Auxiliary Payloads
AST	Associate Administrator for Commercial Space Transportation
BER	Bit Error Rate
BMDO	Ballistic Missile Defense Organisation
BNSC	British National Space Centre
BOL	Beginning Of Life
CATSAT	Cooperative Astrophysics and Technology Satellite
CCD	Charge-Coupled Device
CEC	Cincinnati Electronics Corporation
COMSTAC	Commercial Space Transportation Advisory Committee
COTS	Commercial Off-The-Shelf
CTE	Coefficient of Thermal Expansion
DERA	Defence Evaluation and Research Agency
DET	Direct Energy Transfer
DHS	Data Handling Subsystem
DoD	Department of Defense
DOD	Depth Of Discharge
DPAF	Dual Payload Attach Fitting
EADS	European Aeronautic Defence and Space Company
EDAC	Error Detection And Correction
EGSE	Electrical Ground Support Equipment
EIRP	Equivalent Isotropic Radiated Power
ELINT	Electronic Intelligence
EM	Engineering/electrical model
EMC	Electromagnetic Compatibility
Eng. D	Engineering Doctorate
EO-1	Earth Observing 1
EOL	End Of Life
ERS	Earth Resources Satellite
ESA	European Space Agency
ESTEC	European Space Research and Technology Centre
FMECA	Failure Modes, Effects and Criticality Analysis
FOV	Field of View
GaAs	Gallium Arsenide
GALEX	Galaxy Evolution Explorer
GEO	Geostationary Earth Orbit
GFO	GEOSAT Follow-On
GFRP	Glass Fibre Reinforced Plastic
GNSS	Global Navigation Satellite System
GPS	Global Positioning System
GSFC	Goddard Space Flight Center
GTO	Geosynchronous Transfer Orbit
HEO	Highly Eccentric Orbit
HK	Housekeeping (data)

I/O	Input/Output
IAE	Inflatable Antenna Experiment
ICD	Interface Control Document
IDD	Interface Development Document
IR	Infra Red
IRU	Inertial Reference Unit
ISO	International Organisation for Standardisation
ITSAT	Inflatable Torus Solar Array Technology
ITT	Invitation To Tender
JPL	Jet Propulsion Laboratory
KIPS	Kilo Instructions Per Second
LAN	Local Area Network
LANL	Los Alamos National Laboratory
LEO	Low Earth Orbit
Li	Lithium
LOS	Loss Of Signal
MCM	Multi-Chip Module
MEMS	MicroElectroMechanical Systems
MEO	Medium Earth Orbit
MESA	Modular Experimental Platform for Science and Applications
MFS	Multifunctional Structure
MGSE	Mechanical Ground Support Equipment
MIDEX	Medium Class Explorer
MIL SPEC	Military Specified Component
MIPS	Mega Instructions Per Second
MLI	Multi-Layer Insulation
MMH	MonoMethyl Hydrazine
MMS	Multimission Modular Spacecraft
MOD	Ministry of Defence
MSTI	Miniature Sensor technology Integration
NASA	National Aeronautics and Space Administration
NASDA	National Space Development Agency
NGST	Next Generation Space Telescope
Ni-Cd	Nickel-Cadmium
Ni-H	Nickel-Hydrogen
NMP	New Millennium Program
NO	Nitrous Oxide
OBDH	Onboard Data Handling
OSR	Optical Solar Reflector
PA	Product Assurance
PCB	Printed Circuit Board
PCS	Power Control Subsystem
PEST	Political, Economic, Social, Technological
PFM	Protoflight Model
P/L	Payload
PPT	Pulsed Plasma Thruster
PPT	Peak Power Tracking
PROBA	Project for Onboard Autonomy
QA	Quality Assurance
R&D	Research and Development
REMODEL	Reconfigurable Modular Expendable Lightsat
RF	Radio Frequency
RFP	Request For Proposals
RHESSI	Ramaty High Energy Solar Spectroscopic Imager
ROM	Rough Order of Magnitude
RSDO	Rapid Spacecraft Development Office
RTG	Radioisotope Thermal Generator
SAA	South Atlantic Anomaly

SAR	Synthetic Aperture Radar
S/C	Spacecraft
SEL	Single Event Latch-up
SEU	Single Event Upset
Si	Silicon
SIL	Space Innovations Limited
SM	Structural Model
SMA	Shape Memory Alloy
SMEX	Small Explorer
SNOE	Student Nitrous Oxide Explorer
SOHO	Solar and Heliospheric Observatory
SSC	Swedish Space Corporation
SSTL	Surrey Satellite Technology Limited
STEDI	Student Explorer Demonstration Initiative
STEP	Space Test Experiment Platform
STP	Space Test Program
STRV	Space Technology Research Vehicle
SWOT	Strengths, Weaknesses, Opportunities, Threats
TBD	To be determined
TC	Telecommand
TDRSS	Tracking and Data Relay Satellite System
TEAMSAT	Technology, science, and education Experiments Added to MAQSAT H
TERRIERS	Tomographic Experiment using Radiative Recombinative Ionosperic EUV and Radio Sources
TM	Telemetry
UHF	Ultra High Frequency
UNEX	University Explorer
USAF	United States Air Force
UV	Ultra Violet
WIRE	Wide-Field Infrared Explorer

1 INTRODUCTION

1.1 RESEARCH OBJECTIVES

The key objectives of the research described in this thesis may be stated as follows:

- **To analyse the technical and commercial requirements for a modular, multipurpose small satellite platform**

- **To produce a technically and commercially valid system-level design for such a platform**

The intention is not to solely address the technical aspects of small satellite design, but to cover the commercial and programmatic issues that will make the design viable in a real industrial marketplace.

1.2 ORIGIN AND EVOLUTION OF THE RESEARCH PROJECT

The original idea for research conducted in the small satellite field arose out of the author's involvement in the TEAMSAT project at ESTEC, in 1997. TEAMSAT was a small spacecraft launched as an opportunity payload on the second flight of Ariane 5, and carried a number of experiments and prototypes of new technologies. This project sparked an interest in the benefits of small spacecraft, for demonstration and other applications, and how they could best be produced to take advantage of opportunity launches such as the Ariane 5 test flight.

This idea was further developed, growing out of the realisation that:

1. To a small satellite project, cheaper launches (obtained via sharing/ late availability of spare capacity/ test flights etc) are an extremely valuable commodity
2. As such, competition for such opportunities may be fierce, particularly with the increasing incidence of commercial small satellite ventures
3. To compete successfully for a budget launch slot, a very flexible small spacecraft design, that can be adapted and produced rapidly, is at a considerable advantage

From these points, it also became apparent that a very flexible platform would also have commercial benefits, as it would not only be applicable to a range of different launch options, it would be applicable to a range of different payloads and missions as well. It was therefore decided that the commercial aspects of the design should also be investigated.

These plans fitted in extremely well with a research programme that had been newly introduced at Cranfield University: the Engineering Doctorate. This programme combines technical engineering research, with an additional emphasis on business and management awareness. As well as the research work and technical components, an additional year is spent taking core courses from the Cranfield School of Management

Executive MBA programme. These courses include strategy, operations and project management, marketing, economics and finance. The programme also encourages industry involvement; to ensure that projects undertaken have a good industrial relevance, all EngD students have an external sponsoring company or organisation.

The research work described in this thesis was originally undertaken under sponsorship from Space Innovations Limited (SIL), a UK company specialising in the design and manufacture of subsystems for small satellites and expanding into production of complete "turnkey" small satellites. As a result, the research was initially tailored towards the scope of SIL as a manufacturer and business entity. It was hoped that work could be done in collaboration with the company, which would result in the eventual development of commercial hardware. The initial direction was to look at modification of the existing small satellite designs to incorporate the use of new techniques and technologies, giving a more flexible and multipurpose platform.

Unfortunately, however, SIL ran into financial difficulties in 2000, and was eventually liquidated in the second half of that year. After this event, the research direction had to be somewhat altered. The design work had to become a more general, system-level design study, and the business scope was made more generic, in terms of the types of manufacturer to which the proposed platform may be applicable.

1.3 RATIONALE

The previous section explained the basis for the research concept. Here, the rationale for the use of small satellites in general is introduced, together with the reasons for developing a multipurpose spacecraft platform. This then leads into a rationale for the modular small satellite research work presented in this thesis.

1.3.1 WHY SMALL SATELLITES?

There are a number of properties of small spacecraft projects that make them advantageous. The key benefits afforded by such projects are:

- Reduced spacecraft costs, due to:
    - Lower complexity and fewer components
    - Use of commercial technology
- Reduced launch costs
- Reduced schedule times, giving:
    - Reduced logistical and programmatic costs
    - Opportunity to use/test new technologies in space more rapidly
    - Ability to make use of opportunities such as shared/cheap launches

Reduced spacecraft costs

Small spacecraft are generally less complex than larger ones; this is mainly simply due to the lower parts count. A small satellite with only one or two payloads, and few mechanisms, will give an inherently simpler (and therefore cheaper) design task. Small missions also often use less redundancy, reducing the overall equipment cost. This may not, however, cause great increases in risk; a simpler system using less (but carefully selected) redundancy may nevertheless give a better reliability value than a highly-complex, highly-redundant system. [1] This is illustrated by the fact that overall system reliability, R, is given by:

$$R = (R_0)^n$$

Where R_0 is the reliability (i.e. success probability) of each of the n components that make up the system. A system with a lower parts count can thus achieve the same reliability for a lower reliability value per component.

The above argument also applies to the use of non-space-qualified components (such as commercial processor chips). Great equipment savings can be made via the use of COTS (commercial off-the-shelf) components. This is often also enabled by the fact that small satellite missions have much shorter lifetimes, typically 1-3 years, giving less time for the accumulation of damage due to radiation effects.

Reduced launch costs

The enormous cost of placing a spacecraft in orbit has been a major obstacle throughout the space age. Current average cost-per-kilogram to LEO (Low Earth Orbit) is of order US$20,000 (the actual $/kg varies quite widely – this is illustrated for selected launchers in Figure 1-1). This in itself gives a good reason for using smaller satellites - purely in terms of launch cost, a smaller satellite will give the opportunity for a major saving.

However, true launch costs are not as simple as specific cost multiplied by the mass of the spacecraft. Quoted specific costs assume a payload that utilises the full capacity of the launch vehicle; the whole launch must obviously be paid for whether a launcher is fully-laden or not. And although many vehicles are available in different-capability variants, there may be some launches where the primary payload does not require quite the full capacity of the launch vehicle. This has led to the increasing occurrence of shared launches, with small satellites "piggybacking" with large primaries – using the small amounts of spare capacity that may occur. Small satellites are well-suited to this type of opportunity; as well as their small size, their shorter schedules also mean that they can be made to fit an opportunity occurring quite late in the scheduling of a larger project.

As can be seen in Figure 1-1, the larger launchers give a lower specific cost. Sharing a larger launch gives a mutual cost benefit for both the primary passenger and for a smaller piggyback payload; the prime will not be paying for "dead space" but will utilise the lower specific costs of a larger launcher, and the small satellite avoids the much higher specific costs of a smaller, dedicated launch vehicle.

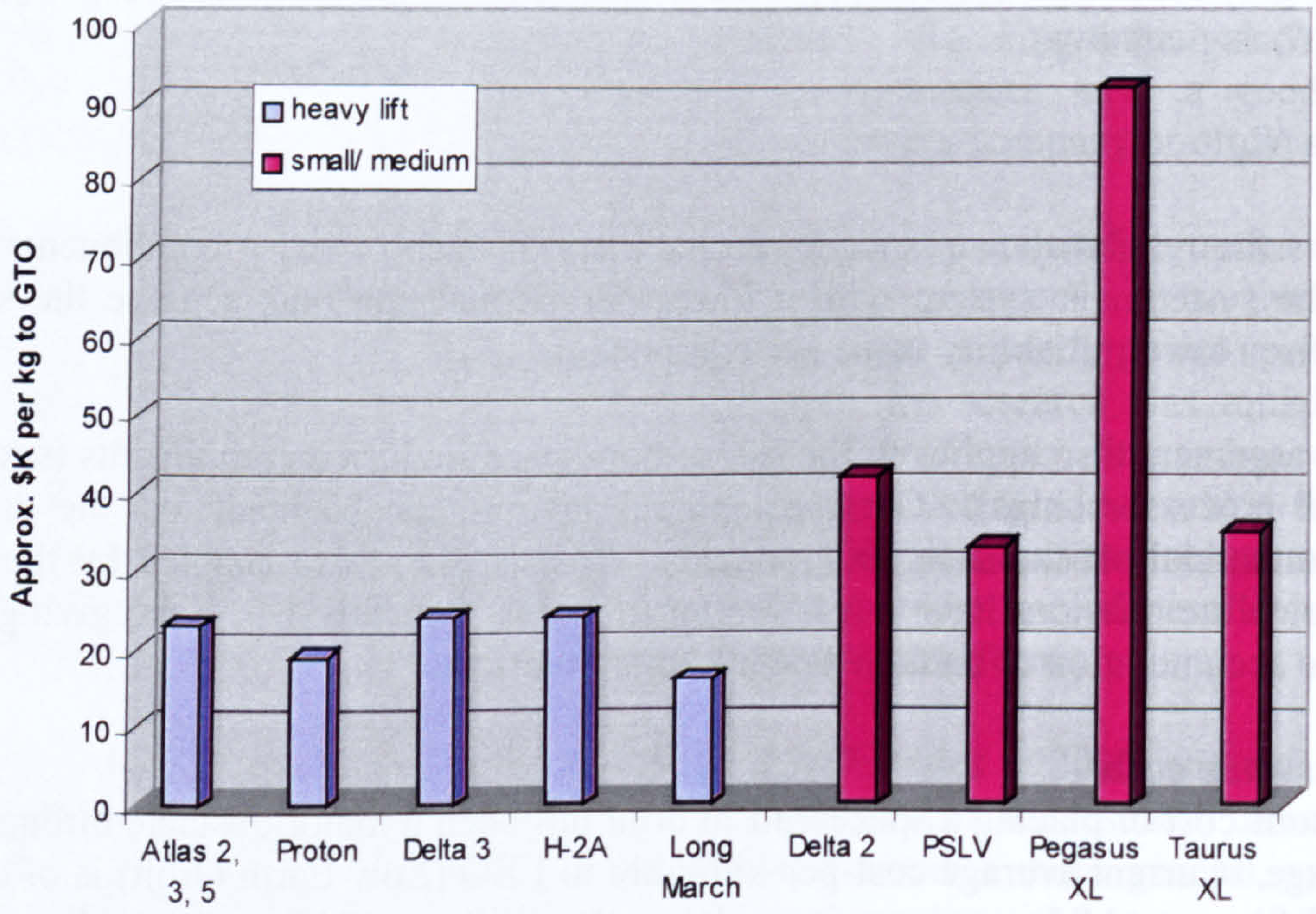

Figure 1-1 Launch costs per kg for different launch vehicles [2]

There are also other factors that may give a cut-price launch to a small spacecraft. New launch vehicles are often given several pre-commercial test flights, on which spacecraft may be flown for low or no cost (but obviously higher risk – insurance may not be possible for such missions). Small satellites are again more suited to this type of opportunity, as it may arise at fairly short notice, and other items, such as test equipment, may need to be accommodated in the launcher fairing. The lower-cost/higher-risk traits of such a launch also tend to be more in keeping with the small satellite philosophy.

Launch options for small spacecraft are discussed in greater depth in Chapter 4.

Reduced schedule times

Large space projects are extremely complex, time-consuming, and expensive. The huge amounts of manpower, documentation and review procedures required to manage such a project, with its many different payload suppliers, contractors, subcontractors and service providers, mean that the timescales may be of order 10-15 years from first concept to launch. Figure 1-2 gives some examples of the duration of large space projects. In contrast, small satellite projects often have durations of less than two years from proposal to launch, and have much smaller project teams, often co-located. This greatly reduces the logistical and programmatic costs.

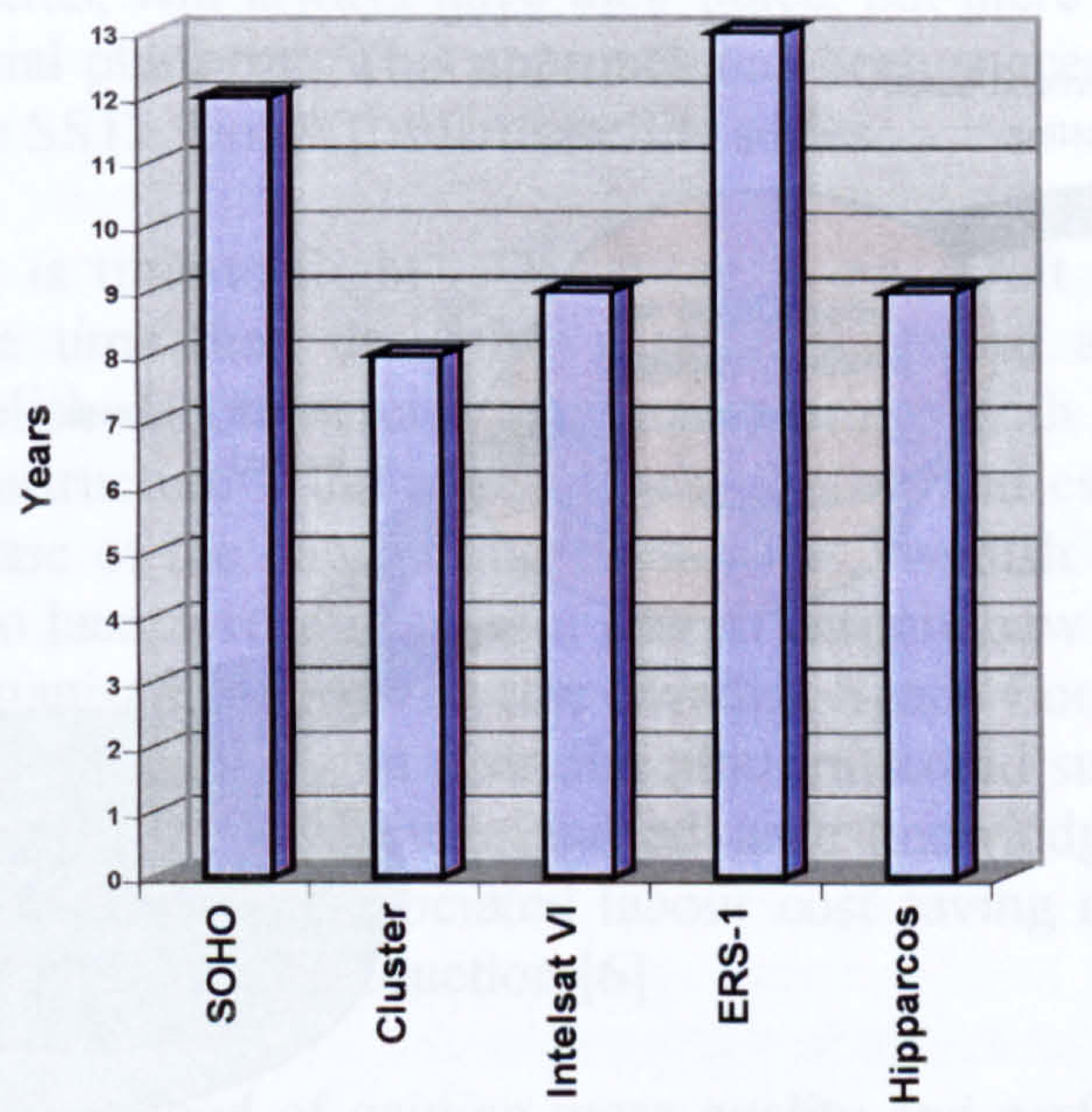

Figure 1-2 Durations from proposal to launch, for some large space projects [3,5]

A key characteristic of the space industry has been the development and use of new technologies, many of which may be transferred to other industries and commercial applications. This transfer of technology has often been cited as one of the justifications for funding space projects. (In fact, with the decline in civil and military spending on space programmes, technology transfer is now increasingly happening "in reverse", with technologies developed in other industries, such as commercial automotive and information technology, being utilised in space.) However, wherever the origin of the technologies, a drawback of large satellite projects is that their lengthy timescales mean that the newest innovations must often remain unutilised, as the design is fixed long before the eventual launch. The end result is that, at launch, a new satellite may be carrying 5 or 10 year-old technology. When it is considered how much the level of computer technology, for example, has changed over the past 5 years, it is clear that this is not an ideal situation.

This gives a further good reason for using simple, small satellites. A small spacecraft project, where there may be only 2 years between the initial concept and the launch, can get these new innovations into space and into use, almost as soon as they become available. This not only allows the benefits of new technology to be reaped early, in terms of performing good science, Earth imaging, and so on, but it provides a platform for validating the use of that technology in a space environment. There have been several proposals for small missions to be used as demonstration pre-cursors to major space projects. Once successfully demonstrated and tested in space, the risks associated with the new technology may be reduced sufficiently to make them acceptable for use on a large, high-criticality project. Plus, the technology provider is then in a position to charge a premium for a fully space-rated product. The flow of new technologies into space use is illustrated in Figure 1-3.

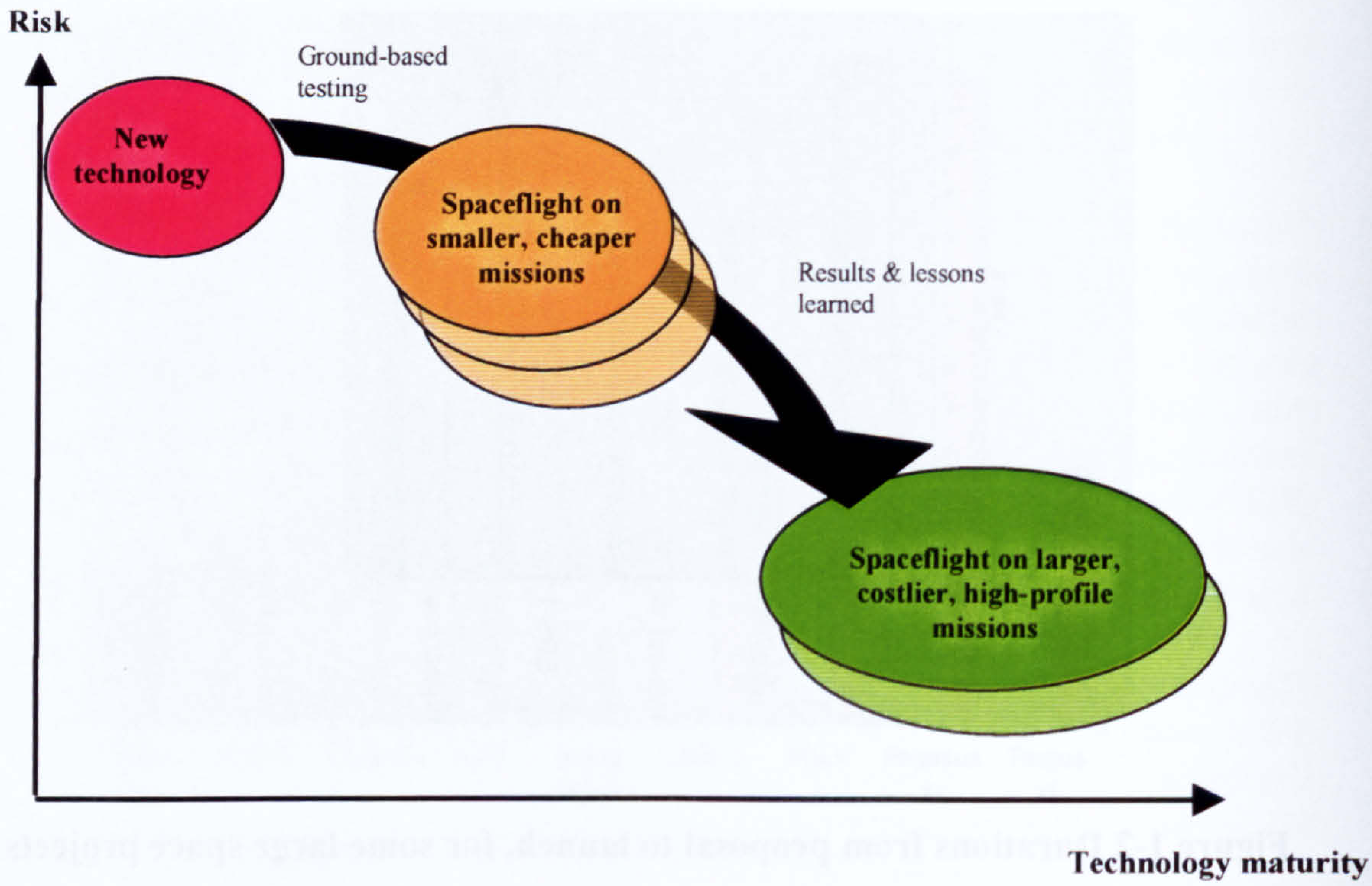

Figure 1-3 Flow of new, unqualified technologies towards use in major space missions – the use of smaller, cheaper missions as demonstrators

Of course, small satellites are not always a suitable solution. While small spacecraft are becoming more capable and sophisticated all the time, there are some applications and instruments which physical laws determine must be of a certain size. These sizes may come from radiation-collecting aperture dimensions, optical focal lengths, detector array geometries or solar array areas for power generation. It would certainly not be possible, at least within the foreseeable future, to pack a Hubble Space Telescope into a 200kg spacecraft. (It may, however, be possible to achieve a similar performance with a constellation of cooperating small spacecraft).

Summary

In summary, while there is undoubtedly still a need for large spacecraft, small satellites can complement the big projects and offer the advantages of:

- Lower spacecraft and launch costs
- Lower logistical and programmatic costs
- Shorter schedule times to get new technologies demonstrated in space more quickly

1.3.2 WHY A COMMERCIAL, MULTIPURPOSE PLATFORM?

Small satellite projects have often been opportunities for organisations such as universities and national research agencies to attain a space presence at a greatly reduced cost. At the very lowest-cost end of the spectrum, small spacecraft that are custom-built in-house, largely by voluntary and student labour and utilising non-space-qualified components, will always have their place, but there are benefits in buying an existing commercial platform. This approach has been successfully demonstrated over many years by the SSTL UoSAT microsatellite series.

One key element is time-to-flight. Designing a spacecraft from scratch will almost always take more time than designing a mission around an existing platform. In addition, an established commercial programme brings with it an existing operational and logistics infrastructure – including supply chains, and established learning curves and knowledge base of the engineering team. The Swedish Freja minisatellite took 5 years to develop to launch readiness, as it was an entirely new design, and a new type of project for the organisation involved (the Swedish Space Corporation). However, it is estimated by the project team that a similar platform could subsequently be built in 24 to 30 months, due to design heritage and advance knowledge regarding procurement and supply chain issues. An associated labour cost saving of 10-15% would also be expected from such a schedule reduction.[6]

There is also the likelihood of gaining more quality and performance per unit cost, as the supplying company, in the course of their own R&D, has already absorbed the bulk of the design phase costs. There is often also lower technical and programmatic risk, due to demonstrated designs/equipment, and greater knowledge about the systems and processes.

These potential benefits to a customer can make a multipurpose platform a marketable commercial item. The positioning of the multipurpose platform in the commercial marketplace, and the different market sectors that exist within the spacecraft industry, are examined in depth in Chapter 3.

1.3.3 WHY A MODULAR/RECONFIGURABLE PLATFORM?

A modular system may be defined as one that is composed of a number of self-contained units, which are easily removed and replaced without requiring significant architectural changes to the rest of the system. The replacing module may have a different performance, but it will still interface with the existing system.

Building up spacecraft systems out of modular "building blocks" has a number of advantages, many of which are particularly applicable to a multipurpose platform.

System upgrading

If all missions using a commercial, multipurpose spacecraft had the same set of requirements, there would be little benefit to designing a platform to be modifiable or upgradeable; a single design that met the requirement set would suffice. However, requirements vary widely (as will be shown in later sections), and what may be a perfect platform for one mission may be entirely inadequate for another. For this reason, an effective multipurpose spacecraft design will have the option to be upgraded to a higher performance level (at increased cost). The easier the upgrade process can be made, i.e. by limiting the impact and redesign incurred on the rest of the system, the smaller the cost increment. A modular spacecraft, at its most idealised, can merely have the under-performing subsystem module unplugged and replaced with a higher-specification one, with the rest of the spacecraft being essentially unaffected.

When producing a multi-purpose spacecraft platform that has different higher-performance options above a standard baseline, there will often be the problem of "wasted performance". If a mission requires just slightly more capability than a particular option can provide, it must move to the next performance increment. Where the increments are large, there is a lot of capability or performance that is not necessary, but that still must be paid for. This is illustrated in Figure 1-4. If there is too much wasted performance, it may be cheaper to produce a purpose-built platform that exactly matches the required performance.

A range of modules with different capabilities, which can be easily interchanged, give a greater number of possible performance increments. This can minimise wasted performance. They also enable only the particular under-performing subsystem to be changed, so that unnecessary capability enhancements to other areas are avoided. In the ideal case, the modular multi-purpose spacecraft "performance curve" can become much closer to that of a purpose-built platform.

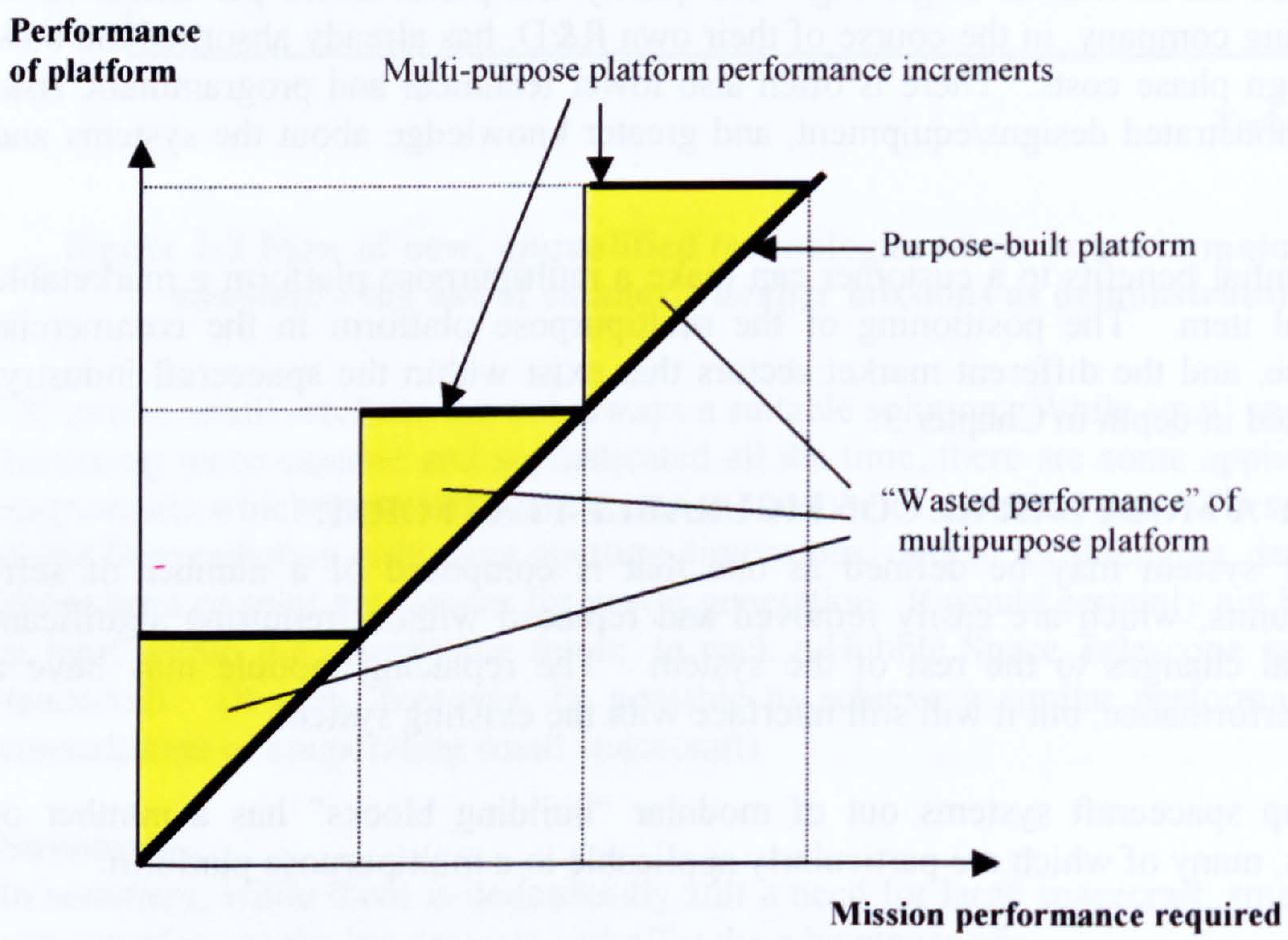

Figure 1-4 Platform performance vs required performance for purpose-built and multipurpose spacecraft

Integration and testing

A spacecraft that is made up of discrete modules can benefit from a greater concurrency in the integration process. Each module may be assembled, and tested at module level, in parallel. Standard interfaces between modules also afford a less complex final integration process, with a more efficient learning curve for the AIT (assembly, integration and test) team, as the method for integrating each module is similar.

Decoupling of the modules, with respect to data and power, reduces the amount of "de-bugging" required when modules are interfaced together[7]. Standard interfaces also mean that test equipment can be much more standardised, and much ground support equipment (GSE) can be re-used for later spacecraft even if modules of different "rating" are being used. The flight qualification process can also be streamlined, by enabling much of the structural testing to be performed at module level.

A full engineering model for each spacecraft produced using the modular platform is not necessary; an appropriate model can be assembled out of a "test suite" containing an engineering model of each module. Test models can be built up of structural and/or electrical models as necessary, and mission-specific flight software and payload test models added. This approach can then enable a protoflight model (PFM) philosophy, with test levels of the PFM minimised. Testing philosophies are discussed in greater detail in Chapter 4.

The reduced integration and test timescales enabled by subsystem modularity have been demonstrated in the past. NASA's Goddard Space Flight Centre compared AIT timelines for spacecraft employing the Multimission Modular Spacecraft platform (discussed in the next chapter), and comparable spacecraft using non-modular designs, and a marked timeline benefit was shown. See Figure 1-5.

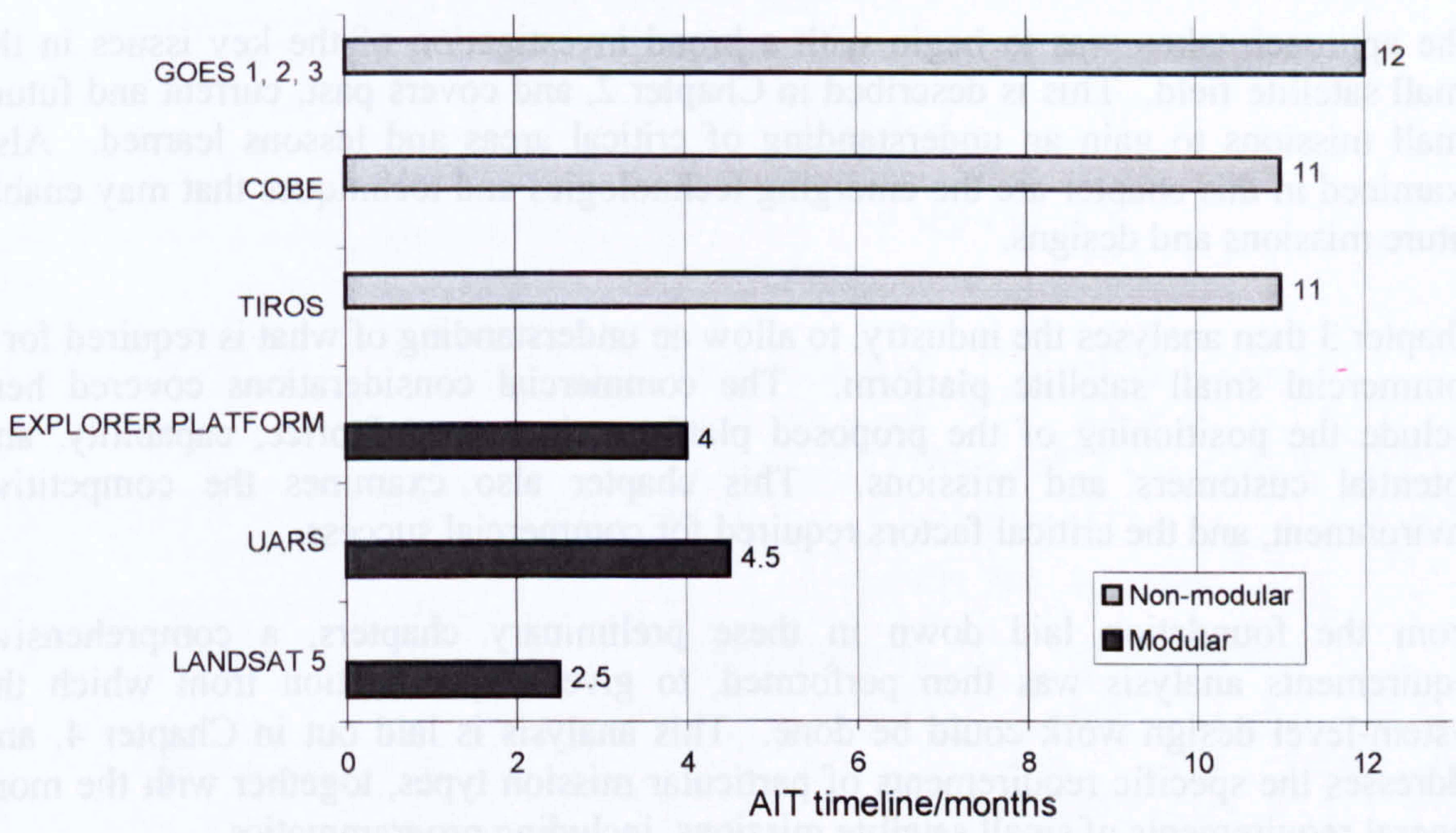

Figure 1-5 AIT timelines for modular vs non-modular GSFC spacecraft [8]

Reducing AIT duration offers valuable cost savings, and helps to meet the goal of reducing time-to-flight. It is also to be expected that lessons learned in the test process of the first spacecraft in the series will further reduce the timeline for successive spacecraft.

Configuration design

With standard modules and standard interfaces between them, the design of the spacecraft configuration is made much simpler, and therefore quicker. Compatibility between subsystems is already "designed in". It is then mainly a case of sizing/selecting the modules according to the requirements of the particular mission.

Simplifying the configuration design process is extremely beneficial from a commercial perspective. When bidding to produce the platform for a particular mission, there may often be very limited time to produce a technical solution proposal. Standard modules, and known configuration options, with clearly understood performance capabilities, can give a competitive edge by ensuring that only missions which are within the scope of the platform are bid for, reducing time and effort being wasted on over-optimistic proposals. This strategy also allows more accurate schedule and cost forecasts to be made.

Summary
In summary, use of modularity can enable a multipurpose spacecraft platform by:

- Providing for easier upgrading of the system
- Streamlining integration and testing
- Simplifying the configuration design for a specific mission

1.4 APPROACH & RESEARCH ROAD MAP

The approach taken was to begin with a broad investigation of the key issues in the small satellite field. This is described in Chapter 2, and covers past, current and future small missions to gain an understanding of critical areas and lessons learned. Also examined in this chapter are the emerging technologies and techniques that may enable future missions and designs.

Chapter 3 then analyses the industry, to allow an understanding of what is required for a commercial small satellite platform. The commercial considerations covered here include the positioning of the proposed platform, in terms of price, capability, and potential customers and missions. This chapter also examines the competitive environment, and the critical factors required for commercial success.

From the foundation laid down in these preliminary chapters, a comprehensive requirements analysis was then performed, to give a specification from which the system-level design work could be done. This analysis is laid out in Chapter 4, and addresses the specific requirements of particular mission types, together with the more general requirements of small satellite missions, including programmatics.

Chapter 5 covers the system level design work. Starting from the requirements derived in the preceding chapter, configuration concepts are defined and traded, leading to a baseline system definition. This chapter also covers the programmatic aspects such as the scheduling of project activities.

To demonstrate the design, in Chapter 6 a case study approach is used, with the baseline platform being applied to several different mission types. The process by which a mission may be matched to an appropriate platform configuration is also proposed.

Finally, Chapter 7 summarises the work done, compares the stated objectives with the results obtained, and draws conclusions as to the contributions and benefits of the research.

All work was performed whilst keeping sight of the links and implications with the real commercial environment.

The research roadmap, shown in Figure 1-6 on the following page, illustrates the logical progression of the thesis, and indicates the questions raised and answered in each chapter.

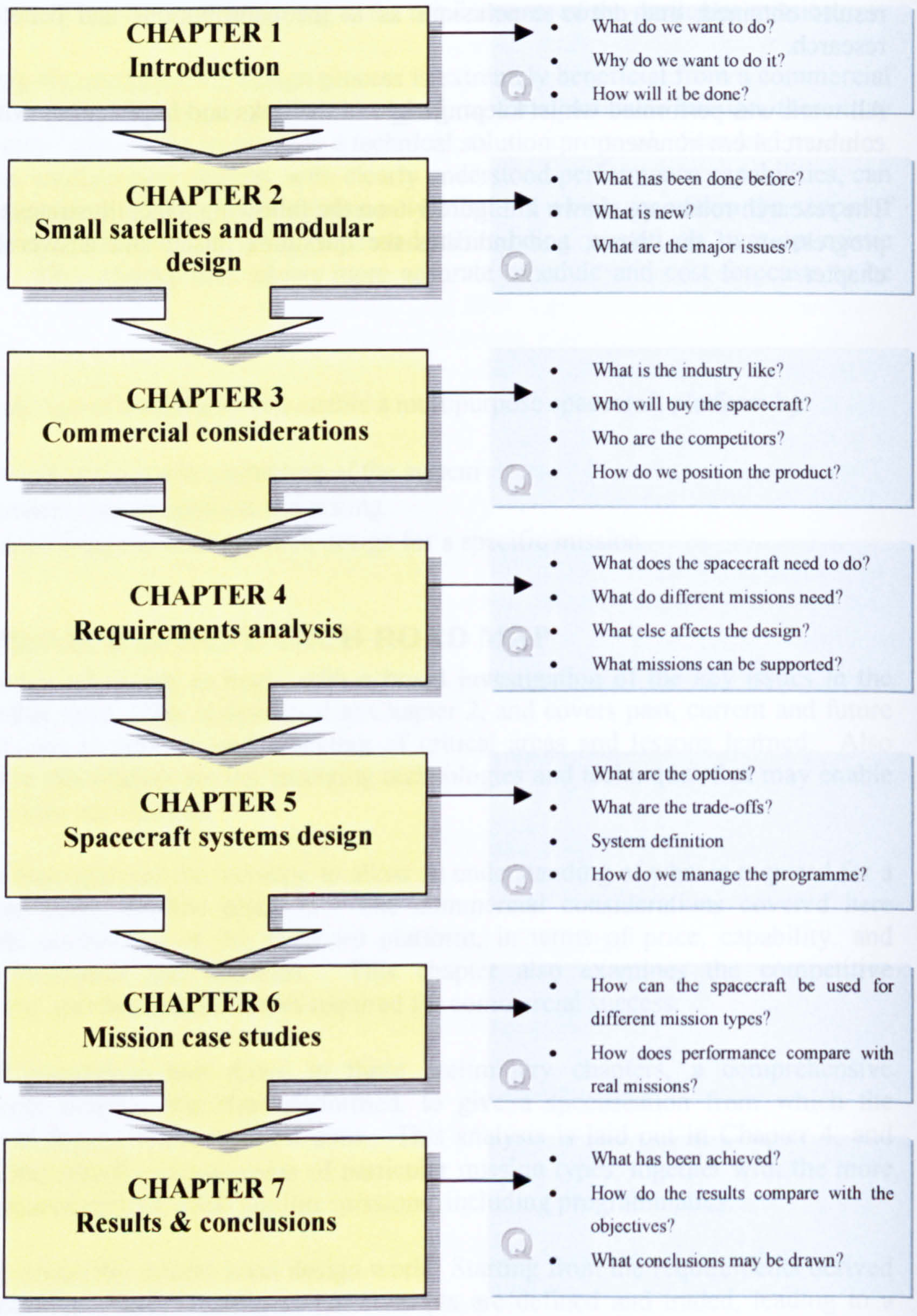

Figure 1-6 Research roadmap

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2 SMALL SATELLITES AND MODULAR DESIGN

This chapter gives a background to the general subject of small satellites, and then continues in greater detail to investigate the specific aspects of multipurpose platforms, modularity and use of new technologies as small mission enablers. The aim of this part of the research work is to identify the main issues relating to the design and use of small spacecraft, and the lessons learned that may be applied to a multipurpose modular platform.

2.1 SMALL SATELLITE DEFINITIONS

The terminology used in the small satellite field varies appreciably between references. A satellite is generally considered "small" if it has a mass of less than a tonne, but the description is often broken down into smaller categories. The very tiniest satellites, emerging as a result of breakthroughs in nanotechnology and microengineering, are known as nanosatellites, and have a mass of only a few kilogrammes. These spacecraft make up a very new, and currently very rare, application area.

The next-largest class, microsatellites, are generally considered to be those which are around the 50kg mark, or which are of the ASAP 4 launch class. The class with which this research is mainly concerned is the minisatellite class. These are generally classified as spacecraft of mass 80kg up to 1000kg.

Of course, such loose categories will have poorly defined boundaries between them, and a single satellite may be described variously as "mini" or "micro" by different authors. Terminology also varies between countries; small satellites are also sometimes known as "lightsats" in the US.

It should also be noted that the definition "small satellite" also tends to refer to the project size and an overall philosophy of "faster and cheaper", and not only to the physical size or mass of the spacecraft.

2.2 LARGE VS SMALL SPACECRAFT: A COMPARISON

A primary driver in the design of practically all spacecraft, large or small, is the constraint of mass. This is of course interlinked with cost drivers, because of the very high specific launch costs mentioned in the introduction. Therefore, the general techniques used to minimise mass, such as choice of materials, will often be used in both large and small spacecraft.

Similarly, the overall environment encountered by the onboard equipment will be comparable for both types of spacecraft category, so the general guidelines for the uses of materials, components and mechanisms in space will apply equally. (There will be certain differences – smaller spacecraft may offer less inherent radiation shielding to electronics housed within them than larger vehicles, as on average there is less structure surrounding each piece of equipment.)

In terms of overall design, small spacecraft generally use similar types of equipment, subsystems and components to those on large spacecraft, but on a smaller scale and lower performance. The key differences are the complexity and redundancy of the systems used. Small spacecraft may often employ single-string designs, and reduce risk via simplicity rather than redundancy.

Structurally, design is simpler for smaller spacecraft, as the load paths are often shorter and less complex. With smaller structural members, it is easier to meet stiffness requirements imposed by the launcher. Thermal design of small satellites is also generally simpler than for larger spacecraft, as the thermal paths are shorter, and the surface area to mass ratio is greater, offering more opportunity for radiating excess heat.

Certain launch opportunities are more constraining on volume or shape than on mass. This may be particularly true of shared or piggyback launches, where a more irregular payload envelope may arise from the demands of the primary passenger. For this reason, small spacecraft are generally considerably denser than larger spacecraft.

In terms of programmatics, small spacecraft projects generally use much smaller teams, often co-located, and few sub-contractors. Procurement and QA policy is also often different, with the use of COTS (Commercial Off-The-Shelf) components being much more widespread.

2.3 USES FOR SMALL SATELLITES

The range of applications that may be serviced by small spacecraft is covered in greater depth in Chapter 3, together with potential customers and users of such spacecraft. However, an overview of the major "typical" uses is given in Table 2-1.

Application	**Remarks**
Science	Lower-cost "budget science" missions University research missions.
Earth observation	Use in land resources and disaster monitoring. Lower-cost weather satellites for developing countries. Surveillance/reconnaissance
Communications	Recent interest in LEO constellations for global communications, voice, messaging, broadband. Amateur store-and-forward Lower-cost option to give developing countries a domestic sat-coms capability. Secure, independent point-to-point communications and messaging for large organisations and the military.
Technology	Demonstration and test of new instruments in space.

Table 2-1 Typical applications for small spacecraft

Small satellites are best suited to missions that are focused on one particular task or goal. They will often have only one or two main payload instruments; in contrast, some of the large NASA science spacecraft may have upwards of 10 or 20 experiments on board.[13] The following section gives some examples of small spacecraft produced around the world.

2.4 OVERVIEW OF THE CURRENT SMALL SATELLITE SITUATION

2.4.1 "SMALLER, FASTER, CHEAPER"

Small satellites are not a new field. As mentioned in the introduction, the earliest satellites were small, simply because launch capacity was limited. However, throughout the space age, small spacecraft have been utilised – about 20% of the total world launches up to 1994 were below 400kg[62]. The difference with the current use of small spacecraft is that progresses in onboard technologies mean that the capabilities of smaller spacecraft can be much greater. This led to a change in the perception of the usefulness of small spacecraft.[24]

In the late eighties, NASA declared a new approach to space missions. The Small Explorer (SMEX) program was started in 1988, to provide frequent flight opportunities for small, low-cost science missions.[25] The first SMEX spacecraft was SAMPEX (Solar Anomalous and Magnetospheric Particle Explorer), launched in 1992. NASA cited the aims of the program as below:

The Small Explorer program is designed to accomplish frequent, high-quality space science investigations utilising innovative, streamlined, and efficient management approaches. It seeks to substantially reduce mission cost through commitment to, and control of, design, development, and operations costs, as well as to improve performance and reduce cost through the use of new technology. Finally, it seeks to enhance public awareness of, and appreciation for, space science and to incorporate educational and public outreach activities as integral parts of space science investigations.
-NASA Office of Space Science.

This type of programme became known as the "Smaller, Faster, Cheaper" approach, and promoted the value of small spacecraft. The new approach was, of course, mainly in response to budget cuts, which required the down-scaling of planned projects, but it did have the effect of heightening interest in small satellite applications.

2.4.2 SMALL SATELLITE USE WORLDWIDE

This section gives an overview of global activity in small satellite projects and programmes.

The USA

There are now a number of small satellite programs in the USA. NASA's SMEX program continues; the latest spacecraft in the series were RHESSI (Ramaty High Energy Solar Spectroscopic Imager), launched in February 2002, and GALEX (Galaxy Evolution Explorer), launched in July 2002. SMEX is now joined by the New Millennium Program and Student Explorer Demonstration Initiative (STEDI).

The New Millennium missions aim to test advanced technologies in space, whilst performing useful science and Earth Observation tasks. The first in the series, the successful comet-encounter mission Deep Space 1, was launched in 1998. Other small missions in this program include the Earth Observing 1 satellite, which was launched in 2000, and is being flown in formation with Landsat 7 in order to compare the respective images obtained. Future New Millennium projects include a mission comprising several nanosatellites, which will demonstrate new technologies and formation flying. This program has seen the use of some exciting new technologies and techniques, many of which would be of great interest to commercial platform providers and are discussed in Section 2.6.

The STEDI program is managed, for NASA, by the Universities Space Research Association, and its low-cost spacecraft are built, tested and operated by universities. The first in the series was SNOE (Student Nitric Oxide Explorer), built by the University of Colorado and launched in 1998. This was followed up in 1999 by the unsuccessful TERRIERS (Tomographic Experiment using Radiative Recombinative Ionosperic EUV and Radio Sources) mission. The third spacecraft in the series, CATSAT (Cooperative Astrophysics and Technology Satellite) is nearing launch readiness and awaiting launch manifest.

The US Department of Defense has employed several small satellite programs, such as STEP and MSTI. The Space Test Experiment Platform (STEP) was a small standardised spacecraft bus designed to take a series of technology payloads developed by the US Air Force Phillips Laboratories. There were some problems with deployment of the solar panels on this platform, however, and this series did not continue. The MSTI (Miniature Sensor Technology Integration) program was run by the US Air Force, and used small satellites equipped with infra-red sensors to demonstrate detection and tracking of ballistic missiles. The last in the series, MSTI 3, was launched in 1996. The current Space Test Program (STP) provides space flights for DoD sponsored experiments. Missions range from "piggy-backing" experiments onto other spacecraft, small and medium-class satellites, to the shuttle. The STP and the Air Force Research Laboratory (AFRL) joined forces for the US$23.5m Mightysat Phase 2 programme, a series of 5 small spacecraft procured from Spectrum Astro. The first in this series was launched in 2000 and carried a number of new technologies including concentrator solar cells, multifunctional structures and a bimodal composite experiment.

Europe

In Europe there has also been increasing interest in the development and use of small spacecraft. ESA's first small satellite, PROBA (Project for Onboard Autonomy) was launched recently, and is being operated successfully.

In the UK, the DERA (Defence Evaluation and Research Agency – now privatised as QinetiQ) Space Technology Research Vehicle program has produced four small technology demonstration spacecraft. STRV 1-a and 1-b were launched in 1994, and demonstrated new solar cell technologies. STRV 1-c and 1-d were launched in 2000 but unfortunately suffered early orbit failures. QinetiQ is also developing a small remote sensing spacecraft, funded by the BNSC/MOD and planned for launch in 2003.

Elsewhere in Europe, national space agencies have been funding national small satellite projects. In 2000, the Italian Space Agency-funded MITA (Microsatellite Italiano di Technologia Avanzata) minisatellite was launched, as a demonstration of technology and technical capability. Champ (Challenging Mini-Satellite Payload) was developed by Germany's space agency, DLR, and launched in 2000 to study ionospheric/atmospheric physics & magnetic fields. The Swedish National Space Board, together with France, Finland and Canada, funded ODIN, launched in 2001, which studied ozone depletion and also conducted a search for interstellar water and oxygen.

Asia-Pacific

Japan has been launching small spacecraft on its own H2 launch vehicle, including a pair of satellites used to demonstrate autonomous docking manoeuvres. 2002 saw the launch of MDS-1 (Mission Demonstration Satellite), a technology demonstrator. Elsewhere, Taiwan launched its first small science satellite, the 400kg ROCSAT-1, in 1999, and Malaysia, Korea, and Thailand have all launched SSTL-produced microsatellites, attaining space presence for the first time.

Middle East

Israel also has its own launch capability, and uses its Shavit rocket to launch small spacecraft, including the Ofek series of reconnaissance satellites. Ofek-5 was launched in 2002. Israel is also working with OHB in Germany to produce the Diamant Earth observation satellite. The Egyptian government have procured a 100kg spacecraft, Egyptsat-1, to carry out Earth imaging and store-and-forward communications.

Russia

The Russian Federation, and the USSR before it, has long employed small spacecraft in its space program. The heavy-lift capability of the Russian launchers means that "batches" of satellites have often been launched together, as in the case of the 6 minisatellites launched in 2001, 3 of which were military Cosmos spacecraft designed for reconnaissance and secure messaging. Similarly, smallsats are also employed in the Gonets LEO data communications network.

South America

The Argentinean National Commission of Space Activities (CoNAE) produced SAC (Satelite de Aplicaciones Cientificas)-A, in 1998, to demonstrate new equipment and technologies for use on later missions. This was followed up in 2000 by the US$30m SAC-C, which performed a remote sensing mission. Brazil launched its SACI-1 small satellite in 1999. This mission studied cosmic rays and plasma physics.

International/commercial organisations

The international Radio Amateur Satellite Corporation (AMSAT) has been involved in the production of amateur radio satellites since AMSAT-OSCAR-5 in 1970. These amateur satellites provide real-time communications and digital store-and-forward services. There are various other groups around the world similar to AMSAT; these organisations generally produce small, low-cost satellites using a lot of volunteer effort and donated equipment. Several of the OSCAR spacecraft were micro satellites built by the University of Surrey. The latest in the AMSAT series, the 400kg AMSAT Phase-3D, was launched in 2000.

Commercial small satellite ventures have also been big news in recent times. The Low Earth Orbit communications constellations such as Globalstar and Iridium enormously increased the numbers of small satellites being produced and launched in the late nineties, but the highly public commercial failure of these enterprises has given a sense of caution to this type of smallsat application. This is discussed in greater detail in Chapter 3.

Also of interest are the relatively large numbers of new launch vehicles being developed. A sudden rush to produce a reusable launcher has seen the proposal of many new and innovative designs for reducing the cost of launches. If any of these new vehicles successfully reach the market, it can only mean good news for satellite manufacturers, as any reduction in launch cost will place satellites within financial reach of more customers. If a satellite design has sufficient flexibility to adapt to new types of launch opportunity without significant changes, then so much the better.

2.4.3 RECENT SMALL SATELLITE MISSIONS

As part of the research work, an investigation was made into recent small satellite missions, in terms of their design, characteristics, and application type. A small selection of spacecraft is described here, to give an overview of the spectrum of designs and application areas covered by this field. An example mission has been chosen from each of the main categories of science, Earth observation, communications and technology demonstration. A more extended summary of recent small satellite missions is given in Appendix A. The aim of this investigation was to begin to gain an understanding of the key issues for small missions. This leads into the more detailed requirements analysis conducted in Chapter 4.

2.4.3.1 WIRE (Wide Field Infrared Explorer)[22] – Science

WIRE was launched in March 1999 on a Pegasus-XL. It was part of the NASA Small Explorer (SMEX) program, and carried an Infrared astronomy payload designed to study the evolution of starburst galaxies. Its main parameters are shown in Table 2-2.

Mass	259kg
Power	158W
Structure	Fibre-reinforced composites
Orbit	540 km, 97° inclination
Payload	30cm Cassegrain IR telescope, solid hydrogen cryostat (7K/12K)
Solar arrays	GaAs cells, deployed panels
Batteries	9Ah super NiCd
Pointing accuracy	1arcminute
ADCS	3-axis gyro package, star tracker, digital Sun sensor, 6 coarse Sun sensors, 3-axis magnetometer, wide-angle Earth sensor, 4 reaction wheels, 3 magnetorquers,
Onboard data handling	80386/7 processor, 1553 data bus
Onboard data storage	30Mbyte solid state recorder
Communications	S-band, 2.25Mbps downlink, 2Kbps uplink
Mission duration	4 months
Cost	US$45M
Delivery time	36 months

Table 2-2 WIRE spacecraft parameters

The spacecraft was configured with the observatory payload mounted on top of the main spacecraft platform via couplings selected to minimise the thermal path between the two. The spacecraft layout is shown in Figure 2-1. The composite structure was selected to minimise the mass of the spacecraft, necessary to allow launch on Pegasus-XL.

The sun-synchronous orbit was required, so that the mission would experience no eclipses, and have a constant orientation with respect to the sun vector. This simplified the thermal control requirements and design of the power subsystem. The orbit also had to be selected so that it would be compatible with the sky surveying that was the primary mission.

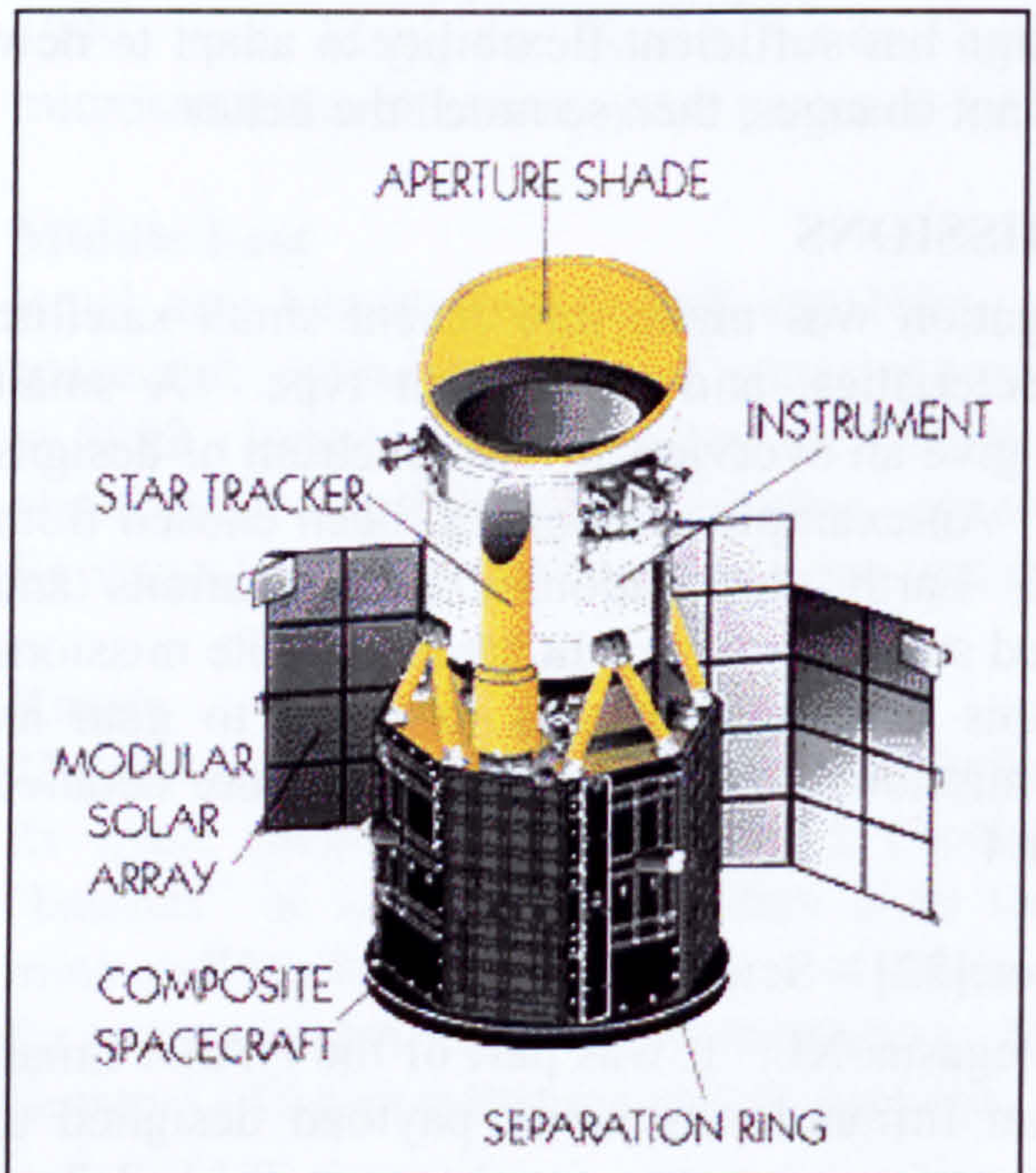

Figure 2-1 The WIRE spacecraft configuration

The struts mounting the instrument to the spacecraft prevented thermal coupling between the two parts of the satellite. The instrument was cooled by a cryostat to 7 Kelvin.

A major aspect of an astronomy mission such as WIRE is the attitude control subsystem. As well as pointing the spacecraft to its required target to arc minute accuracy, and avoiding jitter, it must also prevent the instrument from pointing at or near the sun, moon, or Earth.

Summary of key issues & design points for the WIRE mission:

- Thermal isolation of payload
- Sun-synchronous orbit
- Pointing
- Mass constraint

2.4.3.2 MightySat II.1[53] – technology demonstration

MightySat II.1 was launched in July 2000 by a Minotaur launcher. It was produced by Spectrum Astro for the US Air Force, and flew 10 advanced technology experiments. These included a Fourier Transform Hyperspectral Imager, several new solar array technologies, and a multifunctional primary structure demonstrator. The main MightySat spacecraft parameters are shown in Table 2-3.

Mass	125kg (payload 37kg)
Size	69cm x 89cm x 89cm
Power (average)	Bus 90W, payload 60W
Structure	Composite primary structure with integral VME card cage
Orbit	556km, 98deg inclination
Solar arrays	2 deployed arrays, 2-axis articulated, silicon cells, 330W at EOL
Batteries	12Ah NiCd battery
Onboard data handling	RAD6000 processor, 20MHz, 128Mbyte RAM + 256Mbyte RAM for payload
ADCS	3 reaction wheels, 3 magnetorquers, star tracker, fibreoptic gyro
Pointing accuracy	729 arcsec (control), 540 arcsec (knowledge), 16 arcsec (jitter)
Communications	Via SGLS, AFSCN. Downlink 20Kbps (telemetry), 1Mbps (data). Uplink 2Kbps
Thermal control	Passive radiators. Emergency heaters.
Mission lifetime	>1 year
Cost	US$23.5M for series of 5 buses
Delivery time	24 months

Table 2-3 MightySat spacecraft parameters

The bus was largely based around Spectrum Astro's SA-200B bus, and was a very densely-packed cubic structure, with the VME electronics cards slotted into an integral card cage, maximising the volumetric efficiency of the design. Some of the payloads actually formed integral parts of the spacecraft bus itself, so the configuration was not separated into payload and "service" sections. These integral payloads were known as the "Experimental Bus Components", as opposed to the "Stand Alone Experiments". However, there was provision for large payloads to be mounted on the upper deck. The spacecraft configuration is shown in Figure 2-2.

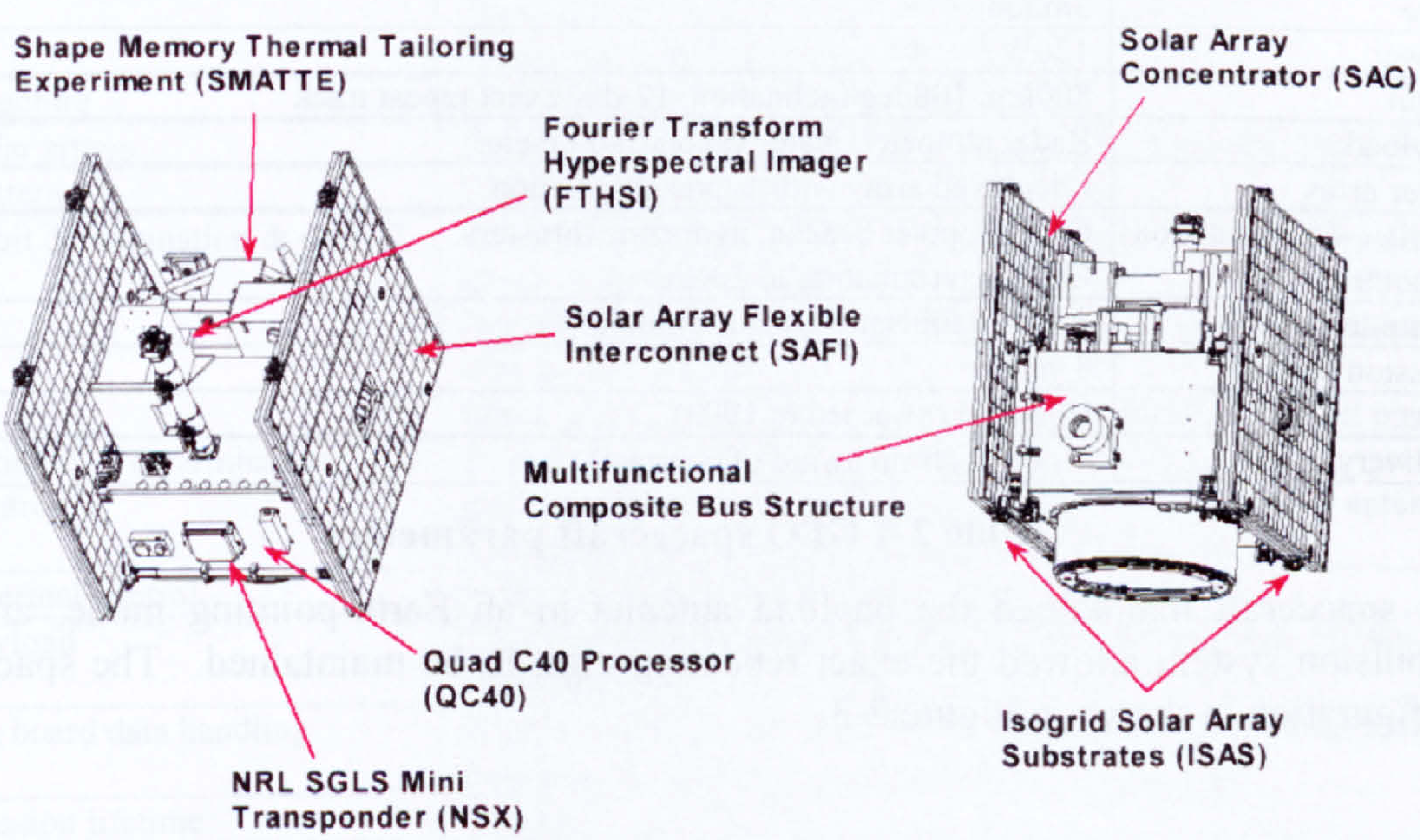

Figure 2-2 The MightySat II.1 spacecraft configuration, showing the position of the payloads

The sun-synchronous orbit simplified the thermal control subsystem, which was able to mainly rely on passive radiator surfaces. The high-precision attitude control was part of the technology being demonstrated by the mission (it is not believed to have been a necessary requirement for any of the other payloads).

MightySat II.1 is intended to be the first in an on-going series of small technology demonstration missions for the US Department of Defense. It is to be expected that the mission configurations of successive spacecraft may be somewhat different, as they will largely depend on the technologies being flown, particularly the Experimental Bus Components.

Summary of key issues & design points for the MightySat II.1 mission:

- Incorporation of payloads as part of the core system
- Large number of payload items
- Rapid production
- Low cost

2.4.3.3 GFO (GEOSAT Follow-On) – Earth observation[5,52]

The GFO spacecraft was launched in February 1998 by a Taurus launcher, as a follow-on to the 1985-1990 GEOSAT program. Its mission was to provide real-time ocean topography data to US Navy users. The data was also archived and later made available to the scientific community. Ball Aerospace produced the spacecraft for the US Navy. Its main characteristics are shown in Table 2-4.

Mass	300kg plus 47kg payload, (plus fuel?)
Size	3m long
Power	126W
Orbit	800km, 108deg inclination, 17-day exact repeat track
Payload	Radar altimeter, water vapour radiometer
Solar array	I deployed array with 1-axis articulation
Orbit determination & control	GPS, Doppler beacon, hydrazine thrusters. Orbit determination to 10cm.
Attitude control	3-axis stabilisation, nadir-pointed
Mission lifetime	8 years
Cost	US$46M (awarded in 1992)
Delivery time	~6 years (from award of contract)

Table 2-4 GFO spacecraft parameters

The spacecraft maintained the payload antenna in an Earth-pointing mode, and the propulsion system allowed the exact repeating orbit to be maintained. The spacecraft configuration is shown in Figure 2-3.

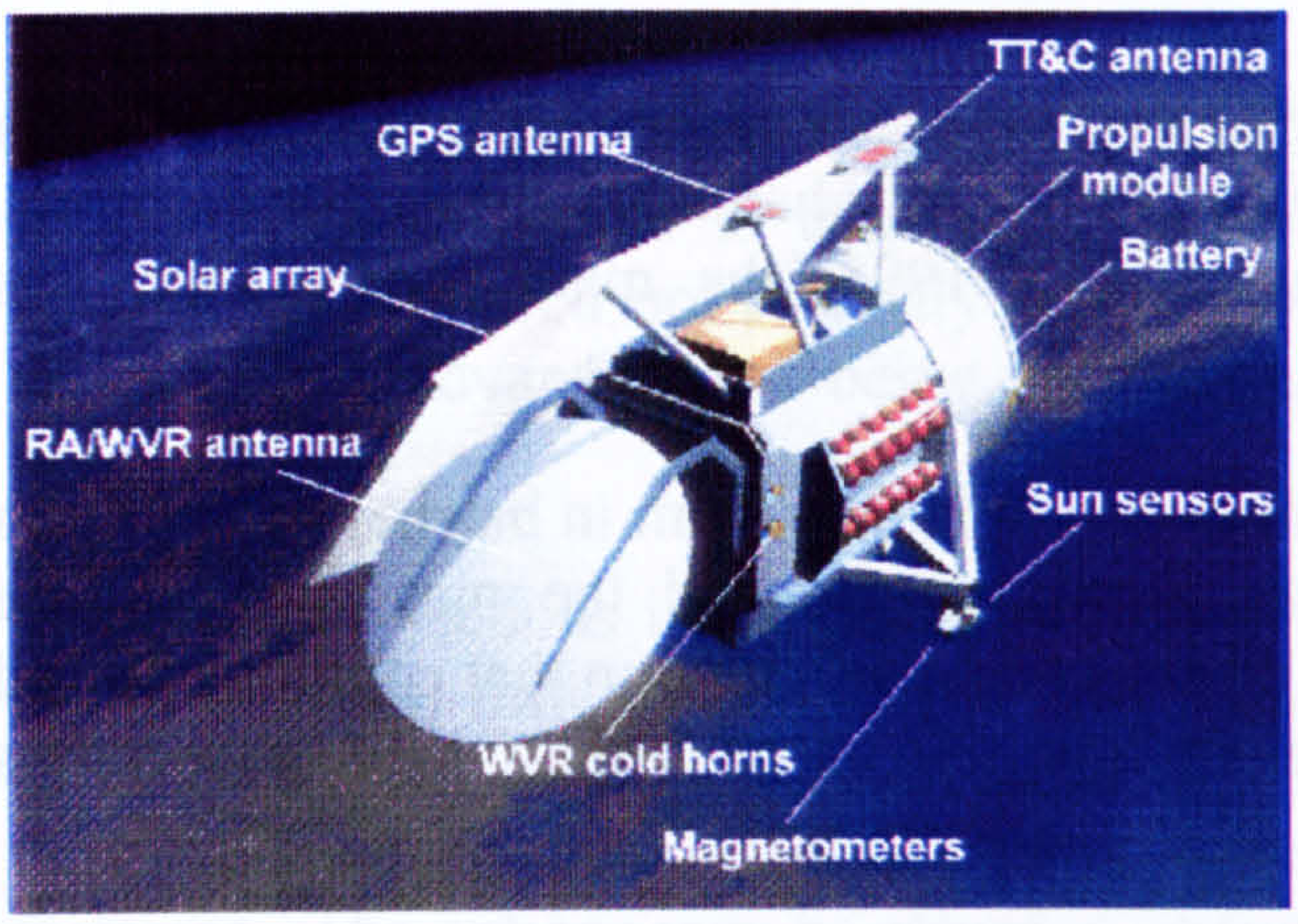

Figure 2-3 The GFO spacecraft configuration

[Image: Space & Naval Warfare Systems Command]

The spacecraft contract award included incentive fees based on long on-orbit lifetime. Options for a further two spacecraft may be exercised in the future.

Summary of key issues & design points for the GFO mission:

- Long mission lifetime
- Precise orbit determination
- Precise orbit maintenance
- Nadir-pointing

2.4.3.4 ORBCOMM – communications[10]

The ORBCOMM spacecraft form a constellation of 26 small LEO satellites, and are designed to provide global messaging services. The main spacecraft parameters are shown in Table 2-5.

Mass	47.5kg
Power	126W
Orbits	~812x824km, 45deg (3 planes of 8 spacecraft) 728x747km, 70deg (1 plane of 2) 781x874km, 108deg (1 plane of 2)
Structure	AlBeMet skins on Al honeycomb
Solar arrays	2 deployed, single-axis articulated arrays, silicon cells, 270W BOL
Batteries	5 CPV NiH2 batteries
Orbit determination & control	GPS. 2 cold-gas thrusters, nitrogen blow-down system. Propulsion along velocity vector only. Principal orbit control is via steering solar arrays during eclipse to alter ballistic coefficient. Spacecraft must remain within +-5degrees within their orbit plane.
Attitude determination & control	2 Earth sensors, 3-axis magnetometer, 6 sun sensors, gravity-gradient tip-mass, integral solar sail/drag sail with boom antennas, 3 magnetorquers. 5 degree nadir-pointing accuracy.
Thermal control	Passive radiators, heaters on batteries & hinges
Payload	VHF quadrifilar antennas, UHF quadrifilar antenna, transceiver, transmitters, receiver.
On board data handling	Flight computer with dual 68302 processors, controlling 5 buses. Processor in each subsystem, distributed architecture.
Mission lifetime	4 years
Cost	US$21.4M for development of initial 2 satellites. Recurring cost estimated at $2M per spacecraft.
Delivery time	3.5 years for initial 2 spacecraft

Table 2-5 ORBCOMM spacecraft parameters

The primary aims of the system are global coverage, near-real-time availability for messages up to 256 bytes, and to be accessed via small, handheld subscriber terminals. Two demonstration satellites were launched in 1995; the main constellation was launched in 1998 and 1999.

Design of the spacecraft centred on the intention to launch them in batches of eight at a time on a Pegasus-XL launcher. This greatly constrained the mass and volume available to each satellite, and drove the disc-shaped configuration that could be stacked to exactly fit the launcher envelope. The configuration of the ORBCOMM spacecraft is shown in Figure 2-4.

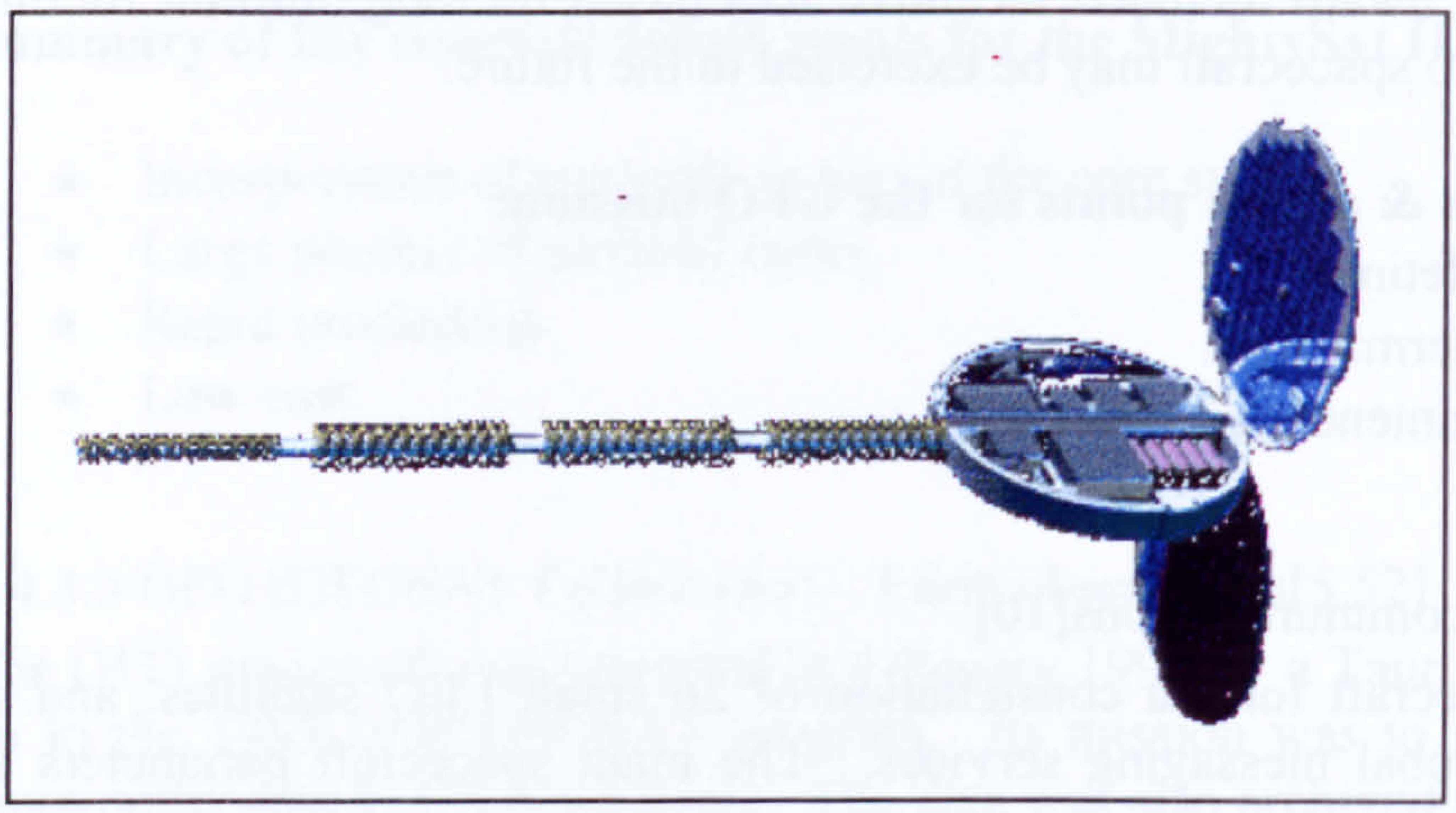

Figure 2-4 The ORBCOMM spacecraft configuration

The tight volume constraints also gave a need for equipment to be stowed for launch, and deployed once in orbit. The communications payload required large quadrifilar antennas; a method was devised for hinging these and stowing them in a folded arrangement. The solar arrays were folded to the spacecraft body for launch, and hinged out once in orbit. Although only articulated about one axis, second axis control could be enabled by yawing the whole spacecraft to follow the sun vector.

Mass was minimised by use of the composite primary structure, and minimising the equipment carried. For example, no reaction wheels were used, as no suitable ones were available at the time of development. Instead, the attitude control scheme was designed around small magnetorquers and use of the gravity-gradient control effects arising from the existing antenna boom structure.

Summary of key issues & design points for the ORBCOMM mission:

- Low recurring cost
- Design for multiple launch, on a specific launcher (Pegasus-XL)
- Minimise mass and size
- Maintenance of relative in-plane position
- Antenna stowage & deployment

2.4.4 MULTIPURPOSE COMMERCIAL PLATFORMS

A multipurpose system is a general-purpose platform, not designed for a specific mission. It may well not perfectly fit the ideal mission requirements, but has the cost and schedule advantages of being an existing design, and, potentially, having flight heritage.

The previous sections illustrated how different small spacecraft can be, and how their key drivers may differ, when they are specifically designed for particular missions. For this reason, some multipurpose systems may have a degree of flexibility, with two or three "options" of differing performance, which broadens their suitability to a range of missions. The options may be afforded perhaps by varying solar array areas or attitude control system specifications This is where a modular approach is an advantage, as outlined in the introduction. Key requirements for a multipurpose platform are therefore a good overall capability that matches the maximum possible number of missions.

When designing a spacecraft platform, it is obviously essential to be aware of the existing spacecraft available, and their performance capabilities. Table 2-6 shows the range of mini- and microsatellites currently available from commercial suppliers, together with a brief description of their characteristics[39],[48].

Manufacturer	Platform	S/C mass/kg (payload)	Av S/C EOL power/W (payload)	Cost/ US$M	Heritage
Aerospatiale	Proteus	600 (300)			
Astrium	Flexbus	300-1000 (100-500)	100-1000 (100-600)	9?	Champ, GRACE
Ball Aerospace	BCP600	290 (90)	(125)	46	GEOSAT Follow-On
Ball Aerospace	BCP2000	988 (380)	(730)		QuikSCAT
Israel Aircraft Industries	Ofek -			50	Ofek 1-5 (Ofek 4 – launch failed)
Lockheed Martin	LM900	962 (470)	(344)		IKONOS, CRSS
Orbital	LeoStar-2	376 (210)	(118)	16.5	GALEX
Orbital	MicroStar	126.6 (68)	(50)	<10.7	BATSAT, ORBCOMM
Orbital	MidStar	1360 (780)	(327)		FUSE
Orbital	MiniStar	125 (25)	(25)	5-10	ACRIMSAT
Orbital	PicoStar	73 (20)	(10)		MightySat-1
Orbital	StarBus	766 (200)	(550)		IndoStar
Space Systems Loral	LS400	450	1100	48	Globalstar
Spectrum Astro	SA200B	130-190 (40-100)	(86)	5	MightySat-2.1
Spectrum Astro	SA200HP	1020 (666)	(650)	81.9	New Millennium Deep Space 1
Spectrum Astro	SA200S	329 (200)	(66)		MSTI-1, MSTI-2, MSTI-3
SSTL	Microsat-70	68 (24)	(1.5-18)	1.15	Tsinghua-1, PoSAT-1
SSTL	Minisat-400	407 (200)	(100)	9	UoSat-12
Swales	EO-SB	568 (236)	(256)	70.4	EO-1
Swedish Space Corp.	Freja-C	214	95 (BOL)	12.5	Freja
TRW	T100	220 (36)	(25)	111	TOMS-EP
TRW	T200A	317 (75)	(94)		ROCSAT
TRW	T200B	373 (95)	(175)	39	SSTI Lewis

Table 2-6 Commercially-available multipurpose platforms

A number of the platforms described are offered through the NASA Rapid Spacecraft Development Office (RSDO). The RSDO manages and directs a program aimed at facilitating the fast procurement of spacecraft and payload space for future missions. It does this by using previously demonstrated spacecraft and vendors, and encouraging the maximum definition of the payload before one of the spacecraft platforms is selected.

Most of the platforms described above have only become available in the last few years or less. Competition is now increasing between the platform providers, as there are several options in most size categories.

Many of the platforms are based on a design developed for a particular mission; the design being then re-used as a commercial platform, offered as a general purpose spacecraft bus. This makes good sense – it would be rather a waste if all the design effort and lessons learned in the production of a "one-off" spacecraft contract were not utilised in future designs. The manufacturer can make use of the design work, which has already been paid for, to reduce the cost of subsequent platforms. However, this does mean that the "offspring" platform may be rather better suited to missions that are similar to the "parent" mission. A platform that is "specifically non-specific", as the proposed platform is intended to be, may have an advantage here, in terms of the range of missions it can accommodate.

2.5 MODULAR DESIGN

A modular system may be defined as one that is composed of a number of self-contained units, which are easily removed and replaced without requiring significant architectural changes to the rest of the system. The replacing module may have a different performance, but it will still interface with the existing system.

The benefits of modular designs have been described in the introduction. Here, some previous spacecraft that employed modularity in their design are examined and their features and use of modularity analysed.

2.5.1 PREVIOUS MODULAR DESIGNS

2.5.1.1 MESA (Modular Experimental Platform for Science and Applications)[7]

Boeing's MESA concept originally evolved in the 1970's, in response to a need for small spacecraft for Department of Defense missions. The platform itself was developed in the early eighties; the configuration being based on the Viking platform that Boeing produced for the Swedish Space Corporation. The bus was designed for a dual launch, on Ariane, with the spacecraft placed between the launch adapter and the main passenger. The basic configuration of MESA is shown in Figure 2-5.

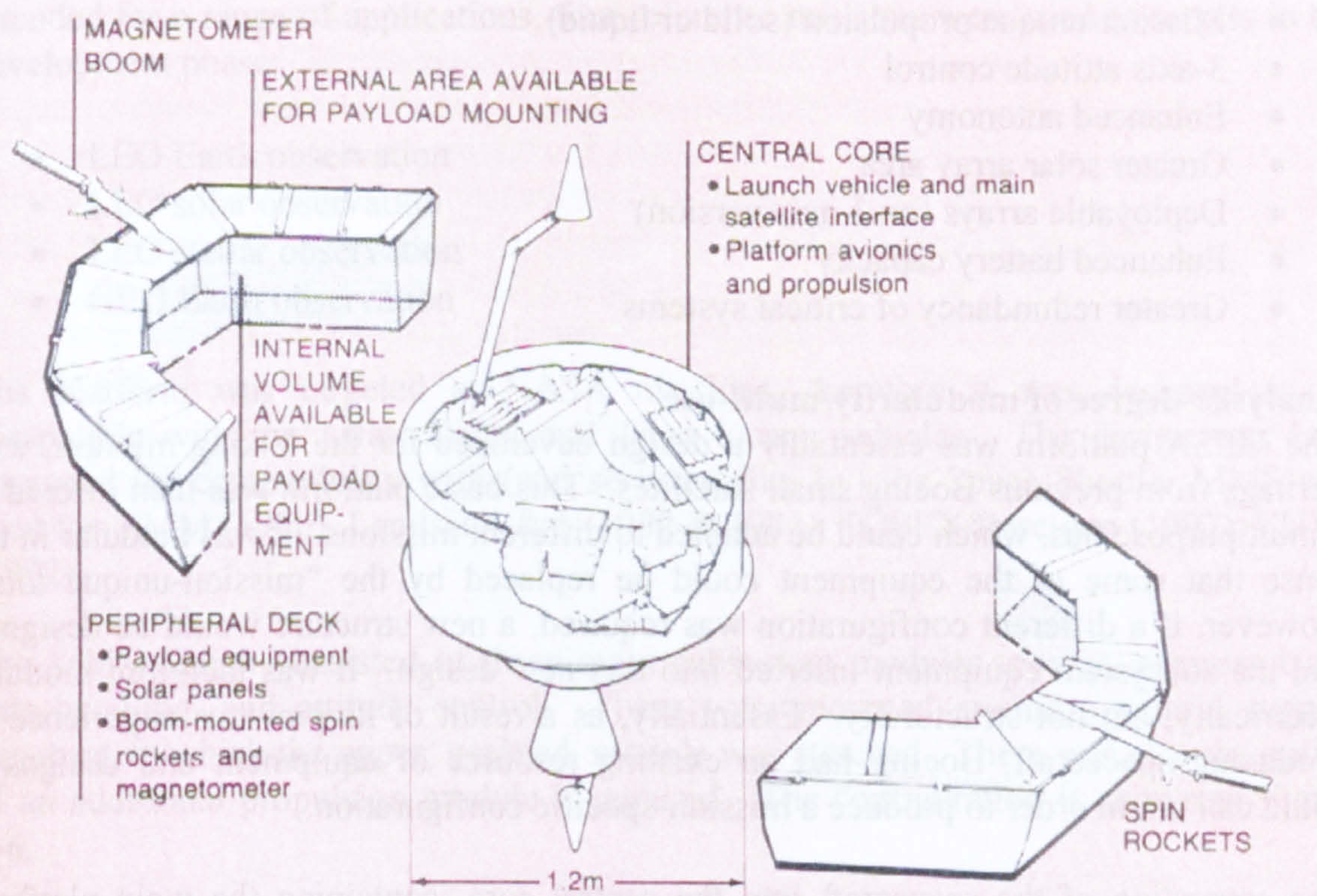

Figure 2-5 Exploded view of the MESA spacecraft. [Image: Boeing]

The main characteristics of the baseline spacecraft are shown in Table 2-7. MESA was expected to be used for Earth observation, data collection and small scientific missions.

Mass of platform (including solid kick motor)	421kg (motor+fuel=265kg)
Maximum stowed width	2760mm
Stowed height	945mm
Payload volume available	1.6m^3
Baseline launch vehicle	Ariane (1194mm interface)
Stabilisation	Spin, 3rpm
Attitude control method	Earth-Sun sensors, magnetometer, magnetorquers, solid motors for spin-up
Pointing accuracy	+-1deg
Power generation	8 body-mounted arrays
Peak power	120W (60W for payload)
Comms.	S-band, 55kbps
Thermal control	Passive: louvres, MLI, radiator plates

Table 2-7 MESA baseline spacecraft parameters

The MESA platform cost from US$(1981)10million, and the baseline spacecraft could also be enhanced by the incorporation of "mission-unique kits" at additional cost. Examples of such kits are shown below:

- Alternative launch vehicle adapters
- Increased data rates
- Compatibility with other ground stations/space relay networks
- Increased onboard data storage

- Mission-unique propulsion (solid or liquid)
- 3-axis attitude control
- Enhanced autonomy
- Greater solar array area
- Deployable arrays (on 3-axis version)
- Enhanced battery capacity
- Greater redundancy of critical systems

Analysis: degree of modularity/multi-use
The MESA platform was essentially a design developed for the Viking mission, with heritage from previous Boeing small satellites. This basic platform was then offered as a multipurpose bus, which could be adapted to different missions. It was modular in the sense that some of the equipment could be replaced by the "mission-unique kits". However, if a different configuration was required, a new structure would be designed and the subsystem equipment inserted into this new design. It was therefore modular electrically, but not structurally. Essentially, as a result of its previous experience in producing spacecraft, Boeing had an existing resource of equipment and designs it could call on, in order to produce a mission-specific configuration.

The separation of the spacecraft into the central core, containing the main platform subsystems, and the outer box-sections and decks which could be used by the payloads, gives further modularity. The platform and the payload are kept largely separate from one another; the two parts could in fact be described as a "service module" and a payload module.

Lessons learned from this design:
The ability to share a launch by stacking the main satellite on top of the smaller platform was a useful concept. It would also allow several of the MESA platforms to be stacked for a dedicated launch. The approach of having a range of standard parts plus the mission kits, from which the complete suite of subsystems required for a particular mission can be selected, is also of interest. This approach is more flexible that having two or three "set" configuration options.

The separation of the payload from the central spacecraft core allows the configuration of the platform itself to be optimised without having to accommodate payload items. At the same time, the available payload volume can be quite clearly defined in advance, and is quite extensive (with the internal volume plus external deck area).

Key points:
- Stackable design
- Equipment kits chosen from suite of subsystems
- Payload and system kept separate

2.5.1.2 MMS (Multimission Modular Spacecraft) [23,34]
The MMS platform was developed by GSFC in the 1970's, as a standard yet flexible spacecraft for large Earth observation and astrophysics missions. As the spacecraft was

intended for a range of applications, four reference missions were used as targets in the development phase:

- LEO Earth observation
- LEO solar observation
- LEO stellar observation
- GEO Earth observation

The platform was targeted at NASA missions, therefore it was designed to be compatible with the Atlas, Delta, and Titan launch vehicles. The design was later upgraded for compatibility with (and serviceability by) the Space Shuttle. MMS was used for: SMM (1980), Landsat 4 & 5 (1982 & 1984), TOPEX/Poseidon (1992), EUVE (1992).

The MMS design consisted of three main subsystem modules, power, command and data handling, and attitude control. These were mounted around a central support structure, to which the upper, payload, module was attached. There was also the option of an additional propulsion module if required. The configuration is shown in Figure 2-6.

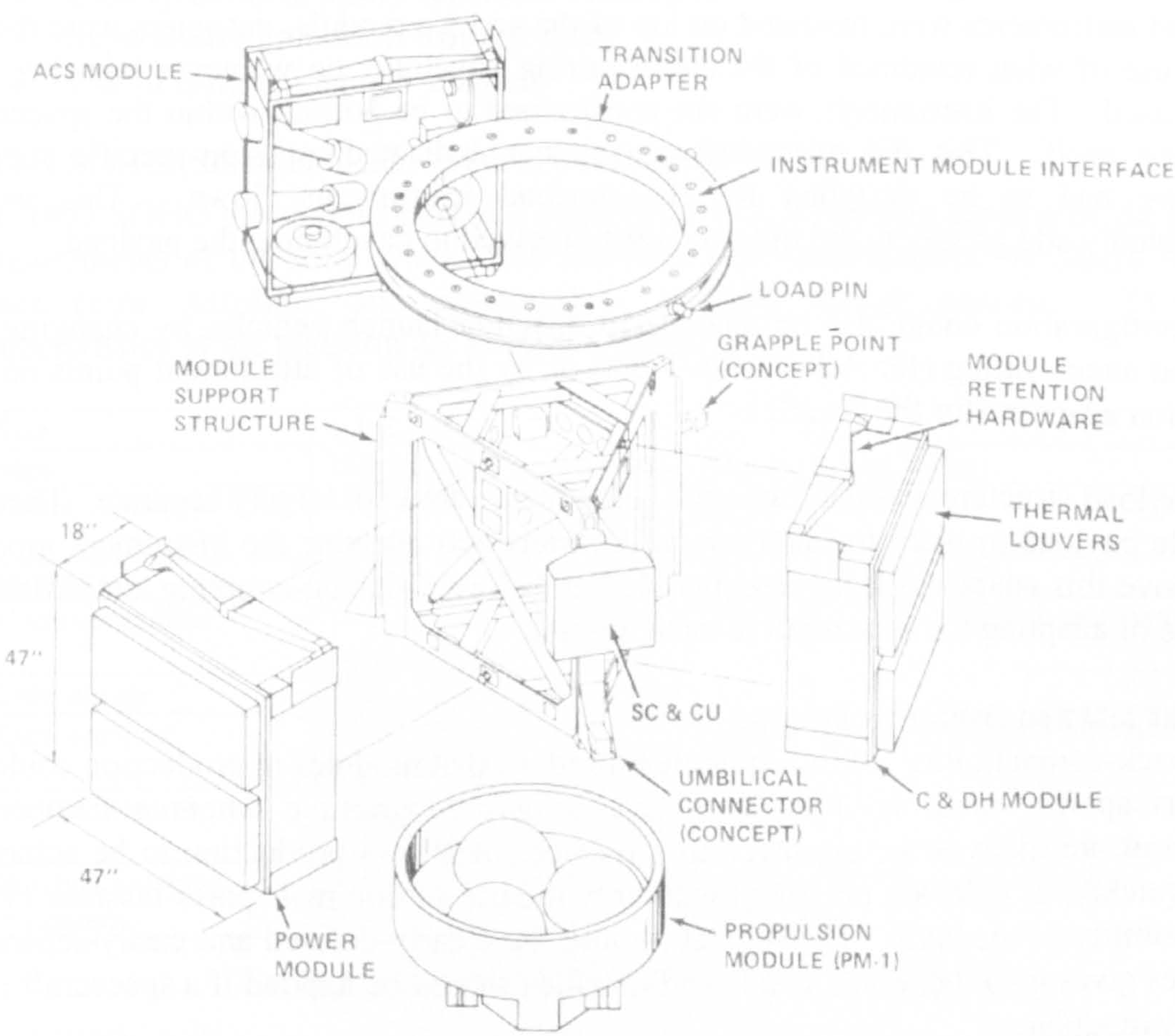

Figure 2-6 Exploded view of the MMS platform. [Image: NASA GSFC]

The subsystem modules had standard interfaces with each other, but their internal design could be modified for adaptation to particular missions. The modules were also "back-compatible" to allow modules in production for a new programme to be utilised as spares for in-orbit servicing of existing spacecraft. The modules were designed to be transparent to the technology contained within them; the interface remained the same even if the components within the module changed.

A particular advantage of the modular design was that it allowed the major part of the testing process to be performed at subsystem level. This greatly reduced the timescale of the AIT campaigns [shown in Fig 1-5 in the Introduction]. This reduced schedule time provided a significant lowering in costs for spacecraft projects utilising the MMS platform.

Analysis:
This was a truly modular design, in that modules could be readily interchanged with those of different performance, and spares could be easily adapted across programmes. The structural configuration of the modules themselves was fixed; the "service" part of the spacecraft was much the same for each mission.

However, the overall configuration was quite flexible, as the service section could support a wide range of different payload and mission-specific equipment. As the payload instruments were mounted on top of the service module, designers were free to make use of what remained of the entire fairing envelope, on whatever launcher was being used. The instruments were not constrained to be housed within the spacecraft structure itself. This did mean, of course, that dedicated, mission-specific support structure had to be designed for the payload instruments flown. This would undoubtedly add greatly to the time and cost involved in developing the payload.

The configuration could also be adapted to different launch vehicles by changing the payload attach fitting (for Atlas/Delta/Titan) or by the use of attachment points on the transition adaptor (for the Shuttle).

The payload structure and the "service" components are kept largely separate. There is a single payload-to-platform interface at the transition adaptor; the instrument module lies above this adaptor, the service module lies below. This de-coupling simplifies the process of adapting the spacecraft to each mission.

Lessons learned from this design:
The "back-compatibility" of the equipment used, so that modules in production could be used as spares, could be very applicable to any programme where a number of spacecraft are likely to be produced in sequence. It allows production to be achieved more quickly, as risks are mitigated with only the use of minimal spares because of the compatibility of the parts. The separation into the clearly-defined and easily-separable modules gives great benefits in AIT, and this idea should be applied if a spacecraft is to be truly modular.

Leaving the design and accommodation of the payload totally separate from the "service" part of the spacecraft has both advantages and disadvantages. It leaves payload designers more freedom in designing their instruments, obtaining required fields of view, and introducing unusual geometries. However, this freedom for the payload may put much more responsibility on the platform itself; it will require the platform and payload designers to collaborate much more closely, and from a much earlier stage, than if the allowable payload envelope and interface was more clearly defined. Loads to be accommodated, centres of mass, vibration modes and coupling effects must all be very strictly controlled when the payload becomes a largely separate structure. In the extreme, this method may become like designing two separate spacecraft.

The separate payload approach may be applicable to the small spacecraft being considered in this work, as long as the allowable interface constraints can be well defined as a part of the platform offered. A suitable template for this type of definition would be that used by launch authorities for the permissible properties of the satellites requiring launch services. The payload would then be treated as a passenger, with services and a specific interface provided to it by the platform, rather than as an integral and distributed part of the whole spacecraft.

Key points:

- Compatibility of equipment
- Clearly-defined, easily separable modules
- Separation of platform from payload
- Use of different launch adapters

2.5.1.3 SOHO (Solar and Heliospheric Observatory)[8,20]

The 1995 SOHO mission investigated solar dynamics, by remote sensing of the sun and measurements of the solar wind. The platform was manufactured by Matra Marconi Space (now Astrium), and employed a modular design concept. The main characteristics of the platform are shown in Table 2-7.

Size	4.3x2.7x3.7m
Mass	1861kg, (of which payload 655kg, propellant 250kg)
Power	1400W max. at 28V, from deployed solar arrays Payload mean power 440W, max 625W
Data rate	40kbit/s continuous, 160kbit/s part-time, 1.3kbit/s HK
Communications	S-band, via NASA Deep Space Network. Pointable high-gain antenna, two quadrifilar omni antennas.
Data storage	2Gbit solid state + 1Gbit tape
Data handling	Based on 16-bit processor, 524KHz data bus, standard interface units to each subsystem/experiment.
Attitude control	3-axis, using gyros, star sensors & sun sensors; actuated by reaction wheels and hydrazine thrusters.
Pointing accuracy	5-15arcmin absolute, <10" over 6 months, <1" over 15 minutes.
Autonomy	>48 hours

Table 2-8 SOHO spacecraft baseline parameters

The SOHO platform consisted of two main sections: the payload module (PLM) and the service module (SVM). The SVM contained all the platform subsystems and included a propulsion sub-module. The spacecraft configuration is shown in Figure 2-7. It was based around a central thrust-tube; equipment compartments being formed by shear walls. The PLM was also arranged around a thrust tube. The two modules were mated via the propulsion module, which itself was based around a short cylinder supporting the propellant tank. Separation of the discrete modules was further achieved via the use of thermal washers at the interfaces, to thermally isolate the PLM from the SVM.

The separation of the payload and platform was done to allow the PLM and SVM to be integrated in parallel. This was important due to the complexity of the payload experiments.

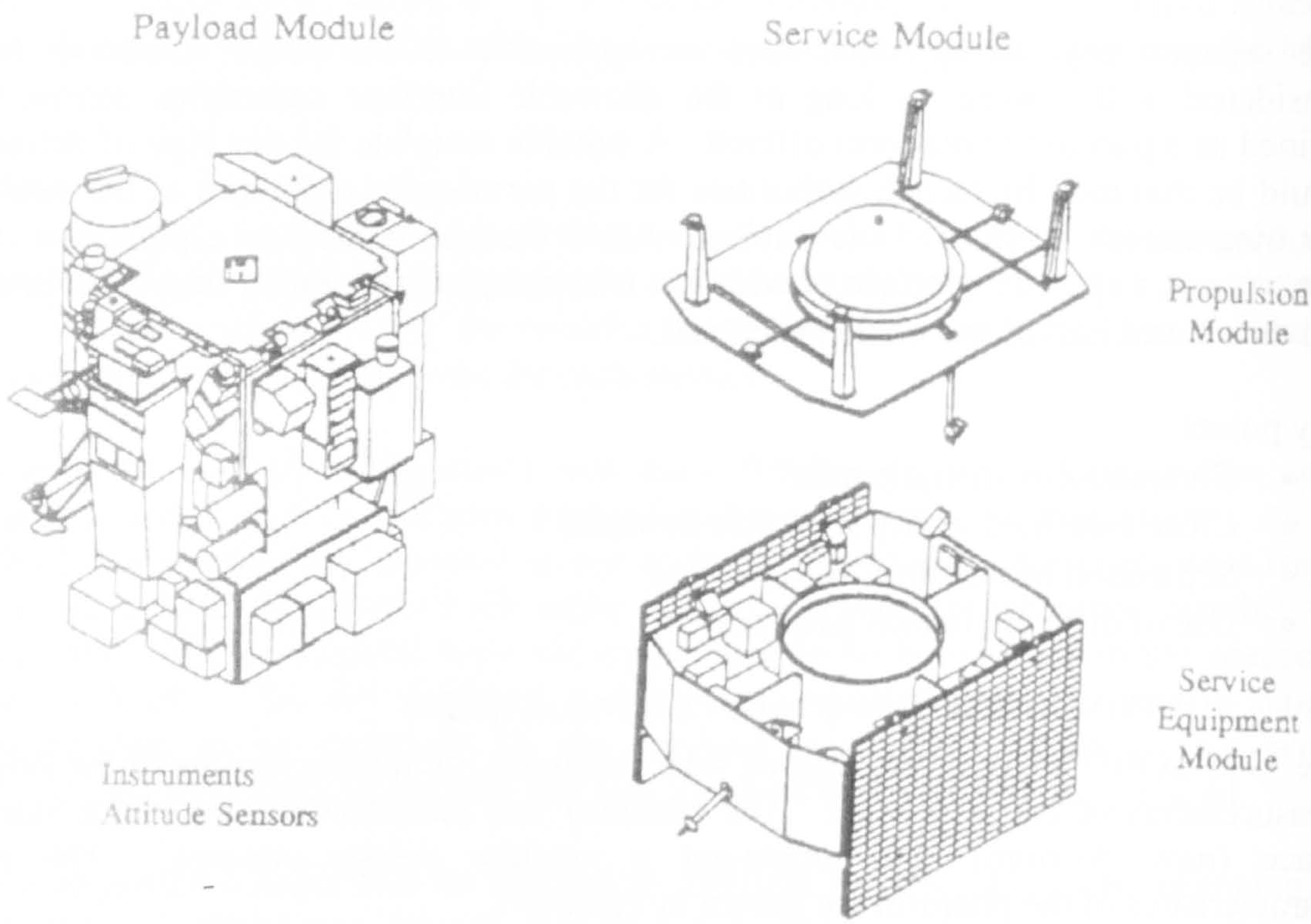

Figure 2-7 The SOHO spacecraft, showing the separate payload and service modules.

Electrically, the SOHO spacecraft also employed modularity. The electrical interfaces were standardised, and validated early on in the integration process. Each major spacecraft component (data handling, attitude and orbit control, two payload) had its own dedicated power distribution unit. An ESA OBDH bus and Remote Terminal Units were used for onboard data handling, allowing each module to exchange data via standard interfaces. The RTU provided all the TM/TC data interfaces to and from each experiment of subsystem.

Within the SVM, the subsystems, each with their dedicated PDU and RTU, were mounted on separate de-mountable side panels. Each panel (and associated subsystem) could then be integrated and tested separately.

Analysis:
SOHO was designed in a modular fashion not to enable it to be reconfigured for different missions, but to make the integration and test process quicker and easier. Modularity was used to enable the spacecraft to be built up in discrete sections concurrently, and at separate locations. It also allowed the payload module to be fully tested in a standalone configuration, using a simulator in place of the service module, before the two halves were mated together. This would allow much easier troubleshooting in the event of an anomaly. This was a very specific type of modularity, that was developed for this one particular mission. However, it did make use of existing ideas and standards to facilitate the module interfaces.

Use of the Remote Terminal Units and Power Distribution Units ensured that each experiment or subsystem "looked" the same electrically, in terms of its interaction with the rest of the spacecraft. As these interfaces were so standardised, a single simulator could be used to verify the correct interaction of each item with the spacecraft, prior to integration. This is an idea which could be carried through to a multipurpose platform, as it would aid the goals of schedule minimisation. Costs could also be reduced by the potential for re-use of the same simulators for successive spacecraft.

Lessons learned from this design:
As with MMS, the SOHO solution of keeping the payload very separate from the general platform is very applicable to a multipurpose spacecraft concept. The use of a standard spacecraft simulator could also be of benefit – it could be supplied as a payload development tool, to back up the usual interface control documentation.

The use of some type of standard interface unit for power and data may be very applicable to a reconfigurable platform, as this maintains the transparency of the modules. Such units could also perhaps be provided for payload instruments, to allow easier payload to spacecraft interfacing.

Key points:
- Modules integrated and tested separately
- Use of simulators
- Use of power and data interface units

2.5.1.4 Cranfield University REMODEL (REconfigurable MODular Expendable Lightsat)[26,32,46]

The Cranfield REMODEL project was a group design study undertaken by MSc students in 1992-93. The aim was to design a multipurpose small spacecraft that could be reconfigured and expanded, giving adaptation to a range of missions. This was a design study only; no prototype or flight models were produced. It is discussed here as it attempted a greater degree of reconfigurability than had been done by previous designs.

The spacecraft was sized for the Pegasus launch vehicle, and designed around different configurations of trapezoidal-box modules. These modules each contained the subsystem equipment, loosely grouped by function, for example AOCS, data handling, power. Different spacecraft "options" could be given by assembling the system from either four or six of the modules; the larger option allowing for greater performance by having additional power and data handling/storage modules.

The spacecraft design concept, with the two different configurations, is shown in Figure 2-8. The transition from a four- to six-module configuration was achieved by rotating the modules both horizontally and vertically; the trapezoid side-angles then permitted the accommodation of six modules around the central core.

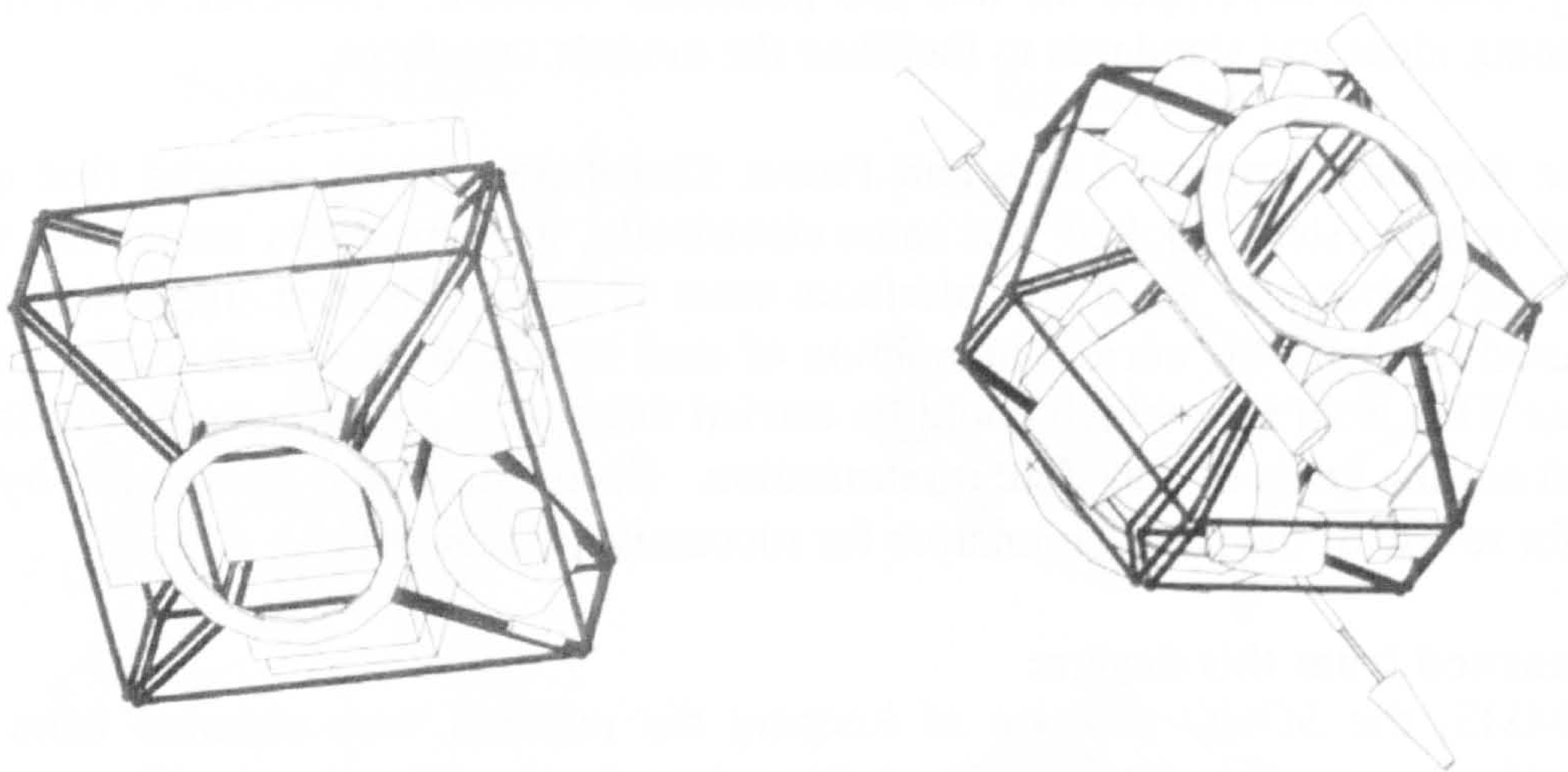

Figure 2-8 The REMODEL platform concept, shown in the four-module configuration (left), and the six-module configuration (right). [Image: King, 1992]

The main characteristics of the platform (in its two configuration variants) are shown in Table 2-9.

Platform mass	125kg (4 modules), 169kg (6 modules)
Payload mass (Pegasus launch)	Up to 154kg (4 modules), up to 110kg (6 modules)
Materials	Titanium tubes form module frames, CFRP covers
Configuration	Load-bearing modules, arranged in a ring of either four or six; may also be stacked.
Orbit	>400km, >30°
Attitude determination	Magnetometers, sun sensors, rate gyro
Attitude control	3-axis, via reaction wheels, magnetorquers
Pointing accuracy	33 arcsec (4 modules), 10 arcsec (6 modules)
Orbit average power (BOL)	180W (4 modules), 285 (6 modules), from body-mounted arrays
Payload power	135W (4 modules), 180W (6 modules)
Power bus	28VDC regulated
Onboard data handling	Computer based on 386 processor, 12kbps data bus
Onboard data storage	62.5MB tape recorder
Communications	S-band, 2 omni-antennas, TM downlink at 2.8Mbps, TC uplink at 2kbps
Compatible launchers	Pegasus, Pegasus-XL, Taurus
Thermal control	Passive; modules largely thermally isolated from one another

Table 2-9 REMODEL platform baseline parameters

Analysis:
In terms of modularity, this design does appear to compare favourably with the other modular spacecraft investigated, in that it can be fully reconfigured to a larger size whilst still using the same basic modular components. However, it should be remembered that whilst the other designs illustrated previously have gone through the full process of detailed design, manufacture, and launch, REMODEL was only developed to the conceptual design stage. The main advantages and potential drawbacks to the design are discussed below.

Advantages of the design:

- Individual qualification of modules – mechanical and thermal.
- Parallelisation of integration and testing process.
- Thermal isolation of each module
- Payload interfaces aimed to be well defined

Drawbacks:
The larger configuration has lower available payload mass even though it gives greater volume and power. This was because it was designed for use on a single launch vehicle; therefore when the platform was expanded, the payload fraction correspondingly reduced. If the concept was expanded to fit other launch vehicles this drawback would be removed. It would also suit the reconfigurable properties of the design.

Expensive materials were used, also very specific to this design. However, again, as all the modules are similar, there will be economies of scale with repeated manufacture.
The structure is also quite complex to fully analyse. However, once the analysis and verification had been done, as all the modules are mechanically similar, it would become easy – again giving economies of scale if production of several platforms is planned.

The trapezoidal modules are not very compatible with the common box-shaped equipment that will often need to be accommodated.

Lessons learned from the design:
The REMODEL concept aims for a truly modular and reconfigurable architecture, both electrically and mechanically. It has some innovative ideas, particularly the use of the trapezoidal module shape that can be used in either of two rotations, giving either the large or small configuration. The previous modular spacecraft were generally only modular in terms of the on-board systems, or the structure was modular to some extent but not reconfigurable. The idea of stackable layers is also interesting, although if the modules are designed to be strong enough to bear the load of an upper layer, there will be mass penalties for the smaller, single-layer configurations.

Key points:

- Reconfigurable structurally and electrically
- Modules can be integrated and tested separately
- Stackable
- Use of standard structural components

2.5.2 KEY ISSUES FOR MODULAR DESIGN

It can be seen from the examination of previous modular designs that the critical factors that enable modularity are the interfaces between the modules. This includes both the properties of the interfaces, and where the interfaces lie, i.e. how the onboard functions are partitioned into the separate modules. This section first analyses the different interface types, and the interface parameters to be considered for each.

Next, determination of the positioning and necessary characteristics of the interfaces is achieved by conducting a breakdown of all the functions that are performed onboard a typical spacecraft. This identifies the inputs and outputs required for each function, and their sources and destinations, and shows how the functions performed must interface/interact with the other subsystem functions on the spacecraft.

Another key consideration for enabling a modular system is that elements of the system that may need to be changed independently must be de-coupled from one another. This means that the minimum number of system elements are affected by a performance upgrade or alteration. For example, if a greater pointing precision is required, then a sensor may need to be exchanged for a different type. If the sensor is not decoupled from the rest of the AOCS subsystem, then the impacts on the system design will be much greater (and therefore more time-consuming and costly). This type of consideration will have to be made once the system requirements has been performed, and the performance increments expected of the system have been determined.

2.5.2.1 Properties of the interfaces

From the earlier definition, a system is modular if its sub-units can be removed and replaced with other sub-units. It therefore follows that the interfaces between these sub-units must be standardised. For a spacecraft, this would imply that if, say, an attitude control module was replaced by an upgrade, the new module would "look" the same as the old one from the point of view of the rest of the spacecraft. To achieve this, we must define what it is that makes a module look the same, i.e. what are the interfaces that must be standardised?

The interfaces that must be considered are as follows[33]:

- Mechanical
- Thermal
- Power
- Data
- Software

Interfaces are generally defined and described by Interface Control Documents (ICDs) and Interface Development Documents (IDDs). These documents should contain sufficient information that no further knowledge of the item described is necessary for the design of a connecting item and the mating interface.

Mechanical

To allow ease of interchangeability, the mechanical interfaces for a module need to be the same as those of the module it will replace. This interface would generally take the form of some type of fastener and associated footprint. The mechanical interface parameters that must be considered are shown in Table 2-10.

Parameter	Remarks
Size, shape, mass, mass properties	
Orientation	Including reference datum
Dimensional relationship between mating items	Including reference datum point(s)
Fastenings	e.g. bolt lengths, threads, sizes, materials, rivet types, nut types, inserts
Force/load transmission requirements	
Tolerances	
Accessibility	For maintenance and integration
Material properties	Including galvanic properties, corrosion, cold-welding
Mechanical properties	
Attachment	Including seals, locks
Handling hard points	
Location/alignment	
Integration considerations	Shock mitigation/limitation, torque requirements for fasteners

Table 2-10 Mechanical interface parameters

Thermal

Thermal design is probably easiest if each module is thermally isolated from the rest of the spacecraft as much as possible. Thermal design and control methods can then be applied on a per-module level. If necessary, the thermal paths between modules can then be tailored to specific requirements; each module being considered as a thermal "black box". The thermal interface parameters that must be considered are shown in Table 2-11.

Parameter	Remarks
Thermal characteristics	Emissivity, absorptivity
Surface finishes	Paints, tapes,
Attachment	Including seals, locks, thermal washers/interface materials. Conductivity, energy exchange.

Table 2-11 Thermal interface parameters

Power

Unless power is separately generated/stored in each module, there must be power lines between subsystem modules. The precise architecture of the power distribution will depend on the design of the spacecraft, but it may be assumed that each module would form a node on the power bus. Each node must be electrically the same for any of the interchangeable modules. This implies that any necessary voltage regulation or conversion from the bus voltage would take place within each module. The power interface parameters that must be considered are shown in Table 2-12.

Parameter	Remarks
Voltage	Should use standard voltage for COTS products (usually 28V DC)
Regulation	
Continuity	
Load current	Including demand variation, transients e.g. inrush current
Grounding	Philosophy used e.g. common signal ground, connected to chassis inside OBC; common power ground, connected to chassis in power distribution unit. Grounding loops can be a problem and may require grounds to be adjusted after spacecraft assembly – leave options open for ground connections e.g. spare pins in connectors.
Switching	
Fault protection	e.g. fuses, switch-out of short circuit
Cabling & connector characteristics	Including max. no. of connect/disconnects for flight connectors

Table 2-12 Power interface parameters

Data

The data interface between modules needs to be simplified as much as possible to better enable making it standardised. The modules should be effectively "transparent" to the onboard communications scheme; if one module is replaced by another, little or no modification to the system should be required. It should theoretically be possible to unplug one module, and plug in another, and it should be able to communicate. The data interfaces that must be considered are shown in Table 2-13.

Parameter	Remarks
Data definition	
Signal characteristics	Analogue or digital, levels, reference
Shielding & EMI prevention	
Grounding	
Load impedance	For impedance matching to minimise losses
Cable & connector characteristics	Including connector pinouts, wire gauges
Physical bus connection	e.g. transformer coupling, optical connection. Method used should prevent failure of one unit affecting operation of the bus.
Data format	Data rates, protocols, coding, timing, updating
Transfer characteristics	Transmission medium, waveform characteristics, losses
Layering	Separation of physical, coding, application layers etc
Circuit protection	
Signal sources & destinations	
Circuit logic characteristics	

Table 2-13 Data interface characteristics

Software

The software interface covers the interchange of information between two items or functional areas. The software interfaces that must be considered are shown in Table 2-14.

Parameter	**Remarks**
Data source & destination	
Data definition	
Event to be controlled	
Message definition	
Initiating condition	
Timing	
Communication characteristics	How the information is communicated between the software items/equipment
Error detection	Including correction and recovery
Priority interrupts	

Table 2-14 Software interface characteristics

2.5.2.2 Functional partitioning: positioning of the interfaces

To be most effective, a modular system should be partitioned such that the sub-units formed are largely single function. This means that individual functions can be upgraded as required, without making any unnecessary changes to subsystems whose performance is already suitable for the mission.

Identification of suitable positions for inter-module interfaces can be achieved by functional breakdown analysis of the spacecraft system. This analysis decomposes all the functions that take place on board into sub-functions, and identifies their inputs and outputs. The process can be continued to deeper and deeper levels, although, once lower levels are reached, the functional analysis becomes much more dependent on the particular hardware being used.

A functional breakdown analysis for a generic small satellite was performed, and is given in Appendix B. This takes each of the following top-level functions, and divides them into sub-functions:

- Attitude determination & control
- Orbit determination & control
- Data handling & onboard communications
- Spacecraft command & control
- Electrical power
- Thermal control
- Communications
- Ground segment

This analysis can be used to generate a series of functional units that have a single main function, and to show their internal and external relationships. This allows the possible positions of interface boundaries to be identified. The interface positions define the points at which the spacecraft would be divided into discrete modules, which would be interchangeable without significant impact on the remaining system. These are shown in the following series of diagrams, together with an analysis of the suitability of the different interface positions.

The function areas studied in this way are attitude control, orbit control, spacecraft command and control, and power. The thermal subsystem is often more a distributed rather than an identifiable single subsystem or cluster of subsystems. Similarly, structural modularity will be studied later, in the conceptual design phase.

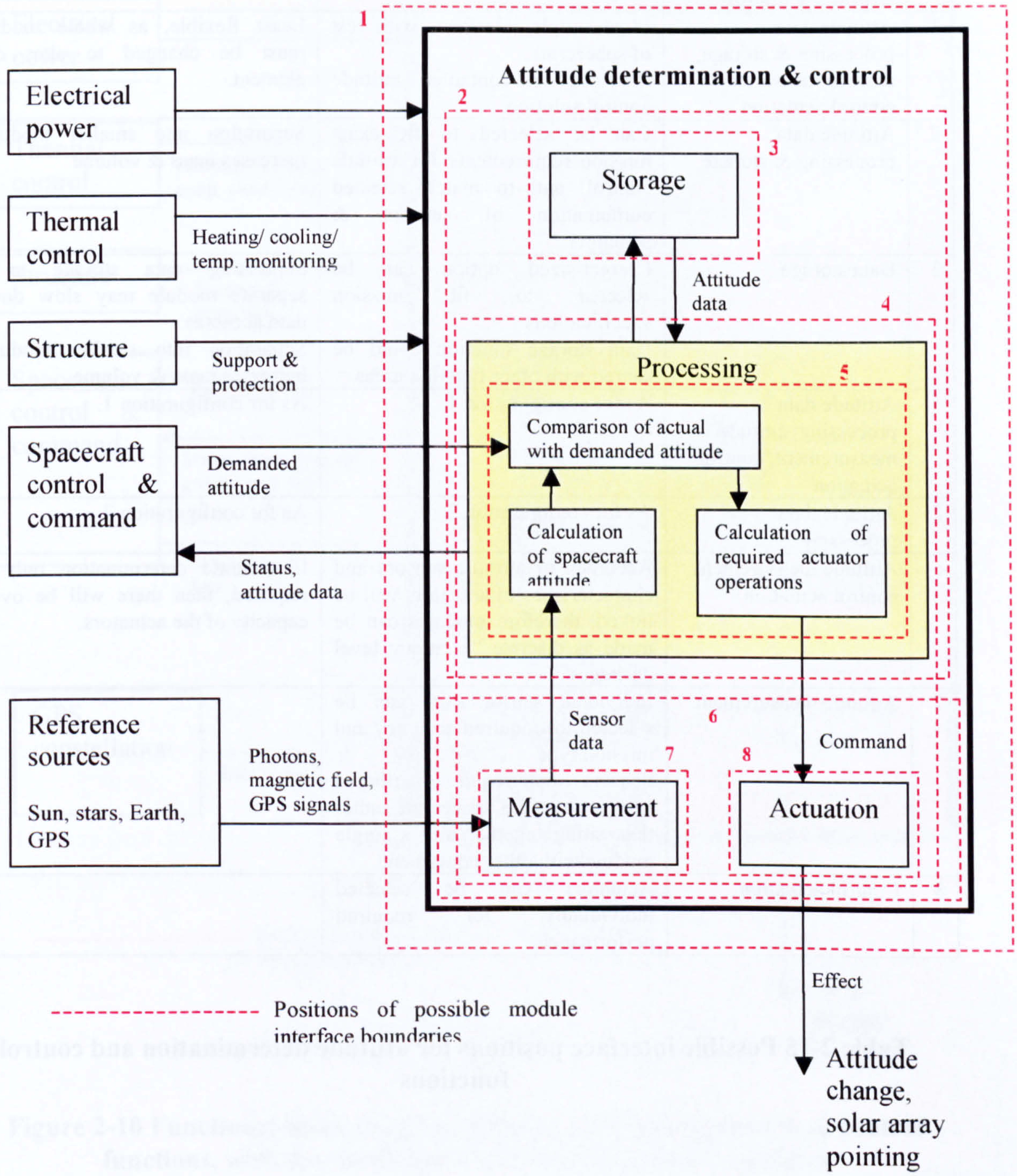

Figure 2-9 Functional block diagram showing attitude determination and control functions, with the associated internal and external interfaces.

Figure 2-9 shows the main functions and inter-relations associated with attitude determination and control. A number of potential module boundaries are indicated; the pros and cons of each of these configurations are shown in Table 2-15.

	Functions performed within the module	Advantages	Disadvantages
1	Attitude data processing & storage, attitude measurement, control actuation	Most simple interfaces with rest of spacecraft. Provides self-contained attitude control solution.	Least flexible, as whole module must be changed to alter one element.
2	Attitude data processing & storage	Can be selected to fit exact mission requirements for attitude control, and to match selected combination of sensors & actuators.	Separation into smaller modules increases mass & volume.
3	Data storage	Correct-sized option can be selected to fit mission specifications. Data storage function could be shared with other function areas.	Separating data storage to a separate module may slow down data accesses. Separation into smaller modules increases mass & volume.
4	Attitude data processing, attitude measurement, control actuation	As for configuration 1.	As for configuration 1.
5	Attitude data processing	As for configuration 2.	As for configuration 2.
6	Attitude measurement, control actuation	Accuracy of attitude sensors and characteristics of actuators will be linked, therefore modules can be made as discrete "accuracy level packages"	If accurate determination only is required, then there will be over-capacity of the actuators.
7	Attitude measurement	Individual sensor suite can be selected for required accuracy and mission type Sensors often require distributed sites around the spacecraft rather than siting together into a single module with other equipment	
8	Control actuation	Actuators can be selected individually for required performance.	

Table 2-15 Possible interface positions for attitude determination and control functions

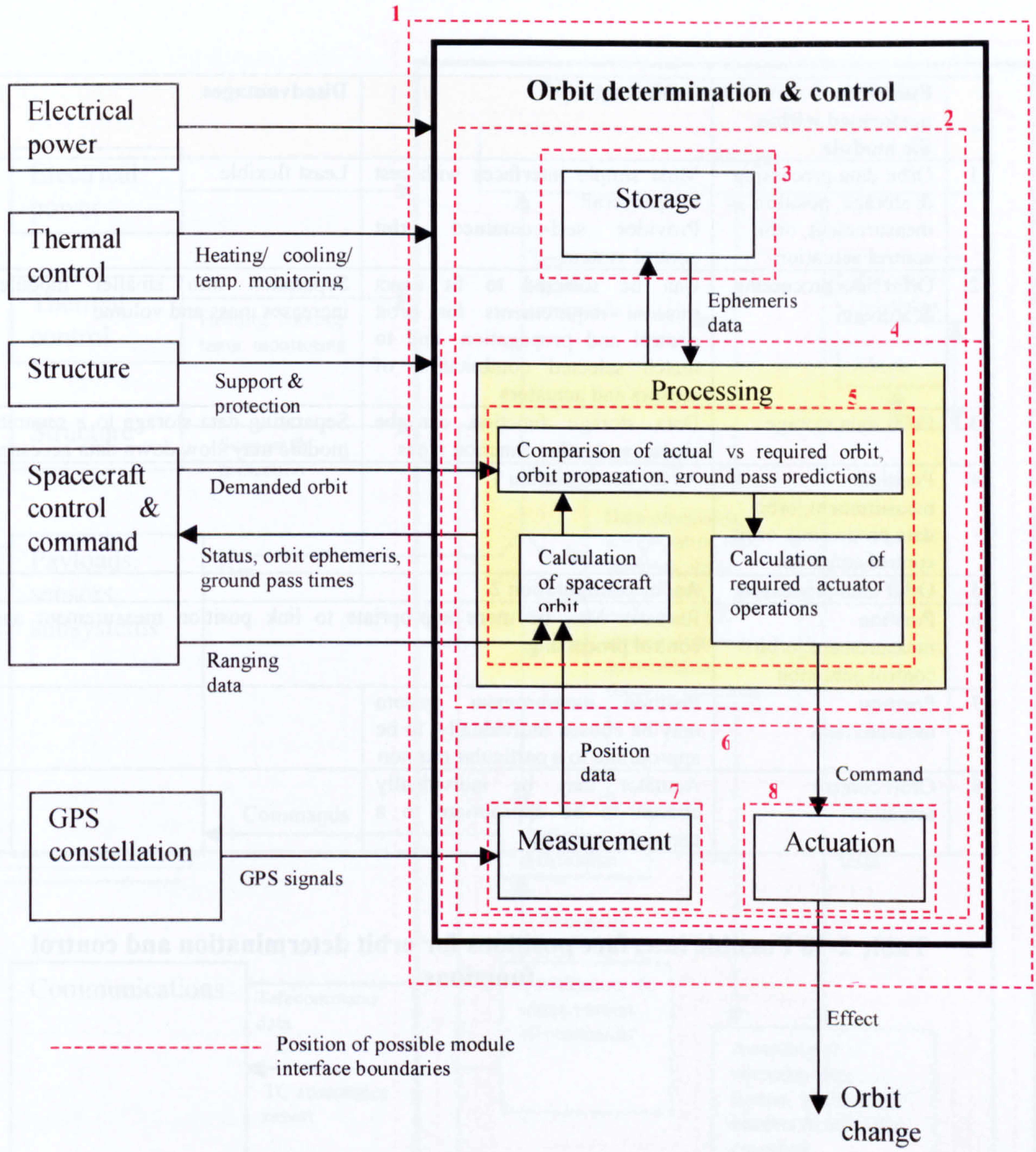

Figure 2-10 Functional block diagram showing orbit determination & control functions, with the associated external and internal interfaces

Figure 2-10 shows the main functions and inter-relations associated with orbit determination and control. It is quite similar to that for attitude control, but orbit determination is orbit data may often be provided from the ground, acquired by performing ranging procedures with the ground station antenna. Alternatively, an onboard GPS receiver may directly obtain position data. Potential positions for interfaces are indicated on the diagram. The advantages and disadvantages of each are discussed in Table 2-16.

	Functions performed within the module	Advantages	Disadvantages
1	Orbit data processing & storage, position measurement, orbit control actuation	Most simple interfaces with rest of spacecraft Provides self-contained orbit control system	Least flexible
2	Orbit data processing & storage	Can be selected to fit exact mission requirements for orbit control and propagation, and to match selected combination of sensors and actuators	Separation into smaller modules increases mass and volume
3	Orbit data storage	Data storage function can be shared with other function areas	Separating data storage to a separate module may slow down data accesses
4	Position measurement, orbit data processing, orbit control actuation	As for configuration 1	
5	Orbit data processing	As for configuration 2	
6	Position measurement, orbit control actuation	Remark: May be more appropriate to link position measurement and control processing.	
7	Position measurement	Position measurement system may be chosen individually, to be appropriate to a particular mission	
8	Orbit control actuation	Actuator may be individually chosen to be appropriate to a particular mission	

Table 2-16 Possible interface positions for orbit determination and control functions

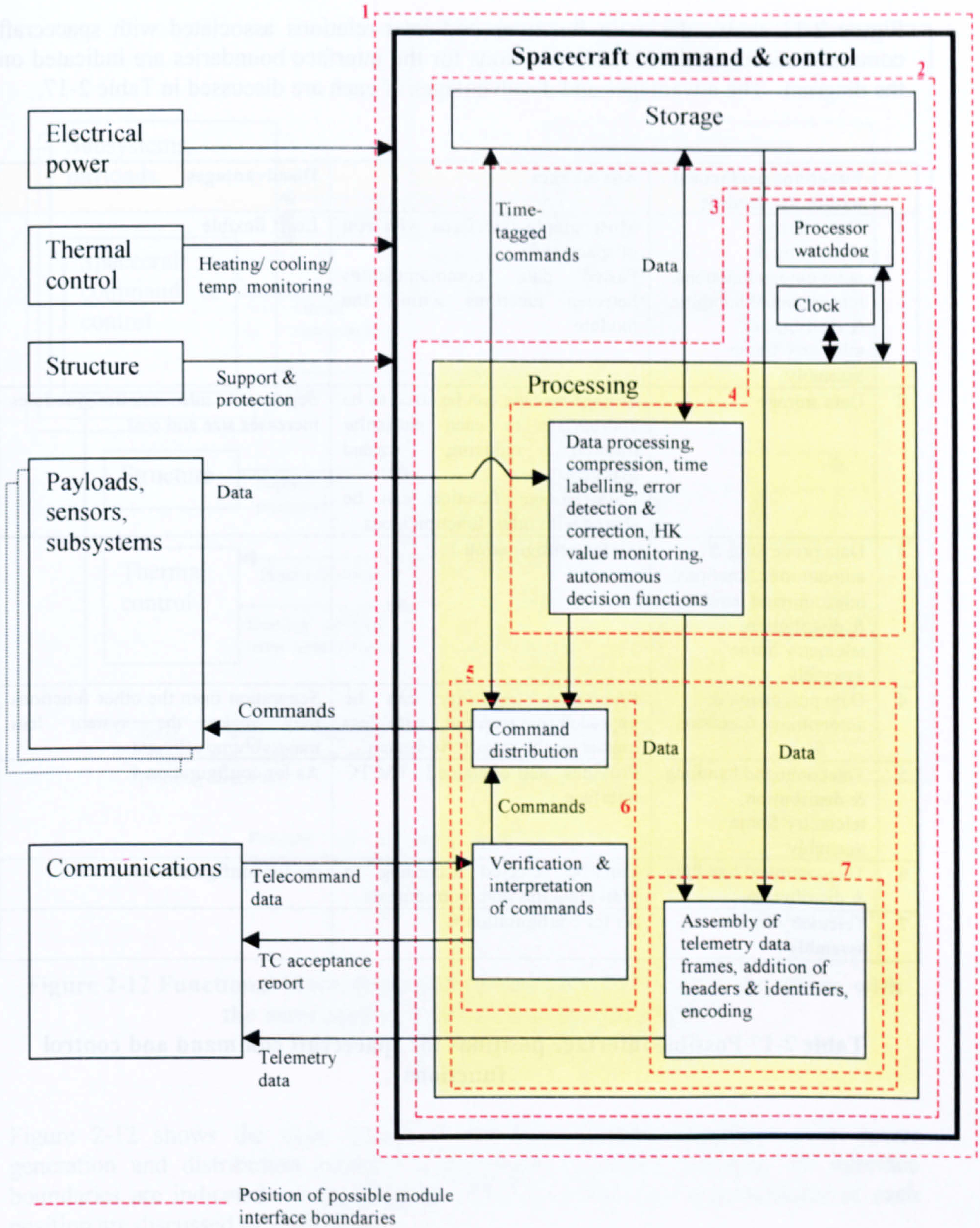

Figure 2-11 Functional block diagram showing spacecraft command & control functions, with the associated external and internal interfaces.

Figure 2-11 shows the main functions and inter-relations associated with spacecraft command and control. Possible positions for the interface boundaries are indicated on the diagram. The advantages and disadvantages of each are discussed in Table 2-17.

	Functions performed within the module	Advantages	Disadvantages
1	Data storage, processing & autonomous functions, telecommand handling & distribution, telemetry frame assembly	Most simple interfaces with rest of spacecraft Faster data communications between functions within the module	Least flexible
2	Data storage	Storage module can be sized to be appropriate to each particular mission, reducing wasted mass/cost. Data storage function can be shared with other function areas.	Separation into smaller modules increases size and cost.
3	Data processing & autonomous functions, telecommand handling & distribution, telemetry frame assembly	As for configuration 1.	
4	Data processing & autonomous functions	Processing capability can be upgraded as required, with less impact on the rest of the system	Separation from the other functions may make the system less mass/volume efficient
5	Telecommand handling & distribution, telemetry frame assembly	Provides self-contained TM/TC interface	As for configuration 4.
6	Telecommand handling & distribution	Can be selected according to individual mission requirements	As for configuration 4.
7	Telemetry frame assembly	As for configuration 6.	

Table 2-17 Possible interface positions for spacecraft command and control functions

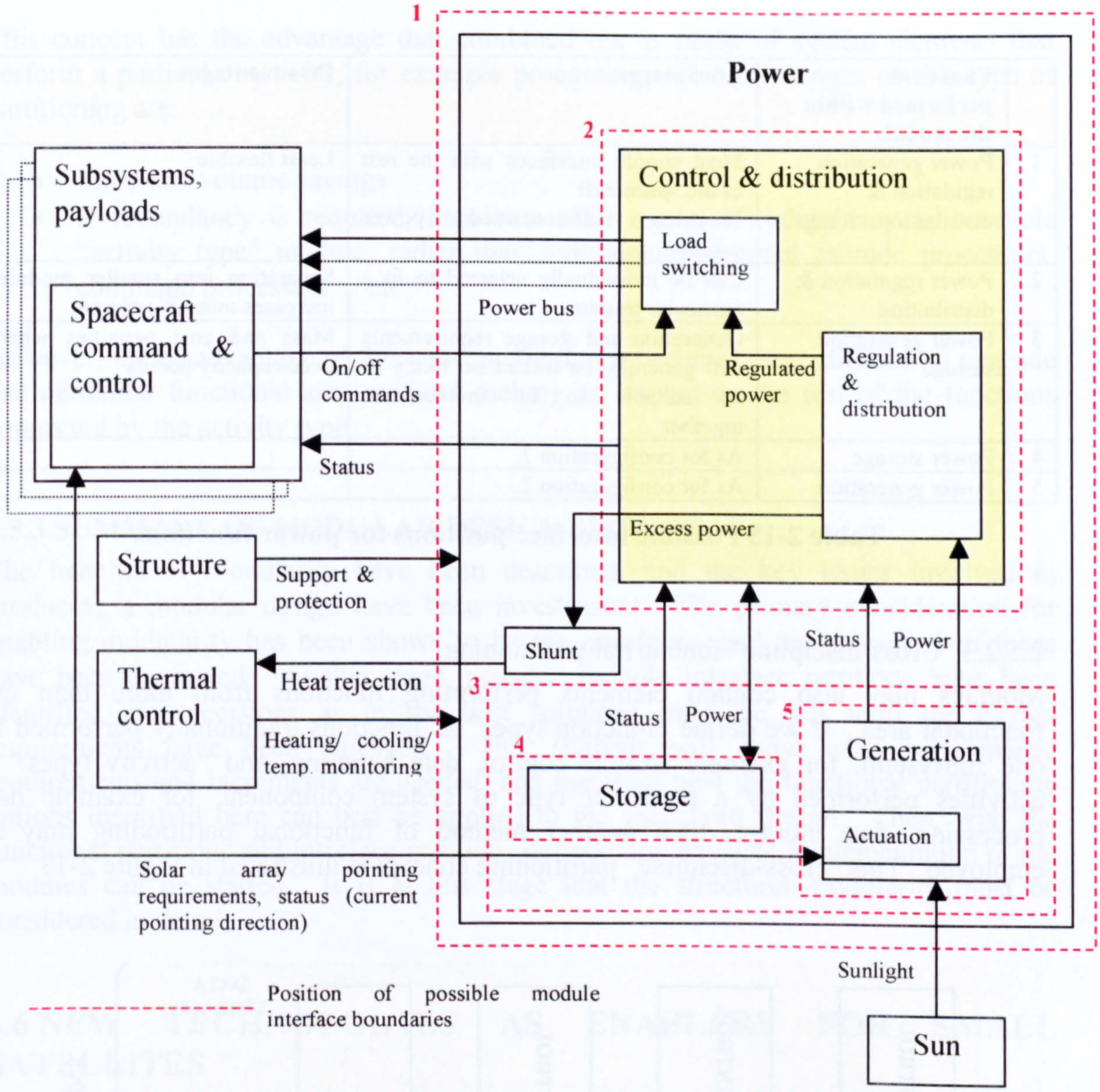

Figure 2-12 Functional block diagram showing spacecraft power functions, with the associated external & internal interfaces

Figure 2-12 shows the main functions and inter-relations associated with power generation and distribution onboard a spacecraft. Possible positions for interface boundaries are indicated on the diagram. The advantages and disadvantages of each position are discussed in Table 2-18.

	Functions performed within the module	Advantages	Disadvantages
1	Power generation, regulation & distribution, storage	Most simple interfaces with the rest of the spacecraft Provides self-contained power solution	Least flexible
2	Power regulation & distribution	Can be individually selected to fit a particular mission	Separation into smaller modules increases mass & volume
3	Power generation, storage	Generation and storage requirements will generally be linked, so like;y to be suitable to be incremented together	Mass and cost penalties where over-capacity occurs
4	Power storage	As for configuration 2.	
5	Power generation	As for configuration 2.	

Table 2-18 Possible interface positions for power functions

2.5.2.3 "Cross-discipline" functio nal partitioning

Modules may also contain elements performing functions from more than one functional area. If we define "function types" as functions traditionally performed by one subsystem, for example attitude control, data handling, and "activity types" as activities performed by a particular type of system component, for example data processing, data storage, then another method of functional partitioning may be employed. This "cross-discipline" partitioning concept is illustrated in Figure 2-13.

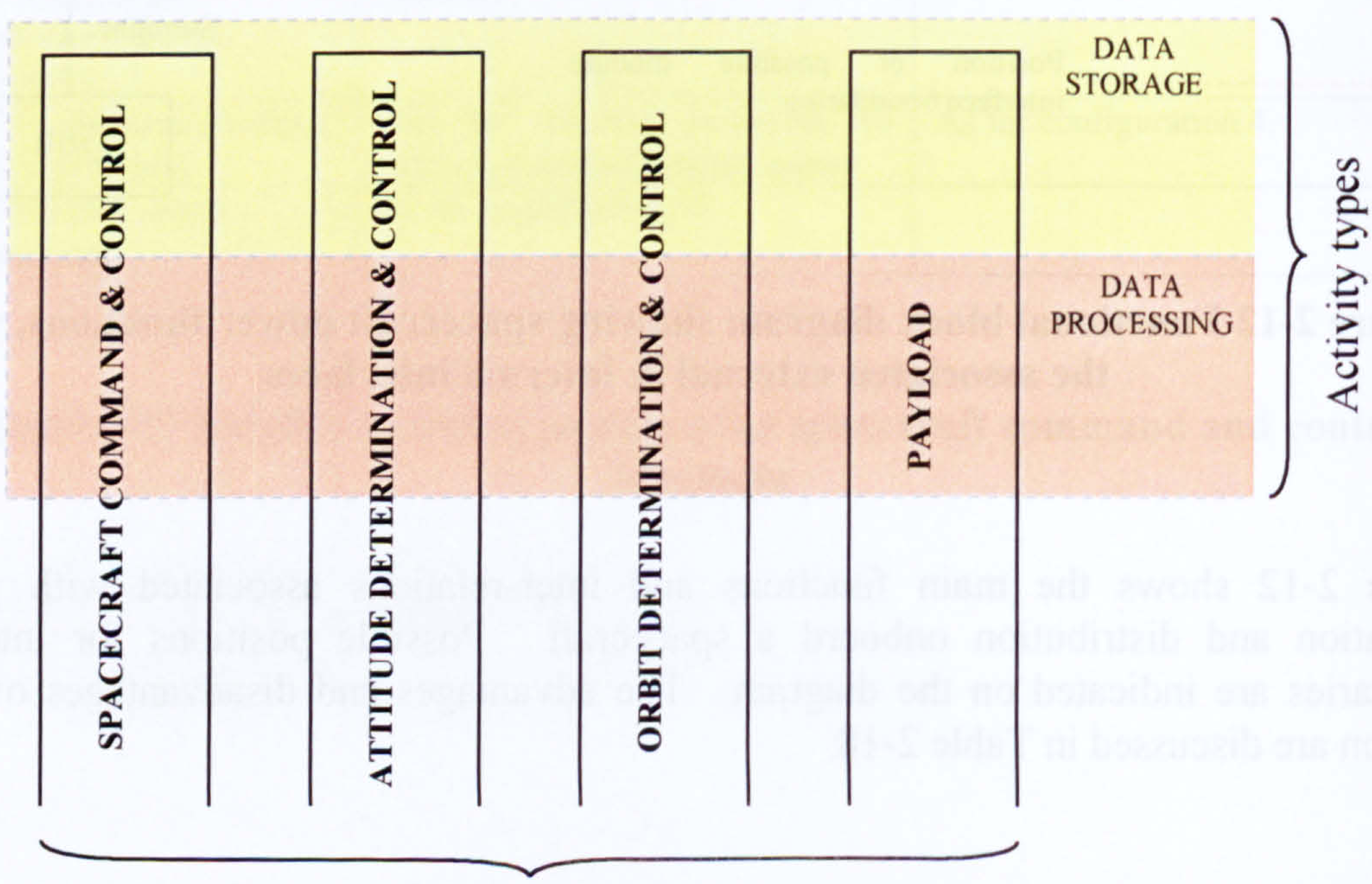

Figure 2-13 "Cross-discipline" partitioning concept

This concept has the advantage that combined use is made of system elements that perform a particular activity, for example processing. The advantages of this type of partitioning are:

- Mass and volume savings
- If redundancy is required, then two units can give redundancy to a whole "activity type" module, rather than individual redundant attitude processors, command processors... etc.

However, the disadvantage of this concept are that it becomes more difficult to upgrade one particular functional area, without making an impact on the rest of the functions connected by the activity type.

2.5.3 SUMMARY OF MODULAR DESIGN SECTION

The benefits of modularity have been described, and the key issues involved in producing a modular design have been investigated. The primary consideration for enabling modularity has been shown to be the interfaces, and the properties of these have been examined. At this stage, possible module interface positions have been identified, but decisions as to the best solutions must be left until the design requirements have been analysed. This analysis will show what performance requirements and increments are needed, and therefore how the functional partitioning options identified here can best be applied to the spacecraft design. Then, with the functional partitions and interface positions defined, the physical implementation of the modules can be started. It is at this stage that the structural modularity must be considered in detail.

2.6 NEW TECHNOLOGIES AS ENABLERS FOR SMALL SATELLITES

Some mission categories have remained outside the scope of small spacecraft, because the typical performance and/or characteristics of a small platform is inconsistent with the requirements of certain payloads. While this continues to be true, new technological developments can widen the scope of smaller platforms by:

- increasing their performance range
- miniaturising subsystems to allow more space to be dedicated to payloads
- miniaturising the payloads themselves to allow them to be flown on smaller platforms
- application to constellations for distributed sensing

Some technological developments may give immediate lower-cost alternatives to existing techniques. With other new technologies, there may be a cost increment involved in choosing them over the more traditional methods. This may arise due to the higher cost of materials and processes, or the need for manufacturers to offset their R&D costs by charging a premium for the technology. To make adopting such a new technology worthwhile, its costs must therefore be traded against its benefits, in terms of performance gains, missions enabled, or launch cost reductions made (via miniaturisation).

In some cases the cost increment may be temporary, as the technology may give benefits from repeatability or manufacturing aspects that only take effect once the programme is running. For example, new equipment may be required for different manufacturing or testing processes, and project personnel may need to be trained in the use of the new and unfamiliar components and techniques. After implementation, the cost savings are then gradually made over time. This type of longer-term benefit would make some technologies better suited to extended programmes producing multiple spacecraft (such as the programme being suggested here), rather than one-off small satellites.

New technologies may also be applicable for small spacecraft as this type of mission can be used as a demonstration testbed. Cutting-edge technologies that may otherwise be too expensive for small missions, may sometimes be obtained more cheaply so suppliers can demonstrate their prototypes in space. The manufacturer then gains the benefit of space qualification, and the small satellite project gains the benefit of useful new equipment.

Of course, there is a degree of risk involved in this approach, as the technology will be relatively untried in the application. But this is itself in keeping with the small satellite philosophy. One consideration with this method of gaining access to new technologies is that this approach may only occur once per 'technology item'. Once the technology is successfully demonstrated, the utility to the supplier of offering equipment more cheaply to a small spacecraft project has gone. The type of commercial, multipurpose platform being considered in this research is an on-going, repeatable design rather than a one-off. Therefore, for a commercial platform, this method may be more applicable to the payloads (which may make better use of a one-off technology opportunity) rather than the platform subsystems. To use a new technology on the platform itself, the benefits afforded have to outweigh the standard 'purchase costs' of using the technology.

The new technologies described in this section are either new developments generally, or are merely quite new to space use. Unfortunately, the very budget cuts that have led to the increased interest in small spacecraft, have also led to the termination of many technology programs that may otherwise have developed applicable technologies. However, the increased interest in itself provides a commercial driver to companies to develop small satellite technologies.

Those technologies investigated were all identified as being of potential applicability to small spacecraft; the more detailed examinations given here identify which are truly suitable for consideration in the later design stages of this work. The benefits afforded by each have been quantified wherever possible, in terms of mass/volume savings, performance gains, schedule reductions, or other advantages. Assessments of the maturities of the technologies have also been made, as well as an analysis of particular applications to which they are suited.

2.6.1 STRUCTURES & MECHANISMS

As already mentioned, volume may often be more of a constraint than mass for small spacecraft. Therefore, in terms of structures and mechanisms, key enablers for small missions are those concerned with post-separation deployment of equipment, or minimising the volume of the existing spacecraft equipment.

2.6.1.1 Inflatable structures

Inflatable structures in space are not a new idea; inflatable technologies were flown as long ago as 1960, on the Echo 1 mission. Since then, work has continued in this field, but at a relatively low level both of investment, and of awareness by the general space community.[21,27] However, the flight of the NASA Inflatable Antenna Experiment in 1996 (shown in Figure 2-14, after deployment from STS-77) stimulated greater interest, and a number of inflatable technology application concepts are now in development.

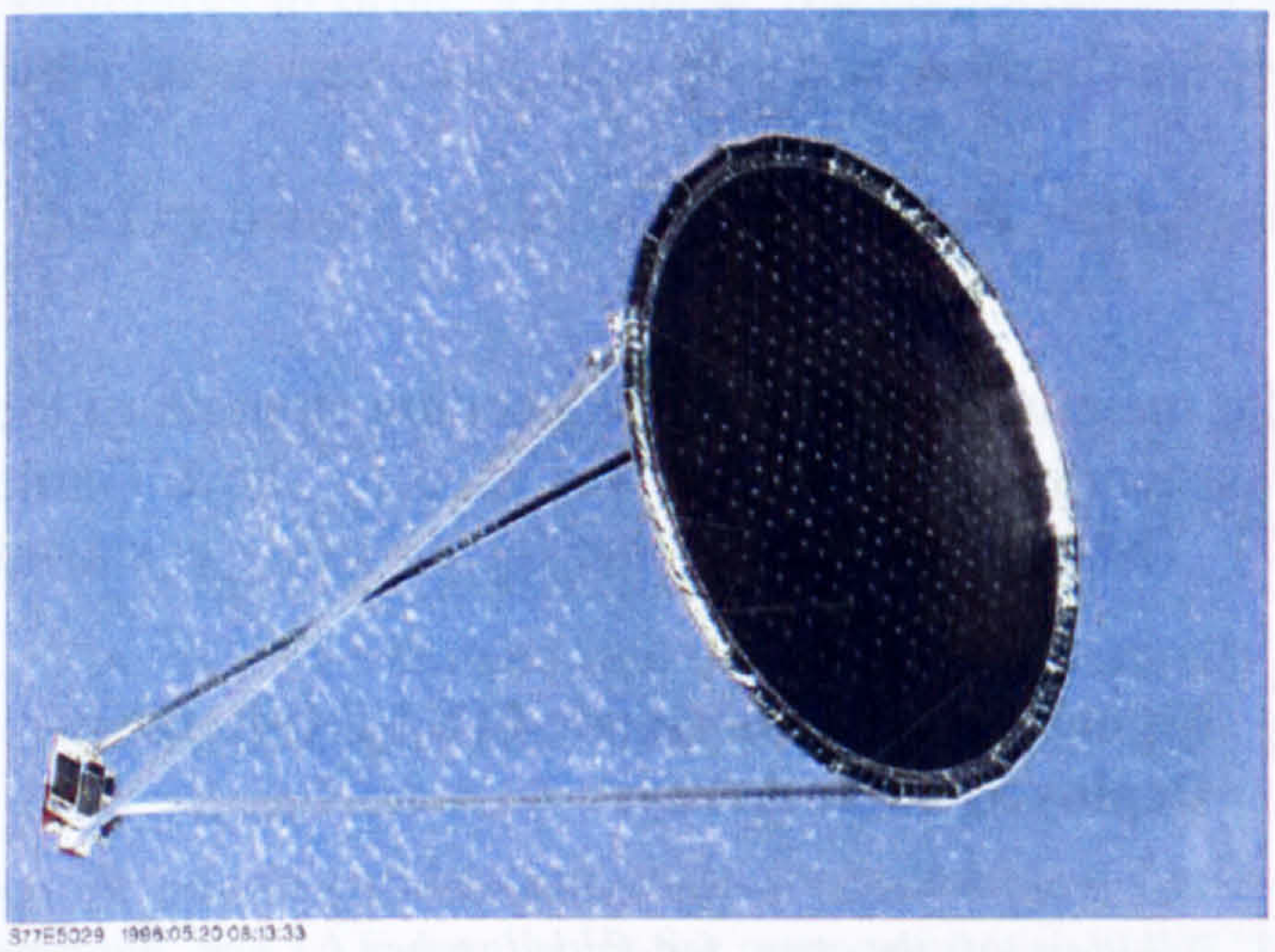

Figure 2-14 NASA Inflatable Antenna Experiment.
Launched in 1996 from the Shuttle, the IAE demonstrated an inflatable reflector supported by an inflated torus/strut assembly. The structure measured 14m x 28m when deployed, yet was stowed in a container the size of an office desk (less than 1.5m³.[27,60] (Image: L'Garde, Inc.)

Overview of the technology

The general principle of operation of inflatable structures is simple; a tubular membrane structure is compactly packaged and then, once in orbit, deployment is effected by pressure of an inflation gas. Various types of deployment scheme have been demonstrated. These are described below.

In the "roll-out" method, a tubular membrane is flattened and coiled; the gas pressure uncoils the tube and inflates it into a tubular strut. Deployment resistance may be provided by Velcro strips on the top and bottom of the tube, a wire brake system, or by constant force springs embedded in the walls of the tube. The roll-out method will be used by ILC Dover, Inc. for the sunshade on the Next Generation Space Telescope (NGST), due for launch in 2007. A 1/3 scale demonstration model is to be flown on the Shuttle in 2003.

In the "mandrel" or "columnation" method, the tubular membrane is drawn over a mandrel and stored behind it. On deployment, gas is introduced through the centre of the mandrel; the resulting pressure build-up pulls the stowed tube back up over the mandrel, which both keeps it at the correct deployment angle and applies deployment resistance through friction. This method has been developed for deployment of the

solar arrays on the Teledesic demonstration model, by ILC Dover, Inc.[28] Deployment of the Teledesic array is shown in Figure 2-15.

Figure 2-15 Inflatably-deployed solar array by ILC Dover, Inc.

The mandrel method of inflatable deployment is illustrated by this image of the Teledesic demonstration array. The central tube is being inflated over the mandrel, to maintain correct orientation. The other two tubes are pressurized after the central tube has fully deployed.

(Image: ILC Dover, Inc.)

In the "fan-folded" method, the tubular membrane is folded; the bending strength of the tube then provides the deployment resistance. This method was used for the L'Garde ITSAT solar array demonstration.[27] The deployment scheme for the array is shown in Figure 2-16.

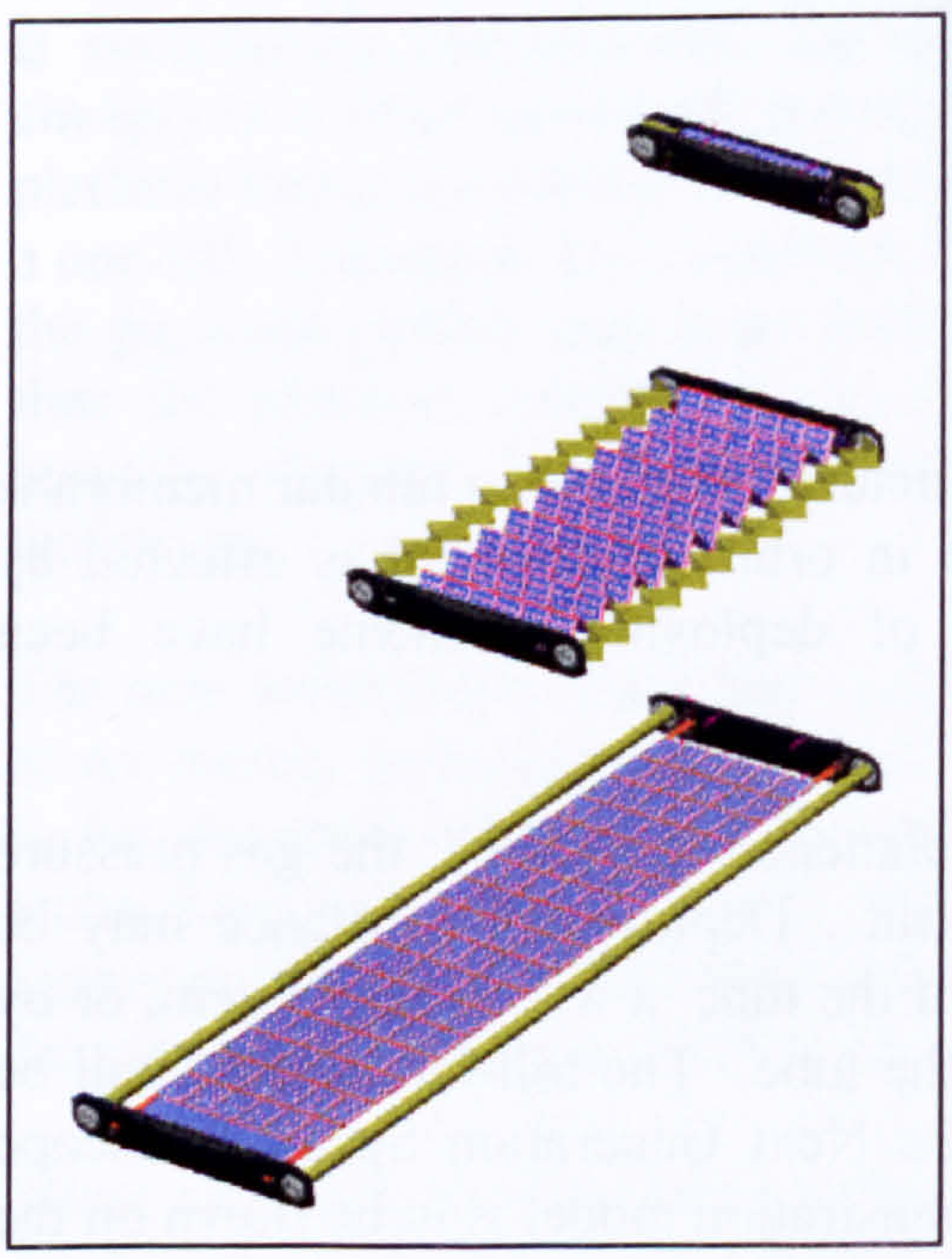

Figure 2-16 The L'Garde ITSAT Inflatably-Deployed Solar Array.

The thin-film solar array membrane is extended by the inflation of the two fan-folded tubes. The diagram illustrates the high packaging efficiency of the stowed structure, compared to the deployed size.
(Image: L'Garde, Inc.)

Shapes other than simple tubes can also be used. Membrane gores can be carefully designed so that once inflated, very precise geometries can be attained.

After deployment, there are several means by which the structure may be maintained in its nominal configuration. Inflatable structures may be divided into three main types. In purely inflated systems, the deployed shape of the structure is maintained by internal pressure of the inflating gas. This type is prone to damage by micrometeoroids and degradation of the membrane, and requires provision of top-up gas to counter leaks.

Mechanically rigidized systems flatten the required shapes for stowage, and then 'pop' them back into shape by an inflation pulse followed by an overpressure pulse to strain the wall into the correct configuration. This type has the disadvantage of a lower strength:weight ratio. However, mechanically-rigidized cylindrical tubes can take high compressive loads, and this method is well developed. It may also be used in low-temperature environments.

With the chemical rigidization method, the material used is pliable in Earth conditions, and becomes rigid once in space due to one of the following: UV radiation exposure, water loss, heating/cooling, or reaction with the inflation gas. This type of inflatable has the highest strength:weight ratio. Some types can also be tested, as they are reversibly rigid.

Benefits of the technology
Inflatable structures are of great interest as lower-cost, lower-mass alternatives to the traditional mechanical deployment techniques used in space. They have therefore been proposed as solutions to the problem of deploying very large (of order 50m or more) structures in orbit. Of course, the characteristics that make inflatables suitable for such very large spacecraft, also give rise to benefits for smaller vehicles.

The key benefits of inflatable structures, with particular respect to small spacecraft, are:
[11]
[21]

- Typically a 50% mass reduction over a comparable mechanical system
- Volume (when stowed) reduced to between 25% and 10% of a comparable mechanical structure
- Can be stowed into a volume of practically any shape
- Structures are inherently strong, as loads are distributed over the large surface area
- Low production costs (reduced by a factor of ~10 for a large antenna structure: the IAE was built for approximately US$1M)
- High deployment reliability (and deployment of smaller systems may be demonstrated more easily on the ground)

Potential applications for small spacecraft
Inflatable structures have potential application to small spacecraft both in enhancing the performance of the bus, and in enabling payloads to be made small enough to fit onto smaller platforms. Applications of particular interest are described as follows:

Inflatably-deployed solar arrays
The smaller stowed volumes for a given deployed area would allow greater power generation even on small spacecraft. The various options for exploiting this technique, and the performance that may be achieved, are described further in section 2.7.2 (Power technologies).

Parabolic RF antennas
Parabolic antennas are often problematic for small spacecraft due to their size and mass. Inflatable antennas may enable small missions to fly higher-gain antennas, enabling much increased data rates to ground. Inflatable parabolic antenna reflectors may also be used in payloads to enable small radio astronomy missions. The practicality of this type of application will largely depend on the surface precision that can be achieved.

Inflatably-deployed SAR antennas

Small spacecraft have previously been unsuited to SAR missions, due to the necessary size and power of SAR antenna arrays. Lightweight, inflatably-deployed SAR antennas are under development by both L'Garde, Inc., and ILC Dover. The L'Garde prototype has a mass of about 1.2kg/m2. The SAR array takes the form of a membrane stretched between inflatable deployment tubes. It is shown in its deployed configuration in Figure 2-17.

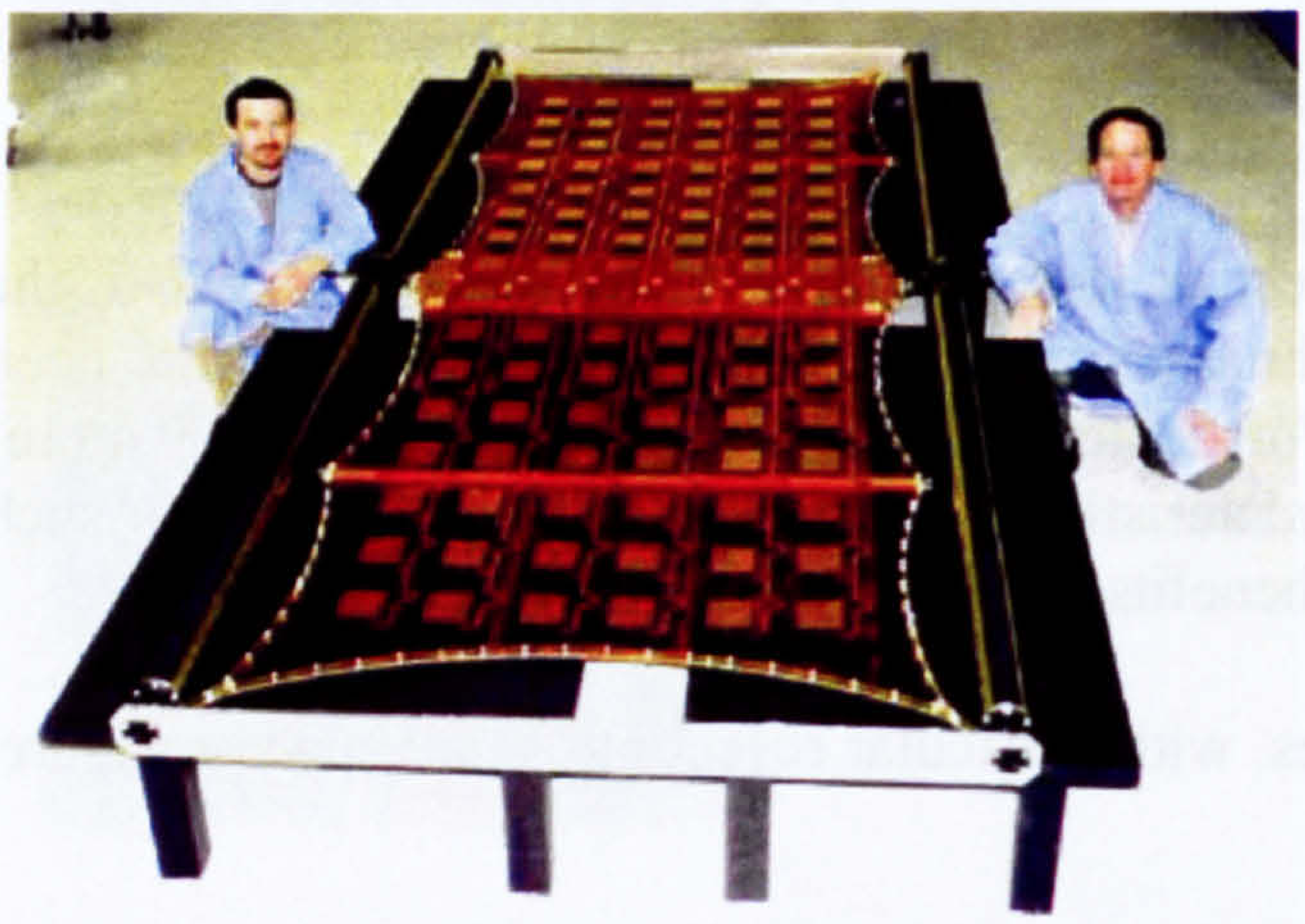

Figure 2-17 Inflatably-deployed SAR antenna.

The tubes at either side are deployed by inflation, stretching the central membrane. A conceptual design for a flight unit has been performed by L'Garde, Inc. (Image: L'Garde, Inc.)

Inflatably-deployed RF waveguides

This uses the same technique for deployment as the deployable SAR antenna described above. The mass is expected to be slightly more than for the SAR, at around 1.6kg/m2, but the stowed volume is slightly less, and the design would give greater sensitivity and allow greater power for radar applications.

Booms

These may be achieved either by simply the extension of a single inflatable tube, or by the deployment of an inflatable truss arrangement. Mechanically-deployable booms of various designs have been used on many small spacecraft, and the enhanced packaging efficiency of the inflatable option would offer great benefits.

Sunshades

Certain instruments and parts of spacecraft may require shading from the sun, for thermal control reasons and also to avoid stray light impinging on sensitive optics. Again, this is an application where a rather large deployed area of structure is required, making inflatables eminently suitable. As mentioned earlier in this section, an inflatably-deployed sunshade is being developed for flight on the NGST mission.

Drawbacks and issues requiring consideration

There are certain issues with inflatables which are potential causes for concern, and which may require further development:

- Purely inflatable structures require make-up inflatant gas to compensate for that lost due to the permeability of the membranes
- Lifetime of the membrane may be limited, due to attack by atomic oxygen and micrometeoroid impact

- Deployment sequence of large structures may become uncontrolled, leading to entanglement (deployment of the IAE went off nominal, and looked at risk of just such an entanglement)
- Venting of inflation gas may cause uncontrolled rotations and/or translations
- Off nominal deployment may cause uncontrolled reaction loads
- May be difficult to achieve extremely high precision (e.g. for reflectors)
- Cannot use traditional FE methods to analyse structures

Summary of technology maturity
Space inflatable technology status is shown in Table 2-19 below.

Application	Status	Maturity
Inflably-deployed solar arrays	Successfully demonstrated in space	High
Parabolic RF antennas	Demonstrated in space, partially successful – reflector did not inflate.	Medium – low (reflector geometry still has outstanding issues)
Inflatably-deployed SAR antennas	Under development, prototypes tested on ground.	Medium
Inflatably-deployed RF waveguides	Under development, prototypes tested on ground.	Medium
Booms	Inflatable booms have been used as a means of deployment for solar arrays, in ground tests and in space.	High
Sunshades	Under development.	Medium

Table 2-19 Summary of space inflatable technology status

Assessment of technology applicability
The growth in interest sparked by the IAE project has resulted in extensive prototyping and testing of inflatable space structures. The technology has been demonstrated in space, and the behaviour of this type of product is becoming better understood, clearing the way for increasing exploitation of the technique for a range of space applications.

This technology is certainly becoming mature enough for use on small demonstration missions, and should be considered for incorporation into future spacecraft and payload designs. Its property of being a self-contained unit that can be packaged into a convenient size and shape makes it particularly applicable to the modular concept. The additional fact that, once the gores are designed, the actual production costs are relatively low, makes it suited to the repeat-manufacture that is characteristic of commercial multipurpose platforms.

The potential drawbacks identified earlier in this section are minimised for the simpler types of structure – that is, tubular struts used as a "backbone" to deploy and support a 2-D structure. These may be best if they are rigidized and the inflation gas controllably vented. Due to the venting, this technology is only really suited to spacecraft with 3-axis attitude control, unless carefully-positioned opposed vents are used.

Another consideration would need to be the implication of only partial deployment. In the case of the solar arrays that are deployed by unrolling inflatable tubes, it may be envisaged that if the deployment halted early, some use may still be made of the deployed section of solar array. Contingency for this type of scenario could be designed in, in terms of layout of the cell strings, so that a partially-deployed array would still be usable. A similar argument would apply to sunshades; however, it would be important to know if a sunshade was incompletely deployed, as the platform pointing may then need to be modified so as to maintain the required instruments in shadow. For inflatable antenna reflectors, it would be expected that incomplete inflation would result in the antenna being unusable.

For the reasons outlined above, it is considered that the most appropriate application of inflatable structures to a commercial small satellite platform would initially be in solar array deployment and sunshades, followed by SAR arrays and inflatable reflectors when the technology has matured a little further.

2.6.1.2 Multifunctional structures (MFS)

Structure is obviously an essential element of any spacecraft, but it makes up around 20% of the mass of a small spacecraft. The aim of multifunctional structure is to enable other functions to share some of this mass (and volume), and allow redundant subsystem-specific mass to be replaced by the existing structural elements on the spacecraft. Suitable candidate areas for this are electronics and power/data harness. The rationale for their applicability is as follows:

Electronics

In general, electronic components are currently mounted onto panel-type structures within electronics boxes. If they could instead be mounted directly onto the spacecraft structure itself, then valuable mass and volume savings could be made. There is then no longer a need for separate electronics housings, which take up a considerable amount of space on board, and employ a large amount of redundant structural mass. This factor is currently limiting the mass savings that can be obtained by miniaturisation of electronics components. Direct mounting of components onto the spacecraft allows miniaturisation to give immediate and direct beneficial effects on overall spacecraft mass and size.

Power/data harness

Much of the mass of the power and data harness on spacecraft is due to casings, supports and connectors rather than the physical wires that actually carry the power or signals. As with the electronics, this element of the spacecraft is carrying a large amount of redundant structural mass. In addition, the large size of the cables and cable bundles means large bend radii are needed, further increasing the volume taken up by harness. Integrating the harness with the structure allows the structural function to be made common across both areas, and the volume used to be greatly reduced. It is estimated that shifting from conventional cabling to MFS for power and data distribution will allow a harness mass reduction of 75% coupled with an increase in available spacecraft internal volume of 40%.[50] The structure of the spacecraft is also inherently suitable for combination with the harness, as all the onboard subsystems, which will require connection to the harness, will already have an existing connection with the structure.

The NASA New Millennium Program investigated and developed this technique for demonstration on the 1998 Deep Space 1 mission. The technology used was developed by the USAF Phillips Laboratory and Lockheed Martin, and employed thin multi-layer copper/polyimide circuit patches bonded to the composite structural panel to form the power and data transmission medium. These circuit patches replaced the electrical interconnect circuits used on conventional motherboards, and flex jumpers interconnected the patches on separate panels. Conventional connectors were not used. Multi-chip modules were attached via specially developed sockets, in interface patches on the panel. The whole panel was then protected and shielded by a thin, formed cover. An example of MFS panels is shown in Figure 2-18.

Thermal control was achieved via the use of heat-transfer devices embedded in the composite panel, the outside surface of which then acted as a radiator. These devices included miniature heat pipes and high-thermal-conductivity straps.[6]

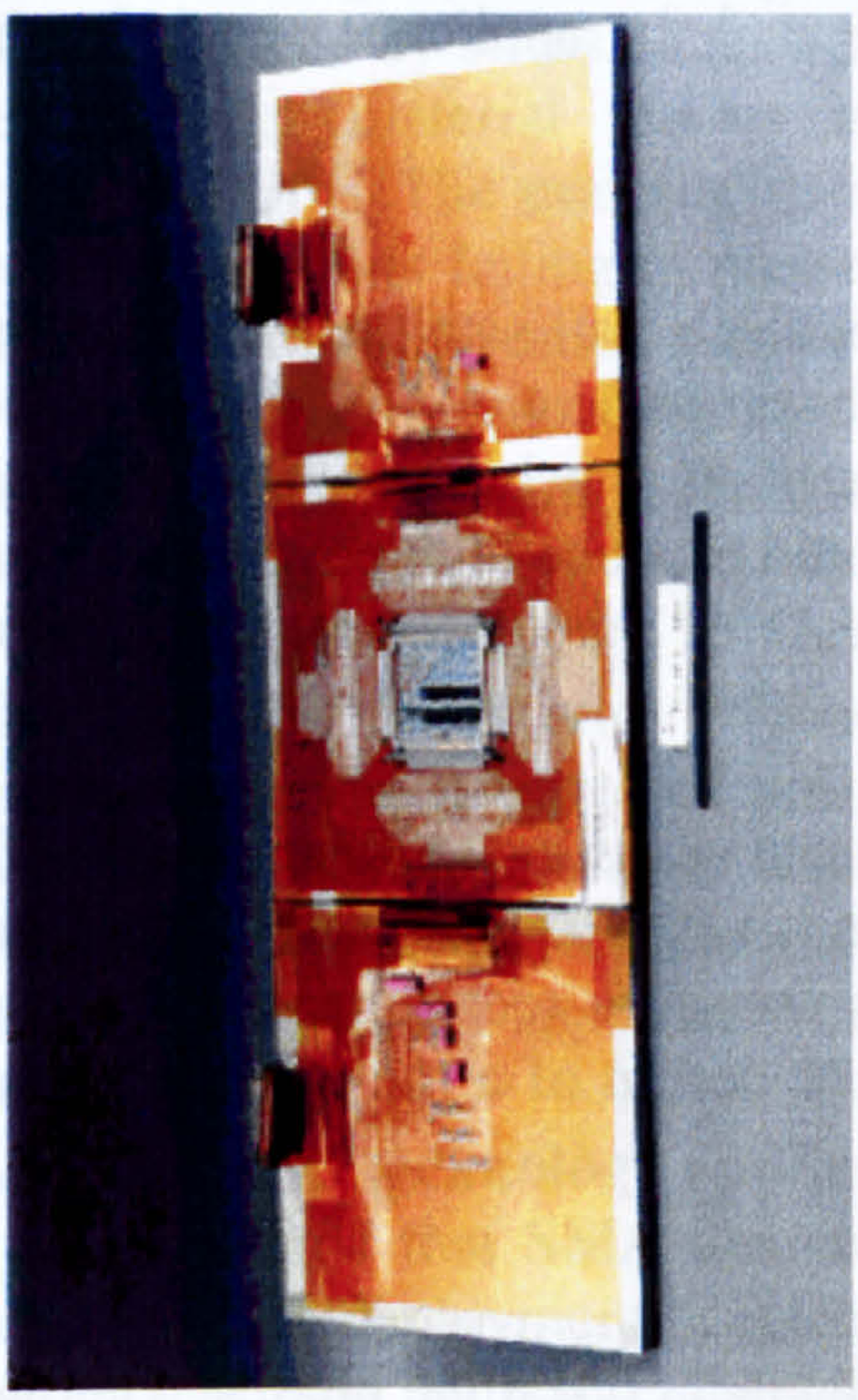

Figure 2-18 Multifunctional Structure Panels.

The circuit patches with multichip modules and surface-mounted parts can be seen, attached to the three panels. The patches are connected between panels by copper polyimide jumper flex strips.

The USAF predicts a twofold reduction in manufacturing costs, and a tenfold reduction in mass and volume, via the use of this technique.[59]
[Image: USAF Phillips Laboratory]

Another idea is to embed electrical harness within composite panels, and then incorporate connectors at the appropriate points where equipment is to be mounted. At its most sophisticated, this could lead to components being "plugged in", both mechanically and electrically, to the spacecraft structure, rather as electronics components are integrated into printed circuit boards. For data connectivity, optical fibres could be laid up within GFRP composites, and provide structural reinforcement as well as allowing on board data communications. This method could utilise many redundant paths, and connectors would only be required at joints between panels; greatly simplifying the data harnessing process and minimising the volume lost to cable routing.

Drawbacks & issues requiring consideration

- Effect of thermal cycling/ differential thermal expansion on patches and components?

- Radiation effects – less shielding than conventional approach?
- More re-working required if wire/connection broken?

Technology maturity
The MFS demonstrator experiment flown on the Deep Space 1 mission performed nominally, and the following aspects of this technology are now considered validated [6]:

- Flex-circuit patches and jumpers mounted directly onto structural panels, for use as electrical interconnects
- Multichip module sockets
- Interconnection of remote sensors via flex-circuits
- Shielding from EMI, radiation, and physical damage, via the protective cover

The MFS concept is currently being further developed by Lockheed Martin, who declare themselves open to collaboration on future projects. In addition, flex-circuitry has also been employed on MightySat-II, STRV-IIc & d and the Advanced Technology Demonstration Satellite, and it is due to be used on the future NMP Space Technology 5 (ST5) mission. Flex-circuitry is also being investigated for application to inflatable structures.

Assessment of technology applicability
Multifunctional structures are ideally suited to a modular small satellite. The volume and mass savings are of immense benefit, and the simplifications in the assembly process are great enablers for fast-track projects. The main drawback is the high initial "cost of adoption". However, the techniques used would produce high payoffs later in a programme, when the repeatability of designs and ease of manufacture and assembly would dominate.

It is considered that this is a key technology area that can act as a significant enabler for modular spacecraft concepts, allow valuable reductions in system mass and volume, and give benefits in simplifying the AIT process. It is therefore recommended that this technology should be considered for inclusion into the design of a multipurpose small satellite.

2.6.1.3 SMA actuators

Shape memory alloys are materials that exist in two distinct crystalline phases. Below a certain phase transformation temperature, the material is Martensitic (possessing a low yield-strength structure) and may be deformed easily into specific shapes. The deformations are retained until the material is heated to above its phase transformation temperature. At this temperature, the material reverts to an Austenitic structure and, in doing so, returns to its original shape. The shape change can generate a large force, which may then be used as a basis for very reliable, repeatable mechanisms. Shape memory materials are now being utilised in shock-free, non-explosive actuators and release mechanisms for space use.[57]

Advantages of shape memory alloy mechanisms over traditional explosive devices are:

- Repeatability – the device may be tested prior to flight, whereas explosive devices are "one-shot" devices
- Shock-free action – no explosive shock loads, which on small spacecraft may not be damped out before reaching sensitive equipment
- Safe to handle
- Lower activation power required
- Fast activation time
- Redundancy may be easily introduced into the device
- Can also be used for rotary actuation

Pin-puller devices can produce forces of over 20N for a mass of only 15g, and power of less than 1 Watt. (Figures based on those for the TiNi Aerospace pin-puller flown on the Mars Global Surveyor spacecraft).

Another useful property of SMAs is superelasticity, (also known as pseudoelasticity). This occurs slightly above the phase transition temperature, where the Martensitic phase can be induced by stress. Large strain for constant stress occurs, but when the stress is relieved, the material returns to its Austenitic phase and the original shape is recovered.[30] The material therefore behaves in an extremely elastic manner, with even large strains failing to produce permanent deformations. This superelasticity has found several applications; it is perhaps currently best known for "indestructible" frames for glasses. It may also be applied to space use, in deployables, where its resistance to severe bending would be extremely useful.

Technology maturity

The nickel-titanium alloy Nitinol has been successfully demonstrated for space applications, and a range of pin-pullers, bolt cutters, rotary actuators and valves are available or in development. Such devices have been flown on a number of missions, including HESSI, Clementine and Rosetta. This is a fairly mature technology, but one for which increasing applications are being found.

Assessment of technology applicability

It is considered that SMA devices may be extremely applicable to the small satellite design, where deployment or release actuators are required. The small size and mass of the devices, and their low power requirement would make them ideal. There are also a number of manufacturers of such devices who appear willing to develop custom designs for specific programmes.

2.6.1.4 Summary

Technology/ technique	Advantages	Disadvantages	Would enable:	Maturity
Inflatable structures	>50% mass reductions 10-25% the volume of a mechanical structure Large deployed structures possible	Micrometeoroids may cause problems if membrane not rigidified Uncontrolled deployment dynamics to be avoided	Missions requiring large structures or booms, e.g. SAR, magnetometry	Demonstrated in orbit (Inflatably-deployed array successful, inflatable antenna partially successful)
Multifunctional structures, e.g. wiring/ heat pipes/ electronics/ sensors integrated into structural members	Final integration of spacecraft is simplified – subsystems simply "plug in" >40% more volume may be dedicated to payload 75% lower harness costs Can halve manufacturing costs	Complexity of manufacture Difficult to repair a faulty embedded component	Volume-critical applications	Thin-film electronics mounted on structural panels have been demonstrated on NASA New Millennium program DS1 mission
Shape-memory alloy actuators	Easily testable & repeatable No pyro-shocks Simple devices Safer than pyros		Reliable solar array deployment Satellite separation systems	Available, more under development Used on HESSI, Clementine, Rosetta

2.6.2 POWER

2.6.2.1 Inflatably-deployed/inflatable thin-film solar arrays

Inflatables have already been discussed in some detail in the structures section (2.7.1). The use of inflatable technology for deploying solar arrays has been greatly enabled by the development of flexible, thin-film solar cells.[29] Amorphous silicon thin-film cells have been in production for use in terrestrial applications for some years, and copper-indium-gallium-di-selenide cells are currently in development. Both these cell types currently have efficiencies of 5-9%.[14] These efficiencies do not compare well with the commonly-used crystalline cells (Si:14.8%, GaAs:18.5%) and new multi-junction cells (22%).[64]

However, the benefits of the thin-film cells, as with inflatables, is their packaging efficiency and low mass. This means that, using a combination of thin-film cells and inflatable deployment, high specific power may be achieved even with the relatively low efficiencies of the cells.

This was demonstrated by the ITSAT (Inflatable Torus Solar Array Technology) array developed in 1993, for DARPA and the US Air Force Phillips Lab, by L'Garde, Inc. The ITSAT configuration and deployment scheme were illustrated in section 2.7.1. It supplied 274W for a weight of only 2.94kg, giving a specific power of over 90W/kg. [12] (Compare the current mechanical state of the art at 44W/kg for the concentrator arrays flown on Deep Space 1.[3])

The inflatably-deployed array in development for the New Millennium Program ST4 comet lander mission, has a specific power of 102W/kg, and is being further developed for NASA in the hope it may reach 250W/Kg. In terms of specific cost, the Teledesic array mentioned in section 2.7.1 had a cost target of $100/W.

Another way of utilising thin-film and inflatable technologies for space power generation has been investigated by ILC. In this scheme, the solar array is itself inflatable, and forms a spherical "balloon" with the outside surface covered with solar cells. The ILC Dover Powersphere™ gives 5W for 600g mass, and is shown in Figure 2-19. It is intended for use on microsatellites, and requires no pointing; its spherical configuration makes it omnidirectional. A proof-of-concept for this design has been produced. [12]

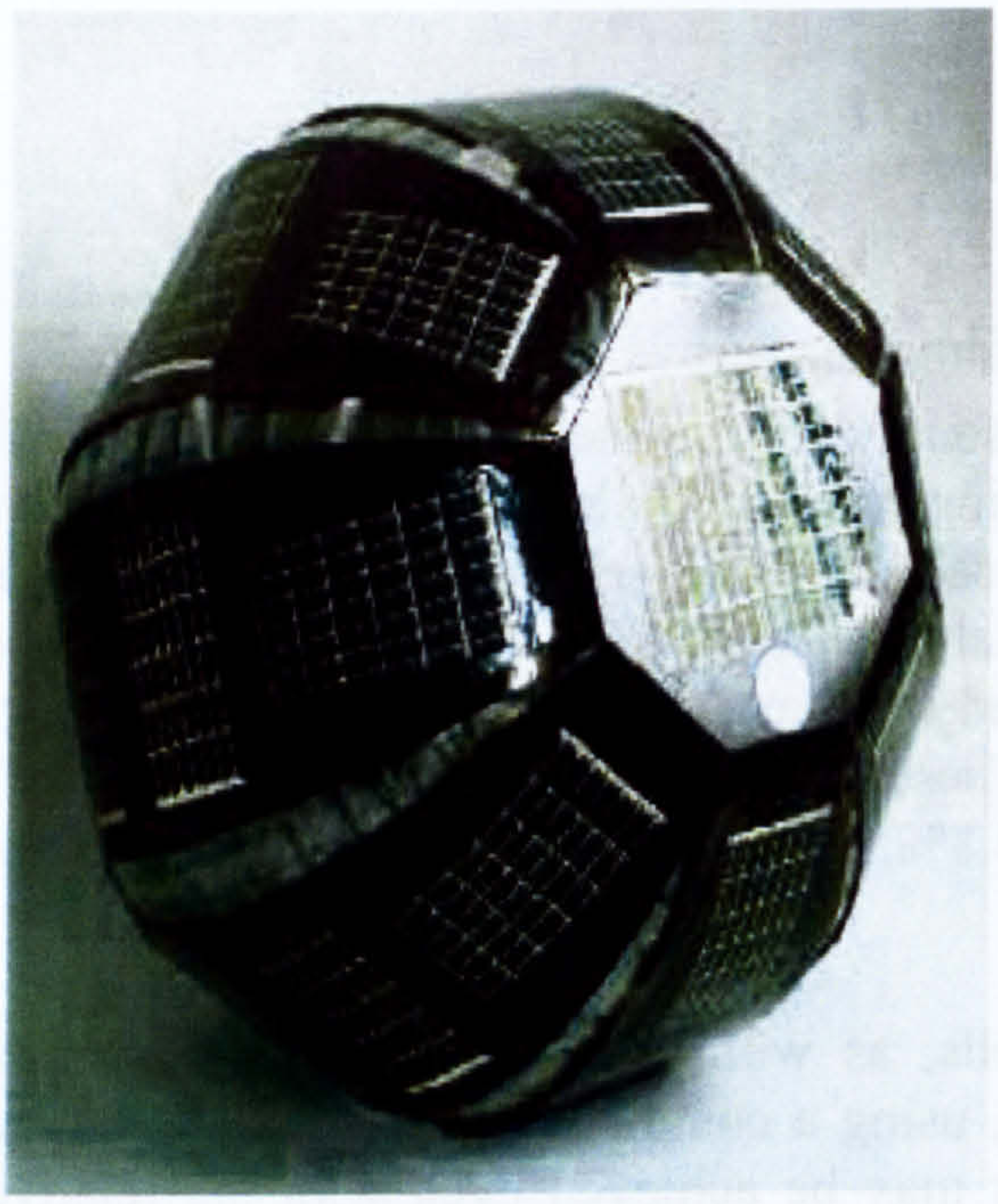

Figure 2-19 The ILC Dover inflatable Powersphere™ solar array

The array design was developed in 1998, and a proof-of-concept unit has been produced. [Image: ILC Dover]

Thin film solar arrays are also suited to deployment by more conventional mechanical means. The Hubble Space Telescope utilises blanket arrays that were deployed by means of a rolled bi-stable tape-type extendible boom. The tape was forced flat whilst rolled, but once deployed formed a rigid cylindrical configuration. Similar deployment concepts could be used for small spacecraft, to benefit from the low packing volume of thin-film arrays.

Key considerations for thin-film solar arrays are:

- Thermal control – emissivity of substrate polymer (this decreases with polymer thickness)
- Solar cell interconnect method

Assessment of technology applicability
The high packaging efficiency of inflatably-deployed thin-film arrays makes them extremely applicable to small spacecraft. They are also inherently modular in that they are effectively de-coupled from the spacecraft due to their deployment. A solar array module may be replaced with a larger or smaller one with minimal effect on the rest of the spacecraft (change in the spacecraft dynamics would of course have to be considered in terms of impact on the attitude control subsystem). This technique is also becoming increasingly mature and well-understood. It is therefore recommended that this technology be considered for inclusion in the spacecraft design.

The fully-inflatable Powersphere™-type array may also be applicable, as it could be used as a "power-boost" option for smaller spacecraft for which a deployed, articulated solar array may be too expensive. It could certainly enable small missions that require a greater power level than could be attained by non-deployable solar arrays, but that could not employ solar array pointing. However, the technology for this device is less mature; it has not yet been demonstrated in space. It should perhaps be considered as a potential future option that may be incorporated into the design at a later date.

2.6.2.2 Concentrator arrays

Concentrator arrays employ Fresnel lenses to focus sunlight onto a reduced area of high-efficiency solar cells. The decreased area of cells reduces system cost and weight. Radiation protection is also enhanced by the greater thickness of material covering the cells.

This technology was used on the 1998 New Millennium Program Deep Space 1 mission, where 720 lenses focused onto 3600 solar cells, to give an array power of 2500W.[35] Specific power for the array was 44W/kg. The demonstration was deemed successful, with the cells operating within their nominal parameters.

This "SCARLET" array has been further developed since the DS-1 mission, and near-term projections are for specific power of 65w/kg at end-of-life. A blanket array system is also under development, with a projected specific power of 150W/kg at EOL.

Assessment of technology applicability

The technology is reasonably mature, and the reductions in system cost and weight that it offers are of potential interest for small spacecraft. However, they are not as great as those that appear to be offered by inflatable technologies. It may be possible to utilise the technology for body-mounted cells; there may be thermal implications to this however, as the SCARLET cells operated at 20°C above that of conventional cells.

2.6.2.3 Battery technology

The power subsystem of spacecraft may make up as much as 30% of the platform dry mass. Of this, energy storage forms a significant proportion. Therefore, technologies that increase the specific energy of batteries are of great interest for spacecraft, particularly small missions.

As small spacecraft are most commonly used in LEO, where (except for dawn-dusk sun-synchronous orbits) there will be an eclipsed period during each orbit, batteries used on smallsat missions may be expected to undergo around 5000 charge-discharge cycles per year. Technologies used must therefore be able to cope with this type of use. However, as the eclipse periods are relatively short (compared to those experienced by Geostationary spacecraft), depths of discharge may be limited to only 40%. Small satellite missions are also generally quite short (up to 2-3 years).

New battery technologies that may be of use to small spacecraft are described below.

Nickel Metal Hydride batteries

NiMH batteries have been used for terrestrial electric vehicles, but are now being developed for aerospace applications. They offer around 30% greater gravimetric specific energy than Nickel-Cadmium batteries and twice the volumetric specific energy of Nickel-Hydrogen batteries.[43] Batteries currently used in electric vehicles store around 100Ah, which is too high for small satellite applications. However, smaller-capacity, sealed aerospace-quality cells have been produced by Eagle-Picher and tested over more than 8000 cycles at 40% DOD.[40]

NiMH cells can be constructed in a simple prismatic configuration, similar to the conventional NiCd cells. This simplicity reduces manufacturing costs, and increases reliability. The geometry also facilitates easy battery construction from multiple cells.

Maturity: Near-term. Cells available from SAFT and Eagle-Picher e.g. 5A-h cell at 150g

Silver Metal Hydride batteries
AgMH cells are a relatively new development, and replace the nickel electrode used in the cells described above, with the higher energy density silver electrode.[44] This gives the potential for energy density increases over those of the NiMH cells. The cycle life is also expected to be suitable for small satellite applications. However, the technology requires significant further development.

Maturity: Not known to have been demonstrated in space. Still a relatively immature technology.

Lithium-ion batteries
Lithium-ion batteries are extensively used in portable electronic equipment such as mobile phones and laptop computers. They have found such uses mainly because of their low mass (lithium is the lightest metallic element). The takeoff in the use of such equipment has driven the interest in developing lithium-ion battery technology. It was estimated that in 1997, cells with a value of more than US$2000m were manufactured in Japan.[17] This level of use obviously drives down prices, and encourages great R&D effort, leading to improved performance.

The characteristics that have made lithium-ion batteries popular for use in portable communications devices, high energy density and ease of packaging, also make them attractive for space flight. Typical energy densities that may be achieved with this technology are 70-110Wh/kg. ESA is developing cells, for GEO applications, with a target of 150Wh/kg.[19] This type of battery is relatively untested in space; the main drawback has been cycle life. However, provided a low depth-of-discharge is maintained, then new lithium-ion batteries are becoming suitable for use on LEO missions.

Only a few spacecraft have so far used this technology. One example is the PROBA spacecraft, which uses a 9Ah battery of 36 lithium-ion cells. This spacecraft, and its battery, are currently operating nominally, and the mission is providing a convincing demonstration of lithium-ion technology.

Maturity: demonstrated successfully in space. Development is continually on-going and the subject of extensive R&D investment.

Lithium polymer batteries
Lithium polymer batteries use a similar chemistry to lithium-ion batteries, but employ a solid polymer electrolyte.[31] The resulting cells have a higher energy density than lithium-ion cells, with estimates of up to 200Wh/kg being made.[2] This type of cell is also more flexible in terms of configuration, as it can be made as thin sheets, or

packaged into different shapes to fit particular applications. This is due to the "sandwich", rather than "rolled" construction.[15]

This type of cell is already in use in some mobile phones and laptops, and is also being developed for use in electric vehicle battery packs. It has not yet been transferred to aerospace use, but its advantages of energy density and packaging may lead to its being adopted in the future. Applications could include a thin-film battery being incorporated into a solar panel to make a self-contained power generation and storage unit. This type of technology could be valuable in future small spacecraft.
Maturity: Gaining maturity for terrestrial use, space use still immature.

2.6.2.4 Combined energy/momentum storage flywheels

The concept of using mechanical flywheels as a means of storing electrical energy was original devised in the early 1970's. However, the level of electronics and materials technology at that time was not far enough developed to make such systems feasible.[45] However, interest in the concept has recently been renewed, and a prototype device is in development by SatCon Technology Corp, USA.

Advantages of the use of flywheels in place of batteries are:

- Fewer temperature constraints
- No capacity reduction over lifetime
- Energy densities of ~60Wh/kg
- Dual use for providing momentum-bias attitude stabilisation

The new flywheel concepts use low-friction magnetic bearings, and high-strength composites that can withstand wheel speeds of up to 50,000rpm. If the flywheel system is not to be used also for attitude control, then a dual-flywheel system can be used, with the wheels rotating in opposite directions to cancel out the angular momenta.

Assessment of maturity and technology applicability

This is an interesting concept, however the system complexity, and the required coupling between the power storage and attitude control subsystems may make this unsuitable for a modular spacecraft. It also does not offer enough of a gain in energy density over the battery options currently available or in development, and it is an immature technology compared to some of the higher-energy batteries described in the previous section. There may also be safety concerns with such high wheel speeds. It is therefore not considered that this is a viable solution for use in the spacecraft design at this stage.

2.6.2.5 Summary

Technology/ technique	Advantages	Disadvantages	Would enable:	Maturity
Inflatably-deployed thin-film arrays	High packing efficiency. High specific power (90W/kg, 250 W/kg predicted) Lower specific cost (target \$100/W)	Micrometeoroids may cause problems if inflated membrane not rigidified.	Higher power on small spacecraft missions.	Demonstrated in space on ITSAT
Inflatable thin-film arrays	Omni-directional Large array area from small undeployed volume & mass	Reliability issues	Passively-stabilised small spacecraft without sufficient surface area for conventional solar arrays	Under development, proof-of-concept produced.
Concentrator arrays	Higher specific power than normal arrays (44W/kg, 65W/kg near-term)	Thermal issues – array operates at 20°C higher temp. than conventional design	As above	Demonstrated in space on DS1
Nickel-metal hydride batteries	Higher energy densities than Nih2 Lower cost than NiH2 Higher specific energy than NiCd		As above Mass/volume critical applications	Space-rated batteries under test (adapted from electric vehicle batteries), and in commercial production
Silver-metal hydride batteries	2x higher volumetric energy density than NiMH		As above	Requires more development
Lithium-ion batteries	High operating voltage Very high energy density (70-110Wh/kg, target 150Wh/kg)	Sensitive to overcharge/ discharge Low cycle life if too high depth of discharge	As above	Demonstrated in space. Also under further development.

Lithium polymer (thin-film) batteries	Higher energy density than Li-ion (~200Wh/kg) May be configured into varied shapes. Could be integrated into a complete array/power-control/power-storage unit		Very small, compact microspacecraft	Under development
Combined energy/momentum storage flywheels	Multifunctional (attitude control & power), hence mass, volume reduction Greater temperature range No appreciable capacity drop from BOL to EOL High peak power Energy densities comparable to nickel-metal hydride	Complexity Couples attitude control and power subsystems Safety concerns (rotor operates at extremely high speeds: 50,000rpm) Would probably need a further power storage device for launch and orbit acquisition phase. Energy density not as high as for Li-ion (60Wh/kg)	Longer-lifetime missions, for example communications/ navigation Electric propulsion where periodic high power draws may be required Inertially-fixed attitudes	Currently immature. Possible experimental demonstrators in orbit within a few years? (Technique only recently became feasible via the development of high-strength composites)

2.6.3 ATTITUDE DETERMINATION

2.6.3.1 Vibratory microgyros[9,38]

The vibratory microgyro is a MEMS (micro-electromechanical system) device, which detects rotation via Coriolis-force-induced energy transfer between oscillatory modes. The device is fabricated from machined silicon, and offers dramatic size and mass savings over conventional gyroscope technologies. Its small size is indicated in Figure 2-20.

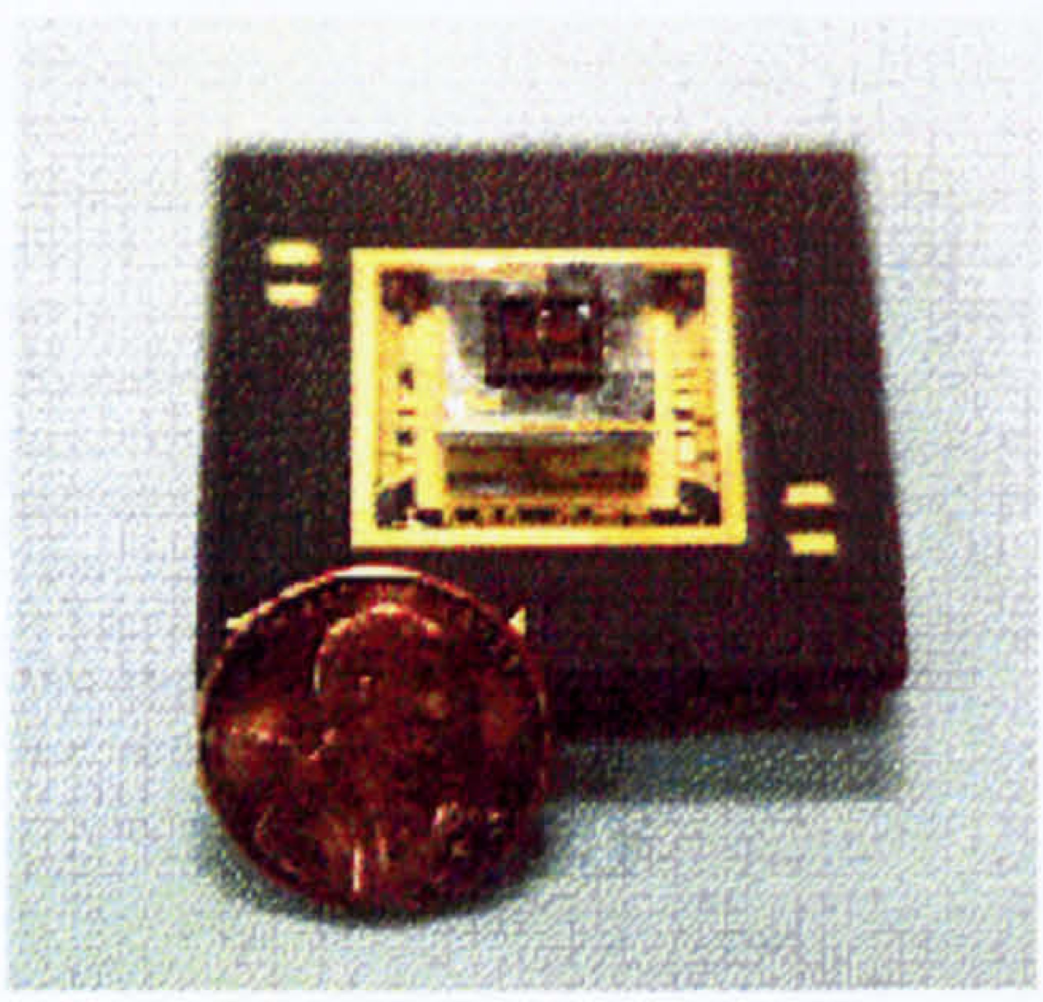

Figure 2-20 Vibratory microgyro

[Image: NASA Center for Space Microelectronics Technology]

The characteristics of the vibratory microgyro are compared with traditional gyros in Table 2-20.

	Traditional technology (Cassini Inertial Reference Unit)	Vibratory microgyro
Mass	5kg (4-axis unit)	<30g (3-axis unit)
Size	28x28x10cm	<1.5x1.5x3cm
Power	18W	1W
Drift	0.01deg/hr	30deg/hr (target: 0.1deg/hr)

Table 2-20 Comparison of vibratory microgyro with traditional gyro technology

Summary of technology maturity & applicability

It can be seen from the table that the key drawback currently with this technology is the drift rate. However, on-going development work on this type of device is expected to improve the drift characteristics. Involvement and investment from other interested parties, especially the military and the automotive industry, should see the technology being improved and matured relatively quickly. This is borne out by an estimate by the DARPA Microsystems Technology Office that the 2003 market for MEMS inertial measurement sensors will be of order $700-1400 million.

It is expected that this type of device will start to be flown on small satellites in the near term, initially as a redundant additional attitude reference sensor. Its small size, mass and power, and its low cost should make it ideal for future small satellite applications. The high drift rates make it less attractive in the short-term, however, and for mini- (rather than micro/nano-) satellites, the more mature ring laser gyros provide higher performance while remaining within acceptable mass limits.

2.6.3.2 GPS Interferometry for attitude determination

The use of GPS on spacecraft has become very widespread in recent years, for accurate orbit determination. It is now finding a new application for attitude determination. Multiple GPS receivers (a minimum of three) positioned at the extremities of the spacecraft can act as interferometers to provide attitude information.[51] The accuracy that may be attained is of order 0.1 degrees when a baseline of 1m is used[19] (this length of baseline may be reasonably expected to be possible on a small spacecraft).

The GPS receivers can also of course be used for the more conventional task of position determination, thus giving a combined attitude and orbit determination system. Such a system was demonstrated on the SSTL UoSAT-12 minisatellite in 1999. This demonstrated 0.5° attitude determination using 4 GPS patch antennas, with the total GPS package having a mass of 1kg.[56]

Assessment of technology maturity and applicability

GPS technology has been successfully used in space for a number of years, and this extension to its use has now been validated. It is a mature technology, and refinement of the interferometric techniques used will probably see continuing improvements in attitude accuracy.

The small size and mass possible for GPS attitude determination systems make them very attractive for small spacecraft, although the shorter antenna baselines possible on smaller vehicles limit the accuracies attained. This would mean that missions requiring greater pointing accuracies would require secondary fine-pointing sensors. However, the ability of GPS to easily go from a "lost in space" situation would still make them a useful coarse-sensing option.

It is recommended that the use of GPS for attitude and position sensing be considered for inclusion in the spacecraft design. This equipment can be backed up by higher-accuracy sensors if required; this is in keeping with the modular, upgradeable concept.

2.6.3.3 Summary

Technology/ technique	Advantages	Disadvantages	Would enable:	Maturity
Solid-state vibratory microgyros	Very small and lightweight (<0.03kg for 3-axis)	High drift-rates (but usable in conjunction with GPS or other reference)	Precise orbit control Control of orbit manoeuvres	Available (developed for missile applications) Performance increasing
GPS interferometry	Attitude determination can be combined with navigation Can go from lost-in-space attitude without requiring sun/Earth visibility	Long baseline required between GPS antennas to allow very accurate attitude measurement		Demonstrated on UoSAT-12

2.6.4 PROPULSION

Small spacecraft employ propulsion mainly for orbit insertion and maintenance, but it may also be used for attitude control. The technologies described here range from totally new developments to evolution of existing systems to make them suitable for use on smaller satellites.

2.6.4.1 Pulsed Plasma Thrusters

Pulsed Plasma Thrusters are electric propulsion devices that use the $\boldsymbol{J} \times \boldsymbol{B}$ Lorentz force to accelerate particles to high velocities. This is not truly a new technology – PPTs have been used on operational spacecraft in the past, but the technology has recently been the subject of a development programme aiming to reduce mass and increase performance, thus making them of interest for small spacecraft.[18] The primary benefits of PPTs are:

- Electrical power requirement which ranges down to a very low level (1-150W)
- Can be made in low-mass, low-volume configurations
- High specific impulse (>1000s)
- Inert solid propellant provides ease and safety of handling (particularly beneficial in small satellites, which often have strict safety drivers when sharing launches)

Overview of PPT operation:

The thruster operates by discharging a capacitor across the surface of a solid propellant bar (generally Teflon). The discharge arc ablates the propellant, creating a heated plasma that is accelerated to produce thrust. Pulses from the capacitor can be produced at any rate consistent with available onboard power.[19] The system is therefore inherently scalable. It also has no moving parts or valves; propellant feed is effected by the Teflon bar being continuously pushed up to the anode by a negator [36]spring.

PPTs have been used for several decades, but new developments in capacitor, integrated circuit and structural material technologies are expected to enable PPTs providing a total impulse of 20,000Ns, specific impulse of 2000s and a fuelled mass of 3.5kg.[37] Use of new polymers for the propellant may also further improve performance by reducing the energy required for ablation and plasma generation.

Analysis of technology maturity and applicability

The low power and mass, simplicity and safety of PPT technology makes these thrusters extremely applicable to small spacecraft orbit-raising and maintenance applications. The mass savings afforded over chemical propellants are considerable. An illustration may be given by comparing the propulsion system mass required to maintain a 100kg spacecraft in a sun-synchronous low-Earth orbit for 5 years. If a PPT was used, the required propulsion system mass would be 8kg, with a power consumption of 2.5W. In contrast, using a monopropellant hydrazine thruster, the mass would be 24kg.[37]

The basic technology is mature and well-understood; the new systems are being developed on a solid foundation of operational space experience. It is expected that

PPTs will be increasingly used on small satellites, and this technology may be considered as a propulsion option for the multipurpose spacecraft.

2.6.4.2 Ion thrusters

Ion thrusters operate by using an electric field to accelerate charged particles to extremely high velocities. The propellant used is generally xenon. They provide the highest available specific impulse, but have generally required too high a power level for use on small spacecraft. However, the increasing use of small satellites has driven the development of ion thrusters requiring power levels of 50-750W.[4,42] Laboratory models have been produced which produce a specific impulse of 2900s at 40W for an ion beam diameter of 5cm, and 2470s at 270W for an ion beam diameter of 10cm. The masses of these propulsion modules are in the range 5-15kg.

It is projected that this type of ion thruster would give significant mass savings over the use of chemical propulsion for orbit insertion of small spacecraft. For example, it is estimated that if an ion thruster of the type described above had been used by the small satellite TOMS-EP for its orbit insertion, in place of its chemical propulsion system, it could have increased its available payload mass by 38kg.[4]

Analysis of technology maturity and applicability

Conversion of ion thrusters technology towards use on small spacecraft is looking more feasible with the lower-power systems mentioned above. However, further development, and on-orbit demonstration is still required. The mass savings afforded by the use of this technology need to be seen to be great enough to push through more R&D effort. This is likely to happen given the growing commercial use of small satellites, but due to the high development costs that would currently be needed to ready this technology for use, it is not considered applicable to the small spacecraft being studied here. It should however be kept as a significant possibility for the future.

2.6.4.3 Low-Power Resistojets

Resistojets operate by resistive heating of the propellant gas. They have been used previously on large spacecraft, particularly for stationkeeping of Geostationary satellites. These mature systems typically operate at 500-1000W and use nitrogen tetroxide propellant.[47] However, a new, lower power resistojet is currently under development by SSTL, which can use nitrous oxide or water as its propellant.[54,55] Other propellants such as ammonia, carbon dioxide and propane are also being studied.

Advantages of low-power resistojets are:

- Low cost
- Low mass and power (compared to other electric propulsion)
- Safe - non-toxic propellants, generally stored at low pressures
- Restartable

The nitrous oxide resistojet has been demonstrated in space on the 1999 UoSAT-12 mission. The unit has a mass of 1.24kg, and can provide thrust of 125mN and specific impulse of 127s. It is shown in Figure 2-21. The power supply required is variable between 100-600W (the thruster is more efficient at higher power).

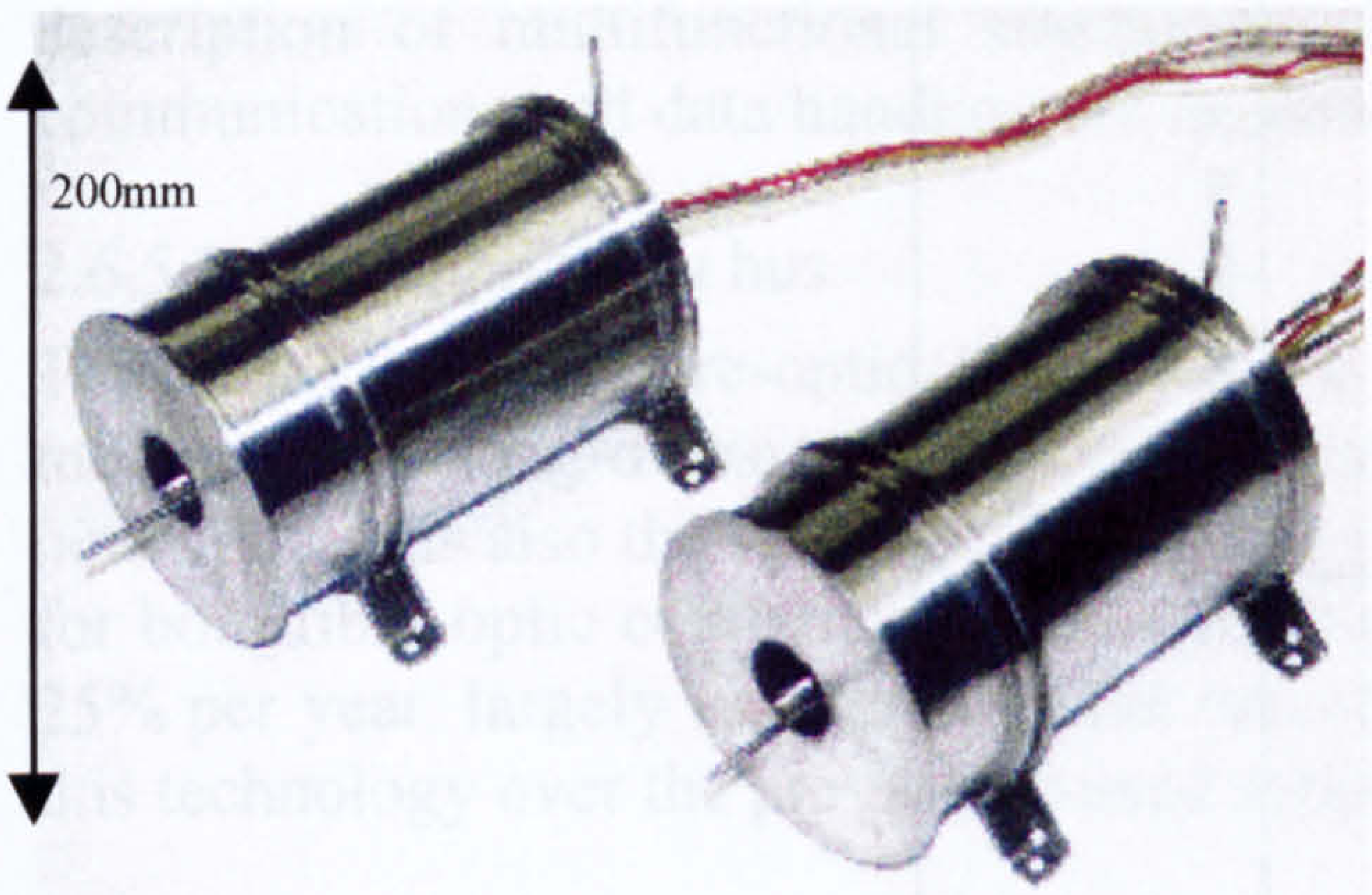

Figure 2-21 Resistojet thrusters [Image: SSTL]

Nitrous oxide is stored as a liquefied gas. It is then fed to the thruster unit, where its temperature is raised to 520K before being expelled through the nozzle.

Analysis of technology maturity and applicability
The extensive ground testing, and successful demonstration of the low-power resistojet in space, has matured this new technology to a stage where it should be considered for use on future small spacecraft. Its characteristics make it likely to find application for stationkeeping and small orbit corrections.

2.6.4.4 Kerosene biprop engine
Bipropellant engines have been widely used throughout the space age, but have generally been too large to be applicable to small spacecraft. Now, a small biprop engine is currently under study at the University of Surrey, which uses kerosene and hydrogen peroxide. Projected thrust levels of 50N, and specific impulse of 270s are quoted.[16] This engine has been extensively test-fired on ground, and a new design is to be constructed.

Advantages of biprop engine:

- High thrust
- Lower mass than equivalent monoprop system
- Potentially similar cost to conventional monoprop engine[49]
- Throttleable, restartable

Analysis of technology maturity and applicability
The kerosene biprop engine is still in the development phase, and has not yet been demonstrated in space, or as a flight-ready model. However, it is based on long flight heritage of bipropellant systems, and design overlaps with smaller systems used for attitude control on large spacecraft should enable the technology to be developed to a level suitable for flight demonstration in the near term. When the technology has matured further it is likely to find extensive use as an orbit acquisition kick motor, in place of the more commonly-used cold gas or hydrazine monopropellant systems currently used.

2.6.4.5 Summary

Technology/ technique	Advantages	Disadvantages	Would enable:	Maturity
Pulsed plasma thrusters	Very high specific impulse (2000s) Low mass (few kg) Lower power than resistojets (1-150W) Safe and simple design	Very low thrust, so would need long "burn" times Contamination	Low-energy orbit raising Stationkeeping	Long flight heritage, new and improved designs in development
Ion thrusters	Highest specific impulse Low propellant mass	High power (50-750W)	Orbit insertion, stationkeeping	Further development required to produce systems suitable for smallsats
Low-Power resistojets	Low cost Low mass (few kg) Non-toxic propellants, low storage pressures	Still requires relatively high power (100-500W)	Small orbit adjustments Stationkeeping	1st space demonstration on UoSAT-12
Kerosene biprop engine	High thrust (50N) Lower mass than monoprop systems Similar cost to monoprop systems	Complexity	Orbit insertion	In development

2.6.5 AVIONICS

Some new avionics technologies were mentioned in the Structures section, as part of the description of multifunctional structures. Other enabling technologies in onboard communications and data handling are described below.

2.6.5.1 Fibre-optic data bus

Terrestrial use of fibre-optic technology is now extremely widespread, with most medium and long-distance data communication being carried out over fibre-optic networks. It is also the subject of heavy and continued R&D effort, as the world market for both fibre-optic components, and for optical communications, is growing at around 25% per year, largely as a result of the takeoff in Internet[58] use. The key benefits of this technology over the previously-used copper cabling are:

- Increased data carrying capacity
- Lower mass (5-15 times lighter than copper cabling)
- Low power consumption
- Unaffected by electromagnetic interference
- Do not cause electromagnetic interference

These benefits have resulted in the technology being successfully crossed over into the aerospace sector. The 1995 Boeing777 was the first commercial airliner to use a fibre-optic avionics local area network. This has provided extensive demonstration of the application of fibre-optics to avionic systems. Spacecraft, particularly military missions, have also begun to adopt fibre-optics for their onboard communications. The US MIL-STD 1773 data bus is a fibre-optic version of the MIL-STD 1553 bus that has been used on many spacecraft. The 1992 SAMPEX mission employed this fibre-optic bus standard, and it was estimated that the use of this technology saved 1kg in harness mass, and several watts of power.[24] MIL-STD 1773 offers data rates of 1Mbs, with a newer version now supporting 20Mbps.

Another recent standard for fibre-optic communications is the IEEE Standard P1393 Spaceborne Fiber Optic Data Bus.[41] This bus uses a redundant fibre-optic ring topology, and up to 127 nodes can accommodate a range of devices from processors to analog sensors. The data rates achievable range from 200Mbps to 1Gbps; the rate is scalable to conserve power. This system offers fourfold power savings over conventional avionics systems.

The main drawbacks to adopting fibre-optics for onboard communications are:

- Harnessing personnel would require re-training as the techniques for fibre-optic cabling differ from those of traditional harness
- Precision splicing equipment is required for joining cables
- Raw materials are more expensive – though not necessarily when considered per bit of data transferred

Analysis of technology maturity and applicability

Fibre-optic technology is now extremely mature in terms of overall use (including terrestrial), and it is increasingly being used on spacecraft. The newer standard bus systems offer high data throughputs, making them extremely applicable for more advanced and data-intensive missions, such as remote sensing. The power and mass reductions will be extremely useful for small spacecraft applications, and the avoidance of EMC concerns simplifies design and testing.

It is considered that there is little reason for a new spacecraft to be designed to use metallic cabling, as the benefits offered by fibre-optic technology greatly outweigh the few drawbacks. The drawbacks themselves, such as the requirement for re-training of harnessing personnel, would in any case generally be less significant to an on-going programme producing successive spacecraft.

2.6.5.2 Autonomy & increased onboard processing capability

Autonomy is not a technology in itself, but rather a technique that has been enabled by advances in onboard processing technology. Spacecraft computers are becoming increasingly capable, to the point where they can now perform many of the tasks that previously had to be performed on the ground.[63] Autonomous functions to which space processors may now be applied include:

- Attitude and orbit determination and control
- Payload data processing and compression
- Battery charging control
- Power resource management
- Operations scheduling
- Thermal regulation
- Identification of anomalies
- (Limited) implementation of corrective actions

The benefits of shifting functional control to the spacecraft arise mainly from the reduction in operations costs. It can remove the requirement for constant staffing of the ground stations, which is extremely expensive. In addition, small missions are often conducted in LEO, where ground contacts are quite brief, and may be infrequent if few ground stations are used. Autonomy is therefore desirable from a survivability point of view; if the spacecraft develops an anomaly while out of ground contact, it should be able to detect the anomaly and place itself in an appropriate safe mode until it can be "rescued" by the ground support team.

Onboard processing of payload data can also be used to reduce the volume of data that requires transmission to the ground. This can reduce costs and enable missions by reducing the number of ground stations required, and/or the downlink data rate of the onboard communications subsystem.

Greater onboard processing power can allow more functions to be implemented in software rather than hardware. This fits in with the multi-mission concept very well, as the physical spacecraft computer architecture can remain the same, with only the flight software being modified. This process can be done in parallel with hardware build and integration, shortening schedule times.

Analysis of technology maturity and applicability
Improved autonomy for small spacecraft is currently being demonstrated in orbit on the ESA PROBA (PRoject for OnBoard Autonomy) satellite, launched in October 2001. PROBA flies various technology payloads, and uses onboard autonomy to plan and schedule operations, and manage onboard resources.[61] Increasing numbers of spacecraft employ autonomous functionality; the UoSAT control centre at SSTL is very rarely manned.

Autonomy is very applicable to small spacecraft, as a means of reducing cost and risk (where spacecraft must be out of ground contact for significant periods). However, the degree of autonomy must be chosen carefully. It is ideally suited to the control and management functions mentioned above, but corrective action implementation is a more dangerous area. If too much decision-making is to be done on board, this leads to a high complexity of onboard processing and extensive software development. This introduces extra cost and risk, which may outweigh any benefits.

It is impossible to plan for all contingencies, and attempting to replace the emergency decision-making abilities of a ground support team with a flight computer is not feasible. For anomaly recovery, the best compromise is to enable the spacecraft to perform sufficient self-test to identify when it has a problem, and place itself into a safe mode from which it will have the best chance of recovery.

2.6.5.3 Wireless onboard communications

Increases in the use of personal communications equipment, and the requirement for the different equipment to interface with one another, has led to the development of a range of wireless communication technologies. This technique was initially seen as a "cable replacement" when installing LANs in buildings. It is now being used to create "personal area networks", where mobile phones, PDAs, laptop computers and computer peripherals such as printers can be connected without the need for cables. The concept is also being taken further, to allow users to interface with resources such as LANs and the Internet whilst on the move, via the use of wireless communications terminals in cafes and shops.

This technology has potential application onboard spacecraft, where there is a situation somewhat analogous to that of a computer with various peripheral equipment requiring data connectivity. There are many potential benefits in adopting a wireless topology on a spacecraft:

- Mass and volume savings due to removal of data harness
- Labour savings from reduced harness manufacture and integration complexity
- Greater flexibility in siting sensors and other equipment
- New equipment could be introduced without requiring changes to architecture and harness

There are two transmission methods currently in use in wireless personal communications – IR and RF. These are discussed in turn below.

2.6.5.3.1 Infrared communications

This technique uses LEDs or laser diodes emitting in the THz frequency band, thus allowing high bandwidths (up to 20Mbps). They are, however, susceptible to interference by sunlight, and they offer no penetration so some degree of line-of-sight must be maintained between transmitter and receiver. The key benefits of this technology are:

- Low cost and simple
- High bandwidths possible (if sunlight interference is avoided)
- Not susceptible to interference by radio sources

Three types are available:

- Unidirectional – these are point-to-point systems using a focussed beam of laser light, and can operate over distances of up to a kilometre. They require careful alignment
- Omnidirectional – these systems broadcast the IR radiation, and make use of scattering by surfaces in the ambient environment to aid transmission. This technique is much simpler, and devices using it can be mobile within the broadcast area. However, the system is more susceptible to interference from sunlight, and the data rates possible are lower (1Mbps)
- Reflective – optical transceivers placed near devices are used to transmit data to a common location, from where it can be redirected to the receiving device

Analysis of technology maturity and applicability

Infrared systems have enjoyed substantial adoption for terrestrial wireless communications – many laptops and mobile phones have IR ports – however, they do not seem likely to be very applicable for use on spacecraft. The problem is that spacecraft, particularly small spacecraft, tend to be very densely packed, with little free volume through which the IR could propagate. This lack of the necessary lines-of-sight is a severe obstacle to using this technology, but it is potentially useful for specific cases where a line of sight is available.

2.6.5.3.2 Radio Communications

Wireless communications using radio frequencies offer the advantage of not requiring line of sight for transmission. Several RF systems are now being adopted for terrestrial personal wireless communications, the best-known of these are Bluetooth and WiFi.

Bluetooth

The Bluetooth system was developed by Ericsson, and is now being promoted by many other communications and IT companies, including IBM, Nokia, Motorola and Microsoft. It is a short-range wireless system, operating over about 10m, which was originally devised to replace the cables connecting PCs to peripherals such as mice, keyboards and printers. The frequency band is 2.4GHz, uses frequency hopping spread spectrum technology, and a data rate of 720Kbps can be achieved.[66] The key benefits of this system are:

- Very low cost
- Very low power (transmit power is 1 milliwatt)[65]
- Good noise immunity (via use of frequency hopping)
- Demonstrated, operational system

The capabilities of Bluetooth are now being extended to allow it to support data connectivity to shared network services, and enhance data security.

Analysis of technology maturity and applicability

This technology is being increasingly adopted for terrestrial applications, and its key benefits of low power and cost, coupled with no line-of-sight requirement would seem to make it ideally suited to use on board small spacecraft. It is definitely recommended that this type of system be considered for possible future inclusion on a smallsat platform, once it has been demonstrated in space. The main potential drawback with this system is that it operates in the S-band, which is the frequency band most commonly used for small spacecraft communications with the ground. Careful frequency selection and testing would be required to ensure that the onboard and space-ground communications systems did not interfere with one another.

Wi-Fi (IEEE 802.11b Wireless LAN)

The WiFi wireless LAN system operates in the same frequency band as Bluetooth, but employs direct sequence spread spectrum technology and supports a higher data rate (up to 11Mbps).[1] It also operates at a higher power, and can transmit over 50m device separations. This system allows comparable performance to a traditional wired Ethernet network. It is intended to be a higher-performance competitor to Bluetooth, but it is a newer standard and its use is not yet as widespread.

Analysis of technology maturity and applicability

WiFi systems are nearly as mature as Bluetooth for terrestrial applications, and share most of the potential space applications and drawbacks of this other system. The greater data rate could make this system more flexible for use in handling payload data on board. However, the power would probably need to be reduced, as a 50-metre transmission capability would be somewhat wasteful of power on a cubic-metre spacecraft.

2.6.5.4 Summary

Technology/ technique	Advantages	Disadvantages	Would enable:	Maturity
Fibre-optic databus	Not susceptible to/does not cause EMI Lower mass (5-15 times lower than copper cables) Greater capacity 4x less power required	May be a problem with radiation hardness Requires re-training for harnessing personnel, new equipment		Increasing use
Autonomy	Reduced ground control requirement Faster response to on-board events Greater flexibility Changes made more in software than hardware – more easily mission-tailored	Greater complexity of s/w & processor capability required	Missions with infrequent ground contact Unmanned ground stations (merely collect downloaded data)	Becoming used to an increasing degree. Various demonstrations including PROBA
Wireless onboard communications	Avoids mass and volume of data harness Low power (milliwats) Simplifies integration Equipment can be added/moved	Interference (RF systems) Need line-of-sight (IR systems)	Flexible and dynamic spacecraft architecture Lower mass and volume required for onboard data handling subsystem	Increasingly used in terrestrial personal communications applications. Not yet demonstrated in space.

2.6.6 RECOMMENDATIONS ON USE OF NEW TECHNOLOGIES

A number of new technologies have been examined in terms of their use as enablers for the small satellite platform. Their maturities are quite varied, with some being relatively untried. The most useful of the technologies identified, with regard to their maturity level and the benefits they offer, are summarised below.

- Inflatably-deployed solar arrays and sunshades
- Multifunctional structures (structure-mounted electronics and harness)
- SMA actuators
- Body-mounted concentrator arrays
- Lithium-ion batteries
- GPS for attitude and orbit determination
- Pulsed-plasma thrusters
- Low-power resistojets
- Fibre-optic data bus
- Wireless onboard communications

With the exception of the wireless onboard communications and multifunctional structures, it is considered that all of the above technologies are at a sufficient maturity to be included in a systems-level design for a spacecraft. The wireless communications and MFS options could be included at a conceptual level, to assess the benefits this technology would offer. This could indicate areas to which future research and development may be directed for enhancement of the platform.

One further consequence of the investigation of new technologies, is the recognition of the variety and number of new developments in this field. It is therefore important that the design should be flexible enough to be able to incorporate new technologies as and when they become available. In this way, the design can be allowed to 'evolve', and gain benefit from new developments. It is anticipated that the modular approach should make this possible, as was found with the MMS platform discussed earlier in this chapter.

As a final note, one of the technologies cited by the Panel on Small Spacecraft Technology as having high potential to make a large impact on cost and capability of small spacecraft is "Technologies to reduce cost and improve efficiency of up-front system engineering and launch and mission operations". It is considered that the modular, reconfigurable approach, where the spacecraft is 'pre-designed' and then the most appropriate configuration for a specific mission is selected, would come into this category.

2.7 CHAPTER SUMMARY

The preceding chapter has introduced the field of small satellites, and established some typical applications and design issues. Some past missions were examined, to gain an understanding of the types of design solutions that have been used, and the equipment and technology available.

The investigation then focused on the issue of modularity, and showed that the key aspects to enable modularity are the positioning and properties of the interfaces. Interface options and parameters were identified, for later use in the design section of the work. These identified possible ways in which the different subsystems may be broken down into interchangeable modular units.

The usefulness of decoupling the payload from the platform, and allowing module "back-compatibility" was identified. Benefits to the AIT process from integration and test mainly at module level were discussed.

Finally, this chapter investigated the new technologies emerging, which may enable the small satellite platform to give greater performance and flexibility. A number of technologies were identified, and the performance benefits analysed.

The next chapter addresses the commercial aspects and requirements of the satellite platform, by looking at markets, customers and suppliers. It also gives a more detailed analysis of likely applications.

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3 COMMERCIAL CONSIDERATIONS

3.1 INTRODUCTION

3.1.1 AIMS

In any engineering design project, it is important that the commercial requirements, as well as the technical requirements, are understood. For a real world application, the item designed must not only perform its intended function, but also result in a commercially viable product. This was key to the original intention of the research – to provide ideas that would be taken up by the original project sponsor, Space Innovations Ltd (SIL). However, even after the end of SIL, it was felt that the commercial emphasis should remain.

The aim of this section is therefore to provide a commercial rationale for the technical work undertaken – see recap of technical work below.

Recap of technical proposal:
To produce a systems design concept of a reconfigurable, multipurpose small satellite, together with the programmatic aspects required for its implementation.

This commercial analysis investigates the market for the spacecraft, likely buyers, and the cost envelope required to compete in the markets identified. This then provides further requirement inputs to the technical work, such as cost targets, and the range of mission types to which the platform should be targeted.

The expected outputs of this commercial rationale section are:

- A justification that there is a market or markets for the proposed satellite platform
- An examination of the likely customers and applications
- Cost regimes
- An analysis of the types of suppliers who may be likely to adopt such a programme and offer this type of spacecraft

Together with the technical rationale given in previous sections, this section reinforces *why* this type of spacecraft is useful, and *what* overall characteristics it should have. The subsequent technical design sections then cover *how* the requirements can be met.

3.1.2 APPROACH

The general commercial strategy of the proposed small satellite programme is introduced, with discussion of the differences between the approach proposed and the more traditional methods. This outlines the expected benefits, and potential problems or drawbacks that must be overcome.

A general introduction to the space industry as a whole is then given. This gives an overview of the types of activities supported by the industry, and the different sectors in which industry participants operate.

The remainder of the chapter describes the sequential approach used to identify the business applicability of the small satellite platform. This sequence is shown in Figure 3-1. The analyses performed build up a picture of the way the proposed platform will fit into the industrial environment.

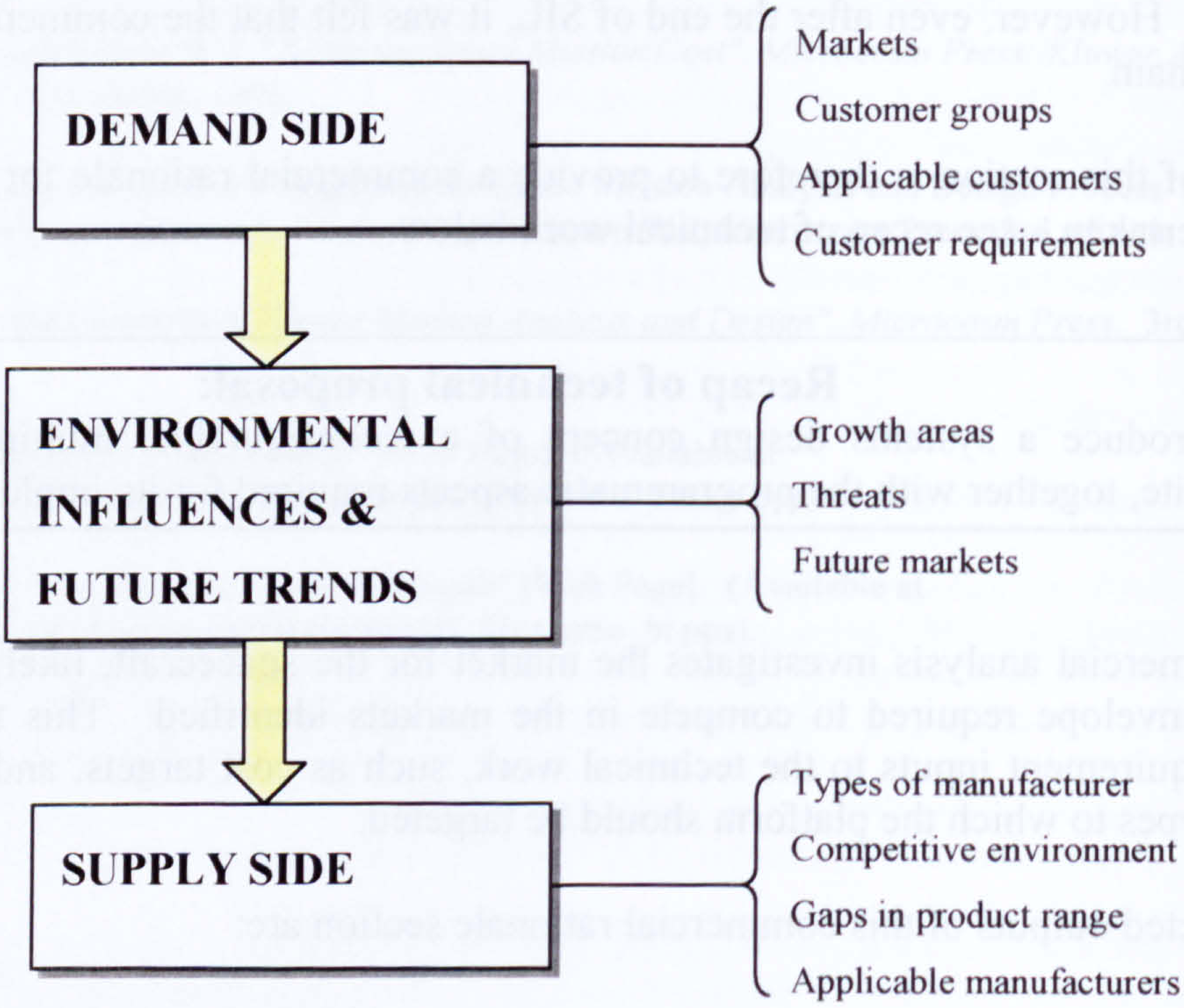

Figure 3-1 Approach used to produce commercial rationale

The areas covered in each of these sections is outlined below:

Demand side
This section characterises what types of customer groups exist in the satellite marketplace, what types of spacecraft functionality they are buying, and for what prices. This allows the buyer groups that fall within the scope of the proposed satellite to be identified.

Environment
This section analyses the structure of the industry, in order to understand the environmental factors and industrial forces at work. PEST and 5 Forces models are

used to characterise each of the sectors, and forecasts are used to identify likely future developments. This shows the possible evolution of the markets and customers, which would need to be accommodated within a commercial spacecraft programme.

Supply side
This section examines the suppliers in the industry, and characterises the different main types. Together with the environmental analyses, this allows the identification of supplier types most applicable to the proposed small satellite programme.

3.2 COMMERCIAL STRATEGY & IMPLICATIONS

This section identifies the way in which the technical proposal – a modular small satellite platform – can also offer strategic commercial benefits. To do this, the typical activities that are undertaken by a platform supplier are first examined, with a view to identifying problem areas.

3.2.1 TYPICAL ACTIVITIES INVOLVED IN A SMALL SPACECRAFT PROJECT

The following analysis assumes that an external customer is procuring a platform, i.e. it does not consider a "self-funded, self-built" spacecraft such as the SSTL demonstrator satellites.

A typical scheme may then be that the customer issues an Invitation To Tender (ITT) for the proposed mission. This will give details of the mission requirements, including payloads and launcher, and the scope of the project – what is to be delivered and when. Companies to which the ITT has been issued then have a (usually very limited) period of time in which to:

- Decide whether the mission is within their capabilities/ interest/ strategic goals
- Perform the necessary mission/ system analysis to produce a project proposal document

This document will usually contain the following items, to a greater or lesser degree of detail depending on the nature of the project, the specifics of the ITT, and the time available for tender preparation:

- Baseline system-level design of the spacecraft platform, including subsystem specifications and performance
- Preliminary operational schemes, including equipment duty cycles, power profiles, ground pass profiles
- Project schedules and timelines, including manpower requirements and milestones/ reviews
- Project costing

When preparing a project proposal, to respond to a competitive ITT, it is obviously beneficial to be able to provide as much detail as possible. This not only gives more confidence in the systems proposed, it also makes it easier to perform accurate cost and schedule analyses. This is particularly important if the supplier is later to be held to these costs and schedules quoted.

The difficulty in the above practice, is that the process of defining the spacecraft system is extremely labour-intensive, and the cost of this activity may be effectively wasted if the tender is unsuccessful. Of course, if the manufacturer already has a platform designed, then the process becomes much quicker and easier, as it is largely a case of providing the details of the satellite platform available.

However, if the platforms available do not match what is required for the proposed mission, the same problems occur, due to the need for re-design effort. The proposed strategy, with a reconfigurable modular platform, is to allow some of the design work to be pre-empted, giving significant labour and cost savings in successive project proposal activities. This strategy is outlined in the following section.

3.2.2 PROPOSED STRATEGY

As described in the introduction, the technical proposal is for a modular small satellite platform. The modularity, and different possible mission configurations of such a platform, allow a different strategy to be adopted by a supplier of the proposed design. Instead of having one design (which was perhaps designed around a previous mission), that is then re-worked to fit new missions, it is suggested that a range of different configuration possibilities are analysed in advance. This requires more labour "up front", but much of this is non-recurring, and means that much less time and effort is required to make good responses to ITTs. The philosophy is shown in Figure 3-2.

The figure shows the higher initial investment required, and the resulting lower level in later stages of the programme, compared with a traditional approach. This strategy obviously only provides a payoff when the programme continues over the production of many successive spacecraft. There are certain similarities with a 'production-line' approach, but it is important to note that with this scheme, the spacecraft are intended to be different and adapted to individual missions, rather than being mass-produced, identical products.

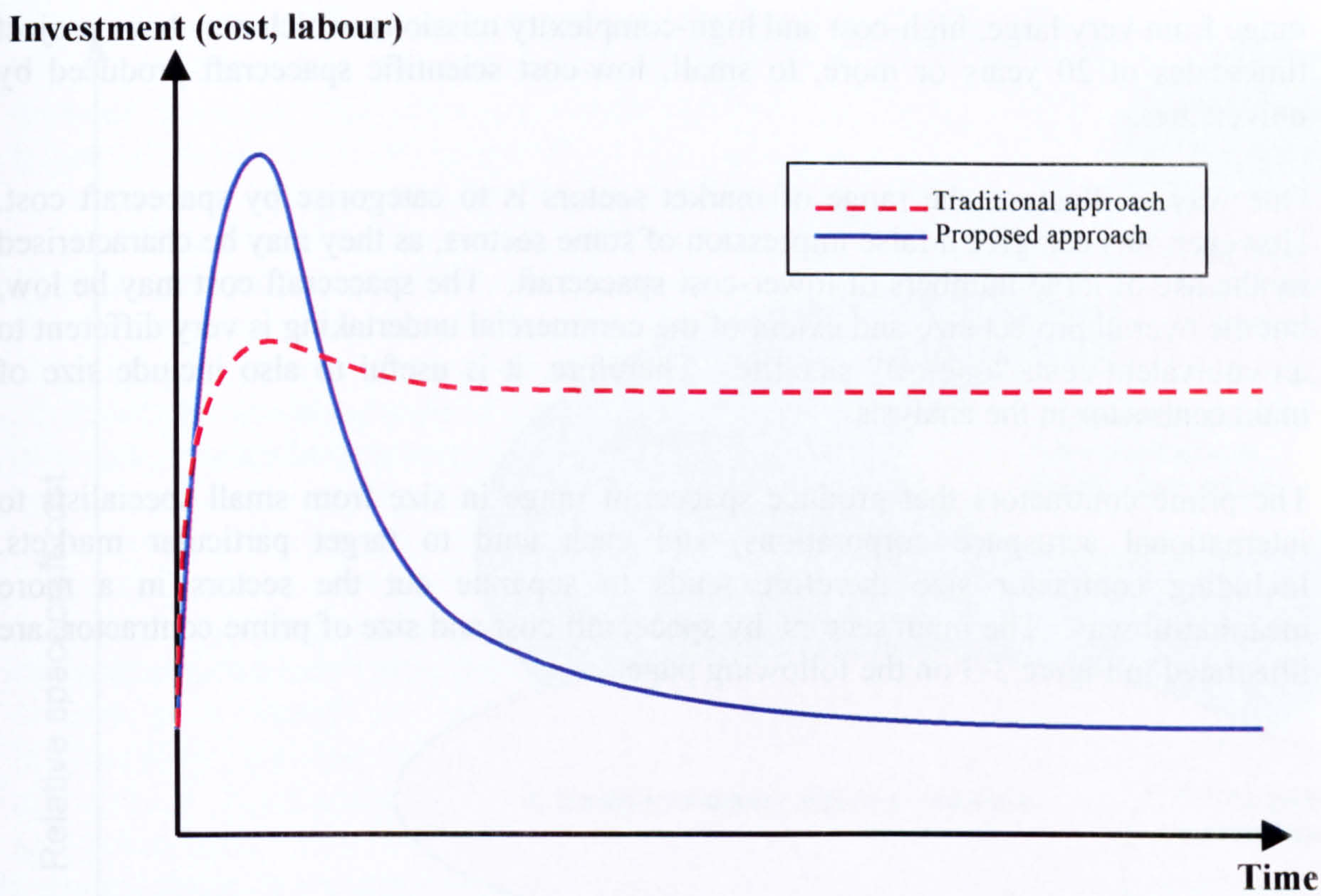

Figure 3-2 Levels of investment over time for a traditional spacecraft production approach and the proposed approach

To be successful, the proposed strategy requires the identification of potential target missions and customers, so that suitable configuration options for the 'pre-designing' can be selected. It is also a strategy that would not be suitable for all types of suppliers, due to the heavy initial investment required. A detailed investigation of customers, markets, and suppliers is described later in this chapter, following an introduction to the space industry itself.

3.3 OVERVIEW OF SPACE INDUSTRY SECTORS

Spacecraft applications may be broadly divided into the following main areas:

- Communications
- Earth Observation
- Technology
- Navigation
- Science

This does not provide a suitable means for categorising market sectors, however. Science missions may be performed on a 60kg microsatellite, or on a manned space station. Within the space industry, a number of broad sectors may be identified. These range from very large, high-cost and high-complexity missions, which may have project timescales of 20 years or more, to small, low-cost scientific spacecraft produced by universities.

One way to illustrate the range of market sectors is to categorise by spacecraft cost. However, this can give a false impression of some sectors, as they may be characterised by the use of large numbers of lower-cost spacecraft. The spacecraft cost may be low, but the overall project size and extent of the commercial undertaking is very different to an equivalent-cost, "one-off" satellite. Therefore, it is useful to also include size of main contractor in the analysis.

The prime contractors that produce spacecraft range in size from small specialists to international aerospace corporations, and each tend to target particular markets. Including contractor size therefore tends to separate out the sectors in a more meaningful way. The main sectors, by spacecraft cost and size of prime contractor, are illustrated in Figure 3-3 on the following page.

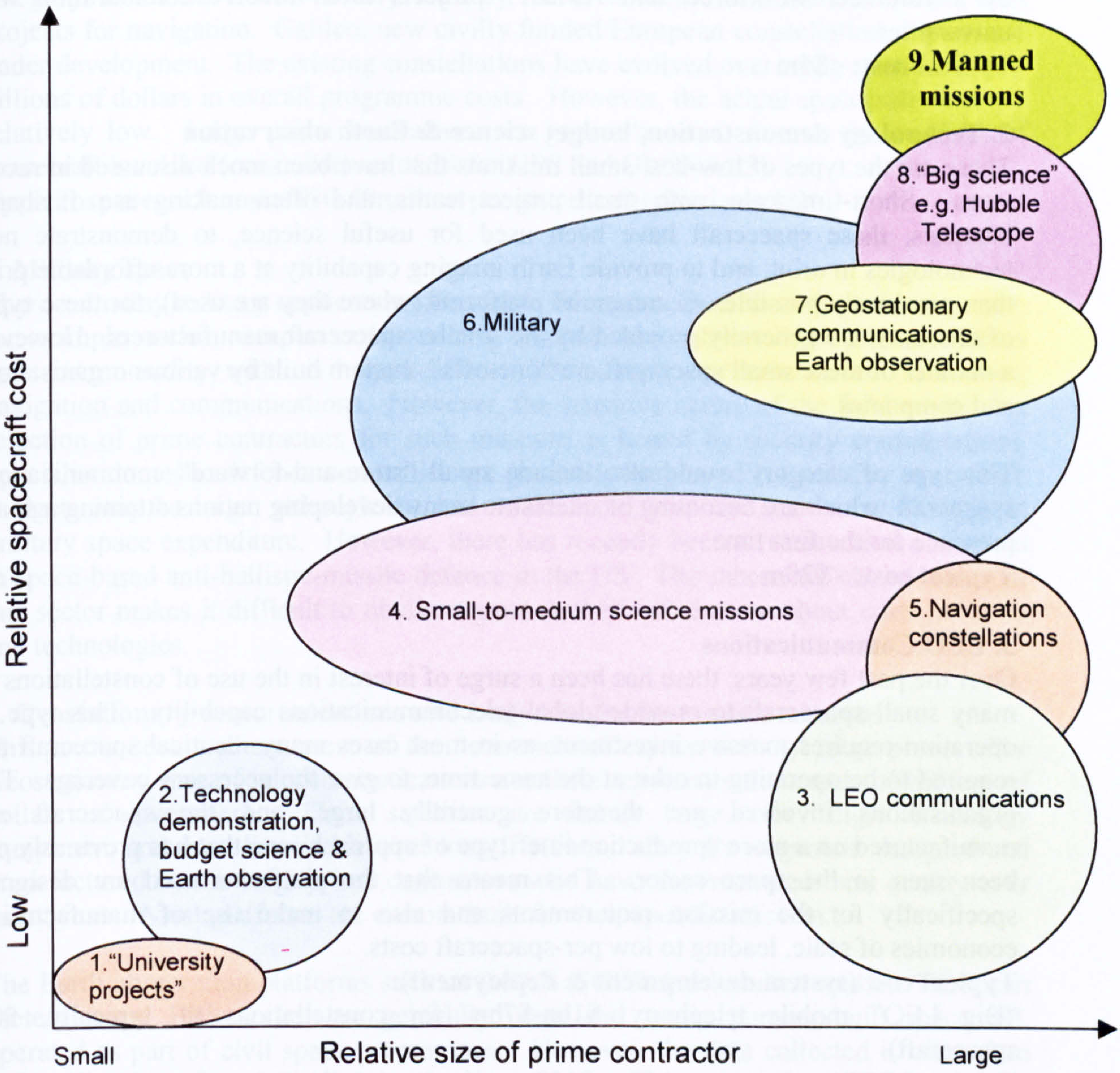

Figure 3-3 Main sectors within the global space industry

An overview of the characteristics of each of the main sector types is given below.

1. "University projects"

At the lowest end of the market are small spacecraft produced in-house by universities, generally on very tight budgets. These therefore make use of commercial components, fairly basic facilities, and are often prepared to accept higher failure risk and non-nominal performance in order to demonstrate experiments in orbit. These programmes often do not use prime contractors as such; as far as possible, the work is done in-house as part of student training. This category also covers the amateur radio satellites produced by the AMSAT organisations in various countries. These spacecraft are built

by a volunteer workforce, and AMSAT projects often involve collaboration with universities.
Typical cost: <$5m

2. Technology demonstration, budget science & Earth observation
These are the types of low-cost small missions that have been much discussed in recent years. Short-timescale, with small project teams, and often making use of shared launches, these spacecraft have been used for useful science, to demonstrate new technologies in orbit, and to provide Earth imaging capability at a more affordable price than previously possible. Commercial platforms (where they are used), for these types of missions, are generally provided by the smaller spacecraft manufacturers. However, a number of these small spacecraft are "one-offs", custom built by various organisations and companies.

This type of category would also include small "store-and-forward" communications spacecraft, which are becoming of interest to many developing nations attaining a space presence for the first time.
Typical cost: <$20m

3. LEO Communications
Over the past few years, there has been a surge of interest in the use of constellations of many small spacecraft to provide global telecommunications capability. This type of operation requires massive investment, as in most cases many identical spacecraft are required to be operating in orbit at the same time, to give the necessary coverage. The organisations involved are therefore generally large, and the spacecraft are manufactured on a more "production line" type of approach; one that has previously not been seen in the space sector. This means that the platforms used are designed specifically for the mission requirements and also to make use of manufacturing economies of scale, leading to low per-spacecraft costs.
Typical cost (system development & deployment):
"Big LEO" mobile telephony: $1bn-$7bn (for constellations of, typically, 50+ spacecraft)
"Little LEO" messaging: $2m-$650m (for constellations of, typically, 2-50 spacecraft)[8]
Approximate spacecraft costs may range from $2m to $50m

4. Small-to-medium science missions
These are the types of missions championed by NASA with its SMEX ("Small Explorer") and MIDEX ("Medium-class Explorers") spacecraft. The spacecraft are fairly small in size and rapid-schedule, but they may often use costly technologies and components that take them well out of the "budget science" category. Similarly, ESA has a science programme that includes medium-sized and small missions.
Typical cost: <$38m (small), <$76m (medium)[14]

5. Navigation constellations
Navigation satellites may be considered a subset of the military sector, as the main global navigation satellite systems, GPS and GLONASS, have been developed by the US and Russian military respectively. However, this has been placed in a separate

category, as GNSS is now widely used by the civil sector, and there are other civil projects for navigation. Galileo, new civilly funded European constellation is also now under development. The existing constellations have evolved over many years, and cost billions of dollars in overall programme costs. However, the actual spacecraft costs are relatively low. Large prime contractors are generally used, due to the numbers of spacecraft involved (constellations of 24 for GPS and GLONASS).
Typical spacecraft cost: $30-40million per spacecraft (plus launch)[6]

6. Military space
There are many military applications for space. Some are specific to the military sector, for example nuclear weapon early-warning systems, and anti-satellite systems. Others have commonality with civil applications, for example surveillance/reconnaissance, navigation and communications. However, the sensitive nature of the field means that selection of prime contractors for such missions is bound by security considerations. Specialist domestic defence companies are generally employed for military spacecraft. Budgets may be large, although the end of the Cold War has seen some slowing in military space expenditure. However, there has recently been a resurrection of interest in space-based anti-ballistic-missile defence in the US. The inherently closed nature of this sector makes it difficult to obtain a great deal of information about costs, markets and technologies.

7. Geostationary communications and large Earth observation missions
This is the sector where most of the "commercial" space activity takes place. The Geostationary ring of telecommunications satellites represents thousands of billions of dollars of investment. These satellites are mostly large, carry high technology equipment, and must be reliable over long lifetimes. They are in general designed and manufactured by the large prime contractors who have demonstrated their quality over many years. Most platforms are custom-built for their specific mission.

The Earth observation platforms such as the US GOES and Landsat, and the European Meteosat and ERS series, are generally large and costly purpose-built spacecraft, operated as part of civil space programmes. However, the data collected is often then disseminated commercially, as it is of great value to a wide range of end users.
Typical cost (for Geo spacecraft): $250m-2500m[17]

8. "Big science"
The staple of the large space agencies, such as NASA and ESA, the large science missions are the high-cost-high-scientific-return spacecraft, which broaden mankind's understanding of the universe. This is truly science for science's sake, in most cases, and a wealth of scientific breakthroughs have been made as a result of this type of mission. The life cycles of these projects are measured in decades, from initial proposal and selection of candidate ideas, through to manufacture and launch. They are funded by civil space programmes, usually via space agencies, and are generally produced by the large manufacturers.
Typical cost (whole life cycle): $1000m +

9. Manned missions

Due to the mission complexity, number of personnel involved, and the need for the highest reliability, manned spaceflight carries the greatest cost and the primes used are those with long experience and extensive resources.

3.4 BUYER/CUSTOME R ANALYSIS – DEMAND SIDE

The buyer-supplier structure in the space industry is an extremely complex one. The different market sectors outlined in the previous section all have very different customer characteristics, and even within each sector there are often many different ways in which satellites are funded and procured.

Across all space applications areas, there are three main categories into which customers may be grouped: civil, commercial and military. These groups procure spacecraft for some or all of the applications identified in the preceding section, as shown in Table 3-1 below.

	Civil	Commercial	Military
Communications	•	•	•
Earth observation	•	•	•
Technology	•	Limited	•
Navigation	•	Limited	•
Science	•		Limited

Table 3-1 Application areas for spacecraft procured by civil, military and commercial buyers

It should be noted that in this context, the "customer", or "buyer", is defined as the group or organisation that is directly purchasing the spacecraft platform hardware from the supplier. This is separate from any other customers further down the chain, for example users of satellite phones, GPS receivers etc.
The following analysis attempts to characterise the different buyer groups, both in terms of their motivations and in terms of the key types of utility they are buying. For each buyer group, the following questions are asked:

- What spacecraft do they buy?
- Why do they buy them?
- How much do they spend?

And in particular:

- What might they use small spacecraft for?
- What are the important factors for choosing a spacecraft platform?

This leads on to evaluation of their applicability as buyers of the small multipurpose satellite, and determination of the basis on which a commercial platform may be selected.

3.4.1 CIVIL MARKETS & BUYERS

This group is composed of the national and international space agencies, such as NASA and ESA, and other non-military government-funded bodies.

3.4.1.1 Characteristics and motivations

The motivations of civil space programmes are varied, but they generally are not intended to generate a profit, at least in the immediate term. The goals are mainly to enable scientific and technological research, and to promote the competitiveness of the space industries of the country or region concerned. (There is, therefore, an interest in the *future* profitability of the industry). For example, the European Space Agency states its purpose to be:

"Provide for and promote, for exclusively peaceful purposes, co-operation among European states in the fields of space research and technology and space applications, for scientific purposes and for operational space applications:
By elaborating and implementing a long-term European space policy
By elaborating and implementing space activities and programmes
By elaborating and implementing an industrial policy"

- (Article 2 of ESA Convention)

Other motivations for civil space programmes may be:[5]

- Promotion of international co-operation
- To encourage skilled workers to remain in the country – avoid "brain drain"
- To provide technological spin-off to other sectors, e.g. medicine
- Education
- To ensure national presence in space – prestige
- To be seen to get "value for money" from the taxpayers' money spent

There is also a motivation to fund missions that will provide benefits to the country/ region and its populace, via:

- Improved weather forecasting
- Gaining better understanding of the country's natural resources & their utilisation
- Disaster monitoring/ rescue operations
- Environment monitoring, e.g. pollution, global warming, ozone layer
- Providing civil navigation infrastructure
- Improving domestic communications infrastructure
- Early warning of potential dangers from asteroids/comets

The budgets available for civil space programs are obviously heavily dependent on the economic status of the countries concerned. Figure 3-4 shows the civil space expenditures for various space faring nations.

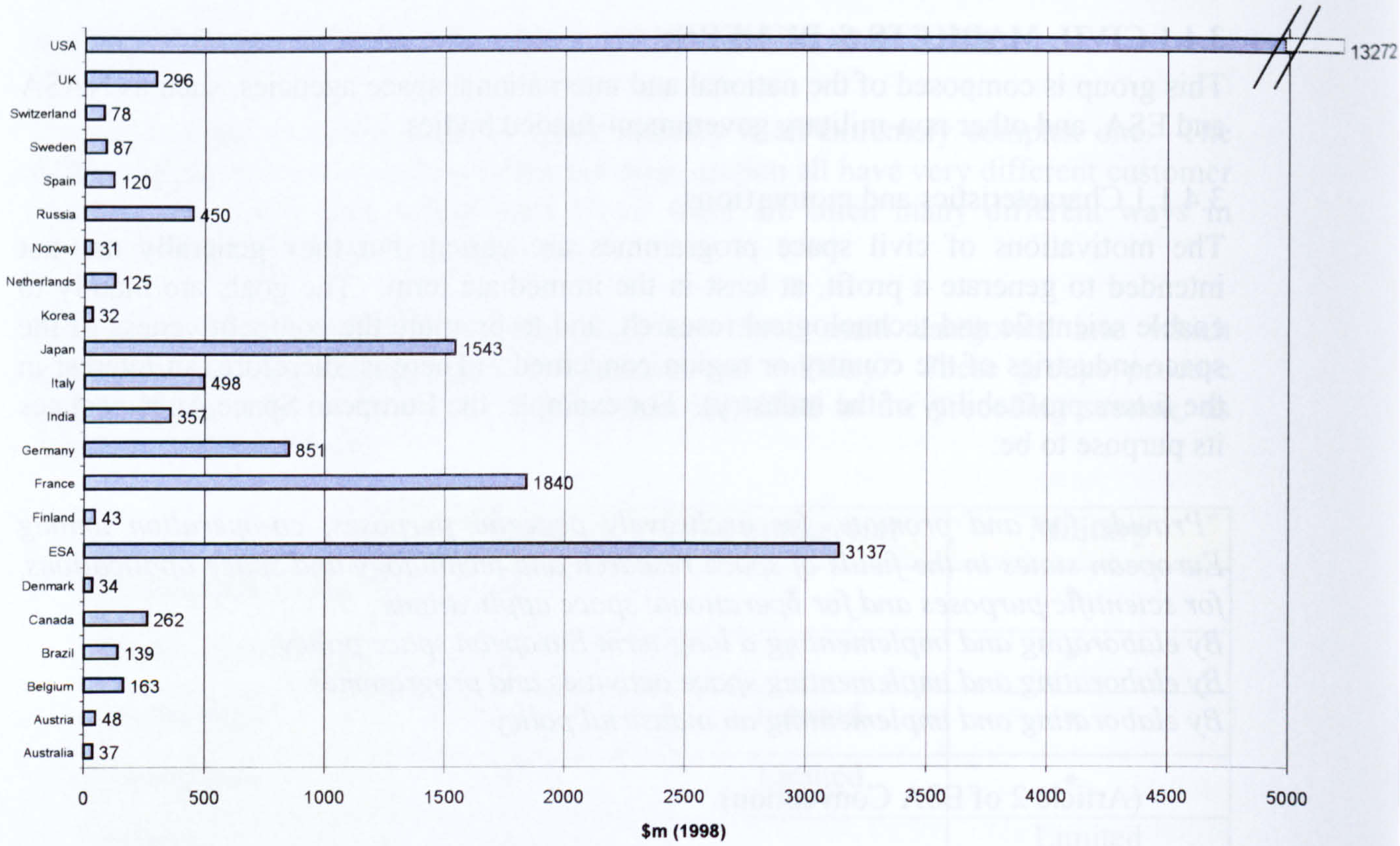

Figure 3-4 Comparison of world civil space expenditures (1998 figures)

The types of projects that are funded will vary from country to country, depending on the particular research/technology/applications priorities of that government or agency. It is generally the case that many more missions are proposed than ever get the funding to be carried through to production and flight.

The areas that are likely to be of most interest for small satellite applications are science, Earth observation, and commercial (i.e. pre-commercial technology demonstration). The numbers of launches for these categories are shown in Figure 3-5, 3-6, and 3-7.[12] These figures also illustrate the proportion of these annual launches that is accounted for by small missions.

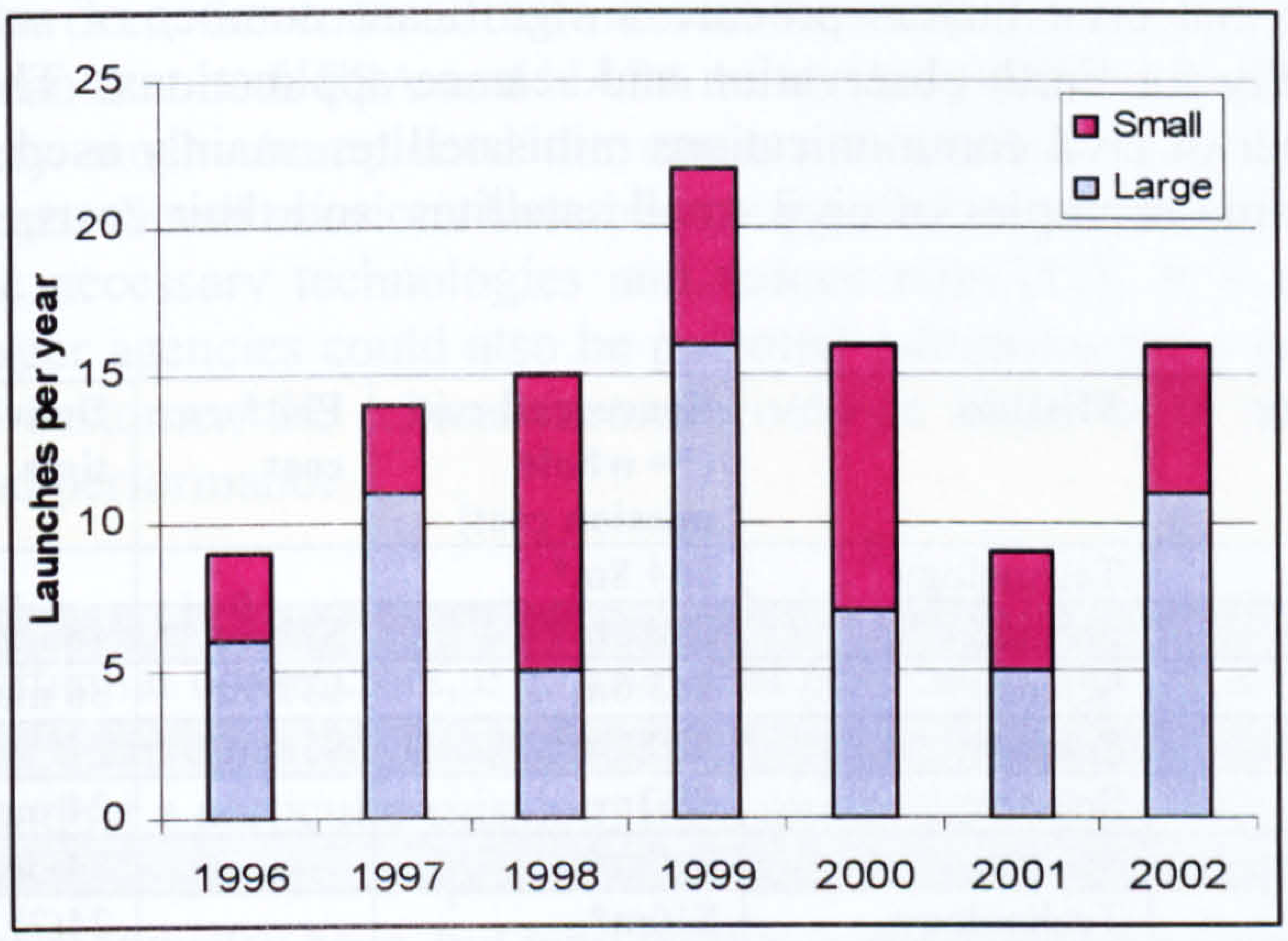

Figure 3-5 Civil Earth Observation launches per year

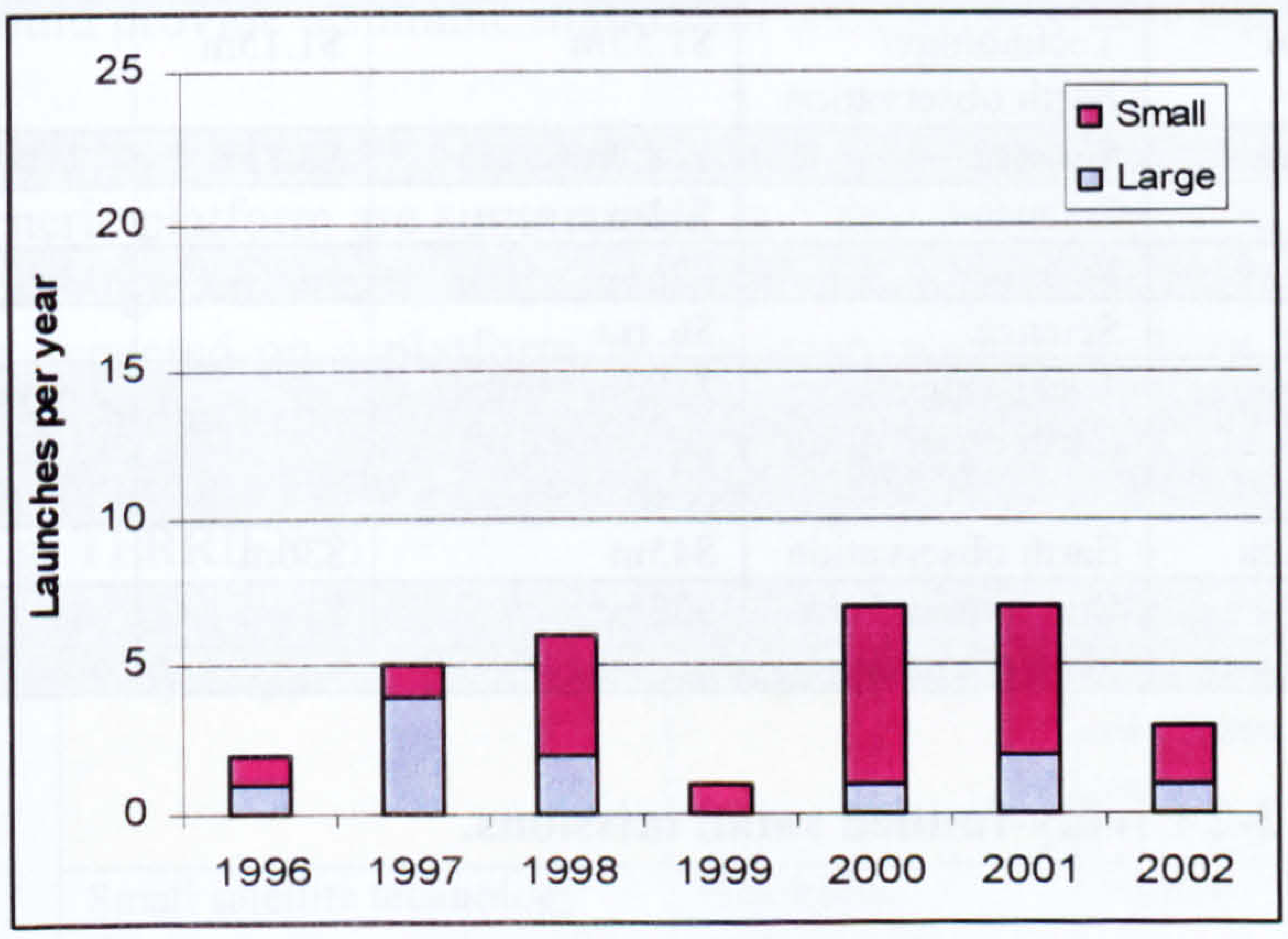

Figure 3-6 Civil technology launches per year

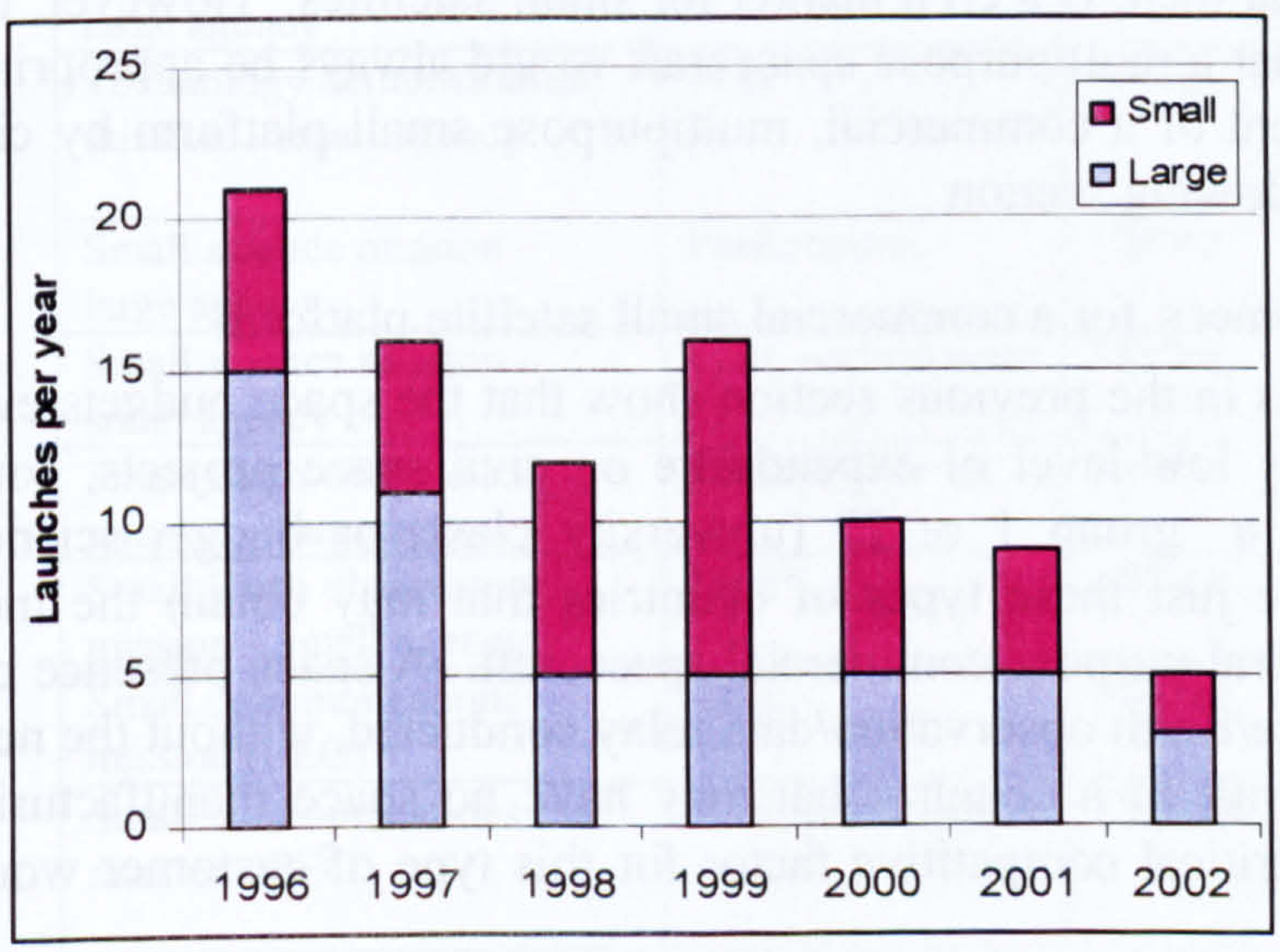

Figure 3-7 Civil science launches per year

The preceding figures show that civil buyers procure a significant number of small spacecraft each year, especially for Earth observation and science applications. There have also been a small number of civil communications minisatellites, mainly used for data collection and relay. Some examples of civil small satellites, and their costs, are given in Table 3-2.

Spacecraft	Country	Mission	Spacecraft cost (*= whole mission cost)	Platform cost	Delivery time
Lewis	USA	Technology	$64.8m*		
Freja	Sweden	Science	$18.5m	$12.5m	48 months
SAMPEX	USA	Science	$43.6m	$31.9m	36 months
HETE	USA	Science	$30.1m*	$5.6m	
ACRIMSAT	USA	Science	$8.3m		24 months
ODIN	Sweden	Science	$40m*		24-36
PROBA	ESA	Technology	$10m*		24(?)
SAC-B	Argentina	Science	$12.5m		
PoSat-1	Portugal	Technology/ Earth observation	$1.53m	$1.15m	
Orsted	Denmark	Science	$18.4m*	$8.8m	
SCD-2	Brazil	Science	$11m		
SNOE	USA	Science	$7m		
TERRIERS	USA	Science	$6.1m		
Earth Observing 1 (New Millennium Program)	USA	Technology/ Earth observation	$193m* (total development cost)		3-4 years
SAC-C	Argentina	Earth observation	$45m	$30m	
SCD 2	Brazil	Communications (data retrieval)	$37m*	$11m	

Table 3-2 Civilly-funded small missions.

The preceding data shows that there is a civil market for small satellites. However, this does not necessarily mean that a multipurpose spacecraft would always be appropriate. The likelihood of procurement of a commercial, multipurpose small platform by civil buyers is addressed in the following section.

3.4.1.2 Applicability as customers for a commercial small satellite platform

The space expenditure figures in the previous section show that the space budgets even of countries with a relatively low level of expenditure on civil space projects, could conceivably run to funding a "group 1 or 2" (university class, or budget science) mission. Indeed, it could be just these types of countries that may obtain the most benefit from a low-cost, general-purpose commercial spacecraft. A space presence can be attained, and useful science/Earth observation/data relay conducted, without the need to develop the entire spacecraft in a country that may have no space manufacturing expertise or facilities. The critical competitive factor for this type of customer would probably be cost.

The larger space agencies, and wealthier and more "space-oriented" countries, have often preferred to concentrate much of their efforts on larger, more prestigious missions.

However, the commercial interest in small spacecraft and its associated technologies mean that both ESA and NASA have also procured small missions. Indeed NASA has very much championed the cause of "smaller, faster, cheaper" spacecraft in recent years. Small missions have been suggested as precursors to big projects, to demonstrate the necessary technologies and reduce risks.[11] It is therefore quite likely that the larger agencies could also be potential customers for a generic platform. For this type of customer, the critical factors would be likely to be mission flexibility, adaptability, and performance.

When a civil space project is funded, funding may often be conditional on selection of particular contractors, e.g. from that particular country, of a particular size or type etc, if the motivation is to encourage domestic industry. However, if the motivation is to conduct a particular mission, for a minimum budget, then foreign manufacturers may be the most attractive option. This is quite often the case with countries with a limited space industry base, but who require particular space capabilities. This type of situation would provide a suitable market for a small, low-cost platform.

The types of civilly funded mission that may lead to the purchase of a commercial generic platform are summarised in Table 3-3. The funding agencies have been divided into large and small, as the available funds (and therefore the maximum spend that may be expected on a platform) for a large agency such as NASA or ESA, are obviously much greater than for the space agency of a single, smaller, country. However, as may be seen from Table 3-2, larger agencies such as NASA also fund very low cost missions (e.g. TERRIERS).

Project type	**Critical factors**	**Estimated platform price**	**Remarks**
Small satellite technology demonstration mission – for large agency	Flexibility, performance	<$40m	Dependent on range of payloads flown and performance requirements.
Technology demonstration mission – small agency	Cost	<$10m	Lower cost than for small science/ Earth observation, as generally shorter mission
Small science mission – large agency	Performance	<$40m	
Small science mission – small agency	Cost, performance	<$15m	
Small Earth observation mission – large agency	Performance	<40m	
Small Earth observation mission – small agency	Cost	<$15m	
Small communications mission (LEO)	Cost	<$15m	Data relay-type application
Small communications mission (GEO)	Cost, performance	<$100m	Price estimate based on reduction from low end of GEO comsat prices

Table 3-3 Potential civil applications for a generic small satellite platform.

The platform price that may expected to be acceptable for each type of application has been estimated from the quoted costs of previous civil missions. For all these applications, it is reasonable to assume that cost will be a factor; however, for some mission types, other factors may become more important. For more "advanced" missions, a cost increment may be less detrimental than a performance decrement. For a small country, launching a spacecraft for the first time, the philosophy may be more one of "What can we get for *x* dollars?"

3.4.1.3 Constraints to applicability

Procurement policies for civil projects are often not simply to choose the cheapest platform that satisfies the technical specifications. Other considerations must often be taken into account, such as the desire to procure from domestic suppliers, for example the "Buy America" Act in the U.S. A similar principle rules ESA procurement. There, the *juste retour* policy demands that ESA member states are awarded contracts of a value proportionate to that state's contribution to the Agency.

There are also restrictions on the transfer of certain technologies and information, such as the Arms Export Control Act, which limits the transfer of technologies applicable to weapons guidance etc.

Another factor that could count against the procurement of a multipurpose platform for a small civil mission would be that it does not involve the development of a new spacecraft bus. If a country or agency intends to drive the acquisition of an "in-house" small satellite capability, via the funding of a small mission, then buying in an existing platform would not be appropriate. However, this problem can be mitigated by the use of technology transfer agreements, such as the training schemes practised by SSTL. Here, engineers from the procuring country are invited to participate in the design and production of the spacecraft, thus acquiring skills to take back for use in developing a domestic space capability. This has been done with Korea, Algeria, and Nigeria.

3.4.1.4 Recommendations

On the basis of the types of missions procured by civil buyers, it is recommended that the multipurpose platform have a capability to be used for small scientific missions, where payloads are being separately funded and developed. Similarly, it is also recommended that the platform be suitable for low-cost Earth observation, again with the payloads to be procured separately. For technology missions, it is expected that a platform capable of supporting science and Earth observation missions would probably be also appropriate for demonstrating technology. Indeed, it has been seen that technology demonstration has been combined with these applications on a number of small missions.

3.4.2 COMMERCIAL MARKETS & BUYERS

This group is composed of companies and organisations within the private sector.

3.4.2.1 Characteristics and motivations

The chief motivation for commercial space activity is obviously to generate profits. Companies generally obtain a return on their investment in the spacecraft and supporting infrastructure, by offering ground-based goods and services arising from the operation of those spacecraft. These goods and services have historically mostly been satellite communications, both point-to-point and broadcast, but recently other commercial markets have emerged. Still more are predicted for the future.

Due to the heavy investment required, commercial spacecraft operators are generally large firms or organisations. However, within the commercial sector, there is a large variation in the way firms operate, and in the way they procure spacecraft. This section examines the main areas within the sector, in which small satellites are employed.

In the past 10 years, there has been increasing interest in the use of small satellites, particularly in Low Earth Orbit, for commercial communications systems. Furthermore, advances in miniaturisation have opened up a market for commercial Earth imaging from small spacecraft. By the late nineties, commercial launches to LEO were at unprecedented levels, and forecasts were for further increases. The following sections give an overview of activities and programmes within the commercial small satellite sector, which can be divided into 4 main areas – "Little LEO", "Big LEO" and Broadband LEO communications, and Earth observation. This shows where there may be potential customer groups for a small satellite platform.

An interesting footnote to the above is that turnover in commercial space activities overtook government-funded activities in 1996.[10] This is probably due mainly to the increasing utilisation of satellite communications, coupled with the general budget cutbacks experienced by the large space agencies.

3.4.2.1.1 "Little LEO" Communications Systems

These are at the lower-cost end of the commercial Low Earth Orbit communications satellite market. They employ store-and-forward techniques to provide narrowband services such as messaging, asset-tracking, and data collection from remote sites. A summary of existing and proposed Little LEO systems is given in Table 3-4.[8]

System	Operator	Prime Contractor	No. of S/C	Mass /kg	1st launch	Remarks
ORBCOMM	International Licensees LLC	Orbital	48	43	1997	Operational Total cost was projected as US$220m [Ref. Jane's space dir 2001] Platform + payload cost $10.7m for 1st s/c, <$1.6m recurring costs [ref. Reducing Space Mission Cost]
VITASat	Volunteers in Technical Assistance	Surrey Satellite Technology Ltd.	2		1993	Comms packages piggybacked on other S/C. Awaiting relicensing by FAA.
FAISat	Final Analysis	Final Analysis	38	151	2003	Under development. 2 test S/C launched.
LEO One Worldwide	LEO One (USA)	Dornier	48	125	2002	Under development. Launch contract signed.
E-Sat	E-Sat Inc.	Alcatel	6	113	2002	Under development. Launch contract signed.
KITComm	KITComm (Australia)	AeroAstro LLC	21	100	TBD	Under development.
IRIS	SAIT RadioHolland (Belgium)	SAIT Systems	6	60	TBD	Under development.
Courier/ Convert	ELAS Courier (Russia)	Moscow Institute of Thermotechnics	12	502	TBD	Proposed.
Gonets-D	Smolsat (Russia)	NPO PM	36	231	TBD	Proposed. 6 test S/C launched.
LEO One Panamericana	LEO One Panamericana (Mexico)	TBD	12	150	TBD	Proposed.
LEOPACK	Space Agency of Ukraine	TBD	28	TBD	TBD	Proposed.
SAFIR	OHB Teledata (Germany)	OHB System	6	60	TBD	
Temisat	Telespazio (Italy)	Kayser Threde	7	40	TBD	Programme on hold. Temisat 1 launched 1993.
Elekon	NPO PM/ Elbe Space (Russia/ Germany)	NPO PM	7	TBD	TBD	Proposed. Comms package as piggyback on navigation satellites.

Table 3-4 Little LEO Systems, operational, in development, and proposed.

The only currently operational Little LEO system is ORBCOMM. This has had its problems, however. The original ORBCOMM organisation filed for US Bankruptcy Court protection in September 2000, after delays in user hardware and software development meant that many orders did not result in actual sales. ORBCOMM is now in the hands of International Licensees LLC, a company composed of European/Asian service providers and investment firms.

Despite the setbacks experienced by this first constellation, many other organisations are taking an interest in this sector, and proposing or developing systems. The end markets these service operators are targeting include:

- Transportation firms – asset-tracking for lorries etc
- International organisations – monitoring of widely-spread assets
- Corporations, governments – machine-to-machine data transfer, paging, email, text-messaging

The operators of this type of service generally seem to be choosing prime contractors at the smaller end of the space manufacturing sector, and the spacecraft are mostly lower-cost satellites. E.g. FAISAT $250m for 26 satellites, Orbcomm $135m for 26 spacecraft.[16]

3.4.2.1.2 "Big LEO" Communications Systems

These are systems that provide global voice communications services. They utilise constellations of spacecraft to give complete coverage of the Earth's surface, allowing use of mobile telephones anywhere in the world. A summary of existing and proposed Big LEO small satellite systems is given below (not included are several systems employing spacecraft outside the mass category of interest here):

System	Operator	Prime Contractor	No. of S/C	Mass /kg	1st launch	Remarks
Globalstar	Globalstar LP	Space Systems/ Loral	48 + 8 spares	447	1998	Operational, but in financial difficulties. US$1.3billion contract for 56 satellites [ref. Jane's space dir 2001 pp71]
Iridium	Iridium Satellite LLC	Motorola (Bus built by Lockheed Martin)	66 + 6 spares	680	1997	Now operated by DoD/Boeing, after assets acquired following bankruptcy. Costs originally projected as US$3.45b for the system & spares, plus US$2.8b over 5 years for operations & maintenance. Lockheed Martin contract US$700m for 120 satellites. [ref. Jane's as above]
ECCO	Constellation Communications	Orbital	46 + 8 spares	703	TBD	Under development. Estimated system cost <US$500m [ref. Jane's as above]
Ellipso	Mobile Communication Holdings	Boeing	16 + 1 spare	998	TBD	Under development. Estimated cost US$564m (16 s/c, construction, launch, 1 year operations) [ref Jane's as above]
ECCO II	Constellation Communications	TBD	46+	585	TBD	Proposed.
Globalstar GS-2	Globalstar LP	TBD	64 + 4 spares	830	TBD	Proposed.
ECO-8	Brazilian Space Agency	TBD	11 + 1 spare	249	TBD	Under study.
Gonets-R	Smolsat (Russia)	NPO PM	48	953	TBD	Proposed.
Koskon	Koskon Consortium (Russia)	AKO Polyot	45	862	TBD	Proposed. Payload tested in 1991.
Rostelesat	Kompomash (Russia)	TBD	115	839	TBD	Proposed. Awaiting funding.
Signal	KOSS Consortium (Russia)	NPO Energia	48	308	TBD	Proposed.

Table 3-5 Big LEO Systems, operational, in development, or proposed

This sector of the market has seen some dramatic failures. The Iridium constellation provided a sound technical solution to global communications, but unfortunately did not attract the customers that the original heavy capital investment had made necessary. Iridium LLC filed for bankruptcy in 1999 after only a year of service, and the constellation is now run by Iridium Satellite, under contract to the US Department of Defense. The DoD contract was for 2 years from December 2000; the future of Iridium after that will depend on its ability to attract new customers.

The Globalstar constellation has also suffered from a lack of customers, and has been unable to meet its financial obligations since its inception. In early 2001, it announced that it may have to seek bankruptcy protection.

A third Big LEO service, ICO, is not mentioned in the above table, as the spacecraft it uses do not fall into the small satellite category. However, this company has also been forced to file for bankruptcy.

The above examples are the result of an over-estimation of the market for the services offered. Massive growth in the much cheaper ground-based cellular phone industry has squeezed the customer base for satellite phones into such a tiny niche that it cannot support the infrastructure required for these Big LEO systems.

3.4.2.1.3 Broadband LEO

The growth of the internet and increasing reliance on its use, has opened up a huge market for broadband data communications. The use of broadband communications satellites can provide service to customers who are inaccessible by landline. There are a number of proposed broadband systems. However, due to the complexity and high transmitter power required by this type of spacecraft, all the proposed systems employ spacecraft that lie slightly outside the small satellite category. This market has been mentioned as it may be a potential future application, as small satellites become more capable.

3.4.2.1.4 Commercial Earth observation/remote sensing programmes

There are a number of commercial Earth Observation programmes in operation or under development. These are summarised in Table 3-6 on the following page.

The images obtained by these commercial projects are used heavily by the fishing industry, aiding in the production of fish distribution maps. They are also used in agricultural management, naval applications, and scientific and environmental research.[4] This is a growing market, with image products (from the whole range of Earth observation spacecraft, large and small) estimated to be worth several billion dollars worldwide.[3]

Spacecraft	Operator	Prime Contractor	Mass /kg	Launch	Remarks
Orbview 1	ORBIMAGE	Orbital Sciences Corp.	74	1995	Cost of 1 Orbimage spacecraft & ground system: <US$100m
Orbview 2	As above	As above	372	1997	
Orbview 3	As above	As above	185		Planned for launch in 2001
Orbview 4	As above	As above	185		Planned for launch in 2001
IKONOS 1	Space Imaging	Lockheed Martin	816	1999	Launch failed Cost of 2-satellite programme plus 5 years operation projected as US$500m
IKONOS 2	As above	As above	816	1999	
IKONOS 3	As above	As above	TBD		Planned for launch in 2004
IKONOS 4	As above	As above	TBD		Planned for launch in 2004
Quickbird 1	EarthWatch	Ball Aerospace	815	2000	Launch failed
Quickbird 2	As above	As above	909		Planned for launch in 2001
EROS A1	ImageSat International	Israel Aircraft Industries	280	2001	
EROS B1-B6	As above	As above	350		First launch planned for 2003
RESOURCE21 1-2	RESOURCE21	Boeing	TBD		Planned for launch in2005
GEROS 1-6	GER Corporation	GER Corporation	225		First launch planned for 2003

Table 3-6 Earth observation small satellites in orbit and proposed [2]

3.4.2.1.5 GEO Communications

Geostationary communications satellites have always been predominantly large, heavy and complex spacecraft. This trend seems set to continue, with spacecraft mass increasing as launcher capabilities increase. There have, however, been proposals for GEO minisatellites, including the BNSC-funded SSTL Gemini spacecraft. These could make use of spare capacity on launches of the large spacecraft, which may have masses of over 5 tonnes.

GEO minisatellites could find a market among developing nations, by providing a lower-cost, “entry level” Geostationary platform for television, radio and telephone services. Advances in power generation and orbit maintenance technologies for small spacecraft may enable this type of mission.

3.4.2.2 Applicability as customers for a small satellite platform

The applicability of each of the commercial small satellite buyers, as customers for the small commercial platform, is addressed in the following sections.

3.4.2.2.1 “Little LEO”

Some of these types of systems employ only a few spacecraft, and are at the cheaper end of the market; therefore a commercial generic platform may be suitable for this application. For a small constellation, on a tight budget, it may not be desirable to use purpose-built platforms. The key requirement will be that the platform must allow the communications payload to perform its mission as required, as, in a commercial scenario, degraded performance would not be acceptable. An acceptable price range for a platform for this application may be estimated to be of order $2m-$10m depending on performance requirement (based on the cost of the ORBCOMM values for the higher range, and “micro” platforms such as UoSats for the lower range).

3.4.2.2.2 "Big LEO"

A generic small satellite platform would not generally be applicable to this sector, as the constellations utilise purpose-designed, assembly-line-manufactured spacecraft. It would not therefore be expected that this type of organisation would form a potential customer group for the satellite. However, there are impacts made from this sector. The high expenditure on the design and development of the spacecraft platforms used for the Big LEO constellations has resulted in the maturing of several technologies and techniques. These could be applied to a new generic platform. The multiple launches used have also pushed forward the capabilities of launchers to insert several spacecraft into orbit at once, thus simplifying the sharing of launch capacity (and hence costs).

3.4.2.2.3 Commercial Earth observation

Small commercial Earth-imaging missions may provide a suitable application for the generic platform, as they will generally use only one or two spacecraft, rather than large constellations to which a dedicated design is more appropriate. The crucial factor is that the generic platform must have sufficient performance to support the imaging payload, at a price lower than that required to produce a tailor-made spacecraft. There will be a trade-off between price and performance, with a certain performance level being necessary to provide images of a commercially viable quality. A suitable generic platform may be an enabler for commercial programmes that would otherwise be unfeasible due to the high development costs for a dedicated platform.

A reasonable price for a platform for a commercial Earth imaging mission may be up to around the $15m mark – slightly more than was estimated for a communications platform because of the generally single-spacecraft nature of these projects (i.e. there is less benefit obtained from developing a specialised platform for repeatability).

3.4.2.2.4 GEO communications

The platform is currently envisaged as being designed for LEO applications, however, a GEO capability could be introduced as an option. Increasing numbers of countries want access to Geostationary satellite services; and a small spacecraft can offer them the autonomy and independence of controlling their own domestic satellite.

3.4.2.3 Constraints to applicability

Commercial buyers may be less willing to compromise on performance and reliability than civil buyers. They are selling an end product, which is dependent on the nominal and timely performance of the mission. Degraded mission performance will impair their ability to obtain the necessary return on their investment. Therefore, constraints may arise from unwillingness to buy a new product, until it has been previously demonstrated. This may even follow through to a wariness of a different *variant* of an otherwise demonstrated platform.

Key success factors here are therefore performance, reliability, and confidence in the manufacturer.

Another constraint to applicability may arise from the number and frequency of spacecraft required. There will be production limits due to the size of the manufacturer, and the facilities and resources available. The production constraints will therefore depend on the type of manufacturer who may potentially adopt this type of spacecraft design strategy.

3.4.3 MILITARY MARKETS & BUYERS

This group is composed of domestic and international organisations concerned with defence and its associated activities.

3.4.3.1 Characteristics and motivations

The primary motivations for military space missions are to improve or maintain national security, and provide support for military operations. Typical applications are:

- National security – reconnaissance, ELINT, early warning systems, missile defence
- Military operations support – communications, weather forecasting, target location & damage assessment, navigation, "anti-satellite" satellites

In support of this, the military also funds R&D missions to demonstrate new technologies applicable to the areas described above.

There are usually around 10-20 military launches every year worldwide. The military satellite launch rate since 1996 is shown in Figure 3-8.[12]

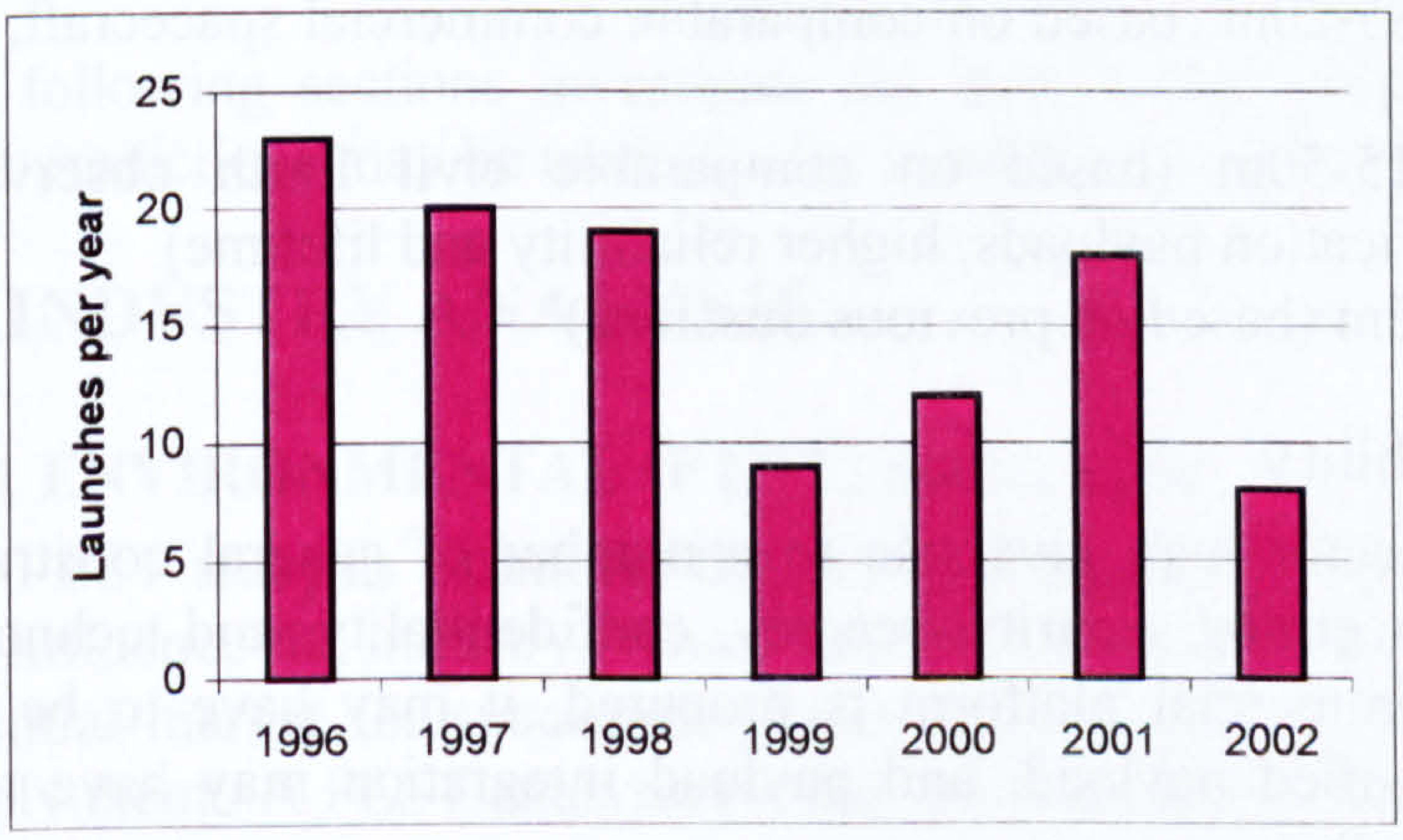

Figure 3-8 Military launches per year

It is not known exactly what proportion of these launches are small spacecraft; many military launches are classified, and even the spacecraft mass is not disclosed. However, it is known that at least some military missions employ small satellites. The potential applicability of a commercial small platform to military missions is discussed in the following section.

3.4.3.2 Applicability as a customer for a small satellite platform

Typical characteristics of military spacecraft are high reliability, high specification, and therefore high cost. A military communications satellite system may cost over £1billion.[9] However, the missions undertaken by small spacecraft may allow some relaxation of military specifications, particularly when budgets are limited. And Western military space budgets *have* generally become much more limited over the past decade or so, since the end of the Cold War.

Where wealthier countries are concerned, it is probably the demonstration missions that are most within the scope of small satellites. For example, France utilised a commercial microsatellite platform, produced by the UK's SSTL, for its Cerise military technology demonstration mission. It would be expected that most of France's other military spacecraft would have specifications far beyond that attainable by a commercial small platform.

The US Ballistic Missile Defence Organisation (BMDO) produced a series of minisatellites to test sensors in orbit rapidly and at relatively low cost. The Miniature Sensor Technology Integration (MSTI) programme has produced 3 spacecraft to date. MSTI-1 cost US$15m; the two follow-on missions cost US$10m each. The sensors demonstrated on these missions probably were, or will be, utilised on higher-cost, higher-performance satellites.

Less wealthy countries may utilise smaller spacecraft for full operational missions, particularly surveillance and secure communications. For example, the Isreali Ofek series of small satellites were used for surveillance, and some of the Russian Cosmos surveillance/communications satellites are only a few hundred kilogrammes.

Reasonable cost ranges for military small satellite missions may be expected to be:

Communications – of order $5-20m (based on comparable commercial spacecraft, with additional lifetime/ reliability)
Surveillance – of order $15-50m (based on comparable civil Earth observation spacecraft, with higher-specification payloads, higher reliability and lifetime)
Technology – of order $10-15m (based on previous missions)

3.4.3.3 Constraints to applicability

The nature of the military sector may give rise to a number of general constraints. These arise out of issues concerning security, secrecy, confidentiality, and technology transfer limitations. If a commercial platform is procured, it may have to be kept largely separate from a classified payload, and payload integration may have to be conducted by military personnel.

The security issues may also restrict military buyers to the use of domestic suppliers only, or suppliers from "friendly" countries. It may also be the case that the military supply chain is generally well-established, and therefore hard to break into.

3.4.4 SUMMARY OF POTENTIAL CUSTOMERS & MARKETS

The analysis of the different customers has identified the following likely buyers and applications for the spacecraft platform. These are summarised in Table 3-7.

	Civil	Commercial	Military
Communications	Low-cost domestic communications Government messaging Cost range: $2-15m	Point-to-point Store-and-forward messaging Asset-tracking Cost range: $2-15m	Low-cost secure communications Cost range: $5-20m
Earth observation	Low-cost weather satellites Disaster monitoring Resources Cost range: $15-40m	Images for fishing, agriculture industries Cost range: ~$15m	Surveillance Operations support Cost range: $15-50m
Technology	Small demonstration missions to promote domestic industry Cost range: $10-40m		Small demonstration missions to test new systems Cost range: ~$15m
Science	Low-cost scientific research Cost range: $15-40m		

Table 3-7 Summary of potential customers & markets

The following sections investigate the factors that affect these industry sectors, and what predictions may be made for the markets, using trends and forecasts.

3.5 INDUSTRY ANALYSIS

3.5.1 ENVIRONMENTAL (PEST) ANALYSIS

The PEST analysis examines the political, economic, social and technological factors that influence the industry. These are the environmental factors that will impact on the potential market for spacecraft platforms, and also on the ability of manufacturers to supply them. A PEST analysis for the space industry overall is shown in Table 3-8.

POLITICAL	SOCIAL
• Restrictions on technology transfer between certain countries • Levels of government expenditure on domestic and international space programmes • Preferences for domestic suppliers/suppliers from preferred countries • Beneficial alliances/agreements with other parties e.g. launch providers, suppliers • Legislation governing uses of space • Government defence policy	• Demand for satellite-based benefits e.g. weather forecasting, disaster monitoring • Demand for mobile personal communications • Growth of the Internet • Pressure for environmental research • Public perceptions on military/ civil spending • Use of small satellite projects as training tools in universities • Public desire for a satellite as a marker of an important event e.g. Millennium? • Demand for defence early-warning systems • Voters' attitudes to space-based military systems
ECONOMIC	**TECHNOLOGICAL**
• Levels of activity in space industry (National/global) • Strength of the domestic economy • Costs of raw materials • Perceived economic benefits from spin-off, technology transfer etc • Globalisation of business • Military budgets • Investor confidence	• Technical capabilities of products • Government investment in space R&D programmes • Availability of materials and components • High rate of change of technological "goalposts" • "Spin-in" of technology from other industries • Requirement to stay up to date with new technologies and components, particularly those of rival countries or organisations • Competition form other technological solutions e.g. unmanned long-endurance aircraft, balloons, airships, terrestrial cable communications

Table 3-8 PEST analysis for the space industry

The civil space sector is heavily influenced by the strength of the domestic economy, and the priorities of the governing bodies. Public perception of the utility of space programmes may also have a significant impact, as government spend on space must ultimately be justified. This factor explains the high levels of effort that large agencies, especially NASA, put in to public outreach projects.

The commercial sector is mainly influenced by customer demand, and competition between rivals. The competitive environment drives new technological developments, which can in turn influence other sectors of the space industry.

The military sector has many similarities with the civil sector, in terms of its influences. It also has a certain degree of similarity with the commercial sector, in terms of countries competing for strategic advantage in their space infrastructure, although this aspect is somewhat reduced from its Cold War levels.

3.5.2 FUTURE TRENDS

Examining trends in the small satellite market, and attempting to forecast what may happen in the future, can give some idea about the on-going need for small spacecraft platforms. It allows some degree of prediction to be made about likely future application areas, and how these may evolve. Also, trends and events that may impact on the popularity and viability of small missions may be identified.

In many cases, valid predictions are very difficult. Examination of trends may not provide reliable projections of what the future may hold for a particular market. However, it may, if nothing else, give an indication of which markets are fairly stable, and which are volatile. This may help to give an idea of the risk of targeting a particular market.

3.5.2.1 Industry forecasts

There are a number of sources of relevant industry forecasts. One such source is launcher payload forecasts. These are prepared by the US Associate Administrator for Commercial Space Transportation (AST) and Commercial Space Transportation Advisory Committee (COMSTAC) groups, and assess likely demand for launch vehicles, from different categories of spacecraft. The 2001 Commercial Space Transportation Forecasts are shown in Figure 3-9. These are the predicted numbers of different types of Non-GEO spacecraft, requiring commercial launch worldwide. This does not therefore include domestic spacecraft that are launched by domestic launchers, e.g. an ESA mission launched by an Ariane.

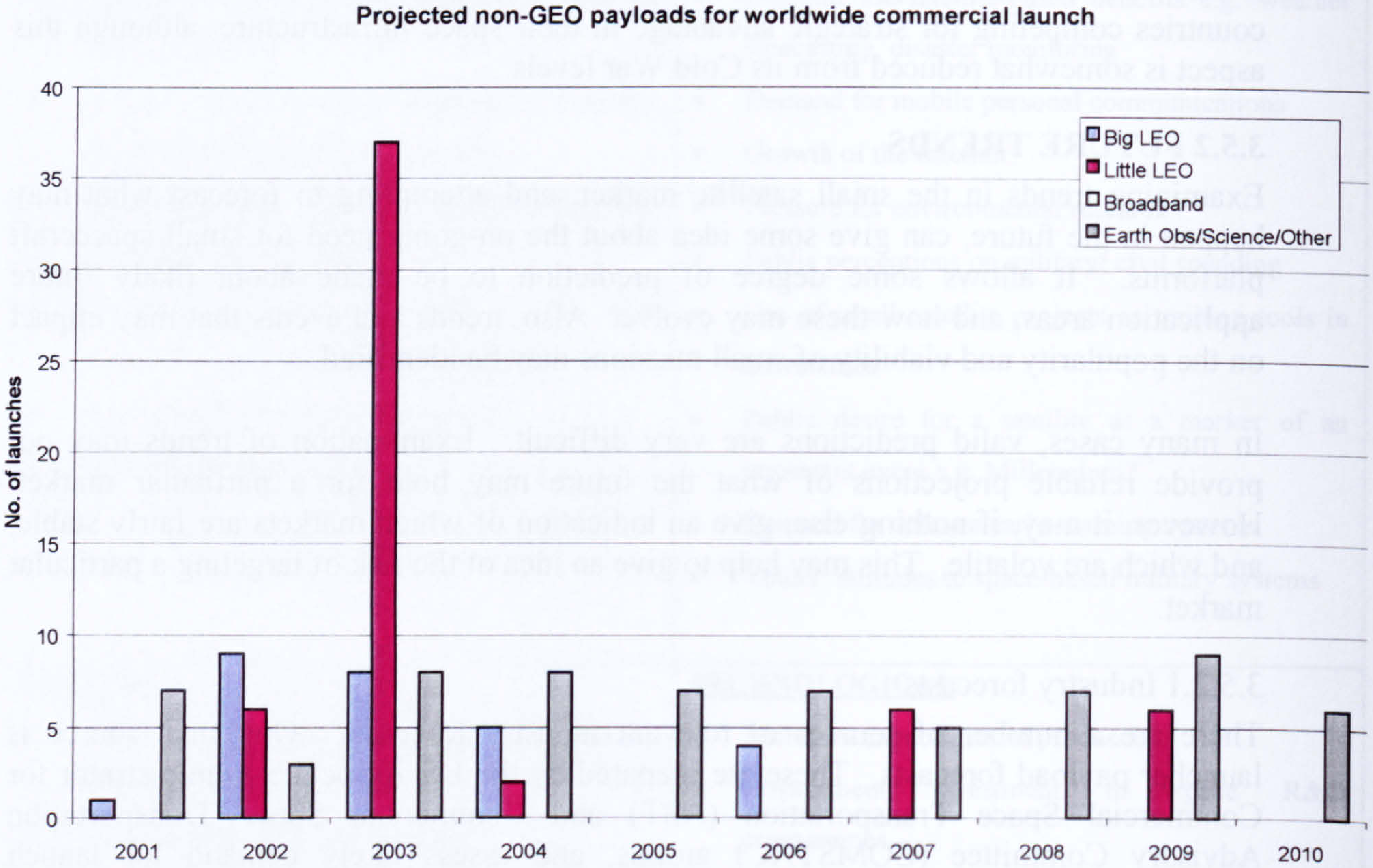

Figure 3-9 Predicted number of non-GEO spacecraft for commercial launch[7]

These predictions show the fairly steady levels of Earth observation and scientific spacecraft production, compared with the greater fluctuations in the communications markets. A high number of Little LEO spacecraft expected in the near term, due to the planned FAISat, LEO One and other constellations (mentioned in Section 3.2.5.1.1). Probably due in part to the bankruptcy suffered by ORBCOMM, it is reported that Little LEO start-ups have had difficulty in securing investment capital.[7] However, despite the setbacks, there is a substantial customer base for messaging services, and it is projected by AST that there will be one to two Little LEO constellations deployed.

Similarly, Big LEO systems have also suffered from the problems experienced by Iridium and Globalstar. A number of applications for FCC licensing of new systems have subsequently been withdrawn. However, there are still some constellations in development, and the AST projects that one new Big LEO constellation will be deployed by 2010.

This projection indicates no LEO broadband satellites are expected by 2010. This is blamed on low investor confidence, high start-up costs, and competition from other areas (terrestrial cable, GEO satellites). However, with the broadband services market expected to be US$100billion by 2006, constellations such as Teledesic may find their required investment. There is therefore some disagreement from differing sources regarding the expectations for broadband satellite (see below).

Another industry projection is compiled by Merrill Lynch. This also analysed by application sector, but shows projected industrial turnover. The Merrill Lynch projections for the period up to 2007 are shown in Figure 3-10.[1]

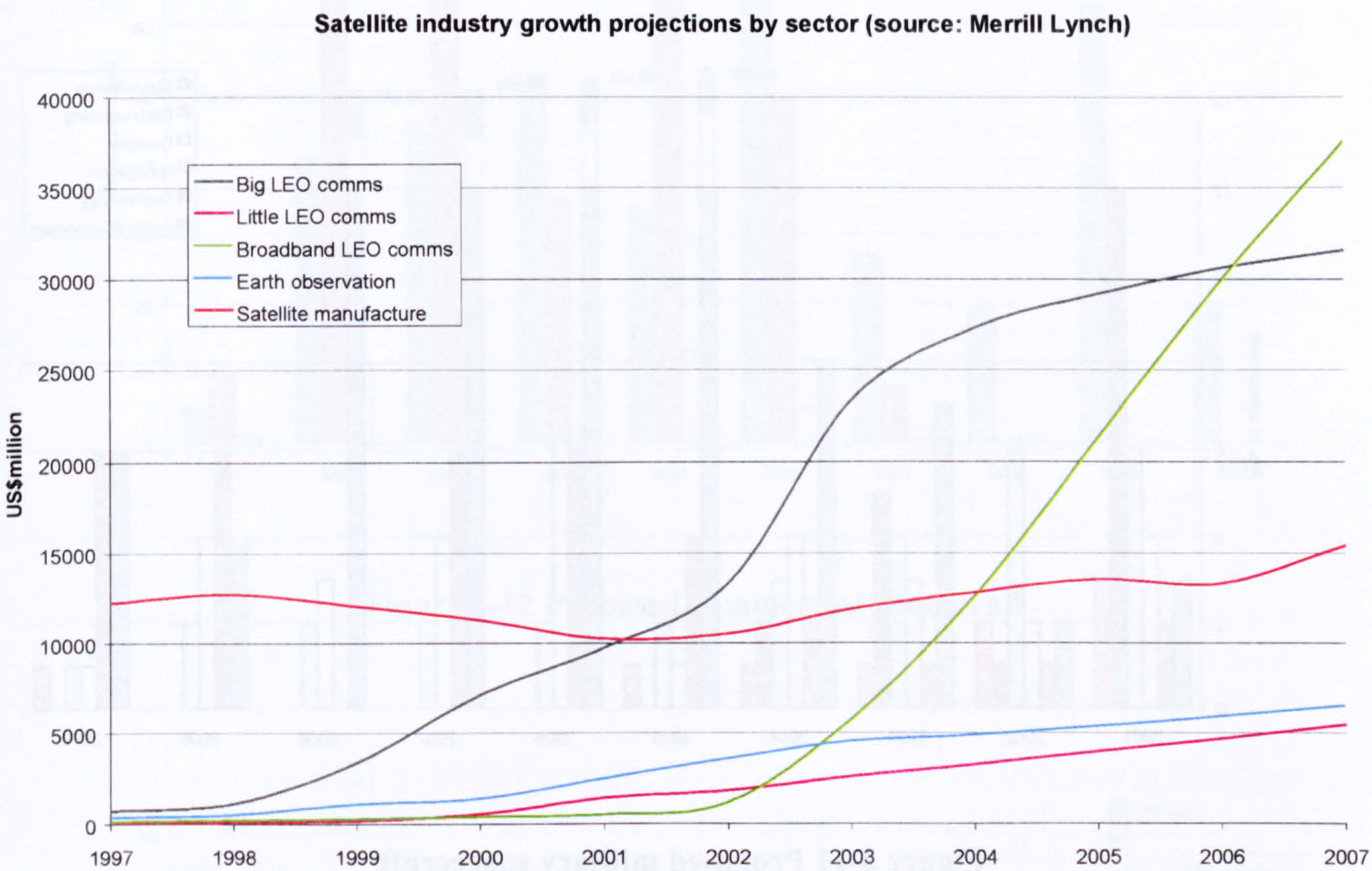

Figure 3-10 Merrill Lynch space industry growth projections

These projections seem somewhat contradictory to the AST forecast, especially in the broadband communications sector. However, this is an indication of expected industry turnover, rather than actual number of spacecraft flown, so it is possible that a small number of broadband satellites could be launched, yet prove very profitable. This is further implied by the fairly steady levels expected in satellite manufacturing.

Figure 3-10 also shows the fairly low, steady level of turnover in the Earth observation sector that was also indicated by the AST forecasts, but it also predicts a similar scenario for Little LEO communications. This level of turnover would be more consistent with a low number of small, independent projects using only a few spacecraft, rather than the deployment of larger constellations implied by the AST forecasts.

However, as it is this more "one-off" type of utilisation that would probably be of most interest for the proposed platform, the presence or absence of Little LEO constellations should not cause too much concern.

3.5.2.2 Proposed future projects

Another method of identifying trends in space use is to look at proposed spacecraft. Although some, perhaps many, of these projects will not be carried to completion, the distribution of different missions may give some useful indicators. Proposed military, commercial and civil spacecraft figures are shown in Figure 3-11, 12 & 13.[1,15]

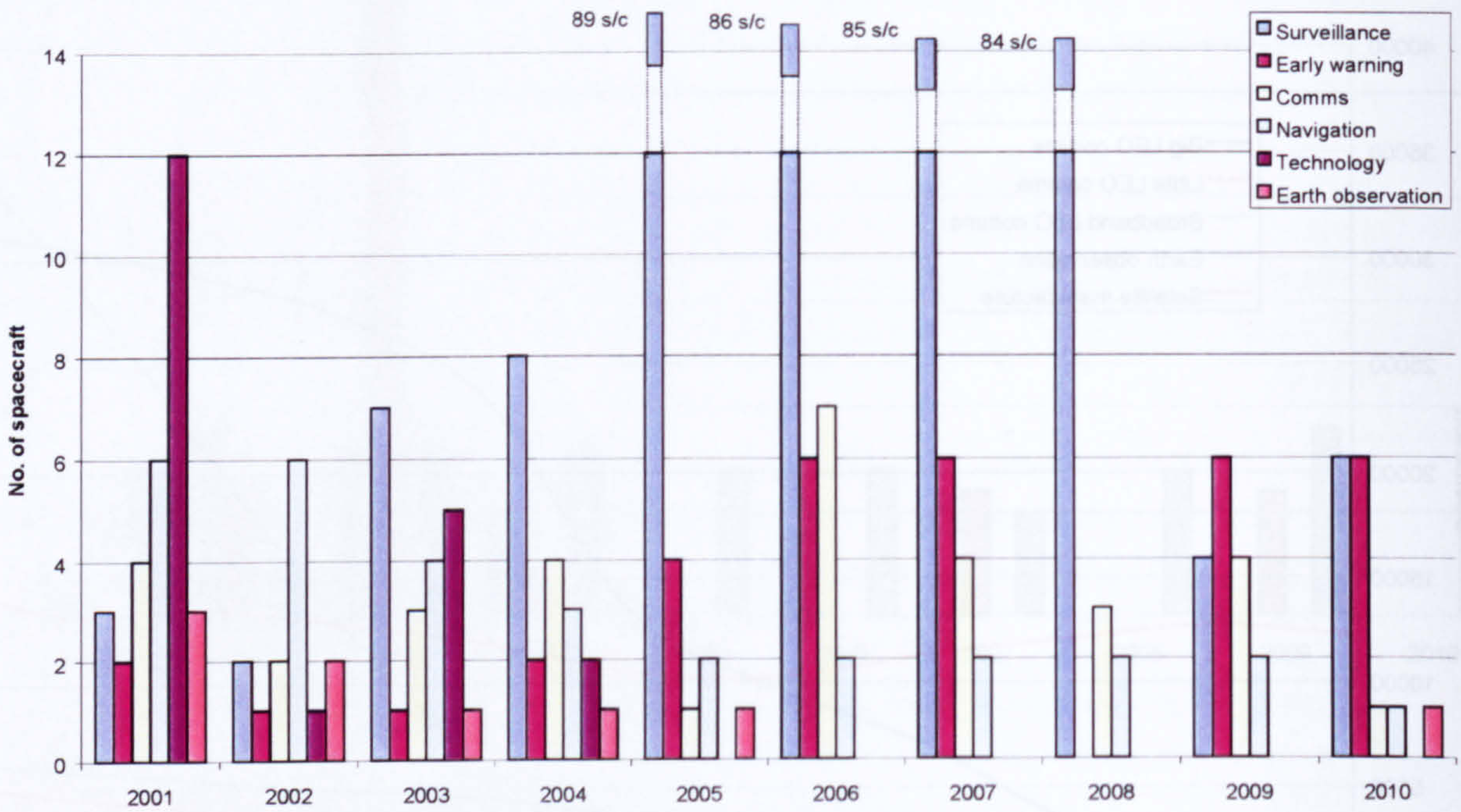

Figure 3-11 Proposed military spacecraft

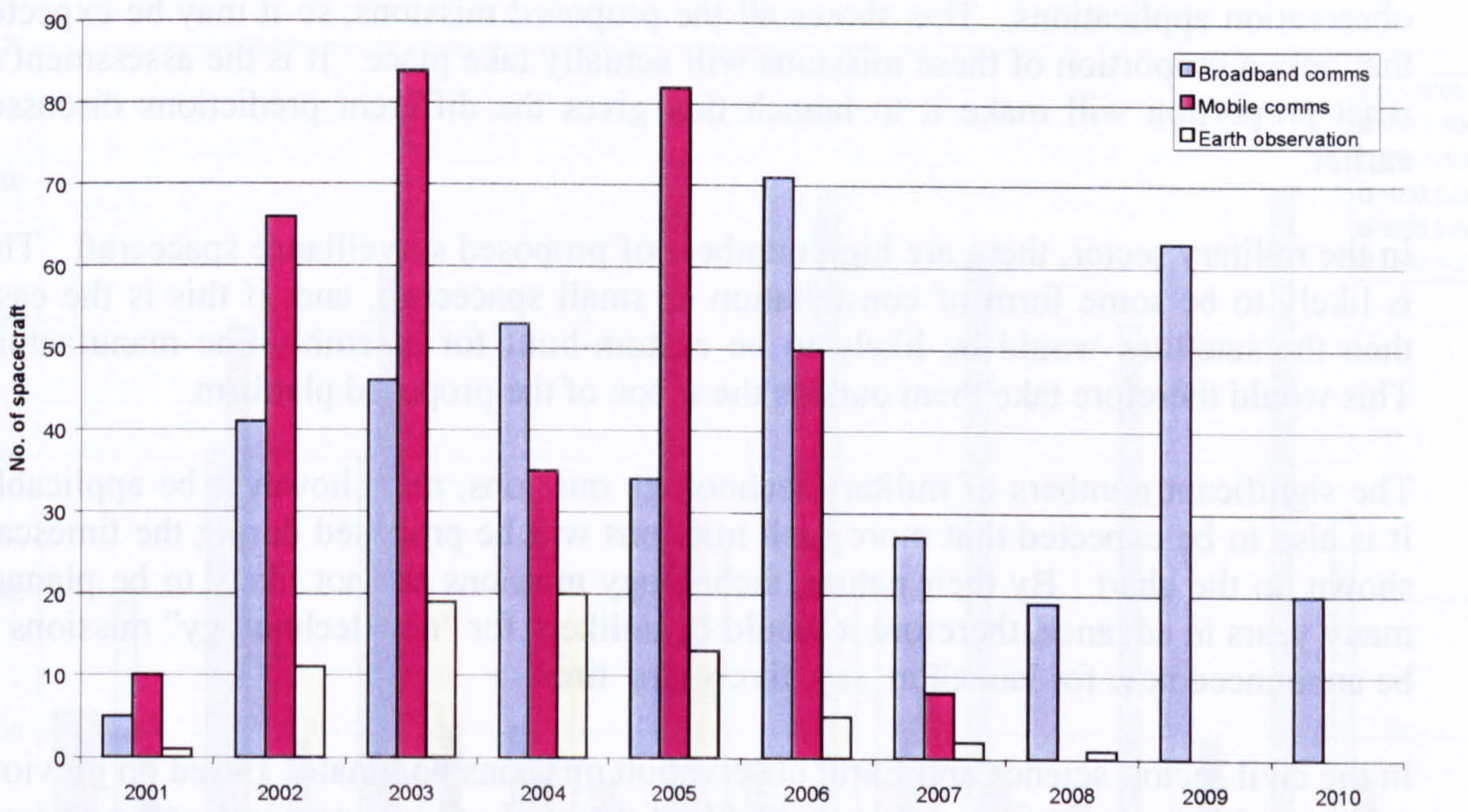

Figure 3-12 Proposed commercial spacecraft

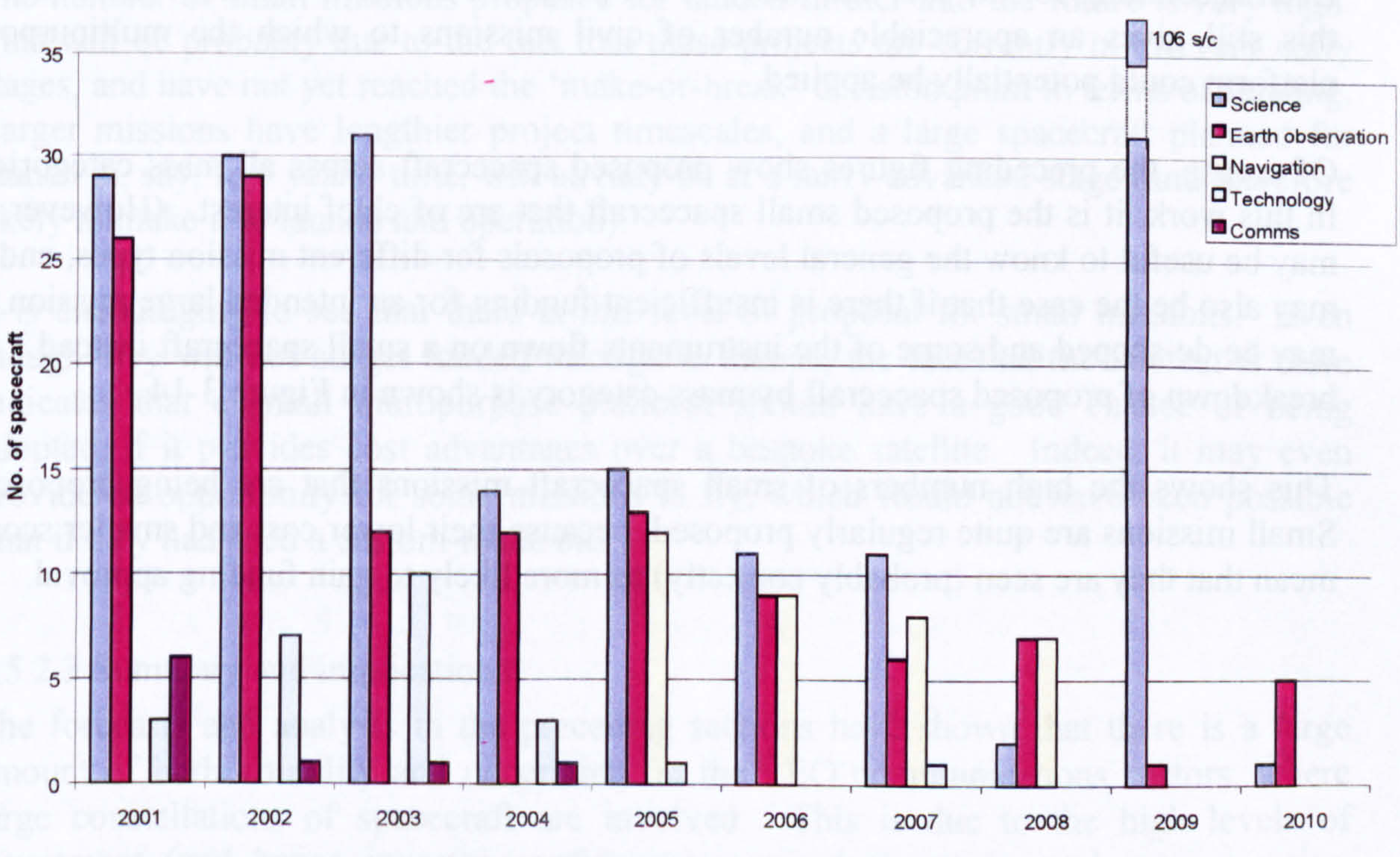

Figure 3-13 Proposed civil spacecraft

These figures again show a similar picture for the commercial sector, with the high interest in broadband and mobile communications, and a lower level for Earth observation applications. This shows all the proposed missions, so it may be expected that only a proportion of these missions will actually take place. It is the assessment of what proportion will make it to launch that gives the different predictions discussed earlier.

In the military sector, there are high numbers of proposed surveillance spacecraft. This is likely to be some form of constellation of small spacecraft, and, if this is the case, then the satellites would be likely to be custom-built for assembly-line manufacture. This would therefore take them outside the scope of the proposed platform.

The significant numbers of military technology missions, may, however be applicable. It is also to be expected that more such missions will be proposed during the timescale shown on the chart. By their nature, technology missions are not likely to be planned many years in advance, therefore it would be unlikely for "new technology" missions to be announced now for launch in, say, five years' time.

In the civil sector, science and Earth observation missions dominate. Based on previous missions, it may be expected that around half the civil science missions and perhaps a third of civil Earth observation missions use small spacecraft (see section 3.3.1.1). There are also a number of civil technology missions proposed, and, as mentioned previously for the military sector, new technology missions are likely to be continually proposed for quick development. Again based on previous missions, a large proportion of technology missions utilise small spacecraft.

Even accounting for the fact that many of the proposed missions will not reach launch, this still gives an appreciable number of civil missions to which the multipurpose platform could potentially be applied.

Of course, the preceding figures show proposed spacecraft across all mass categories. In this work, it is the proposed small spacecraft that are of chief interest. (However, it may be useful to know the general levels of proposals for different mission types, and it may also be the case that if there is insufficient funding for an intended large mission, it may be de-scoped and some of the instruments flown on a small spacecraft instead.) A breakdown of proposed spacecraft by mass category is shown in Figure 3-14.

This shows the high numbers of small spacecraft missions that are being proposed. Small missions are quite regularly proposed, because their lower cost and smaller scope mean that they are seen (probably correctly) as more likely to gain funding approval.

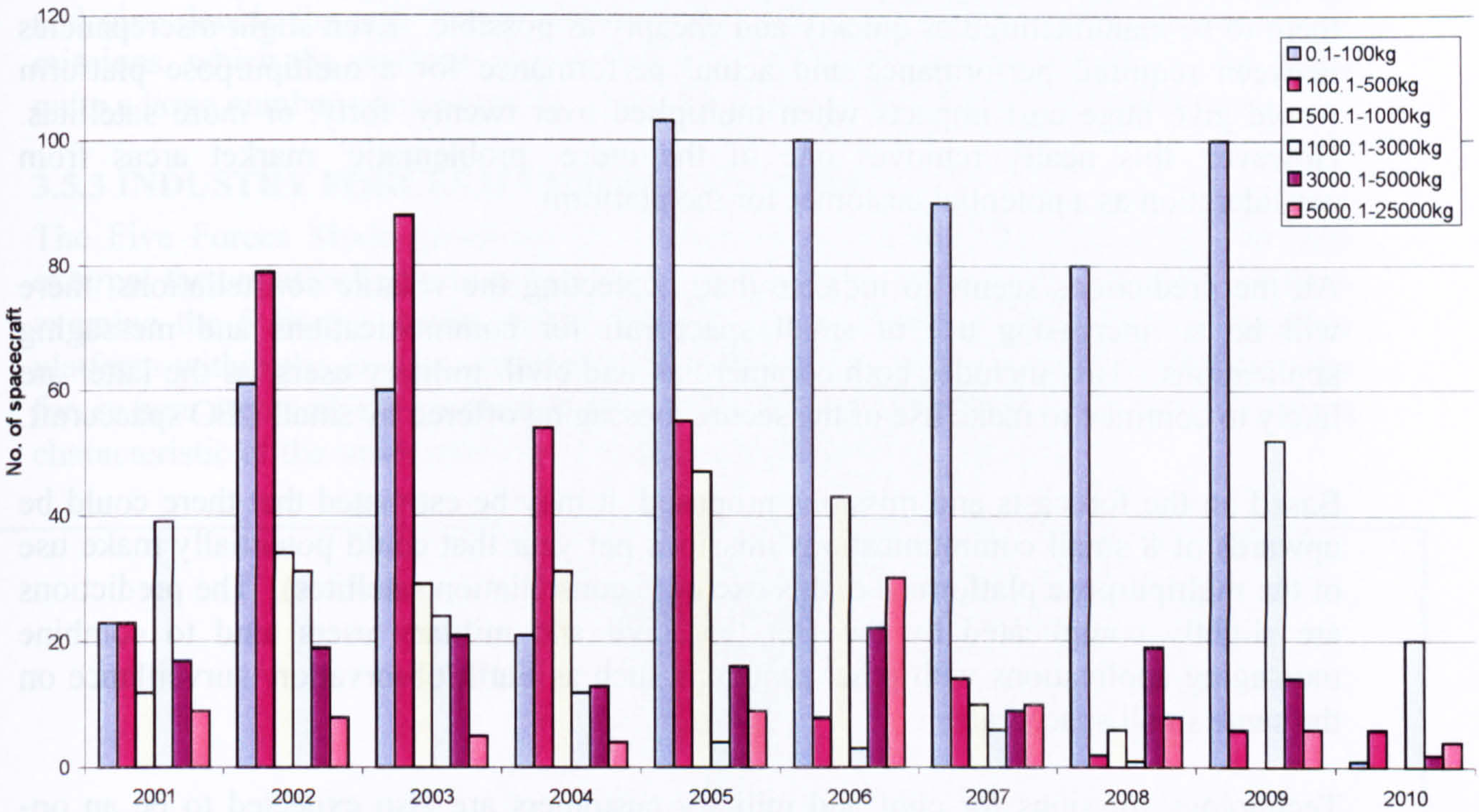

Figure 3-14 Proposed spacecraft by mass category[15]

The number of small missions proposed for launch further into the future is very high. This will be probably due to the fact that these projects are currently in the very early stages, and have not yet reached the 'make-or-break' decision point in terms of funding. Larger missions have lengthier project timescales, and a large spacecraft planned for launch in, say, five years' time, will already be at a fairly advanced stage (and therefore likely to make it to launch and operation).

It is encouraging to see that there is this level of proposal for small missions. Even though they will not all get funded through to launch, the fact that the interest is there indicates that a small multipurpose platform should have a good chance of being adopted, if it provides cost advantages over a bespoke satellite. Indeed, it may even provide an opportunity for some missions to fly, which would not have been possible than if they had used a custom-made bus.

3.5.2.3 Summary and implications

The forecasts and analysis in the preceding sections have shown that there is a large amount of both volatility and uncertainty in the LEO communications sectors, where large constellations of spacecraft are involved. This is due to the high levels of investment (and hence investor confidence) required to set up such an enterprise. However, the fact that these systems use so many spacecraft actually takes them outside the scope of the multipurpose platform.

Where large numbers of identical satellites are to be produced, they will almost certainly be custom-designed precisely for both their in-orbit application, and to enable them to be manufactured as quickly and cheaply as possible. Even slight discrepancies between required performance and actual performance for a multipurpose platform would give huge cost impacts when multiplied over twenty, forty, or more satellites. However, this neatly removes one of the more 'problematic' market areas from consideration as a potential customer for the platform.

All the predictions seems to indicate that, neglecting the volatile constellations, there will be an increasing use of small spacecraft for communications and messaging applications. This includes both commercial, and civil/ military users, as the latter are likely to continue to make use of the secure messaging offered by small LEO spacecraft.

Based on the forecasts and missions proposed, it may be estimated that there could be upwards of 8 small communications missions per year that could potentially make use of the multipurpose platform (i.e. this excludes constellation satellites). The predictions are slightly complicated by the fact that civil and military users tend to combine messaging applications with other functions such as Earth observation/ surveillance on the same small spacecraft.

Technology missions for civil and military customers are also expected to be an on-going, and perhaps increasing, opportunity for the use of a multipurpose small platform. Based on the forecasts and proposed missions, it may be estimated that there could be upwards of 5 small technology missions per year that could potentially make use of the multipurpose platform. (This covers both civil and military sectors). Numbers of technology missions may increase due to military requirements for new space-based infrastructure to respond to the current world terrorism threat. Countries may also use such missions to encourage domestic high technology industries and promote their international competitiveness.

Civil science missions are expected to be a continuing application area, and to include the use of small satellites for perhaps half of their number. Based on the forecasts and proposed missions, and on the previous launch rates, it may be estimated there may be upwards of 7 small science missions per year that could potentially make use of the multipurpose platform. The scientific sector is not expected to show any particular growth or volatility, as it is not concerned with financial return. Overall levels of spending and therefore mission numbers are likely to generally be dependent on the strengths of domestic economies.

Finally, Earth observation offers perhaps the strongest growth area, as 'non-space' countries use small observation missions to attain space presence, and commercial enterprises make use of the increasingly capable small satellites to perform imaging for fisheries and agriculture. Based on the forecasts and proposed missions, it may be expected that there could be growth to around 8-10 civil, 10-12 commercial, and 5-7 military Earth observation missions per year, that could potentially make use of the multipurpose platform.

The above projections are annual worldwide levels, and assume that most small missions (that do not use the constellation, mass-production philosophy) are potential targets for use of the multipurpose platform. Even if the estimates are over-optimistic and give double the real "target market", this would still leave a potential number of missions, which the platform may be used for, of more than 20 per year. This gives quite a large number of missions for which to compete with other manufacturers.

3.5.3 INDUSTRY FORCES (5 FORCES) ANALYSIS

The Five Forces Model proposed by Porter[13] is a useful tool to identify the key external factors affecting a company, within its industry sector. The following sections examine the 5 Forces model in the context of a company offering a small satellite platform within the civil, commercial and military markets previously described. The forces here are much the same regardless of the market sector being targeted; they are a characteristic of the small satellite manufacturing industry in general.

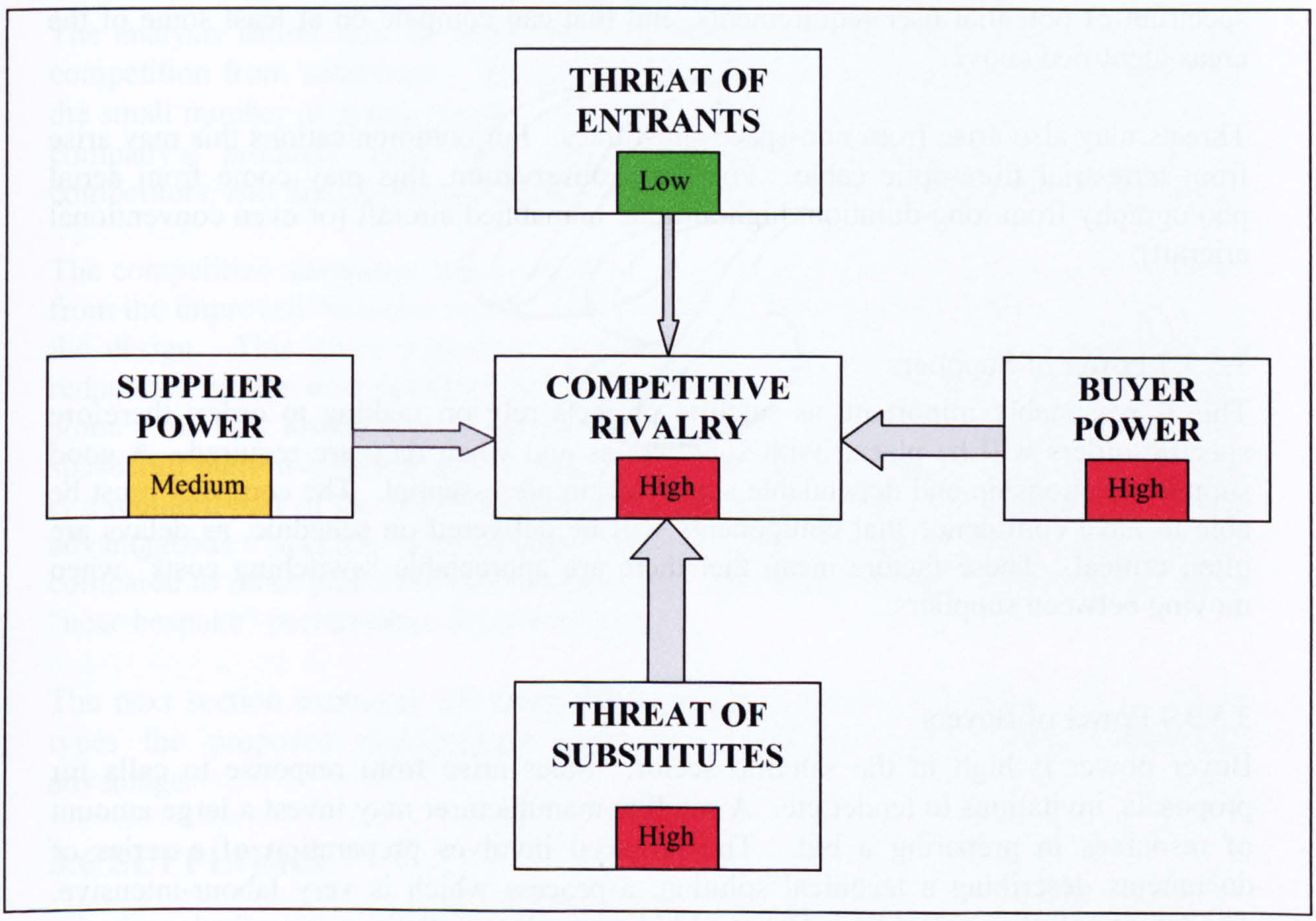

Figure 3-15 Porter's Five Forces and their relative importance for a small satellite manufacturer

3.5.3.1 Threat of Entry

This is not one of the primary threats in this situation, as there are significant entry barriers in the space industry. Companies must build up expertise, contacts, and customer confidence before they can take on the existing players. Sudden new entrants to the small satellite industry are unlikely; any threat of this type would be likely to come from an existing participant in the space industry choosing to diversify into the sector.

3.5.3.2 Threat of Substitutes

This is a strong threat. A customer for a satellite platform is generally buying a capability; either to provide some overall function to a user, e.g. produce images of the Earth, or to provide a support function for their payload instrument. There are quite a large number of suppliers, considering the relatively small number of satellites produced. Therefore there may be a number of satellite platforms offered by different manufacturers that could meet the technical specifications. These substitutes will then compete on cost, schedule, flexibility, political considerations and flight-proven heritage, as well as the acceptability and "good fit" of their technical solution. This threat forces a company to offer a product that is both capable of meeting a wide spectrum of potential user requirements, and that can compete on at least some of the areas identified above.

Threats may also arise from non-space substitutes. For communications this may arise from terrestrial fibre-optic cable. For Earth observation, this may come from aerial photography from long-duration, high-altitude unmanned aircraft (or even conventional aricraft).

3.5.3.3 Power of Suppliers

This is reasonably important, as satellite projects rely on making to order; therefore specific orders will be placed with suppliers as and when they are required. A good supplier relationship and dependable supply chain are essential. The company must be able to have confidence that components will be delivered on schedule, as delays are often critical. These factors mean that there are appreciable "switching costs" when moving between suppliers.

3.5.3.4 Power of Buyers

Buyer power is high in the satellite sector. Sales arise from response to calls for proposals, invitations to tender etc. A satellite manufacturer may invest a large amount of resources in preparing a bid. The proposal involves preparation of a series of documents describing a technical solution, a process which is very labour-intensive. The buyer will then select their favoured bid, according to their own particular criteria; these may be purely technical, but more often will include the political and commercial factors mentioned previously.

There are a small number of buyers, and a small number of individual contracts "on offer" in any period. The individual projects are of high value, and to win or lose a contract may have the power to make or break a small company. Companies must ensure that they remain up to date with any new contracts which may be offered, and make efforts to maintain links with potential buyers. These activities often take place at worldwide exhibitions, conferences and workshops dedicated to the industry.

There is a trade-off between the cost to the company of preparing a bid, and the likelihood of winning the contract. The company must decide whether it is worth its while to respond to a contract proposal, which can be a difficult decision.

3.5.3.5 Competitive rivalry

Competition is high in this industry, because of the competitive bidding process for projects and the need to have a technically and programmatically superior product. In order to compete, margins may be cut very low and ambitious scheduling targets made. In some instances, a contract with zero, or even negative, margin may be taken on, if it is expected to lead to future competitive advantage.

3.5.3.6 Summary and implications of competitive environment

The analysis shows that the main forces at work in the competitive environment are competition from substitutes. Because of the relatively high number of suppliers, for the small number of platforms procured each year, it is obviously very important that a company's products must have some competitive advantage over those of its competitors, and that the buyer's requirements must be well researched and understood.

The competitive advantage that is expected to be offered by a modular platform, arises from the improved "mission match" that can be obtained from the ability to reconfigure the design. This gives a potential performance advantage. The expected schedule reductions confer cost benefits, but the platform is not foreseen to compete on cost alone – the very lowest cost platforms are largely composed of fairly low-performance spacecraft built in-house at universities or other institutions, where the design and build is a required part of the whole experience. Rather, the platform is expected to give an advantageous "specific performance" i.e. it should give increased performance compared to other platforms of a similar cost. The multipurpose platform aims to give "near-bespoke" performance for lower price.

The next section examines the competing manufacturers in the market, and to which types the proposed multipurpose platform programme could confer competitive advantage.

3.6 SUPPLIERS

3.6.1 INTRODUCTION

A summary of the small satellite platforms available, and their technical specifications has been given in Chapter 2. This section examines the manufacturers of these platforms - the major players on the supply side of the small satellite industry. The key manufacturer types are then characterised, and analysed in terms of their strengths and weaknesses, and the types of opportunities and threats that they are subject to.

The original intention of this work was to research the competitors of SIL, and show how the generic small satellite programme would allow SIL to gain advantage over those competitors. The revised intention is to identify what types of manufacturer would be most likely to benefit from adopting the proposed programme.

3.6.2 OVERVIEW OF SMALL SATELLITE MANUFACTURERS – THE SUPPLY SIDE

The characteristics of space platform suppliers are very diverse, and range from companies of only a hundred staff, to the giant multinational aerospace and defence corporations. A summary table of companies who produce spacecraft is given in Appendix C. It should be noted that there are other manufacturers, who may supply domestic markets, and for which information has not been available (for example in China, India etc). However, it is believed that the table is a fair representation of companies supplying the international commercial satellite market.

The following sections give an overview of the main structure of the small satellite supply side.

3.6.2.1 Europe

The UK makes a suitable starting point, as it is the base for one of the pioneers of low-cost small spacecraft, Surrey Satellite Technology Limited (SSTL). SSTL was formed in 1985, as a company wholly owned by the University of Surrey. It is now an independent commercial company of over 100 staff, which has had considerable success in developing and marketing low cost small satellites, particularly for technology transfer programmes with Asian-Pacific countries such as Malaysia and Thailand.

The other main participant in the UK space industry is Astrium, a joint venture owned by EADS (European Aeronautic Defence and Space Company). It is the leading European satellite manufacturer, and has sites in France, Germany and Spain, as well as the UK. Astrium operates in a different sector to SSTL; although it does offer smaller satellites, it primarily produces large commercial communications, Earth observation, and scientific spacecraft. The company is also heavily involved in the launch vehicle sector.

The only other satellite manufacturer in the UK is the former government Defence Evaluation and Research Agency (DERA), now privatised as QinetiQ, which has launched several small satellites. These are in-house technology demonstration projects, without an external customer, and QinetiQ is not currently commercially offering spacecraft platforms. This may potentially change, however, now that the organisation is a commercial enterprise. The company is currently working on a small satellite for Earth imaging, although this spacecraft, TOPSAT, utilises an SSTL microsatellite bus.

In Germany, OHB-System produces small satellites for communications, science and Earth observation. They also supply subsystems and sensors, and can arrange launch services via a Russian affiliate. This 120-employee company has already designed and manufactured several small spacecraft since 1994. It is part of a larger group of companies, which include a commercial telecommunications company. OHB is currently collaborating with the US on a further satellite project.

Another German company, Kayser-Threde, has a small division targeted at the small satellite market, and reportedly has several orders. TERMA Elektronik, a Danish organisation, also produces comparable products to SIL. These companies probably have a similar strategy to SIL, though Kayser-Threde has a broader manufacturing scope, having the capability to produce large structures.

The Swedish Space Corporation has a history of producing small satellites and subsystems, and reports several more planned spacecraft. Their strategy is very much one of low cost, build in-house, and, as a government-owned company, they obtain much of their work from Swedish national space projects.

The Belgian company Verhaert produced the PROBA satellite for the European Space Agency. This design may become used as a commercially-available small satellite platform. The company has over 150 employees, but covers other technology areas as well as space systems.

Italy has both the huge Alenia Aerospazio organisation, which has nearly 3000 employees, and the smaller Carlo Gavazzio Space, with less than 150 people. Alenia is prime contractor for Italian Space Agency and many ESA missions, and its products range from instruments and subsystems, through small satellites, to Space Station modules. Carlo Gavazzio Space produces small spacecraft, payloads and ground stations. Its first small satellite was launched in 2000.

France's Alcatel Space employs over 6000 people, and has produced many spacecraft over its thirty year life, including satellites for ESA, CNES, Eutelsat, and Intelsat. It is also producing small spacecraft in association with CNES. Alcatel Space is itself a part of the even larger, 130,000-employee Alcatel group.

3.6.2.2 The USA

The main low-cost small satellite provider in the US sector is AeroAstro. This small company specialises in bringing down the cost of spacecraft, through innovation, wide expertise and close partnership with its supply chain and other related organisations. It has been very successful in its efforts to miniaturise spacecraft and make them affordable, and has developed and built three microsatellites. Their strategy, however, is to build very small spacecraft, so they cannot currently compete where a slightly larger satellite is needed.

The larger Spectrum Astro also specialises in small spacecraft and systems, and has produced many small platforms for NASA and DoD programmes, including Deep Space 1, RHESSI, MightySat II and the MSTI series. It also offers a range of standard small satellite buses, and satellite subsystems.

Another smaller company is Swales Aerospace, which provides engineering support, design and build for spacecraft.

As with Europe, the US space industry is dominated by a few very large players. The USA is responsible for a large proportion of the world's space industry turnover and activity. The companies and organisations involved are too numerous to comprehensively list here, however, the following are some of the key participants:

- NASA – the US government agency whose strategy covers space research and development, collaboration with industry and other agencies, technology transfer and a variety of manned and unmanned space missions. NASA's procurement procedures preclude bids from non-US prime contractors.
- Boeing – the largest aerospace corporation in the world, main industrial partner to NASA and the US Department of Defense.
- Lockheed Martin – large aerospace company which produces space systems, missiles and high technology products for commercial and military customers.
- Ball Aerospace – manufactures complete spacecraft systems, subsystems and sensors, commercially and for NASA.
- TRW – provides scientific, communications and military spacecraft, and subsystems.
- Orbital Sciences Corp – large aerospace corporation with a wide capability in spacecraft design and manufacture.
- Space Systems Loral – produces large numbers of communications spacecraft for constellation applications.

These companies have similar strategies to the large organisations in Europe; large spacecraft, large high cost projects, heavy investment. However, some of them are also successfully competing in the small, low-cost satellite sector. In many cases the customer is NASA, but commercial organisations are also utilising their experience and facilities, particularly for communications spacecraft.

The military is also a considerable customer in this market, and also generally procures from domestic suppliers.

3.6.2.3 Other

Other countries such as Israel, Japan, Korea, Argentina and South Africa have fairly recently entered the space industry. Russia and the Ukraine have been participants since the start of the space age. Because of the limited information available, it is hard to estimate the types of strategies adopted by the companies operating in these countries. However, it is known that NEC of Japan and IAI of Israel are both offering commercial small satellites; IAI has already launched four spacecraft. Russian manufacturers produce quite large numbers of spacecraft for the domestic market, particularly communications and military satellites, but do not appear to be marketing platforms to international customers. (Launch services are, however, being quite heavily marketed internationally).

Stellenbosch University in South Africa has produced a small satellite and also supplies components, particularly extendible boom structures. The Argentinean company

INVAP is producing small spacecraft for their national space programme. The Russian organisations presently seem more focussed on larger spacecraft or military satellites.

It will be worthwhile maintaining a watch on these other countries, to see how their strategies are developing. Japan in particular may become an important contender in the small satellite industry, given their strength in technology innovation and miniaturisation, and also that they have developed their own launch capability.

3.6.3 CHARACTERISATION OF MANUFACTURER TYPES

This gives an overview of the range of sizes and types of suppliers. From the survey performed, some general groups of supplier types are identified, and characterised by SWOT (strengths, weaknesses, opportunities, threats) analyses. The main types identified are:

- Large, multidisciplinary aerospace companies – e.g. Astrium (EADS), Boeing
- Large space subsidiaries – e.g. Alcatel Space, Mitsubishi Electric space division
- Large-to-medium-sized specialists – e.g. Spectrum Astro
- Small space subsidiaries – e.g. Verhaert, Kayser-Threde
- Small specialists – e.g. SSTL, AeroAstro

Each of these types is now characterised, to identify which types would benefit from offering the proposed platform and programme.

3.6.3.1 Large multidisciplinary aerospace companies

These are the largest aerospace organisations, with employees numbering in the thousands, and they offer a wide range of products and services. In the case of Boeing, this ranges from aeroplanes to launchers as well as spacecraft. These large companies are generally publicly-traded, and have many different sites in different countries. Key characteristics of this type of organisation are shown in the SWOT (Strengths, Weaknesses, Opportunities and Threats) analysis in Table 3-9.

Strengths:	**Weaknesses:**
Experience & heritage (generally) Large resource pool (staff & facilities) Stability (business not totally dependent on single projects or markets) Greater financial resources Customer confidence	Complex internal structure Greater levels of bureaucracy Resistance to change Potentially less efficient due to size (therefore less likely to take on marginal small projects) High overheads High cost products

Opportunities:	**Threats:**
Investment to research, develop and mature new technologies Entry into new markets at lower risk (risk can be absorbed by the rest of the organisation) Development of new products Leveraging techniques/activities across different disciplines	Major changes to overall target markets Restrictions in technology transfer to different countries Competition from other large companies e.g. for large defence contracts Lack of innovation (i.e. becoming "stuck in their ways")

Table 3-9 SWOT analysis for "large multidisciplinary" aerospace companies

Suitability for the proposed small satellite programme

This type of company would have the necessary financial strength to undertake a programme requiring up-front investment, and would be large enough to absorb a degree of risk if future profitability is likely. However, the spacecraft could be targeted at too much of a "budget market" to be of interest to the largest suppliers.

The facilities and infrastructure of the organisation may also be too geared towards large projects and spacecraft, for a small satellite programme to be desirable. If there are only, say, facilities for the assembly and testing of spacecraft of order 5 tonnes, then the overheads for the assembly and testing of a 500kg may be likely to be uneconomic. An answer may be to set up a new division dedicated to small spacecraft, with dedicated facilities and procedures. However, this adds to the up-front investment cost.

It is therefore expected that this type of organisation would only be likely to adopt the proposed smallsat programme if:

- Existing facilities/ infrastructure are suitable for small satellite production

Or

- The programme is expected to be sufficiently profitable that the building of a whole new facility is economic

3.6.3.2 Large space subsidiaries

These form the "space division" of a larger, non-aerospace parent firm, but are still large organisations in their own right. The parent firm is often involved in related areas such as technology, communications or other engineering. In the case of Alcatel Space, the parent firm, Alcatel, is the world's leading supplier of telecommunications infrastructures. The space division is one of four segments of Alcatel, the others being Networks, Optics and e-Business, all of which combine to give Alcatel an 'end-to-end' capability in telecommunications. However, the space division benefits from the group's large overall R&D expenditure (nearly 3 billion Euros in 2001), and also offers spacecraft platforms for applications other than communications, including science and Earth observation.

The key characteristics of this type of organisation are largely the same as for the large aerospace companies, but the parent company may be even larger, and the areas covered by the rest of the parent company may vary widely. This can further enhance the strengths arising from the large resource pool, and the leveraging of techniques and activities from different disciplines. Further strengths and opportunities are obtained from the often larger and more well-distributed portfolio of customers of the parent company.

Suitability for the proposed small satellite programme

The arguments here are much the same as those for the large multidisciplinary aerospace companies, described previously. However, it may be easier to set up the small satellite programme as a separate division, and keep it separate from the larger space projects. This may be easier to implement in an organisation that is already divided into different functional areas.

3.6.3.3 Medium-sized specialists

These are the companies that are dedicated to one main area – producing satellite platforms and subsystems, and providing supporting services. There are not many larger specialists, as most of the larger companies cover other product areas. (The world small satellite market is probably not large enough to support many large specialists.) This group may employ in the region of 500-1000 staff, and have one or two main sites. Examples of this type of company are Spectrum Astro and Swales Aerospace, both of which are privately-owned enterprises. Key characteristics of this type of organisation are shown in Table 3-10.

Strengths:	**Weaknesses:**
Activities focussed onto specialist area Specialist, highly-skilled employees Reputation for main product Strong enough financial position for significant R&D investments (but less than for previous categories) Mix of large and small projects Specialist facilities	Sensitive to market fluctuations due to more specific target market
Opportunities:	**Threats:**
Become main brand for specialist products Develop "main supplier" relationships with repeat customers e.g. agencies	Changes in demand for products Competing products from other companies

Table 3-10 SWOT analysis for "medium-sized specialist" space companies

Suitability for the proposed small satellite programme

This type of company would probably be in a position to fund the up-front investment for the proposed programme, and it could employ its large pool of specialist experience and lessons learned, to enable the detailed design work required. As this type of organisation is likely to offer equipment that may be suitable for small spacecraft, much of the required technology and subsystems can be sourced from in-house, with modifications made as necessary by engineers already familiar with the specific equipment. The facilities and infrastructure would probably be appropriate for small spacecraft production, and overheads more curtailed than those of the previous company types examined.

It is therefore expected that this type of company would be potentially likely to be suitable for adoption of the small satellite programme.

3.6.3.4 Small space subsidiaries

These are small space divisions of larger parent companies. These parent companies have other interests, often, as in the case of Verhaert and Kayser-Threde, in other high technology and engineering areas. This group has fairly low numbers of staff; a few hundred for the whole parent company, of whom 100 or less are in the space division. These smaller companies will take on smaller projects, often as subcontractors rather than primes, and have only a few sites – the space segment may be at just one of the sites. Key characteristics of this type of organisation are shown in Table 3-11.

Strengths:	**Weaknesses:**
Small, focused space team, but with the greater financial back-up of the larger parent company Small enough to be able to target niche markets Lower overheads Lower-cost products	Scarcer facilities Lower manpower – less able to absorb high activity periods Lower financial strength
Opportunities:	**Threats:**
Expansion of space division Inclusion of further stages of the supply chain within the parent company Leveraging techniques/activities across different company divisions	Fluctuations/ trends in target markets (both space and those of the other company divisions) Danger of "spreading too thin" over too many different areas

Table 3-11 SWOT analysis for "small space subsidiaries"

Suitability for the proposed small satellite programme

The main potential obstacle for this type of company to the implementation of the small satellite programme, is the up-from investment and manpower required. Once implemented, the programme would probably be very appropriate to the type of business and customers with which this group is concerned. If the programme could perhaps be gradually introduced, in parallel with other on-going projects, then it could work well. It would be favourable that the employees would be likely to be familiar with the requirements of small spacecraft and lower cost techniques.

3.6.3.5 Small specialists

These are small companies, with one to two hundred employees or less, which specialise solely in small spacecraft, systems and related services. One of the best known examples of this type of company is SSTL. This group generally has a single site, and takes on smaller projects. Key characteristics of this type of organisation are shown in Table 3-12.

Strengths:	**Weaknesses:**
Specialisation Small enough to target niche markets Very low overheads Simple management structure Ability to offer low-cost products Ability to become a recognised specialist brand (e.g. UoSats) Encouragement of innovation	Very sensitive to financial risk, and fluctuations in target market Fewer resources and facilities Individual projects extremely important Sensitive to customer confidence and perceptions – especially with respect to quality and stability of the company Lower technical capabilities of spacecraft Potential difficulty to break into foreign markets (due to small size and lack of international offices)
Opportunities:	**Threats:**
Support from civil programmes to promote small enterprises Strategic alliances with universities and other institutions Demand for low-cost spacecraft from developing nations	Low-cost products offered by competing organisations

Table 3-12 SWOT analysis for "small specialist" space companies

Suitability for the proposed small satellite programme

It would probably not be likely for the smallest companies to have the necessary financial or manpower capabilities to implement the proposed type of programme. However, it may be possible for implementation to be achieved if the organisations concerned were to receive some form of assistance from government grants, or collaboration with universities etc.

3.7 CHAPTER SUMMARY

This chapter has shown that the market for small satellites covers Earth observation, science, technology demonstration and communications missions, with customers being drawn from civil, commercial and military sectors. From examination of these customers, the following utility areas and cost ranges were obtained (table repeated from Section 3.4.4):

	Civil	Commercial	Military
Communications	Low-cost domestic communications Government messaging Cost range: $2-15m	Point-to-point Store-and-forward messaging Asset-tracking Cost range: $2-15m	Low-cost secure communications Cost range: $5-20m
Earth observation	Low-cost weather satellites Disaster monitoring Resources Cost range: $15-40m	Images for fishing, agriculture industries Cost range: ~$15m	Surveillance Operations support Cost range: $15-50m
Technology	Small demonstration missions to promote domestic industry Cost range: $10-40m		Small demonstration missions to test new systems Cost range: ~$15m
Science	Low-cost scientific research Cost range: $15-40m		

The examination of the industry environment and future trends showed that, in the world market, using an "optimistic scenario" there might be more than 40 small satellite missions per year for which a commercial platform may compete. Even if this estimated number is over optimistic, and if some of the missions are removed from the target market due to political procurement preferences, for example, then it may still be expected that there may be more than 20 suitable target missions per year. Out of these target missions, the approximate distribution across the identified mission applications that may be expected is shown in Figure 3-16.

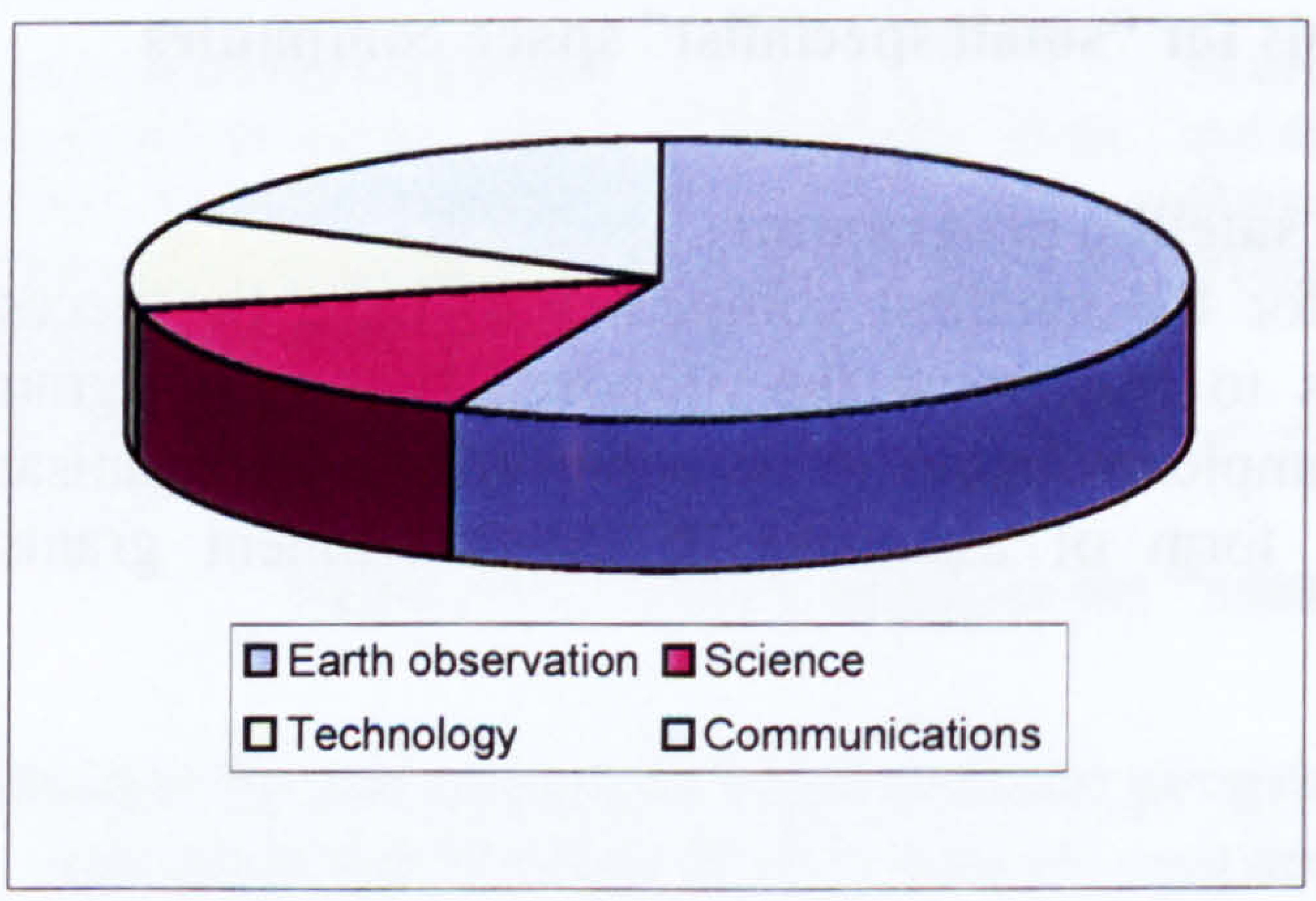

Figure 3-16 Approximate distribution of applications expected within target missions

It is expected that Earth observation and communications applications sectors will exhibit the most growth.

The relatively high proportion of expected target missions that perform Earth observation, and the anticipated growth in this sector, indicate that it is very important the multipurpose platform should effectively support this type of application. The flexibility to configure for all these types of mission is important, but if one type must "slip", it should not be Earth observation.

The proposed programme, with its emphasis on "pre-empting" customer requirements and investing heavily in a great deal of design work up-front, was shown to be best suited to the "medium-sized specialist" segment of space companies. The restricting factor for smaller companies is the level of investment required before a return is made. For very large organisations, applicability is limited by the difficulty in producing cost-effective smaller spacecraft using infrastructure and facilities designed for large projects.

Chapter 4 will now perform a detailed analysis of the technical requirements that must be met by the proposed multipurpose spacecraft, in its different configurations, if it is to compete for the markets and customers identified here.

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4 ANALYSIS AND DEFINITION OF REQUIREMENTS

4.1 INTRODUCTION

For the design of the spacecraft platform to proceed, a detailed requirements specification must be produced. This chapter analyses the technical and programmatic requirements that must be fulfilled, and addresses the implications of these requirements on the design. As the platform is intended to be multipurpose, the requirements definition process is somewhat more complex than that for a specific mission.

In most spacecraft projects, the definition of requirements has as its starting point a top-level statement of the mission objectives, usually provided by the user (customer). For example, an astronomy mission may have the objective of detecting high-energy cosmic ray bursts and determining their location to within a few arc minutes, or a remote sensing mission may have the objective of imaging the Earth's surface with a specific resolution and repeat time. Requirements are then derived from this high-level objective, becoming more detailed as payload instruments and mission scenarios are defined.

The customer will also generally impose constraints, such as schedule and cost, and may also impose other, "directed" requirements. These may be minimum-performance requirements, or requirements stemming from political considerations, such as use/avoidance of particular launchers or use of a particular supplier's equipment.

The requirements generation process for a typical space project is illustrated in Figure 4-1 on the following page. This shows the directed requirements coming from the customer, the requirements derived from the mission objectives, and the process flow towards design, development and management.

For a generic spacecraft platform, this overall process still holds, but, as the platform is being developed with no specific customer in place, there are no directed requirements as such; all requirements must be derived. This derivation must come from the anticipated requirements for the range of missions to which the platform is to be targeted. The previous chapter identified the mission and customer types that the multipurpose small satellite should be capable of satisfying. The goal in this chapter is to investigate what requirements are necessary to produce a platform that can be adapted to as many of the target missions as possible.

The approach used is firstly to investigate the particular requirements that might be encountered from different mission types. These provide an envelope of mission and payload requirements. From this, the range of requirements for all target missions can be identified, and decisions made as to how broad the scope of the multipurpose platform may be.

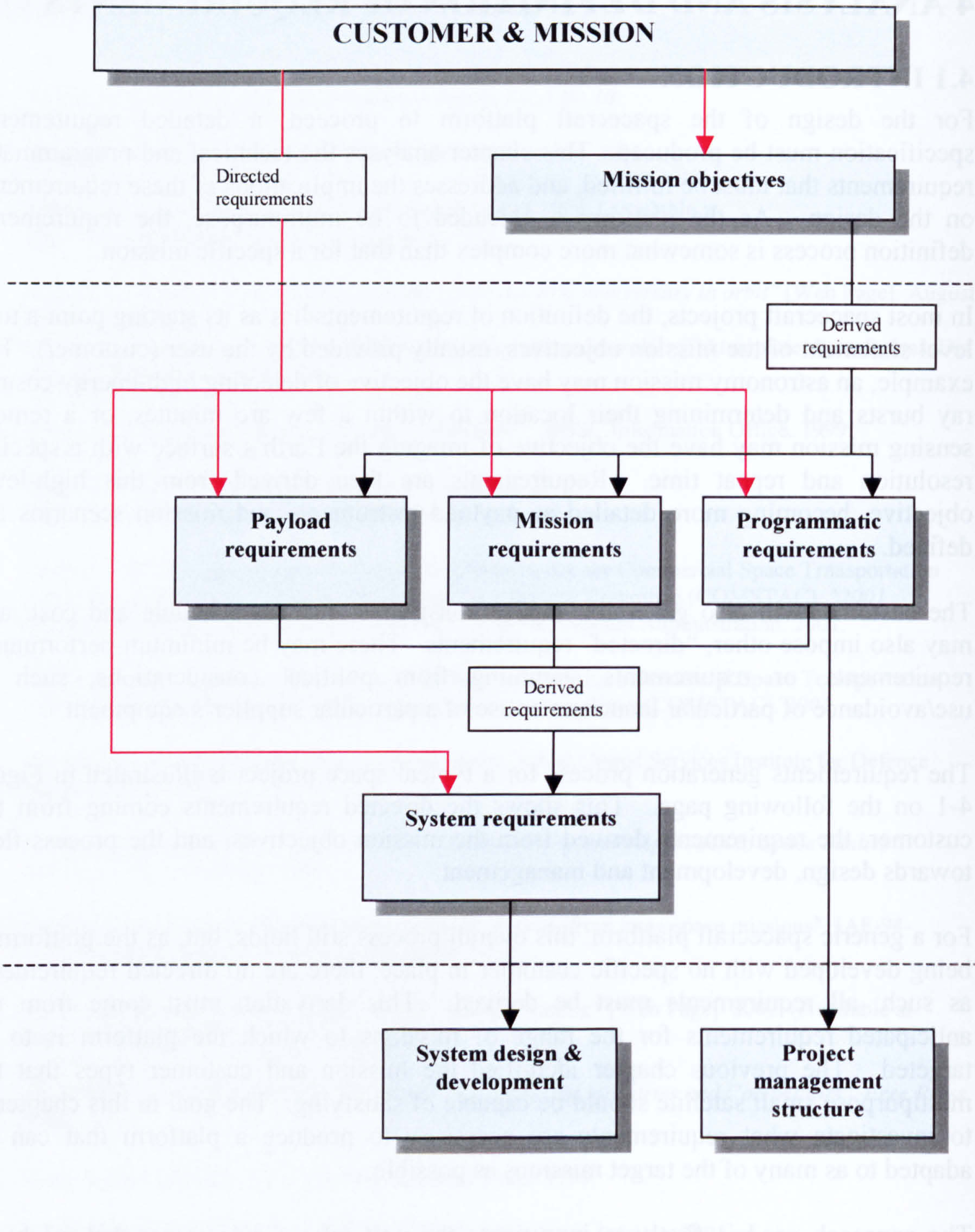

Figure 4-1 Requirements generation process for a typical space mission

The payload requirements drive the system requirements to a large degree. Figure 4-2 shows a breakdown of the system, mission, and programmatic requirements for the spacecraft. This indicates the factors that must be covered when considering the differences in requirements between mission types.

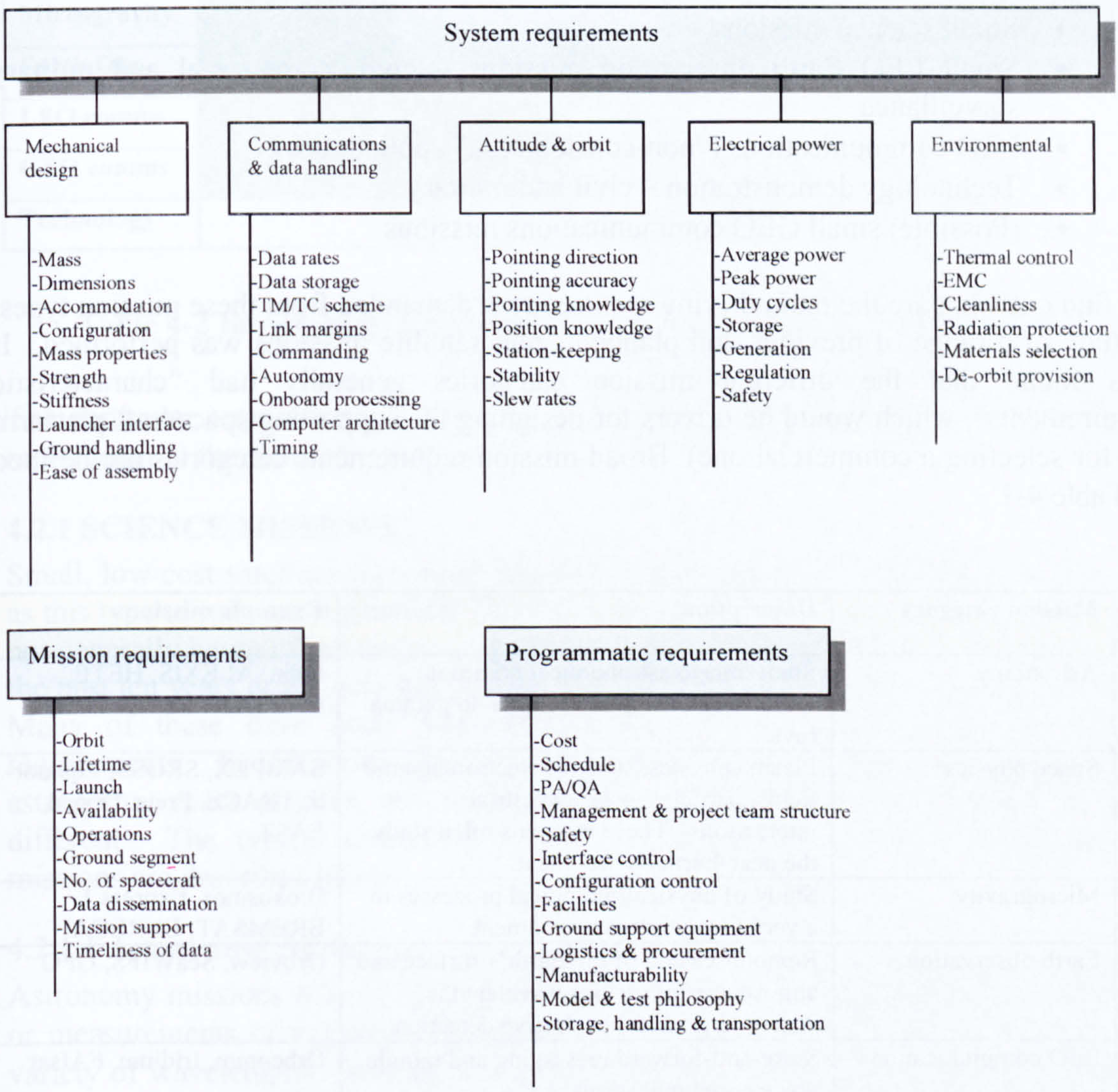

Figure 4-2 Spacecraft requirements breakdown

The next section now examines the specific requirements that may be anticipated for different mission types.

4.2 MISSION-SPECIFIC REQUIREMENTS ANALYSIS

When attempting to provide a multi-purpose platform for use by many different customers, a range of payload types must be examined, so that a typical baseline requirements profile can be built up for the various mission categories. It is obviously impossible to predict exactly what a future payload may require, but this method should give a design that is close enough to be easily tailored to a specific mission.

Chapter 3 identified the likely types of mission to which the multipurpose platform should be targeted. These are:

- Small science missions
- Small LEO Earth observation missions – civil, commercial and military surveillance
- LEO communications ("non-constellation" applications)
- Technology demonstration – civil and military
- (Possible) small GEO communications missions

To find out what are the main driving requirements demanded from these mission types, a study of a range of previous and planned small satellite missions was performed. It was found that the different mission categories generally had "characteristic requirements", which would be drivers for designing the supporting spacecraft platform (or for selecting a commercial one). Broad mission requirements categories are defined in Table 4-1.

Mission category	Description	Example missions
Astronomy	Study/image astronomical bodies in various wavelengths, from RF to gamma rays.	Odin, ALEXIS, HETE, CATSAT
Space physics	Plasma physics, study of electromagnetic fields, particles, solar-terrestrial interactions. These missions often study the near-Earth environment.	SAMPEX, SROSS, Equator-S, TRACE, Freja, Orsted, FAST
Microgravity	Study of physical/biological processes in a very low gravity environment.	Biokosmos, Express 1, BREMSAT, EURECA
Earth observation	Remote sensing of the Earth's surface and atmosphere, in various wavelengths. Active (e.g. radar) or passive detection.	Orbview, SeaWIFS, GFO
LEO communications	Store-and-forward messaging and mobile voice communications.	Orbcomm, Iridium, FAIsat
GEO communications	Broadcast services.	
Technology	Demonstration/validation of new technologies and techniques in space.	STRV, MSTI, PROBA

Table 4-1 Mission requirements categories

Although most missions will have a "basic level" for all the requirement categories identified, there are often one or two requirement areas for which a higher performance is generally needed for that particular mission type. The requirement areas of particular importance to the different mission categories are indicated in Table 4-2.

MISSION CATEGORY	MAIN REQUIREMENT AREAS				
	Attitude/orbit	**Power**	**Comms/ data handling**	**Environmental**	**Mechanical/ configuration**
Astronomy	•		•	•	•
Space physics	•				•
Microgravity	•			•	•
Earth Obs.	•		•		
LEO comms	•	•			
GEO comms	•	•			
Technology					

Table 4-2 Importance of different requirement areas by mission category

The particular requirements found for each of the mission categories are described in more detail in the following sections.

4.2.1 SCIENCE MISSIONS

Small, low cost satellites have been traditionally the preserve of scientific applications, as this type of mission is generally the most difficult to fund – science for science's sake not generally being of interest to commercial investors, as described in Chapter 3. Over the past ten years or so there has been a quite large number of small scientific missions. Many of these have been purpose-built satellites, designed around the payload requirements. In this study, the scientific mission category has been separated into astronomy, space physics, and microgravity, as the requirements for these are quite different. The typical characteristics and requirements of each of these types of missions are described below.

4.2.1.1 Astronomy/ astrophysics

Astronomy missions may be defined as those with the aim of performing observations or measurements of extra-terrestrial objects. These measurements may be made in a variety of wavelengths, from radio to gamma rays, and the targets may range from the relatively local (within the solar system) to extremely distant, weak extra-galactic objects. A selection of astronomy missions and their key characteristics are shown in Table 4-3.

Spacecraft and mission	Key characteristics to enable mission
ODIN – astronomy and aeronomy[33,34]	Inertial, high-accuracy (±15 arcsec), long-duration pointing. Capability to perform an all-sky survey. Cooling of detectors, via use of cryostat and cooling straps to spacecraft external radiator. Up to 4kbyte per sec data rate, 3Mbyte per orbit. Thermal stability and accuracy of alignment crucial for platform. Telescope must be protected from direct sunlight.
HETE – gamma ray astrophysics[13]	Excellent accuracy of knowledge of absolute time and position of the spacecraft (to give information on position of the gamma ray sources). ± 2degrees pointing accuracy, ± 0.2 degrees pointing knowledge
CATSAT – gamma ray astrophysics[26]	Large field of view free of Earth occultation and interference from the sun. Spectrometer x-ray detectors require cooling to 160K, and shielding from the sun. Altitude of 450km is a compromise between the lower backgrounds and particle dose rates of lower orbits, and the required minimum 1-year lifetime
XTE – x-ray astronomy	Rapid pointing (>6 degrees per minute), can point anywhere in the sky, accuracy <0.1degree, knowledge ~1 arcminute High data rates (transmits nearly constantly via TDRSS)
ROSAT – x-ray astrophysics	Capable of fast slewing (180degrees in ~15 minutes) Pointing accuracy of 1 arcminute, <5arcsec per second stability, jitter radius ~10 arcsec, post-facto attitude knowledge of 6arcsec, orbit 580km, 53degrees Used for staring modes and scanning modes
ISO – infrared astronomy	Pointing accuracy 5 arcsec Two 750l tanks of liquid hydrogen and liquid helium for detector cooling Lifetime 18 months Data rate 44kbps, telemetered in real-time Orbit 1000km x 70500km, 5.25 degrees (detectors were only operated during the parts of the orbit outside the van Allen belts) Star trackers part of payload module (presumably to ensure exact alignment)
SWAS – submillimetre wave astronomy[10,19,31]	Pointing accuracy 38 arcsec, stellar pointing Points at 3-5 targets per orbit 12Kbps instrument data rate Sunshade protects instrument

Table 4-3 Astronomy missions and mission-enabling characteristics

The key requirements and their impacts on the different subsystem areas are now addressed.

Mechanical design and configuration

Highly mechanically-stable platforms are required for payloads such as telescopes, and the alignment of the payload instrument onto the platform must be extremely precise. This enables high accuracy pointing. For this reason, the high-accuracy attitude sensors (usually star trackers) are sometimes mounted on the same support structure as the payload, to ensure that the instrument pointing direction is accurately known and controlled. Similarly, materials selection for supporting structures must be made such that thermoelastic deformations are avoided.

Payload dimensions may be large, for example long optical 'benches' for telescopes. Sizes and masses of some astronomy payloads are given in Table 4-4.

Payload	Power /W	Mass /kg	Size /m	Remarks
Extreme UV telescope	164	128	2.78x0.86x0.254	Data rate of 1.28Mbps
Soft X-ray camera			0.1x0.1x0.175	0.91 steradian field of view Flown on HETE
X-ray/ gamma-ray imaging spectrometer[18]	110	120	0.45 diameter, 1.7 long (inc. support structure)	Flown on HESSI Data: 10Gbits in 10mins
Cassegrain telescope[36]	35		0.3 diameter, 1.6 length	Detectors passively cooled to -65°C Flown on TRACE
Submillimetre wave telescope	59	102		Flown on SWAS
Radiometer telescope		80 (inc. electronics)	.1 diameter reflector	Flown on ODIN

Table 4-4 Characteristics of typical astronomy payloads

A wide payload field of view may be required, and the field of view must be free from occultation by any other apparatus. To enable this, and also due to their frequently large size, astronomy payloads are often mounted on top of spacecraft. A typical configuration for an astronomy satellite is shown in Figure 4-3. This shows the large telescope payload mounted on top of the spacecraft platform, and the sunshade protecting the instrument aperture.

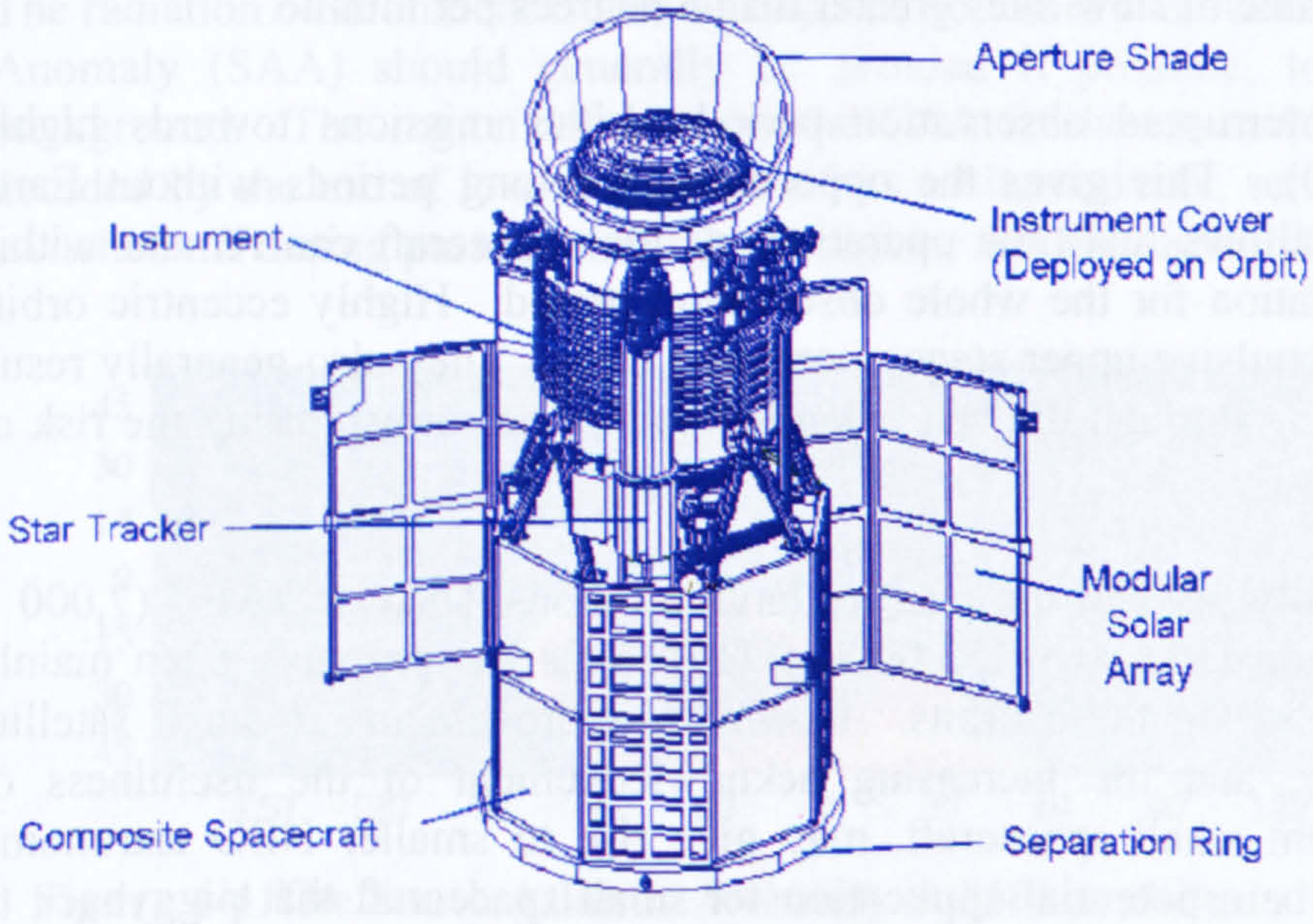

Figure 4-3 Configuration of a typical astronomy spacecraft – WIRE [Image: NASA]

Detectors for astronomy applications (particularly in the infra-red domain) often require cooling to cryogenic temperatures. This implies the carriage of liquid cryogens, such as helium, on board. Siting and design of the storage tanks must be planned such that the spacecraft does not become unbalanced as the cryogen is used up. (This is a similar case to that of propulsion fuel tanks).

Communications and data handling

Data rates may be high if the payload is in a continuous "observatory" mode (note that XTE transmits constantly via TDRSS, and the 10Gbits in 10 minutes of the HESSI spectrometer). This may require high onboard data storage capability, high downlink data rates, and/or the use of several ground stations to obtain sufficient data transfer capacity. Alternatively, it may require a high-eccentricity orbit to maintain the spacecraft in view of one ground station for long periods (see below).

Attitude and orbit

Astronomy missions are generally characterised by a need for a very stable platform, which can be accurately pointed to the observation targets. The payloads are often imagers; any disturbance of the platform will therefore degrade the images produced. This is of even greater importance where very faint sources are being studied. Pointing of the payload to the desired target must often be achieved to arcsecond accuracy, and, once pointed, the spacecraft may be required to "stare" uninterrupted at the target for long periods, to allow integration of weak signals. The spacecraft attitude is generally inertially-referenced. Anti-sun orientation often required, to avoid stray light interference into the optics. For many instruments, there is a large avoidance zone around the sun vector, which is critical for preventing damage to the sensors.

The spacecraft must generally have the ability to point to specific targets of opportunity. If the pointing must be achieved quickly (for example, where transient sources are being targeted), then the spacecraft should be capable of fast slew rates. For example of the XTE spacecraft is capable of slew rates greater than 6 degrees per minute.

Desire for long, uninterrupted observation periods drives missions towards highly eccentric orbits (HEO). This gives the opportunity for long periods without Earth occultation, and also allows real-time operations as the spacecraft can remain within view of the ground station for the whole observation period. Highly eccentric orbits generally require a propulsive upper stage, increasing costs. They also generally result in the spacecraft passing through the van Allen belts twice per orbit, raising the risk of radiation damage.

HEOs have been mainly used in the past by large missions, such as XMM (7,000 x 114,000km) and Integral (10,000 x 153,000km).[15] This has probably been mainly due to the costs of reaching these orbits. However, improvements in small satellite propulsion technology, and the increasing acknowledgement of the usefulness of science conducted from small spacecraft, may give rise to smaller HEO astronomy missions. This could be a potential application for small spacecraft that piggyback to GTO.

Generally, however, most small astronomy missions have been to Low Earth Orbit. In LEO, the spacecraft will pass into eclipse during most orbits (unless in a sun-synchronous terminator orbit). This produces thermal challenges. LEO also reduces observation efficiency due to the greater Earth occultation, and gives a much greater onboard data storage requirement, due to the more limited ground contact.

Sun-synchronous orbits do not offer direct benefits to the payload, but may offer secondary benefits to thermal and power design. A terminator orbit would give a more constant thermal environment, avoiding thermal shocks. However, higher inclination orbits may be more expensive to reach.

Power

The power requirements for astronomy payloads are not generally driving factors. However, the requirements to point at target objects for long periods may place additional requirements on the orientation of the solar arrays. However, as payloads must generally be pointed well away from the sun vector, fixed arrays may be possible, oriented in opposition to the payload pointing direction. This may not always the give ideal, normal incidence onto the arrays, but may be sufficient if articulated arrays are to be avoided. This approach can also give additional sun shielding to the payload instruments.

Environmental

The detectors used in astronomy payloads may often require cooling. For some systems, cooling by means of passive radiators may be sufficient. For others, particularly infra-red instruments, may require cryogenics. Other thermal environment considerations are the instance of jitter induced by thermal shock, when the spacecraft enters or leaves eclipse. This may be difficult to prevent, therefore implying operational constraints (see below).

The radiation environment is also of importance. In low Earth orbits, the South Atlantic Anomaly (SAA) should generally be avoided if possible, to reduce the radiation background. The region affected by the SAA is shown in Figure 4-4, and may be avoided by the use of low-inclination orbits. Alternatively, sensitive detectors may be switched off during passage through the region, and additional shielding used.

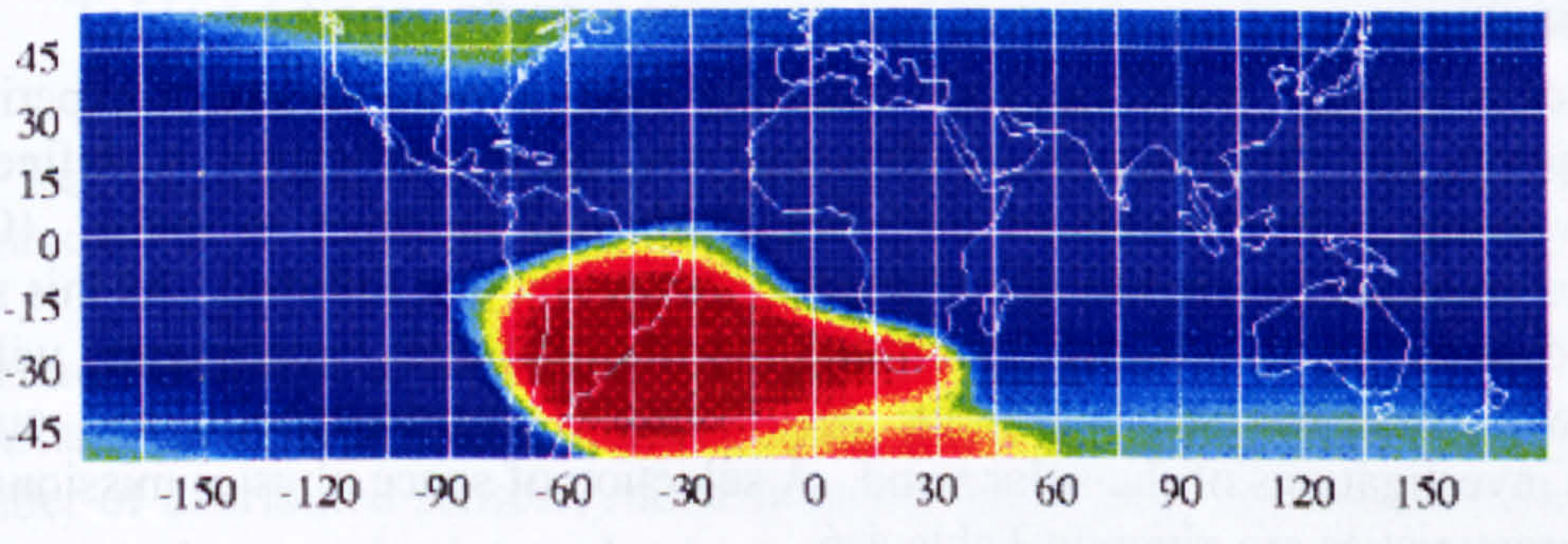

Figure 4-4 The South Atlantic Anomaly at 560km altitude, indicated by particle background data collected by the ROSAT spacecraft [Image: NASA]

Similarly, if the spacecraft is placed in an orbit that passes through the van Allen belts, (for example a HEO), then additional shielding and/or rad-hard electronic devices may be required. Again, payload instruments may be switched off during transit periods, but the bus subsystems will often be required to be operational during these times.

The optical surfaces of payload instruments must be kept extremely clean. This impacts on choice of materials, avoiding thruster plume impingement, and good cleanliness practices during the AIT process.

Mission and operations

The long observation times may often require careful scheduling around other spacecraft activities, for example momentum-dumping and orbit maintenance burns.

Summary of key requirements

The main requirements and performance capabilities that would make a platform suitable for use in an astronomy mission are shown in Table 4-5.

Requirement area	Required performance/ characteristics	Relative importance	Remarks
Payload accommodation	Large volumes required. Clear fields-of-view. Precise alignment.	High	
Data rate	Up to several Mbps.	Med-high	
Orbit	HEO desirable. LEO acceptable (usually).	Medium	
Propulsion	Very mission-dependent		Needed if HEO used.
Attitude	Inertial, avoid sun-pointing.	High	
Pointing knowledge	Up to arcsecond accuracy	High	
Pointing accuracy	Up to tens of arcseconds accuracy	High	
Manoeuvring	Slew rates up to ~10°/minute	High	
Power	100-200W total bus power	Low	
Lifetime	1-2 years.	Medium	Often limited by supply of cryogenic coolant.
Other	Detector cooling often required. Cleanliness for optics.		

Table 4-5 Astronomy mission requirements summary

4.2.1.2 Space physics

Space physics is a rather broad category, and may cover a wide variety of experiment types. However, for the purposes of this analysis, the classification is defined as missions involving measurements of particles, fields and radiation in space. (Other space physics areas are covered by the Microgravity section that follows). As this study is concerned with Earth-orbiting spacecraft, much of these experiments will be concerned with solar-terrestrial physics, for example magnetospheric and auroral physics, and investigations of the solar wind. A selection of space physics missions and their key characteristics are given in Table 4-6.

Spacecraft and mission	Key characteristics to enable mission
Astrid – magnetospheric physics[17]	Spinning spacecraft to allow sensors to scan the sky.
Oersted – solar-terrestrial physics[6,25]	Apogee 850km, perigee 400km to allow study of low altitude ionospheric currents. Coverage around noon/midnight local time to study effect of solar wind. Magnetometer mounted on a long boom to avoid spacecraft magnetic field. Star imager and magnetometer mounted on common optical bench to allow 20 arcsec attitude determination. GPS for accurate position data (required for magnetic field mapping). 5 instruments.
IMAGE – magnetospheric physics	Spinning spacecraft to allow instruments to scan. 7 instruments.
POLAR – magnetospheric physics[20]	Highly elliptical (9 Earth radii apogee, 1.8 Earth radii perigee), 86 degree orbit allows observation of polar magnetospheric regions, and both high and low altitude perspectives. Spinning spacecraft to allow instruments to scan. Long wire and rigid booms allow accurate electric and magnetic field measurements. 13 instruments.
FAST – auroral physics[1,29]	Elliptical, high inclination orbit (348x4159km, 83 degrees) (Apogee height was made as high as possible within launch constraints) Collects high-rate, high-resolution data while passing through the auroral zones (4 times per orbit). 5 instruments. Spinning at 12rpm, spin axis normal to orbit plane Attitude knowledge ~0.1 degrees Sun angle maintained at less than 60 degrees Due to the high-radiation environment in this orbit, electronic equipment is shielded by aluminium honeycomb radiation shield, rad-hard parts were used, and solar cells have extra thick cover glass. To maintain EM cleanliness, the solar array incorporates Faraday cage, the harness is designed to cancel EM fields, and the spacecraft is made electrically conductive. (<1nT DC magnetic fluctuations detected by the onboard fluxgate magnetometer.) Deployed arrays avoided due to the unwanted shadowing and wake effects they would cause.

Table 4-6 Space physics missions and mission-enabling characteristics

The key requirements and their impacts on the different subsystem areas are now addressed.

Mechanical design and configuration

On examination of previous projects, space physics missions seem more likely to fly a number of distributed sensors, rather than one main large instrument. Mass and size of some typical space physics payloads are given in Table 4-7.

Payload	Power /W	Mass /kg	Size /m	Remarks
Ion mass spectrometer	334	80	0.5x0.5x0.4	
Beam plasma	38	17	0.6x0.7x0.7	+ Two 0.7 diameter antennas
Doppler imaging interferometer	620	100	0.25x0.25 x0.25	
Electron/ ion telescope assembly	21 (peak 29)	52		Flown on SAMPEX
Hot plasma experiment	16	10		Flown on Freja
Boom structures		22		Assembly includes 2 rigid booms and 6 wire booms. Flown on Freja
Overhauser magnetometer	3	2.5		Requires mounting on a boom Flown on Orsted
Boom + mechanism		4.2	8m long Approx. 0.7x0.3x0.3 stowed	Flown on Orsted
Charged particle detectors	1.5	2		Flown on Oersted

Table 4-7 Characteristics of typical space physics payloads [38,39]

A typical configuration of a space physics spacecraft is shown in Figure 4-5 on the following page. This illustrates the use of instrument-mounting booms. Field-measurement instruments generally require deployment away from the main body of the spacecraft to avoid electromagnetic disruption from on-board systems. These booms may be rigid, or wire tensioned by the centripetal forces of a spinning spacecraft. The configuration must often be suitable to provide the necessary fields of view for scanning instruments.

The radiation environment of the spacecraft may also require sensitive electronics (e.g. the onboard computer) to be placed near the centre, where there will be most shielding. The choice of materials for the spacecraft may be influenced by requirements to provide shielding, and also to give an electrically continuous and conductive structure. This avoids the incidence of potential differences across the spacecraft when it flies through plasmas.

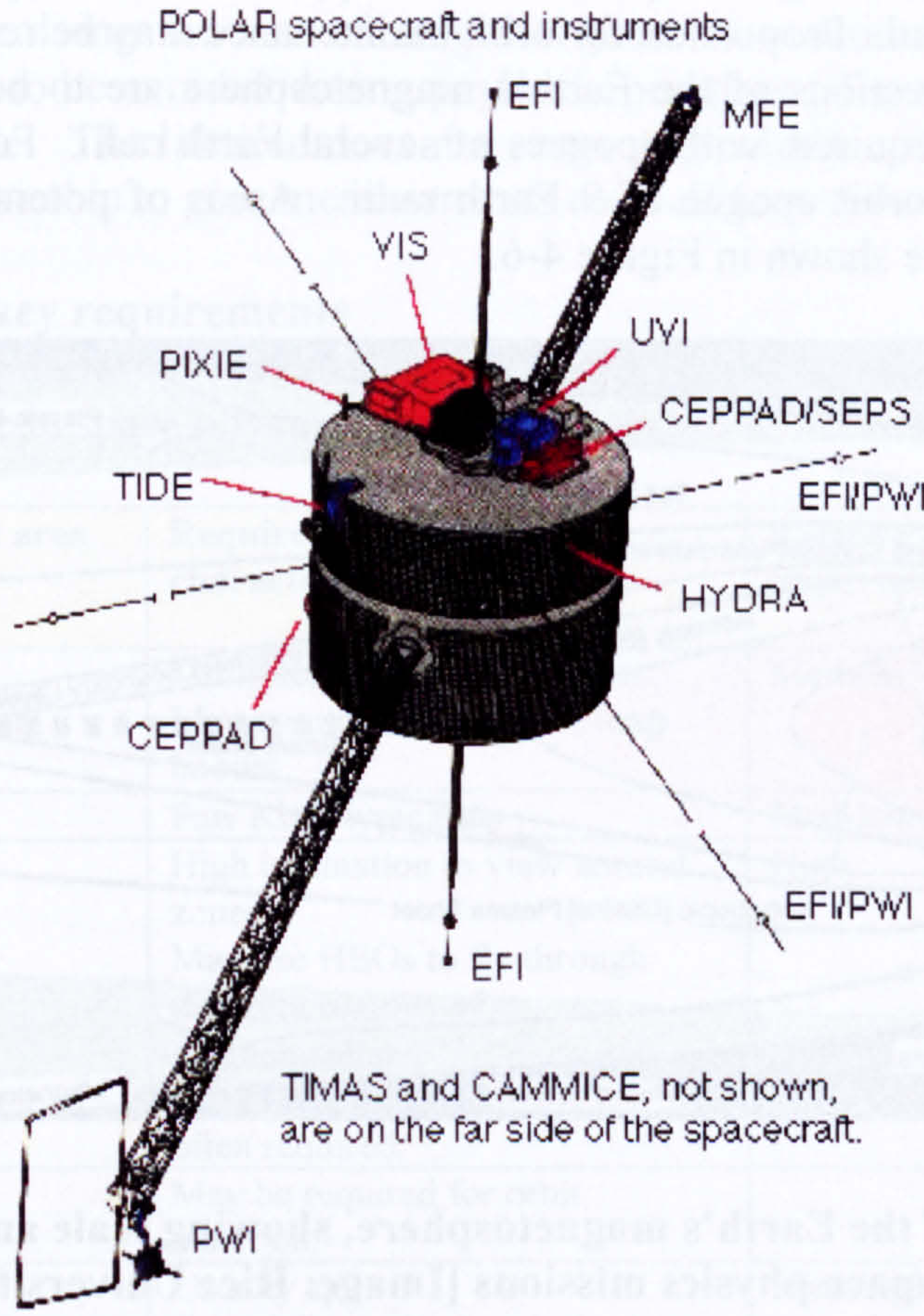

Figure 4-5 Configuration of a typical space physics spacecraft – POLAR [Image: NASA]

Communications and data handling

The data rates from individual instruments may not be particularly high, but they may often be in continuous operation. If the spacecraft is spinning, an omnidirectional system may be required. Alternatively, the antenna may be mounted along the spin vector, if this is aligned in a suitable direction to enable communications.

Attitude and orbit

Space physics spacecraft may need to spin, to remove errors due to local (spacecraft-induced) fields and differentiate between these and the environmental fields to be measured. The spin also allows deployment and tensioning of long wire booms. Spun spacecraft seem to be more common in this application area than any other.

Pointing accuracy is generally not critical (~±5°) but accurate pointing knowledge is required for some aspects e.g. limb scanning. Similarly, magnetospheric mapping experiments require high degree of pointing knowledge (a few arcseconds), although not necessarily precise pointing control.

For investigations of auroral regions (quite common applications in this area), polar or near-polar orbits a required. Propulsion for orbit maintenance may be required. Where investigations of wider sections of the Earth's magnetosphere are to be made, highly eccentric orbits may be required, with apogees of several Earth radii. For example, the POLAR mission has an orbit apogee of 9 Earth radii. Areas of potential interest for space physics missions are shown in Figure 4-6.

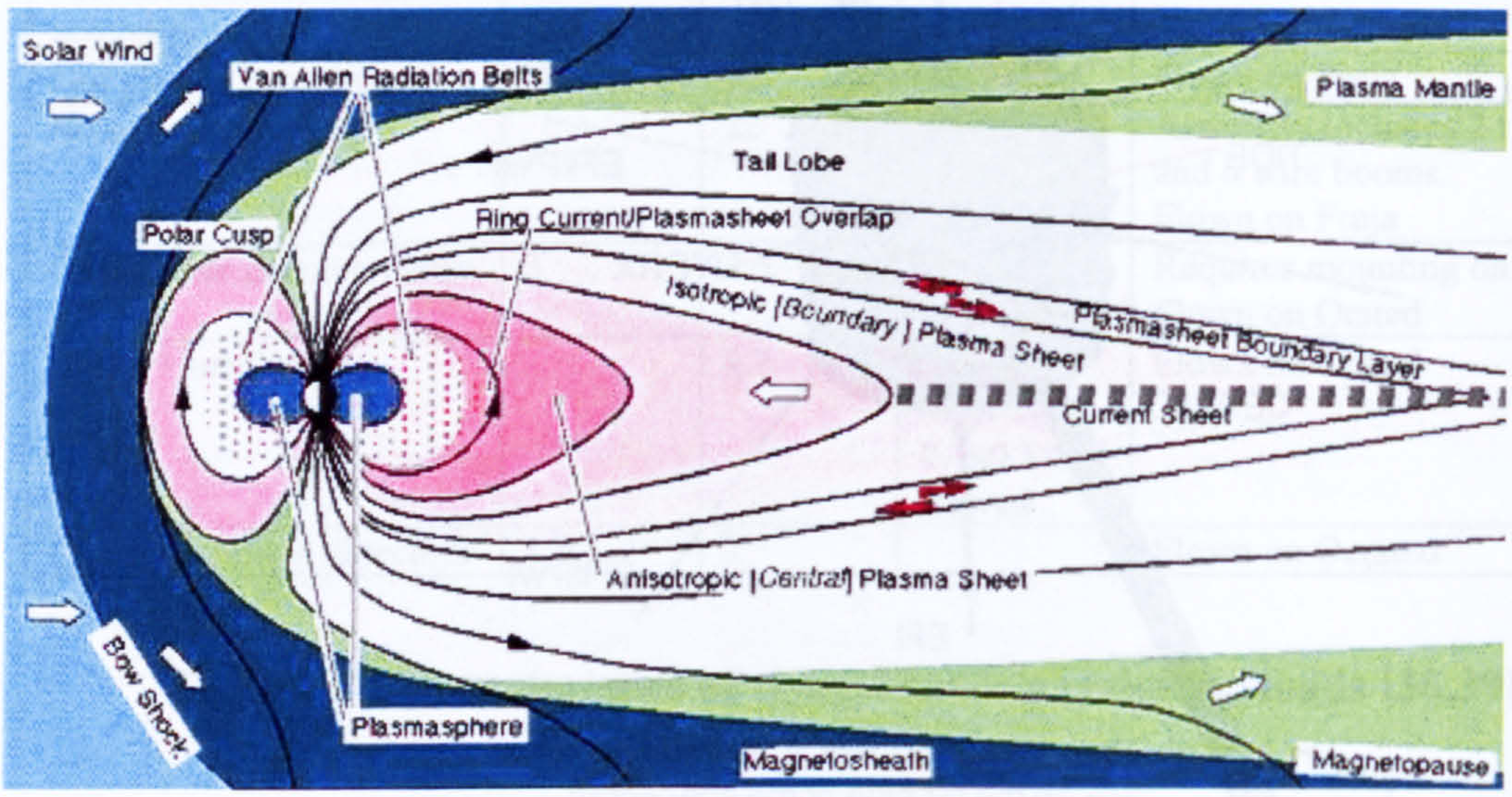

Figure 4-6 Diagram of the Earth's magnetosphere, showing scale and regions of interest for space physics missions [Image: Rice University]

Highly eccentric orbits will generally require the use of a propulsive upper stage. Propulsion will also be required if the spacecraft orbit is required not to precess (however, precession may be desirable as it means the spacecraft would pass through additional regions of the magnetosphere).

Power
The power requirements for most space physics payloads do not seem to be particularly high, but they may be required to operate continuously. Power requirements may be made slightly more problematic by the fact that many space physics spacecraft are required to be spinners. This limits the use of deployed arrays.

Environmental
The characteristics that make particular orbits of interest to space physics missions can cause environmental problems for the spacecraft itself. High radiation fluxes may require the spacecraft to use additional shielding, and drive the selection of radiation-hard electronic components.

Sensitive field-measurement instruments require a high degree of electromagnetic cleanliness on board the spacecraft. This gives a requirement on power harness for power and return wires to be twisted together to minimise magnetic fields. It also drives selection of equipment to avoid units having residual magnetic fields.

Mission and operations

Operations involving the use of magnetorquers or reaction/ momentum wheels will probably have to be avoided during payload operations, if sensitive field measurements are being made. The lifetime does not seem particularly critical, but at least a year is likely to be desirable to give measurements over all four seasons.

Summary of key requirements

The main requirements and performance capabilities that would make a platform suitable for use in space physics missions are shown in Table 4-8.

Requirement area	Required performance/ characteristics	Relative importance	Remarks
Payload accommodation	Multiple, smaller instruments. May require mounting on long booms.	Medium	Deployable booms may require quite large volumes.
Data rate	Few Kbps typically	Med-low	
Orbit	High inclination to view auroral zones. May use HEOs to fly through different regions of magnetosphere. Accurate position knowledge often required.	High	
Propulsion	May be required for orbit insertion.		
Attitude	Usually spin. Spin axis usually inertially-fixed.	Med-high	
Pointing knowledge	Up to arcsecond accuracy.	Med-high	
Pointing accuracy	Few degrees.	Low	
Manoeuvring	Not often required.	Low	If required, spin will mean higher torques required to manoeuvre.
Power	Typically in region of 100-200W	Low	
Lifetime	1-2 years	Low	
Other	Requires high electromagnetic cleanliness onboard.	High	
	May often fly through regions of high particulate radiation, electronics may require shielding.	High	

Table 4-8 Space physics mission requirements summary

4.2.1.3 Microgravity

Microgravity missions make use of the extended periods of near-weightlessness that are possible in orbit. Short-duration experiments are possible on Earth, via parabolic flights or drop-tubes, but for experiments where, for example, biological samples are being grown, a space platform is needed. A selection of microgravity missions and their enabling characteristics are shown in Table 4-9.

Spacecraft and mission	Key characteristics to enable mission
EURECA[16,40]	Very low residual onboard accelerations (<10^{-5}g for periods of more than 1 week) Active thermal control. High degree of autonomy (up to 48 hours without ground contact).
Foton 11	No manoeuvres made during mission, to maintain microgravity. Deliberately re-entered and recovered.
SFU	Retrieved by the Shuttle.
FSW 2	Returned a capsule to Earth.
METEOR-1	Recoverable payload module of materials processing and medical research experiments, intended to be de-orbited after 20-30 days. Service module was to remain in orbit for ~1 year.

Table 4-9 Microgravity missions and mission-enabling characteristics [9]

The key requirements and their impacts on the different subsystem areas are now addressed.

Mechanical design and configuration

Mass and size of some typical microgravity payloads is given in Table 4-10.

Payload	Power	Mass	Size	Remarks
Morphological Transition and Model Substances (MOMO)[7]	90	43	3.5x6.5	Studies directional solidification of transparent media. Flown on STS-84
Materials experiment assembly[39]	500	900	1x1x2	

Table 4-10 Characteristics of typical microgravity payloads

Many microgravity payloads have been flown on the Shuttle. They have therefore been quite large; some have been designed for operation by astronauts whilst in orbit. However, these payloads have sometimes taken the form of suites of many smaller experiments. Some of these may be suitable for flight on small spacecraft (if human intervention is not required during flight).

Microgravity experiments are typically housed inside a pressurised container, containing the requisite environmental control systems. This may make payloads quite large and heavy. As the experiments are utilising the microgravity aspect of the flight, they will not generally require any apertures or sensor fields of view to the exterior for the spacecraft. This may make design of the structure simpler. However, late access to the payload may be required, which may impact on the positioning of the payload and configuration of the spacecraft bus.

Communications and data handling
Microgravity missions may require real-time operations, where (remote) human interaction is necessary. This may require the use of a data-relay satellite, or careful choice of orbit and ground stations (see below).

Attitude and orbit
To maintain microgravity (generally considered to be $<10^{-5}$g), the spacecraft must make minimal attitude and orbit control operations. This gives a requirement for the spacecraft to be placed into an orbit that will be stable for the duration of microgravity required.

Orbit selection may also be influenced by the potential need for the satellite to be retrieved, or to jettison a return capsule. This would require a low altitude. Eccentric orbits may be selected to allow the spacecraft to remain in view of the ground station for long periods of real-time operations.

Power
Environmental control systems, and materials experiments may require high (and continuous) power levels.

Environmental
Environmental control for life sciences experiments, crystal growth, etc may need strict control. This covers temperature, pressure, humidity and ambient atmosphere. However, the environment will often be a self-contained system within the payload (but this may increase power requirements).

Mission and operations
Late access to the payload may be required, before launch, and launch delays may cause problems. This could be a factor against choosing to launch as a secondary payload, where the primary may exercise its right to adjust launch schedules or deny access to payloads.

The required mission durations for microgravity applications may be relatively short. This will be heavily dependent on the type of experiments being conducted. After the experiment period, recovery of samples may be required. This would imply the inclusion of a return capsule, to be re-entered, or the retrieval of the spacecraft by the Shuttle. In some cases, e.g. fluid sciences, adequate data transfer may be sufficient.

Summary
It is not expected that this type of application would be likely to make up a significant portion of the target missions for the small spacecraft. This is based both on the history of such missions (relatively few small satellite missions), and also the presence of the International Space Station as a platform for microgravity studies. Therefore, requirements for microgravity missions are not considered to be key drivers for design of the multipurpose platform.

4.2.2 COMMUNICATIONS MISSIONS

LEO is also now being extensively used by constellations of small communications satellites. However, as explained in Chapter 3, large constellation applications are not considered to be a feasible target area for the multipurpose platform. The mission requirements addressed here are those for small messaging and communications spacecraft, which may be used singly or in very small constellations. A selection of small communications spacecraft, and their enabling characteristics are shown in Table 4-11.

Spacecraft and mission	Key characteristics to enable mission
FAISat – messaging	1000km circular orbit, 66 degree and 83 degree inclinations 600W BOL power from 2-axis articulated solar arrays 7-10 year design life VHF/UHF/L-band communications payload 150kg spacecraft mass Cosmos launch for high inclination orbits
ORBCOMM – messaging	270W BOL power, with spacecraft dry mass of only 33kg Orbits chosen so that the constellation would provide near real-time availability. Spacecraft design was driven to permit launch of 8 spacecraft on a single Pegasus-XL. 328cm long deployed VHF/UHF antenna.
SCD-2 – data collection	Intended to fly at 180 degrees to sister spacecraft SCD-1 in 744x768km, 25 degree inclination orbit. Carried S-band and UHF communications payload. 115kg spacecraft mass. Provided 70W of power.

Table 4-11 Communications missions and mission-enabling characteristics[8,12]

The key requirements and their impacts on the different spacecraft subsystems are now addressed.

Mechanical design and configuration

The payload of a small communications satellite will mainly consist of transponders and a main antenna. Transponders tend to be reasonably high mass-low volume, and require a large amount of power. The characteristics of some small communications payloads are given in Table 4-12.

Payload	Power	Mass	Size	Remarks
ORBCOMM Transceiver & antenna	94	12	Antenna 328cm long	
2.4kbps transceiver	20	2.2	0.05x0.27x0.2	Proposed for Dial-A-Sat[14]
Antenna		4	0.57x0.57x0.03	

Table 4-12 Characteristics of typical communications payloads

The communications antenna will require mounting on a spacecraft face, and this may have implications in terms of shadowing of solar arrays. If the antenna is large, it may require deployment once in orbit.

Communications and data handling
Spacecraft command and communications will be likely to be very low rate, as there is no requirement to return payload experiment data. Spacecraft telemetry will mainly consist of housekeeping data. Therefore, a fairly simple system may be acceptable. This frees up power and other resources to the communications payload.

Attitude and orbit
In terms of the attitude control, communications spacecraft would generally be 3-axis stabilised, with the solar arrays maintained in a sun-facing attitude (to maximise power output), and the antenna Earth-pointing. High-accuracy pointing is unlikely to be required; control to the order of a few tenths of a degree should suffice. This is obviously dependent on the beamwidth of the spacecraft antenna, which will vary according to the specific application of the satellite.

If spacecraft are to be part of a small constellation, then they will be required to maintain their orbital position to some accuracy. This requires accurate position knowledge (probably via GPS), plus the ability to make orbit control manoeuvres.

For messaging and data-collection applications, high-inclination orbits are likely, to allow global coverage. However, other inclinations may be selected for coverage of specific regions. Propulsion may be required for orbit insertion and maintenance. This will also be true for a possible Geostationary spacecraft, which would require an apogee kick stage to circularise its orbit after insertion into a GTO. Piggyback launch to GTO is quite likely to be available, due to the large number of large launches to this orbit. Once in GEO, thrusters for station-keeping propulsive manoeuvres would be required.

Power
Power requirements will be dictated by the parameters of the communications payloads, and its operational duty cycles – how often and for how long it must transmit. Communications spacecraft generally have rather high power budgets, to allow freedom to operate the payload as much as is necessary.

Environmental
Communications payloads do not generally have any additional thermal requirements, and commercial transponders will usually be radiation-tolerant.

Mission and operations
As communications satellites are required to provide a continuous service (rather than to perform a specific experiment or collect a data set), a longer lifetime is expected – perhaps 10 years. This will mean a requirement for greater redundancy levels, higher reliability components and larger volumes of consumables (propellant).

In the case of a commercial communications application, it may be expected that the payload supplier be responsible for their own ground network infrastructure, to provide a service to its own customers. However, the satellite platform provider may be responsible for the routine housekeeping monitoring of the spacecraft. Also, for a LEO messaging service, a customer may require their own 'personal messaging satellite', and therefore also will require their own portable ground receiving systems.

Summary of key requirements
The main requirements and performance capabilities that would make a platform suitable for use in communications missions are shown in Table 4-13.

Requirement area	Required performance/ characteristics	Relative importance	Remarks
Payload accommodation	Antennas may be large.	Med-high	Antennas may require deployment.
Data rate	Low (Kbps)	Low	Little or no payload data, just housekeeping.
Orbit	LEO, probably high inclination but could be tailored to particular user's coverage requirements. Possible GEO.	Medium (High if GEO)	
Propulsion	May be required		
Attitude	Nadir pointing	High	
Pointing knowledge	Few tenths of a degree	Medium	
Pointing accuracy	Up to a few tenths of a degree	Medium	Depends on antenna beamwidth.
Manoeuvring	Maintain nadir pointing	Medium	
Power	May be up to 500-600W	High	
Lifetime	Longer lifetime an advantage – 5-10 years	Med-high	
Other			

Table 4-13 Communications mission requirements summary

4.2.3 EARTH OBSERVATION MISSIONS

Earth observation spacecraft have generally been quite large and heavy – the well-known ESA ENVISAT spacecraft massed over 8 tonnes, the French SPOT satellites nearly 2 tonnes. In recent years, however, use of smaller satellites for this type of application has grown. A summary of recent small missions is given in Table 4-14.

The previous use of large spacecraft has been mainly due to the high power and pointing accuracy requirements of imaging payloads, and the large size of most instruments. However, the attitude control and power systems on small satellites have been much improved, making them more suitable for Earth observation.

Spacecraft and mission	Key characteristics to enable mission
GFO – ocean altimetry[5,32]	800km, 108° orbit, giving a 17-day exact-repeat track. Data downlinked continuously to Navy ships and facilities. Payloads 47kg in total, and require 121W for operation.
TOMS-EP – ozone-mapping [4]	High-inclination, low-altitude orbit (955km, 99.3 degrees), giving coverage of the entire Earth every 24 hours. (Orbit is sunsynchronous, ascending node 11:00 to 12:00 local time). Nadir-pointed. Onboard time knowledge to <100msec accuracy w.r.t. UTC. Pointing accuracy <0.5° (roll/pitch), <1.0° (yaw). Pointing knowledge <0.25° in all axes. 3-year lifetime goal. Continuous mapping mission. Contamination control critical – contamination budget established in design phase. Hydrazine propulsion, with redundant thrusters, for orbit acquisition. Deployed arrays, supporting ~130W orbit average load.
EO-1 – multi-band Earth imaging[27]	Pointing accuracy 0.03° in all axes, <5arcsec jitter. Inertial or nadir pointing. 705km sunsynchronous orbit, 10am descending node. May be slewed to celestial objects for instrument calibration. Solar panels produce 600W EOL power. (Experiment 80W) 50a-hr super-NiCd battery. Hydrazine thrusters for orbit insertion, maintenance, and de-orbit. 105Mbps data transmission via X-band. S-band backup system. 40Gbit onboard data storage. 529kg total spacecraft mass. (Main payload 90kg). Payload-spacecraft alignment measured to 20arcsec accuracy.

Table 4-14 Earth observation missions and mission-enabling characteristics

The key requirements and their impacts on the different spacecraft subsystems are now addressed.

Mechanical design and configuration

The payloads of Earth observation spacecraft are often quite large, and will require an unobstructed field of view towards the Earth. Therefore, instruments are often mounted externally, on the nadir-pointing face of the spacecraft. Smaller instruments may be mounted internally, and provided with an aperture in the spacecraft body structure. Characteristics of some typical Earth observation payload instruments are shown in Table 4-15.

Payload	Power /W	Mass /kg	Size /m	Remarks
Thematic mapper	280	239	2x0.7x0.9	Data rate 85Mbps
Doppler imager	165	191	1.25x0.6x0.8	Data rate 20Mbps
V High resolution camera	9	12	0.7x0.35	Designed for traffic-monitoring microsatellite
Infrared temperature sounder	72	66		
Total Ozone Monitoring System	25	30		50° x 3° field of view. <0.03° mechanical alignment accuracy required. Isolated thermal interface required (<5W transfer). Data rate 150Kbps
GOES sounder	150	130	Aperture 31cm	
AVHRR/3	28.5	32.5	0.3x0.4x0.8	Used on NOAA satellites
Earth imaging spectrometer	<100	25	0.3x0.2x0.1	

Table 4-15 Characteristics of typical Earth observation payloads[4,24,39]

Small spacecraft may typically only carry one or two main instruments, due to size limitations. A typical configuration of an Earth observation spacecraft is shown in Figure 4-74-8. This shows the instruments mounted on the nadir face of the satellite, and their quite large size.

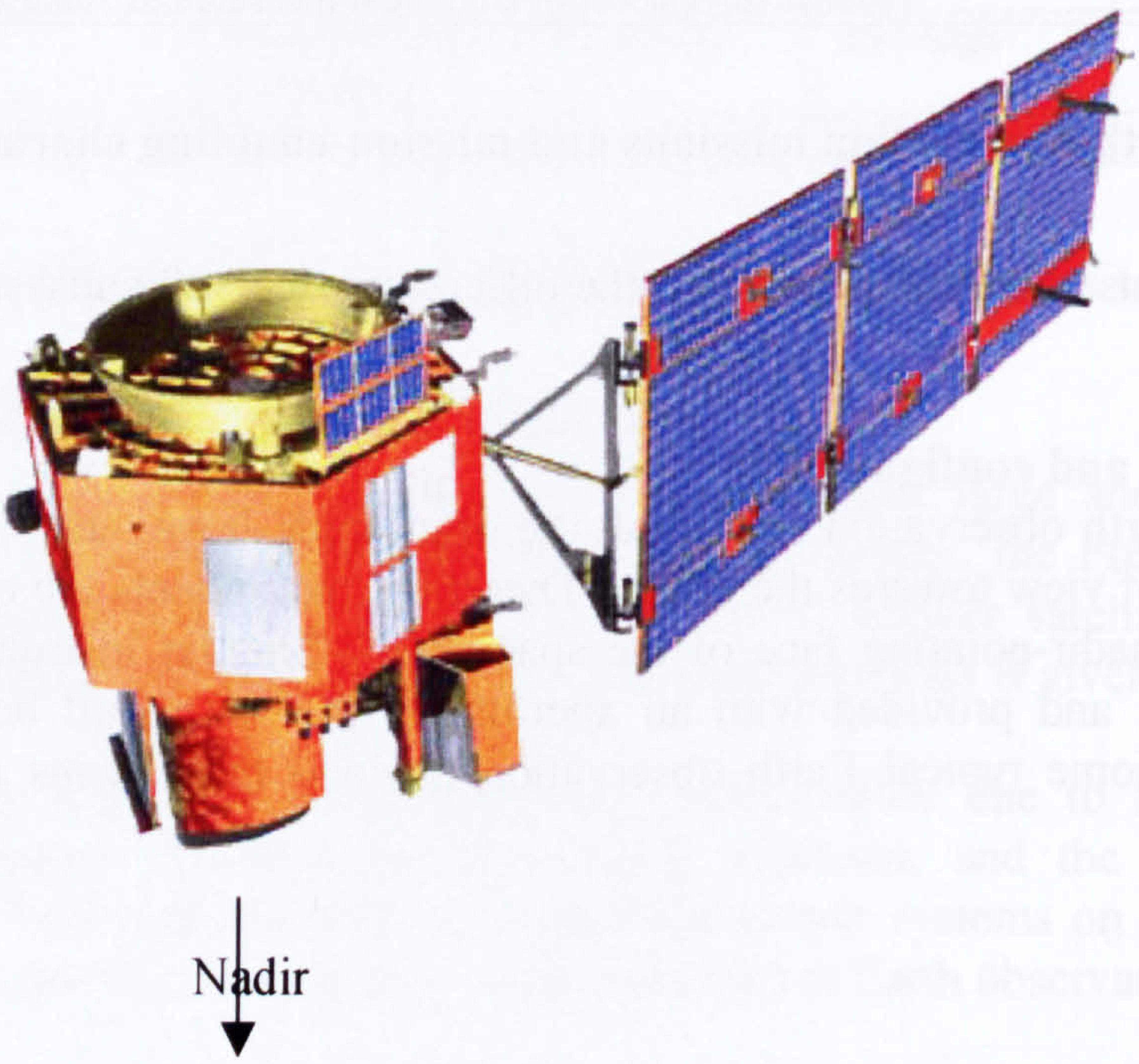

Figure 4-74-8 Configuration of a typical Earth observation spacecraft – EO-1 [Image: NASA]

The nadir face will also probably have to be used for mounting of the communications antenna, further increasing demand for space in this area.

Communications and data handling

Earth observation instruments typically generate large quantities of data, as they often operate continuously, at high data rates. As the spacecraft often operate in LEO, ground passes are short, and there may often be many orbits without ground access (depending on the number of ground stations used). This can drive the requirement to:

- Store large quantities of data on board (the EO-1 spacecraft, which may be taken to be state-of-the-art, has 40Gbit storage capacity)
- Perform onboard data-compression processes
- Provide high downlink data rates, perhaps by moving to X-band

Alternatively, ground contact can be increased by using a larger number of ground stations, and downloading data in near-real-time (as with GFO).

Attitude and orbit

In order to allow global observation and high repetitivity of observation, Earth observation spacecraft are generally deployed into low altitude (700-1000km), high inclination orbits. It is often helpful to use sun synchronous orbits so that observation is always performed under similar light conditions. Equatorial/tropical countries may also desire spacecraft in low-inclination orbits, where coverage of the same area can be achieved several times a day. Orbit parameters for recent LEO Earth observation spacecraft are shown in Figure 4-9.

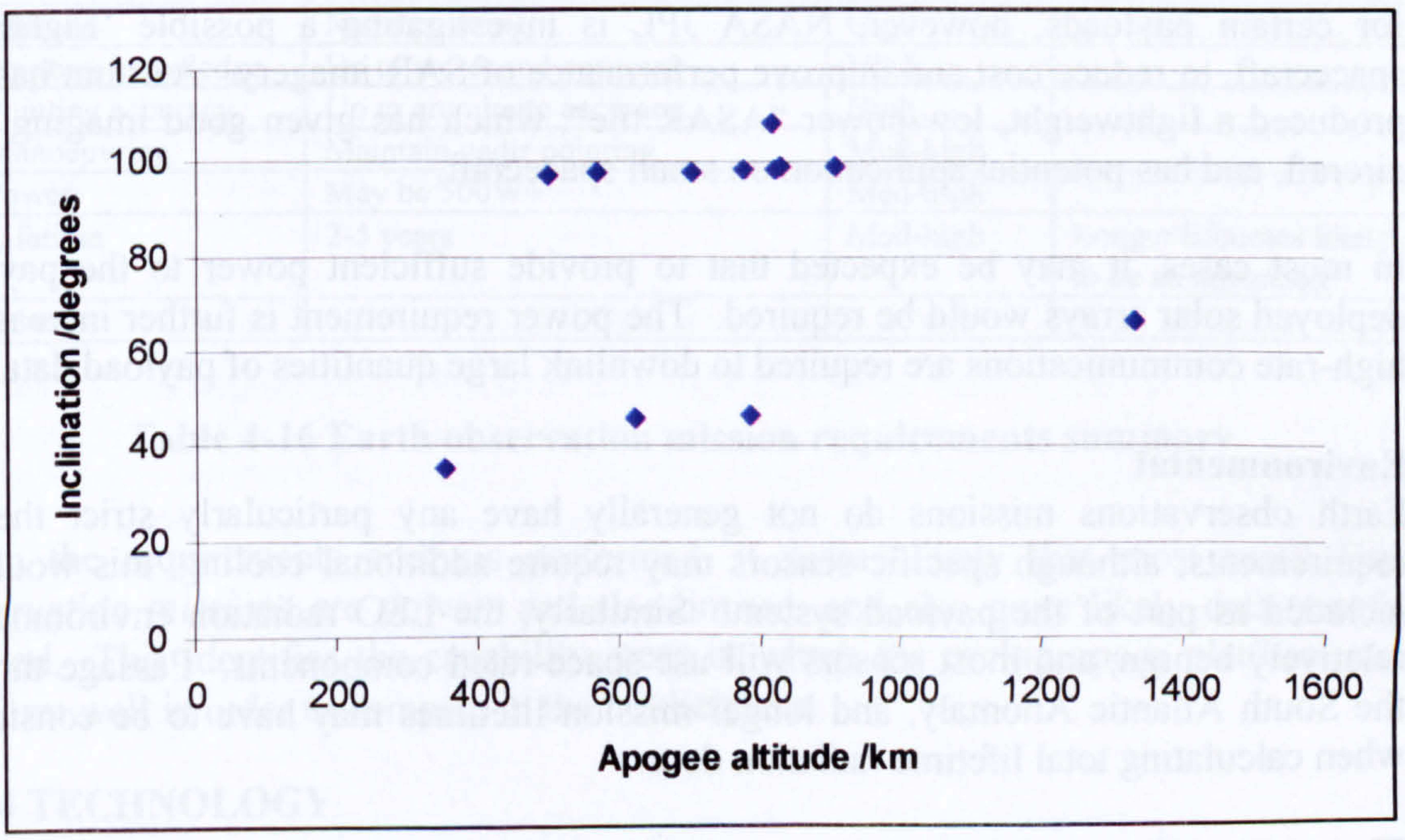

Figure 4-9 Altitude and inclination of LEO Earth observation spacecraft

This figure shows the common use of the sunsynchronous inclination, and the additional use of lower-inclination orbits for spacecraft observing specific locations.

Lower altitudes may allow improved instrument resolutions. However, the increased aerodynamic drag in lower orbits will give greater propulsion requirements for orbit maintenance.

Geosynchronous spacecraft are also used for Earth observation, giving very good coverage of specific areas. From this high altitude, however, resolution is poorer. It is probably less likely for a small satellite to be used for observation missions from high orbits, as the required instrument apertures for higher altitudes may make the payloads too large.

Propulsion may be required to maintain the spacecraft in its correct orbit, particularly if a longer mission life is required. For repeating ground-track orbits, this is particularly important, and will also be required for accurate initial orbit acquisition.

To produce clear images, accurate control and knowledge of spacecraft attitude is required. In general, pointing accuracies of up to 0.1°, or even greater, are necessary. Higher pointing capabilities are likely to be a key point in the selection of a platform for an Earth observation mission. Accurate position and timing data are also often required. The spacecraft will often be 3-axis stabilised, nadir pointing, but spun spacecraft are also sometimes used (where scanning instruments are employed).

Power

The power requirements of payloads are often quite high, especially for SAR (synthetic aperture radar) applications. This has meant that small satellites have not been suitable for certain payloads; however, NASA JPL is investigating a possible 'LightSAR' spacecraft, to reduce cost and improve performance of SAR imagery. Astrium has also produced a lightweight, low-power "ASAR tile", which has given good imaging from aircraft, and has potential application on small spacecraft.

In most cases, it may be expected that to provide sufficient power to the payload, deployed solar arrays would be required. The power requirement is further increased if high-rate communications are required to downlink large quantities of payload data.

Environmental

Earth observations missions do not generally have any particularly strict thermal requirements; although specific sensors may require additional cooling, this would be included as part of the payload system. Similarly, the LEO radiation environment is relatively benign, and most sensors will use space-rated components. Passage through the South Atlantic Anomaly, and longer mission lifetimes may have to be considered when calculating total lifetime radiation dose.

The main environmental consideration for Earth observation spacecraft is cleanliness. This is important during the AIT process on the ground, and once the spacecraft is in orbit. Condensation of out-gassed material onto optical surfaces must be avoided, which impacts on permissible choice of materials. Positioning of thrusters must also be made such that plumes do not impinge on instrument optics.

Mission and operations
Longer lifetimes are likely to be advantageous for Earth observation missions, as there is likely to be an on-going requirement for the data generated. Therefore, when a spacecraft reaches the end of its life, a new one must replace it.

Spacecraft operations are generally characterised by continuous or near-continuous operation of the payload. This means that the spacecraft must be capable of autonomous function while not in ground contact.

Summary of key requirements
The main requirements and performance capabilities that would make a platform suitable for use in Earth observation missions are shown in Table 4-16.

Requirement area	Required performance/ characteristics	Relative importance	Remarks
Payload accommodation	Requires mounting or apertures on nadir face. Instruments may be quite large.	High	
Data rate	Up to tens of Mbps.	High	
Orbit	Often sunsynchronous. May require repeat ground track. Lower orbits for higher image resolution.	High	
Propulsion	Likely to be required for orbit maintenance.	Med-high	Also for accurate orbit insertion.
Attitude	Nadir pointing	High	
Pointing knowledge	Up to arcsecond accuracy.	High	
Pointing accuracy	Up to arcminute accuracy.	High	
Manoeuvring	Maintain nadir pointing	Med-high	
Power	May be 500W+	Med-high	
Lifetime	2-5 years	Med-high	Longer lifetimes likely to be an advantage.
Other			

Table 4-16 Earth observation mission requirements summary

From the requirements analysis performed, it seems likely that most small Earth observation missions are power- and size-limited, and also quite likely data-transfer-limited. This identifies the capability areas in which the multipurpose platform must perform well in order to compete in the marketplace.

4.2.4 TECHNOLOGY
New technologies for space applications are constantly being developed, and their use may enable new types of missions. Before they can be used in a space mission, however, they must be qualified for space use. This generally takes the form of accepted environmental testing on the ground such as thermal vacuum tests and

vibration tests. But the true test only comes when the equipment is operated in real space conditions.

The problem with this is that, if the equipment fails, perhaps via some unexpected mode that was not encountered during ground testing, it could result in the failure of a large and expensive mission. This is why technology demonstration missions can be extremely useful. The idea of these missions is to launch new equipment on a small, cheap satellite, and qualify it in true space conditions.

The spacecraft itself can be basic, using older, proven techniques. The lifetime need not be particularly long, merely long enough to show that the payload operates as expected. If the mission cost can be minimised, it may be worthwhile for a manufacturer to use such an opportunity, as it can then advertise its new product as 'space qualified'. It may also be possible to reduce safety margins on later missions flying the proven equipment, which will also save money.

It is hard to define any typical requirements for a technology demonstration mission, as the payloads to be flown may vary immensely. Driving factors, however, are likely to be that the cost be low, and the schedule be short; therefore such missions are likely to be opportunistic, taking advantage of whatever cheap launches may be available.

The technologies to be demonstrated will be either prototype subsystem or payload systems. Therefore, it may reasonably be expected that if a platform is suitable for the range of mission applications described previously, it can also provide a suitable testbed for demonstration of both subsystem and payload test items.

4.3 SYSTEM REQUIREMENTS

These are the requirements that influence the design of the spacecraft itself. They are driven from the top level by the programmatic and mission requirements, and also by the specific needs of particular payloads. This section forms the basis for the spacecraft design work in the next chapter.

4.3.1 MECHANICAL DESIGN AND CONFIGURATION

The mechanical design and overall configuration of the spacecraft is driven by a number of different requirements, arising from a range of different sources. These requirements are shown in Table 4-17.

Configuration driver	Impacts on:	Remarks
Launch vehicle requirements	Size, shape, mass of spacecraft.	
Accommodation of equipment	Size and shape of spacecraft.	Mass and volume of items to be accommodated (inc. payload.)
Power requirements	Choice of body-mounted or deployed arrays. Need for mechanisms.	Examined in Section 4.3.4.
Pointing requirements	Positioning of equipment.	Covered in accommodation of equipment.
Field of view requirements	Positioning of equipment. Provision of apertures in structures.	Covered in accommodation of equipment.
Environmental requirements	Structural and mechanical properties (strength, stiffness). Selection of materials. Thermal design.	From the launch, and on-orbit environments.
Assembly and ground handling requirements	Inclusion of handling hard-points in structure.	

Table 4-17 Requirements driving spacecraft mechanical design and configuration

Each of these drivers is now examined, to derive the requirements applicable to the spacecraft design.

4.3.1.1 Requirement inputs from the launch vehicle

Many of the mechanical, structural and configuration requirements for a spacecraft are driven by the launch vehicle. Selection of a particular launcher constrains the mass, size and shape of the spacecraft, and the method of interfacing mechanically with the rocket. As the spacecraft is intended to be suitable for a wide range of missions and customers, it follows that it should also be suitable for launch by a range of different vehicles. This also confers a potential advantage in terms of cost and schedule – flexibility in choice of launch vehicle may allow use to be made of short lead-time 'opportunity launches' at a lower price.

There are a number of different possibilities for small satellite launch. These may be categorised as "piggyback", shared, and dedicated launch.

"Piggyback", or secondary payload, launches may be offered on the larger launchers. These take advantage of spare capacity when launching large spacecraft, with the piggyback passenger fitting into the available free space. Several large launchers, such as Ariane 5, Delta, and Tsyklon have special support structures designed to carry small piggyback payloads. The Ariane5 ASAP (Ariane Structure for Auxiliary Payloads) is shown in Figure 4-10.

In other cases, a dedicated support and interface structure may be made to fit the launcher, or, on rare occasions, the main satellite. The piggyback option is only available to smaller spacecraft, and the secondary payload has little or no influence on launch schedule and mission parameters. The available envelope size and geometry, and the interface, may be somewhat unconventional (for example, the Delta 2 secondary payload interface ring is in a vertical, rather than the usual horizontal, plane).

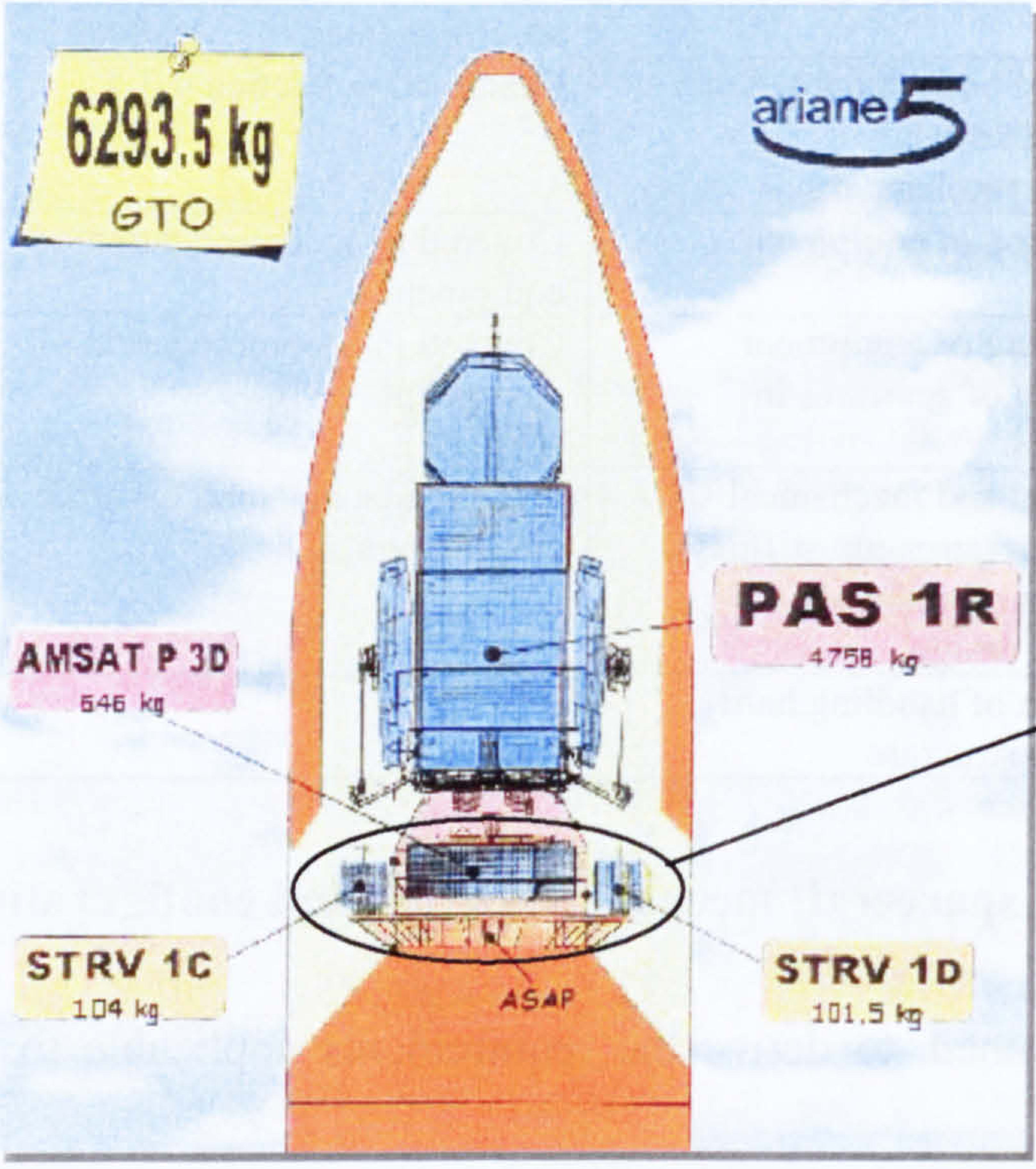

Figure 4-10 The first flight of the Ariane ASAP5, on A507. [Images: Arianespace]

Two STRV microsatellites are supported on the ASAP ring, and launched piggyback with the PAS 1R main satellite, and AMSAT P3D.

Piggyback launches are more often available to GTO rather than LEO, as the heavy-lift launch vehicles are most commonly used to orbit large Geostationary spacecraft (for example, 90% of Ariane launches are to GTO).

Shared launches are distinct from piggyback launches. These are dual-manifest, with each passenger having – if not necessarily equal – at least a significant influence on the launch schedule and mission parameters. Many launchers offer structures to facilitate dual launch, for example the Delta 2 DPAF (Dual Payload Attach Fitting). This was used in 2000 to launch the JASON and TIMED spacecraft, and the EO-1 and SAC-C spacecraft.[41] The dual payload arrangement is shown in Figure 4-11 on the following page.

Dedicated launch is generally the mostly costly of the options, but gives the advantage of providing the exact specifications required by the spacecraft and mission, without the need for compromise. For small spacecraft, a dedicated launch would only be feasible on the small launchers, such as Pegasus and Taurus.

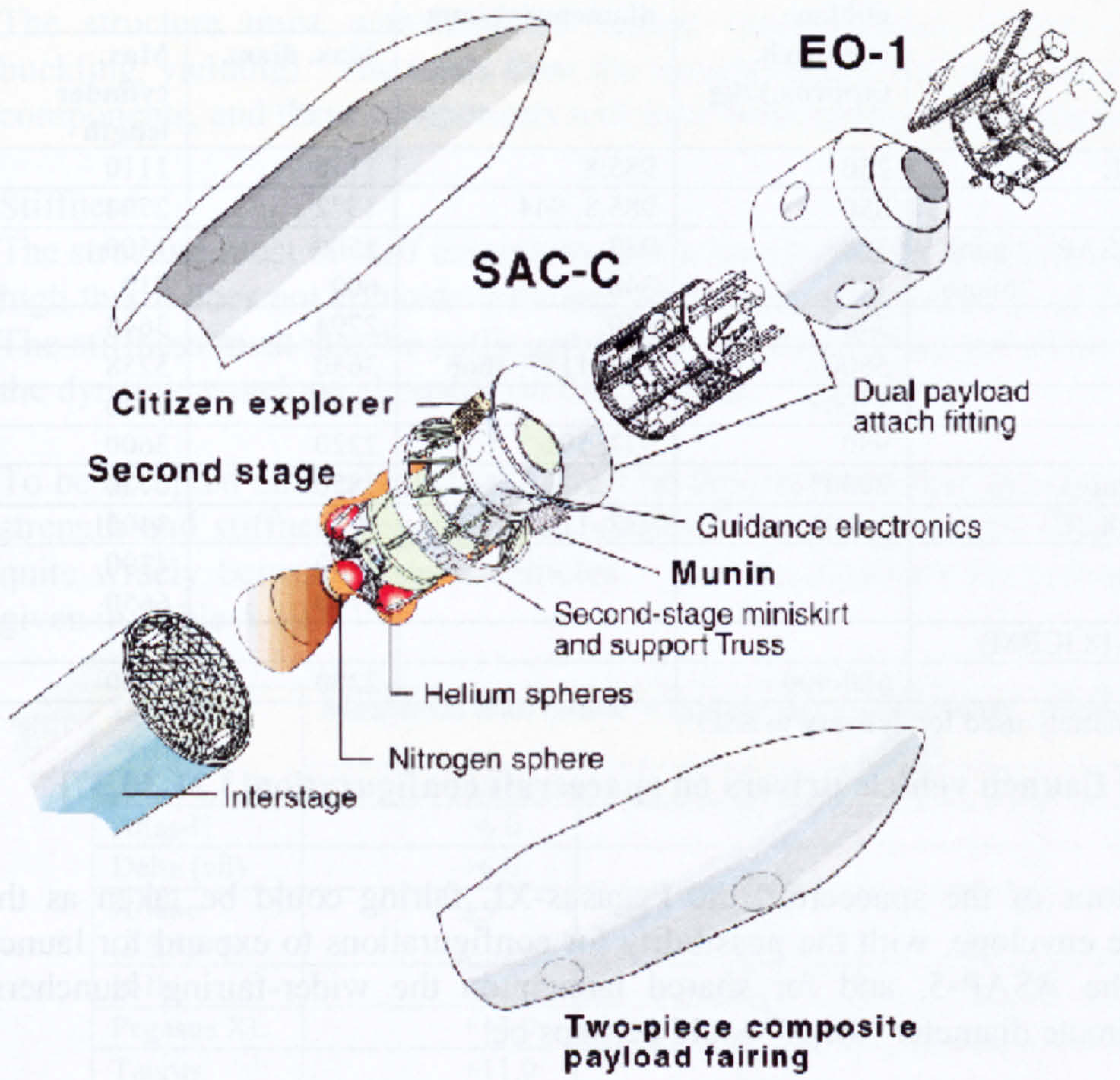

Figure 4-11 Dual payload arrangement on the Delta 2 vehicle [Image: NASA]

The allowable envelopes within the launcher fairings, and payload interface specifications can be found in detail in the user guide of each vehicle. An overview of the main interfaces and launch envelopes likely to be used by small spacecraft is given in Table 4-18 on the following page.

From the table, it can be seen that many launchers offer a 937mm interface. It would therefore make sense to use this interface as a baseline in the spacecraft design. The Pegasus and Taurus interface is slightly larger, but an adapter to the standard 937mm is probably available, or could be made quite easily.

Launch vehicle	Mass to 600km sunsynch. (approx.) /kg	Adapter diameter(s) /mm	Spacecraft envelope /mm	
			Max. diam.	Max. cylinder length
Pegasus-XL	250	985.8	1118	1110
Taurus	850	985.8, 944	1372	2794
Ariane 5 ASAP – "mini" "micro"	300 120	937 348	1500 600	1500 710
Delta 2	3400	940	2794	3648
Atlas 2	5500	937, 1147, 1666	3650	5258
PSLV	1200+		2900	2740
Rokot	950	937, 591	2220	3600
Soyuz/Molniya	2000*		2850	6720
Long March 2C	5000	937, 1194, 1497	3070	3400
Athena 1 2			1980 2740	4290 6650
Dnepr (SS-18 ICBM)				
Cosmos	680-900		2200	4700

* Not generally used for delivery to SSO

Table 4-18 Launch vehicle drivers on spacecraft configuration[3,21,30,37]

For the dimensions of the spacecraft, the Pegasus-XL fairing could be taken as the smallest baseline envelope, with the possibility for configurations to expand for launch on Taurus or the ASAP-5, and for shared launch on the wider-fairing launchers. Suitable approximate diameter "steps" could perhaps be:

- 1100mm – for launch on Pegasus-XL
- 1300mm – for launch on Taurus
- 1500mm – for launch on ASAP5
- 1900mm(+) – for launch on Athena and larger-fairing vehicles

Other size requirements may come from a desire to share the available space with other satellites, or perhaps to be able to stack two of the spacecraft if the design allows this. The 1900mm envelope for launch on the larger vehicles is suggested rather than prescribed; the fairings are considerably wider than this. However, a narrower diameter would allow the spacecraft to be inserted below a main passenger, within a dual launch adapter. These fairing and interface dimensions will be drawn on during the design concept phases in the next chapter.

As well as the size of the spacecraft, the launch vehicle also largely drives the mechanical design. The spacecraft structure has the task of supporting and securing all the other on-board subsystems and instruments, both on the ground, through launch, and in space. It must therefore satisfy a wide range of requirements, the most crucial of which is arguably the ability to withstand the launch loads. Launch is the most physically demanding stage of most space missions, and launch environment therefore has a major influence on the mechanical design of spacecraft. The prime requirements on the structure for launch environment survivability are:

Strength
The structure must withstand the launch loads without failing (e.g. by bending, buckling, yielding). The loads from the launch vehicle will have axial and longitudinal components, and these components will have both steady-state and transient levels.

Stiffness
The structure must be stiff enough to give a lowest natural frequency that is sufficiently high that it does not coincide with any of the natural frequencies of the launch vehicle. The stiffness must also be sufficient to prevent any deflections of the structure outside the dynamic envelope allocated on the launcher.

To be accepted onto a launcher, it must be demonstrated that the spacecraft satisfies the strength and stiffness requirements specified for that particular launcher. These vary quite widely between launch vehicles – the specifications for potential launchers are given in Table 4-19.

Launch vehicle	Maximum load (static + dynamic) (g)		Minimum fundamental frequency (Hz)	
	Axial	Lateral	Axial	Lateral
Atlas-II	+6.0	+1.6	15	10
Delta (all)	+6.0	±2.0	35	15
Ariane 5 (ASAP-5)	±5	±3.0	60	30
H-II	+5.0	±2.0	30	10
Pegasus XL	+13.0	±6.0		
Taurus	+11.0	±2.5	25	
Rokot	+8.1	±0.8	Avoid 16-33	15

Table 4-19 Launch environment specifications for selected launchers

If the spacecraft is to be compatible with a number of different launchers, it must either be designed to meet the environmental requirements of the 'worst case' launcher that may be expected to be used, or be easily upgradeable to meet them if required. From the table above, it can be seen that the ASAP5 carries very high stiffness requirements, requiring a fundamental frequency higher than 60Hz. However, it is easier to achieve high stiffness on smaller spacecraft due to the shorter length of the structural members.

The launcher will also impose mass properties requirements, defining limits for the position of the spacecraft centre of mass (axial location and lateral offset). The limits are determined by the structural properties of the separation system.

4.3.1.2 Mass targets

The spacecraft mass targets for the design section arise from:

- giving comparable payload mass capability to those platforms with which the design is to compete,
- being compatible with a range of launch vehicle options,

whilst allowing sufficient payload mass for the accommodation of payloads for the identified missions.

The platform is intended to be flexible and so a rigidly defined target mass is not particularly appropriate in this case. However, this should not mean that system mass should be allowed to creep up unnecessarily – the lower the mass of the platform, the greater the mass allowance for the payload for a particular launch.

The platform and payload mass capabilities of commercial small spacecraft platforms are shown in Figure 4-12.

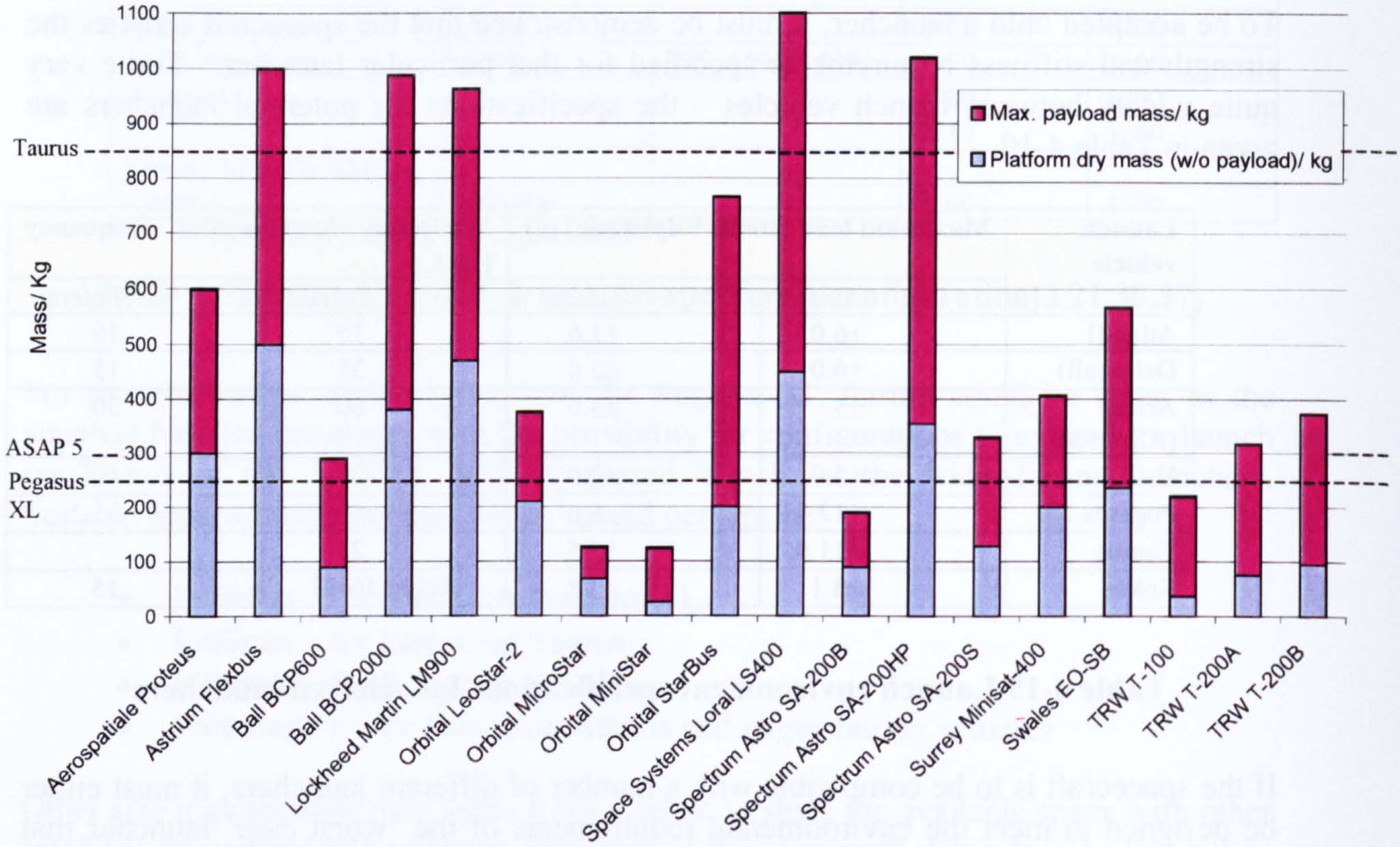

Figure 4-12 Platform and payload mass for commercial multipurpose small spacecraft platforms, with small launcher capabilities (to 600km SSO) indicated

This figure shows that the commercial platforms divide approximately into two groups. The smaller group may be launched by Pegasus-XL or on an ASAP 5, and have payload capabilities of 100-200kg. Some of these may, however need to launch on a larger launcher (e.g. dual launch on a Taurus) if their full payload mass capability is used. The larger group, with payload mass capability of around 500-600Kg, is sized for launch on a Taurus-class launcher, or dual launch on a larger vehicle.

The key factor is to allow the platform dry mass, in the various configurations, to fall sufficiently below useful mass "cut-off points", driven by launcher options, to give an acceptable payload mass. These cut-off points would reasonably be given by:

- Pegasus-XL/ASAP 5 launch (the small difference in launch capabilities between these two could be given over to additional payload/propellant with the same basic platform mass)
- Shared Taurus launch (sharing gives some play in the mass fraction used, also applies for piggyback launch on larger vehicles)
- Dedicated Taurus launch

This then gives approximate platform mass targets of up to 100-150kg for Pegasus-class launch, 200-300kg for "dual-Taurus" or piggyback-class launch, and in the region of 400kg for a dedicated Taurus-class launch. The largest of these cut-offs will give quite a large spacecraft; it may be considered to be unfeasible to take even a modular design from the Pegasus-size to Taurus-size. This question will be addressed during the design phase. These mass boundaries must also fit in with the volumetric accommodation constraints of the launchers, identified in the previous section.

4.3.1.3 Accommodation of on-board equipment

Spacecraft structures may be categorised as either primary or secondary; primary structures are responsible for the principal load-bearing and provision of main attachment points, secondary structures include equipment boxes and casings, brackets, and supports for smaller components.

Much of the equipment that must be accommodated within the primary structure will be electronics boxes of various sizes. These have traditionally been rectangular boxes (due to their containing rectangular circuit boards), often with attachment by bolting through flanges at the base. Although new technologies are now beginning to expand the range of possible configurations for electronics, it is likely that accommodation of a certain number of the conventional 'black boxes' will be required on board most spacecraft for some time to come. Investigation of possible alternatives is given in the subsystem design sections.

Allowances must also be made in the structure for the passage of electrical cables between the various subsystems, and also to the outside for testing and communication with the spacecraft once it is on the launcher.

Table 4-20, on the following page, summarises the types of spacecraft subsystem equipment that may need to be accommodated on board, and gives a top-level outline of their mounting requirements.

Equipment Item	Relative size	Remarks & particular mounting considerations
Data handling/computer system	Large	Likely to dissipate quite large amounts of heat; this may influence positioning near a radiator.
Receiver	Medium	
Transmitter	Medium	May be physically housed with the receiver. Recivers/transmitters often quite heavy.
Antenna	Small-medium (depending on type)	Externally mounted. Paired omnis mounted on opposite faces, high-gains mounted on Earth-viewing face. Require clear FOVs.
Battery	Large	May have sensitive thermal range, which may influence positioning.
Power control system	Medium	Shunt resistors may dissipate large amounts of heat, this may influence positioning near radiators.
Reaction wheel	Medium	Generally 3 mounted orthogonally, plus 1 offset redundant.
Magnetorquer rod	Small-medium	As above. Should not be placed near equipment sensitive to magnetic fields.
Attitude control system	Medium	May be physically part of the data handling/ onboard computer system.
Cold gas thrusters	Small	Mounted externally, should be positioned to avoid plumes impinging on sensitive equipment (e.g. optics). Correct alignment of thruster pairs critical.
Propellant tank	Medium-large	Require specialised supports to avoid high shell stresses. Should be mounted near CG so this does not shift with propellant use. (This also applies to tanks for other onboard consumables such as cryogenic coolants).
Solar array	Large	May be externally body-mounted or deployed. Deployed arrays require deployment mechanism. Articulation mechanisms may also be used.
Inertial reference unit	Small-medium	Does not require any external aperture.
Sun sensor	Small	Enough sensors must be mounted on different faces to allow sun vector to always be known. Need clear FOV.
Star tracker	Small-medium	May be mounted externally, or provided with an aperture to the exterior. Need clear FOV. Alignment must be precisely known for accuracy of attitude determination.
Earth sensor	Small	Mounted externally; many different types & levels of sophistication possible.
GPS system	Small	Generally an external patch antenna plus small electronics package.
Launcher mechanical interface	Medium	Interfaces to main structural load paths.
Power harness	Small-medium	Needs to be distributed throughout the spacecraft to most of the equipment on board. Bend radii require consideration. May require strain-relief loops. Must be secured sufficiently to avoid launch vibrations causing fatigue failure of wires.
Data harness	Small-medium	As above.

Table 4-20 Mounting considerations for onboard subsystem equipment

Specific requirements in terms of sizes and shapes of equipment will follow on from definition of each subsystem. Some ballpark figures may, however, be identified to give a preliminary idea of the types of items that might need to be accommodated. These are obtained from previous small spacecraft or commercially-available equipment, and are shown in Table 4-21.

Item	Mass (kg)	Dimensions (mm)	Remarks
Reaction wheel	5	350 (diameter)	
Magnetorquer rod	0.8	500 (length)	
Data handling system	5.5	290x290x90	Based on SIL equipment
Transmitter (S-band)	0.7	150x90x40	Based on SIL equipment
Receiver (S-band)	1	170x150x40	Based on SIL equipment
Battery	5	250x60x60	7Amp-hour pack
Power control system (electronics)	3.5	290x290x50	Based on SIL equipment
Attitude controller (electronics)	2.5	200x170x110	

Table 4-21 ROM sizes of small satellite subsystems

An attempt has been made to specify equipment that is close to the current "state-of-the-art", as it does not seem sensible to design around out-dated equipment. Use of the new technologies discussed in Chapter 2 will also be incorporated in the design phase, where appropriate and beneficial. Only the larger pieces of equipment are included here, as this is only intended to give a ROM idea of the scales that must be dealt with. As a general rule, increased capability or capacity of a particular subsystem will be associated with an increase in mass and size. This must be accommodated within the design reconfiguration.

The mission-specific requirements analysis performed earlier in this section addressed similar requirements in terms of size and mounting for different payloads (Table 4-4, Table 4-7, Table 4-12 and Table 4-15).

These tables give a preliminary idea of what the structure may be required to contain and support. This allows for preliminary configuration studies to be performed. These can be then developed further, and trade studies done, after more detailed definition of the subsystems has taken place.

4.3.1.4 Ease of assembly and ground handling

Design of the spacecraft will obviously be mainly driven by both launch survivability and performance on-orbit. However, consideration must also be given to the integration process, and how the spacecraft may be handled on the ground. Inclusion of ground handling hard-points allows the spacecraft to be manoeuvred during assembly, test and integration with the launcher, and safely transported between sites.

The design should also allow access to internal subsystems and payload, so that equipment can be removed and replaced if necessary. The method of integration must be carefully considered during design, particularly when ensuring that fastening points are accessible. Designing the spacecraft so that sections can be assembled in parallel will help to reduce the duration of the integration process.

4.3.1.5 Materials

Selection of suitable materials for the spacecraft structure is driven by a number of requirements. These requirements may be divided into the following areas:

- Structural
- Manufacturing
- Environmental

The structural requirements arise from the mechanical properties required of the materials to be selected: specific strength, specific stiffness. These in turn come from the launcher requirements described previously, the spacecraft configuration and equipment to be supported, and the spacecraft mass budget.

Manufacturing requirements arise largely from cost and schedule concerns. The materials selected, the intricacy and number of parts to be made, and the manufacturing techniques required will determine the ease and cost of manufacture.

The environmental requirements arise from the particular aspects of the space environment that can adversely affect materials. These are covered in the environmental requirements section.

Structural materials used for small satellites

A traditional material for the primary structure of small spacecraft is machined aluminium. It is popular due to its relatively low cost, light weight and high strength. It can be machined precisely to produce stiff ribbed panels, space-frame struts, and equipment boxes. Sheet aluminium can also be folded to shape where lower precision is acceptable. This sheet-forming is cheaper than machining. A major problem with aluminium for certain types of application is its high coefficient of thermal expansion.

As small satellites are becoming more widely used, more advanced materials are now starting to be employed. Composites such as carbon fibre reinforced epoxy have been employed where a very low coefficient of thermal expansion is required, for example where telescope optics must remain precisely aligned. Composites remain a more expensive option, and have further disadvantages for small missions: inserts are required for fastenings – this makes late-stage changes (which can be a possibility in budget missions) much more difficult than with metal structures.

Where panels are required, composite sandwich materials, with a cellular honeycomb centre (often aluminium) between two thin skins, are often used. These panels are very light and stiff, but again require inserts where attachment points are needed.

The electrical conductivity of materials can also be a consideration. Where structures are formed from insulating materials, differential charging can occur in orbit, causing potentially hazardous arcing. When a metal structure is used, this hazard is generally avoided.

Special considerations must be made in selection of materials, adhesives and lubricants, as many substances routinely used for terrestrial applications out-gas in vacuum. Condensation of out-gassing products onto optical lenses rapidly degrades their performance, and therefore must be avoided as far as possible. The metals cadmium, zinc and tin are volatile in space conditions, and are not generally permitted. Certain plastics such as PVC and PVA are subject to outgassing. A large amount is know known about the behaviour of materials commonly used in space; accepted and proven materials should not give any problems.

4.3.1.6 Structural configuration of small spacecraft

A main driver for the structural configuration of a spacecraft is that it must maintain integrity throughout the extreme environment of launch. This environment takes the form of axial loads, both steady-state and transient, from the acceleration of the launcher, plus mechanical and acoustic vibrations. In addition, shock loads occur when pyrotechnic release mechanisms are fired.

To withstand the high axial loads, spacecraft are often designed around a central thrust tube, which attaches directly to the launcher interface and carries the loads up through the structure. This may be a solid cylinder, or a framework structure. The other spacecraft components are then mounted to this, either inside or outside, and often on 'decks', or shelves. An example of a framework structure with payload decks is shown in Figure 4-13. This is the bus structure of the LANL FORTE spacecraft, and is constructed wholly of composites.

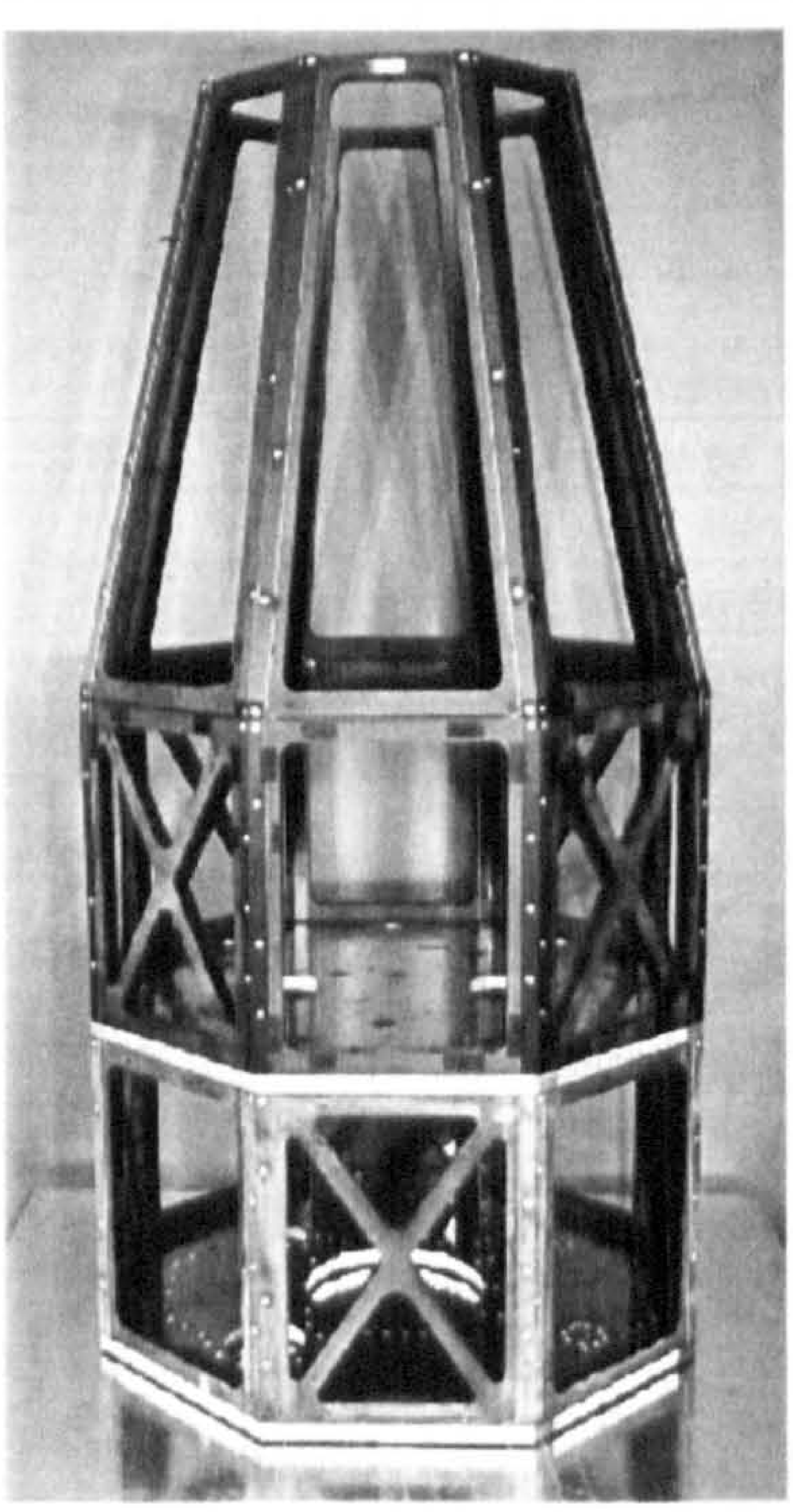

Figure 4-13 The FORTE spacecraft primary structure [Image: LANL]

The space-frame structure tapers at the top end to fit the fairing of the Pegasus on which the spacecraft was launched. In its final configuration, the decks supported the subsystems and payloads, and the outer frame supported the solar array panels.

It is also possible to use subsystem or payload modules as load-bearing structures. This is an area of interest for a modular spacecraft, where the individual modules containing the equipment fit together and themselves make up the primary structure. It is planned that this idea will be investigated in the design phase of this research.

Configuration also depends on the type of stabilisation to be employed – a spin stabilised spacecraft will generally be symmetrical about its spin axis, and be spinning about the axis of greatest moment of inertia. This will make the spin stable.

A 3-axis stabilised spacecraft may be more asymmetric, with deployed arrays and instruments. However, it must still conform to the moment of inertia and centre of mass requirements of the launch vehicle. For precise pointing applications, flexibility of the deployed spacecraft should be avoided.

4.3.2 COMMUNICATIONS AND DATA HANDLING

This subsystem forms the vital link between the space and ground segments. Data from the spacecraft consists of both payload data, which must be passed to the experimenter or customer, and housekeeping data – information for monitoring the health and status of the platform as a whole. Data sent to the spacecraft consists of commands, and in some cases software uploads or patches. This must be distributed to the correct subsystem. Transfer and management of data on the spacecraft is carried out by the on-board data-handling system.

4.3.2.1 Communication of data

When planning a mission, it is necessary to know what amounts of data will be produced, and how often it will be possible to download it. This will be dependent on the payload, the orbit, and the ground stations used. The amounts of data to be stored on board, and the rate at which it must be relayed to the ground can be calculated. The inputs that must be considered when producing a requirements specification for the communications subsystem are shown in Table 4-22.

Parameter	**Impact on communications subsystem**
Downlink data rate	Transmitter power, frequency, antenna type, modulation, coding
Uplink data rate	Ground station, receiver frequency, antenna type, modulation, coding
Ground station(s)	Total ground pass duration (time available to downlink data)
Power budget	Transmitter & receiver power, transmitter duty cycle
Orbit altitude	Link budget, ground pass durations
Orbit inclination	Ground track (and therefore ground station pass geometries)
Orbit eccentricity	Link budget
Spacecraft attitude	Position of antennas, capability to point at ground station
Mission budget	Limits type of equipment that may be used
Mission lifetime	More redundancy for longer missions
Requirement for ranging	Transponder with ranging capability
Phase coherence	Use of coherent turnaround for the downlink generation allows computation of s/c range-rate, for faster signal acquisition.

Table 4-22 Requirement input parameters for communications subsystem

All of the parameters mentioned are designed iteratively, through trade-off studies, as it is often impossible to have exactly what is initially requested. For example, a certain combination of orbit choice, ground station availability and data rate may require an impossibly high RF power budget and antenna size.

Most small spacecraft use S-band communications. Up-links are generally of lower data rate than the downlinks, often a few kbps. The downlink may be in the region of several Mbps. This order of data rate is easily achievable from LEO using an S-band transmitter and small, low-gain "patch" antennas. Higher rates can be achieved in the

S-band if higher gain cone or parabolic antennas are used. If a much higher data rate is required, an X-band link may be used.

There is a trade-off between antenna gain and required transmitter power. A higher-gain antenna radiates into a narrower beamwidth, thereby allowing a lower transmitter power to give the same power at the receiver. However, higher-gain antennas are generally larger, heavier, and more complex than low-gain or omnidirectional antennas. They also need to be actively pointed towards the ground receiver.

The data rate requirements obtained in the mission-specific analysis range from very low, a few Kbps, for the spacecraft data of communications missions, to tens of Mbps for the imaging payload data of Earth observation missions. Up to the order of 2-3Mbps can be achieved via an S-band system. The higher-performance requirements may be met by the use of an X-band system. A suitable target to aim for in the design will be a 3Mbps rate for general applications, and a 50+Mbps high-capability rate. This will compare favourably with rates available from other small commercial platforms. A low-rate, possibly UHF system may be investigated for spacecraft housekeeping data on communications missions.

The requirement for downloading payload data also drives the number and location of the ground stations. The option to downlink via more than one ground station allows more data to be transmitted whilst using a lower rate, due to the longer contact times. In LEO, ground pass durations are very short: around ten minutes for a 600km orbit (assuming AOS/LOS occurs at 5° above the horizon). Therefore, a requirement for more than one station would be assumed for most missions.

The requirement for ranging is likely to be less critical with the common usage of GPS onboard spacecraft. It will be assumed that the platform will carry GPS, and that this will be used for position determination.

Flexibility in terms of mounting configurations for both antennas and payloads will be necessary.

4.3.2.2 On board data handling

This subsystem collects the data produced by the payload and the housekeeping monitoring instruments, stores and formats it in readiness for its transmission to the ground as telemetry. It is also responsible for reception and decoding of commands, and their distribution to the appropriate areas. The system responsible for handling commands and data may consist of a single, central unit, or a system of distributed units.

The requirement input parameters that must be considered for the data handling subsystem are shown in Table 4-23.

Parameter	Impact on data handling subsystem
Number of payloads	No. of data channels, no. of command addresses, data communication architecture, mass of cabling
Number of subsystem equipment items	As above
Command format	Command processing s/w & h/w
Telemetry formats	Telemetry processing s/w & h/w, A-D converters, harness types & configuration
Payload data interfaces	As above
Payload data rates	Data buffering, bus protocols
Data buffering within payloads	Polling of payloads for data collection
Requirement for time-tagged commanding	Implementation in onboard s/w, requires provision of time signal
Sensor sampling rates/schemes	Polling schemes
Testability	Inclusion of umbilical connectors for testing on ground.
Requirement for data processing	Implementation of processing algorithms in the onboard s/w
Time stamp for telemetry	Need onboard time signal, need to get time signal updates e.g. from ground, GPS etc
Requirement for onboard computer	
Onboard time accuracy	Frequency of time signal updates, drift of local oscillator
Data storage requirement	Provision of solid-state mass memory
Degree of onboard autonomy	Implementation in onboard software Determines sophistication of processors required
Spacecraft mass	Choice of hardware, including onboard data comms medium (wire, fibre-optic, RF), no of processors, amount of mass memory
Available power	Choice of hardware Duty cycles Hibernate states
Mission budget	Technology & qual. level, use of hardened electronics
Redundancy	No. of redundant systems – affects mass & cost

Table 4-23 Requirement input parameters for data handling subsystem

The data handling subsystem must interface to all of the onboard subsystems and payloads, therefore it is difficult to design a truly "generic" system. Numbers of inputs and outputs, volumes and rates of data, and the types of processes that must occur on board may not be quantified in advance. However, increasingly modular electronics enables a much greater flexibility than before, and an increased ability to make changes in software to accommodate different requirements. An over-capacity in terms of number of payload channels that can be handled, or volume of data that can be stored, will probably make minimal impacts in terms of extra mass, volume or cost.

From the mission-specific requirement analysis performed earlier, it was found that space physics missions are likely to have a higher number of different payloads per mission. Many small satellite missions may only have two or three payloads, but a flexible multipurpose platform must have the capability to accommodate more. This may be achievable via a range of interchangeable I/O boards. For flexibility, the data handling system should be able to handle at least ten payload inputs, preferably more. A ten-payload capability would handle all the missions studied except for the POLAR mission. This was a NASA MIDEX mission and its parameters would fall at the very top end of the scope of the proposed platform. If the platform were to be used at the edge of its performance area, some slight re-design may be acceptable. This would involve additional cost and some increase in delivery time.

The physical method and protocol by which the payloads interface with the data handling subsystem also requires consideration. The key requirement is that it should be a recognised, commonly-used system. Examples of such protocols are RS232, RS422, and the MIL-STD1553/1773 bus standard. This makes it easier for payload suppliers to develop and test their instruments, and is also more likely to have been designed for in existing equipment.

The above considerations equally apply to the bus subsystems.

There may also be a requirement to provide a data-buffering service for payloads without internal buffering. Some payloads may have the capability to store their data until polled by the onboard computer, but if they do not, an alternative provision must me made available. Additionally, A-D conversion may be required for some sensors. Discrete processing dedicated to the payloads may also be required.

The level of data storage required on board is determined by instrument data rates and the regularity and downlink data rate of ground passes. Generally, a greater storage capability will always be desirable, as it offers some flexibility in operations and avoids data loss in the event of temporary problems with the communications link. Other commercially-available platforms offer data storage ranging from 2 –100 Gbit. Improved solid-state storage has facilitated such increased levels. To be competitive, the proposed platform should be capable of offering a comparable range of onboard storage capacities.

4.3.3 ATTITUDE & ORBIT DETERMINATION AND CONTROL

Most spacecraft require some form of stabilisation, to maintain a preferred and known attitude. This may be to allow payload instruments to be pointed at specific targets, such as the Earth's surface, or a particular star system, or to allow the antenna to point at the ground station. Attaining and maintaining a desired orbit may also be required.

The attitude and orbit control subsystem is perhaps the spacecraft element with the most scope for diversity. Depending on mission requirements, AOCS may consist of no more than a deployed boom and simple sun sensors. At the other end of the scale, it may require thrusters, reaction wheels and an array of high-accuracy direction and rate sensors. The parameters that must be considered when defining requirements for the attitude and orbit control subsystem are shown in Table 4-24.

Parameter	Impact on AOCS subsystem
Pointing accuracy	Accuracy of actuator hardware, attitude sensors and control system
Pointing knowledge	Accuracy of attitude sensors
Payload pointing direction(s)	Spacecraft attitude(s) e.g. nadir-pointed, inertially-fixed, etc
Slew rates	Maximum torques from actuators
Stability requirement	Provision of damping systems to reduce jitter, nutation, and perhaps libration

Mission lifetime	Mass of fuel, reliability of equipment
Spacecraft mass	Types of sensors and actuators available at low mass, minimum impulse bits
Mission budget	Choice of hardware limited by cost. Complex software algorithms also costly
Power budget	Limits available actuators and sensors.
Magnetic cleanliness	Magnetorquers & electric motors have residual magnetic fields. May not be suitable if very sensitive payloads are on board.
Orbit accuracy	Accuracy of orbit determination method (e.g. GPS), provision of orbit control (i.e. thrusters)
Position knowledge accuracy	Accuracy of orbit determination
Orbital altitude & eccentricity	Limits use of gravity-gradient and magnetic actuation methods. Eccentricity introduces oscillatory gravity gradient effects
Orbital inclination	Local magnetic field direction for magnetic torque actuators
Operational modes and duty cycles/ schedules	Fuel mass required/power required,
Presence of payload optics	Avoid impingement of thruster plumes

Table 4-24 Requirement input parameters for attitude and orbit control subsystem

When designing a bespoke spacecraft, over-engineering the subsystem should be avoided for missions that don't need high accuracy; as this will obviously drive up cost and subsystem mass. For a generic platform, however, the AOCS may need to err on the side of too much capability. The difficulty comes, therefore, in choosing a level of control performance that would be sufficient for most missions, but is not uneconomic for less demanding ones. The criterion must be, would it be cheaper to redesign the platform with simpler AOCS equipment, with the extra time, effort, and risk involved, than to accept the existing solution?

The solution is probably to make actuator hardware available in performance "steps" which have similar electrical & mechanical interfaces, so if necessary the spacecraft capability can be stepped up to the next level. This will be addressed in the following chapter.

For many of the missions identified, the key requirement is to maintain one particular attitude (such as nadir pointing) to the defined accuracy and stability. This impacts more on the choice of attitude sensors, and the onboard attitude processing and control algorithms.

In addition to performing operational manoeuvres, the attitude control system must also be capable of counteracting environmental disturbance torques. These are torques acting on the spacecraft as a result of external environmental influences:

- Gravity gradient effects
- Solar radiation pressure
- Aerodynamic drag
- Magnetic field interaction

or due to unwanted/unavoidable onboard phenomena:

- Thruster misalignment and/or mismatching
- Uncertainty in centre of gravity (this is likely to be minimal in a small spacecraft)
- Sloshing of fuel (or other onboard liquids e.g. cryogenics)
- Rotating mechanisms (e.g. solar array deployment mechanisms)
- Flexibility of structures

However, it may be reasonably assumed that the maximum torque levels needed to fulfil mission requirements will be higher than those needed to overcome environmental disturbances. It is therefore recommended that the platform capability be designed around fitting the different mission requirements identified, with the availability of some "spare performance". The resulting attitude performance increments will then allow for missions with a range of requirements, which may arise from different sources. For example, a particular mission may have lower "operational requirements", in terms of slewing torques for target pointing, but have high requirements for environmental disturbance compensating torques, due to its configuration.

4.3.4 POWER

All spacecraft require electrical power for the onboard equipment, but the levels required vary widely between missions. The usual procedure for defining the power subsystem is to identify the primary requirements from the mission type and the payload characteristics. The requirement input parameters that must be considered for the power subsystem are shown in Table 4-25.

Parameter	Impact on power subsystem
Orbit altitude	Determines max. eclipse duration, and hence power storage requirement (and no. of charge-discharge cycles)
Orbit inclination	Sun vector geometry, and hence solar array design
Launch epoch	Initial sun vector geometry
Duty cycles (payload & platform)	Sizing for arrays and batteries
Payload average/peak power requirement	Sizing for arrays (and batteries if payload to operate in eclipse)
Platform average/peak power requirement (BOL and EOL)	Sizing for arrays and batteries
Minimum survival power	As above
Payload required voltage(s)	Bus voltage, and regulation/conditioning system
Mission lifetime	Selection of solar cells. Sizing of solar arrays for necessary EOL performance. Sizing of shunts for dumping excess power at BOL. Battery selection for required cycle-life.
Spacecraft size & configuration	Surface area available for body-mounted arrays. Envelope available for stowed-deployed arrays.
Launch vehicle	Envelope for stowed solar arrays.
Cost constraints	Limits on complexity of deployment devices, and technology of solar cells.

Table 4-25 Requirement input parameters for power subsystem

The amount of power that must be made available for the payload is difficult to estimate unless the mission is quite well defined. This is because it not only depends on the physical power consumption of the payload(s) in question, but also the operational regimes, duty cycles etc. Different payloads also require different types of supporting subsystems, which may consume more or less power. In the case of designing a generic platform, it is therefore a problem to choose the baseline level of power to be made available to the payload.

The orbit, orbit epoch, and spacecraft attitude also heavily affect the definition of the power subsystem. Different altitudes and eccentricities determine maximum eclipse duration, inclination and epoch determine sun angle geometry.

Figure 4-14 gives mass and power data for a range of small spacecraft missions, and illustrates the wide range of power generation requirements. Also shown is an estimate of the maximum level of power that may be generated from an "idealised" spacecraft with only body-mounted solar cells, using the following assumptions:

- Spacecraft is cubic, with density of $150kg/m^3$
- Best-case illumination is assumed (i.e. sun vector normal to one spacecraft vertex, thus lighting 3 faces)
- Entire illuminated surface of spacecraft is covered with GaAs solar panels
- GaAs cell efficiency is 18.5%
- Solar radiation flux is $1358W/m^2$
- A 3-year mission lifetime, with 2.75% cell degradation per year
- Inherent loss factor of 0.77 due to cell connections reducing active area, and temperature-related losses
- Orbit average available power is for 500km orbit, with 35min eclipse
- Constant power use is assumed over whole orbit (sunlit & eclipsed)

The power levels indicated in Figure 4-14 are the power *generated* during the sunlit part of the orbit (the higher curve), and the power *available* to be constantly used by the spacecraft payload and systems (the lower curve). Available power is significantly lower than that generated due to the power required for battery charging, and the power drawn from the batteries during eclipse.

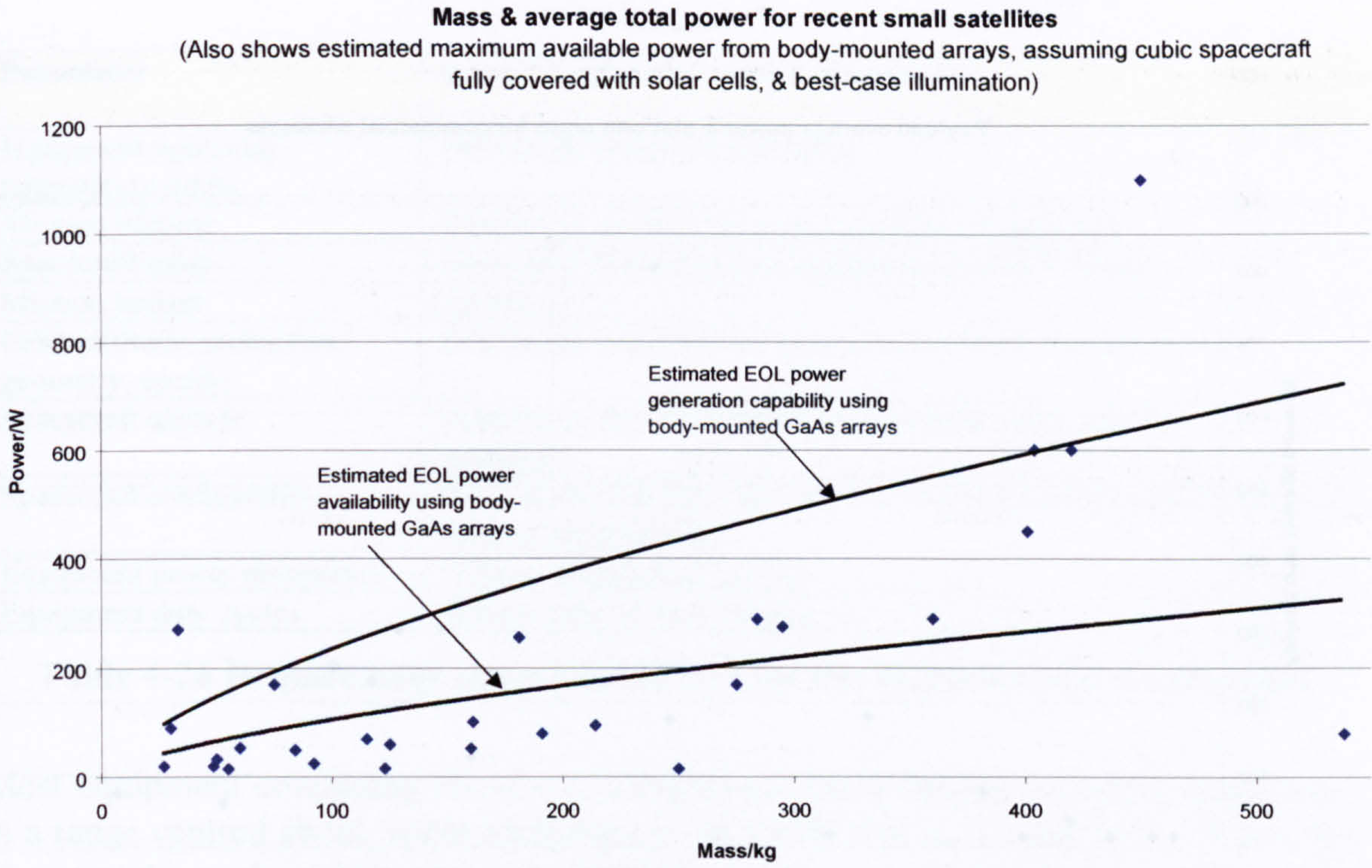

Figure 4-14 Mass and power of recent small satellites

The figure shows that while some of the spacecraft investigated could feasibly rely on body-mounted solar cells, deployed arrays would be necessary for many others, with some being borderline. A particular problem is that, at a certain power requirement level, there will be no choice but to switch to some form of deployed array, due to area limitations for body-mounted arrays. The change from body-mounted to deployed arrays has considerable impacts on the spacecraft design as a whole. If a range of "add-on" options will be used to enhance the capability of a basic platform, then this would have to be considered. These areas will be addressed in the following chapter.

Figure 4-15 shows the payload power capabilities of currently available commercial small spacecraft. It is the available payload power, rather than the total bus power, that will be of interest to potential customers. From the mission-specific analysis conducted earlier, a range of power requirements was identified, from <100W to 500+W. At the high-power end, more power is seen as offering competitive advantage. From the graph, it can be seen that only one commercial platform offers payload power of over 500W.

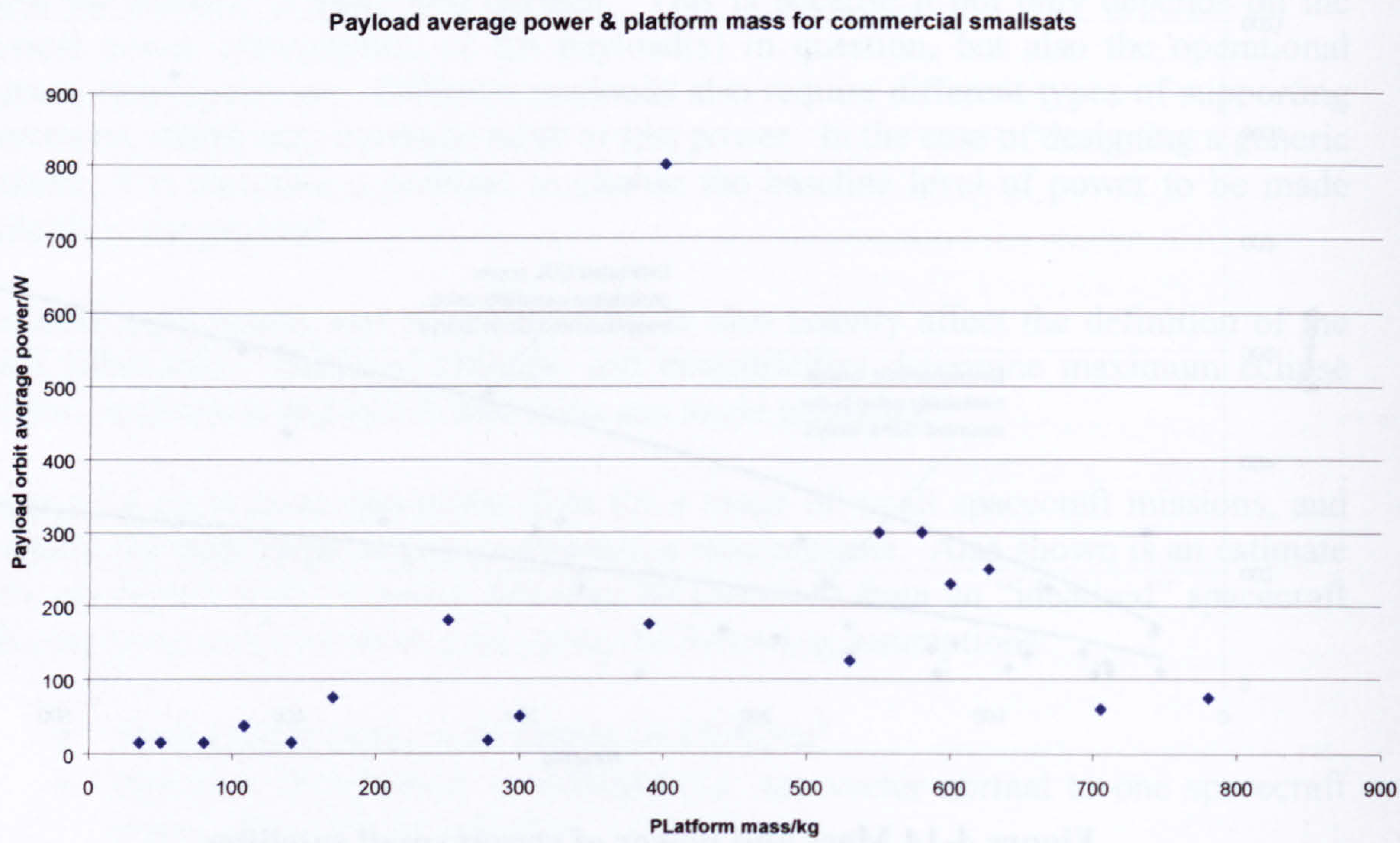

Figure 4-15 Mass of commercial small satellite platforms and power available for the payload

This would suggest the requirement to offer a range of different levels of power capability, using different levels of sophistication in the power subsystem. The modular approach should allow a good degree of tailoring to be available in the power subsystem, using different cell types and areas, and different energy storage solutions. Standard interfaces between the different component modules will also be required, to allow additional or replacement modules to be introduced.

4.3.5 ENVIRONMENTAL

These are the factors that concern the space environment, the environment on board the spacecraft, and the interactions between them.

4.3.5.1 Thermal control

Design of the thermal control subsystem for a generic spacecraft should present fewer problems than many of the other subsystems, as most spacecraft components will share at least some of their nominal operating range. However, although the desired thermal range may be similar for different missions, varying mission parameters will impact on the exact method required to achieve it. The parameters impacting on the thermal control subsystem and its requirements are shown in Table 4-26.

Parameter	Impact on thermal control subsystem
Equipment operating temperature limits	Defines the target thermal range
Mission lifetime	Determines amount of coolant required, if applicable
Spacecraft mass	Constrains thermal control equipment that may be used
Mission budget	As above
Orbit (altitude, inclination, geometry, epoch)	Determines solar thermal inputs and cycles for the spacecraft
Spacecraft attitude	Determines thermal balance of spacecraft with respect to solar radiation
Spacecraft configuration	Determines heat paths through the structure, thermal equilibrium with external environment
Equipment power dissipation	Affects internal heat balance
Equipment duty cycles	Affects internal heat balance

Table 4-26 Requirement input parameters for the thermal control subsystem

Most equipment containing electronic components has a desired operating temperature in a range centred about 'room temperature' on Earth, that is, around 20ºC. Most other components, such as batteries and mechanical devices will operate nominally at this temperature. Therefore, spacecraft are generally designed to maintain temperatures within this area. Nominal operating ranges may be broader or narrower, depending on the type and design of the equipment. Some examples of such ranges for space subsystems are given in Table 4-27.

Equipment item	Nominal operating temperature range/ °C	Remarks
Transponder	-30 to +65	
Antenna	-65 to +95	
Reaction wheel	-30 to +50	
Magnetometer	-80 to +80	
Ni-Cd cells	+5 to +20	
Ni-H cells	-5 to +20	
Lithium cells	0 to +40	
Solar arrays	-105 to +110	Efficiency is temperature-dependent
Onboard computer	-10 to +50	
Most electronics units	-10 to +55	
Propulsion system	+7 to +55	Thrusters to +65
Structures	-45 to +65	Smaller range where thermoelastic deformations will adversely affect critical alignments
Pyrotechnics	-100 to +120	
Internal harness	-15 to +55	
External harness	-100 to +100	
Electric motors	-45 to +80	
Optical payload sensor	+15 to +25	

Table 4-27 Typical temperature requirements for space subsystems and payloads [11,22,39]

Difficulties may arise when a particular payload has specific thermal requirements that are outside the normal range. Certain types of payload experiments may require cooling, sometimes to very low temperatures. They must then generally be thermally decoupled from the rest of the system, as far as possible. However, in some cases, the thermal isolation system will be built into the payload module, and it may be integrated thermally without any major impact on the spacecraft. For example, an infrared payload module may have an internal sensor thermal range requirement of –200 to –80°C, but the range for the whole module is –40 to +30°C.

4.3.5.2 Electromagnetic compatibility

As the platform may be used to support instruments that are highly susceptible to electromagnetic interference, (e.g. space physics payloads), the control and test scheme must reduce the effects wherever possible. This will include aspects at the design phase, such as the requirement for using shielded twisted pairs for power cabling, and electrical data connections. For compatibility and avoidance of coupling between subsystems and payloads, a frequency utilisation plan must be used early in the mission planning phase, defining the emitted frequencies and susceptibilities of both payload and bus subsystem items. This type of planning and analysis will require data to be available for the payload subsystems, obtained via prior EMC testing. This is a further area where an approach that uses pre-tested modules will confer an advantage.

4.3.5.3 Cleanliness

This covers the requirements for assembly and testing on the ground and onto the launcher, and operation in the space environment. On the ground, cleanrooms giving a controlled environment will be required for assembly and test operations, to avoid contamination by particulates.

4.3.5.4 Requirements for operation in the space environment

The space environment has particular aspects that present challenges for spacecraft design. These are:

- Vacuum
- Radiation
- Atomic oxygen
- Temperature cycling
- Spacecraft charging
- Debris/micrometeoroid impact

Table 4-28, on the following page, describes the effects of these environmental aspects, and the resultant requirements placed on space materials and equipment.

Aspect	Effects	Requirements/constraints
Vacuum	Sublimation of metals	Cannot use Cd, Zn, Sn.
	Cold welding of metals	Must be considered when designing mechanisms.
	Outgassing	Must use resins/adhesives/coatings that have acceptable outgassing rates. Potting compounds must be de-gassed to remove bubbles.
	Evaporation and 'creep' of lubricants	Lubricants containing graphite cannot be used.
	Hardening of adhesive tapes, plastic films and thermoplastics	Use only tested & qualified tapes. PVC, acetate, cellulose, polyamide, PVA cannot be used.
	Depolymerisation of rubbers	Use only tested & qualified rubbers.
	Expansion of trapped gas pockets.	Use venting holes.
Radiation (particle)	Altering of properties of some composites, via modification of resin matrix	Property changes must be addressed if mission is long duration.
	Evolution of corrosive products from lubricant oils and greases.	Lubricated mechanisms may require radiation shielding. Lubricants must be carefully selected.
	Degradation of surface paints and other coatings.	Specially-selected coatings must be used.
	Hardening and discolouration of thermoplastics.	Choose less susceptible plastics; PTFE should be avoided in high-rad orbits.
	Loss of transparency of glass.	
	SEU/SEL events in computer hardware.	Use shielding. Employ radiation-tolerant devices with EDAC.
Radiation (UV)	Darkening of adhesives, coatings, and paints.	Select materials for optics with care.
	Discolouration of plastics and films.	
Atomic oxygen	Surface attack of metals; can cause cracking.	Use metals which form stable oxide layers; this provides a protective coating.
	Attack of composites (resin matrix and then fibres).	Composites may require a protective coating, particularly for long duration missions.
	Degradation of solid lubricants.	Enclose lubricated areas to shield lubricants.
	Attack of thermoplastics and plastic films.	Avoid hydrocarbon-based plastics in susceptible areas; use silicones.
Temperature (thermal cycling)	Can change CTE of composites.	
	Can cause failure due to thermal shock or thermal fatigue.	
Spacecraft charging	Can result in spark discharge within the spacecraft.	Maintain electrical conductivity around the structure to avoid large potential differences.
Debris/ micrometeoroid impact	Damage to spacecraft from hypervelocity impacts	Estimate impact probabilities using models, and provide shielding if necessary/feasible.

Table 4-28 Requirements and constraints arising from the space environment

The considerations in the table above are particularly important where a spacecraft is being developed at minimum cost, and there is a desire to use commercial components and materials. Care must be taken to avoid unsuitable materials that may be used in commercial equipment; certain items may require modifying, replacing or shielding. For example, commercial circuit boards may have connectors containing cadmium, which would have to be changed for gold-plated replacements.

4.3.5.5 End of life de-orbit

The increasing build-up of post-mission spacecraft and other debris in near-Earth space has led to many calls for satellites to be de-orbited at the end of their missions. Although any regulatory system would be difficult to enforce, responsible space users may be expected to adopt this approach where possible in the future. This study will not address particular methods of de-orbiting, but the possible requirement to add on a device to remove the spacecraft at the end of life should be considered.

4.4 MISSION REQUIREMENTS

These cover what the spacecraft will require to perform its intended mission. They are derived from the mission-specific requirements analysis performed in Section 4.2.

4.4.1 ORBIT

The orbit requirements for particular mission types have been addressed in the mission-specific analyses, and stem from the impact of factors such as sun geometry, altitude & inclination, eccentricity, radiation environments, Earth coverage & repeat cycles, and eclipses. Higher altitudes give longer ground passes, lower aerodynamic drag (and therefore longer lifetimes without orbit-boosting thrusters), slightly shorter maximum eclipse durations, and better Earth coverage. Lower altitudes give more benign radiation environments, are easier to reach (in terms of launcher performance), and give better instrument resolution for Earth observing missions. The effect of orbit on lifetime is addressed further in the next section.

4.4.2 LIFETIME

Mission lifetime that can be reliably achieved by a given spacecraft is influenced by a number of factors. These are shown in Table 4-29.

Influencing factor	**Remarks**
Orbit altitude	Orbit decays due to aerodynamic drag. Propulsion required to maintain altitude.
Rate of consumable usage	Trade-offs between mass of consumables (i.e. propellant, cryogenic coolant) carried and mission lifetime.
Wearout rates	Wear in mechanisms, cycle life of batteries, degradation of solar cells.
Failure rates, reliability	Mean time between failure of components and equipment.

Table 4-29 Factors influencing mission lifetime

As described in the preceding section, mission lifetime is heavily influenced by the orbital altitude. Low orbits will decay rapidly without station-keeping propulsive manoeuvres. The lower the orbit, the greater the delta-v required to maintain altitude. Lifetime is also therefore influenced by the mass of fuel that can be carried, lower orbits requiring greater propellant mass for a given lifetime. As the spacecraft should be capable of achieving lifetimes of 5 years or more (a requirement identified for communications and Earth observation missions), and to be suitable for use in a range of orbit regimes, it will require a propulsion option to allow for orbit maintenance.

An idea of the amount of fuel required to maintain a spacecraft in lower orbits is given by Figure 4-16. For illustration, a hypothetical 250kg (wet mass) spacecraft is assumed to be maintained in orbit for one year at solar maximum. This shows that for orbits below 400km or so, the propellant mass requirement will cause difficulties where longer lifetimes are needed. Of course, electric propulsion is a possible alternative here, but this has impacts on the spacecraft power requirement. This could lead to a larger solar array area, potentially affecting the ballistic coefficient of the spacecraft and further increasing the orbit-maintenance delta-v requirement.

These considerations must be taken into account when the performance of the final spacecraft design is being specified.

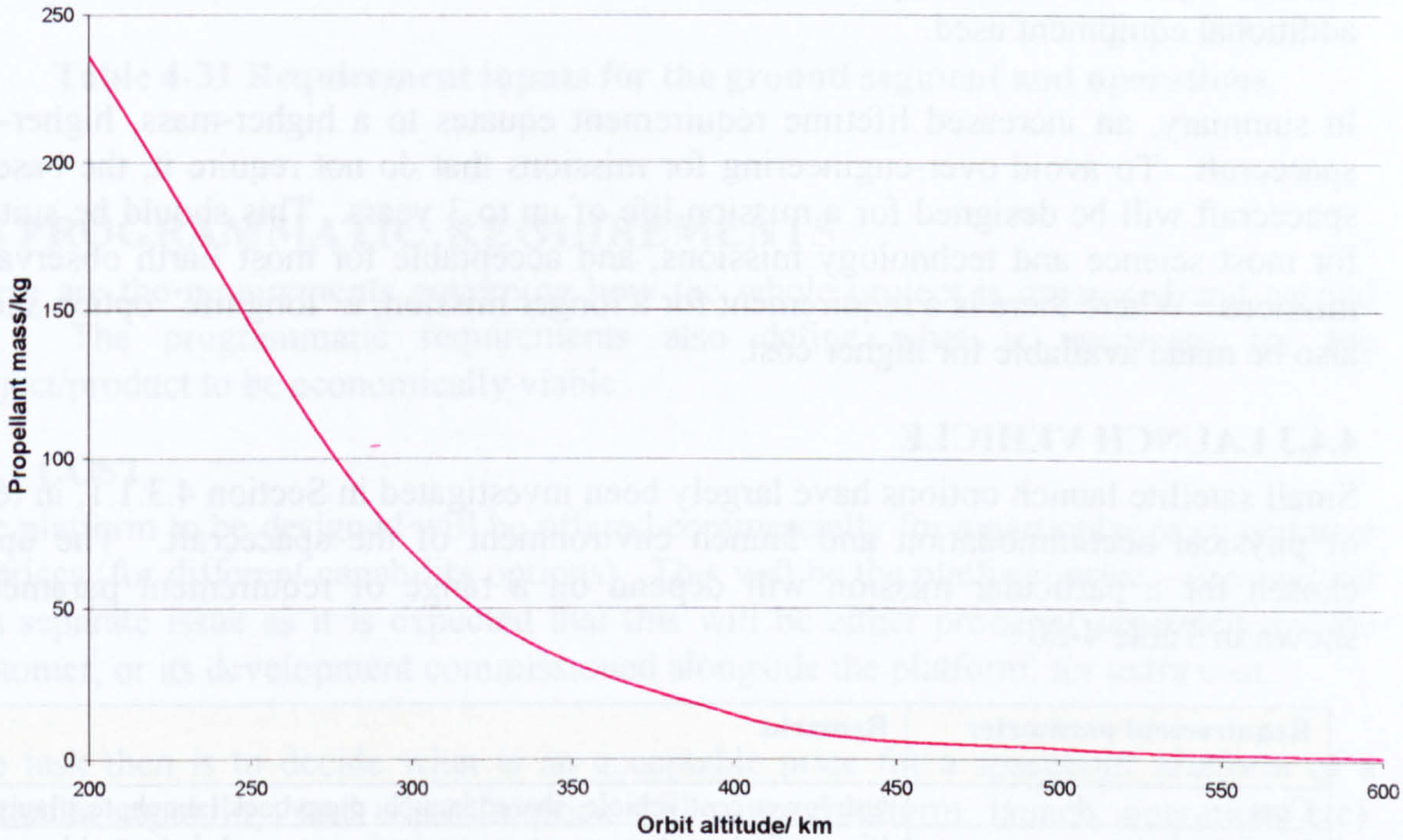

Figure 4-16 Estimated propellant mass to maintain orbit altitude for 1 year at solar maximum. [Based on a 250kg spacecraft, using cold gas propulsion, and with ballistic coefficient of 200kg/m^2. Delta-v requirements from [39].]

Another factor influencing lifetime is the use of onboard cryogenics for cooling detectors. Once all the cryogen has evaporated, the detector can no longer produce useful data. This is likely to be a limiting factor for lifetime of astronomy payloads, and some Earth observation payloads. However, this is a payload feature and places no direct requirements on the platform (aside from accommodation of cryogenics).

The final key influence on mission life is the lifetime and reliability of the spacecraft components and equipment themselves. Component and equipment lifetime is influenced by wearout rates (e.g. for batteries, bearings in moving parts, degradation of solar cells etc), failure probabilities, and the level of redundancy employed. Failure and wearout rates will be influenced by their operational environment (particulate radiation, atomic oxygen, micrometeoroid – see 4.3.5.4). Quoted failure rates for components may often specify radiation dose. For relatively short-duration missions, of 2-3 years, in LEO, radiation doses are generally survivable by COTS-level electronics[35]. This has been demonstrated on numerous small missions. However, to attain required reliability for longer missions and/or higher-dose orbits, radiation-hard components may be required.

Due to the higher cost and more involved procurement processes for rad-hard parts, and the likelihood that the majority of missions are likely to be of order 2-3 years, it is recommended that COTS parts be used as a baseline, with the option to "upgrade" (for a cost increment) to the higher-tolerance parts when specifically required.

Higher reliability (and hence longer design life) may also be achieved by the use of redundancy. This inherently increases the mass (and cost) of a subsystem, due to the additional equipment used.

In summary, an increased lifetime requirement equates to a higher-mass, higher-cost spacecraft. To avoid over-engineering for missions that do not require it, the baseline spacecraft will be designed for a mission life of up to 3 years. This should be suitable for most science and technology missions, and acceptable for most Earth observation missions. Where there is a requirement for a longer mission, a "long life" option should also be made available for higher cost.

4.4.3 LAUNCH VEHICLE

Small satellite launch options have largely been investigated in Section 4.3.1.1, in terms of physical accommodation and launch environment of the spacecraft. The option chosen for a particular mission will depend on a range of requirement parameters, shown in Table 4-30.

Requirement parameter	Remarks
Cost	Use lower cost vehicle, shared launch, piggyback launch, test launch.
Launch window	If specific launch parameters required, may need dedicated launch, or to be the primary passenger if a sharer can be found.
Launch site	May arise from political requirement to use particular launcher, or from desired orbit inclination.
S/C mass, dimensions	Impacts on range of launch options available.
Campaign requirements	E.g. late access to spacecraft – may require negotiation if S/C is not primary passenger.

Table 4-30 Requirement inputs impacting choice of launch vehicle

The aim of the design is that it should be compatible with launch on a range of different vehicles. This then allows a decision to be made, on a mission-by-mission basis, as to which is the best launch option for that particular spacecraft.

4.4.4 GROUND SEGMENT, OPERATIONS AND AVAILABILITY

This area is likely to require definition on a mission-by-mission basis. Requirements will vary widely, as will the possibilities for negotiating use of existing facilities (this may depend on the type of mission being conducted, and the customer – e.g. a military small satellite could make use of domestic military communications infrastructure, but a foreign customer would be unlikely to be allowed access). The key requirement inputs are shown in Table 4-31.

Requirement parameter	Remarks
No. & position of ground stations	To give required contact frequency with s/c Use of dedicated, purpose-built stations vs existing facilities
Data timeliness & dissemination	Means of data transfer to experimenters etc
Autonomy of s/c and ground stations/ control centres	Impacts on staffing requirements, and hence costs
Ground station characteristics	Impacts on link design and s/c communications subsystem
Security	To prevent interference with the s/c, disruption of systems
Mission operations schedules	
Spacecraft commissioning	
No. of spacecraft	To perform the required mission

Table 4-31 Requirement inputs for the ground segment and operations

4.5 PROGRAMMATIC REQUIREMENTS

These are the requirements governing how the whole project is organised and carried out. The programmatic requirements also define what is necessary for the project/product to be economically viable.

4.5.1 COST

The platform to be designed will be offered commercially for a particular price or range of prices (for different capability options). This will be the platform price – the payload is a separate issue as it is expected that this will be either procured separately by the customer, or its development commissioned alongside the platform, for extra cost.

The task then is to decide what is an acceptable price for a spacecraft platform of a particular capability, and what the price will cover (platform, launch, operations etc). This will then form one of the constraints for the design. The expected customer groups and their characteristics have been addressed in the business rationale section, as have the costs of previous spacecraft and current commercial platforms.

The use of the modular platform should enable cost "steps" so that the platform can be applied to both basic and more advanced missions. Where an extremely basic mission is intended, with cost the only driver, it may not be applicable to attempt to target such missions. The hardware used for such missions may be below the scope of that used for the rest of the envelope of applicable missions. Based on previous missions and civil budgets, suitable cost ranges are expected to be:

- "Basic" platform, lowest performance level: $5-10m
- Higher capability, larger payload mass, more power etc: $10-20m
- Advanced platform, providing more "mission tailoring": $20-25m

These aim to bring a high-capability platform in at below the costs often incurred by using a custom-built platform for the more advanced missions, while also providing options for the more basic mission requirements. These are general guidelines, which should keep the platform competitive on cost. The technical flexibility, and resulting schedule benefits should also enhance competitiveness.

4.5.2 SCHEDULE

Inputs from previous projects, launch vehicle lead-times – if going to share a launch, will only know of opportunity after primary payload is manifested (probably?)
To obtain a target schedule requirement (and hence platform delivery time), a key factor that was taken into account was launcher schedule.
Many large spacecraft are manifested for launch many years ahead. However, at this early stage, a large contingency mass will be allocated, which is likely to prevent the offering of spare capacity to a secondary payload. It is considered likely that more accurate estimates of spare capacity will only come at the Authority To Proceed stage, or even at the issue of the preliminary Interface Control Document. Table 4-32 shows the mission cycles and milestones for a range of launch vehicles.

Launcher	**Typical mission cycle (months to launch)**		**Remarks**
	From ATP	From prelim. ICD	
Pegasus-XL	24	21	
Taurus	24	20	
Delta	24	14	24 months from initial s/c questionnaire
Rockot	18	15	EUROCKOT launch service provider.
Tsyklon (Cyclone)	18-24	12	
Soyuz	24	21	
Ariane 5	24	21	
Sea Launch	18	10	
Athena (LMLV)	24	20	
Atlas	12	11	
Proton	30	21	
H-vehicle	36		

NB ATP=Authority To Proceed, ICD=Interface Control Document

Table 4-32 Typical launch vehicle mission cycles

In fact, due to the risk of mass additions due to late design changes in the main passenger spacecraft, spare mass may be offered even later in the launcher mission cycle. Part of the basis for a small platform being competitive is to have a shorter schedule time, thus allowing a greater chance of cheaper shared launches. Therefore, on the basis of the table above, ideally the spacecraft should aim to have a lead-time of less than 18 months, and preferably a year, to be able to co-manifest on any of the launchers mentioned above. This is a rather ambitious target, however, and anything under a two-year delivery time would still make the platform competitive over most other commercial buses, as indicated in Table 4-33.

Spacecraft	Manufacturer	Delivery time/ months	Remarks
ACRIMSAT	Orbital	24	Based on MiniStar bus
Globalstar	Space Systems Loral	18	Multi-S/C, production line approach
Grace	Dornier (now EADS)	36-48	Based on FlexBus platform
MightySat 2.1	Spectrum Astro	24	
ODIN	Swedish Space Corp.	24-36	
RHESSI	Spectrum Astro	36	Based on SA-200S bus
WIRE	GSFC	36	

Table 4-33 Typical delivery times for small spacecraft

It is therefore recommended that as a target, the platform should be capable of being delivered for launch in 18 months from mission definition. This would obviously be heavily dependent on the timeliness of requirement inputs and other data, finance, and support being supplied by the customer. Such a delivery time would allow a small mission to be developed within the mission cycle of most of the launch vehicles, thus increasing the options for finding a launch-sharing opportunity.

4.5.3 QUALITY AND SAFETY

There are recognised quality standards that may be applied to all areas of engineering, management, and product assurance. The International Organisation for Standardisation (ISO) is a worldwide federation of national standards bodies from 100 countries, and sets standards and guidelines for quality assurance systems. The widely-used ISO 9000 is a set of standards that aid organisations in the definition and maintenance of a quality system. These standards are not specific to any type of industry or business type; they do not define particular quality assurance requirements, but provide global guidelines for the management of quality.

Within the space industry, NASA uses a standards framework based around ANSI/ASQC standard specifications; ESA employs standards defined by the ECSS. Both of these sets of standards are rooted in the ISO 9000 system, but are more targeted towards the specific issues of space manufacture. They define the required procedures for aspects such as tracking and reviewing documentation and processes, standardising business practices, and give strict specifications for equipment, environment, and the training and certification levels of the staff. Similarly, there are definite procedures for acceptance testing and qualification, including documenting test results. Components, parts and spacecraft produced to such standards will then be supplied with accompanying documentation to provide traceability and proof of qualification level.

For most space projects, the standards must be rigorously applied, thus generating large amounts of documentation and requiring the employment of large amounts of administration effort. In small satellite programmes, the standards may sometimes be slightly relaxed, to enable shorter schedule times and the use of lower cost components. However, some level of quality systems must be enforced, as the spacecraft must still be flight worthy, and, more importantly, acceptable for launch.

Even if lower-level standards are to be adopted to minimise cost and schedule, the requirement areas shown in Table 4-34 should all be considered for the proposed small satellite programme. These requirements are adapted from the 20 elements of the ANSI/ASQC Q9001-1994 standard. Decisions can then be made regarding the applicability of each particular requirement area to smaller, cheaper spacecraft. NASA uses this approach in its Small Explorer missions.[28]

	Quality system requirement	**Remarks**
1	Management responsibility	Covers quality policy, organisation (including personnel responsibility and authority), management of resources, and management review.
2	Quality system	The quality system is the means of ensuring that the product conforms to specified requirements. A quality manual covering procedures and planning is generally prepared.
3	Contract review	Covers procedures for contract review, amendment and records. Ensures that requirements are properly documented, contractual differences are resolved, and suppliers are capable of meeting the requirements.
4	Design control	Covers procedures for the control and verification of the product design. Includes planning, organisational and technical interfaces, design inputs and outputs, review, verification and validation, and control of design changes. Also covers reliability analysis and FMECA.
5	Document and data control	Covers document and data approval and issue, and procedures for controlling changes to documents and data.
6	Purchasing	Covers evaluation of subcontractors, data required on product ordered, and verification of the purchased product.
7	Control of customer-supplied product	Procedures for the control of verification, storage and maintenance of customer-supplied product, and for reporting products that are unsuitable for use e.g. due to damage.
8	Product identification and traceability	Procedures for identifying and tracing a product from receipt, through production stages to delivery.
9	Process control	Covers the procedures, equipment, working environment, reference codes/standards, parameters to be monitored, workmanship criteria, personnel qualification, and equipment maintenance required in the production process.
10	Inspection and testing	Covers inspection and test of incoming products, products in-process, finished products, and the keeping of proper test records.
11	Control of inspection, measuring and test equipment	Covers the identification, control, calibration and maintenance of test equipment.
12	Inspection and test status	Procedures for identifying the conformance or nonconformance of a product w.r.t. tests performed.
13	Control of nonconforming product	Procedures to ensure that nonconforming products are prevented from unintended use and installation. Covers identification, documentation, evaluation, segregation and disposition.
14	Corrective and preventive action	Covers corrective or preventive actions to eliminate nonconformities, including handling customer complaints and determination of necessary corrective/preventive action.
15	Handling, storage, packaging, preservation, and delivery	Procedures for these activities.
16	Control of quality records	Including identification, collection, indexing, access, filing, storage, maintenance and disposition of records.

17	Internal quality audits	To verify that quality activities and results comply with the planned quality system.
18	Training	Procedures to identify training needs, and provide for such training.
19	Servicing	Procedures for performing and verifying servicing.
20	Statistical techniques	For establishing, controlling and verifying process capability and product characteristics.

Table 4-34 Quality system requirements, adapted from [2]

A particular area for consideration in the Design Control requirement, (element 4 in the table above), is the quality level of the parts and components used. Components may be categorised by quality level, as shown in Table 4-35.

Quality Level	Description
Commercial/Industrial grade (COTS)	Lowest cost. "Hardware store" grade components. Little or no supporting documentation or traceability. Any quality checks are on a random-sample basis (e.g. 1 per 10,000).
MIL-SPEC/MIL-STD	Higher cost. Military-grade components. Materials, processes, packaging, transportation and test samples per batch all specified by military procurement standards. Availability may be an issue, due to increasing use of commercial components in place of MIL-SPEC.
Hi-REL	Higher cost, longer lead-time. Components meeting or exceeding MIL-SPEC requirements, but with individual testing. May also include some or all of the S-Level criteria described below.
S-Level (Space qualified)	Highest cost, longest lead-time. Components satisfying some or all of the following: Successful operation in space in a similar application, for a significant time, and under comparable conditions. Successful environmental testing (e.g. thermal-vacuum, radiation), with documented results. Full traceability, component history documented from raw material through all manufacturing processes to the finished article, also transportation and storage conditions. Failure Modes and Effects Analysis performed. Reliability analysis performed (quantifies likelihood of component failure). Declared Materials List (includes properties, procedures, procurement specifications). Declared Parts List (traceability, analysis, specifications, suppliers, screening and burn-in requirements for each component/subcomponent). Declared Processes List (step-by-step instructions, inspection criteria, operator skill levels and tooling for each specific process applicable to a material or part).

Table 4-35 Component quality levels[23]

It can be seen from the table above that the main difference between normal COTS components and S-level parts is the level of traceability and testing. In some cases, there may be little difference in the actual manufacturing method or technology, but there is a much greater degree of quality control. Of course, S-level parts have also been extensively demonstrated in space. However, increasing use of COTS components on spacecraft is enabling greater confidence levels in their tolerance of space conditions. Ideally, decisions on the use of COTS components must be made on a case-by-case basis, by subsystem experts.

Similarly, in terms of workmanship levels required, commercial standards may be considered acceptable *where the subsystem experts concerned have carefully considered the suitability of the standards for space applications.* Typical areas for this type of consideration are design and manufacture of PCBs, soldering, harnessing, conformal coating, and staking processes. (These areas are covered by NASA Handbook guidelines – which are followed in "conventional" space missions.)

It is assumed that in the detailed design phase of the proposed programme, subsystem engineers with specialist knowledge of these areas would be called upon to judge the implications of using (lower-cost) commercial parts and workmanship standards (and, possibly, personnel). There would be an inherent trade-off here; using commercial parts and practices may save time and money, but the resulting products may then require more extensive testing to prove their suitability for space use – potentially negating the earlier savings.

4.5.4 MANAGEMENT AND CONTROL STRUCTURE

Small spacecraft projects typically have small teams and a simple management structure. Even if a small spacecraft was to be produced by a larger manufacturer, this approach is still probably the most appropriate. The main roles required within a project are: (NB one person may have more than one role on a very small project)

- Project/mission manager
- Systems engineer
- AIT manager
- Procurement manager
- Payload manager
- Subsystem/payload specialists – comms, OBDH/computer hardware & software, AOCS, structures & mechanisms, thermal, power, payload instruments
- Operations manager
- Quality/safety assurance manager

The team would ideally remain the same throughout the project; this is more likely for small satellite projects, as the schedules are likely to be less than two years. The design team can then be involved in the assembly and test stages of their particular specialist area. This reduces training and troubleshooting time during AIT. Furthermore, if personnel are retained over the course of several projects, then the lessons learned on one project can easily be applied to successive projects.

The small team approach also implies a greater level of personal responsibility and empowerment for project personnel. Requiring the project manager to sign off every detail or purchasing order would cause costly and unnecessary delays. However, individual responsibilities and authorities must be well defined and understood, and a very good level of communication must be maintained. This is helped by the use of the small team. A brief daily meeting may be a good idea, to ensure all project staff are aware of any new developments.

4.5.5 FACILITIES AND LOGISTICS

Production of a spacecraft requires a number of specialist facilities. The main requirements are as follows:

- Secure areas for storage of components (probably with the capability for some form of asset-tracking or logging)
- Cleanrooms with sufficient area for integration of separate spacecraft parts by a number of different teams
- Different cleanroom cleanliness grades for assembly of different equipment (e.g. class 100-1000 for optics)
- Mechanical test facilities – shaker tables, mass properties balances, acoustic chambers
- Thermal vacuum test chambers
- RF test ranges

Some of these facilities will be required for constant use, and are therefore necessary on-site. Others, such as test facilities, may be provided by a third party for a (sizeable) fee.

4.5.6 MANUFACTURABILITY

To be competitive, a commercial satellite platform must not only perform well, it must also be suited to relatively easy (and low-cost) manufacture. Potential problems that impair manufacturability include:

- Overly high quality level components
- Complex machining
- Dangerous processes (e.g. use of toxic substances
- Large numbers of processes
- Complex parts

Conversely, the following approaches improve manufacturability, and can bring manufacturing costs down:

- Use of repeated/standard parts
- Good management of supply chain
- Use of lessons learned and promotion of learning curve
- Use of appropriate quality levels and testing

4.5.7 ASSEMBLY, INTEGRATION & TEST

The activities required in a typical spacecraft AIT phase are as follows:

Unit-level acceptance testing
Incoming equipment is subjected to functional and physical testing prior to acceptance as a flight unit. The functional tests verify correct operation of the equipment. These are standalone tests, which will usually involve connection to a power supply and test software on a PC. The physical tests verify that the following are consistent with the design specifications:

- Mass
- Dimensions
- Power draw, including start-up inrush currents
- Mechanical attachment footprint
- Correct structural manufacture, e.g. no materials or processes incompatible with space flight

Platform integration and functional testing
After equipment has been accepted, and all the necessary items are available, platform integration can commence. This is usually conducted in stages, with the platform being divided into "integration levels". Testing is conducted at intervals throughout the integration process, as successive items are added. The process culminates with a full spacecraft functional test, once the integration is complete.

Mechanical testing
These tests verify that the spacecraft structure is capable of withstanding the launch loads. Sine and random vibration tests are performed on a mechanical shaker, and acoustic tests within an acoustic chamber. Acoustic tests are not always necessary, especially for smaller spacecraft, as their smaller surfaces make them less susceptible to acoustic vibrations. Shock testing, simulating the transient loads introduced during pyrotechnic separation events, is performed by striking the spacecraft with a calibrated hammer blow.

For traditional projects, a dedicated test prototype model is used for testing at qualification levels, and the flight model is tested only at (lower) acceptance levels. Many small spacecraft projects use a protoflight approach instead, where only one full spacecraft model is produced and tested. To avoid the requirement to test the flight spacecraft at the high qualification levels, a simpler structural model is produced.

Thermal balance testing
A thermal model of the spacecraft (usually a mass dummy with heaters added to represent power dissipation within equipment) is used with a solar simulator. This validates the spacecraft thermal model. The thermal environments are then replicated for the thermal-vacuum testing.

Thermal vacuum testing
This simulates the space environment as closely as possible. The complete spacecraft is operated within the chamber, to verify correct function in its orbital configuration.

RF testing
The spacecraft antennas are tested in their flight configuration, in a specially constructed anechoic chamber. This allows measurement of antenna beam patterns.

EMC testing
Electromagnetic compatibility checks may sometimes be required if very sensitive instrumentation is being flown.

4.6 CHAPTER SUMMARY

This chapter has used an examination of previous spacecraft to characterise the expected requirements for the following mission types:

- Astronomy
- Space physics
- Earth observation
- Communications
- Technology

These requirements are summarised in Table 4-36.

Requirement area	**Required performance/ characteristics**	**Relative importance**	**Remarks**
	Astronomy missions		
Payload accommodation	Large volumes required. Clear fields-of-view. Precise alignment.	High	
Data rate	Up to several Mbps.	Med-high	
Orbit	HEO desirable. LEO acceptable (usually).	Medium	
Propulsion	Very mission-dependent		Needed if HEO used.
Attitude	Inertial, avoid sun-pointing.	High	
Pointing knowledge	Up to arcsecond accuracy	High	
Pointing accuracy	Up to tens of arcseconds accuracy	High	
Manoeuvring	Slew rates up to ~10°/minute	High	
Power	100-200W total bus power	Low	
Lifetime	1-2 years.	Medium	Often limited by supply of cryogenic coolant.
Other	Detector cooling often required. Cleanliness for optics.		
	Space physics		
Payload accommodation	Multiple, smaller instruments. May require mounting on long booms.	Medium	Deployable booms may require quite large volumes.
Data rate	Few Kbps typically	Med-low	
Orbit	High inclination to view auroral zones. May use HEOs to fly through different regions of magnetosphere. Accurate position knowledge often required.	High	

Propulsion	May be required for orbit insertion.		
Attitude	Usually spin. Spin axis usually inertially-fixed.	Med-high	
Pointing knowledge	Up to arcsecond accuracy.	Med-high	
Pointing accuracy	Few degrees.	Low	
Manoeuvring	Not often required.	Low	If required, spin will mean higher torques required to manoeuvre.
Power	Typically in region of 100-200W	Low	
Lifetime	1-2 years	Low	
Other	Requires high electromagnetic cleanliness onboard. May often fly through regions of high particulate radiation, electronics may require shielding.	High High	
Communications missions			
Payload accommodation	Antennas may be large.	Med-high	Antennas may require deployment.
Data rate	Low (Kbps)	Low	Little or no payload data, just housekeeping.
Orbit	LEO, probably high inclination but could be tailored to particular user's coverage requirements. Possible GEO.	Medium (High if GEO)	.
Propulsion	May be required		
Attitude	Nadir pointing	High	
Pointing knowledge	Few tenths of a degree	Medium	
Pointing accuracy	Up to a few tenths of a degree	Medium	Depends on antenna beamwidth.
Manoeuvring	Maintain nadir pointing	Medium	
Power	May be up to 500-600W	High	
Lifetime	Longer lifetime an advantage – 5-10 years	Med-high	
Earth observation missions			
Payload accommodation	Requires mounting or apertures on nadir face. Instruments may be quite large.	High	
Data rate	Up to tens of Mbps.	High	
Orbit	Often sunsynchronous. May require repeat ground track. Lower orbits for higher image resolution.	High	
Propulsion	Likely to be required for orbit maintenance.	Med-high	Also for accurate orbit insertion.
Attitude	Nadir pointing	High	
Pointing knowledge	Up to arcsecond accuracy.	High	
Pointing accuracy	Up to arcminute accuracy.	High	
Manoeuvring	Maintain nadir pointing	Med-high	
Power	May be 500W+	Med-high	
Lifetime	2-5 years	Med-high	Longer lifetimes likely to be an advantage.

Table 4-36 Summary of mission-specific requirements

Technology mission requirements were also addressed, but it was decided that these spacecraft could be considered as "special cases" of the above mission types, depending on the technology being demonstrated.

The general requirements for the platform as a whole were then examined. Requirement input parameters for the onboard subsystems were identified, for use in the following chapter. Analysis of mechanical design requirements covered launch vehicle drivers, and led to the suggestion for platform diameter steps, as follows:

- 1100mm – for launch on Pegasus-XL
- 1300mm – for launch on Taurus
- 1500mm – for launch on ASAP-5
- 1900mm(+) – for launch on Athena and larger-fairing vehicles

Approximate platform mass boundaries were also identified, as follows:

- 100-150kg for a Pegasus/ASAP-class launch, with 100-200kg payload
- 200-300kg for a "dual-Taurus" launch, with 100-200kg payload
- Around 400kg for a dedicated Taurus launch, with up to a 500kg payload

Programmatic requirements were also addressed, including cost and schedule. Based on previous missions, and civil budgets, some approximate cost ranges for the platform were proposed:

- "Basic" platform, lowest performance level: $5-10m
- Higher capability, larger payload mass, higher power: $10-20m
- Advanced platform, with "mission tailoring": $20-25m

These costs are proposed targets. As it is envisaged that the platform will use generally COTS equipment, its hardware costs will be similar to an equivalent "bespoke" spacecraft, as similar equipment will be used. The cost savings mainly arise from the different programmatic approach, which uses more efficiency to design, assemble, and test the spacecraft. Therefore, it is less meaningful to use absolute platform costs.

A delivery schedule target of 18 months was proposed. This was based on launcher mission cycle times, and the desired ability to co-manifest on a launch at a late stage. It is also highly competitive compared to delivery times for other commercial platforms.

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5 SYSTEM DESIGN

5.1 AIMS AND APPROACH

This section covers the conceptual design of the spacecraft platform, and aims to provide a systems-level bus architecture as an output, with mass and power budgets, and performance characteristics for the different configuration variants.

The approach used is first to decide on a suitable configuration. This "top-down" method is considered more suitable for this study, as the design is being driven more by the requirements for the overall platform and its ability to be reconfigured, than by the specific requirements of a particular payload or subsystem. In this type of approach, it is more important to fit the subsystem equipment into a configuration that is best suited to supporting a range of payloads, rather than the more usual method of designing the platform around particular subsystem equipment. This approach obviously requires later iteration in order to balance the needs of payload, structure, and subsystems.

A range of basic configuration types is identified for analysis. Parameters are chosen that provide useful metrics for evaluating the applicability of these different configuration concepts. This allows trade-off studies to be performed and an appropriate concept to be chosen for more detailed development.

At this stage, subsystem level design can proceed, with further configuration iterations being performed as needed. The subsystem designs give phased performance options, which supply the modularity and reconfigurability required at a functional level. These are then integrated into the different platform performance/capability variants.

The methodology flow for the System Design chapter is shown in Figure 5-1 on the following page.

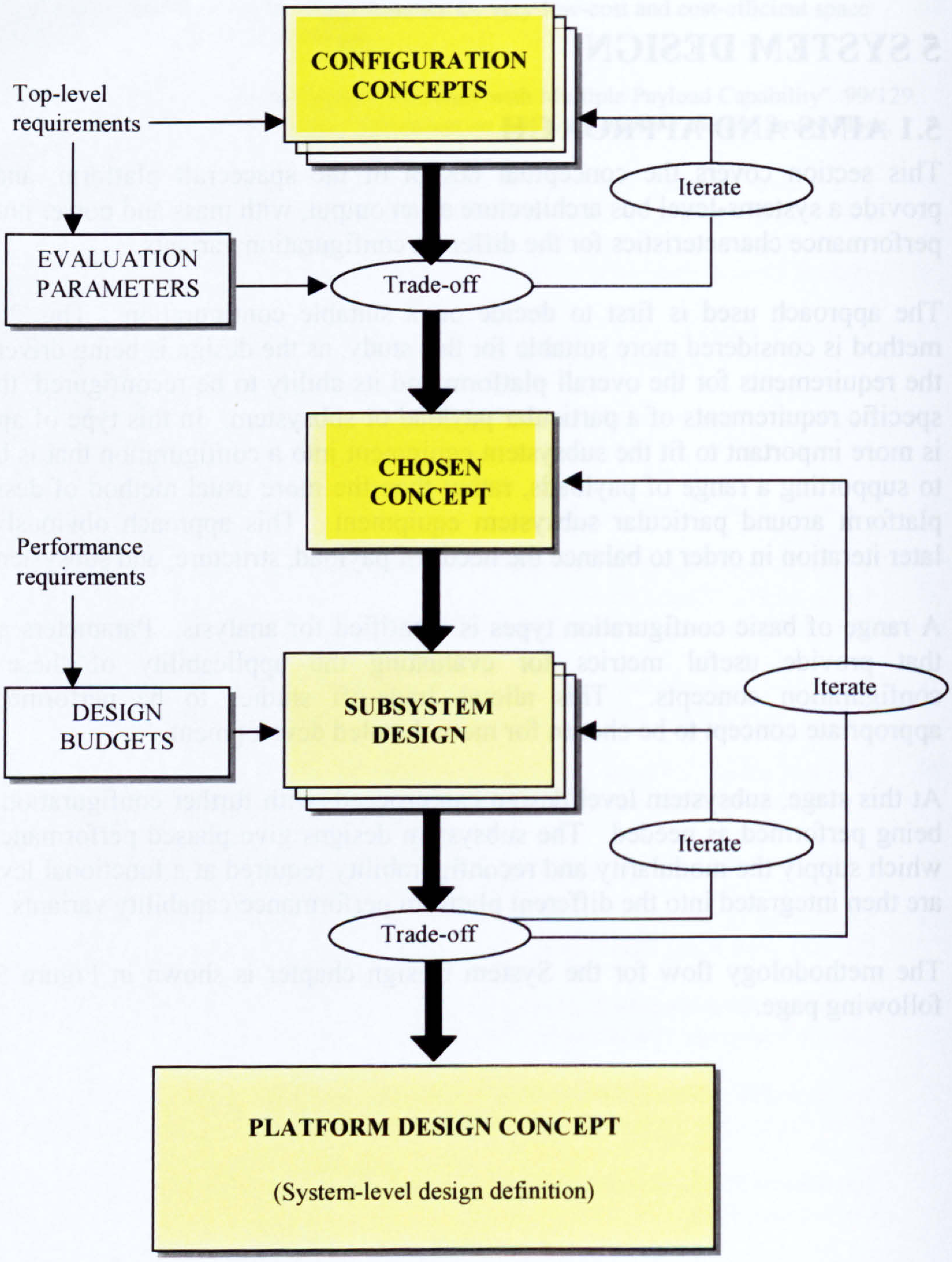


Figure 5-1 Design methodology flow

5.2 CONFIGURATION CONCEPTS

5.2.1 PARAMETERS FOR CONSIDERATION IN CONCEPT EVALUATION

The requirements identified in Chapter 4 lead to the definition of key parameters to use when evaluating the proposed concepts. For each concept, these parameters are assessed and the concept assigned a rating for trade-off purposes. Multipliers are used to assign greater weightings to those parameters considered most critical.

The parameters are adapted from the requirements governing the configuration of the spacecraft as a whole. For example, they are not the specific power level requirements, but requirements such as the need for the spacecraft to be adaptable to raising a range of different power levels. The subsystem-specific requirements are considered later, in the subsystem design section.

The top-level requirements that impact on the overall configuration may be summarised as:

- The platform must be adaptable to a range of payload types, sizes, and configurations
- The platform must be adaptable to a range of launchers
- Schedule reduction and flexibility should be enabled by the use of reconfigurable modules and common parts
- The different configurations should allow for a range of performance/ cost/ capability levels

This allows the definition of the following parameters, which describe how well a particular configuration can meet the driving requirements:

5.2.1.1 Mass

This may actually be divided into two separate parameters, platform mass efficiency and payload mass capability. The former is a measure of how much superfluous structural mass is employed in the platform configuration (compared with, say, a platform of comparable size designed specifically for optimum mass-efficiency). The mass capability describes the ability of the platform to support heavy payloads.

The mass efficiency is less important in this instance, as it may be considered acceptable to sacrifice some mass efficiency for the sake of allowing the design to employ a greater degree of modularity. This will be reflected in the parameter multipliers defined later in this section.

5.2.1.2 Volume

As with mass, this parameter may be divided into measures of the overall volume efficiency of the platform, and the volume available to the payload. Again, volume efficiency is less important than the volume available to the payload, but it should obviously be recognised that a volume-inefficient design is likely to leave less volume (restricted as it always is by the launcher envelope) free for the payload. The payload volume parameter must also give consideration to the variety of different payload shapes and sizes, and their numbers per mission.

5.2.1.3 Aperture/ field-of-view provision

This parameter gives a measure of the ease with which the platform can provide access to the exterior for payload instruments. For example, an enclosed box architecture, where the payload is accommodated inside the body of the platform, would score poorly in this category, as apertures must be cut into the structure of the spacecraft (thus weakening the structure and reducing exterior surface area for externally-mounted equipment).

5.2.1.4 Solar array surface area available

Although in the previous chapter it was determined that the platform is likely to require deployed solar arrays for most missions, for missions at the lower end of the cost/capability spectrum it may be useful to offer a variant with body-mounted arrays only. Body-mounted arrays are also more likely to be used if a spin-stabilised variant is produced. This parameter gives a measure of the suitability of the design to this type of configuration. However, this is a low-priority requirement, which is reflected in the parameter multiplier.

5.2.1.5 Cost

This parameter takes account of the estimated relative costs of materials and manufacturing for the different structural configurations. Therefore, designs that use larger amounts of structure, are more complex, or require more expensive materials or manufacturing processes, rate a lower score than simpler, cheaper structures.

Although cost is a significant factor, the platform is not intended to be the very lowest cost option available; it is intended to be low cost *for a given capability*. It should be noted, however, that this parameter does not take account of the cost savings that may arise from a more expensive design that is highly modular. These savings are accounted for in the modularity parameter described later.

5.2.1.6 Size adaptability

One of the main requirements identified is the ability to adapt to a range of different launch opportunities, implying a need to offer a number of different size configurations. Different size configurations will also be required for supporting different size, shapes, and masses of payload. This parameter gives a measure of how suitable each design is to be configured into different sizes. A higher score is given if the design can be re-sized with minimal changes to the parts required. Size adaptability is a high priority; this is reflected in the multiplier assigned.

5.2.1.7 Suitability for modularity/ reconfigurability

This parameter describes another high priority requirement, and gives an indication of how well the design is suited to division into separable modules that may then be reconfigured.

5.2.1.8 Degree to which platform and payload may be decoupled

As seen in Chapter 2, several spacecraft that have employed a multipurpose platform largely decoupled the payload from the supporting "service" platform. The advantages of this decoupling were described in the previous chapter, and it is considered to be a fairly high priority, as it helps to both enable modularity and allow parallel integration and testing.

5.2.1.9 Suitability for accommodating COTS equipment

Many spacecraft equipment items, particularly electronics boxes, take the form of square or rectangular prisms. This gives a good packing efficiency as long as the accommodating volume is of a similar geometry. However, it becomes more difficult to mount such items where there are curved surfaces or tight corners, resulting perhaps in the additional expense of modifications or custom building.

Although custom building is permissible within the philosophy of the multipurpose platform proposed, it will generally be an advantage if the configuration offers suitable accommodation for this shape of equipment box (both for platform subsystems and for payloads). This parameter therefore gives an indication of the ease with which "standard" boxes may be accommodated within the design.

5.2.1.10 Other considerations

Configuration also impacts on the interface with the launch vehicle. It is likely to be easier to produce an adapter to attach a smaller diameter spacecraft to a larger launcher interface than vice versa. Therefore, designs with a more central, smaller diameter load-bearing structure will probably be easier to attach to a range of sizes of launch vehicle.

Furthermore, the shape of the load-bearing structure also determines the way that the loads are transferred to the launch adapter; for example, a hexagonal spacecraft would require an adapter to receive point loads at the six vertices, and transfer them to the cylindrical launch vehicle while avoiding any local overloading[11].

Other areas to consider are the complexity and numbers of parts involved in assembling the different structures examined. Cost of materials is generally relatively insignificant compared with labour costs[9], so designs that reduce assembly labour will provide a cost advantage (and also reduce schedule time).

5.2.1.11 Parameter multipliers

In the design concept trade-off analysis, each of the parameters described above are rated with a score on a scale of 1 to 5, with 5 indicating best fulfilment of the requirements. In order to account for differences in the relative importance of the parameters, each is then assigned a multiplier, or weighting factor, of 1, 2, or 3. The parameter multipliers are shown in Table 5-1.

Ref.	Parameter	Multiplier
1	Platform mass efficiency	1
2	Payload mass capability	3
3	Platform volume efficiency	2
4	Payload volume	3
5	Aperture/ field-of-view provision	3
6	Solar array surface area available	1
7	Cost	2
8	Size adaptability	3
9	Suitability for modularity/ reconfigurability	3
10	Degree to which platform and payload may be decoupled	2
11	Suitability for accommodating COTS equipment	2

Table 5-1 Design evaluation parameters and multipliers

These parameters provide a reference framework, which is now used to evaluate a range of possible design concepts.

5.2.2 CONFIGURATION CONCEPTS AND ANALYSIS

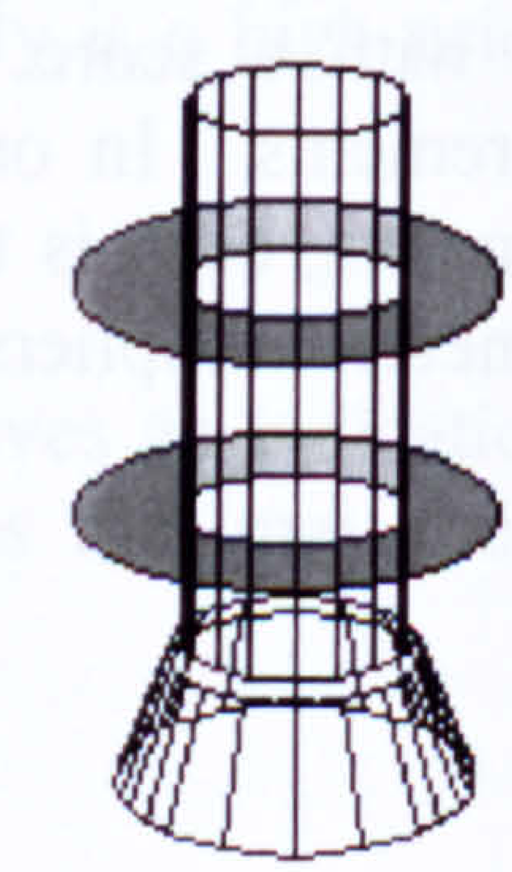
The initial configuration concepts are adapted from the main types of spacecraft configuration identified by study of past and current missions. The configuration types identified are:

- Thrust-tube/decks
- Skin-stringer/longeron with bulkhead decks
- Box modules
- Space-frame/decks

These configuration types are now analysed using the derived evaluation parameters. Based on the analysis results, the main "parent" configurations are then adapted to give further concepts, which are tailored more towards the requirements defined.

5.2.2.1 Thrust-tube/deck configuration

The thrust-tube/deck configuration is analysed in Table 5-2.

Configuration	**Thrust-tube and outer decks**	
Examples	Minisat,	
Description	Central cylindrical thrust-tube, with equipment mounting decks attached to outside. May be closed, with outer shear panels with solar arrays, or open. Bottom of tube interfaces to launch adapter cone.	
Components/ materials	Thrust tube generally filament-wound C fibre or honeycomb sandwich. Isogrid also possible. Decks often Al honeycomb. Outer panels (if present) honeycomb.	

General Applications & Characteristics	
Pros	Propellant tanks or bulky payloads (e.g. optics) can be accommodated within thrust tube. Open architecture gives wide range of sensor FOVs. Cylindrical thrust tube provides simple interface to launch adapter, with uniform loading.
Cons	Hard to mount equipment directly to thrust tube (curved surface). If open, electronics boxes may require additional shielding. Decks and fixings must be very stiff – only one edge fixed. Shape not very compatible with typical rectangular equipment boxes. Integration of equipment within the tube may be difficult.
Particularly suited to	Astronomy or Earth Observation missions. Spin-stabilisation. Larger spacecraft.
Not suitable for	Microsatellites (inner thrust-tube becomes superfluous). Volume-limited missions where a highly-compact design is needed.
Equipment mounting	On decks, within thrust-tube. Onto thrust tube if adaptable to curved surface. Possibly on outer shear panels (if present), for low-mass equipment.

Evaluation Parameters

	Multiplier (1-3)		Rating*	Total
Mass efficiency	1	Med-high – thrust tube is generally quite mass-efficient as the main structural element. Little superfluous structure.	4	4
Payload mass	3	Large, heavy payloads can be supported by the thrust tube, especially if tube is short and payloads mounted on top. Less on the decks.	4	12
Volume efficiency	2	Med-high if equipment mounted within thrust-tube. Good use of launcher volume due to similar shape.	3	6
Payload volume	3	High, if inside thrust tube used. Can also mount on top.	4	12
Aperture provision	3	Very good, if use open architecture	5	15
Solar array surface	1	Requires additional structural panels around exterior.	1	1
Cost	2	Medium – single main structure, but filament winding quite costly (but reduces for successive windings on same mandrel).	3	6
Size adaptability	3	Can make different diameter, thickness and length of thrust tube – requires different mandrels and attachments.	3	9
Suitability for modularity/ reconfigurability	3	Poor to medium – can vary length/diameter of thrust tube, position of decks, use different I/F cone for different launchers but needs different parts. Not very modular structurally. Could make decks standardised?	2	6
Platform/ payload decoupling	2	Medium to good – depends on whether payload is inside thrust tube, on decks, or on top.	4	8
COTS equipment accommodation	2	Poor, due to curved surfaces of thrust tube.	2	4
Total evaluation score: total of ratings [1-5] x multipliers [1, 2 or 3] (i.e. out of possible 125)				83

* Parameters rated on a 1-5 scale, where a rating of 5 indicates greatest suitability

Table 5-2 Thrust tube/ outer deck configuration description and evaluation

Discussion

The table above identifies the main strengths and weaknesses of this configuration. The main weaknesses are poor size adaptability, poor suitability for modularity and reconfigurability, and the difficulty of accommodating COTS-type equipment. Each of these weaknesses is now discussed, to attempt to identify ways of adapting the configuration to improve these areas.

The classic thrust-tube configuration has been much used for large spacecraft, but does not perhaps scale down particularly well. If the tube becomes too small, it becomes difficult to use the volume inside it. Alternatively, to avoid this, the thrust-tube must then be used as the external structure, with all the equipment mounted inside it. This then changes the entire design philosophy. This lack of scalability is one of the principal drawbacks of the design; as indicated in the table above, it reduces its ability to be reconfigured. However, some degree of reconfigurability is possible by:

Using different numbers and sizes of decks
These could be varied to accommodate different sizes and masses of equipment. The central tube would have to be strong enough to support any of the possible configurations, if it was to be a standard, "off the shelf" component. However, it may be possible to vary the number of windings, to give a thicker-walled tube for more robust configurations. Accommodation of heavy and bulky equipment on the outer decks may present problems for several reasons:

- As the shelves are only constrained at one end, they must be made very stiff (implying quite a large structural mass)
- Attaching directly to the filament-wound tube requires special inserts and fasteners
- The equipment is mounted at some distance from the centre of mass, adversely affecting the mass properties of the spacecraft if the equipment is heavy.

Using different diameters and lengths of central tube
As a key element of the price of a filament-wound structure comes from producing the mandrel, offering many different diameters of tube would not be very cost-effective. However, different lengths of tube can be made on the same mandrel, so it would be possible to perhaps offer only two different diameters (corresponding to a small and a medium-sized launcher), but then a very flexible range of lengths. This would obviously affect the stiffness of the tube.

It is also more difficult to see how this type of configuration can be easily divided into discrete modules, which can be assembled and tested separately. However, each deck may be separately assembled, and then mated with the central tube towards the end of the integration process. Equipment to be situated within the tube could be mounted on a secondary structure that fits into the tube and then is fastened in place in a pre-assembled configuration.

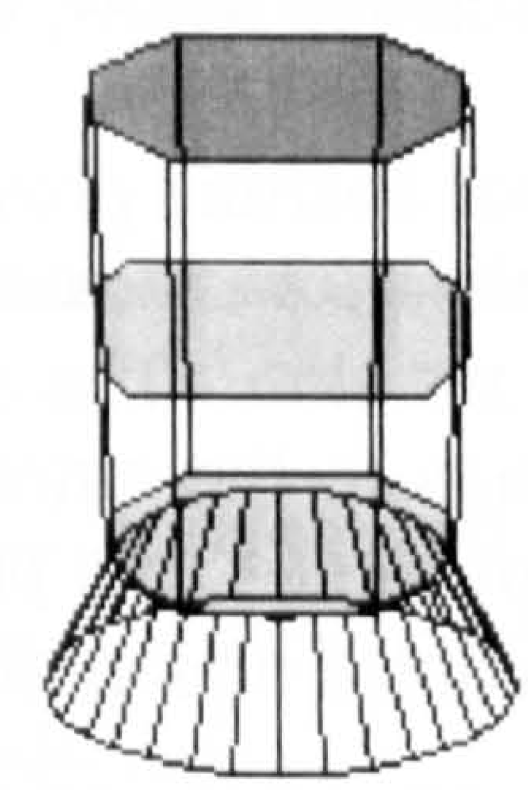
Despite the modularity scheme described above, it would be difficult to employ this method to allow any "pre-qualification" of the structure, as it would greatly depend on the exact positioning of each piece of equipment on the decks. It may be possible, however, to define a qualified "envelope" of allowable equipment positions and masses, or total deck loading, for different tube/deck configurations, which would still allow for some streamlining of the design and qualification process.

Another drawback identified is the difficulty of accommodating COTS equipment. The curved geometry will result in "lost" volume where the (often rectangular) equipment boxes are mounted on or near the thrust tube. This may be overcome, in the case of the spacecraft systems, by using custom-built equipment, although this will add to the cost unless the cost to develop dedicated hardware is outweighed by the improved packaging solution offered. This is an area where some of the new technologies identified in Chapter 2 may provide solutions. Particularly useful are the multifunctional structures and inflatably-deployed solar arrays, as these may be adapted to more unusual geometries.

One final consideration for this type of design is that the equipment is largely mounted externally. This implies a lower inherent shielding of potentially sensitive electronics (as they are not enclosed within a secondary layer of structure). Additional shielding may therefore be required; this will be mission- and equipment-dependent.

5.2.2.2 Skin-stringer/longeron/bulkhead deck configuration

The skin-stringer/longeron/bulkhead deck configuration concept is described and analysed in Table 5-3.

Configuration	**Skin-stringer/longeron with bulkhead decks**	
Examples	ERS-1, STRV, Clementine	
Description	Outer skin-stringers structure, may be square/hexagon/octagon etc prism. Bulkhead decks prevent lozenging & provide mounting points.	
Components	Stringers generally Al. Skins & decks honeycomb, sheet Al or isogrid.	

Applications	
Pros	Can be made very compact by choice of shape – e.g. cuboid will enclose typical electronics boxes very effectively. Many suitable sites available for mounting equipment Solar cells can be mounted on the outer skins Integration can be made easy if panels can be assembled sequentially
Cons	May be difficult to accommodate odd-shaped payloads. Holes must be made in skin to provide payload apertures- this weakens the structure.
Particularly suited to	Smaller spacecraft Missions with several small experiments Spacecraft relying solely on body-mounted solar arrays
Not suitable for	Large payloads, e.g. telescopes with long optical sections
Where can equipment be mounted?	Top and bottom of decks. Inside & outside panels.

Evaluation Parameters

	Multiplier (1-3)		Rating	Total
Mass efficiency	1	High – little superfluous structure	4	4
Payload mass	3	Heavy payloads may be supported on top, or on the decks, depending on construction.	4	12
Volume efficiency	2	High-density packaging possible, depending on shape (square prism probably best).	4	8

Payload volume	3	Limited, but payloads could be placed on top, and/or protrude from sides, depending on launch envelope.	2	6
Aperture provision	3	Low – require holes to be made in skins	2	6
Solar array surface	1	High – cells may be easily mounted on outer skin	5	5
Cost	2	Low-med, depending on materials used	4	8
Size adaptability	3	Good – can vary size and shape of prism, vary number & position of bulkhead decks	4	12
Suitability for modularity/ reconfigurability	3	Medium – size and configuration may be changed, but not easily separable into discrete modules	3	9
Platform/ payload decoupling	2	Medium – best if payload is mounted on the top	3	6
COTS equipment accommodation	2	Good, particularly if a square shape is used	4	8
Total evaluation score:				84

Table 5-3 Skin-stringer/ longeron configuration description and evaluation

Discussion

As indicated in the table, the main weaknesses of this type of configuration are lack of apertures, lower payload volume, and limited options for modularity. These aspects are discussed below, and suggested improvements offered.

Poor aperture provision

This arises because a key factor governing the strength of the structure is the presence of the skin. The structure is weakened when holes are made in this skin, and there are limits to the number and size of holes that are permissible, thus limiting the aperture provision. This problem may be avoided if payloads requiring significant apertures are mounted on top of the main spacecraft.

Poor payload volume

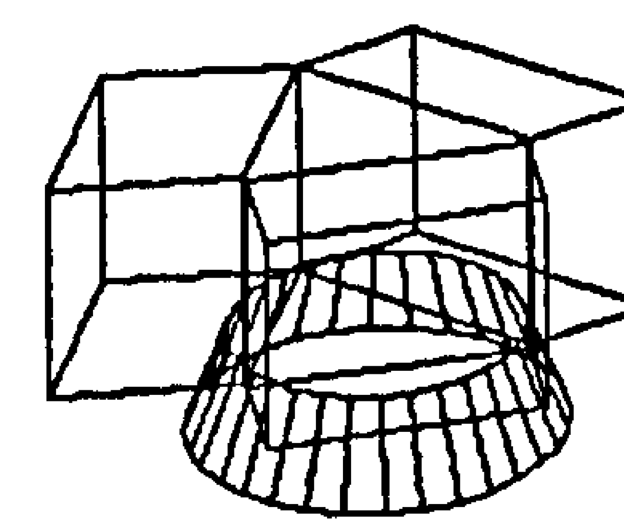
Part of the problem here is the presence of the closed sides of the spacecraft, and the bulkhead decks. These limit the accommodation options for larger payloads. Again, this problem may be avoided by mounting large payloads on the top of the main body of the spacecraft. However, this configuration is still less suited to accommodating payloads with one long dimension (such as telescopes), as these are probably better mounted longitudinally if they are to fit within usual launch envelopes.

Modularity

The options for making this configuration separable into modules are generally quite limited, due to the overall construction. Some modularity is possible by employing a system of "trays", which slot inside the main body. This is used for the SSTL UoSAT series. This does not, however, provide for much reconfigurability, although it can allow parallel integration to reduce schedule time.

5.2.2.3 Box module configuration

The box-module configuration concept is described and analysed in Table 5-4.

Configuration	**Box modules**	
Examples	MMS, AMSAT P111-C, SNAP1	
Description	Equipment mounted in boxes, which form part of the primary structure	
Components/ materials	Box modules, varying in number Interface ring Possibly a frame structure to which the modules are joined, or corners of boxes are reinforced & joined together Boxes may be machined Al or honeycomb. Frame (if present) may be Al, carbon-fibre	
Applications		
Pros	Each module can be integrated and tested separately, greatly parallelising AIT Different numbers of modules can be used, depending on what equipment must be accommodated Payloads can be mounted on top of the modules if necessary Secondary electronics box-enclosures may not be necessary – this would increase the mass efficiency	
Cons	Not very mass and volume efficient If payloads are placed within module boxes, provision of apertures may be difficult	
Particularly suited to	Multipurpose platforms	
Not suitable for	Missions with tight mass budgets Body-mounted solar arrays (unless "filler panels" are added between modules to provide additional area)	

Evaluation Parameters

	Multiplier		Rating	Total
Mass efficiency	1	Low – lot of superfluous structure	2	2
Payload mass	3	Variable – could be reasonable if structure made strong and payload mounted on the top, but load paths ill-defined	3	9
Volume efficiency	2	Medium – modules can be filled quite densely with equipment, but modules not very densely packed together	3	6
Payload volume	3	Low if only in module boxes, but higher if payload can be mounted on top of modules	3	9
Aperture provision	3	Low if payloads in module boxes, but high if top-mounted	3	9
Solar array surface	1	Low - med	2	2
Cost	2	Low-med, depending on materials	4	8
Size adaptability	3	High	5	15
Suitability for modularity/ reconfigurability	3	Very suitable – different numbers and types of modules can be used	5	15
Platform/ payload decoupling	2	Good – can use separate payload module, and/or mount payload on the top	5	10
COTS equipment accommodation	2	Good, if module shape is appropriate	5	10
Total evaluation score:				95

Table 5-4 Box-module configuration description and evaluation

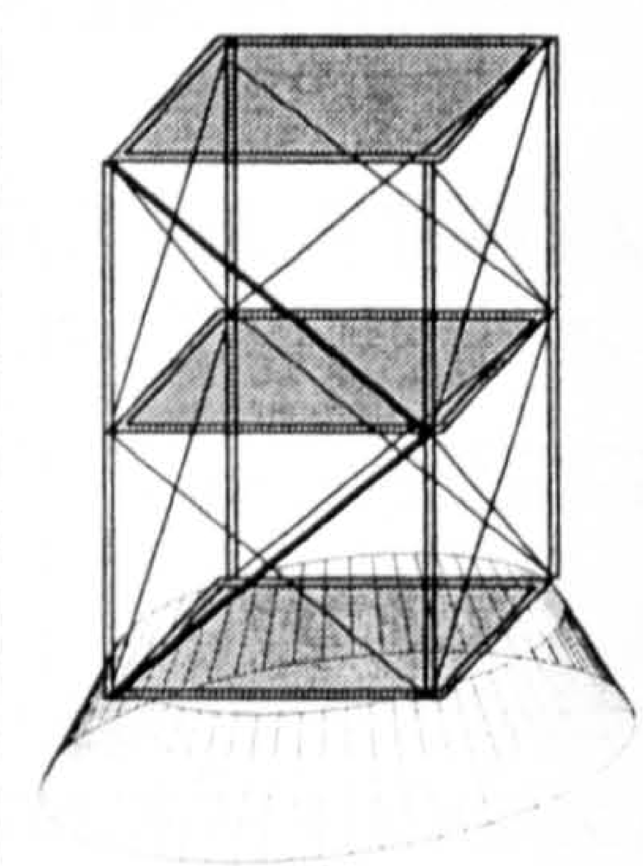
Discussion

This configuration is probably the most compatible with the modular/reconfigurable concept. Its main drawback is its mass inefficiency, but this can be minimised by reducing the amount of separate equipment containment structure within the box modules, and using the box-modules themselves as the electronics housings. It may also be improved if the central volume contained by the modules themselves can also be used.

This type of design scores very highly in the analysis, and could be considered as a viable configuration concept.

5.2.2.4 Space-frame/deck configuration

The space-frame/deck configuration concept is described and analysed in Table 5-5.

Configuration	**Space-frame and decks**			
Examples	FORTE			
Description	Horizontal decks supported by open framework truss structure. Non-primary-load-bearing panels may be added to the outside to carry solar panels, additional equipment			
Components	Space-frame made of machined/ extruded aluminium/titanium struts, or carbon-fibre composite. Structure may be prefabricated whole or in large parts, or may consist of separate frame & truss members fastened together Honeycomb/ isogrid/Al plate decks			
Applications				
Pros	Lightweight, especially if made of composites Can be prefabricated in large sections – cuts integration time & cost Good access to spacecraft interior for AIT Can accommodate range of equipment shapes & sizes due to open structure			
Cons	Frame structure causes point loads on interface ring/ launcher structures If joints are welded, QA/repeatability issues Spacecraft grounding problems if carbon-fibre frames used If structure is open, may have to provide extra radiation shielding			
Particularly suited to	Larger spacecraft Spacecraft with deployable structures e.g. arrays, booms etc			
Not suitable for				
Where can equipment be mounted?	Top & bottom of decks. Hard-points on frame structures, inside & outside			
Evaluation Parameters				
	Multiplier		Rating	Total
Mass efficiency	1	High – little superfluous structure	5	5
Payload mass	3	High	5	15
Volume efficiency	2	High – good packing efficiency possible	5	10
Payload volume	3	Med – can mount payloads so they protrude through frame if necessary, or on the top, but harder to fit very large payloads	3	9
Aperture provision	3	High	5	15
Solar array surface	1	Med – would have to add outer skins, which adds mass (and reduces aperture availability	3	3

Cost	2	High but would decrease if many were built, particularly if composites were used	2	4
Size adaptability	3	Med-high	4	12
Suitability for modularity/ reconfigurability	3	Possible to change size of frames by changing lengths of structural members Use different numbers of frame "boxes"	3	9
Platform/ payload decoupling	2	Difficult unless payload entirely mounted on top deck	2	4
COTS equipment accommodation	2	Good	5	10
Total evaluation score:				96

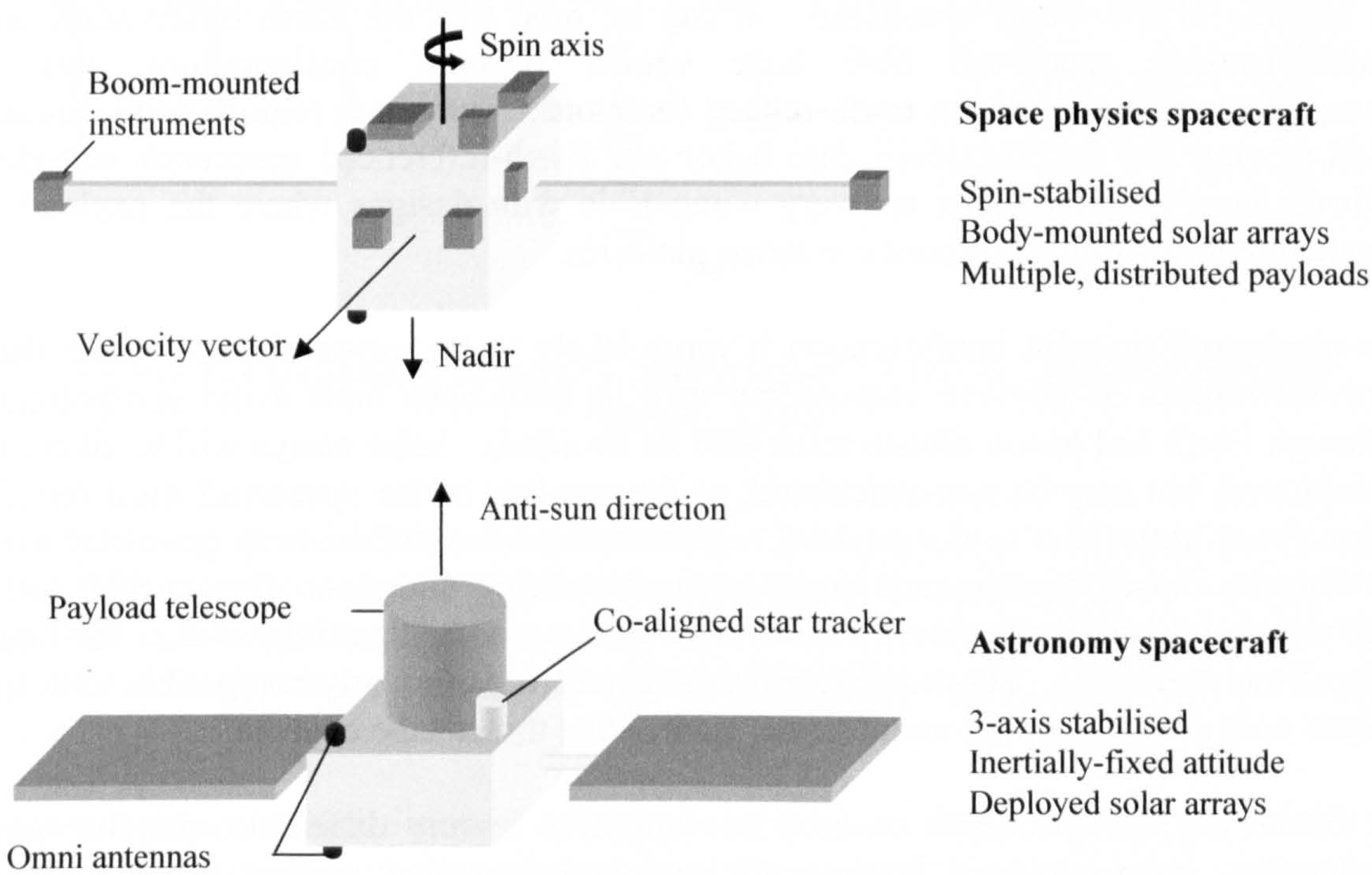
Table 5-5 Space-frame/deck configuration description and evaluation

Discussion

As with the box-module configuration, this design scores very highly in the analysis. Its main weakness is that it is a little more difficult to incorporate the desired modularity scheme. The design is perhaps modular at too low a level (i.e. at component level).

5.2.3 DRIVERS FROM ON-ORBIT CONFIGURATION

The on-orbit configuration for different types of mission application also affects the final choice of design. Consideration must be given to the fields of view for payloads and sensors, the attitude of the spacecraft, and the general positioning of subsystem equipment. To be suitable, the design chosen for the proposed platform must be compatible with all these on-orbit configurations. The generalised on-orbit layouts for the main mission application types are shown in Figure 5-2. These are derived from the requirements identified in Chapter 4.



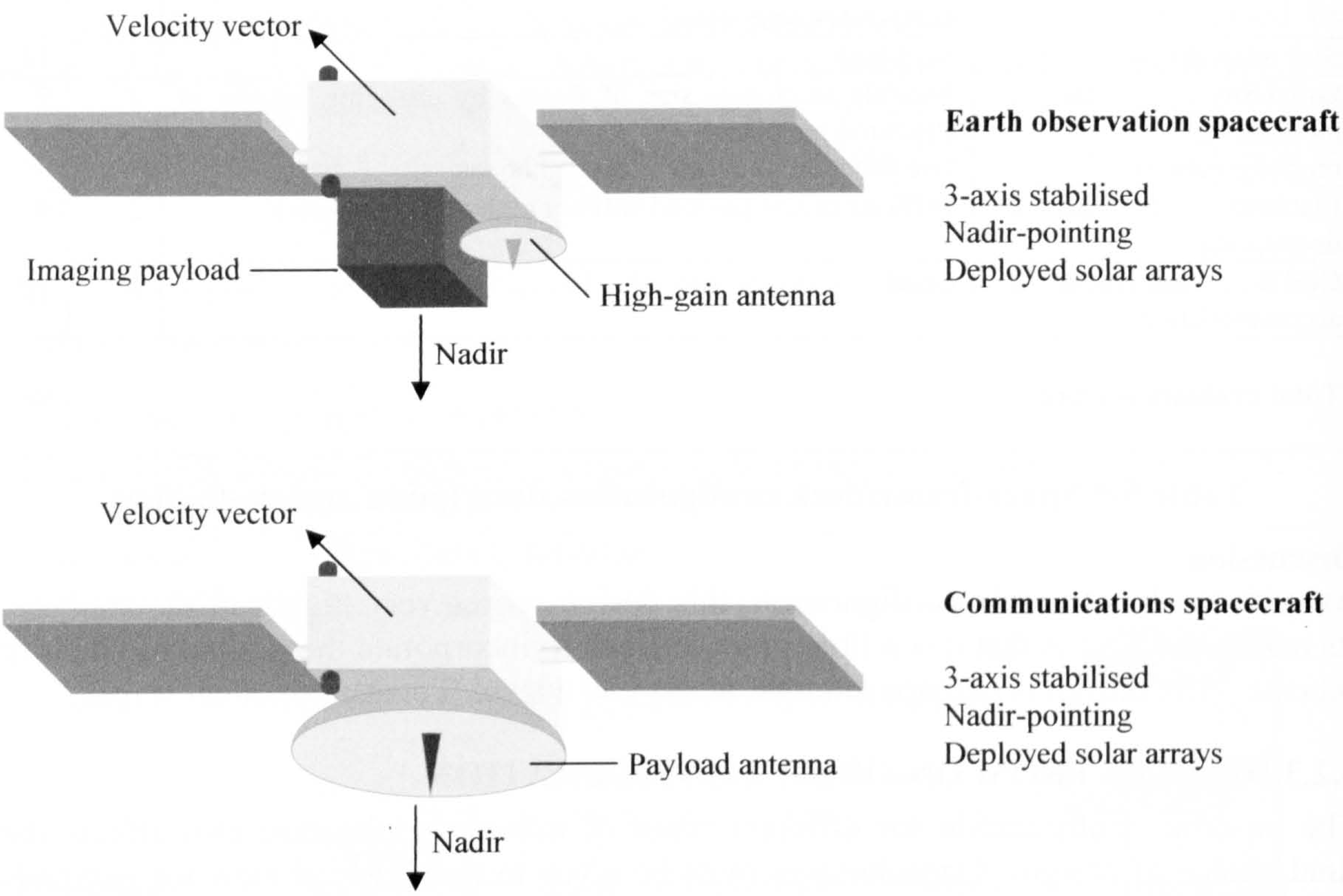


Figure 5-2 General on-orbit configuration characteristics for the main mission application types

The main spacecraft platform is represented by a generic cuboid block in the diagrams, as these are merely to indicate the general positioning and pointing of the payload, and the attitude of the whole spacecraft. It can be seen that the Earth observation and communications spacecraft both have similar on-orbit configurations; this is unsurprising, as both perform Earth-related functions. They both require nadir-pointed fields of view for their payloads, and hence use Earth-referenced spacecraft attitudes. Both of these configurations are very compatible with designs where the payload is mounted onto the top of a separate systems platform.

The astronomy on-orbit configuration is more likely to be sun-referenced rather than Earth-referenced, as payload instruments such as telescopes must avoid sun-pointing (although Earth and Moon albedo must also be avoided). Solar arrays will be likely to be deployed, but may be non-articulated, as the sun-line of the spacecraft must remain within fixed limits due to the payload requirements. A suitable array geometry may therefore be chosen by reference to the payload pointing direction. Due to their often large size, telescope payloads may need to be accommodated partially within the main body of the spacecraft. This configuration is therefore particularly compatible with the designs with an open central architecture, such as the thrust-tube concept.

The Space physics spacecraft on-orbit configuration is very different, with the spin-stabilisation, boom-mounted instruments, and body-mounted arrays. This is more suited to designs with closed architectures (allowing solar cells to be mounted on the outer surfaces. The skin-stringer architecture is therefore probably most suited to this type of mission configuration, but the box-module design could be quite easily adapted by the addition of some "closing panels" between the modules, which would carry solar cells. The chief payload accommodation difficulties with this type of configuration will probably be the stowing of the payload booms. With the box-module architecture, booms could perhaps be stowed in the voids between the modules.

5.2.4 DERIVATIVE CONCEPTS

A number of possible baseline concepts have been identified. Each has particular strengths and weaknesses, as described in the analysis. These are summarised by Figure 5-3.

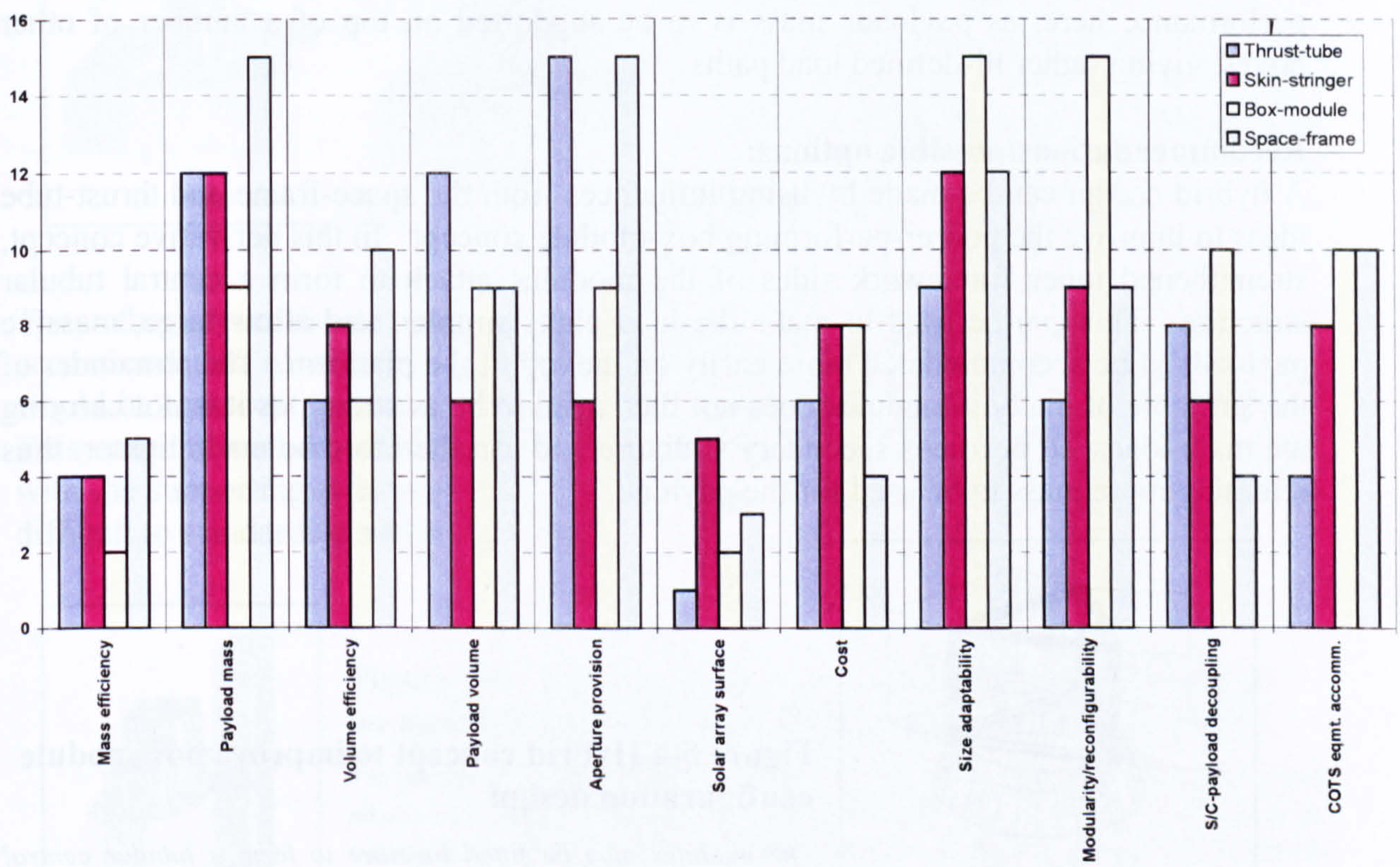


Figure 5-3 Graphical summary of evaluation scores for the four baseline configurations

In order to arrive at the best configuration solution, an attempt is now made to produce some "derivative concepts", which take positive aspects of the different concepts and use them to strengthen the weaker areas identified. These may then be used to form hybrids of the baseline configurations, which "fill in the gaps" identified in the figure above.

The strongest performers in each category can be identified, and the characteristics that give these strengths can be combined where possible. Of course, some of the characteristics may be incompatible, which gives the different possible hybrids. To simplify the task, the most important parameters (those given a multiplier of 3 in the analysis), are considered first.

5.2.4.1 Payload mass

In this category, there is no clear strongest performer; none of the designs performs particularly poorly, and good performance is obtained for different reasons. The space-frame/deck design performs well due to the inherent strength and stability of trusses and frames, and the ability to support large masses on any of its decks. However, it is slightly disadvantaged by the need to transfer point-loads onto the launcher interface. The skin-stringer design performs similarly, but provides less strength if apertures are cut into the outer skins. The thrust-tube provides good stiffness and strength if it is not subject to local non-uniform loads. It is therefore good for supporting a large mass on the top of the tube, but not well-suited to supporting large masses on the outer deck structures. These could induce buckling. The box-module design gives the poorest performance here, as payloads mass is to be supported on top of a number of other boxes, giving rather ill-defined load paths.

Recommendations/possible options:

A hybrid design can be made by using influences from the space-frame and thrust-tube ideas to improve the poorer-performing box-module concept. In this derivative concept, strengthened inner framework sides of the modules attach to form a central tubular structure. This can be used to make the load paths simpler, and allow large, massive payloads to be accommodated more easily on the top of the platform. The remainder of the structure of the box-modules does not then need to be as strong, as it is not carrying the main loads. It becomes secondary structure and can therefore be made lighter, thus releasing more mass to be used for the payload.

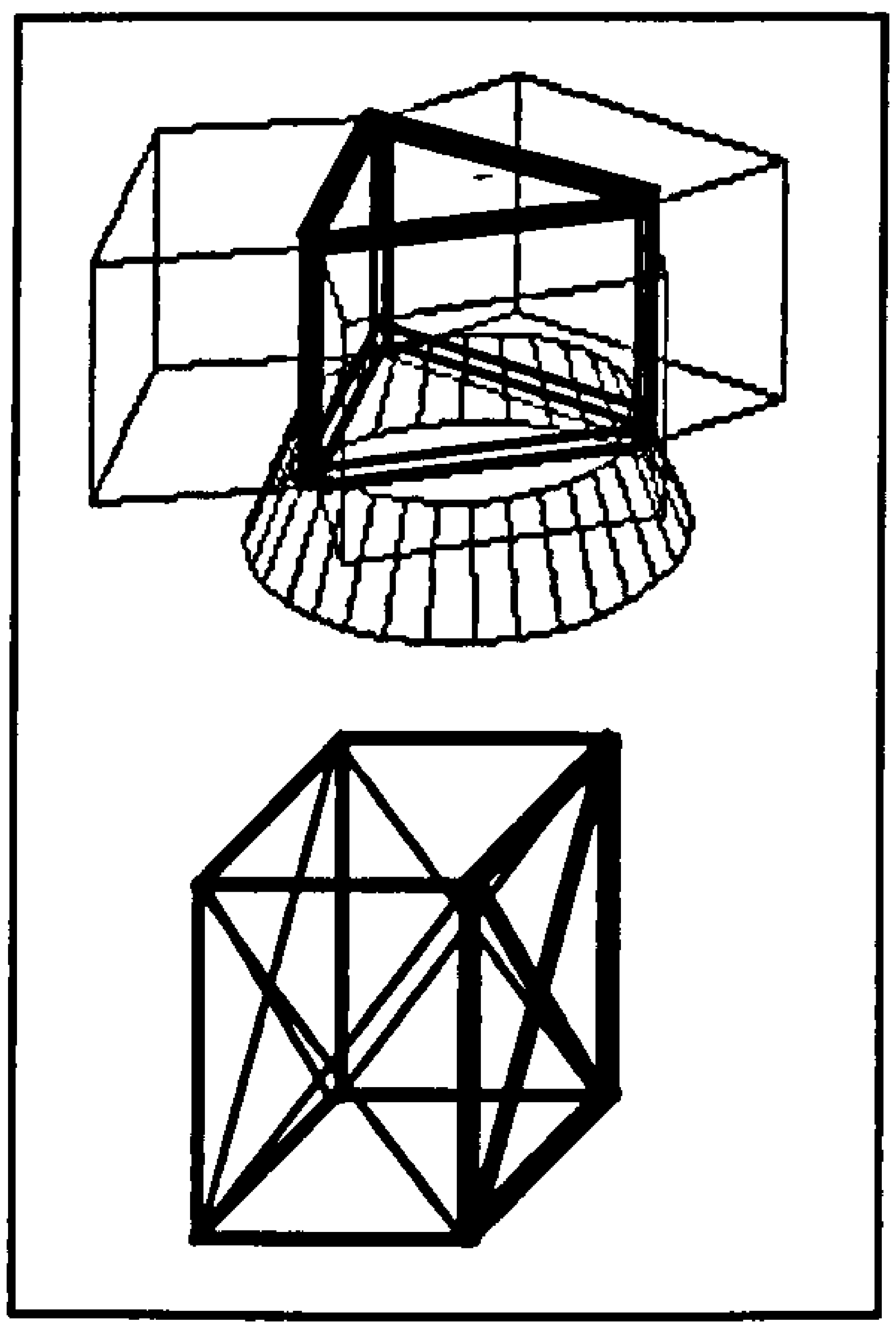


Figure 5-4 Hybrid concept to improve box-module configuration design

The modules may be fitted together to form a tubular central frame structure. This gives strength and allows the mounting of heavy payloads on the top of the spacecraft platform. The lower diagram illustrates a single module, showing how one of the sides is reinforced to form the central "tube".

5.2.4.2 Payload volume

The performance of the different configurations varies quite widely in this area, the strongest performer being the thrust-tube design. The determining factor that enables a strong performance is the use of a central tube or open space within the spacecraft structure. This is present with the thrust tube, and the box-module configurations, and permits the accommodation of long, bulky payload items such as telescopes. The other two designs score more poorly in this area, as they do not have the ability to accommodate large, lengthy payload instruments due to the internal decks. While cut-outs could be made in the decks, these would weaken the structure if large.

Recommendations/possible options:
As with the recommendation described above, adapting the box-module design to form a central tube gives an open volume for accommodation of long payloads. Using different numbers of modules could provide a larger central void in which to mount larger payloads. This is illustrated in Figure 5-5.

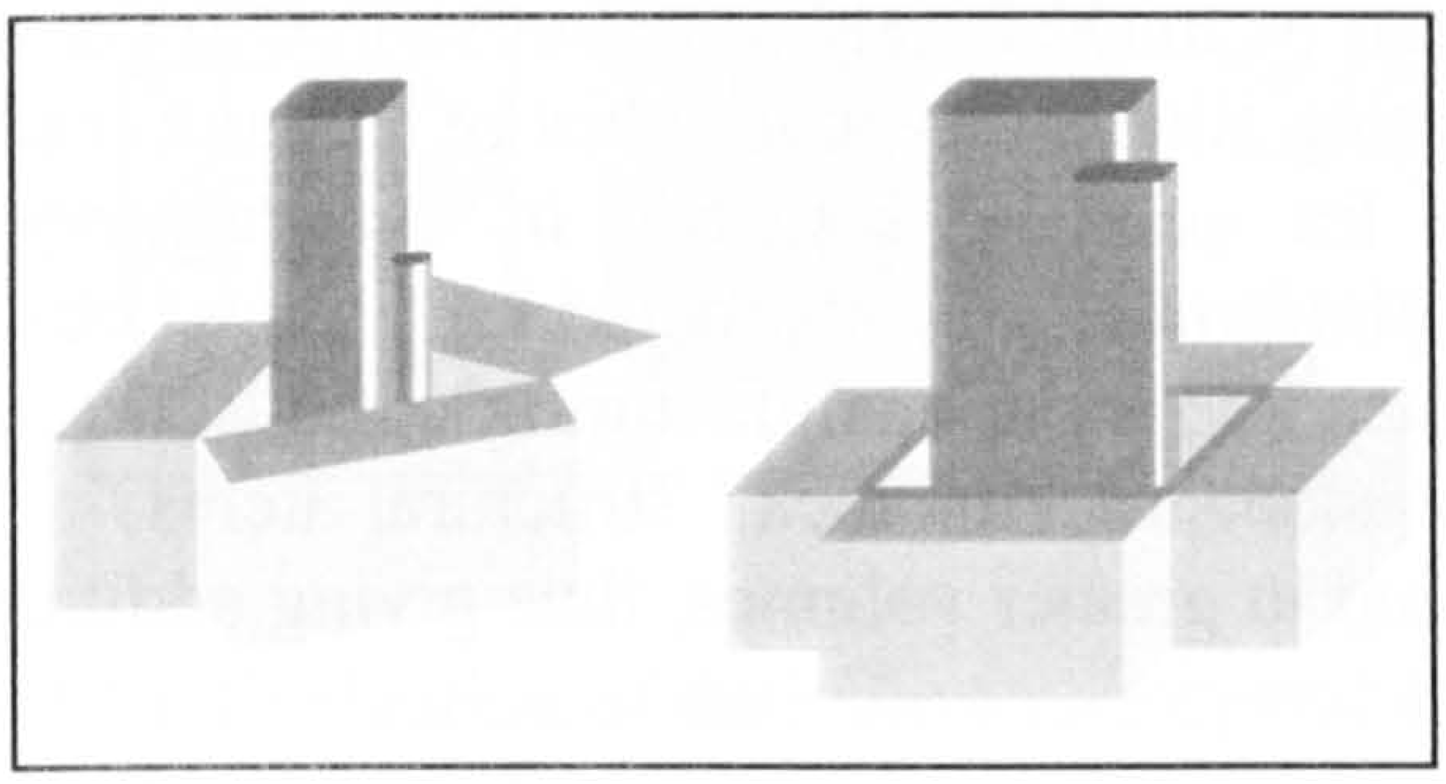


Figure 5-5 Accommodation of lengthy payload instruments on the modified box-module design concept

Payload instruments are mounted within the central void formed by attachment of the box-modules. Instruments may be attached to points on the central frame. Payloads may also be accommodated within the modules.

The other space-frame and skin-stringer designs lack the suitability for axial accommodation of a long payload within their own structure. A possible option here would be to accept that long payloads must be mounted along the side of the platform. This concept is illustrated in Figure 5-6. However, this presents additional difficulties with the supporting structures that would be required, and makes the architecture more difficult to standardise and pre-design.

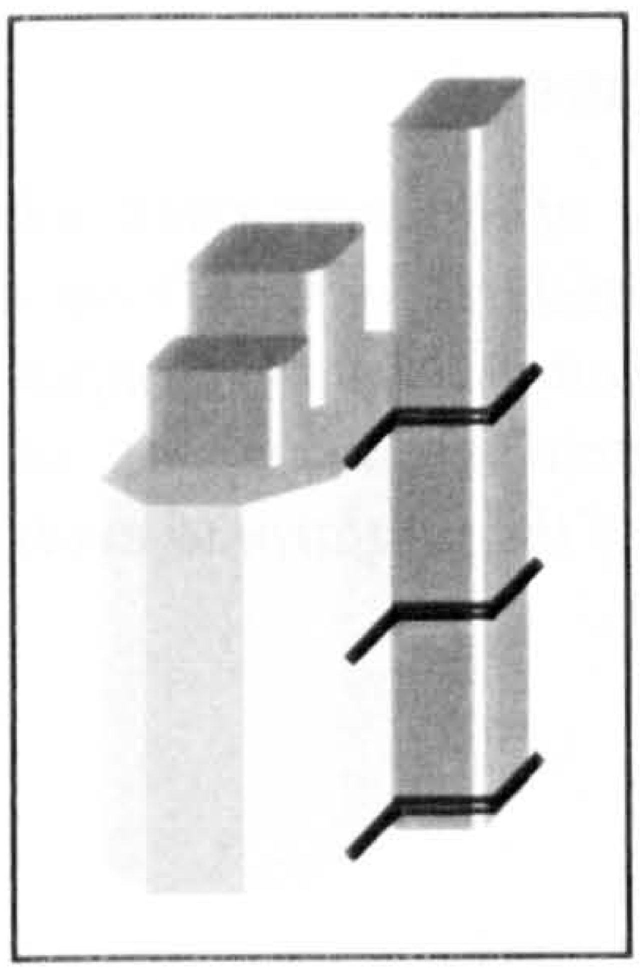


Figure 5-6 Accommodation of lengthy payload instruments on modified skin-stringer design: "side-mounted payload"

The payload is attached to reinforced mounting points on the sides of the main platform structure. The platform is made narrower to allow the payload to be accommodated within the launch volume. Additional, smaller payload items may be mounted on the top of the platform, as before. A similar scheme may be proposed for the space-frame/deck design.

5.2.4.3 Aperture provision

Performance in this area is governed largely by the outer construction of the platform. Therefore, any configuration that relies on solid skins for its structural strength and stiffness, is less likely to offer good aperture provision. This then implies that configurations constructed of any type of solid boxes will be less suitable; this is reflected in the performance scores. However, the performance of this type of design can be improved if the main payload mounting area is accepted as being on top of the main platform.

Recommendations/possible options:
For instruments requiring significant apertures, mounting on top of the main platform is the best option for most of the configurations. To enable apertures in the main spacecraft body (e.g. for attitude sensors), frames rather than skins may be more appropriate as the outer structure. Where numerous smaller apertures are acceptable, machined isogrid[7] structures may be employed instead of solid skins. These recommendations do not give new designs as such; they are merely suggestions that may be implemented on the existing designs.

5.2.4.4 Size adaptability

The configurations studied all performed quite well in this category. The poorest performer, the thrust tube configuration, can be made in any size, but scored lower due to the additional cost involved in constructing filament wound tubes of varying (rather than standard) sizes. The key element for good performance in this category is therefore that it should be possible for the platform to be constructed in varying sizes by using common, standard elements. This then keeps the manufacturing costs lower, and allows the fabrication of platforms from standard "in-stock" structural items. The standard elements may also be manufactured in greater volumes, thus giving additional possibilities for economies of scale.

Recommendations/possible options:
It is probably best to avoid the filament-wound tube design, if good size flexibility is to be obtained. The use of structures that may be made up of fairly standardised common elements should be used, as these can be produced in bulk and then assembled to the appropriate sizes. This implies framework-type structures and panels, which can be machined in sections and then cut to size.

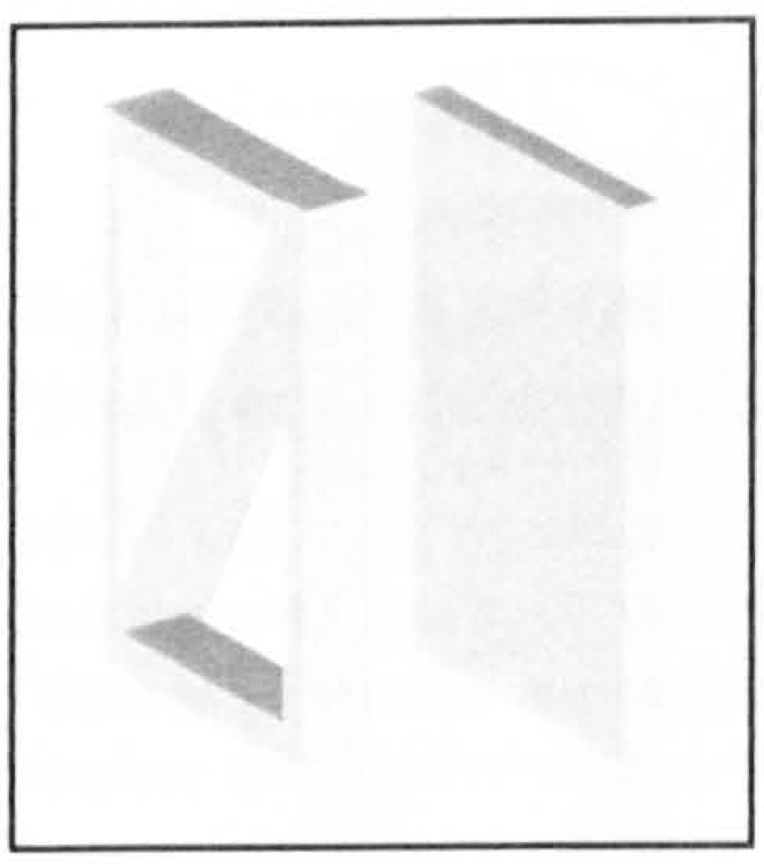


Figure 5-7 Standardised common elements

These may be easily assembled into structures of different sizes, whilst retaining the manufacturing advantages of standard units. These can also be easily interchanged with different functional units if required, for example, a radiator panel, multifunctional structure, or other specialist unit. This then enhances the ability of the platform to evolve.

5.2.4.5 Suitability for modularity/reconfigurability

The strongest performer in this area is the box-module design. Performance here is mainly driven by the way in which the structure is built up. Where the structure is composed of discrete sub-units, the configuration may be more easily altered, or divided into separate functional modules for integration/ testing. (Note that this is different to the idea of the standard elements outlined above. Standard structural elements do not necessarily permit the configuration to be divided into – or assembled in – modular parts. The sub-units described here are composed of both structure and spacecraft/ payload systems.)

Recommendations/possible options:
The skin-stringer and space-frame designs could be modified using ideas from the box-module concept, and be divided into separate "compartments", which are then stacked. These could be assembled separately, and different numbers and sizes could be used, giving reconfigurability. Payload could be either accommodated within a separate compartment module, or mounted on the top of the stack for greater field-of-view freedom. These modified concepts are shown in Figure 5-8

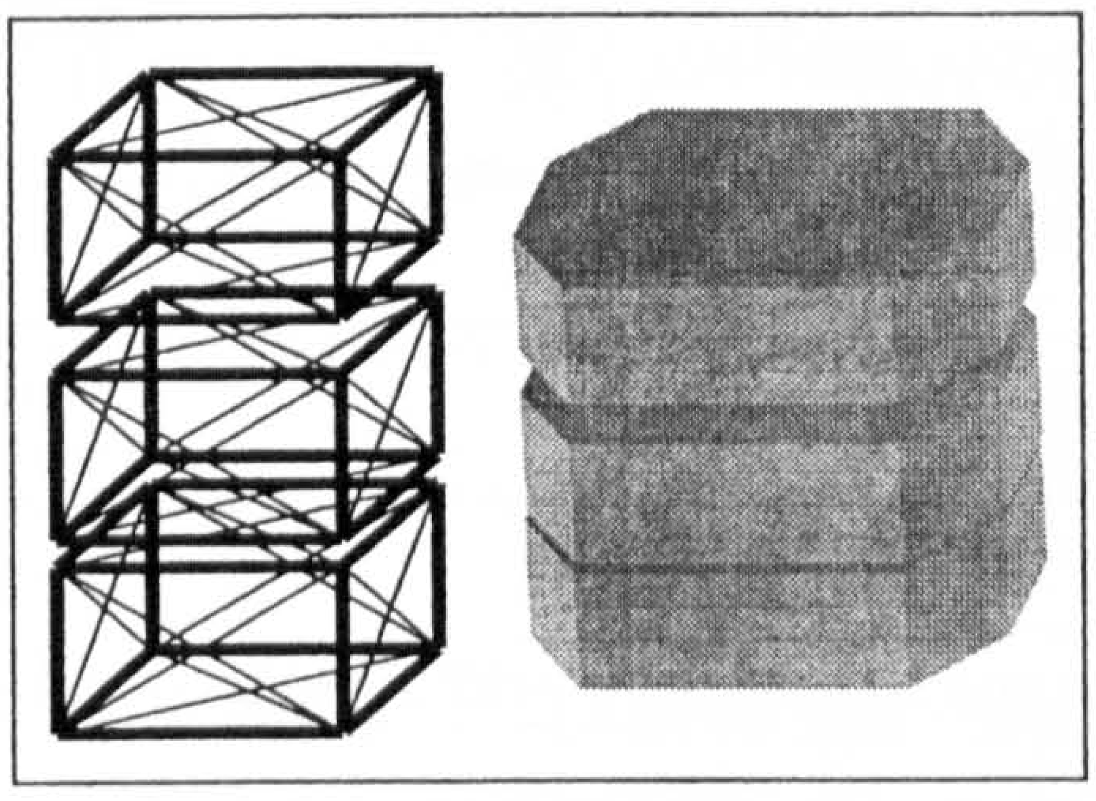


Figure 5-8 Modified space-frame (left) & skin-stringer (right) design concepts

The two original parent concepts are modified to improve their suitability for modularity and reconfigurability.

5.2.4.6 Evaluation of derivative and hybrid concepts

The four derivative concepts are now evaluated using the same parameters as before, to identify the most suitable choice of baseline configuration. The advantages and disadvantages given by the modification to the original designs are shown as follows, and new evaluation ratings assigned.

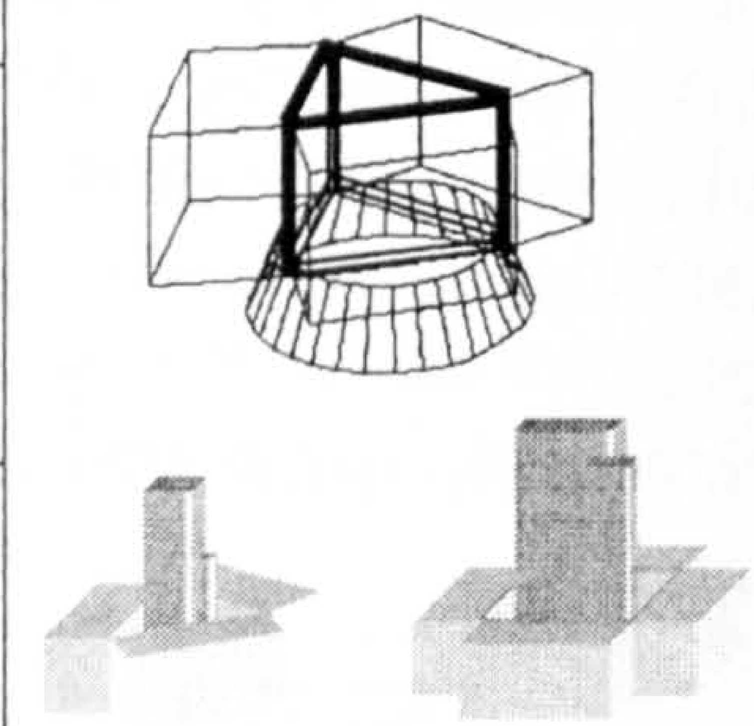
Configuration		**Modified box-module**		
Description		Modules with reinforced side form primary structure central tube, in which payload may be mounted. Different numbers of modules may be assembled without alteration being required.		
Components		Strut-frame primary structure. Remainder of boxes assembled from interchangeable lighter strut-frames, machined isogrid/ composite panels.		
Effect of modification				
Pros		Enhanced modularity Ability to easily reconfigure Better-defined load paths Improved payload mass capability Accommodation of long payloads, propellant tanks etc within central void		
Cons				
Evaluation Parameters				
	Multiplier		Rating	Total
Mass efficiency	1	Improved from original concept	3	3
Payload mass	3	High – large mass can be supported by the central frame structure formed by the modules	5	15
Volume efficiency	2	Medium – modules can be densely packed with equipment, but modules not densely packed together	3	6
Payload volume	3	High – payload may be mounted on top, within the central void, and within additional modules if necessary	5	15
Aperture provision	3	Good, if payloads are mounted on top/ in central void	5	15
Solar array surface	1	Quite poor, but cells may be mounted on outer surfaces of modules, and additional panels used to "close in" the outside if required.	2	2
Cost	2	Low-med, as may be made from standard components produced in larger quantities. Quite easy to assemble, reducing labour costs.	4	8
Size adaptability	3	High – different numbers of modules, modules may also be varied in size.	5	15
Suitability for modularity/ reconfigurability	3	Very suitable	5	15
Platform/ payload decoupling	2	Easy – payload can be mounted on top or within the central tube, at the final stage of AIT. Also easier to decouple/isolate thermally and from vibrations if required.	5	10
COTS equipment accommodation	2	Good – using cuboidal module shape suits the shape of COTS equipment.	5	10
Total evaluation score:				114

Table 5-6 Modified box-module configuration description and evaluation

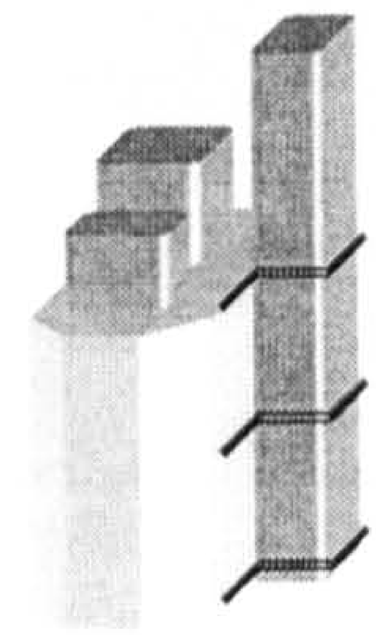
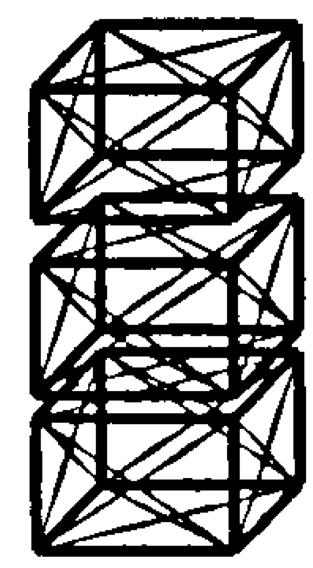
Configuration	**Side-mounted payload**			
Description	Large, long payloads mounted beside main platform body, and attached to hard-points built into the spacecraft platform structure. Modification of both skin-stringer and space-frame platform designs, which were unsuited to accommodation of long payload instruments.			
Components	As with original designs, but structure reinforced to take loads arising from attachment of the side-mounted payload.			
Effect of modification				
Pros	Accommodation of long payloads Good volume efficiency			
Cons	Difficult to standardise Unbalanced mass properties Load paths made very different			
Evaluation Parameters				
	Multiplier		Rating	Total
Mass efficiency	1	Lower than before, as reinforced structure required to support side loading. (If design is to be standardised, this extra structure would be present whether required or not).	4	4
Payload mass	3	Good – as before	4	12
Volume efficiency	2	High-density packing of equipment is possible, depending on prism shape	4	8
Payload volume	3	High – much better than before	5	15
Aperture provision	3	Good – better than before if payloads side- or top-mounted	5	15
Solar array surface	1	Good except for lost side face where the payload is mounted, therefore slightly lower than before	4	4
Cost	2	Low-med, depending on materials used	4	8
Size adaptability	3	Good – can vary size and shape of prism, vary number and position of bulkhead decks	4	12
Suitability for modularity/ reconfigurability	3	Low-med – not easily separable into discrete modules	2	6
Platform/ payload decoupling	2	Good, if payload mounted on side or top. Internally-mounted payloads more difficult	4	8
COTS equipment accommodation	2	Good if square prism shape used	4	8
Total evaluation score:				100

Table 5-7 Side-mounted payload configuration description and evaluation

Configuration	**Modified space-frame**	
Description	Separate space-frame "compartments" are stacked to form the platform structure. Gives discrete, interchangeable modules, which may be assembled independently. Size, shape, and number of modules may be varied.	
Components	Frames and trusses, as before. Deck structures to support equipment.	
Effect of modification		
Pros	Improved modularity Improved reconfigurability	
Cons	Cannot accommodate larger payloads Decreased mass efficiency	

Evaluation Parameters	Multiplier		Rating	Total
Mass efficiency	1	Lower than before – more superfluous structure	3	3
Payload mass	3	High	5	15
Volume efficiency	2	High – good packing efficiency possible	5	10
Payload volume	3	Medium – not suited to long payloads	3	9
Aperture provision	3	Good	5	15
Solar array surface	1	Low – medium, as would have to add outer skins – adding mass and reducing aperture provision.	2	2
Cost	2	Medium (higher if composites used)	3	6
Size adaptability	3	High – different sizes and numbers of compartments can be used	5	15
Suitability for modularity/ reconfigurability	3	High	5	15
Platform/ payload decoupling	2	Good – payload can be mounted on top and/or within its own separate compartment	5	10
COTS equipment accommodation	2	Good – based on squares, which is suited to the shapes of standard COTS equipment	5	10
Total evaluation score:				110

Table 5-8 Modified space-frame configuration description and evaluation

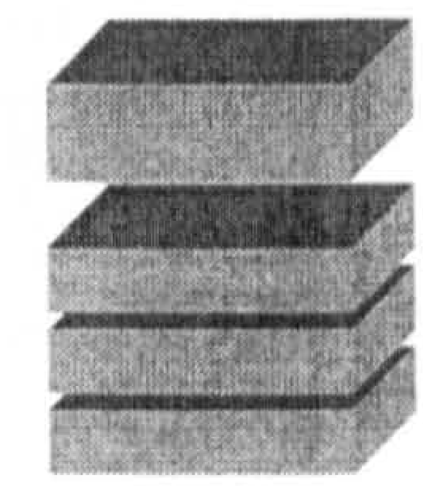
Configuration	**Modified skin-stringer**			
Description	Separate skin-stringer compartment "boxes" are stacked to form the platform structure. Size, shape and number of modules may be varied.			
Components	As before.			
Effect of modification				
Pros	Improved modularity Improved reconfigurability			
Cons	Cannot accommodate larger payloads Decreased mass efficiency			
Evaluation Parameters				
	Multiplier		Rating	Total
Mass efficiency	1	Lower than before – more superfluous structure	3	3
Payload mass	3	Quite high – payloads may be supported on the top, or in a dedicated compartment	4	12
Volume efficiency	2	High-density packing possible, depending on shape of prism	4	8
Payload volume	3	Medium – not suited to long payloads	3	9
Aperture provision	3	Low-medium (depends if payloads mounted on top)	3	9
Solar array surface	1	High – cells easily mounted on outer skin	5	10
Cost	2	Low-medium, depending on materials used	4	8
Size adaptability	3	High – different sizes and numbers of compartments can be used	5	15
Suitability for modularity/ reconfigurability	3	High	5	15
Platform/ payload decoupling	2	Good – payload can be mounted on top or within its own separate compartment if required	5	10
COTS equipment accommodation	2	Good, if square prism is used	4	8
Total evaluation score:				107

Table 5-9 Modified skin-stringer configuration description and evaluation

The analysis of these modified designs appears to indicate that the modified box-module configuration offers the best solution for the multi-use spacecraft platform. It gives the best performance in the key areas, and it is also compatible with the idea of using standard structural parts that was proposed earlier. Checking with the on-orbit configuration drivers identified previously, it also appears suitable for the range of missions to be targeted.

5.2.5 CHOICE OF CONFIGURATION CONCEPT

The parameter evaluation analysis of the different derivative concepts appears to indicate that the modified box-module concept offers the best overall performance. It combines the most useful attributes of several of the initial configurations, which enhance its performance in the payload mass, payload volume, and aperture provision areas. In addition, it is highly suited to the modular, reconfigurable architecture that was considered necessary for the platform.

The remaining design concepts are less comprehensive in the advantages they offer. The modified skin-stringer and space-frame designs both suffer from a more limited payload accommodation and remain less suited to the modular/reconfigurable ideal. The side-mounted payload arrangement, while improving the payload accommodation on these two designs, also introduced further complexity into the design and made it more difficult to offer standard configurations.

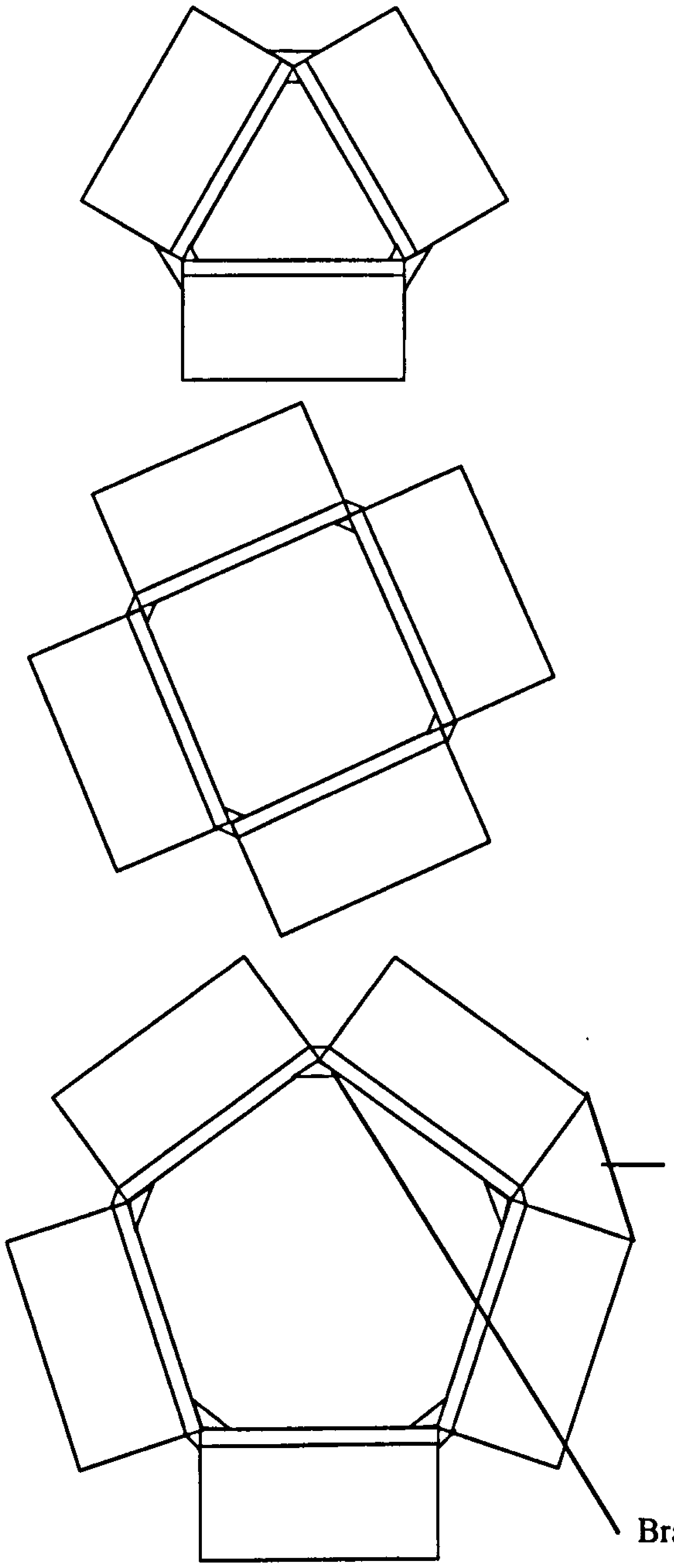
The modified box-module configuration is therefore selected as the baseline design for the platform.

The key characteristics of this candidate design are described as follows:

- The thrust-tube principle is combined with the box-module approach
- The box-modules are mainly formed as space-frame structures, but can also be closed with panels or isogrid sheets
- The central faces of the box-modules form an effective thrust-tube around a central void
- Different configurations can be formed, by the use of different numbers of the standard modules
- The general form of the modules is standard, but they can be made in different sizes to the same basic scheme
- The modules may be rotated to give a "short, wide" or "tall, thin" configuration
- The configuration may be stackable, depending on specific structural design

The configuration selected combines the desirable features of several of the original "parent" configurations analysed.

Concepts for general module layout, and the module construction, are shown in Figure 5-9 and Figure 5-10. These indicate the general configuration concept; further details, including dimensions and equipment accommodation, will be addressed in the structures subsystem design section.



Different configurations may be assembled by fastening different numbers of modules together. A taller, narrower configuration is achieved by turning each module through 90°, as previously described.

Mechanical fastening may be achieved by bolting the module frames onto brackets, sized and shaped specifically for each of the possible configurations.

The mechanical footprint for each module may then be kept the same, independent of the final platform configuration.

The "external voids", between the modules, may be enclosed by the use of additional external shear panels. Alternatively, they may be utilised by allowing equipment to extend outside its own supporting module, through apertures in the external frames. They may also be used for stowing equipment that will later be deployed, e.g. booms, inflatably-deployed arrays, radiators, payload SAR arrays, etc.

Example of closure by addition of external shear panel, to form an extra external compartment. A base plate may also be attached to the module frame, for support of equipment within the additional compartment.

Figure 5-9 General module layout, showing different configuration options

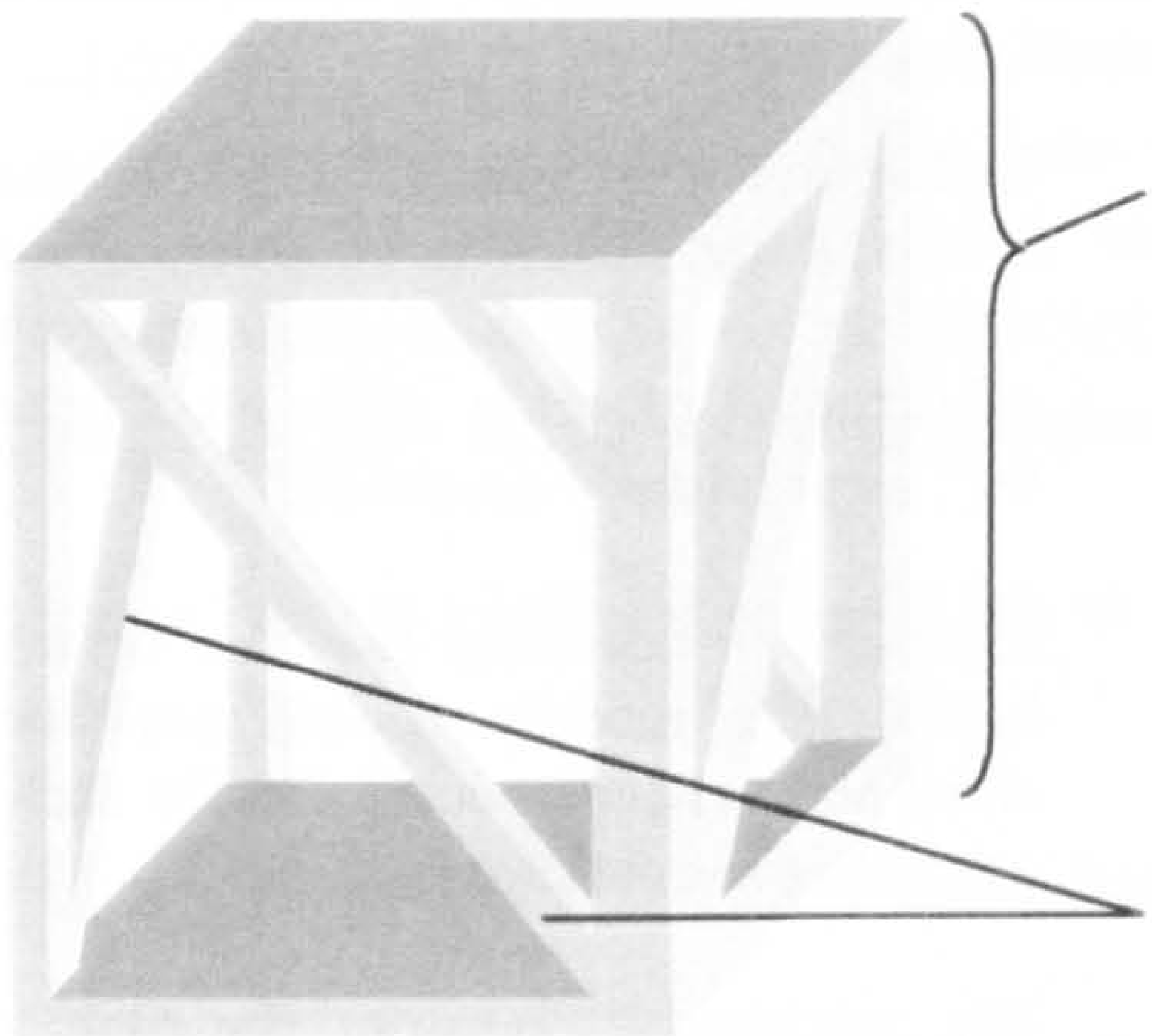


Figure 5-10 Module construction

Key benefits of the configuration concept:

- Can be assembled into a variety of configurations
- Is constructed from simple, standard parts, which are easily repeatable and give manufacturing benefits
- Structural models for all the different configurations may be assembled from a "standard kit", and tested to qualification levels. Flight units may then be tested at acceptance levels, and possibly only at module level. This reduces time and cost of testing, and enables it to be conducted at smaller facilities.

This section has introduced the baseline configuration for the platform. The required subsystems to be accommodated are now addressed. With design and sizing of the subsystems complete, the full platform architecture may then be determined. This will include decisions on the module division scheme – i.e. the interface positions. Options for this were identified in Chapter 2, and are addressed again with respect to the proposed configuration.

5.3 SUBSYSTEM TRADEOFFS, DESIGN AND SIZING

This section addresses selection of the platform subsystems. For each subsystem, design budgets are taken from the requirements previously derived, and the different solution options are examined. The options are then traded off, and the most appropriate solution selected. This includes sizing and selection for the different platform performance steps, and identification of the most appropriate interface positions (i.e. where the different modules can be broken out from the rest of the system).

5.3.1 COMMUNICATIONS & DATA HANDLING

5.3.1.1 Communications

The requirements derived in Chapter 4 for the communications subsystem are summarised as follows:

- 3Mbps downlink data-rate for general applications
- 50+Mbps high-performance option
- Low (Kbps) rate for communications missions housekeeping data/backup system

The required components for a spacecraft communications subsystem are shown in Figure 5-11.

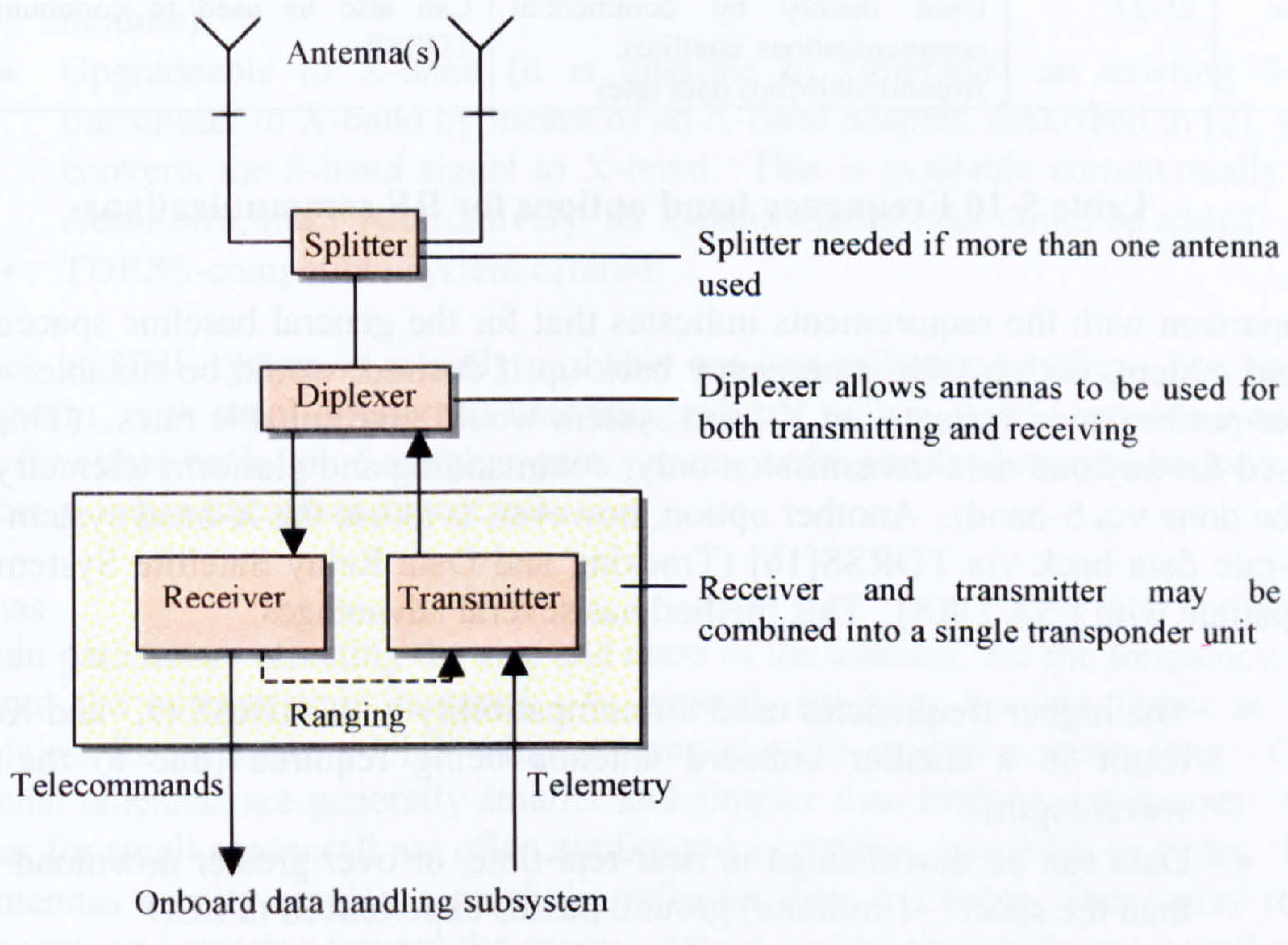


Figure 5-11 Key elements of spacecraft communications subsystem

The mass and volume of the communications subsystem is principally taken up by the transponder unit. This unit may be designed as separate receiver and transmitter, although this is likely to give an associated increase in size and mass due to the additional housing structure. However, it may enable a greater flexibility; for example, redundancy may be desired in the receiver, but not in the transmitter. The receiver could then be replaced with a redundant unit, without needing to use a fully-redundant transponder. Positioning two smaller units may also be easier in some architectures than positioning one larger one.

Frequency band

There are several options available for RF communications with spacecraft. These are discussed in Table 5-10.

Band	Frequency/ GHz	Typical usage, & data rates from LEO	Remarks
UHF	0.2-0.45	Small, v low cost spacecraft. Kilobit-per-second data rates	Not generally viable for commercial platforms unless as just a back-up system, as data rates too low.
S	2.5-2.69	General widespread use in LEO. Around 3Mbps data rates	In use for many years. Many COTS units available, including small units specially aimed at minisatellite market.
X	7.25-8.4	General widespread use for higher-performance applications. Around 150Mpbs data rates	Quite widespread use; increasing on smaller spacecraft. Several COTS units available. Can also be used to communicate via TDRSS.
Ka	25-27	Used mainly by commercial communications satellites. Around 600Mbps data rates	Can also be used to communicate via TDRSS.

Table 5-10 Frequency band options for RF communications

Comparison with the requirements indicates that for the general baseline spacecraft, an S-band system, with a UHF emergency back-up if desired, would be suitable. For the higher-performance variants, an X-band system would give suitable rates. (This would be used for payload data transmission only; commanding and platform telemetry would still be done via S-band). Another option, however, is to use the X-band system to send high-rate data back via TDRSS[16] (Tracking and Data Relay Satellite System – also compatible with ESA DRS). This method has several advantages:

- The higher frequencies used for compatibility with TDRSS (X- and Ka-band) result in a smaller onboard antenna being required (due to the shorter wavelength)
- Data can be downloaded in near-real-time, or over greater download periods than the short (~10minute) ground passes experienced in LEO
- If the TDRSS method is extended to include command, then the spacecraft can be monitored and controlled in near-real-time – this may be advantageous in the event of an anomaly, or if an unexpected science target of opportunity arises

However, there will be high costs involved in the use of this service, although its use may mean that a mission-dedicated ground station is no longer required. Depending on the need for greater data download, and the cost of using several ground stations to achieve the necessary ground contact, it may become cost-effective to make use of TDRSS. This type of decision would have to be made on a mission-by-mission basis. There may also be other restrictions arising from the use of NASA infrastructure by non-US project teams. However, it would make sense to offer a TDRSS-compatible system as an option for data-intensive missions, and those for which a more real-time level of spacecraft contact is desirable.

A decision must also be made as to whether to offer a transponder that supports ranging. This has always been considered a beneficial feature, as it allows the spacecraft position to be accurately measured during ground passes. However, now that the vast majority of spacecraft carry GPS systems on board, as part of their own attitude and orbit determination subsystem, the need for ranging is somewhat obsolete. Despite this, it may be useful as a safety back-up, in the event that there is a malfunction with the onboard GPS, or there is a problem or change of policy with the (US military-controlled) GPS infrastructure. There appears to be little advantage in not offering a coherent system supporting ranging.

Recommendation:

- Baseline S-band system
- S-band system supporting ranging
- Low- and high-power variants (via the use of an optional power amplifier module)
- Upgradeable to X-band (it is possible to "upgrade" an existing S-band transmitter to X-band by means of an X-band adapter, described in [2], which converts the S-band signal to X-band. This is available commercially from AeroAstro, Inc.) Alternatively, an X-band transponder could be added.
- TDRSS-compatible system offered.

The back-up UHF system is actually probably not cost-effective to offer. The basic S-band system, with omni-directional antennas and low power, would provide a suitable system for either back-up of a higher-gain system, or for use for low-rate housekeeping data from a communications spacecraft.

Antennas

The main parameters affecting the size and mass of the antenna, are the frequency being used, and the antenna gain required. In general, antennas become larger at lower frequencies; for example a UHF whip antenna may be around a metre long. Omni-directional antennas are generally smaller and simpler than high-gain antennas. Omni antennas for small spacecraft are often configured as patches, turnstiles or cones. High-gain antennas usually employ a parabolic reflector dish and feed. These may require deployment, and steering toward the ground station (unless the whole spacecraft can be used to point the antenna). An alternative to reflector antennas is the phased array antenna, which is "steered" electronically. These are smaller and more easily accommodated on smaller spacecraft, and do not require any mechanical steering assembly. They take the form of a flat panel, with associated electronics. Phased array antennas are generally more expensive than reflectors.

For general applications, omni-directional antennas are the most appropriate choice. One antenna on each end of the spacecraft will enable communication from any attitude, which is extremely useful for both initial acquisition of the spacecraft if operational attitude has not yet been attained, and for maintaining contact if the spacecraft enters an anomalous attitude. It also means that the spacecraft does not require specific pointing toward the ground station, which could be incompatible with other operational pointing requirements.

For higher data rates, a higher-gain antenna will be needed. However, omni-directional coverage should always be present via a back-up system, for safety. This could be an omni S-band or UHF system, which can be used for "rescue" commands and monitoring spacecraft response, even if signal has been lost from the main antenna due to pointing loss.

Recommendation:

- Use 2 omni-directional S-band antennas on all spacecraft, either as primary system or as a back-up
- For higher data rates in the S-band, use an S-band phased array antenna
- For the highest data rates, use an X-band phased array antenna, in combination with the back-up S-band omnis

Some examples of COTS communications units are given in Table 5-11. This gives an idea of the mass and size of the equipment.

Equipment	Manufacturer	Mass /kg	Size /mm
TDRSS Telemetry transmitter	Cincinnati Electronics Corp.	4.1	177 x 177 x 107
UHF Telemetry transmitter	Aydin Telemetry	0.45	89 x 64 x 34
UHF receiver	Aydin Telemetry	0.36	
S/L-band Telemetry transmitter	Aydin Telemetry	0.3	89 x 64 x 27
S-band telemetry transmitter	Fujitsu Laboratories Ltd	1.8	110 x 110 x 122
S-band transmitter	SSTL	0.5	160 x 120 x 20
S-band receiver	SIL	0.97	170 x 150 x 40
S-band transmitter	SIL	0.7	150 x 86 x 35
TDRSS user transponder	Cincinnati Electronics Corp.	5	
Conical spiral S-band antenna	SIL	0.19-0.3	70 x 70
S-band patch antenna	Physical Sciences Lab.	0.11	102 x 102 x 5
S-band patch antenna	Swedish Space Corp.	0.11	73 x 102 x 102
UHF turnstile antenna	Swedish Space Corp.	0.39	240 x 300
X-band antenna	Physical Sciences Lab.		25 x 102
X-band patch antenna	Physical Sciences Lab.		37 x 37 x 5
X-band phased-array antenna	Boeing (for GSFC's EO-1)	5.5	

Table 5-11 COTS communications equipment

Costs

Estimates for ROM costs of the various communications equipment is given as follows (based on mission costings given in [14], and cost estimating relationships from [1]:

- Low-gain antennas $10-50k
- Phased-array antennas $100-400k
- Transponders $100-500k (depending on power, performance, frequency)
- Typical overall subsystem cost $0.3-3m

Performance

The performance that may be expected of the various COTS systems examined has largely been derived from comparable spacecraft data and quoted product specifications. As the proposed platform is intended for use with a variety of different mission parameters, expected performance can only be estimated. The exact capabilities will be dependent on the subsystem units selected, and their detailed design characteristics. However, to give an idea of the basic performance level that may be anticipated, a sample link budget for the baseline omni-directional S-band downlink is shown in Table 5-12.

Parameter	**Symbol**	**Value**	**Remarks**
Frequency	F	2 GHz	S-band
Transmitter power	P	15 W	Reasonable estimate for small satellite baseline value
Transmitter power (dB)	P	11.8 dBW	
Transmit antenna gain	G_t	0 dBW	Omni antenna
EIRP	EIRP	10.8 dBW	Equivalent Isotropic Radiated Power
Propagation path length	S	2000 km	Assuming a ~600km LEO
Space loss	L_s	-164.5 dB	From: L_s= 20log c –20log S – 20log f
Propagation loss	L_a	-0.3 dB	Estimate, from[15]
Receive antenna diameter	D_r	5 m	Assumed – small ground station dish
Peak receive antenna gain	G_{rp}	37.8 dBi	From: G_{rp}= 20logπ + 20logD + 20logf +10logη - 20logc, with efficiency η=0.55
Receive antenna beamwidth	θ_r	2.1°	From: $\theta = 21/fD$
Receive antenna pointing error	e_r	0.2	Estimate, as above
Receive antenna pointing loss	L_{pr}	-0.1 dB	From: $L_{pr} = -12(e/\theta)^2$
Receive antenna gain	G_r	37.7 dBi	
System noise temperature	T_s	135 K	Estimate, as above
Data rate	R	4×10^6 bps	Should allow for a dedicated payload data rate of 3Mbps
E_b/N_0	E_b/N_0	25 dB	From: E_b/N_0 = EIRP + L_{pr} + L_s + L_a + G_r + 228.6 – 10logT_s – 10logR
Carrier-to-noise density ratio	C/N_0	91 dB-Hz	From: $C/N_0 = E_b/N_0$ + 10logR
Bit Error Rate	BER	10^{-5}	
Required E_b/N_0	Req E_b/N_0	4.5 dB	Using BPSK modulation and Viterbi coding
Implementation loss		-2	Assumed
Margin		18.4 dB	

Table 5-12 Link calculation example for omni S-band system

The example link calculation indicates that an S-band system, using simple omni-directional antennas and relatively low transmitter power, should be able to comfortably offer a payload data rate of at least 3Mbps.

Communications subsystem equipment list

The equipment required for the communications subsystem for each of the platform variants (the different "capability steps" in which the platform can be offered), are shown in Table 5-13. This includes estimates for the mass and size of each of the main components. These estimates are based on the data for available COTS units, indicated previously in this section. Exact dimensions may be changeable; if the platform were to go into production, some units may be developed or modified in-house by the platform manufacturer. Alternatively, if a relationship can be built with the equipment providers, they may offer units configured to better suit the particular geometry of the platform.

Platform variant	Equipment list	Mass/size/power estimates	Remarks
Basic	S-band receiver S-band transmitter Diplexer 2x omni-directional S-band patch antennas Harness, fastenings and mass margin	1kg, 170x150x40mm, 10W 0.7kg, 170x150x20mm, 20W 0.1kg, 100x50x20mm 0.1kg, 102x102x50mm 0.5kg	Estimate (Each) Estimate Includes general subsystem margin
Basic w. ranging	As above plus: Transponder link module	0.5kg, 170x150x20	Estimate
High-power S-band	As above plus: Power amplifier	0.2kg, 100x100x20, 20W	Estimate
High-gain S-band	As above plus: S-band phased array antenna	4kg, 600x300x80	Estimate, including control electronics
High performance	As for basic w. ranging plus: X-band Tx adapter X-band phased array antenna	0.3kg, 100x100x50, 5W 5kg, 250x300x100	Estimate Estimate based on EO-1
TDRSS option	TDRSS-compatible transponder 2x omni-directional S-band antennas S- or X-band phased-array antenna (option)	3kg, 170x150x70, 35W As before As before	Estimate, assuming can be made smaller than CEC unit For higher data rates

Table 5-13 Communications subsystem equipment list

This allows an estimated mass budget to be defined for the different variants. This is shown in Table 5-14.

Variant	Communications subsystem mass and power estimates
Basic	2.5kg, 10W (30W when transmitting)
Basic with ranging	3kg, 10W (30W when transmitting)
High-power S-band	3.2kg, 10W (50W when transmitting)
High-gain S-band	7.2kg, 10W (30W when transmitting)
High-performance	8.3kg, 15W (35W when tranmsitting)

Table 5-14 Estimated communications subsystem mass and power budget for the different performance variants

These estimates are considered to be towards the higher range of likely mass values, and include some additional margin.

A block diagram of the communications subsystem is shown in Figure 5-1, with the different equipment used for the different performance options.

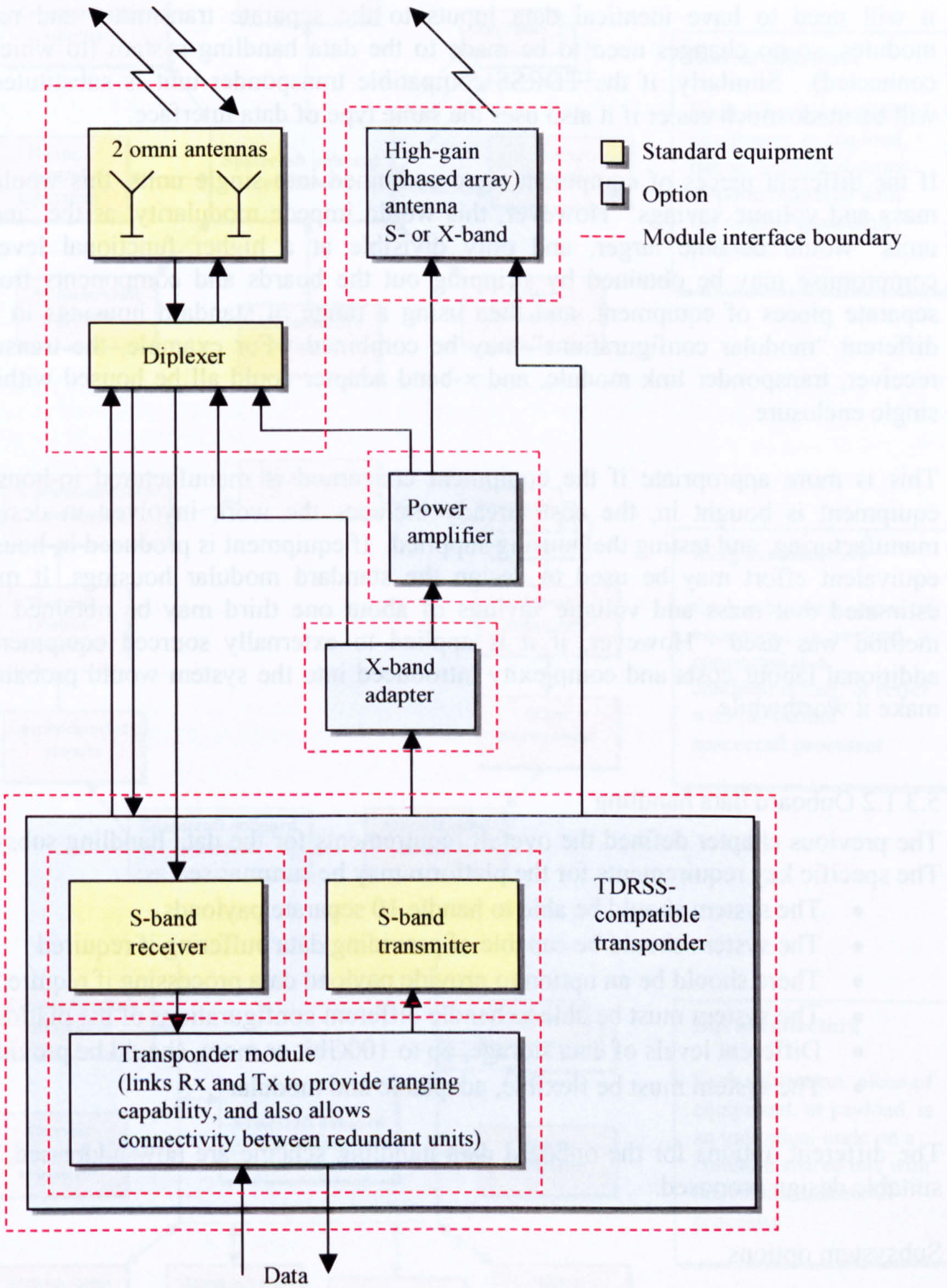


Figure 5-12 Communications subsystem block diagram

The different optional "add-ons" fit in with the modular concept; suitable functional interface boundaries are also indicated on the block diagram. These show where it should be possible to add, remove, or replace self-contained modular units. For example, it should be possible to transparently insert the transponder module. For this, it will need to have identical data inputs to the separate transmitter and receiver modules, so no changes need to be made to the data handling system (to which it is connected). Similarly, if the TDRSS-compatible transponder unit is substituted, this will be made much easier if it also uses the same type of data interface.

If the different pieces of equipment were combined into single units, this would give mass and volume savings. However, this would impede modularity, as the "modular units" would become larger, and only divisible at a higher functional level. A compromise may be obtained by stripping out the boards and components from the separate pieces of equipment, and then using a range of standard housings in which different "modular configurations" may be combined. For example, the transmitter, receiver, transponder link module, and x-band adapter could all be housed within one single enclosure.

This is more appropriate if the equipment concerned is manufactured in-house. If equipment is bought in, the cost already includes the work involved in designing, manufacturing, and testing the housing supplied. If equipment is produced in-house, the equivalent effort may be used to design the standard modular housings. It may be estimated that mass and volume savings of about one third may be obtained if this method was used. However, if it is applied to externally sourced equipment, the additional labour costs and complexity introduced into the system would probably not make it worthwhile.

5.3.1.2 Onboard data handling

The previous chapter defined the overall requirements for the data handling subsystem. The specific key requirements for the platform may be summarised as:

- The system should be able to handle 10 separate payloads
- The system should be capable of providing data buffering if required
- There should be an option to provide payload data processing if required
- The system must be able to handle different configurations of the platform
- Different levels of data storage, up to 100Gbit or more, should be provided
- The system must be flexible, adaptable and modular

The different options for the onboard data handling scheme are now addressed, and a suitable design proposed.

Subsystem options

There are a number of options for data handling architecture. These largely depend on the data communication topology used, and the distribution of the different subsystem functions. The main architecture groups are shown in Figure 5-13 on the following page.

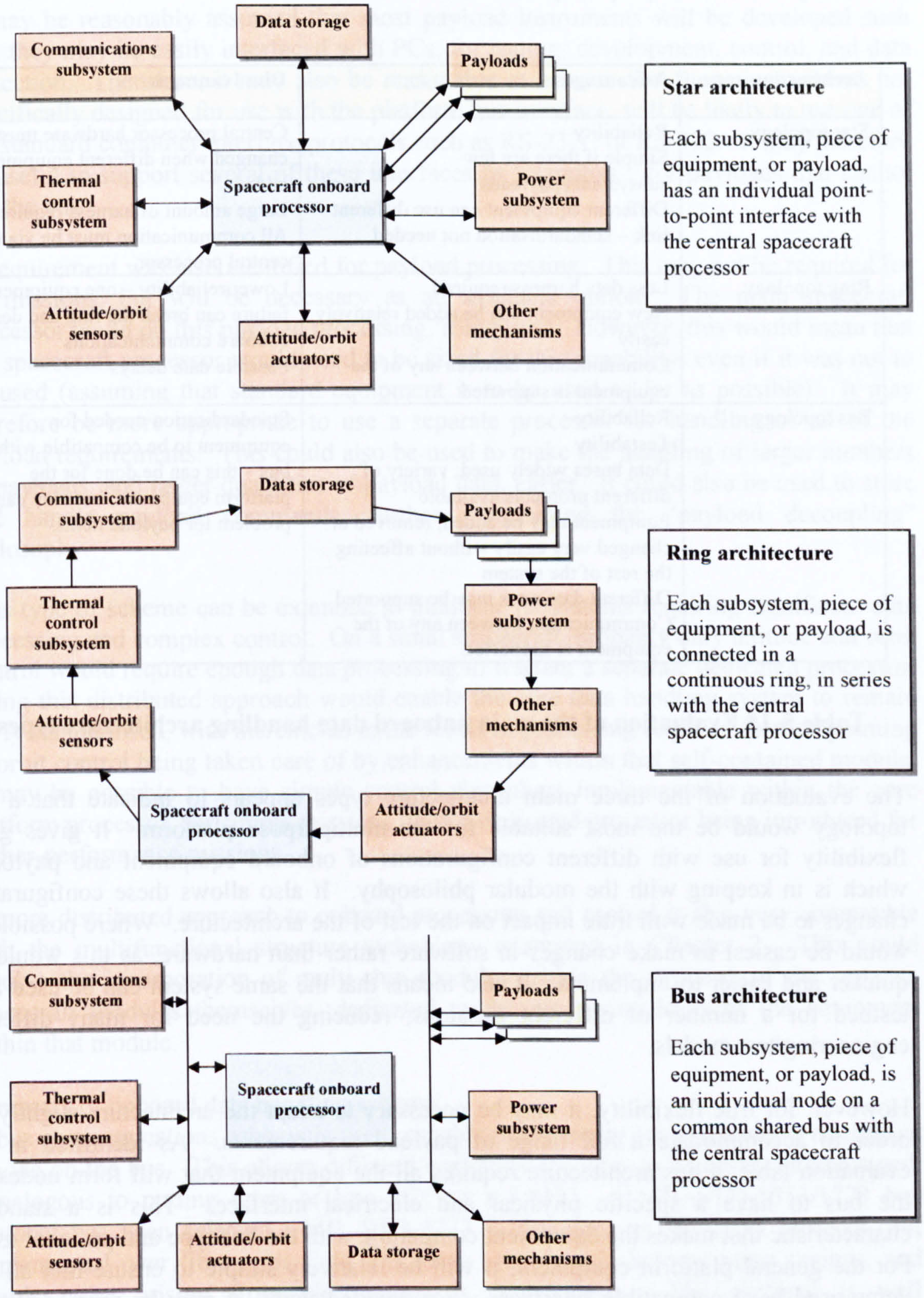


Figure 5-13 Onboard data handling subsystem: main architecture options

Each of these architectures has advantages and disadvantages that make them applicable to different types of spacecraft and mission. These are evaluated with respect to their suitability for the multipurpose platform, in Table 5-15.

Architecture type	Advantages	Disadvantages
Star topology	Reliability Simple if there are few subsystems/payloads Different equipment can use different link – standardisation not needed	Central processor hardware must be changed when different equipment added Large amount of harness required All communication must be via the central processor
Ring topology	Less data harness required New equipment can be added relatively easily Communication between any of the equipment is supported	Lower reliability – one equipment failure can break the ring and destroy onboard communications Possible data delays
Bus topology	Reliability Testability Data buses widely used; variety of different protocols available Equipment may be added, removed or changed very easily without affecting the rest of the system Different data rates may be supported Communication between any of the equipment is supported	Standardisation needed for equipment to be compatible with the bus – this can be done for the platform equipment, but may cause a problem for payload items.

Table 5-15 Evaluation of the main onboard data handling architecture types

The evaluation of the three main architecture types appears to indicate that a bus topology would be the most suitable for the multipurpose platform. It gives good flexibility for use with different configurations of onboard equipment and payloads, which is in keeping with the modular philosophy. It also allows these configuration changes to be made with little impact on the rest of the architecture. Where possible, it would be easiest to make changes in software rather than hardware, as this would be quicker and easier to implement. It also means that the same system can be used as a testbed for a number of different missions, reducing the need for many different engineering/test models.

However, for true flexibility, it may be necessary to adapt the architecture slightly, in order to accommodate a full range of payload requirements. As identified in the evaluation table, a bus architecture requires all the equipment that will form nodes on the bus to have a specific physical and electrical interface. This is a standard characteristic that makes the equipment compatible with the bus type and protocol used. For the general platform equipment, it will be relatively simple to ensure that all the items used have compatible interfaces. For payload items, it may be more difficult. Some payload items may have been developed for use with other interfaces, and the payload provider may not wish to (or have the time and money to) make changes to the design. This may be the case if a payload has already been developed, ahead of the selection of the platform on which it has the opportunity to fly.

It may be reasonably assumed that most payload instruments will be developed such that they may be easily interfaced with PCs, for testing, development, control, and data collection. Therefore, it may also be reasonable to assume that those instruments not specifically designed for use with the platform bus interface, will be likely to use one of the standard computer interface protocols such as RS-232C or RS-422. It may therefore be useful to support several of these interfaces as "standbys", for payloads that cannot use the bus.

A requirement was also identified for payload processing. This may not be required for all missions, but will be necessary as an available option. The main spacecraft processor could do this payload processing, if required. However, this would mean that the spacecraft processor would need to be sized for this capability, even if it was not to be used (assuming that standard equipment is to be used as far as possible). It may therefore be more appropriate to use a separate processor for handling of all of the payload requirements. This could also be used to make the handling of larger numbers of payloads, and larger quantities of payload data, easier. It could also be used to store and handle payload commands, further maintaining the "payload decoupling" philosophy.

This type of scheme can be extended to platform subsystems requiring significant data processing and complex control. On a small spacecraft, probably only attitude and orbit control would require enough data processing to warrant a separate dedicated processor. Using this distributed approach would enable the core data handling system to remain fairly standardised, with increments in the levels of processing required for fine pointing or orbit control being taken care of by enhancements within that self-contained module. It may be possible to have simple control algorithms implementable within the core platform processor, for simple missions, with a dedicated processor being introduced for higher-performance missions.

A more distributed approach to onboard processing and control is also very compatible with the multifunctional structure technology examined in Chapter 2. This could involve the incorporation of multi-chip modules within the structure of the separate spacecraft modules themselves, dedicated to the processing tasks of the equipment within that module.

Summary of onboard data handling scheme:
A bus communications scheme is used, capable of accepting each subsystem module as a node on the bus. This allows different capability modules to be easily interchanged. (Analogous to putting more or less PCs on a LAN.) A MIL-STD-1553/1773 bus protocol has been selected. MIL-STD-1773 is the same as the 1553 protocol, but implemented over fibre-optic link; this gives considerable harness mass savings, and avoids noise arising from electromagnetic interference. The protocol is well established, and compatible with the majority of COTS spacecraft equipment. Couplings to the bus should be made in such a way that the failure of a single unit will

not affect the operation of the bus itself; with a fibre-optic bus this is inherent, with traditional cabling inductive couplings should be used.

For payloads, there is an optional "payload controller", which forms its own node on the bus. This can poll payload instruments, buffer data, and pass on commands received via the bus. This controller can then be modified (mainly in software), to interface with different numbers and types of payload, without affecting the rest of the data handling subsystem. A separate payload bus, controlled by the payload processor, allows differing numbers and types of payloads to be incorporated easily into the system. If necessary, this controller can be used to process (i.e. compress) payload data, before on-board storage and transfer to the ground.

For missions with very simple payload requirements, payloads can interface directly with the main platform processor. In both of these schemes, additional, back-up payload interfaces are available via RS232C or RS422, for payloads incompatible with the bus.

For attitude determination and control applications, a separate processor is used. This processor shares the main platform bus, and may also be used to give a degree of redundancy, if it also carries a basic back-up emergency control system for the main platform.

The architecture of the onboard data handling scheme for the platform, including the optional items, is shown in Figure 5-1.

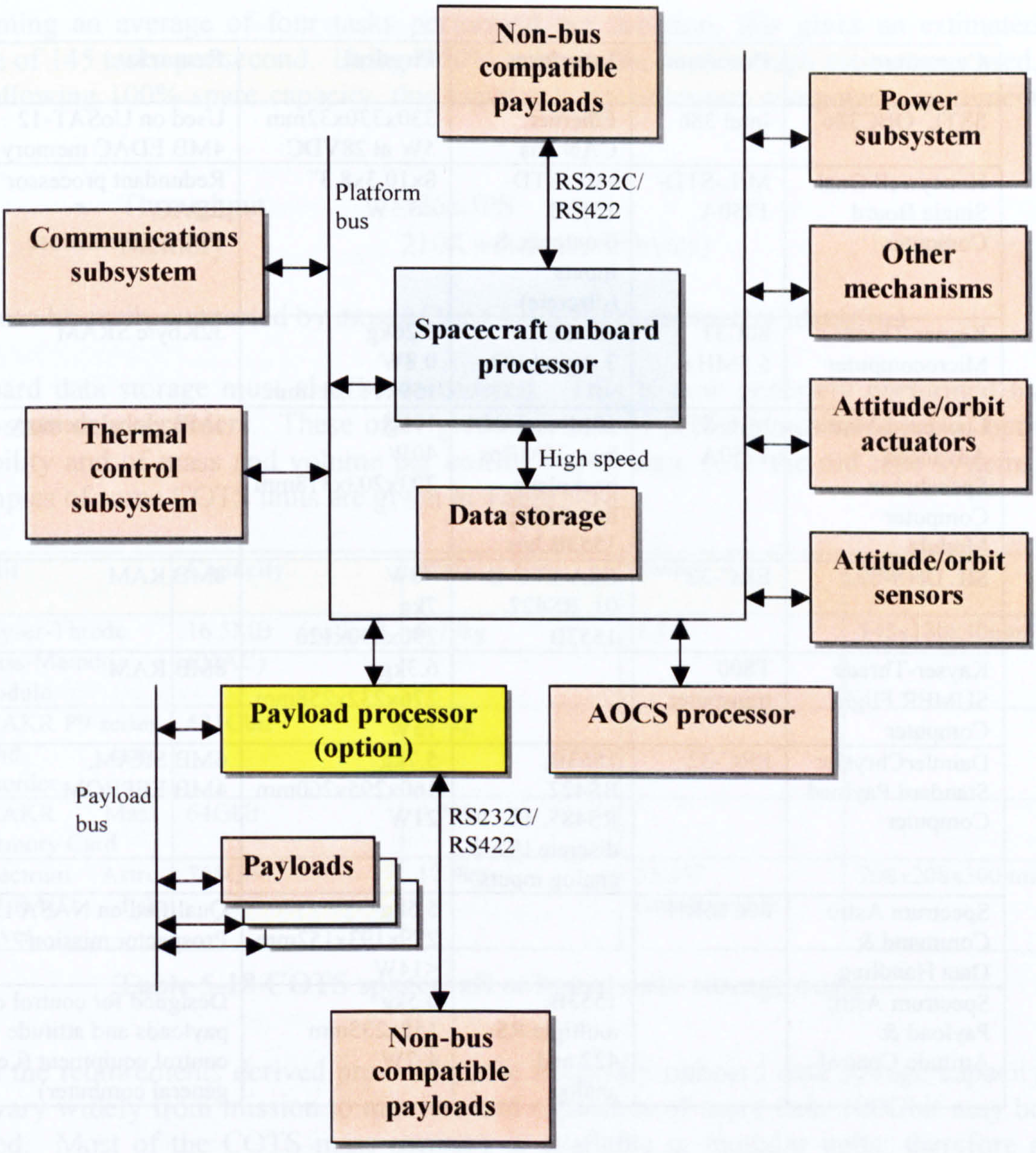


Figure 5-14 Onboard data handling architecture

Equipment selection

There are now a reasonable number of COTS spacecraft computer systems available, including equipment suitable for small satellites. For the proposed platform, it is likely to be best to purchase a COTS unit, and then make any necessary (minor) modifications, rather than develop a system from scratch. Commercial space-qualified equipment would be most appropriate for a platform that may be required to operate in a range of orbits (and therefore radiation environments), and for lifetimes of 5 or more years. The characteristics and performance of space-demonstrated computers is shown in Table 5-16.

Computer	Processor	Interfaces	Physical	Remarks
SSTL OBC386	Intel 386	Ethernet, CAN bus	330x330x32mm 5W at 28VDC	Used on UoSAT-12 4MB EDAC memory
Honeywell Dual Single Board Computer	MIL-STD-1750A	MIL-STD-1553B 6 outputs, 8 inputs (discrete)	8x10.3x8.3" <5.5W	Redundant processor EDAC 100Krad total dose
Kayser-Threde Microcomputer	80C31 5.5MHz	RS422 3 digital I/O	0.26kg 0.8W 159x113x14mm	32Kbyte SRAM
Lockheed-Martin Advanced Spaceborne Computer Module	MIL-STD-1750A	10Mbps FDDI bus, 25MBps backplane bus, 1MBps 1553B bus	10kg 40W 203x203x318mm	1MB global memory
SIL DHS-S32	ERC-32	ESA TTC-B-01, RS422, 1553B	<8W 7kg 290x290x120	8MB RAM
Kayser-Threde SUMER Flight Computer	T800 transputer		6.3kg 276x232x258mm 15W	8MB RAM
DaimlerChrysler Standard Payload Computer	ERC-32	1553B, RS422, RS485, 12 discrete I/O, 8 analog inputs	5.7kg 160x295x260mm 21W	6MB SRAM, 4MB EEPROM
Spectrum Astro Command & Data Handling	80C86RH		5.5kg 279x193x152mm <14W	Qualified on NASA Lunar Prospector mission
Spectrum Astro Payload & Attitude Control	-	1553B, multiple RS-422 and analog	0.5kg 158x233mm 4-7W	Designed for control of payloads and attitude control equipment (i.e. not a general computer)

Table 5-16 COTS spacecraft computers

Selection of the onboard computer is dependent on the size and throughput of the applications it must support. Typical values for these are given in Table 5-17.

Function	Size (code+data) /Kwords (16-bit)	Typical throughput /KIPS	Typical execution frequency /Hz
Command processing	5	7	10
Telemetry processing	3.5	3	10
Autonomy	15	10	10
Fault detection & correction	17	20	5
Power management	1.7	5	1
Thermal control (if used)	2	3	0.1
Operating system s/w	25	0.3n + 0.05m *	-

*Where n = no. of tasks per second (calculated from the above functions and execution frequencies), and m = no. of data words handled per second (assumed to be 1000)

Table 5-17 Size and throughput for typical onboard data handling applications

Assuming an average of four tasks performed per function, this gives an estimated figure of 145 tasks per second. Using a 50% margin, due to the rough estimations used, and allowing 100% spare capacity, this equates to an estimated computer requirement of:

- Throughput 426KIPS
- Memory 210Kwords (430Kbytes)

This can be easily supported by most of the COTS flight computers identified.

Onboard data storage must also be considered. This is now generally performed by solid-state data recorders. These offer greatly increased performance in terms of both reliability and of mass and volume per unit of stored data, over the old tape systems. Examples of some COTS units are given in Table 5-18.

Unit	**Capacity**	**Mass**	**Power**	**Size**
Kayser-Threde Mass-Memory Module	16.5MB (without EDAC)	0.79kg	1.5W	145x130x40mm
SEAKR P9 series solid state recorder	521Gbit	8.9kg		
SEAKR Mass Memory Card	64Gbit	0.99kg		
Spectrum Astro SQ-RAID disk drives	768Gbit	12.9kg	55.4W (standby 15W)	208x208x300mm

Table 5-18 COTS spacecraft onboard data storage units

From the requirements derived previously, the necessary onboard data storage capacity will vary widely from mission to mission, and capacities of more than 100Gbit may be needed. Most of the COTS mass memory is available in modular units; therefore a range of onboard data storage capacities can easily be offered. The larger mass memory recorders shown above have storage densities of around 60Gbit/kg. To include some margin, it will be assumed that a commercial data storage equipment provider could supply units in 25Gbit modules of 0.5kg each.

The data storage modules would be expected to be housed within the data handling unit itself, to allow high-speed (probably parallel) access between the processor and the recorder. The data storage would probably take the form of individual cards, the dimensions of which could be specified so that they would easily slot into appropriate housings within the data handling unit. With successive additions of mass memory cards, a larger data handling housing will therefore be needed. These will, however, all be quite standardised, so will not require much additional design effort. It is not expected that the additional boards will introduce sufficient mass that the whole housing structure would require significant modification.

Based on the data for the available COTS equipment, a baseline equipment list and mass budget for the different variants of the data handling subsystem can be obtained. This is shown in Table 5-19.

Variant	Equipment list	Mass, size estimates	Remarks
Basic	Main platform data handling unit, with single mass memory module Data harness	6kg, 15W 300x300x120mm 2.5 kg	Payloads controlled from main processor Includes fastenings
Basic with extra data storage	As above, plus: 1 additional mass memory card per 25Gbit extra storage	As above plus: 0.5kg, 10W per extra card extra 20mm box depth	Increments of ~25Gbit
Redundant	Main unit, plus: Second processor for redundancy	As for main processor	Mass reduction can be obtained by housing both units together
Payload support	Main unit, plus: Second processor for payload processing and control Additional harness (if more payloads)	Estimate 4kg, 3W (smaller unit as only for payloads) 300x300x60mm 0.5kg	Dedicated processor for payload control and data processing (May be housed separately)
Redundancy + payload support	Combines redundant and payload support variants		

Table 5-19 Data handling subsystem equipment list

Assumptions:

1. Main data handling unit mass and dimensions based on available data on COTS units, assuming a reasonably basic main system (for missions without much payload processing or fine attitude control requirement).
2. Redundant processing unit assumed to be similar to main unit. In reality, mass savings would be possible, as not all the components must be doubled up.
3. Payload processing unit is assumed to be smaller than the main unit, as it is dedicated to the payload only (i.e. it will not require telemetry/telecommand processing boards or to perform other platform functions).
4. Harness mass assumes use of fibre-optic cables

This equipment list allows a mass budget for the different data handling subsystem variants to be estimated, shown in Table 5-20.

Variant	Data handling subsystem mass and power estimates
Basic	8.5kg, 15W
Basic with extra data storage	9.25kg, 45W (for 100Gbit storage capacity)
Redundant	14.5kg, 20W
Payload processing	13kg, 18W
Redundant with payload processing	19kg, 23W

Table 5-20 Data handling subsystem mass and power budgets

These are maximum expected masses, and include margin. Continuing advances in processing technology, and qualification of this technology for space use, will be expected to bring the mass of this type of equipment down further.

Costs
ROM cost estimates (references/sources as before) for the data handling system are given as follows:

• Main data handling unit	\$300-750k
• Software	\$250-500k
• Payload processor	\$200-400k
• Mass memory per Gbit	\$2-15k
• Harness	\$50-150k
• Typical data handling subsystem total cost	\$0.5-3m

Interfaces and module boundaries
The modularity analysis in Chapter 2 identified the interfaces and possible module boundaries for the onboard data handling subsystem. Referring to this analysis, the most appropriate module boundaries within the data handling subsystem appear to be as follows:

- Storage – modularity achieved by the use of mass memory units, the required number of which may be added as additional boards within the data handling unit. This also forms a "cross-discipline" modular function, as the data storage can also be used by other modules, such as the payload processor and attitude processor.
- Processing – the processing capability is upgraded in a modular fashion by the use of the optional separate payload processor. The "processing module" includes the command, telemetry, watchdog and autonomous functions. The module boundary is therefore in position 3 on the block diagram shown in Figure 2-11.

Other considerations
The data handling system must be easily testable at all stages in integration and test, up to and including on the launch vehicle while awaiting launch. Therefore, some form of test connector is used. This is employed such that it forms a "transparent" interface that looks like the normal spacecraft-to-ground RF communications link. While at the platform-level test phase, this interface can be used to check out both the spacecraft systems and the EGSE/ground operations equipment, without the need to transmit via RF within the test environment.

Once attached to the launcher, the spacecraft is not usually permitted to produce any RF signals, so again, any contact for testing, last-minute software updates or commanding is performed via the test connector. This is achieved via an umbilical link, generally provided by the launcher. (This link is specially designed to break apart on separation from the launch vehicle).

Harness and harness routing must also be considered. The detailed design and manufacture of the harness loom is quite complex, but the key factors to be considered in preliminary design are:

- Numbers and destinations of the cables – what is connected to what
- Harness routing – holes required through the structure, cable supports
- Cable bend radii
- Integration and test issues – access to connectors, integration sequences (to allow connections to be made at the correct times)
- Harness loom manufacture – how the loom is broken down into separate manufacturing units

It is suggested that the harness looms for the individual box-modules are assembled separately. These can then be integrated with the subsystems within each box, and tested as a stand-alone unit, with a simulator being used in place of the rest of the platform.

At the platform integration stage, the modules are mechanically attached to each other, and the inter-module harness connected. Due to the use of separate modules containing, as far as is possible, functionally-grouped equipment, the inter-module harness can be reduced. Module integration can be made quicker and easier by the use of a "patch-panel" type of arrangement, where a single connector (or a few connectors) simply links the different sections of harness across the module interfaces. The alternative is to leave free ends of harness for later insertion into the appropriate connectors on the other modules. This makes stand-alone testing more difficult and could require complicated routing of harness into other modules – increasing the risk of errors and/or damage to harness and connectors.

The patch-panel interface would also simplify integration of the payload with the platform. Patch-panels introduce additional mass due to the extra connectors (and possibly support brackets) used, but they will simplify the integration task, shortening AIT duration and therefore costs.

For enhanced sophistication and reduced integration time, use can be made of the multifunctional structure technique examined in Chapter 2. Data harness can be constructed as a thin "film" which is attached to the structure of the module itself. For standard box modules, the subsystem units can be simply "plugged in" to the structure and the harness. This is particularly suitable if composite structure, and fibre-optic data cables are used. It would give mass and volume savings, and simplify the integration process. This approach may be introduced as an evolution of the standard platform, when the different equipment requirements are well understood, and the plug-in integration method has been demonstrated and suitably qualified.

A further evolution would be to use wireless onboard communications – the Bluetooth™-type technology described in Chapter 2. This technique would require extensive testing and demonstration to qualify it for commercial space use, but the potential to replace most of the data harness with low-power RF links would offer:

- Volume and mass benefits – huge reduction in required cabling
- Integration effort benefits – equipment could be placed anywhere, without harness routing and connection concerns
- Time savings – little or no data harness to design and manufacture

This is probably some years off, but the overall modular architecture of the platform means that it would be possible to gradually implement this technology as it becomes more mature. A suggested starting point would be to add some redundant sensors (such as sun sensors) employing the technology, and examine their performance as part of the baseline system. This could give a step-wise approach to a shift to an all-wireless platform.

5.3.2 ATTITUDE & ORBIT DETERMINATION & CONTROL

In the requirements definition in Chapter 4, it was assumed that all missions targeted to use the proposed platform will require active attitude control, and that some would require propulsion for orbit acquisition and control. Furthermore, it was also determined that for some attitude requirements, thrusters would be required to meet the necessary torque demands. For design purposes, therefore, attitude control and propulsion are considered separately. Propulsion may form part of the attitude control subsystem, but is better handled as a separate subsystem module. Attitude determination and control is considered first.

5.3.2.1 Attitude determination and control

Attitude determination

A range of different pointing requirements was identified in Chapter 4, with precisions ranging from arcseconds to the order of one degree. The accuracy achievable is dependent on the accuracy of the pointing knowledge, the control algorithms used, and the characteristics of the attitude control actuators used to effect attitude adjustments.

Pointing knowledge is obtained via the use of onboard sensors. The type, number, and position of the sensors used dictate the possible accuracy of the attitude determination system. Typical performance and characteristics of the different types of attitude sensors are shown in Table 5-21.

Sensor	**Accuracy**	**Specifications**	**Remarks**
Miniature Earth horizon sensor	± 0.15° to 1.15°	0.15-0.28kg, <0.7W 1kg, 1-5W for redundant 3-telescope assembly	Based on EDO-Barnes COTS units [3,4]
High-accuracy Earth horizon sensor	0.02-0.03°	<2kg, 1.5W for redundant, 3-telescope assembly	Based on EDO-Barnes COTS unit
Sun sensor	0.03°	100g 75x90x25mm No power draw	Based on EDO-Barnes COTS unit 4 units give full coverage of celestial sphere
Star tracker	15arcsec	1.8kg, 2.8W 144x144x280mm	Includes baffle Based on SSTL COTS unit
High-accuracy star tracker	<3arcsec	5.4kg, 10W 180x260mm	Based on Ball Aerospace COTS unit

Inertial Reference Unit		0.75kg, 86x89mm 12W	Based on Litton COTS unit Uses ring laser gyros
Solid state microgyros		0.05kg, 2W (for 2-axis rate gyro)	High drift rates (i.e. need regular updating)
GPS interferometry	0.5°*	1kg, <7W 300x160x35mm (plus antennas)	*Depends on separation of antennas Based on SSTL COTS equipment
Magnetometer	10nTesla sensitivity 10Hz	0.295kg 130x90x36mm <<1W	Based on SSTL COTS unit

Table 5-21 Attitude sensor performance and characteristics

From the table above, it can be seen that for missions requiring high attitude accuracies, a star tracker will be required. This unit will generally be mounted co-aligned with the instrument requiring accurate pointing, to avoid any mounting misalignments causing payload pointing errors. For missions with lower accuracy requirements, Earth and sun sensors will be acceptable. To give full attitude knowledge, sun sensors will generally need to be mounted on all faces of the spacecraft. This gives the advantage of allowing the spacecraft to determine its attitude from a "lost in space" situation. Earth horizon sensors are particularly suited to nadir-pointing missions such as Earth observation.

Gyroscope- and accelerometer-based inertial reference units provide information on changes in velocity and rotation, but absolute orientation must be updated at intervals from an externally-referenced source. Drift rates (and therefore required update intervals) vary according to the technology used. IRUs are useful for use in conjunction with star trackers, when fine attitude determination is required, and during attitude and orbit control manoeuvres.

GPS interferometry utilises signals received by 2 or more GPS antennas, to give information on the relative position of the antennas (and hence spacecraft orientation). This technique obviously gives greater accuracy with increasing separation of the antennas – thus limiting the accuracy possible on board a small spacecraft. However, GPS can also allow attitude recovery from a lost in space situation (depending on the number of antennas used). It also gives valuable orbit position data.

Recommendations:

- Sun sensors as general baseline for all missions (gives initial attitude acquisition)
- Star tracker for high-accuracy missions, in conjunction with IRU
- Earth horizon sensors for Earth observation/ Earth-referenced missions
- Magnetometers for missions employing magnetorquers

Attitude control

Required torques for different slew rates have been identified in Chapter 4, together with the likely slew-rate demands for different mission types. The following analysis examines the performance requirements to achieve different attitude manoeuvres.

Assumptions for the following figures:

- Spacecraft are assumed to be cubic, and of a uniform density of 150km/m3.
- For the thrusters, the moment arm is assumed to be half the spacecraft linear dimension, i.e. it is assumed that the thrusters are mounted in the centre of the spacecraft side faces.

Figure 5-15 shows that for larger spacecraft, and those requiring higher slew rates, the torques required may not be achieved by the use of wheels.

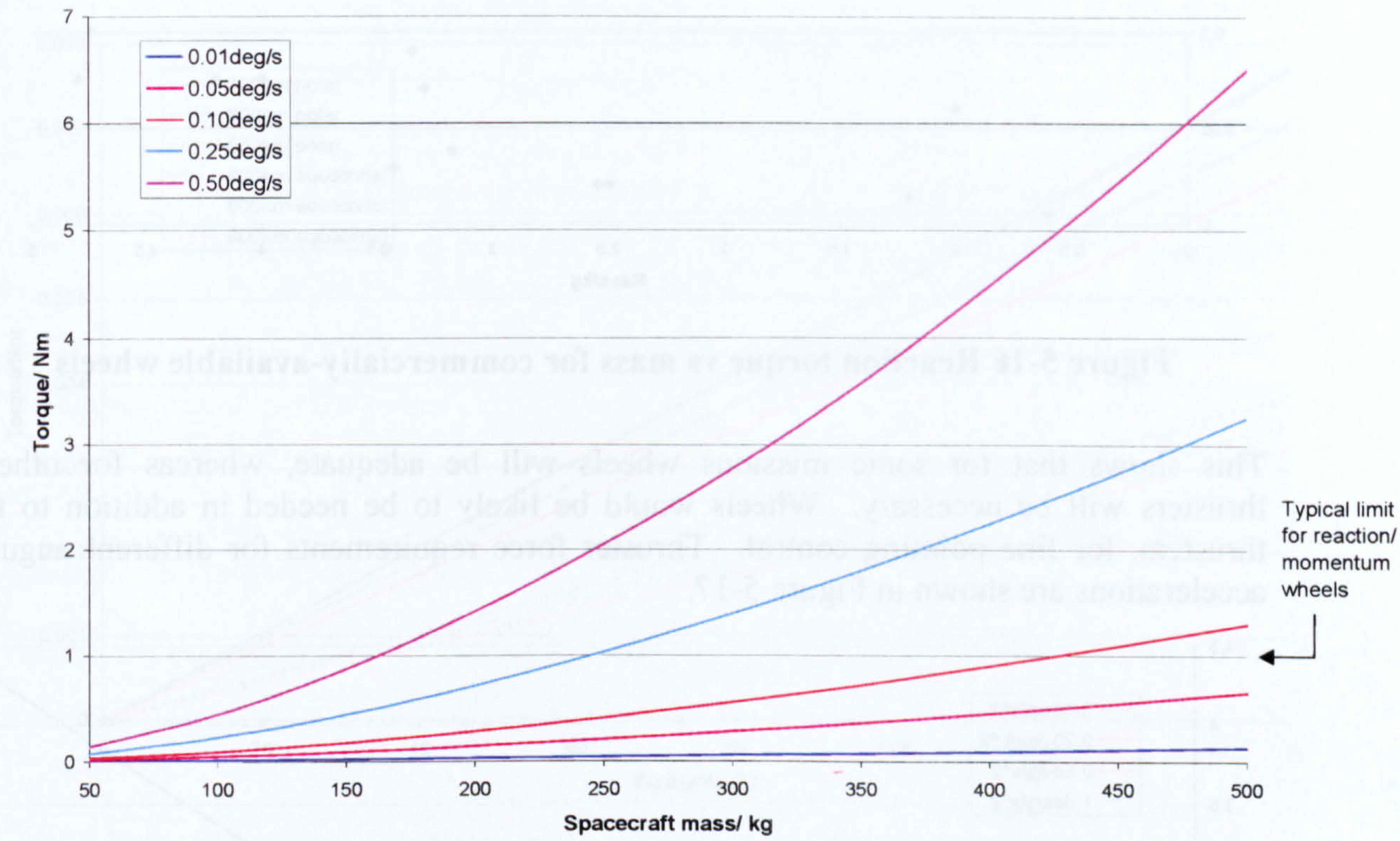


Figure 5-15 Torques required to achieve different slew rates, vs spacecraft mass

In fact, the limits for wheels suitable (in size, mass, and power) for small spacecraft applications is lower than the generally accepted level of around 1Nm, as is indicated in Figure 5-16.

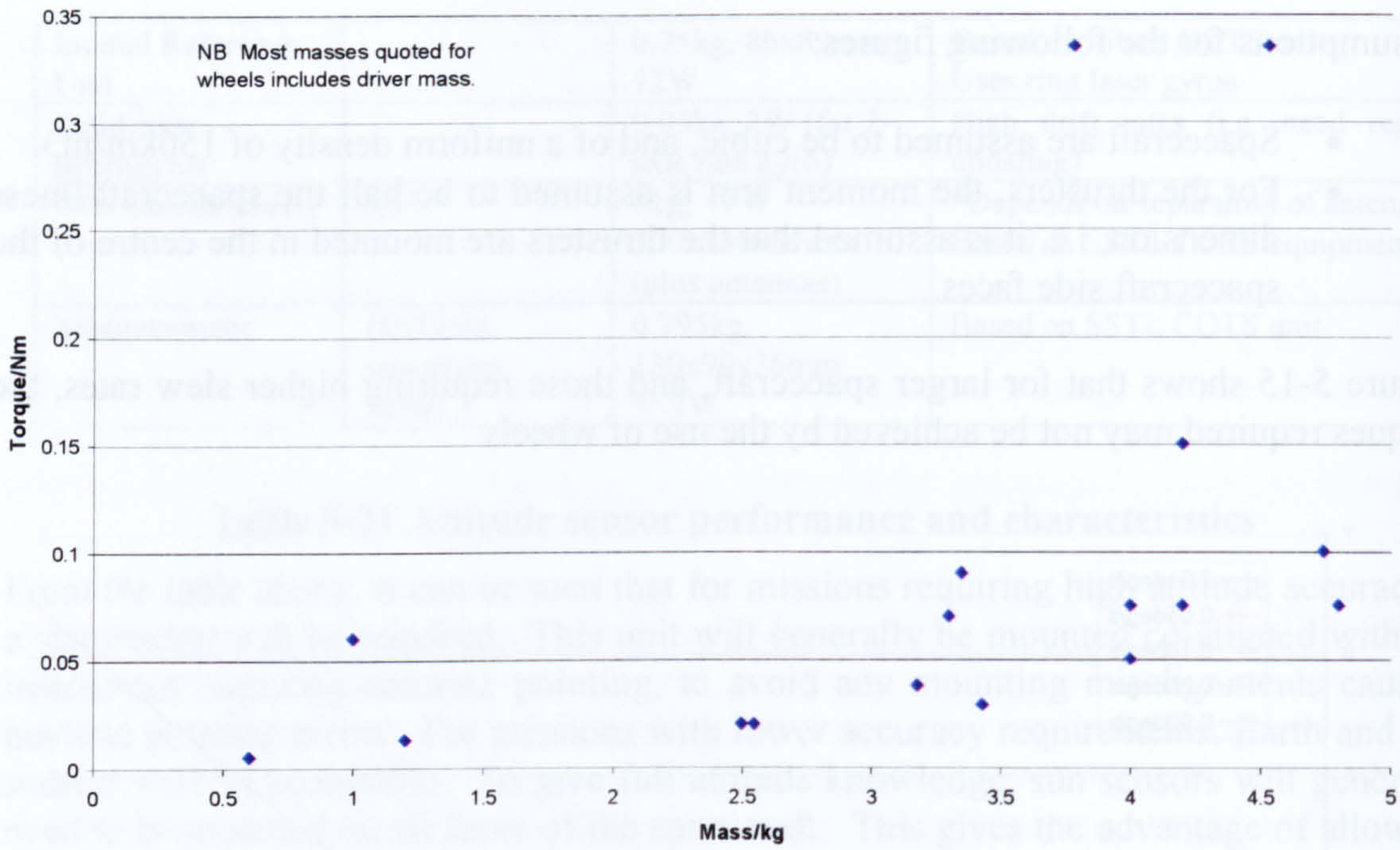


Figure 5-16 Reaction torque vs mass for commercially-available wheels

This shows that for some missions wheels will be adequate, whereas for others, thrusters will be necessary. Wheels would be likely to be needed in addition to the thrusters, for fine pointing control. Thruster force requirements for different angular accelerations are shown in Figure 5-17.

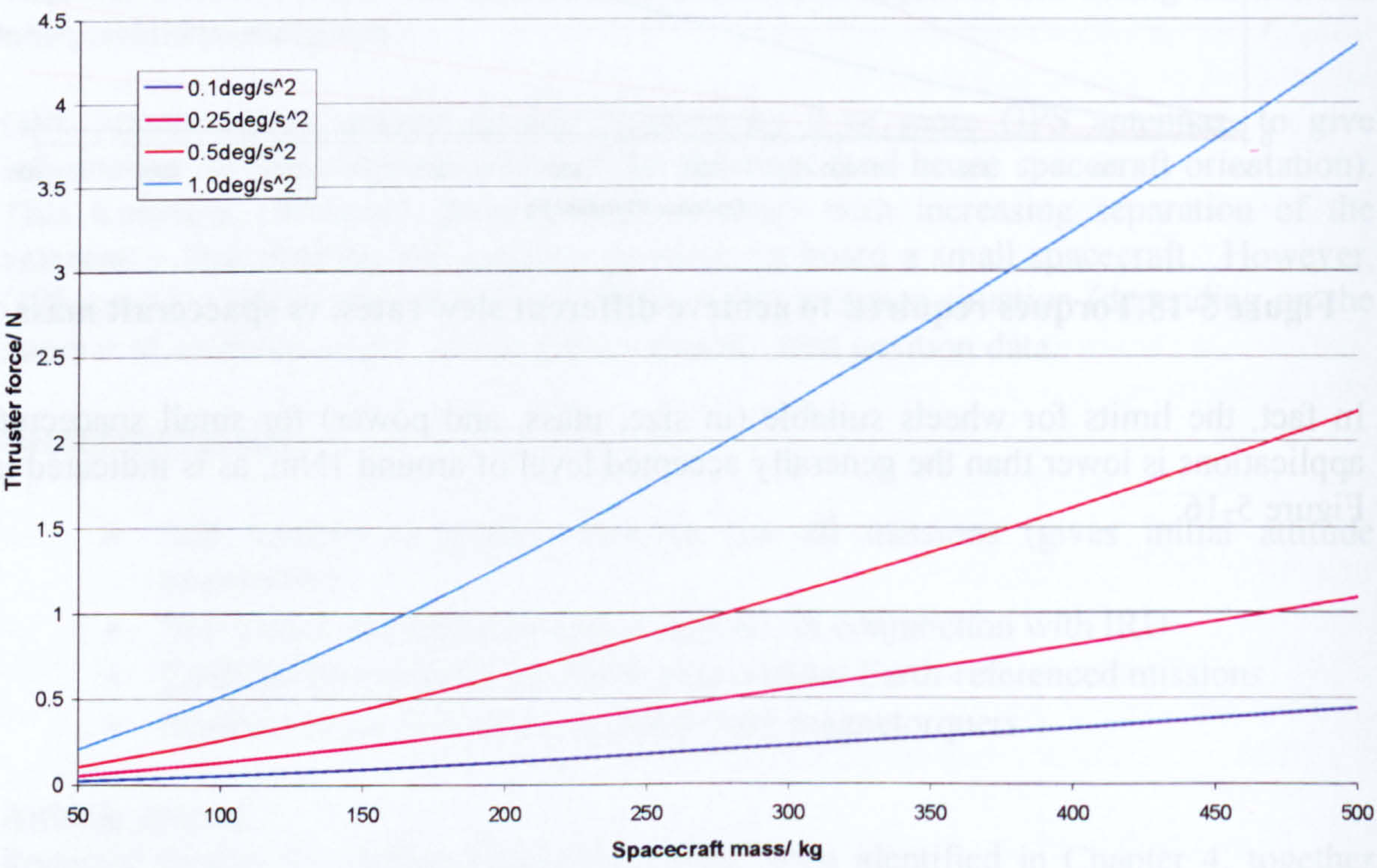


Figure 5-17 Estimated thruster forces required vs spacecraft mass, to give a range of angular accelerations

This would be likely for astronomy missions requiring fast response to transient events, for example. Thrusters will also be required if orbital adjustments are needed (such as in GEO, or for inclination changes). For many of the missions identified, however, the key requirement is to maintain one particular attitude (such as nadir pointing) to the defined accuracy and stability. This impacts more on the choice of attitude sensors, and the onboard attitude processing and control algorithms.

For low torque requirements, magnetorquers may be used (in LEO). Torque levels of magnetorquers are shown in Figure 5-18.

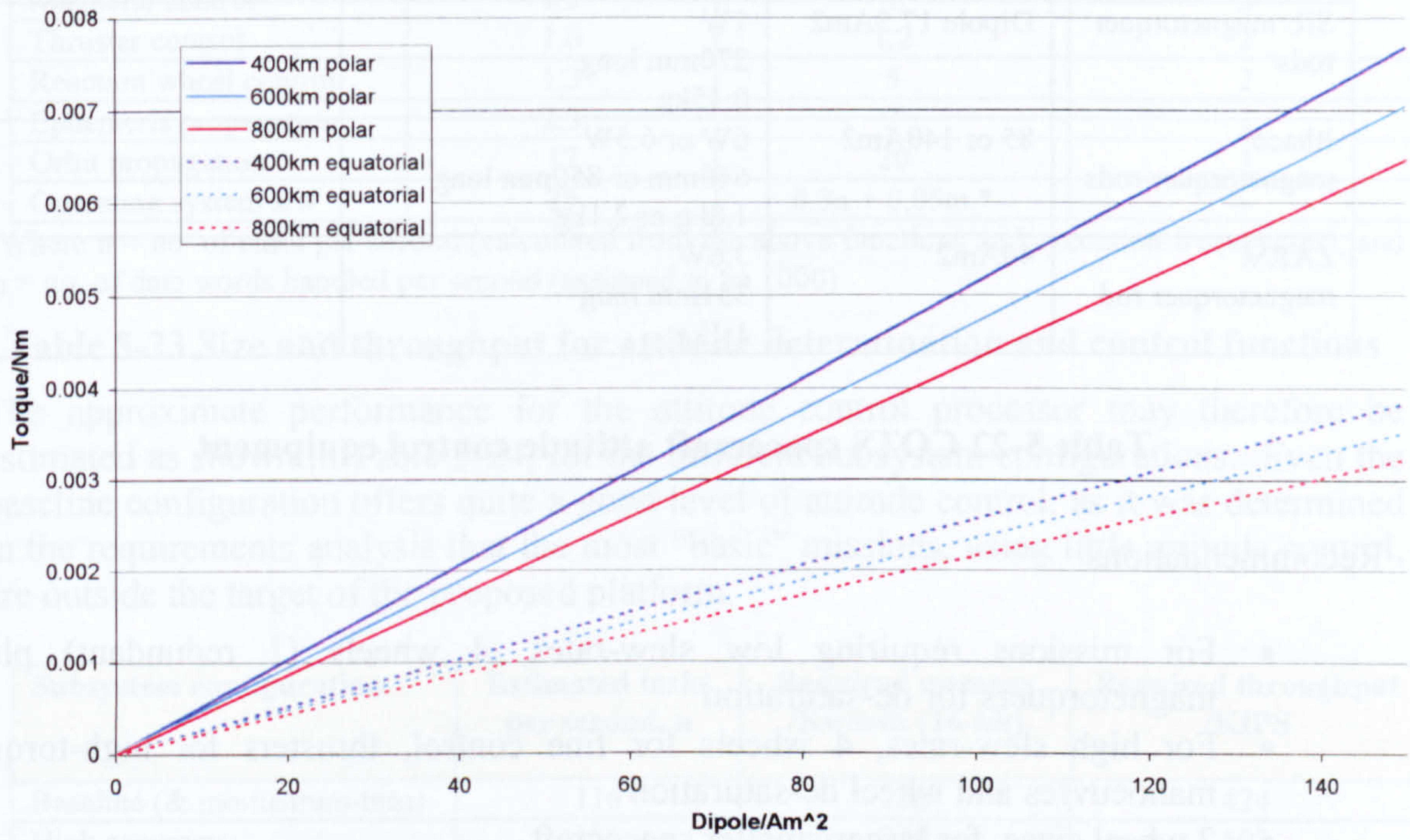


Figure 5-18 Torque produced vs magnetorquer dipole moment, for different low Earth orbits

Magnetorquers produce lower torque levels, but are simple and relatively cheap. They can also be used to de-saturate wheels. Typically, a 1kg magnetorquer rod will give a magnetic dipole of around 40-50Am^2, a 2kg rod a dipole of around 100Am^2.

Where wheels are used, with no thrusters, magnetorquers will be required to enable periodic wheel de-saturation. Characteristics of some commercial attitude control equipment are given in Table 5-22. (Note that thrusters are considered in the following section on propulsion).

Equipment	Performance	Specifications	Remarks
Honeywell HR 0610 Reaction Wheel	6Nms 0.075Nm	80W peak power 120mm ht. x 267mm diam. 4kg	Integral electronics Rad hard, 7.5 year lifetime
Teldix reaction wheel	2Nms 0.15Nm	85mm ht. X 222mm diam. 4.2kg	
Teldix RSI 01-5 reaction wheel	0.04Nms 5-10mNm	6W peak power 102mm ht. X 95mm diam. 0.6kg	5 year lifetime in orbit
Space Sciences Corp. momentum wheel	20Nms 0.335Nm	8W 4.53kg 114mm ht. X 343mm diam.	
SIL magnetorquer rods	Dipole 17.5Am2	1W 270mm long 0.45kg	
Ithaco magnetorquer rods	85 or 140Am2	6W or 6.5W 640mm or 850mm long 1.8kg or 2.1kg	
ZARM magnetorquer rod	40Am2	3.6W 351mm long 1.1kg	

Table 5-22 COTS spacecraft attitude control equipment

Recommendations:

- For missions requiring low slew-rates, 4 wheels (1 redundant) plus magnetorquers for de-saturation
- For high slew-rates, 4 wheels for fine control, thrusters for high-torque manoeuvres and wheel de-saturation
- 2 wheel sizes, for larger/smaller spacecraft
- For spin-stabilisation, thrusters for spin-up and precession control
- Momentum-bias system using single wheel

Processing

The software required to perform processing of sensor data, and attitude determination and control functions, may be estimated using the size and throughput estimates given in Table 5-23, on the following page.[8]

Function	Size (code+data) /Kwords (16-bit)	Typical throughput /KIPS	Typical execution frequency /Hz
Gyro processing	1.3	9	10
Sun sensor processing	1.5	1	1
Earth sensor processing	2.3	12	10
Magnetometer processing	0.3	1	2
Star tracker processing	17	2	0.01
Kinematic integration	2.2	15	10
Error determination	1.1	12	10
Precession control	4.8	30	10
Magnetic control	1.2	1	2
Thruster control	1.0	1.2	2
Reaction wheel control	1.3	5	2
Ephemeris propagation	2.3	2	1
Orbit propagation	17	20	1
Operating system s/w	25	0.3n + 0.05m *	-

*Where n = no. of tasks per second (calculated from the above functions and execution frequencies), and m = no. of data words handled per second (assumed to be 1000)

Table 5-23 Size and throughput for attitude determination and control functions

The approximate performance for the attitude control processor may therefore be estimated as shown in Table 5-24, for the different subsystem configurations. Even the baseline configuration offers quite a good level of attitude control, as it was determined in the requirements analysis that the most "basic" missions, using little attitude control, are outside the target of the proposed platform.

Subsystem configuration	Estimated tasks per second, n	Required memory /Kwords (16-bit)	Required throughput /KIPS
Baseline (& momentum-bias)	116	156	424
High accuracy	156	208	502
High manoeuvrability	188	232	552
Spin-stabilised	108	160	432

Table 5-24 Estimated attitude control computer requirements for the different subsystem configurations

Assumptions:

- 50% margin due to rough estimates used
- 50% spare computer capacity allowance

The table indicates that there is not an enormous variation in the performance required for the different subsystem configurations, implying that it would be reasonable to use a standard processor system, capable of supporting any of the different configurations. This would be more cost-effective, as it would minimise the number of redesigns between missions.

Software modules could be implemented separately, to support the different configurations, as not all will be needed for all missions (for example, thruster control, star tracker control etc). The required memory and throughput is easily compatible with most COTS spacecraft computer boards.

The above further implies that, as a standard attitude processor will be used, there is no advantage to housing it separately from the main platform computer, as it will not need to be substituted in response to changes to the attitude control subsystem. The attitude control electronics will therefore form part of the standard data handling enclosure. If necessary, a separate sensor/actuator controller could be used to handle interfaces with different numbers and types of attitude sensors and actuators.

Attitude control equipment list and mass budget

An equipment list, and estimated mass budget, for the different configurations of the attitude control subsystem is given in Table 5-25, on the following page.

Configuration variant	Equipment list	Mass, size estimates	Remarks
Baseline	4 x wheels 4 x magnetorquers 3-axis magnetometer 6 x sunsensors Earth horizon sensor Processing electronics	4kg, 270mm diam. x 120mm, 80W per wheel (large option) 1kg, 100mm diam. x 100mm, 15W per wheel (small option) 1kg, 350mm, 4W each (large) 0.5kg, 270mm, 2W each (small) 0.3kg, 130x90x36mm, <<1W 0.1kg, 75x90x25mm, each (0W) 1kg, 200x120mm, 3W (3-telescope) *	*Electronics considered as part of onboard data handling subsystem
High accuracy	4 x wheels 4 x magnetorquers 3-axis magnetometer 6 x sunsensors Star tracker Inertial reference unit Processing electronics	As above " " " 1.8kg, 3W, 144x144x288mm 5.4kg, 180x260mm, 10W (high acc. option) 0.75kg, 90x90mm, 12W	
High manoeuvrability	3 x dual paired thrusters Propellant tank + propellant 4 x wheels 6 x sunsensors Earth horizon sensor or star tracker* Inertial reference unit	0.1-0.2kg each pair Estimate 8kg (highly mission-dependent)	See propulsion section for more details on thrusters/propellant etc *Dependent on required accuracy
Spin stabilised	3 x dual paired thrusters Propellant tank + propellant 6 x sunsensors Passive nutation damper	 Estimate 1kg	As above
Momentum-bias	Single wheel 4 x magnetorquers 3-axis magnetometer 6 x sunsensors Earth horizon sensor or star tracker*	4.5kg, 10W, 343mm diam. x 114mm	Larger wheel than for reaction control *Dependent on required accuracy

Table 5-25 Attitude determination and control subsystem equipment list

The equipment list estimates shown in the table are based on the available information on state-of-the-art COTS units. This list allows an approximate mass and power budget to be produced for the different attitude control subsystem variants. This is shown in Table 5-26. There are obviously other implications in changing between the different configuration variants, such as the greater complexity of software development for the higher accuracy variant.

Note that when estimating the power budget, it is assumed that maximum torque demand would never be required from all wheels or rods at the same time, therefore a maximum anticipated power demand is estimated as all wheels/rods at half maximum power level.

Configuration variant	Subsystem mass & peak power estimates*
Baseline	22kg (8kg), 175W (43W)
High accuracy	23.5kg (8.5kg), 196W (62W)
High manoeuvrability	28kg (16kg), 188W (58W)
Spin stabilised	10kg, <5W
Momentum bias	11.5kg, 34W

*Lower mass/power option in brackets – for smaller mission configurations, using smaller wheels/rods

Table 5-26 Attitude determination and control subsystem mass and power budgets

Costs

Estimates for ROM costs of attitude determination and control equipment (references/sources as before) are given as follows:

- Wheels $50-200k
- Sun sensors $5-20k
- Star tracker $200-400k
- Earth horizon sensor $100-400k
- Magnetorquer rods $5-20k
- IRU $150-500k
- Typical attitude control subsystem total cost $0.5-3m

Interfaces and module boundaries

Referring to the modularity analysis performed in Chapter 2, the internal interface positions are 5, 7 & 8 in Figure 2-9, with a shared access to the cross-discipline data storage module. This gives the following as modular units:

- Processing – the attitude control processor is a separate board within the main data handling unit. Within this processor, the different functions are carried out with modular software modules, dedicated to the different equipment items used for a particular mission.
- Measurement – each sensor is considered separately, as attitude sensors often need to be distributed across the whole platform, therefore cannot really be contained within a single module. However, each sensor used should have a standard data interface so that they may be interchanged without the need to make changes to the rest of the system.
- Control – sets of wheels and rods can be considered as a "functional module"; however, to allow for spacecraft configuration to be varied, it is recommended that they be considered, and attached, separately. As above, standard data interfaces should be used, to allow actuators to be easily interchanged.

5.3.2.2 Orbit determination

Orbit determination has historically been achieved by means of ranging conducted by the ground station. This provides the instantaneous orbital elements, which can then be propagated either on the ground, or onboard the spacecraft. The disadvantage of this method is that it can only be performed during ground passes. Any autonomous navigation therefore depends on the accuracy of the onboard orbit propagator. However, the use of an onboard GPS receiver now allows spacecraft to determine their orbital position to within a few tens of metres, at any time during the mission.

It has already been recommended that the transponder used should support ranging, as a back-up orbit determination method. Primary orbit determination will, however, be via GPS. A COTS unit will typically have a mass of less than 1kg for the receiver and four antennas (such a system can also then be used for attitude determination if required). Power requirement is under 7 Watts. This equipment will be used on all mission configurations.

5.3.2.3 Propulsion

Propulsion system can be made as a separate self-contained module, although if it is to be used for attitude as well as orbit control, then thrusters may have to be positioned at distributed points around the platform. In this event, the pipework will also require routing around the platform to the thruster positions. The propellant tanks will generally require mounting centrally within the spacecraft, to avoid centre of mass migration as the propellant is used up over the mission life.

Propulsion equipment characteristics and performance are shown in Table 5-27.

System	**Performance**	**Specifications**	**Remarks**
Laben ion thruster	2-12mN thrust 3000s Isp	Max power 500W 24kg dry mass without tank Sample propellant mass 16kg inc. tank	Sample mass based on drag compensation for 500kg satellite at 300km for 3 years, plus deorbit[10]
Spectrum Astro Xenon Ion propulsion system	20mN at 500W 100mN at 2.3kW 2200s Isp at 500W 3100s Isp at 2.3kW	41kg system mass	COTS system Likely to be too high power for this application
Nitrogen cold gas system	Typical 60s Isp 0.05-200N thrust	~8kg system mass + propellant	Tanks form about a third of the system dry weight, due to pressurisation. Based on SSTL cold gas system
TRW hydrazine thrusters	0.09-0.22N thrust ~200s Isp	0.1-0.2kg	Mass per thruster (not whole system)
Marquardt hydrazine thrusters	2.2N thrust ~200s Isp	0.1-0.2kg	As above
Bipropellant (MMH/MON) system based on LEROS-20 engine	20N thrust ~290s Isp	9kg	Based on system developed for UoSAT-12[12]

Low-power NO resistojet	125mN thrust 127s Isp	1.24kg 100-600W	SSTL unit, qualified on UoSAT-12. See chapter 2
Pulsed plasma thruster	~2000s Isp (v low thrust)	3-4kg	Not yet COTS. See Chapter 2

Table 5-27 Propulsion subsystems for small spacecraft

The mass of the propulsion system is very dependent on the mass of the propellant required for the particular mission. This is itself dependent on the type of system used, and the total delta-v requirement of the mission. Figure 5-19 shows the mass of propellant required to achieve a given delta-v, for a range of propulsion system options. Some example mission delta-v requirements are also indicated.

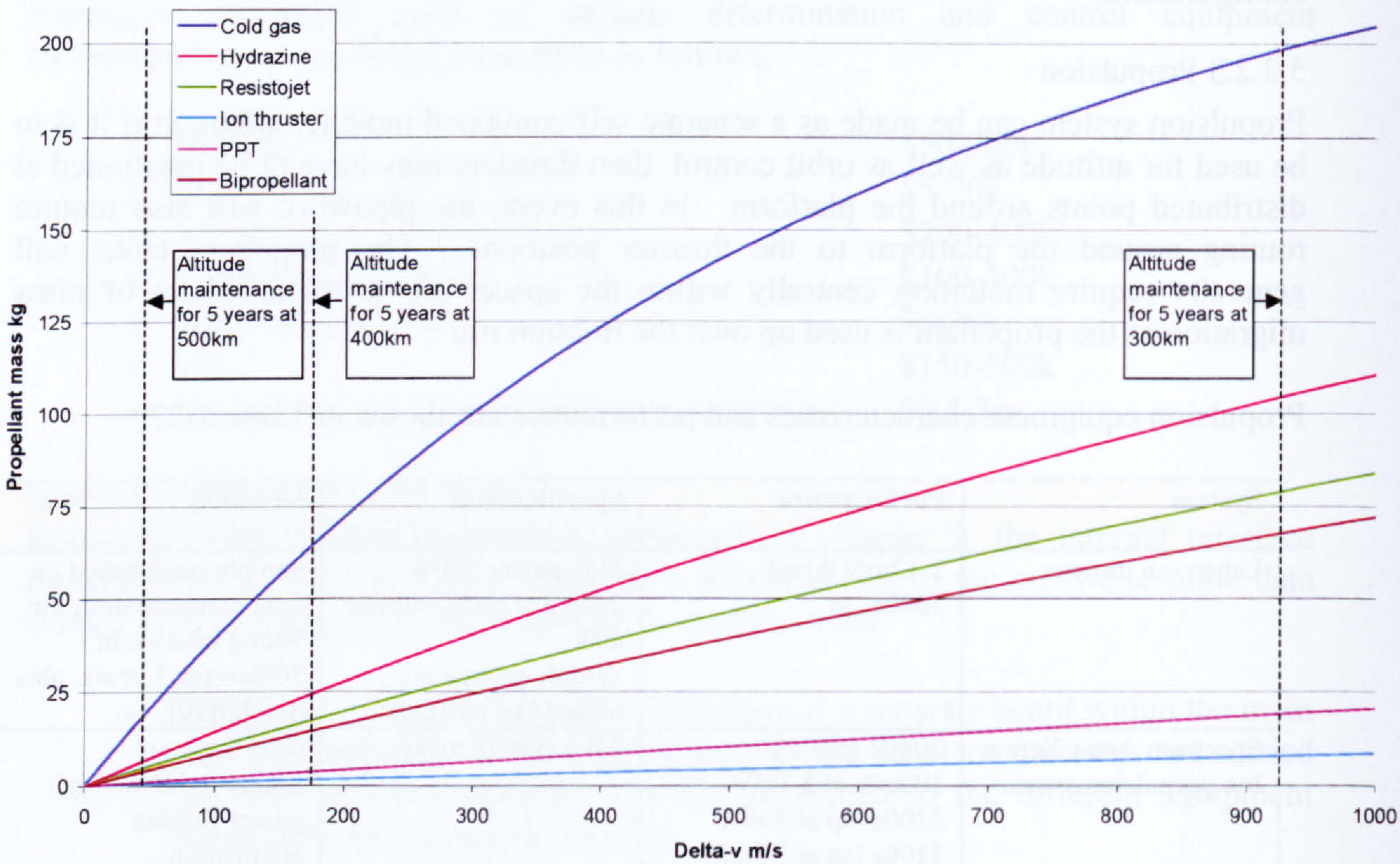
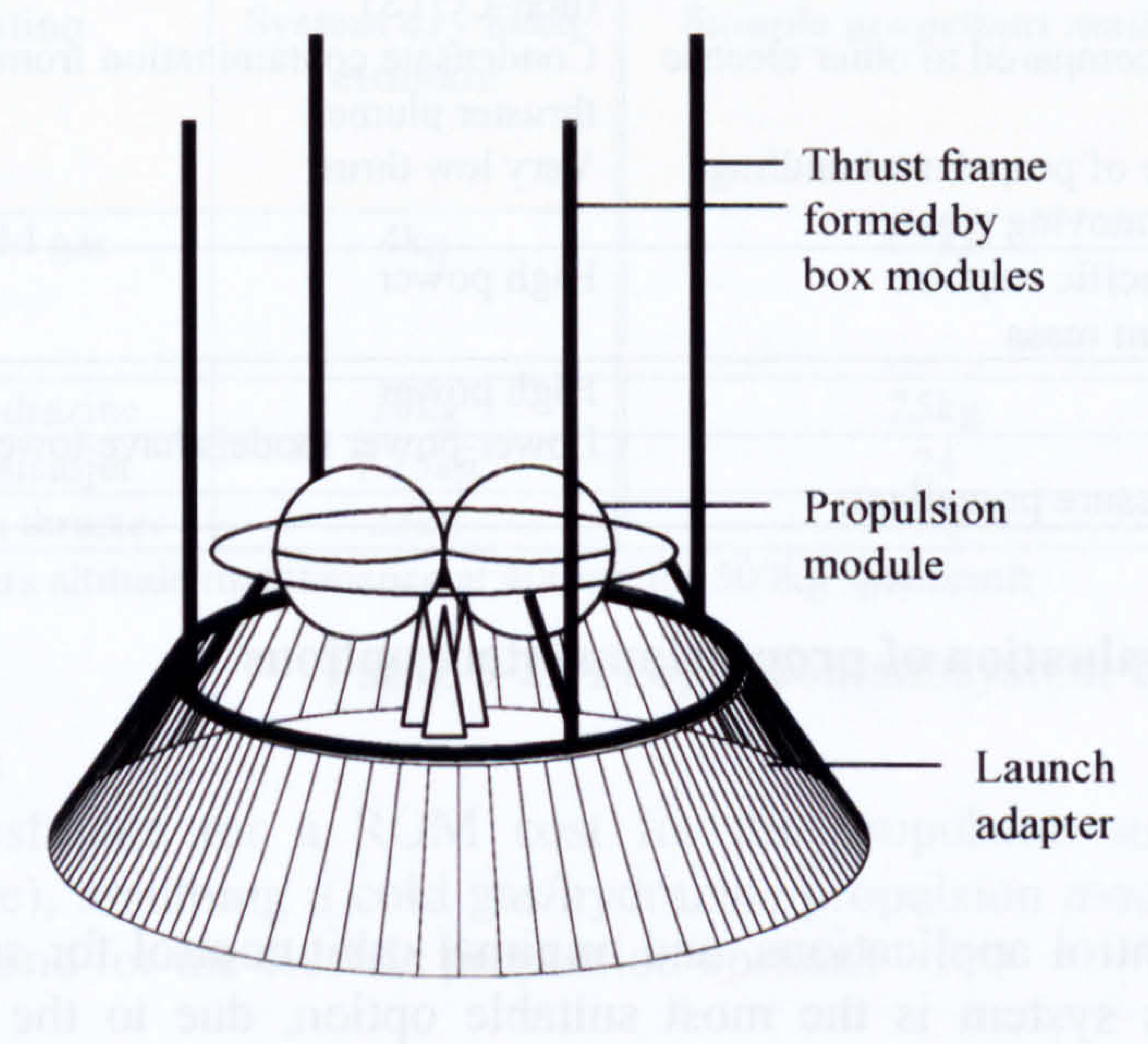


Figure 5-19 Propellant mass required for mission total delta-v, using different types of propulsion system

Assumptions:

- 250kg spacecraft dry mass
- I_{sp} values used: cold gas (60s), hydrazine (175s), resistojet (250s), ion (3000s), PPT (1200s), bipropellant (300s)
- Ballistic coefficient used: 200kg/m^2
- Mission during solar maximum

It is recommended that the propulsion system be considered as a separate, self-contained module, as it will only be required for some target missions. The most suitable position for the propulsion module is at the base of the spacecraft, within the central void formed by the module boxes. This allows the central volume to be used for the various sizes of tank, and central positioning of these tanks means that the centre of mass of the spacecraft will suffer minimal disturbance as the propellant is depleted over time. The scheme for positioning of the propulsion module is shown in Figure 5-20.



The propulsion module is positioned in the central void, within the thrust frame formed by the platform modules.

The main propulsion module load is taken by the launch adapter, rather than through the thrust frame. Heavy tanks can therefore be supported without the need to reinforce the main structure.

Various sizes of tank can be accommodated. For attitude control, propellant can be easily piped through to the distributed thrusters.

Figure 5-20 Positioning of propulsion module

For orbit control, a single thruster or cluster of thrusters directed along the longitudinal axis of the spacecraft will be used. This can be contained within the main propulsion module. For attitude control, propellant must be supplied to paired thrusters distributed around the platform; thus requiring the propulsion subsystem to be integrated with the platform to a greater degree. However, the central void enables this to be achieved more easily, as propellant pipework can be routed through each module and then down through the centre for connection with the propulsion module at the appropriate stage in integration.

Propulsion system selection

The advantages and disadvantages of the different propulsion options considered are shown in Table 5-28.

Option	Advantages	Disadvantages
Cold gas	Simple Lower parts count Safe propellants Small impulse bits Low cost Less contamination risk from thruster plumes	Pressurised tanks (heavy) Limited performance High mass relative to performance

Hydrazine	Simple Low cost Well-understood and widely used Higher thrust than cold gas Can be used for attitude and/or orbit control	Performance and mass efficiency lower than biprop system
Bipropellant	Higher performance	Higher complexity
PPT	High specific impulse Low mass Low power (compared to other electric propulsion) Safety & ease of propellant handling No valves or moving parts	Small units only just being developed (non-COTS) Condensate contamination from thruster plume Very low thrust
Ion thruster	Very high specific impulse Low propellant mass	High power
Resistojet	Low cost Low mass Safe, low pressure propellants	High power Lower-power models have lower Isp

Table 5-28 Evaluation of propulsion system options

Recommendations:

- For most attitude control applications, and minimal orbit control for smaller missions, a cold gas system is the most suitable option, due to the small impulse bit and lower risk of thruster-plume contamination.
- For high-thrust orbit control, a hydrazine or bipropellant engine is suitable. The components for these systems are quite readily available commercially, and can be supplied configured as a "black box" module that is interfaced with the rest of the spacecraft. Hydrazine is recommended as a baseline, but may be substituted by a bipropellant system as they become more qualified for small spacecraft applications. This system can also be used to supply attitude control thrusters.
- Where high thrust is not required, and/or mass and volume are limited, a low-power resistojet or small ion thruster is suitable. Although this option requires high power, for initial orbit acquisition it is not likely to be operating at the same time as the payload, making power draw less critical. It will also be used most intensively at the beginning of the mission, when power raising capability is likely to be at its highest. Ion thrusters are most appropriate where a high delta-v is required (as long as a long transfer is acceptable – i.e. probably not for short missions).

The propulsion module is very difficult to "pre-design", as requirements vary so much between mission profiles. Therefore, the best compromise is probably to establish a supply-chain relationship with manufacturers of each type of system, and liaise with them during the detailed platform design phase to ensure that propulsion modules provided by them can be easily interfaced with the main platform.

Subsystem mass estimates

A propulsion system mass estimate can be made based on information on commercially-available equipment, and adding additional mass for the support structure. For missions requiring large propellant loads, additional mass will be required for the larger tankage necessary. Mass estimates for the different propulsion options are given in Table 5-29.

Option	System dry mass estimate	Sample propellant mass*	Total for sample mission, inc. ~10% margin for module support structure
Cold gas	4kg	65kg	69kg, but not meaningful – would not be used for this application
Hydrazine	10kg	25kg	39kg
Resistojet	1.25kg	24	28kg
Ion thruster	25kg	1.5	30kg

*5 years altitude maintenance at 400km for 500kg spacecraft

Table 5-29 Propulsion subsystem mass budgets

Costs

An estimate for a ROM cost for the propulsion subsystem (references/sources as before), assuming a cold gas/hydrazine propulsion module, is $0.5-2m. No data could be found for the electric propulsion options.

5.3.3 POWER

The key requirements derived in Chapter 4 for the power subsystem are as follows:

- Power level should be offered in capability increments
- The lowest acceptable platform power level (peak) is around 100W
- For high-power missions, platform power in excess of 500W would be desirable

The three main areas that must be considered for the power subsystem are generation, storage, and distribution. These are considered in turn, and solution options offered.

Power generation

Power generation on board Earth-orbiting satellites almost always uses solar cells, and other methods such as fuel cells and RTGs will not be considered here due to their costs, complexities, and safety/environmental issues. If solar cells are to be used, there are still a number of different options for the specific method of power-raising. These are discussed in Table 5-30.

	Advantages	Disadvantages	Remarks
Cell type			
Silicon	Easily available Lower cost Well-developed and tested	Lower efficiency (15%)	Longest-established technology
Amorphous thin-film silicon	Low mass Packaging efficiency	Lower efficiency (5-9%)	
Gallium-Arsenide	Higher efficiency (18.5%) Available	Higher cost (3 times that of silicon)	
Indium Phosphide	High efficiency (18%) Greater radiation tolerance	Higher cost (as for GaAs)	
Multijunction	Highest efficiency (22%)	Newer technology (therefore immature, expensive)	Technology is being matured on a number of missions
Array type			
Body-mounted	Simple	Limited area	See Figure 5-21 for limitations on power-raising
Deployed-fixed	Fairly simple Improved array area Back of solar panel can better radiate heat	Fixed position of array may not give ideal sun incidence angle at all times	Good for SSO missions with fixed sun angle
Deployed-articulated	Sun-tracking optimises sun incidence angle and therefore power	Cost and complexity Mechanisms may introduce vibrations	
Concentrator	Increased cell power output	Higher cell temperatures may result, reducing efficiency. Also recent problems with out-gassing on Boeing 702 platform.	Can be used in combination with the previous array types
Inflatable omni-directional	Simple Does not require pointing Small stowed volume	Relatively untried new technology	See details in Chapter 2

Table 5-30 Power generation options using solar cells

For the platform, the actual power-raising method used is of less importance than its ability to be easily tailored to a particular mission-specific power level. Therefore, a phased, modular approach to the system design is necessary, with the technology selection giving the best power for each platform variant.

For the lowest-capability variant, it makes sense to use the simplest option – body-mounted arrays. These will give a fairly low performance, but will also provide a basic minimum-cost option for the budget end of the market. An idea of the maximum possible power-raising that may be achieved by body-mounted arrays is given by Figure 5-21. This shows the power that could be raised by a cubic spacecraft with its outer surface fully populated with GaAs or Si solar cells. The true operational value that could be achieved would obviously be far less than this, due to incomplete coverage of the spacecraft with cells, and non-nominal illumination conditions.

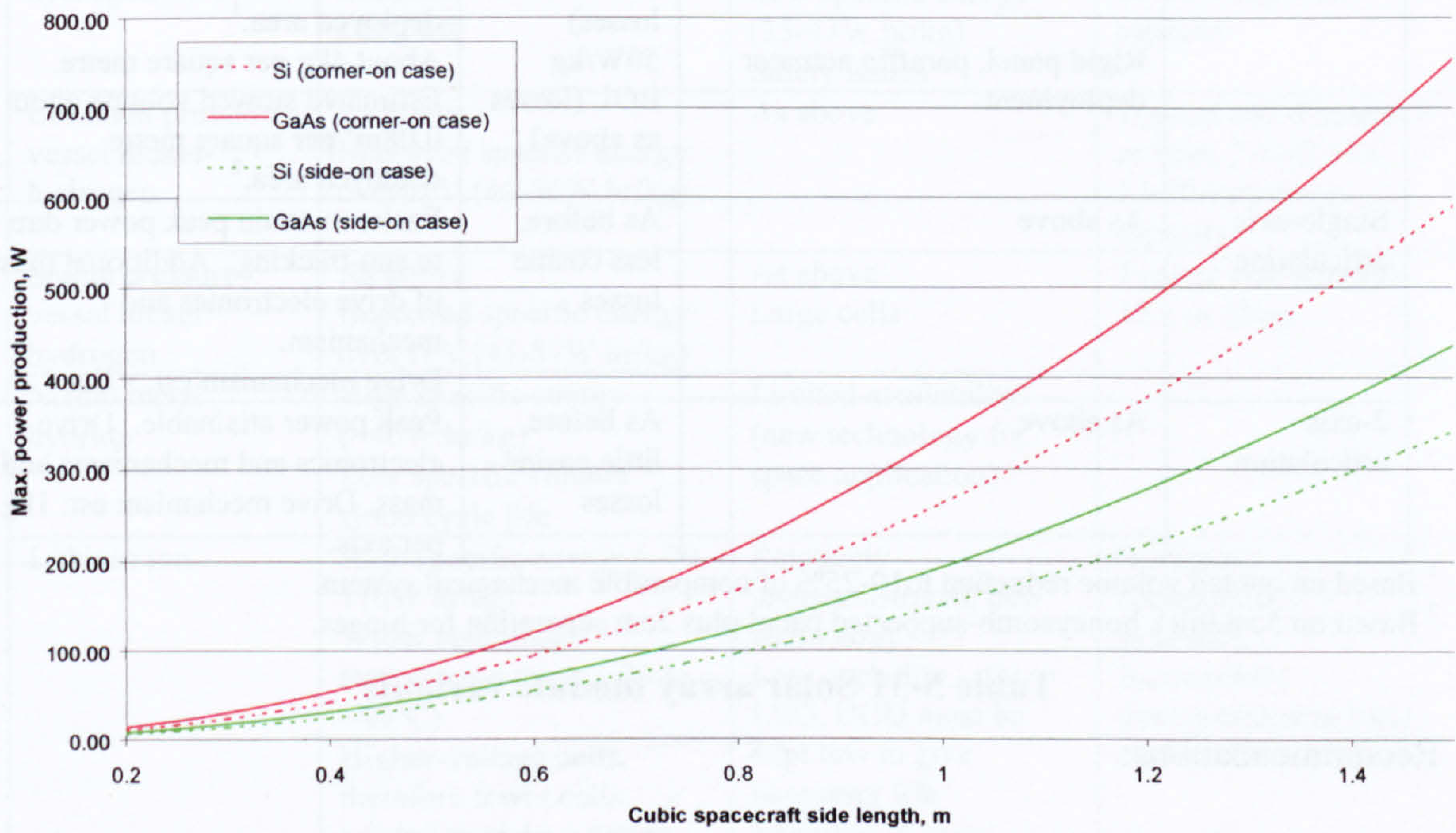


Figure 5-21 Maximum power production from ideal body-mounted arrays, with spacecraft size

For the higher-power platform variants, various degrees of array deployment will be required. The inflatably-deployed flexible arrays examined in Chapter 2 are ideal candidates, as they may be stowed into relatively small volumes, and these volumes may also be of more flexible geometries than those needed for deployable rigid panels. Modularity may be employed by offering different size options of panel, and the option to use one or two panels. The simplest of the deployed variants would be a deployed-fixed array, with the options going on to cover single-axis and two-axis sun-tracking variants.

Another, simpler, way to boost the power-raising of a lower-cost platform variant would be to include a Powersphere™ inflatable omni-directional array. This technology was discussed in Chapter 2.

The estimated parameters for the different deployed array modules are given in Table 5-31. These show both conventional technology and the inflatably-deployed arrays discussed in Chapter 2. Offering either of these as an option may be advisable as some customers may be averse to using such new technology. The rigid panels are assumed to be populated with GaAs cells; the flexible arrays use thin-film silicon.

Variant	Description	Peak power generation	Remarks
Powersphere™	Inflatable sphere covered with thin-film solar cells.	5W from 0.6kg mass	Estimated volume <0.01m^3
Deployed-fixed	Inflatably-deployed thin-film array	100W/kg BOL (but cosine losses)	About 1kg per square metre deployed. Estimated stowed volume 0.02m^3 per square metre deployed area.[1]
	Rigid panel, paraffin actuator deployment	50W/kg BOL (losses as above)	About 4kg per square metre. Estimated stowed volume about 0.08m^3 per square metre deployed area.[2]
Single-axis articulation	As above	As before, less cosine losses	Easier to attain peak power due to sun-tracking. Additional mass of drive electronics and mechanism. Drive mechanism est. <1kg
2-axis articulation	As above	As before, little cosine losses	Peak power attainable. Drive electronics and mechanisms add mass. Drive mechanism est. 1kg per axis.

[1] Based on quoted volume reduction to10-25% of comparable mechanical system.
[2] Based on 5cm thick honeycomb-supported panel plus 2cm separation for hinges.

Table 5-31 Solar array module variants

Recommendations:

- Offer a basic body-mounted panel array as the budget option. This could be enhanced by the use of body-mounted concentrator arrays (as discussed in Chapter 2) at a slightly greater cost.
- Offer deployed-fixed arrays in standard modular increments in either traditional or inflatable technology options. A reasonable "module unit" for the rigid panel array type would be 0.5m^2. For the flexible arrays, which are likely to be larger, but have a lower power generation per unit area, larger area increments, e.g. 1m^2, would be more useful. (The same inflatable deployment mechanism could be used for the different sizes of array – just extended to the required length.) These would give maximum power-raising capability increments of approximately 100W (BOL) for both types of array.
- Offer the above arrays (with the same modular size increments) with either one- or two-axis articulation. This would require an additional drive mechanism for each axis of articulation, and would also involve additional design and analysis work (and hence additional costs on top of the cost of the extra mechanisms)

Power storage

New technologies for power storage were discussed in Chapter 2. These, together with the more established options, are summarised in Table 5-32.

Battery type	Advantages	Disadvantages	Remarks
Nickel-Cadmium	Well-demonstrated Widely commercially available	Low specific energy (25-30W.hr/gk) Narrow temperature range (-5 to +25°C)	Cells typically 5-100Amp-hour capacity
Individual pressure vessel nickel-hydrogen	Well-demonstrated Widely commercially available Long life	Narrow temperature range (-5 to +25°C) Low specific energy (35-43W.hr/kg) Safety issues	Typical cell diameter 9-12cm, 1.22-1.25V, 20-300Amp-hour capacity
Common pressure vessel nickel-hydrogen	As above Improved specific energy over IPV (40-56W.hr/kg)	As above	Typical cell diameter 6 or 9cm, 2.44-2.50V, 12-20Amp-hour capacity
Single pressure vessel nickel hydrogen	As above Improved specific energy over IPV (43-57W.hr/kg)	As above Large cells	Typical cell diameter 12.5 or 25cm
Nickel metal hydride	High specific energy (~40W.hr/kg) Low specific volume Good cycle life	Limited availability (new technology for space application)	
Lithium ion	High specific energy (~70-110W.hr/kg) Wider operating temperature range (-20 to +40°C) Higher-voltage cells, therefore fewer cells needed to make a battery Low self-discharge	Relatively undemonstrated, new technology Low cycle life – for LEO, DOD must be kept low to give necessary life Sensitive to over-charging	Becoming commercially available. Successfully demonstrated in LEO on PROBA.
Lithium polymer	Very high specific energy (~200W.hr/kg) Very flexible configuration – can be made in sheets or variety of tailored shapes.	Untried in space	Should be considered for potential future inclusion on the platform, once technology has matured.

Table 5-32 Power storage options

As this is to be a new platform, which is intended to form the basis of an on-going commercial programme, it does not seem sensible to base subsystems around older, less capable technology. The lithium-ion technology offers clear benefits in terms of mass, size, and thermal constraints, and is the subject of concerted development effort. As recommended in Chapter 2, it will be selected for use. The lithium polymer battery is also of great interest. The modular approach used for the platform design should enable this technology to be inserted in future, once it is sufficiently mature.

Specifications of some commercially available Lithium-ion cells are given in Table 5-33.

Manufacturer	Details	Remarks
Sony	70-80 W.hr/kg	Commercial cells, tested at JPL for space application
SAFT	5 Amp-hour cell, 17x65x60mm	Tested to 10,000 cycles at 20% DOD
Yardney	5 Amp-hour cell, 3.7V, 0.16kg	Developed for military/ aerospace applications
MAXELL	1.35 Amp-hour cell, 3.6V, 0.04kg, 18mm dia. x 65mm height	Designed for terrestrial use, but probably also suitable for space applications

Table 5-33 Commercially-available lithium-ion cells

From the requirements section, it was determined that in LEO the battery must supply power for up to 36 minutes during eclipse periods. For payload power increments of 100W, this would equate to a power capacity requirement from the battery of 60W.hr for each increment. Using lithium-ion technology, and a depth of discharge of 20% (a greater DOD may be acceptable for shorter missions, due to the reduced number of discharge cycles), this then gives a figure of 300W.hr battery capacity for each 100W payload power increment.

Assuming a 28V power bus, (which is the most common for space equipment, and therefore likely to be the best choice for a multipurpose platform), this equates to a just over 10Amp-hours battery capacity to give 100W payload power through the worst possible eclipse case. From the manufacturers' details given previously, assuming 3.6V per cell, a string of 8 cells connected in parallel would give the required 28V. Based on a cell mass of 0.16kg per cell, this would give a battery mass of 1.28kg (excluding housings and interconnects). Assuming an energy density of 80W.hr/kg, this would give a battery capacity of approximately 100W.hr.
To give the desired 10Amp-hour capacity, at least 300Watt-hours are required at 28V. Therefore, it may be assumed that three of the parallel 8-cell strings, connected in series to form a battery module of 24 cells, should provide approximately the 10Amp-hour capacity desired.

From the manufacturers' details previously give, the size of this battery may be estimated:

- Mass of 24 cells at 0.16kg per cell = 3.84kg
- Mass including extra to account for battery housing and interconnects = 4kg
- Size of battery 150x200x90mm (cell dimensions plus estimated margin for mounting, container, and interconnects)

[Note: compare a 10Amp-hour battery module using Nickel-Cadmium cells: 8kg, 230x185x120mm (SIL equipment).]

Platform subsystem power requirements must also be allowed for. If a slightly greater depth of discharge was allowed, or the eclipse was shorter, then there should be sufficient energy stored in one battery module to both service the platform subsystems, and still offer 100W to the payload. This could require some operational trades. However, one battery module should be sufficient for both platform and payload for the most basic missions. The low baseline depth of discharge also provides a reasonable margin, especially for shorter missions (and the lowest-cost missions are likely to be shorter).

Recommendations:

- Baseline minimum performance variant carries 1 of the 10 Amp-hour battery modules. This will allow at least 100W to be drawn during eclipse. It will also allow for additional peak power draw above that which can be provided by the solar arrays. This may be useful for specific operations.
- Higher-power variants can then be produced by the addition of one or more extra battery modules, each providing an extra 100W to power the payload during eclipse/peak-power operations.

Power distribution

The power distribution system must be capable of regulating, controlling, and distributing electrical power on board the spacecraft. Additionally, for the multipurpose platform, it must also be capable of accommodating different configurations of power storage, solar cells, and different numbers of switchable and hard-wired loads. A functional block diagram of the power subsystem, showing the functions performed by the control and distribution system, was given in Chapter 2. Essentially, this system must take electrical power from the solar arrays, then route it to those units requiring it, via commandable switches. It must also use some of the power to charge the batteries, and switch in battery power during eclipse, and periods of high power demand. Finally, it must dump excess power.

The "modular increments" that the power distribution may have are therefore driven by the variability in:

- Number of switchable loads (in turn driven by number of platform equipment items and payload units)
- Power sources (i.e. number of solar cells, and the amount of power coming from them)
- Power storage (i.e. number and configuration of the batteries used)
- Amount of excess power to be dumped

The number of loads may be estimated from a list of platform equipment plus an additional number for allocation to the payload. Where there are multiple loads within a subsystem (for example in the attitude control subsystem), the number of loads to be switched by the power control system may be reduced by moving the load switching to the subsystem. In this method, the attitude control subsystem would then be responsible for controlling switching on or off equipment such as reaction wheels, or magnetorquers. The power control system would simply provide a power line to the attitude control subsystem. This has the added benefit of simplifying the power harness from the power subsystem.) This essentially distributes the power control around the spacecraft. It also allows specific power conversion to be performed for particular units as required (for example, voltage conversion, regulation, levelling and isolation from bus noise), removing the need for a regulated bus.

Using this method, an estimate for the number of switched loads may be made as shown in Table 5-34.

Load	Remarks
Attitude and orbit subsystem	May need a separate switched line to the orbit control subsystem, if propulsion is used.
Data handling subsystem	Main processor may be hard-wired, not switched, for safety
Deployment mechanism(s)	If present
Thermal control subsystem	If active heaters are used
Solar array drive mechanism(s)	If present
Communications subsystem	Receiver generally hard-wired, not switched, for safety. Transmitter could be switched from the data handling subsystem?
Payloads	10 instruments specified in the data-handling requirements, therefore assume this number of switchable loads is possible

Table 5-34 Switched loads onboard a generic spacecraft

Even using this scheme, it can be seen the number of loads is highly variable. As switching boards can be quite large and bulky, it would make sense to employ modularity in the configuration of the power control system, to allow interchangeable "switching modules", capable of switching different numbers of loads. For example, a module capable of handling 10 switched loads could handle a fairly simple platform with only one or two payload instruments. Then, for a more complex configuration, two modules could be used – one to handle payload switching, the other to handle platform switching. This would also give freedom to decide whether to centralise all the switching in the power control system, or whether to distribute the function around the spacecraft on a subsystem-by-subsystem basis.

Load-failure protection, detection, and isolation could also be provided at these distributed control modules. This would allow more sophisticated modules to be inserted in place of the basic ones if necessary.

The method used to control power provided by the solar array must also be considered. Two options are available: Peak Power Tracking (PPT) and Direct Energy Transfer (DET). The PPT system uses a DC-DC converter in series with the array, and extracts the exact spacecraft power requirement, up to the peak power point. It uses 4-7% of the total power, but allows very good power extraction from the array at BOL. This gives a greater bias towards power availability at BOL than EOL.

The DET system is more efficient; any excess power is dumped in shunt resistors, and, as they operate in parallel with the array, they use little power. This system gives a more even efficiency from BOL to EOL, and is simpler and lower mass.

Size and mass of the power control system is quite difficult to estimate. A commercial system offering a fully-regulated 28V bus, redundant shunt regulators (i.e. a DET system), short-circuit protection, and user-configurable switched and non-switched loads had the following characteristics: (SIL equipment)

- Mass 3.5kg
- Dimensions 290x290x50mm (volume 0.0042m^3)

Although this is a different configuration to that suggested for the platform, it may be used as a baseline for estimating the size of the power control unit. As the proposed system uses some distributed control, it may be estimated that the main unit would be reduced in size from this baseline, and additional smaller switching control modules would be required around the other subsystems, and for the payload. (The payload switching module could be housed as part of the power control system, or as part of the payload control/data handling subsystem). An estimate for the power control system modules is made in Table 5-35, assuming that total volume of the subsystem will be slightly increased (the penalty for enhancing modularity).

Module	Details
Power control main unit (battery charge/discharge control, shunt regulation, primary load switching, primary load fault detection & isolation)	2.5kg, volume 0.003m^3
Payload switching module (5 switched loads)	0.25kg, volume 0.0005m^3
Payload switching module (10 switched loads)	0.4kg, volume 0.0008m^3
Subsystem switching modules	As for payload switching modules (no. of loads subsystem-dependent)

Table 5-35 Power control system modules

Recommendations

- Unregulated 28V bus
- DET system (for simplicity and efficiency)
- Individual battery charging (as Li-ion batteries sensitive to over-charging. Also, gives flexibility in number of batteries used)
- Distributed switching and conversion for the subsystems (different subsystem configurations can then be interchanged without need to alter the power control system), with different fault isolation/detection options
- Interchangeable switching modules for the payload, handling different numbers of switched loads. Also with different fault isolation/protection options

Interfaces and module boundaries

The interface positions identified in the modularity analysis in Chapter 2 that are most suitable are 2, 4 and 5 in Figure 2-12. This gives the following as the separate module units:

- Control and distribution – this is in fact further broken down into control and switching, to allow for different complexities of switched loads.
- Storage – the battery cells are formed as modular units, which may be duplicated as required for a particular mission. (Note – if and when polymer battery technology has matured for space use, storage and generation can then be combined into a single modular unit, with the battery forming a thin panel on the reverse of the solar array.)

- Generation – the solar arrays are formed from modular units, which again may be built up into the required size and complexity for a particular mission

The power subsystem equipment for the different variants is shown in Table 5-36.

Variant	Equipment list	Mass/size estimates	Remarks
		Solar arrays	
Basic body-mounted	Cells mounted on honeycomb panels	Dependent on platform configuration	Populate all free outer surfaces with cells. Cell dimensions usually 2x2cm, 2x4cm, or 5x5cm
Deployed-fixed flexible – per ~100W	Flexible array membrane, $1m^2$ Deployment mechanism (inflatable rigidizable tube)	$0.02m^3$ stowed 1kg inc. deployment mechanism	For applications where stowing array presents problems
Deployed-fixed rigid – per ~100W	Array panel, $0.5m^2$ Deployment mechanism (hinge + lock-out)	$0.04m^3$ stowed, 2kg inc. deployment mechanism	Rigid panel array deployment probably easier than flexible array for small areas, more difficult for large areas
Deployed-articulated	As for deployed-fixed plus: drive mechanism	<1kg 100x100x150mm estimated	Per axis of articulation Use with either array type
		Batteries	
Basic	10Amp-hour battery module	4kg, 150x200x90mm	
Higher power	Additional battery modules as required	As above	Each module gives an additional 10Amp-hour capacity
		Power control & distribution	
Basic – 1 payload instrument	Main power control unit Subsystems switching units for AOCS, DHS. Power harness	2.5kg, $0.003m^3$ 0.25kg, $0.0005m^3$ each 3kg	Payload switched from main power control unit Single line to AOCS, DHS
Basic – up to 5 payload instruments	As above, plus: Small payload switching module	0.25kg, $0.0005m^3$	
Basic – up to 10 payload instruments	As basic, plus large payload switching module	0.4kg, $0.0008m^3$	
Complex – up to 5 payload instruments	As basic, plus: Small payload switching module Additional subsystem switching module	As details above	For switching of additional mechanisms/equipment
Complex – up to 10 payloads	As basic, plus: Large payload switching module Additional subsystem switching module	As details above	

Table 5-36 Power subsystem equipment list

Notes:

- The performance quoted is at BOL; degradation is 2.75% per year for GaAs cells and 3.75% per year for Si cells.
- These mass and size estimates are intended to err on the side of over-estimation, therefore they may be taken to include mass margin, and some flexibility in terms of size.

As all the different variants are modular and interchangeable, a subsystem may be built up to tailored requirements regarding power level, power storage level, and switching and regulation.

Mass budget

When increasing the platform power capability, it may reasonably be assumed that one battery module would be added every time a solar array module is added, as both give operational power increments of about 100W. (The actual power provided by the arrays obviously may be less than this figure if they are non-articulated, due to cosine losses when sun-angle geometry is unfavourable). Assuming then that these two options are linked, power subsystem mass budgets may be offered for a number of platform variants. These are shown in Table 5-37. Other configurations would be possible with the modular options proposed; these budgets merely give an idea of the mass range.

Platform variant	Subsystem mass inc. harness /kg	Remarks
Basic	12kg	Body-mounted arrays (assume 1.5m^2) 1 payload 1 battery module
Basic w. deployed-fixed panel arrays	18.25kg	2 array modules (rigid panel) 1 battery module <5 payloads
Basic high power deployed-fixed flex-arrays	31.5kg	Using 5m^2 flexible arrays 5 battery modules <5 payloads
Complex high power articulated panel	~33kg	Using 5m^2 flexible arrays Single axis articulation (use yaw-steering if necessary) 10 payloads

Table 5-37 Estimated power subsystem mass budgets for a range of different performance variants

There is obviously greater complexity involved in articulating arrays in one or two axes, and there will be implications involved in increasing the number of modular array elements (e.g. precise method of deployment for the rigid panel arrays, stowed configuration of the flexible arrays). However, this has given an idea of the equipment masses and sizes that must be accommodated on board. It has also offered an architecture that gives the necessary modularity to enable a range of different platform variants to be produced, and to enable future technology improvements to be easily inserted into the system.

Costs
ROM cost estimates for power subsystem equipment (based on sources/references as before and in Chapter 2), are given as follows:

- Solar arrays — $100k-$5m (depending on size, construction, articulation)
- Batteries — $100-200k
- Power control — $50-100k
- Typical power subsystem total cost — $0.8-6m

7.1.1 THERMAL CONTROL SUBSYSTEM

Spacecraft thermal control may be achieved either passively or with an active control system. Passive control involves the use of surface finishes specially chosen for their absorptivity and emissivity characteristics, insulating blankets, passive louvre radiators, and passive heat pipes. These methods are simpler, lower cost, and high reliability.

Active control methods include heaters (with control of varying sophistication), pumped-loop heat pipes, active louvre radiators, and active coolers (Peltier and Stirling cycle). These methods introduce more complexity, have higher risk of failure, require power and control systems and are higher cost, but they give a greater degree of temperature regulation. They are typically used for spacecraft that will experience very challenging thermal environments, such as interplanetary probes, or that carry instruments with very demanding temperature requirements.

The thermal balance of a spacecraft in orbit depends on the incident radiation flux, the onboard energy dissipation of the spacecraft systems, and the absorptivity/emissivity properties of the spacecraft. In Earth orbit, incident radiation arises from direct solar illumination, solar radiation reflected by the Earth (Earth albedo), and Earth infrared radiation. Thermal balance is described by the expression

$$A_{sc}\varepsilon\sigma T^4 = \alpha(A_sJ_s + F_aA_eJ_a) + \varepsilon F_eA_eJ_e + Q_{total}$$

where:

A_{sc}	= Total spacecraft area
α	= Solar absorptance
σ	= Boltzmann constant (5.67×10^{-8} $Wm^{-2}K^{-4}$)
A_s	= Area normal to the sun
J_s	= Solar flux (1481 Wm^{-2})
F_a	= View factor of Earth albedo on spacecraft
A_e	= Area normal to the Earth
J_a	= Earth albedo flux (0.34 x Solar flux)
ε	= Infrared emittance
F_e	= View factor for Earth thermal radiation
J_e	= Earth thermal flux
Q_{total}	= Total spacecraft power dissipation
T	= Spacecraft surface temperature

Using the above energy balance equation, the equilibrium temperature of the spacecraft can be approximated for different conditions and surface properties. For this approximation, the following assumptions are made:

- Spacecraft is isothermal, and cubic
- Two sizes are used, of side areas $0.5m^2$ and $1m^2$ (same power dissipation)
- A hot case condition is considered to occur when one spacecraft face is in full sunlight, and one face is Earth-pointing
- A cold case occurs during eclipse, when the spacecraft sees only the thermal radiation of the Earth, on one face
- Onboard power dissipation of 300W is assumed
- An orbital altitude of 500km is assumed, giving an Earth view factor of 0.86
- Surface properties taken to be as follows: white paint (α=0.20, ε=0.85), black paint (α=0.95, ε=0.85), solar cells (α=0.70, ε=0.80), OSR (α=0.10, ε=0.81), aluminium (α=0.38, ε=0.035)

(Note: OSR = Optical Solar Reflector, a second surface mirror produced by overlaying a highly reflective surface with a transparent cover that has a high IR emissivity. This type of reflector gives the lowest temperature in sunlight of any surface.)

The equilibrium temperatures for each case are shown in Table 5-38. (Note that equilibrium temperatures are different due to the spacecraft being different sizes but dissipating equal heat internally.)

Surface finish	Temperature /K			
	Small spacecraft		Large spacecraft	
	Hot case	Cold case	Hot case	Cold case
White paint	251	227	233	201
Black paint	308	227	298	201
Solar cells	296	230	285	203
OSR	243	230	220	202
Aluminium	579	475	543	400

Table 5-38 Equilibrium temperatures for different energy balance cases

In Chapter 4, the thermal ranges required for typical onboard equipment were identified. The strictest limits were found to be required by the batteries, at 0 to +20°C (although some propulsion equipment will tolerate temperatures only down to +7°). An onboard temperature of around 280K is therefore a suitable target.

The figures in the table indicate that passive thermal control is generally possible in LEO, by selection of appropriate surface finishes and multi-layer insulation to control energy absorption and radiation. However, the modular construction of the spacecraft makes the thermal properties and control requirements more complex.

To allow modules to be interchangeable and as independent as possible, it is recommended that they be thermally isolated from one another, via the use of multi-layer insulation (MLI) blankets and thermal isolation washers at the mechanical interfaces. Regulated heat flow between the modules can then be permitted as required using heat pipes, or by replacing isolation washers with conducting material. Simple, low cost, flexible heat pipes are commercially available and would be ideal for this purpose. These consist of an evacuated sealed tube that is partially filled with a working fluid. Local heat input is taken up by the fluid, which then changes to a vapour state. The resulting local increase in pressure forces vapour transport to the cooler parts of the pipe, where condensation dumps the heat to the colder region.

For mission profiles involving a constant attitude with respect to the sun (i.e. inertial attitudes in sun-synchronous orbits), there is likely to be one "hot" module. The module selected to be sun-facing should then be that which contains the equipment that is most tolerant of higher temperatures. Heat pipes from this module can be routed to the "cold" modules to provide heating, and to radiators to dump excess heat if required. Where the spacecraft has one nadir-pointing module, it may be best to make this the module containing the batteries. The Earth-pointing face of the spacecraft will see less temperature variation whilst having less risk of becoming too cold.

Mission configurations with deployed arrays are unlikely to have any difficulties in accommodating radiators on the exterior where required. Configurations with body-mounted solar cells will have limited exterior surface area for radiators; however, the cells typically have a high emissivity (around 0.80), so will effectively act as radiators when not sunlit. The key problem with body-mounted cells is that it is more difficult to transfer heat away from the sunlit cells (cell efficiency is reduced at higher temperature). Deployed arrays can easily radiate to space from the rear of the panel, maintaining the sun-facing cells at a lower operating temperature.

Recommendations:

- The platform will employ passive thermal control, with simple bimetallic strip-controlled heaters if required for particular missions
- Each module will be thermally isolated from the other modules as far as possible and considered separately, to make reconfiguration and separate testing easier
- Thermal links necessary between modules will be made using passive heat pipes and selected structural conduction paths
- Specific payload requirements for heating/cooling will be considered on a mission-by-mission basis, and any additional equipment necessary will be included as part of the payload
- A baseline mass budget of 2kg will be assigned to the baseline thermal control subsystem. This covers tapes, paints, an additional radiator panel, heat pipes and several simple heater elements.

Costs

A typical ROM cost estimate for a small satellite thermal control subsystem (based on mainly passive control), is $30-100k.[5],[6],[13]

5.4 STRUCTURAL DESIGN & EQUIPMENT ACCOMMODATION

The following sections address design and sizing of the chosen configuration concept for compatibility with the candidate launch vehicles. Accommodation of the platform subsystems is also covered, including recommendations for positioning of externally-mounted items and payload instruments.

5.4.1 STRUCTURAL DESIGN & SIZING

Chapter 4 identified candidate launch vehicles, and the spacecraft accommodation envelopes associated with them. Spacecraft envelopes were derived as follows:

- Pegasus-XL envelope 1100mm diameter circular
- Taurus envelope 1300mm diameter circular
- ASAP-5 (mini) envelope 1500mm diameter circular
- Large-fairing envelope 1900mm diameter circular

As mentioned in the previous chapter, the largest envelope is much more flexible, as the larger-fairing launchers have a greater diameter than the envelope proposed here. It is assumed that, due to the high cost and "over-capability" for a smaller spacecraft, a large launcher would be more likely to be used in special circumstances. These may be where a very high propellant load requires a greater lift capacity, or a specific payload requires a larger than normal envelope.

Based on the above envelope dimensions, and the configuration concept selected, a suitable module size and platform configurations could be chosen. The process involved iterative trials of different module dimensions, to achieve a configuration suitable for launch with any of the envelopes identified.

The module dimensions, and configurations within the launcher envelopes, are shown in Figure 5-22 and Figure 5-23. The baseline configuration consists of three modules in a triangular assembly. For the smallest launch envelope (Pegasus-XL), the platform is made "tall and narrow", by orientating each module such that the longest side is vertical.

In a Taurus launch envelope, the modules may be orientated such that the longest side is horizontal, giving a "short and wide" configuration. This gives a larger upper payload volume, and makes more efficient use of the available envelope. Alternatively, a four-module configuration can be fitted within the Taurus envelope, by again using the modules in the "tall" orientation. This gives a larger central volume within the platform, making it more suitable for a propulsion module or long payload instrument.

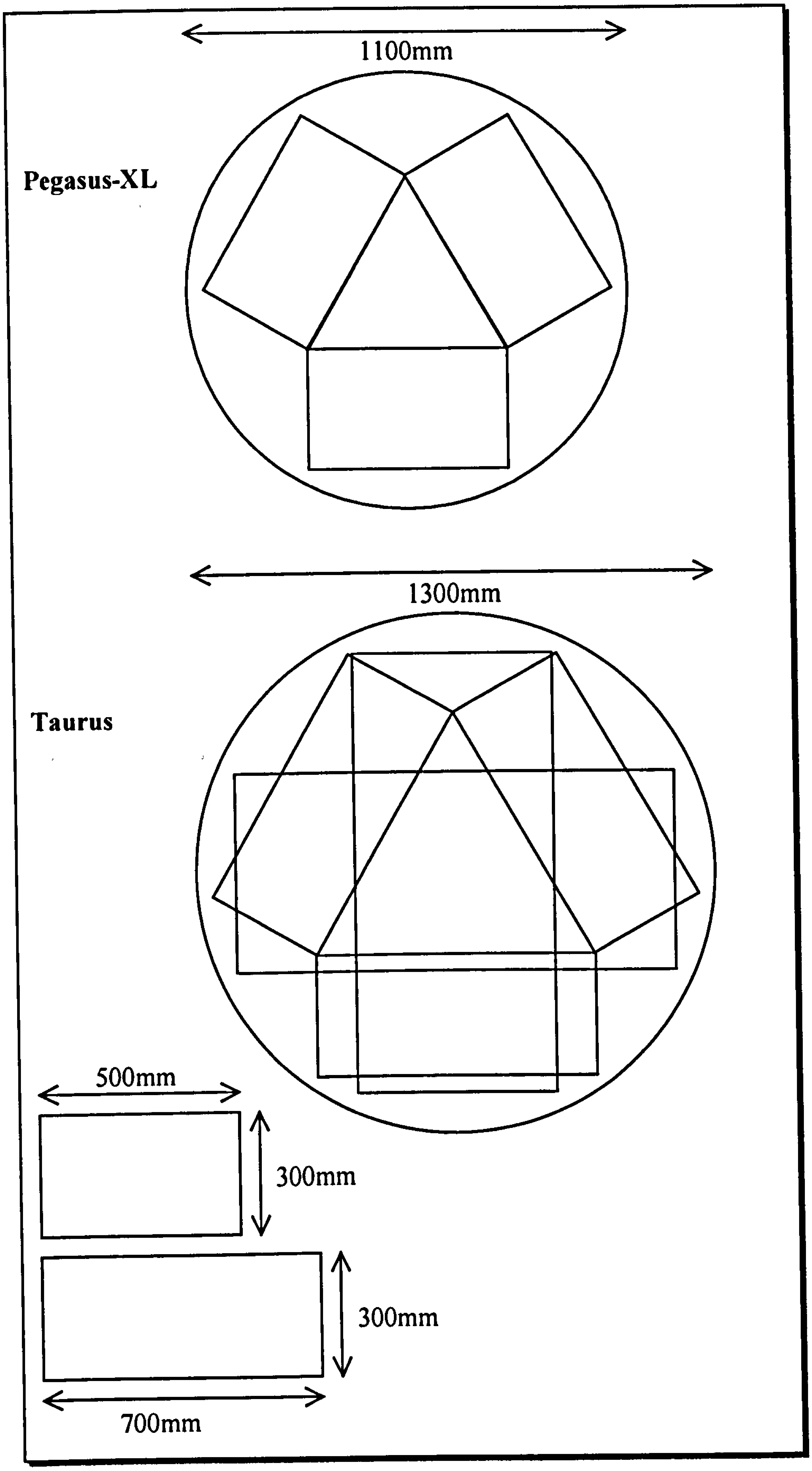
In the ASAP-5 minisatellite envelope, a four-module configuration in the wide module orientation can be accommodated. Obviously, the smaller configurations can also be accommodated within the Taurus/ASAP envelopes, and using a narrower platform would allow greater freedom to mount items on the exterior of the modules. (Note: the Pegasus-XL and Taurus envelopes used are slightly smaller than the actual allowable envelopes quoted in the User's Guides, thus allowing some additional margin in platform size).

For a very large configuration, launched on a wide-fairing launcher, a five-module configuration is also possible. This gives a very large volume for payload and additional platform equipment (e.g. fully redundant subsystems). However, this variant is considered unlikely to be within the usual scope of the platform.

The different configurations are summarised in Table 5-39.

Configuration	No. of modules	Orientation of modules	Launcher compatibility
Small narrow	3	Tall	Pegasus-XL, Taurus, ASAP-5, Larger launchers
Small wide	3	Short	Taurus, ASAP-5, Larger launchers
Large narrow	4	Tall	As above
Large wide	4	Short	ASAP-5, Larger launchers

Table 5-39 Summary of configuration options and launcher compatibility


1100mm
Pegasus-XL
1300mm
Taurus
500mm
300mm
300mm
700mm

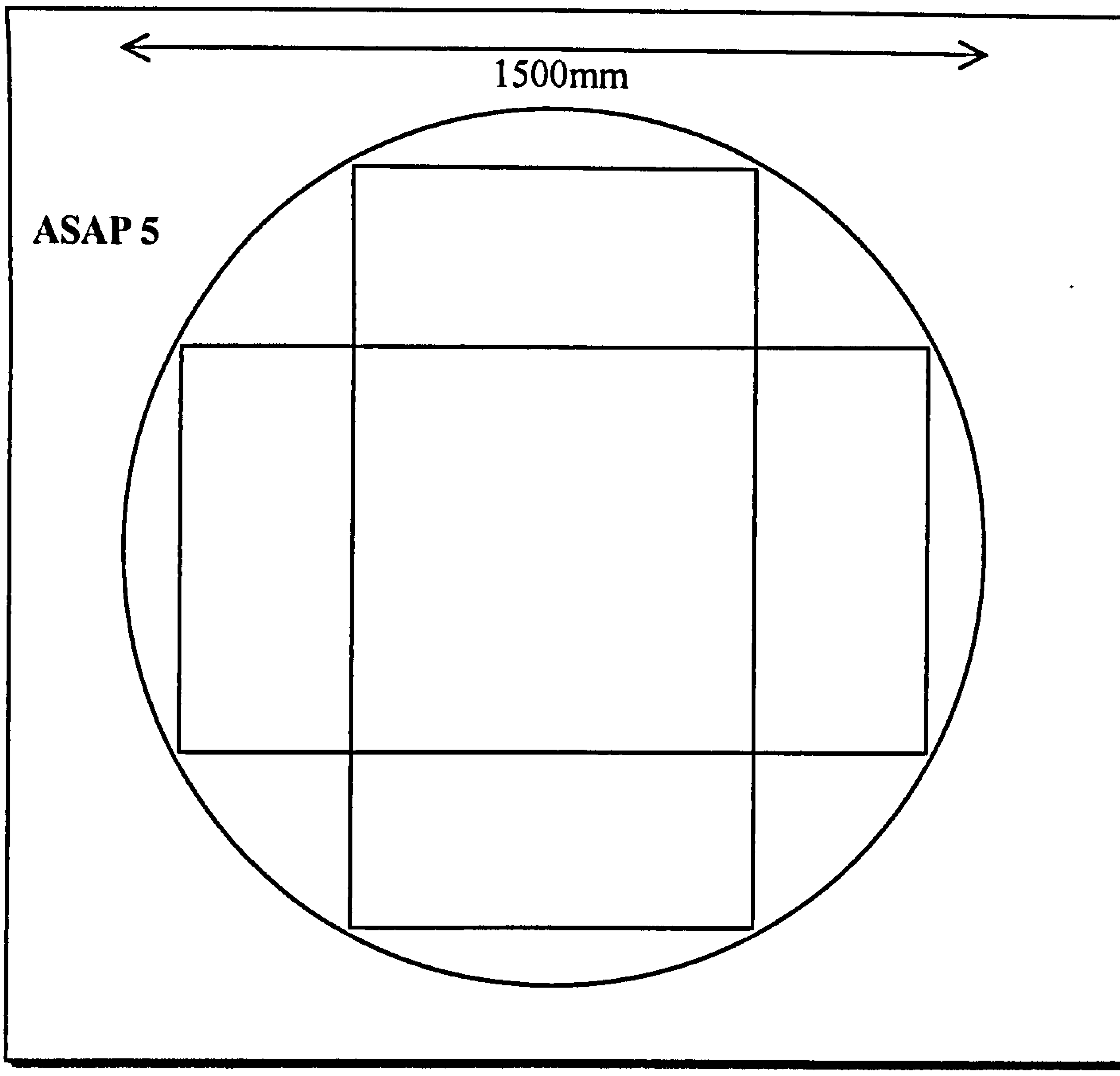


Figure 5-22 Module accommodation and configurations within different launcher fairing envelopes, including module dimensions

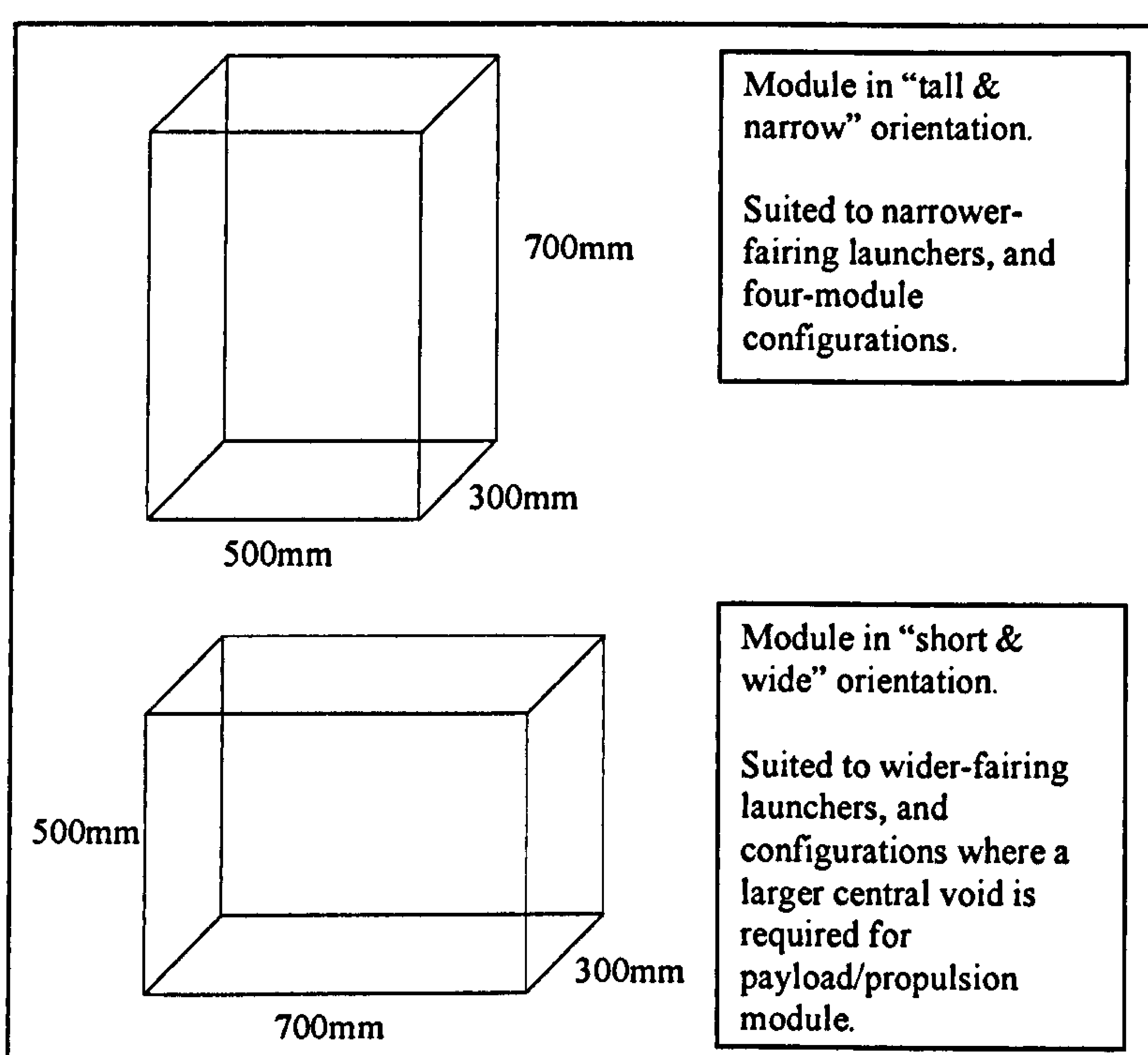


Figure 5-23 Module dimensions & orientation options

The module sizes specified are the external dimensions. As previously discussed in the configuration concept section, the modules can be formed in an open construction, such that equipment items may be permitted to protrude through the sides (where envelope constraints allow). The thickness of the module sides will depend on the material and construction type (e.g. frame-truss, solid panel, isogrid panel).

Equipment will be bolted to the side walls, and module bases and tops. Choice of construction for each panel (side/top/base) can be made depending on the particular requirements for equipment mounting. Modularity and interchangeability are enabled by using panels with the same strength and stiffness properties. Any panel can then be used in any position. This obviously only applies to the "secondary" structural panels; the inner, primary load-carrying panels are of a different, stronger, construction. However, a similar interchangeable scheme can be used between these panels.

To allow the platform to compete in the two payload size categories identified in Chapter 4, two different strengths of the load-bearing panels can be used. The stronger panels can be used for support of heavier payloads, or to allow a stacked configuration, while the lighter panels are suitable for smaller payloads and avoid the mass penalty otherwise incurred. Only the primary structural panels need to be replaced. This strategy minimises the changes between the "light" and "heavy" payload configurations, whilst reducing the wasted structural mass when a lighter payload is flown.

The inner void volume enclosed by the modules is available for payloads and/or the propulsion module. Instruments and electronics boxes can be bolted to the inner faces of the modules. However, this can make parallel integration more difficult, if payload equipment is being mounted by bolting to the interior of the panel. Parallel integration is made easier if payload is mounted to a shelf or shelves, which are then bolted into the centre of the platform. Long instruments, which must run the length of the central volume, can be supported via struts bolted to the sides, top, and base of the modules.

Further payload volume is available above the main platform and modules. Equipment can be mounted here by attachment to the top covers of the modules, or bolting onto the top of the reinforced side panels (for heavier instruments).

Finally, payload equipment can be accommodated inside one or more modules, if necessary. The volume available will depend on the platform configuration and capability variant used.

Sizing of the structural members constructing the modules will require detailed structural modelling and analysis. This forms part of the detailed design phase of the proposed platform programme, and is not covered here. However, a conservative mass estimate for the module of each structure can be made by analogy with previous small satellites.

The following assumptions are made:

- Each individual module can be considered as analogous to a "box-type" small satellite

- Each individual module is responsible for supporting its internally- and externally-mounted equipment, plus an equal share of the payload mass
- The platform equipment mass supported by each module is assumed to be up to 35kg
- The platform should aim to support a separate payload mass of 50kg (for the "light" payload option) or 125kg (for the "heavy" payload option) per module. This gives a payload mass capability of up to 150kg or 375kg for the three-module configuration, and 200kg or 500kg for the four-module configuration. (The four-module configuration can also support additional payload within the fourth module). This is consistent with the mass capability requirements identified in Chapter 4.
- A structure mass fraction per module is assumed to be 20% (this is based on the structural mass fractions of small spacecraft using an aluminium single box structure)

These assumptions result in the following estimates for the structure mass of each module:

- 21kg for the "light" payload option
- 40kg for the "heavy" payload option

This is a very conservative estimate, and the actual structure mass required is likely to be lower than these values, especially if composite materials are used.

5.4.2 MODULAR PARTITIONING & SUBSYSTEM ACCOMMODATION

The basic configuration identified in the previous section consists of three identical modules, which is expanded to four modules in the larger configurations. The baseline for accommodation of all the required platform equipment should therefore be using three modules only. This then leaves the fourth module in the larger configurations free for accommodation of payload instruments and electronics boxes, and for "overspill" of platform equipment if required.

To allocate equipment between modules, the following factors must be considered:

- The modules should be independently testable, as far as possible
- The modules should contain approximately equal mass
- Some equipment requires external mounting/aperture to the exterior

To enable independent testing of each module, it makes sense to allocate functions to single modules as far as possible. This makes subsystem functional testing easier, and allows greater levels of integration and testing to be done in parallel. Externally-mounted equipment, such as sun sensors and antennas, can be considered separately, as integration of these is more flexible.

Examination of the equipment list derived earlier in this chapter, and the equipment mounting requirements identified in Chapter 4, leads to the following proposal for equipment partitioning across the modules:

Attitude control module, containing:
- Wheels
- Magnetorquers
- Inertial Reference Unit

Communications and data handling module, containing:
- Data handling main unit
- Payload processing unit
- Transponder
- Amplifier
- Diplexer
- X-band adapter

Power module, containing:
- Power control unit
- Battery modules
- Switching units

This partitioning gives approximately equal mass distribution across the modules, depending on the exact equipment (i.e. capability variant) selected. The remaining equipment requires external mounting or particular field of view. This will be considered later in the section.

Accommodation of the subsystem equipment in the three modules is shown in Figure 5-24, Figure 5-25, and Figure 5-26.

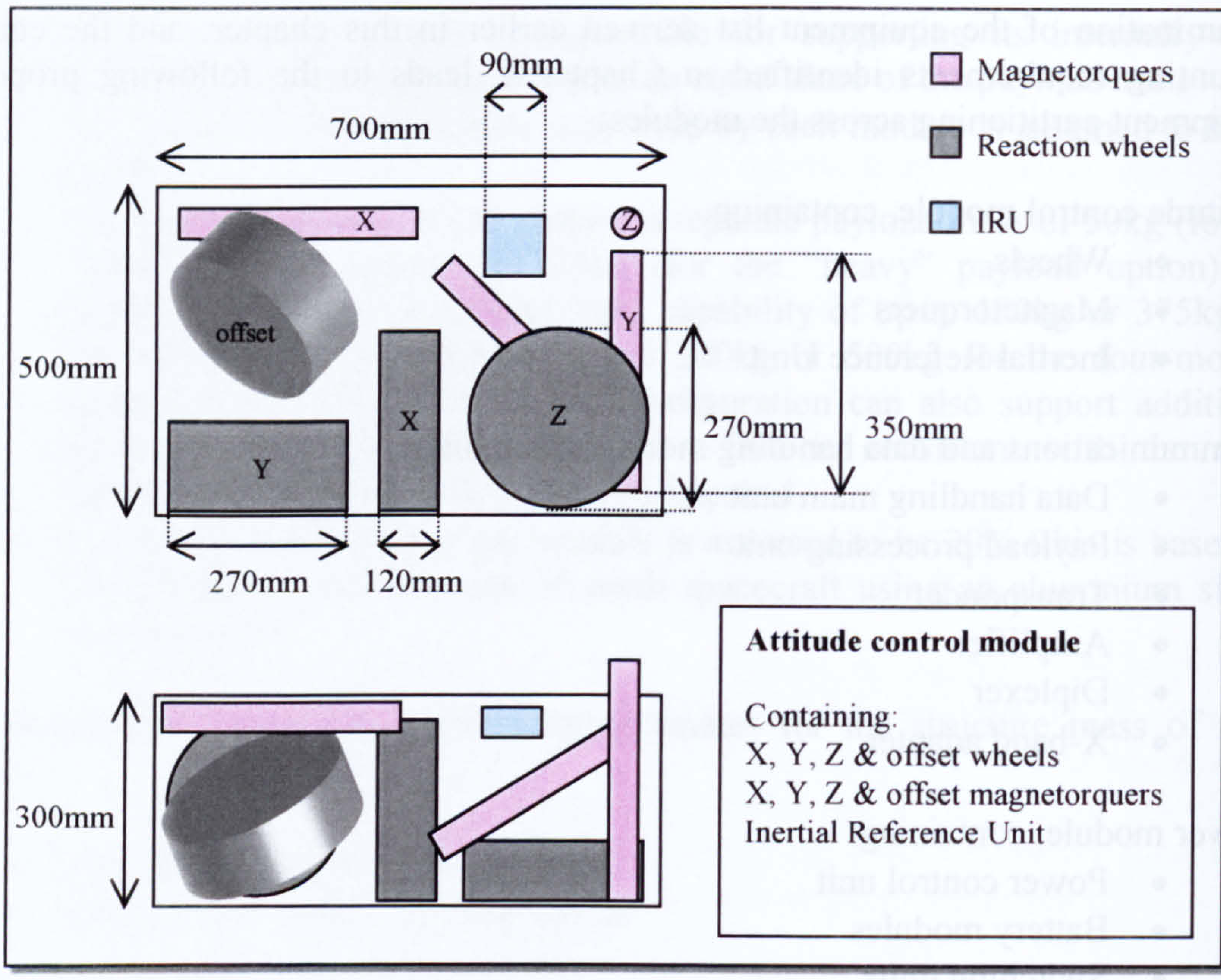


Figure 5-24 Equipment accommodation within the attitude control module. *The large sizes of magnetorquers and wheels are illustrated. The larger magnetorquers are longer than the shortest module dimension, implying that one rod must protrude from the side of the module. This will probably be acceptable; alternatively, it could be mounted in a different module, in the centre, or a different rod configuration could be used. All rods could be skewed to the spacecraft body axes; it is less important for the rods to align with the body axes, as it is orientation with respect to the ambient magnetic field that is more relevant.*

The offset wheel and rod will require mounting brackets for attachment. Other platform variants will use different equipment; the illustration shows the configuration with the maximum expected population of the module.

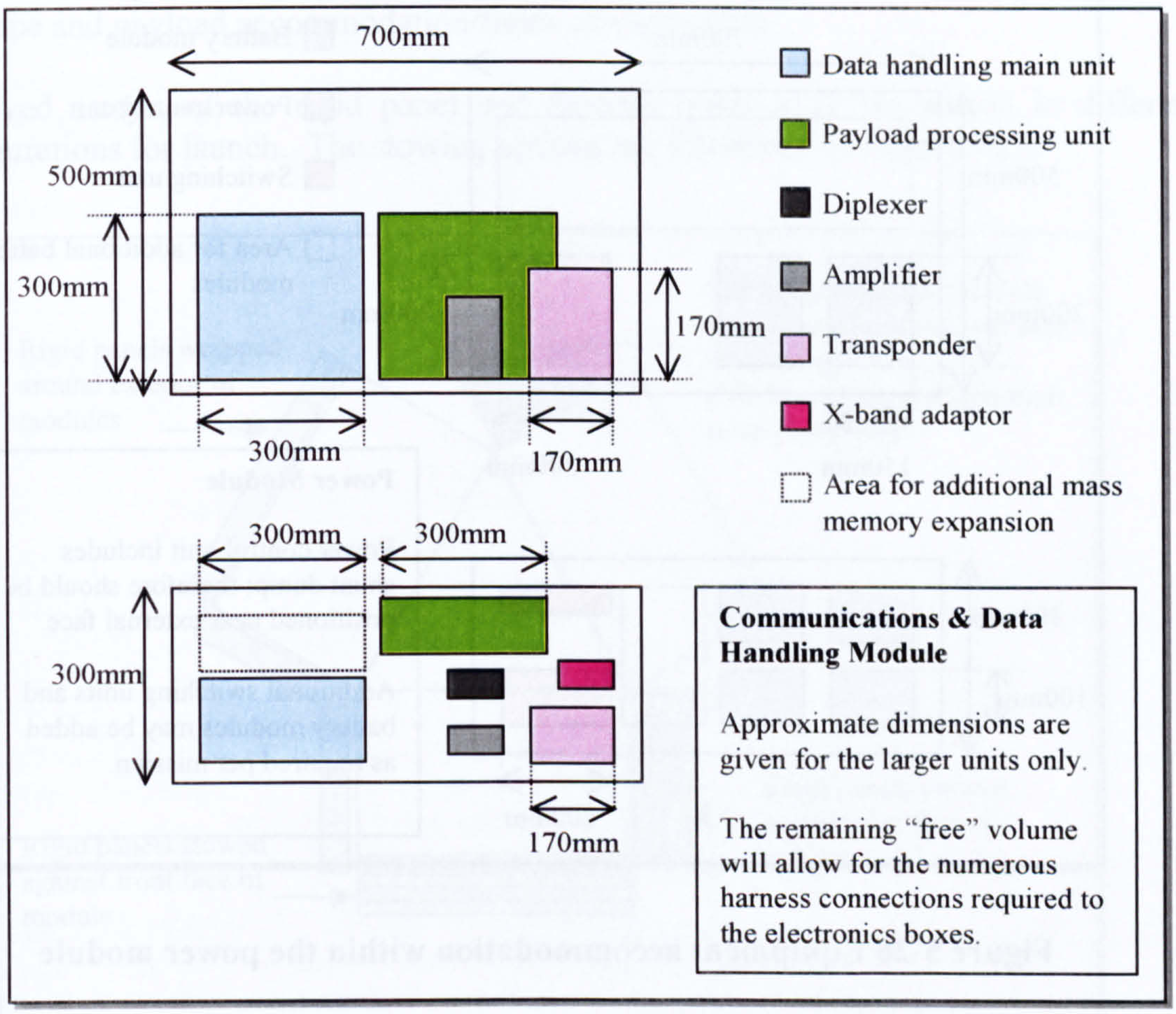


Figure 5-25 Equipment accommodation within the communications and data handling module. *The amplifier and diplexer may alternatively be positioned nearer to the antennas, on the exterior of the spacecraft.*

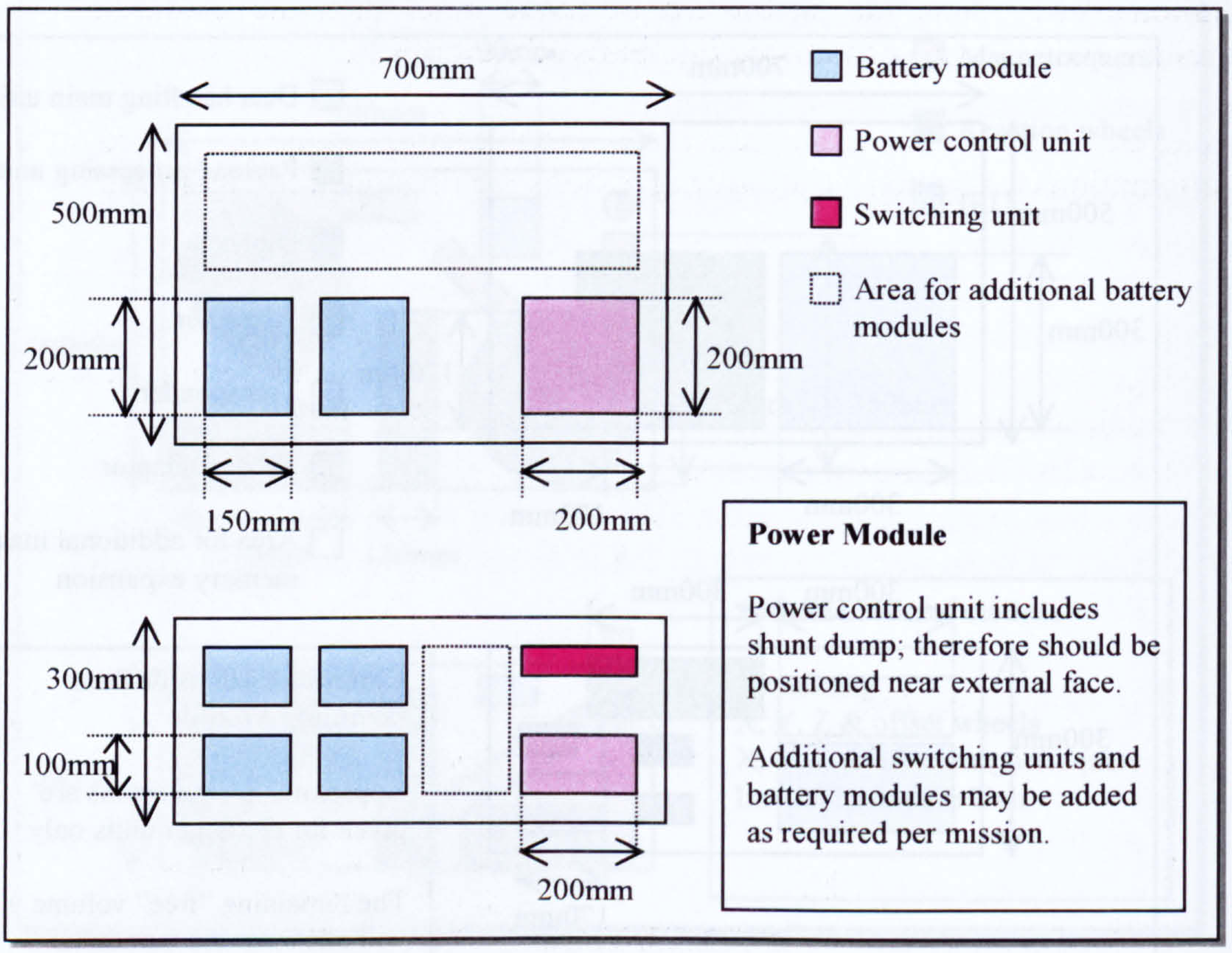


Figure 5-26 Equipment accommodation within the power module

There are other considerations for equipment mounting within modules, due to the option for turning the modules to the alternate orientation. Equipment boxes must be designed such that they can take either axial or lateral loading (i.e. they can be safely mounted in either orientation).

The other internal equipment – power and data harness, and thermal control equipment – is distributed through the modules as required. This is performed on a mission-specific basis.

The externally mounted equipment consists of the following:

- Solar arrays
- Communications antennas
- Sun sensors
- Magnetometer
- Earth sensor
- Star tracker

The solar arrays may be of various different types and configurations, as described earlier in this chapter. Body-mounted arrays will be simply attached to the outer faces of the modules, and additional panels to "close" the gaps between the modules. For additional array area, the panels can be made taller than the modules, if launcher envelope and payload accommodation/fields of view allow.

Deployed arrays, of the rigid panel and flexible types, may be stowed in different configurations for launch. The stowing options are illustrated in Figure 5-27.

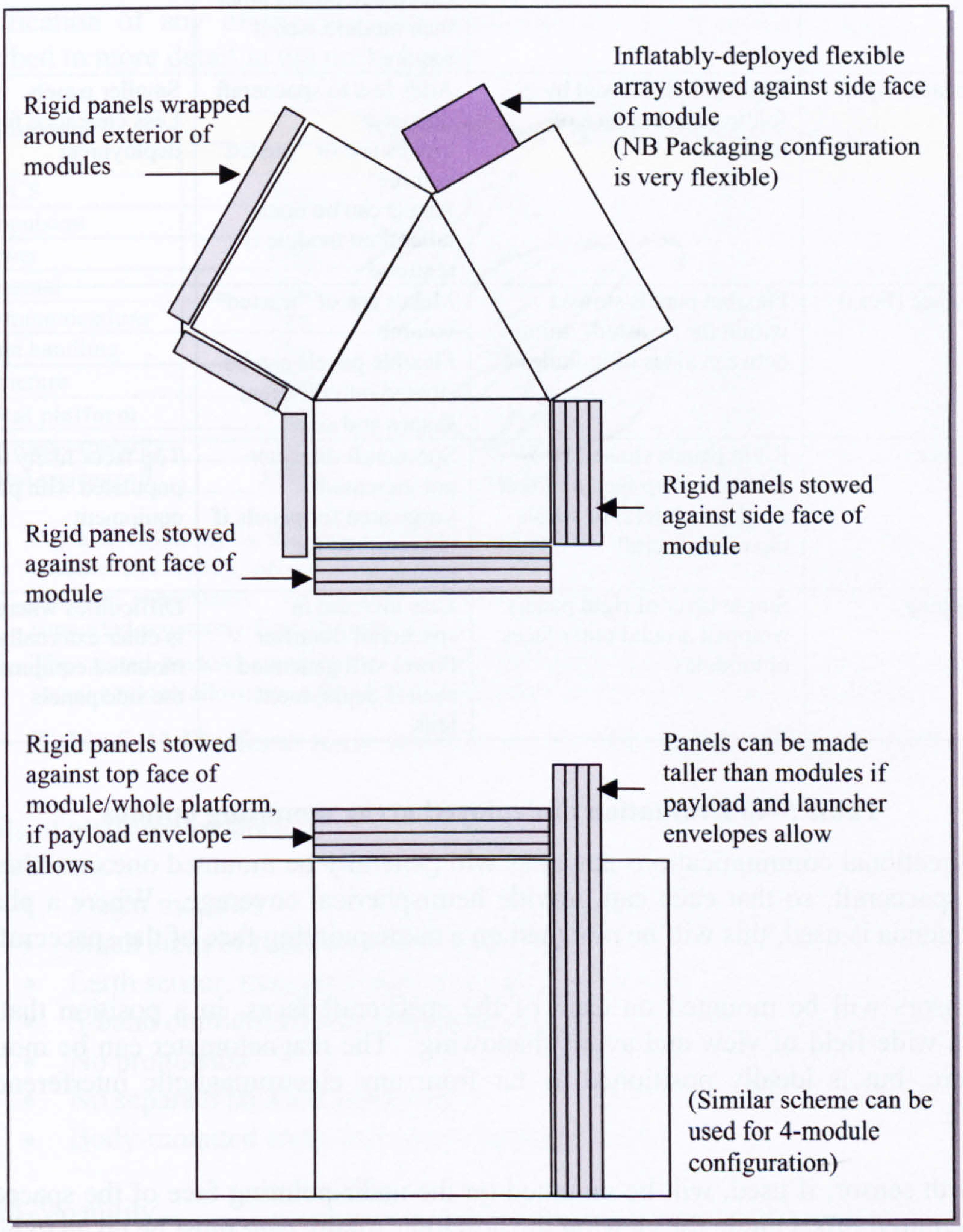


Figure 5-27 Options for stowing deployable solar arrays

The advantages and disadvantages of these stowing configurations are shown in Table 5-40. The configuration selected will be largely mission-dependent.

Configuration	Description	Advantages	Disadvantages
Front face	Rigid panels stowed by z-folding on front face of modules	Large area possible Easy access for mounting Good clearance for deployment Can make panels taller than module face if required	Adds to the spacecraft diameter (limited by launch envelope)
Side face (rigid)	Rigid panels stowed by z-folding on side face of modules	Adds less to spacecraft diameter Makes use of "wasted" volume Panels can be made taller than module if required	Smaller panels Less clearance for deployment
Side face (flexi)	Flexible panels stowed within the "wasted" volume between sides of modules	Makes use of "wasted" volume Flexible panels can be stowed into different shapes and sizes	
Top face	Rigid panels stowed by z-folding on top face of either single module(s) or whole top of spacecraft	Spacecraft diameter not increased Large area for panels if not required for payloads	Top faces likely to be populated with payload equipment
Wrapping	Single layer of rigid panels wrapped around outer faces of modules	Less increase in spacecraft diameter Power still generated even if deployment fails	Difficulties where there is other externally-mounted equipment on the side panels

Table 5-40 Evaluation of deployed array mounting options

Omnidirectional communications antennas will generally be mounted one on either end of the spacecraft, so that each can provide hemispherical coverage. Where a phased-array antenna is used, this will be mounted on a nadir-pointing face of the spacecraft.

Sun sensors will be mounted on each of the spacecraft faces, in a position that will afford a wide field of view and avoid shadowing. The magnetometer can be mounted anywhere, but is ideally positioned as far from any electromagnetic interference as possible.

The Earth sensor, if used, will be mounted on the nadir-pointing face of the spacecraft, at the required offset angle for viewing the horizon. Again, care must be taken to ensure that the required field of view is free of obstructions. Where a star-tracker is used, this will normally be mounted alongside the payload instrument(s) requiring accurate pointing. This is to minimise the effect of any mounting misalignments between payload and platform.

5.4.3 PLATFORM MASS BREAKDOWNS

A mass breakdown for the platform in selected different variants is given in Table 5-41. Selected capability steps are given for illustrative purposes; many slight differences in exact configuration are possible with the standard variants described for each subsystem. Details of the different variants are given below. At the detailed design level, each of these variants would be detailed in a database, allowing easy selection and identification of any of the possible combinations of subsystem variants. This is described in more detail in the next chapter.

Subsystem	Mass by platform variant (see separate details)			
	1	2	3	4
ADCS	8kg	22kg	28kg	23.5kg
Propulsion	0	0	40kg	0
Power	12kg	20kg	33kg	33kg
Thermal	2kg	2kg	2kg	2kg
Communications	3kg	3kg	3kg	8.5kg
Data handling	9kg	14kg	14kg	20kg
Structure	63kg	63kg	84kg	160kg
Total platform	**97kg**	**124kg**	**204kg**	**247kg**
Payload capability	221kg	194kg	260kg	587kg
Total spacecraft	**318kg**	**318kg**	**464kg**	**834kg**

Notes:

1. Payload capability is the "top-mounted" payload mass allowance (based on 50kg per "light" module and 125kg per "heavy" module), plus any additional mass capability not used by platform subsystems. Propulsion is considered separately, as it is a self-contained module, supported separately from the modules.
2. The quoted payload capability is the mass anticipated to be supportable by the platform. The actual payload allowable will depend on launcher selection and spacecraft target orbit.

Table 5-41 Platform mass breakdown for selected capability variants

Details of platform variants:

1. Small basic
    - 3 light modules
    - Small sizes of reaction wheels and rods
    - Earth sensor, magnetometer and sun sensors only
    - S-band omnidirectional communications
    - No propulsion
    - No separate payload processor
    - Body-mounted array and single battery module

2. Mid-capability
    - 3 light modules
    - Large sizes of reaction wheels and rods
    - Earth sensor, magnetometer and sun sensors only
    - S-band omnidirectional communications
    - No propulsion
    - Separate payload processor and extra mass memory
    - Deployed-fixed 300W array, extra battery modules

3. High manoeuvrability
    - 4 light modules
    - Large sizes of reaction wheels, cold gas thrusters
    - Star tracker, sun sensors, IRU
    - S-band omnidirectional communications
    - Hydrazine propulsion
    - Separate payload processor and extra mass memory
    - Deployed articulated 500W flexible array, extra battery modules

4. Large, high-spec
    - 4 heavy modules
    - Large sizes of reaction wheels and rods
    - Star-tracker, magnetometer, sun sensors, IRU
    - High-performance communications (X-band)
    - No propulsion
    - Redundant data handling system, with separate payload processing and extra mass memory
    - Deployed, articulated 500W flexible arrays, extra battery modules

This gives an idea of the range of sizes and configurations that can be constructed from the platform "standard parts list". This will be further illustrated in Chapter 6, where the platform is applied to several case study mission profiles.

The effect of the use of standard parts, modular design, and a "pre-design" approach on programmatics is now examined.

5.5 PROGRAMMATICS

A life-cycle flow of a typical spacecraft programme is shown in Figure 5-28 on the following page. In order to achieve the desired schedule (and cost) reductions, one or more of the activities in the programme must be compressed. The proposed programme should achieve reduction in the time required for many of the activities identified, and allow for a greater chance of success at the project bidding stage. Each of these activities is addressed in turn, to identify the ways in which the design and programme approach can give time and cost savings.

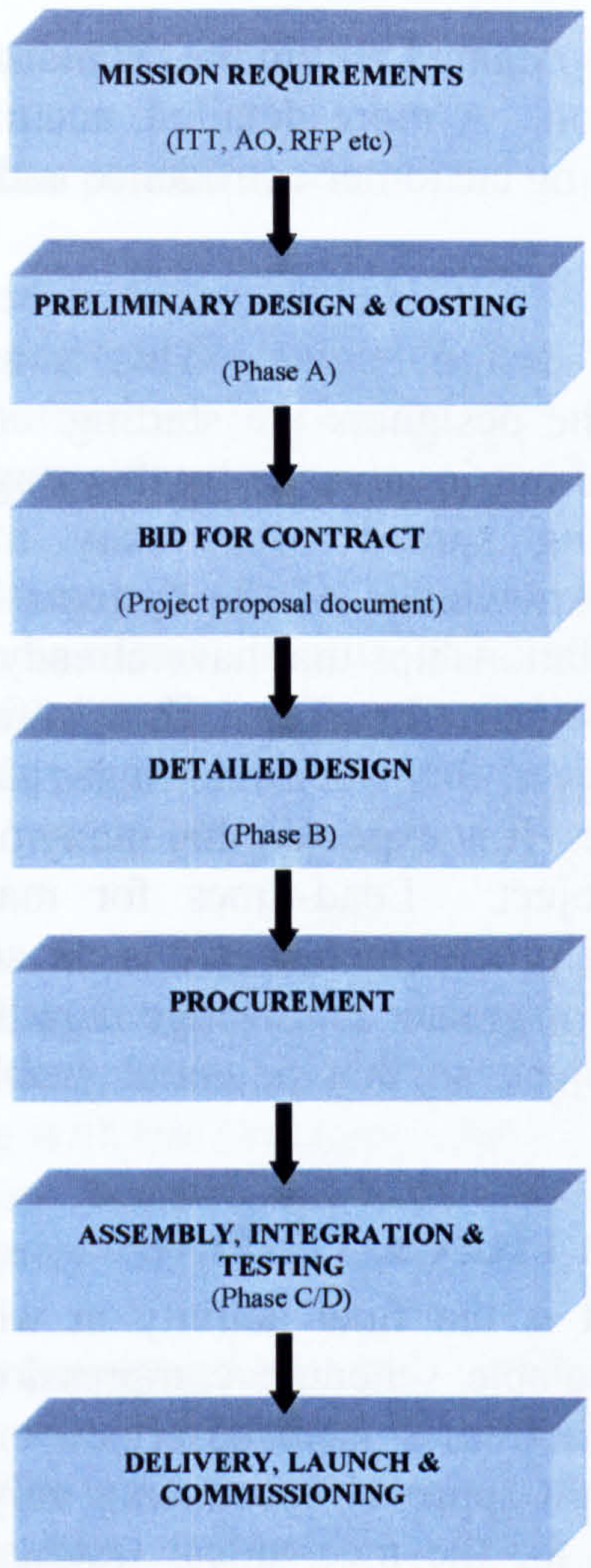


Figure 5-28 Programme flow for a typical spacecraft

5.5.1 PRELIMINARY DESIGN & COSTING

Most projects commence with an Invitation To Tender, Announcement of Opportunity, Request For Proposal, or similar. Essentially, this is the initial announcement for an intended mission, for which platform providers may wish to propose the use of their products. The interested parties must then:

- Firstly, assess whether the proposed mission is applicable to their platform, to decide if a bid will be made
- Perform the preliminary design and provide a feasibility study based on their products
- Produce a costing and schedule for the project

A bid for the project, containing the project proposal, can then be submitted.

At this stage, the proposed approach confers several advantages. The "pre-design" of different platform variants should allow a design fitting the mission specifications to be derived more quickly and easily than a from-scratch approach. Furthermore, information on performance, mass, power, cost, and availability for the bulk of the equipment is already in place, contained within the "standard parts list". This simplifies costing and schedule preparation. A more detailed, accurate, and convincing bid can therefore be assembled, increasing customer confidence and hence competitiveness.

5.5.2 DETAILED DESIGN

When a bid is won, detailed design begins. This is again helped by the greater knowledge base from which the designers are starting. A large part of the detailed design has already been carried out in advance by this stage, the non-recurring cost of this up-front investment being spread over many missions, for greater cost-effectiveness. The improved knowledge of the systems being used to make up the design, and the supply-chain relationships that have already been established, mean that the procurement process can be started earlier. The platform manufacturer may keep some equipment in-stock; however, this is a rather high-risk strategy, particularly at the beginning of such a programme. It is expected that many units will be bought in as per the requirements of each project. Lead-times for many space flight items are considerable, therefore starting procurement early is an advantage. Existing supply-chain relationships also allow for greater knowledge regarding scheduling and delivery lead-times early on in the project, so that potential problems or bottlenecks can be identified more quickly.

5.5.3 ASSEMBLY, INTEGRATION & TESTING

Assembly, integration and test is the final activity in which adopting the proposed programme can lead to appreciable schedule compressions and cost savings. AIT requirements, approaches and activities were described in Chapter 4. The proposed programme will use a protoflight approach, producing only one full spacecraft model. To reduce the levels required for the mechanical testing, a representative structural model will be tested at qualification level. The protoflight model will then be tested only at acceptance level.

The advantage with the proposed approach and design, is that the different structural designs can be qualified in advance, as standard configurations and modules are used. The appropriate modules can be constructed, with mass dummies for platform equipment and the payload. As most of the platform equipment comes from the standard parts list, it is largely only the payload mass dummies that must be specially made for each project. The platform can be constructed from a "kit" of representative mass dummies of all the standard platform parts (equipment and structure). This can be kept in-house, and configured as required for modelling each spacecraft produced. This further reduces time and cost.

Once full platform-level structural qualification tests have been passed, it may be acceptable to perform a large part of the mechanical testing at the structural module level. For example, it may be possible to accept flight modules based on satisfactory mechanical acceptance tests performed at the module level. This would give significant advantages:

- The modules could be tested as soon as each individual module was completed. This would allow mechanical testing to be performed in a

staggered fashion, without the need to wait until the whole platform was complete.
- The testing could be performed in smaller facilities, reducing testing costs.

However, if a large payload is being supported on the modules, and/or the payload structural model is not considered sufficiently representative, this approach may not be acceptable to the launcher authority. In any case, this approach may be able to limit the duration and cost of full spacecraft mechanical acceptance tests.

RF and thermal balance testing can also be carried out using the full structural model. The flight antennas can be mounted in their appropriate positions, and the RF beam patterns and performance validated. For thermal balance tests, appropriate heaters can be attached to the equipment mass dummies, to simulate operational power dissipation. These tests can be performed in parallel with integration and testing of the flight modules.

The structural model will also be useful for producing the wiring loom. Although the data loom may be minimised if a multifunctional structure panel approach is adopted (as the loom is integrated along with the structure), there will still be the power cabling and any additional data harness. Working with the structural model also progresses the learning curve of the team prior to PFM integration.

Where there is an on-going programme of small satellites, the modularity of the system can be exploited by using common equipment as both engineering model and flight-spare. This can reduce the amount of "wasted" equipment, whilst retaining the ability to replace equipment if a problem occurs in a flight unit. The modular spacecraft construction further assists in making it easier to de-mount and replace equipment items: as testing can proceed to an advanced stage before the whole platform is finally assembled, faults can be identified while the platform is in a more accessible state.

The AIT for the protoflight model will start with acceptance testing of equipment as it arrives. Assembly and test of each module is then carried out in parallel. Equipment is delivered, and "bench-level" test of the connected equipment is performed, to verify the functional interfaces. At this stage, a test harness and external power supply will be used, together with appropriate simulators and control EGSE. With nominal operation confirmed at this level, the equipment can be integrated into the module structure, and connected with the flight harness. Module-level functional testing is then repeated. At this stage, module-level mechanical acceptance testing can be performed if required.

In parallel with the module AIT, the payload will be being assembled and tested. This may be performed externally to the platform manufacturer, at a separate location. In this case, it may be useful to supply the structural model for fit-check purposes. Once all modules and payload are complete, assembly of the full spacecraft can be performed. A preliminary functional test can be performed using a test harness to make the inter-module and platform-payload connections. This will allow a check-out to be performed before the spacecraft is physically assembled. After assembly, a full system test will be performed.

The completed spacecraft will then be subjected to the required mechanical acceptance tests, and mass properties checks. Any deployables must also be tested. The thermal-vacuum test is then performed, in the orbital configuration, using the thermal inputs derived from the thermal balance test. System tests are performed after each test, to ensure that no faults have occurred. The spacecraft is also tested with the ground station in the loop. This may be a specially-built ground system for the project, or an existing, established ground station. Finally, the spacecraft can be packed for shipping to the launch site. Propellant (if used) is loaded in the controlled environment of the launch site integration facilities.

The time taken for the AIT phase of a space project may be 4 or 5 years for a large, complex spacecraft, down to less than a year for a simpler small satellite. The modular, parallel integration and testing approach proposed for the platform should allow integration time to be reduced to a level more consistent with a spacecraft of much lower performance and complexity. The standardised nature of the platform also means that the AIT teams will be able to apply lessons learned in initial projects, making further schedule reductions in later projects more likely.

A proposed timeline for AIT, with estimates for durations of the activities, is shown in Table 5-42. Actual durations will depend on a variety of factors, especially:

- Selected configuration
- Level of mission-specific equipment
- Payload configuration
- Manpower levels
- Available facilities
- Heritage, and team experience level

The timeline shown is an estimate for a first project using a particular platform configuration. It would therefore be anticipated that the schedule would be reduced for subsequent projects, through increased experience and familiarity.

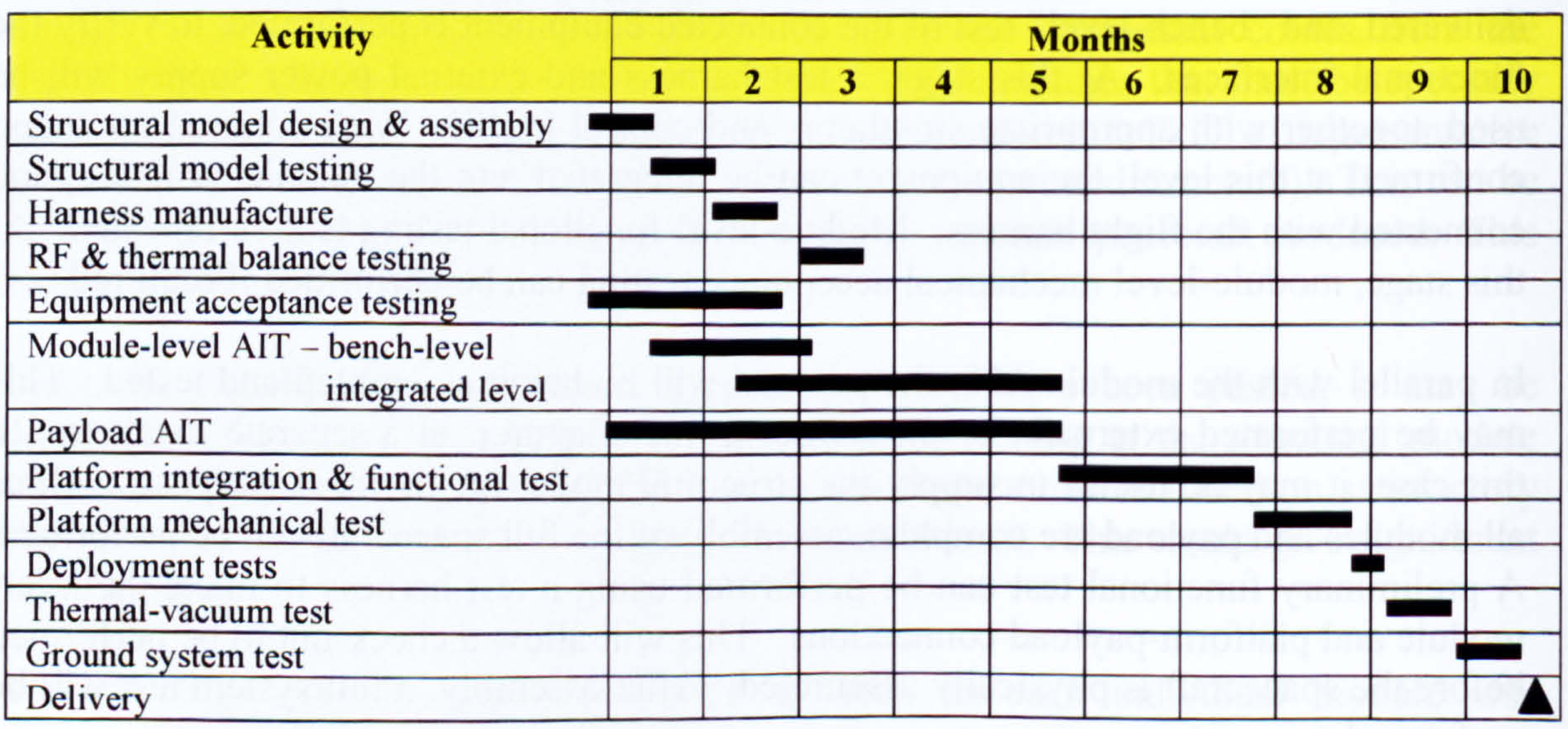


Table 5-42 Outline AIT schedule for a spacecraft using the proposed platform

This schedule is comparable to a microsatellite timeline[13]. This should be valid, as the integration and test process can be considered analogous to the parallel AIT process for several microsatellites (each module). Furthermore, the structural model can be more quickly designed, produced and tested, as it is largely a standard design that can be assembled from a suite of standard mass dummies and structural members.

5.5.4 SUMMARY

The overall philosophy is to shift as much of the time and effort "upstream", so it is shared across all of the spacecraft produced in the programme. The design effort is taken as far as possible in advance of the actual project specifications. This reduces the design time and speeds the response to customer requirements. Procurement can be started earlier, reducing delays to starting assembly. AIT duration is streamlined by the use of parallel processes, pre-qualification of modules (where permissible) through qualification of standard structural models, and application of lessons-learned to subsequent spacecraft.

An estimate of around 10 months or less is estimated for platform AIT and delivery. A similar or lower duration would be reasonable for spacecraft design and development, as only certain areas would require mission-specific design effort. There will also be some overlap with AIT, as some activities, such as software development, can be performed concurrently with integration. It is therefore expected that the target of an 18-month delivery time, proposed in the previous chapter, could be met quite comfortably, especially after the first couple of missions.

5.6 CHAPTER SUMMARY

This chapter has identified a suitable configuration for the modular platform, which may be configured in different ways to suit a range of missions. It is also suitable for a modular, parallel approach to AIT, for reduction in schedule time. The different configurations are sized to suit a range of different launch options and payloads.

The platform subsystems have also been defined, with different capability variants providing step levels in platform performance. Payload accommodation can range from around 200kg to over 500kg, depending on configuration. High-power options are possible via the use of deployed flexible arrays and multiple lithium battery modules. High data rates are possible via the use of an X-band adapter upgrade and phased-array antenna. Enhanced manoeuvrability can be offered through the use of thrusters, and orbit control can be provided by a hydrazine or electric propulsion module.

The system-level design proposed in this chapter gives a preliminary idea for configuration of the platform. The aim is that this preliminary design would then be taken into a more detailed design phase, for all the proposed variants. This would include production of a structural model. When this had been done, the manufacturer would be in a very strong position to bid for a wide range of space missions. The initial investment in manpower for the detailed design should then be recouped in the time and labour savings afforded by the "pre-designing" approach. This is quite a high-risk, high-reward strategy. However, the analysis of the market for small spacecraft conducted in Chapter 3 indicates that there should be sufficient target missions to support a competitive product.

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6 MISSION CASE STUDIES

6.1 INTRODUCTION

This chapter firstly addresses the process by which an appropriate platform configuration can be selected for a particular mission. The idea is to identify the most suitable configuration or configurations quickly and easily. The proposed platform, even with its limited standard parts list and capability variants within each subsystem, still has quite a considerable number of permutations. A set of standard payload evaluation and characterisation parameters is therefore used to describe the payload. These can be used to find a "match" with the most compatible platform configuration. The matching process can also identify if there is a difficulty in supporting the payload, which would require additional mission-specific design work.

A platform User's Manual is also proposed. This is analogous to the manuals produced by launch vehicle providers, and details the interfaces, performance, and envelopes available to prospective payloads.

Several illustrative mission case studies are also given, to demonstrate the application of the platform to different requirements. The platform is configured to sets of example mission requirements. These are derived from actual missions and payload instruments, identified in Chapter 4.

6.2 MISSION TO CONFIGURATION MATCHING

When a set of mission requirements is released in an ITT, the time available to define and submit the project proposal may be quite short. Therefore, it would be useful for the proposed programme to have a procedure for identifying the most appropriate configuration match, and highlighting any under-performing or conflicting areas. These would indicate that some mission-specific platform modification would be required. If this was minor, then the mission may be deemed worth bidding for. However, if the platform proved highly incompatible with the mission, a decision could be made that the effort of preparing a bid was not worthwhile.

Payload/mission characterisation may be divided into resource, performance, and operational parameters, as follows:

- Resource parameters – physical resources that must be supplied by the platform (supporting the payload mass, providing electrical power, accommodating the payload volume)
- Performance parameters – the capability of the platform to perform particular functions (manoeuvrability, pointing accuracy, stability, downlink data rates, data storage, data processing)
- Operational parameters – the ability of the platform to meet the operational profile of the mission (attitude, orbit, launch vehicle, lifetime)

Each of these parameters is discussed in Table 6-1 on the following page.

Parameter	Description	Remarks
Resource parameters		
Payload mass	Total mass of all payload equipment	Determines choice of heavy or light modules, and number of modules
Payload power	Peak power required by payload	Determines choice of array type and number of battery modules
Payload volume	Total volume of all payload equipment	Determines choice of module configuration. The choices may need to be refined later, if the payload is an unusual shape.
Performance parameters		
Manoeuvrability	Maximum slew rates required	Determines choice of attitude control option. (e.g. Slew rates >0.05° probably require thrusters.)
Pointing accuracy	Knowledge, accuracy, and stability	Determines choice of attitude sensor options
Data rate	Total payload data downlink rate	Determines choice of communications option (e.g. move to X-band for high rate, or to TDRSS-compatible for real-time)
Data storage	Amount of payload data storage required on board	Determines number of mass memory modules. Is related to data rate – these two areas can be traded off.
Processing	Requirement for onboard processing of payload data/ complex payload control e.g. high autonomy	Determines need for payload processor option
Operational parameters		
Attitude	Type of pointing required, e.g. inertial, nadir-pointing, spinning, momentum-bias	Determines choice of attitude control subsystem option
Orbit	Orbit type, altitude, eccentricity	Determines need for propulsion (in low orbits or if accurate orbit needed) Non-LEO orbits may determine need for extra shielding of electronics
Launcher	Choice of launch vehicle option(s)	Determines module configuration. Also impacts on payload mass and volume capabilities
Lifetime	Required mission life and reliability figures	Determines redundancy scheme and propellant budgets

Table 6-1 Payload/mission characterisation parameters

After the proposed platform has been taken to a more detailed design stage, the details of all the different configurations can be entered into a searchable database. This would contain details of the capability variants for each of the subsystems, allowing any of the possible configurations to be put together into a "virtual platform". This could be generated in accordance with the identified mission parameters, allowing rapid mission-to-configuration matching.

The computer database could also be configured to flag some platform permutations as non-recommended, for example if a deployed articulated array is selected in combination with a spinning spacecraft, or if an electric propulsion module is selected in combination with a low-capability power subsystem.

At the initial mission-definition stage, there are parameters that may be traded, for example downlink data rate and onboard data storage, or the mass of liquid propellant versus additional capability in the power subsystem in order to use electric propulsion. A database approach would be a useful system-engineering tool to make rapid checks of different possible solutions to the mission requirements.

It is important to note that ideal mission-configuration matching occurs when the mission parameters are equalled by (rather than merely satisfied by) the platform parameters. For example, a mission with a pointing accuracy parameter of 0.5° would be satisfied by a platform configuration with a pointing accuracy of 5 arcseconds. However, this configuration is *over-performing* with respect to the mission parameter. Over-performing is nearly as undesirable as under-performing, as the customer will be paying for capability that is not required. It is obviously unlikely that a perfect match will be found. In this case, a range of close matches can be selected. These may not necessarily all be taken from the over-performing side; it may be useful to include an additional "budget" option, which almost meets the mission specifications, in the project proposal, in case the customer is willing to trade slightly on requirements.

After selection of possible configuration candidates, these can then be examined in greater detail, and the most suitable selected for more traditional study. The database can then be used to generate equipment lists for the baseline designs. Some missions may not exactly "fit" any of the possible configurations. In this case a performance, resource, or operational "gap" is identified, and non-standard systems may have to be used or designed as mission-enablers.

The above approach outlines the method to be used when responding to a customer-generated ITT. Procurement of the platform may also arise when a customer produces a payload to fit the platform offered. This is, in effect, analogous to a spacecraft being produced to fit a particular launcher. Therefore, it would make sense to use a similar strategy to launch vehicle providers, and produce a platform User's Manual. This would follow a similar format to a launch manual, and include the details shown in Table 6-2.

Information area	**Details**	**Remarks**
General platform description	Introduction to the platform, general design and applicability. Also information on the manufacturer, flight heritage and qualification of the platform and equipment etc.	To provide customer confidence in the capabilities of the manufacturer, and the quality and validity of the platform. Also to give an overview of the type of platform being offered.
Payload mass capabilities	Allowable masses for the different configurations	Will vary dependent on launcher selected. Some examples can be given for illustrative purposes.
Payload power capabilities	Power production capabilities for the different power subsystem options	
Launcher compatibilities	Launch options for the different configurations	Payload envelopes could be shown for the different launch configurations

Payload mechanical interfaces and allowable volumes	Analogous to launcher envelope specifications. To be given for the different module configurations.	Will vary depending on the launcher selected. It would be assumed that potential customers would perform preliminary mission analyses based on the launcher compatibilities given.
Payload electrical interfaces	Data interface protocols, connector types, earthing schemes, allowable data rates, buffering requirements, voltages	
Performance capabilities	General performance options for the different variants, in pointing, data rates, and data storage	
Payload environment	Thermal, vibration, cleanliness and EMC environments anticipated	The manual would indicate that more controlled environments could be provided at a cost increment.
Management approach	Organisational responsibilities and interfaces, reviews, documentation, control activities, quality assurance	Example of how a typical project would be run
Schedule	Activities, durations, milestones	
Facilities	Facilities that could be made available for payload AIT activities	
Quality & safety	Requirements for the payload e.g. declared parts/materials lists	

Table 6-2 Information to be contained in the platform User's Manual

It is expected that more customers may be obtained using this approach, especially once the platform has been demonstrated successfully. It would be in the interests of both customer and payload manufacturer for payloads to be designed specifically for compatibility with the platform. The platform would require little or no tailoring to the payload, reducing the cost and schedule time. The payload could be produced to a specific plan, rather than design effort being wasted in producing a more generic design that could be accommodated on a range of possible platforms.

As with launch vehicle user guides, the information contained within the platform manual cannot be fully comprehensive. However, sufficient configuration examples could be compiled from the database that a good range of capabilities could be described. The user manual would probably evolve and become more detailed over time, as it could be partially compiled from the results of previous project studies. (Compilation of a detailed user manual as an end in itself would be quite expensive).

Customers wishing more detailed information can then approach the manufacturer directly, and submit a mission profile by providing the mission characterisation parameters derived earlier in this section. Mission definition can then proceed as previously described, and a proposed solution and costing offered.

The following section now takes several sample sets of mission characterisation parameters, and provides a baseline mission configuration for each.

6.3 CASE STUDIES

To demonstrate the applicability of the platform to different missions requirements, different example mission specifications are created, and suitable baseline spacecraft designs produced from the platform options. To make the mission requirements authentic, they are based on payload instruments and requirements from some of the missions identified in Chapter 4. The missions selected are:

- X-ray astronomy
- Magnetospheric physics
- Earth observation

These have been selected as they have very different mission parameters. The spacecraft resulting from mission-configuration matching will be quite different, yet will originate from the same platform standard parts list. This will illustrate the wide scope of the proposed platform.

6.3.1 X-RAY ASTRONOMY MISSION

The proposed mission for this case study is to perform astronomical observations at x-ray wavelengths. The mission will involve long durations of observation of x-ray sources, with the spacecraft being re-orientated to observe new sites of interest as required. This is based mainly on the HESSI and XTE missions. The mission parameters are shown in Table 6-3.

Parameter	Requirement	Remarks
Payload mass	120kg	Instrument based mainly on HESSI x-ray imaging spectrometer
Payload power	110W	
Payload volume	Main instrument 0.45m diameter, 1.4m long Plus electronics boxes	
Manoeuvrability	0.1° per second	To point to new targets of opportunity
Pointing accuracy	Pointing knowledge 5 arcsec Pointing accuracy 25 arcsec	
Data rate	10Gbits over 10-minute observation period	
Data storage	Unspecified	
Processing	Not required	
Attitude	Inertial	
Orbit	450km sunsynchronous	Low orbit avoids radiation Constant sun vector makes anti-sun-pointing easier
Launcher	Unspecified	
Lifetime	2 years	

Table 6-3 X-ray astronomy case study mission parameters

To arrive at a baseline configuration for the above mission, each of the spacecraft subsystems is addressed, to select the required platform options. Iteration is required to ensure compatibility between the chosen options.

6.3.1.1 Structural configuration

A tall, four-module configuration would allow the payload to be partially accommodated within the centre of the platform structure. The payload electronics boxes can be accommodated within the fourth module. This arrangement would permit launch on Taurus, (the smallest launcher that could accommodate the payload instrument dimensions). This also allows for inclusion of a propulsion module if required.

6.3.1.2 Communications and data handling

Due to the large quantities of data produced while the instrument is observing, a high downlink data rate is required. Therefore, the high-performance subsystem option will be used, with the X-band adapter and phased-array antenna. Two omnidirectional antennas are also used. The phased-array antenna is mounted on the outside face of one of the modules. This face is nadir-pointed for high-gain communications.

The data handling subsystem selection is again mainly influenced by the large quantities of data produced by the instrument. As payload processing is not required, the basic data handling option will be used, with additional mass memory modules to store the payload data until it can be down-linked. A baseline level of 100Gbit is used, which will allow data from 10 observation periods to be stored between downloads. At an X-band down-link data rate of 100Mbps, it would take at least two ground passes to transfer this quantity of data to the ground.

The use of more than one ground station would be recommended for this mission, to increase the frequency of data download.

6.3.1.3 Attitude and orbit determination and control

The high pointing accuracy and knowledge requirement implies selection of the high-accuracy subsystem option. However, high manoeuvrability is also required, so a combination of the high-accuracy and high-manoeuvrability options will be selected. This substitutes cold-gas thrusters for the magnetorquer rods in the high-accuracy option. Dual paired thrusters will be used in each axis.

6.3.1.4 Power

The power subsystem requirements are derived from the spacecraft peak power budget shown in Table 6-4.

System	**Peak power requirement /W**
Payload	110
Communications	35
Data handling and mass memory	45
Attitude and orbit determination and control	58
Power control	2
Total	**250**

Table 6-4 Estimated spacecraft peak power budget for x-ray astronomy mission

As the spacecraft is in a sun-synchronous orbit, and will have a limited attitude geometry with respect to the sun, the most appropriate choice for the solar arrays is to use a deployed-fixed configuration. Deployed-fixed arrays avoid jitter from drive mechanisms, and the power requirements of the spacecraft are fairly modest so some cosine losses are acceptable. Rigid, rather than flexible, arrays are chosen to avoid any array flexing affecting the pointing accuracy and stability of the spacecraft.

For a 2-year mission, cell performance degradation of (GaAs cells) will be 5.5%. Therefore, a 300W BOL array will give an EOL power of 267W. This meets the peak power requirements plus some margin. Three 100W rigid-panel array modules will be used. They will be stowed on the outside of three of the platform modules, and deployed to face the opposite direction to the payload instrument.

Two battery modules will supply at least 200W during eclipse. This should be sufficient if peak power-draw is not used during eclipse.

6.3.1.5 Spacecraft configuration s ummary

The spacecraft mass budget is shown in Table 6-5.

System	Equipment	Mass /kg
Payload		120
Communications subsystem	High-gain X-band + S-band omnidirational backup	8
Data handling subsystem	Main unit plus 100Gbit mass memory	10
Attitude & orbit determination & control subsystem	Wheels, cold gas thrusters, IRU, star tracker	35
Power subsystem	Deployed-fixed arrays, 2 battery modules	17
Thermal control subsystem	Passive	2
Structure	4 light modules	84
Total		**276**

Table 6-5 X-ray astronomy spacecraft mass budget

The relative mass fractions for the different spacecraft systems are shown in Figure 6-1. These mass fractions appear to be fairly standard, except for the structure mass perhaps being a little high. This is not unexpected, as it was suggested in the previous chapter that a full structural analysis would show that structural mass had been overestimated.

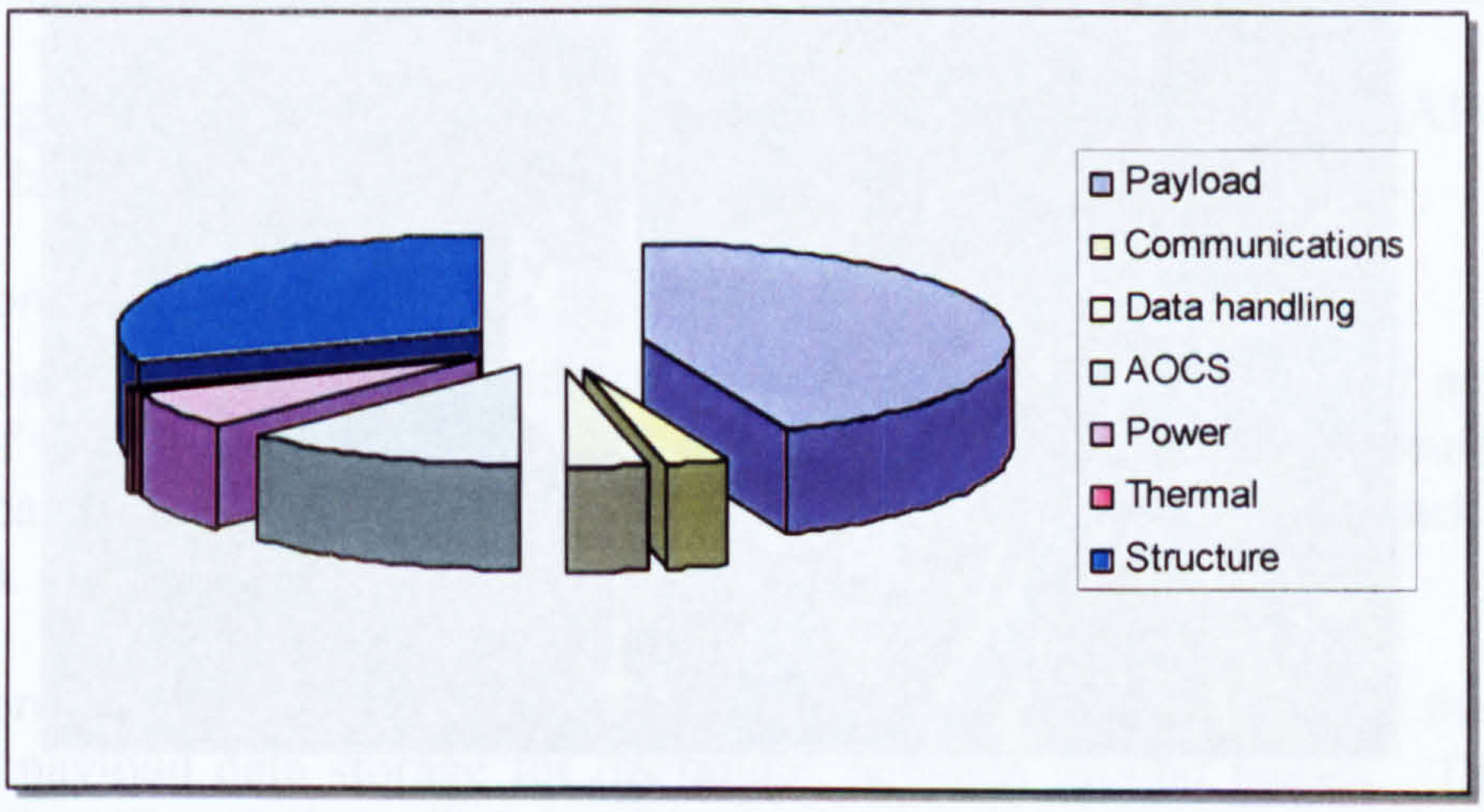

Figure 6-1 Spacecraft system mass breakdown

The general arrangement of the spacecraft, indicating the configuration of the platform modules, is shown in Figure 6-2. An overview of the on-orbit configuration is given in Figure 6-3.

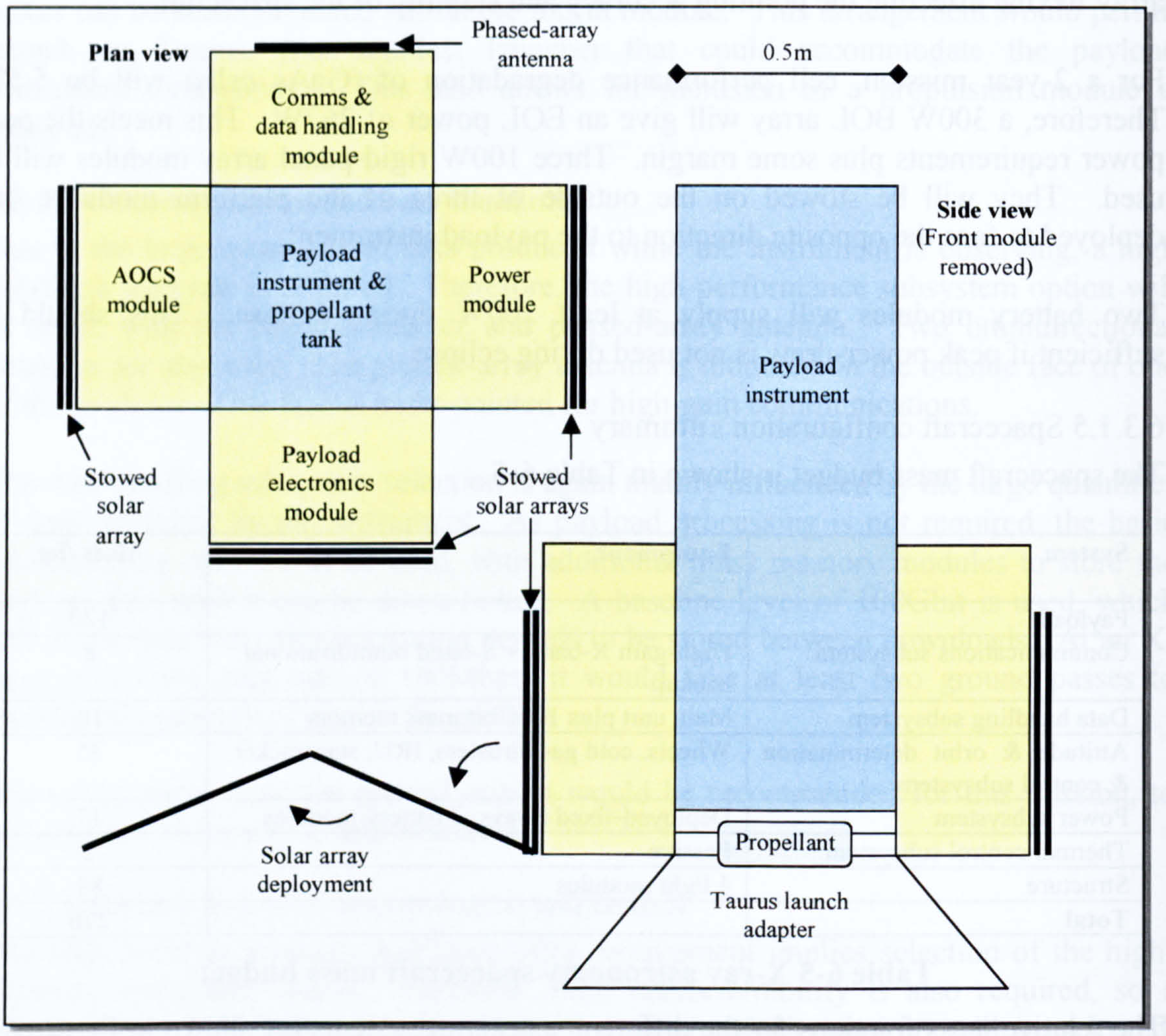

Figure 6-2 X-ray astronomy mission general spacecraft arrangement

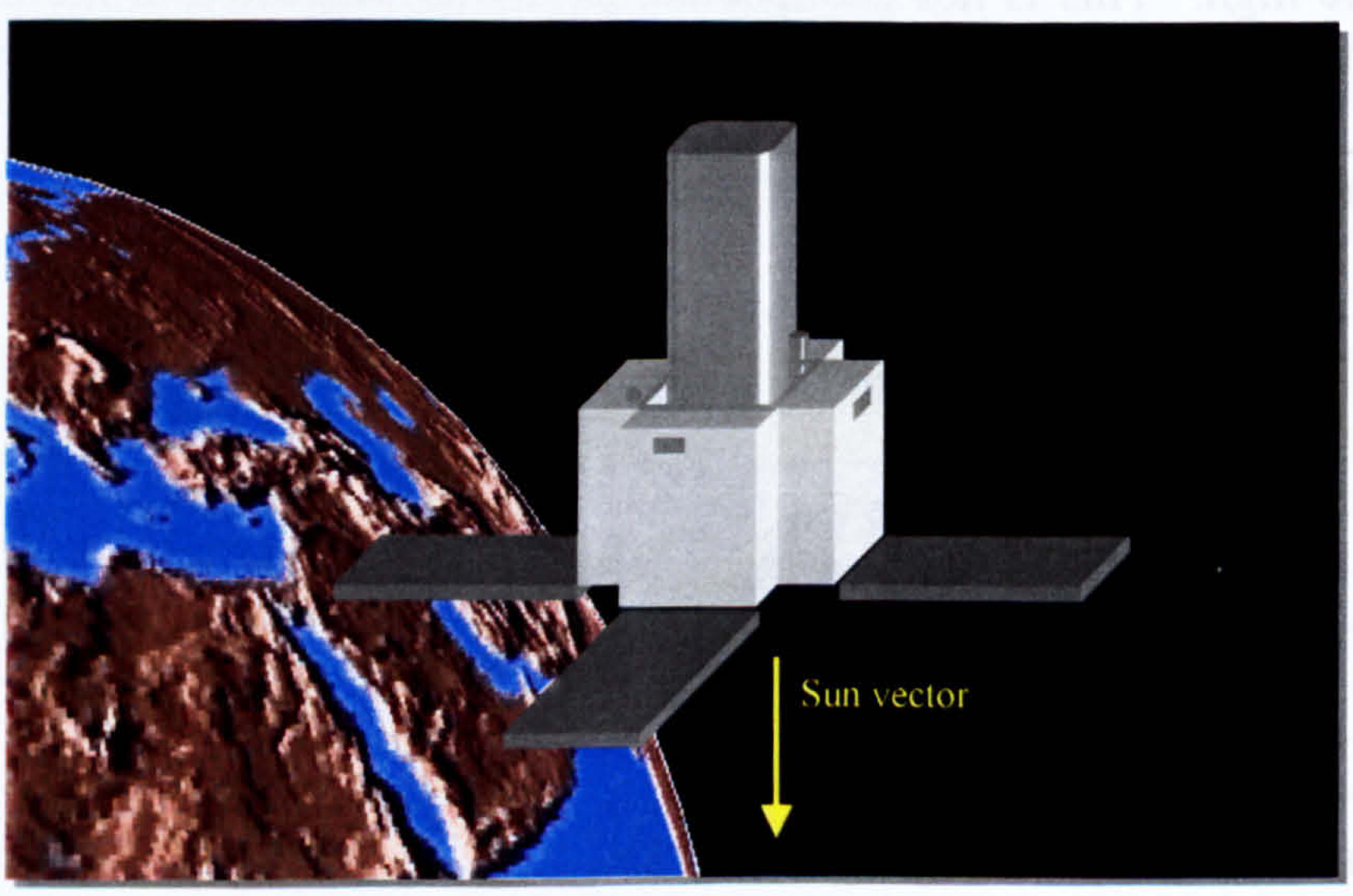

Figure 6-3 X-ray astronomy mission on-orbit configuration

6.3.2 MAGNETOSPHERIC PHYSICS MISSION

The proposed mission for this case study involves investigation of the Earth's magnetosphere, particularly the polar auroral regions. The mission parameters are shown in Table 6-6.

Parameter	Requirement	Remarks
Payload mass	Booms 3x2.5kg Magnetometers 3x2.5kg Charged particle detectors 2x2kg Plasma analyser 10kg	Based on STACER boom
Payload power	Total 28W	
Payload volume	Booms 102x115x264mm each Detectors 0.25x0.25x0.1m each Plasma analyser 0.5x0.4x0.4m	Magnetometers mounted on end of 6m-long booms
Manoeuvrability	Unspecified	
Pointing accuracy	5° (knowledge 0.1°)	
Data rate	20Kbps total	
Data storage	Unspecified	
Processing	Not required	
Attitude	Spin-stabilised, spin axis normal to the sun	
Orbit	Polar, 600km	
Launcher	Unspecified	
Lifetime	2 years	

Table 6-6 Magnetospheric physics case study mission parameters

The spacecraft subsystem options are now assessed, and the most appropriate configuration selected for the above mission.

6.3.2.1 Structural configuration

The spacecraft is required to be spin-stabilised, and has several medium-sized payloads. A tall three-module configuration is used, with one of the stowed booms accommodated in each of the three platform modules. This is possible as the spin-stabilised platform requires much less hardware in the attitude control module, and the other modules have sufficient spare volume with the subsystem options selected. The largest payload instrument, the plasma analyser, is mounted centrally on the top of the modules.

This configuration is compatible with launch on Pegasus, Taurus, or ASAP-5 (although an ASAP launch is less likely to LEO).

6.3.2.2 Communications and data handling

The payload data rate is low, and assuming that data is only collected near the polar regions of the orbit, basic S-band omni-directional communications should satisfy the mission parameters. Omnidirectional antennas will be mounted at each end of the spacecraft.

Payload processing is not required, and a single mass memory module should provide sufficient payload data storage for operations between ground passes. Therefore, the basic data handling option will be used.

6.3.2.3 Attitude and orbit determination and control

The spacecraft is spin-stabilised, and will therefore require thrusters for spin-up and precession control. As the spacecraft is quite small, and orbit control is not required, a cold gas system is suitable. Dual paired thrusters in all three axes will allow for spin rate management and control of the angular momentum vector. The thrusters will be the only actuators required, although passive nutation dampers are also included in the spin-stabilisation module option.

Sun sensors will give the required pointing knowledge accuracy. These will be mounted on the exterior panels.

6.3.2.4 Power

The power subsystem requirements are derived from the spacecraft peak power budget, shown in Table 6-7.

System	Peak power requirement /W
Payload	28
Communications	30
Data handling and mass memory	15
Attitude and orbit determination and control	<5
Power control	3
Total	**<81**

Table 6-7 Estimated spacecraft power budget for magnetospheric physics mission

The power requirement is very low, and can be achieved with body-mounted cells. To increase the available area, closure panels are inserted between the modules. With the spin axis normal to the sun vector, this gives a projected cell area of approximately $0.7m^2$. Populating with GaAs cells, and allowing for "lost" area due to sun-sensors and the particle detectors, a BOL power-raising capability of at least 100W should be obtainable. For a two-year mission, this gives an EOL power of at least 94W, assuming a cell performance degradation of 2.75% per year.

A single battery module will be used, which will give 100W of power during eclipse.

6.3.2.5 Spacecraft configuration summary

The spacecraft mass budget is given in Table 6-8.

System	Equipment	Mass /kg
Payload		29
Communications subsystem	S-band, omnidirectional antennas	2.5
Data handling subsystem	Main unit, with 1 mass memory module	8.5
Attitude and orbit determination and control subsystem	Spin-stabilisation, cold-gas thrusters, passive nutation dampers, sun sensors	10
Power subsystem	Body-mounted cells, 1 battery module	12
Thermal subsystem	Passive	2
Structure	3 light modules	63
Total		**127**

Table 6-8 Magnetospheric physics spacecraft mass budget

The mass fraction for each of the spacecraft systems is illustrated in Figure 6-4. This shows the high structure mass fraction, due to the platform being somewhat over-sized for the mission requirements.

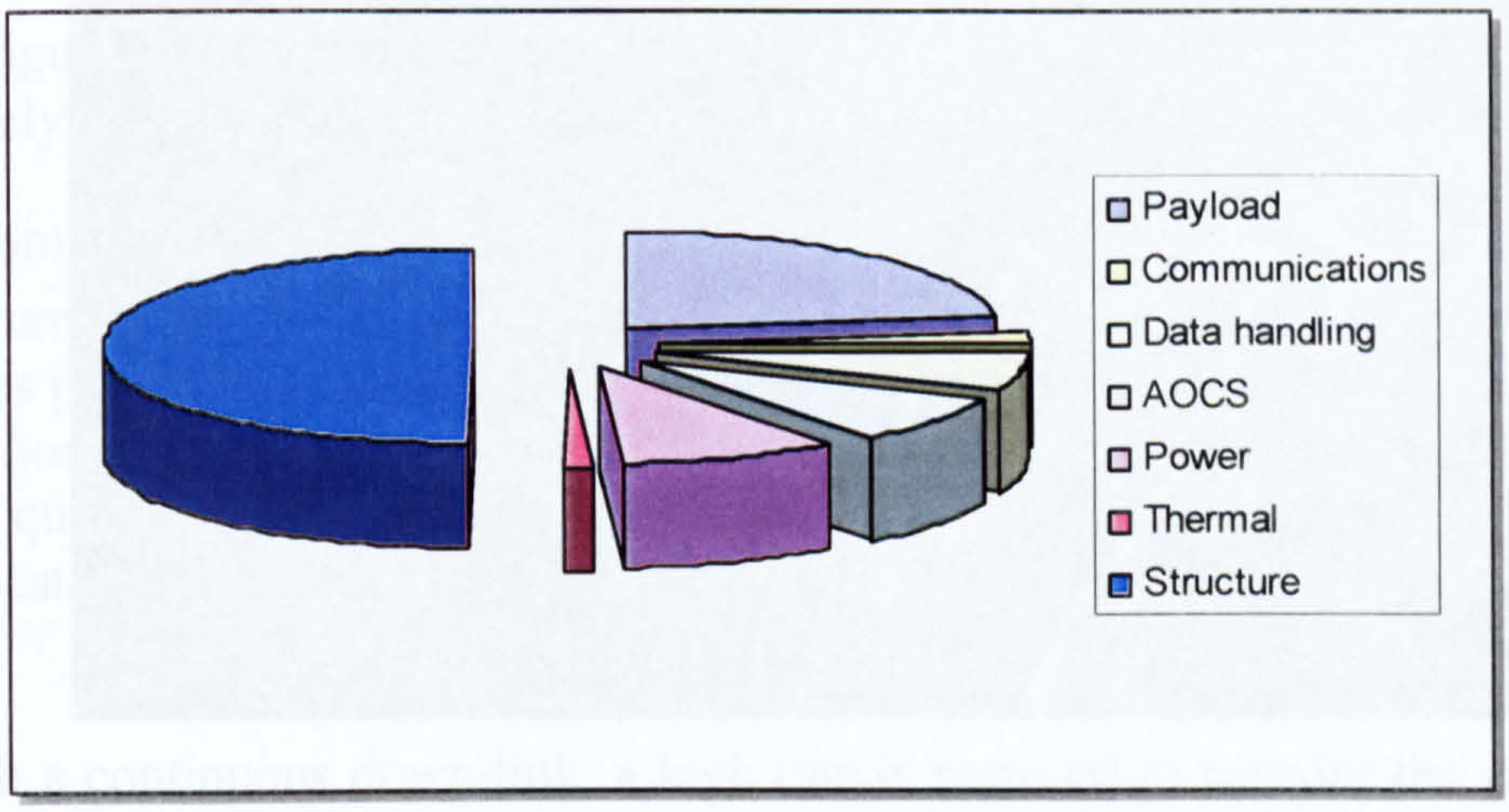

Figure 6-4 Spacecraft system mass breakdown

The general arrangement of the spacecraft, including the platform modules, is shown in Figure 6-5. An overview of the on-orbit configuration is given in Figure 6-6.

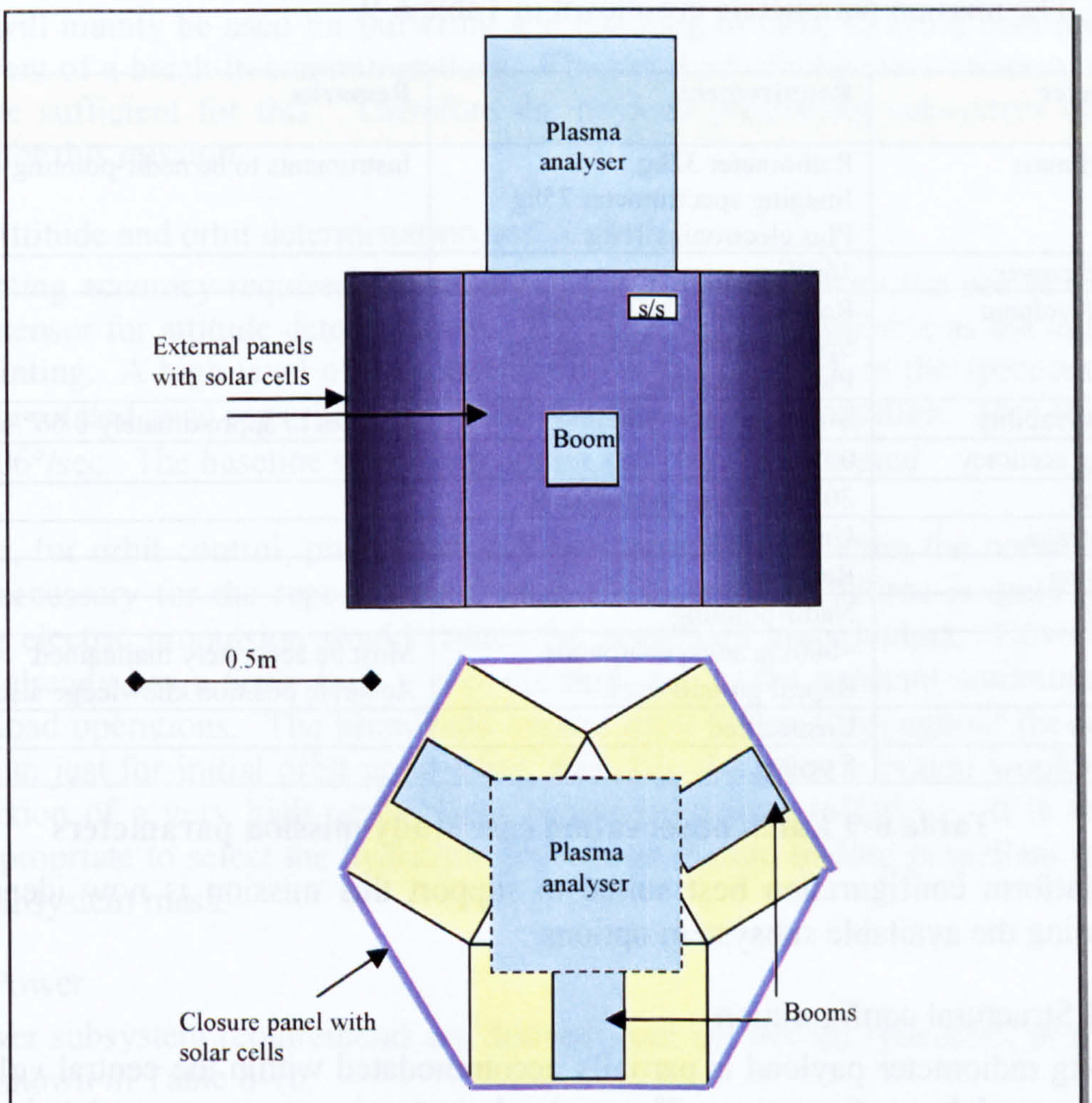

Figure 6-5 Magnetospheric physics mission spacecraft general arrangement

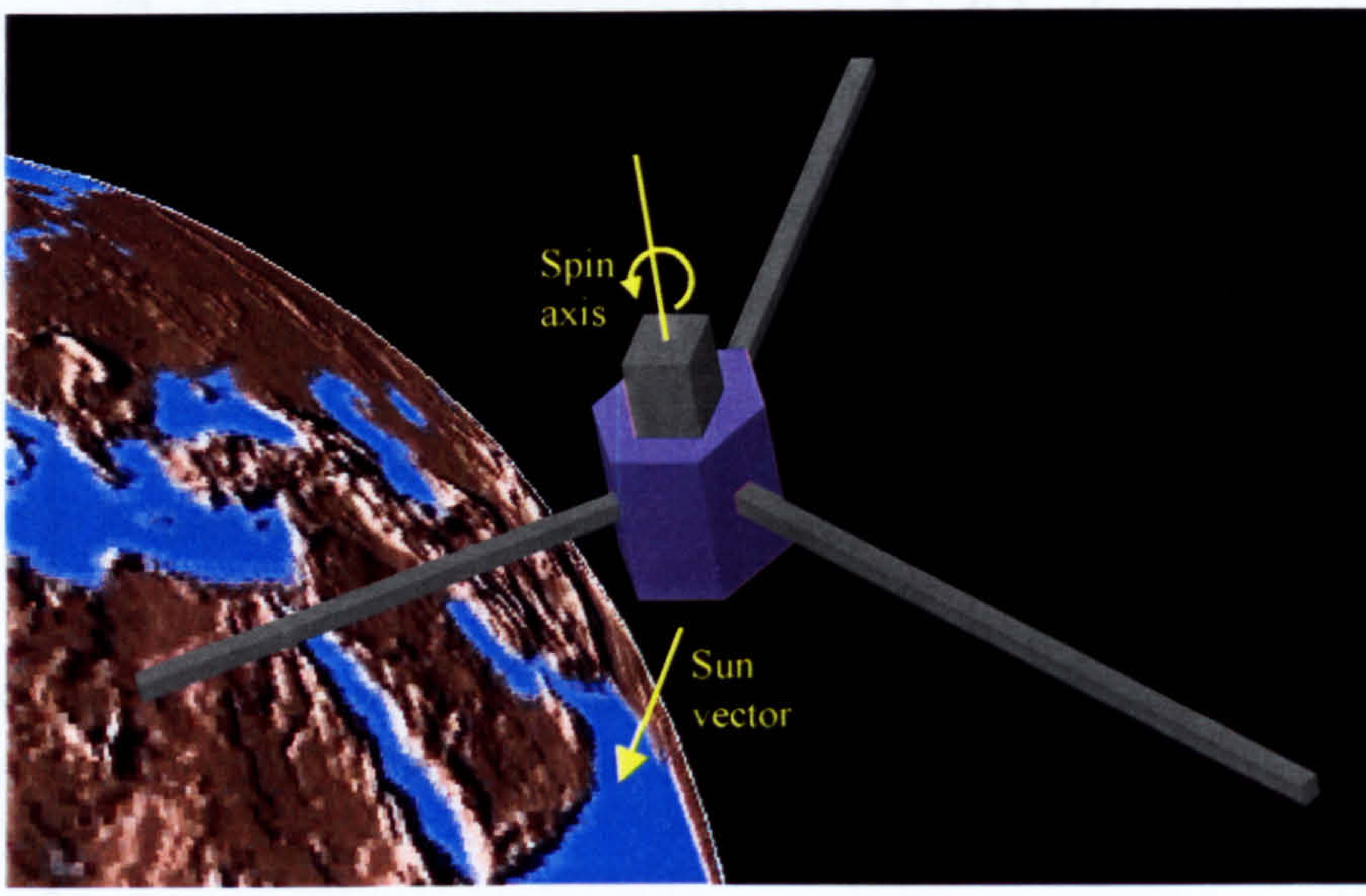

Figure 6-6 Magnetospheric physics mission on-orbit configuration

6.3.3 EARTH OBSERVATION MISSION

The proposed mission for the final case study involves global imaging from Low Earth Orbit. The mission parameters are shown in Table 6-9.

Parameter	Requirement	Remarks
Payload mass	Radiometer 32kg Imaging spectrometer 25kg Plus electronics 10kg	Instruments to be nadir-pointing
Payload power	160W	
Payload volume	Radiometer 0.3x0.4x0.8m Spectrometer 0.3x0.2x0.1m Plus electronics	
Manoeuvrability	Maintain nadir-pointing	Equates to approximately 0.06°/sec
Pointing accuracy	0.05°	
Data rate	30Mbps from instruments	
Data storage	Unspecified	
Processing	Required	
Attitude	Nadir-pointing	
Orbit	~600km sunsynchronous Repeat ground-track	Must be accurately maintained. Accurate position knowledge also required.
Launcher	Unspecified	
Lifetime	5 years	

Table 6-9 Earth observation case study mission parameters

The platform configuration best suited to support this mission is now identified, by addressing the available subsystem options.

6.3.3.1 Structural configuration

The long radiometer payload is partially accommodated within the central volume of a tall, four-module configuration. The payload electronics are accommodated within the fourth module, together with the Earth horizon sensor. The propulsion module is also accommodated within the central volume, near the launch adapter. The deployable arrays are mounted on the outside faces of two of the modules, with the solar array drive mechanism housed within the body of each module.

This configuration is compatible with a Taurus or ASAP-5 launch (although the ASAP is less likely for a LEO mission).

6.3.3.2 Communications and data handling

The instruments produce data at high rates, and it is expected that there will be continuous payload operation to achieve global mapping. Therefore, it is not feasible to use the short ground passes experienced in LEO for data download, or to store the required quantities of payload data on board. For this mission, near-real-time communications are more suitable, so the TDRSS-compatible subsystem option is selected.

Even with a continuous down-link, a high rate is required to transfer the payload data. The spacecraft will therefore transmit via TDRSS at x-band. S-band omnidirectional antennas are used for commanding and back-up downlink.

The main data-handling requirement specified is for payload processing. With real-time data transmission, there will be a lower requirement for onboard data storage. The storage will mainly be used for buffering and handling of data, to avoid data drop-outs in the event of a break in communications. The standard 25Gbit mass memory module should be sufficient for this. Therefore the payload processing subsystem option is selected for this mission.

6.3.3.3 Attitude and orbit determination and control

The pointing accuracy required for the mission is achievable with the use of an Earth horizon sensor for attitude determination. This is particularly suitable as the mission is nadir-pointing. A high level of manoeuvrability is not required, as the spacecraft must be merely rotated once per orbit to maintain its Earth-facing orientation. This equates to about 0.06°/sec. The baseline subsystem option can therefore be used.

However, for orbit control, propulsion will be required to maintain the nominal orbit. This is necessary for the repeat ground track. The mission lifetime is quite long, so selecting electric propulsion would reduce the propellant mass budget. However, the mission already has a fairly high power demand, due to the constant communications and payload operations. The propulsion system must be used throughout the mission, rather than just for initial orbit acquisition, therefore the electric option would require the selection of a very high-performance power subsystem (600W+). It is therefore more appropriate to select the hydrazine propulsion option, trading propellant mass for power subsystem mass.

6.3.3.4 Power

The power subsystem requirements are derived from the overall spacecraft peak power budget, shown in Table 6-10.

System	Peak power requirement /W
Payload	160
Communications	35
Data handling and mass memory	18
Attitude and orbit determination and control	175
Power control	3
Total	**391**

Table 6-10 Estimated spacecraft power budget for Earth observation mission

The spacecraft peak power requirement is quite high, with the payload and communications equipment in constant operation. To achieve the requirements, a deployed, single-axis articulated option is selected. A deployed-fixed option would give high cosine losses due to the variations in sun angle (since the spacecraft is not in a sun-synchronous orbit). With one-axis articulation and the use of yaw-steering, the arrays can be maintained in a sun-pointing geometry.

Two $2m^2$ flexible deployed arrays would give an EOL power of 381W at the end of the 5-year mission. Assuming that any peak-power-draw would take place only in sunlight, when array power can be boosted from the batteries if necessary, this option should satisfy the mission criteria. Four battery modules supply 400W in eclipse periods.

6.3.3.5 Spacecraft configuration summary

The spacecraft mass budget and system summary is given in Table 6-11.

System	Equipment	Mass /kg
Payload		67
Communications subsystem	TDRSS-compatible, X-band (payload data) + S-band (command & backup)	10
Data handling subsystem	Main unit + payload processing unit	13
Attitude and orbit determination and control subsystem	Wheels, magnetorquers, Earth horizon sensor, hydrazine propulsion	42
Power subsystem	Two $2m^2$ deployed flexible arrays, 1-axis articulation, 4 battery modules	26
Thermal subsystem	Passive	2
Structure	4 light modules	84
Total		**244**

Table 6-11 Earth observation spacecraft mass budget

A proportional breakdown of the mass of the spacecraft systems is illustrated in Figure 6-7. This shows the relatively high mass of the AOCS subsystem, due to the requirement for a propulsion module. The power subsystem is also quite heavy, although mass savings are made through using the inflatably-deployed flexible arrays.

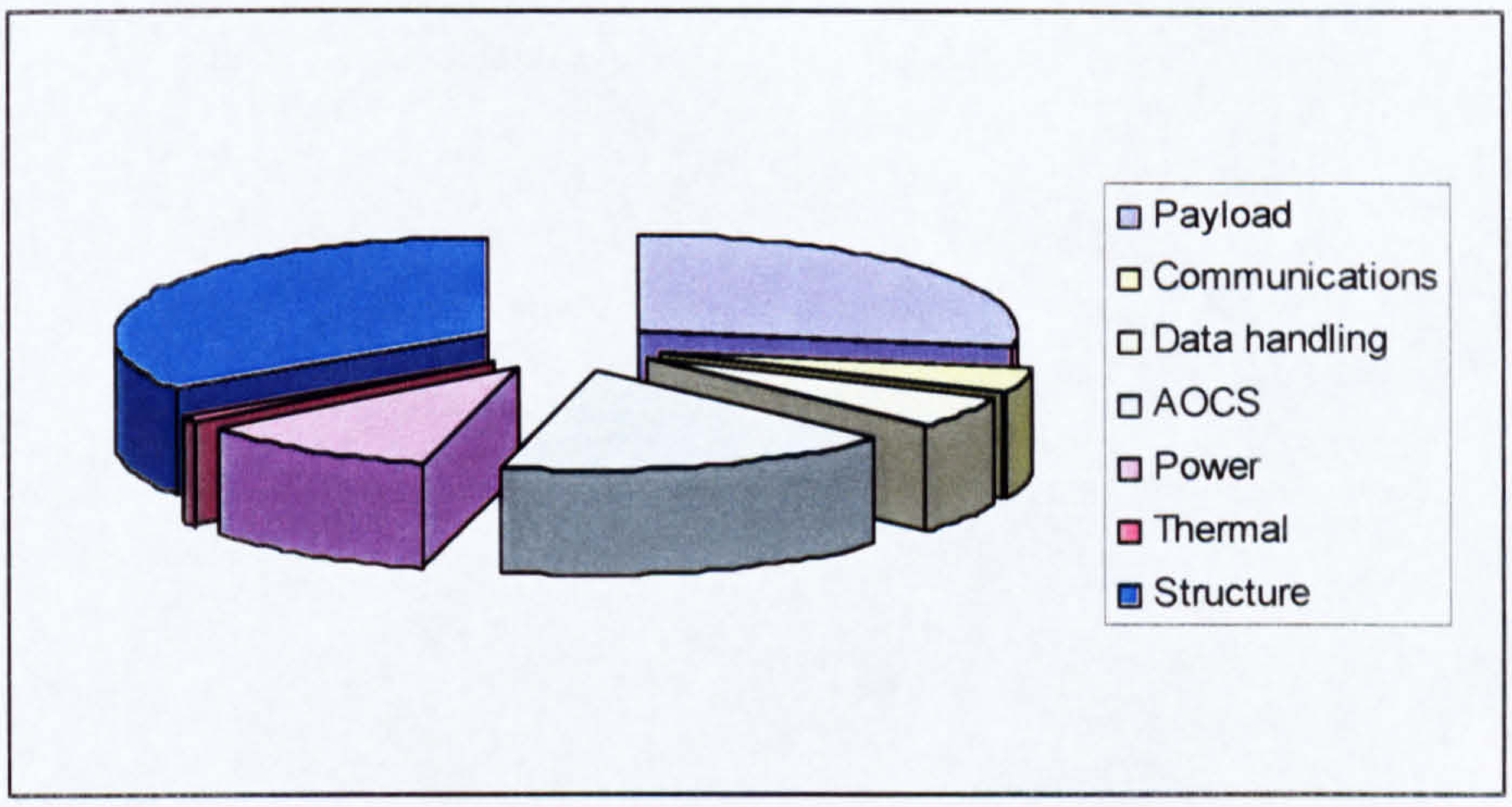

Figure 6-7 Spacecraft mass breakdown

The general arrangement of the spacecraft is shown in Figure 6-8. The on-orbit configuration is illustrated in Figure 6-9.

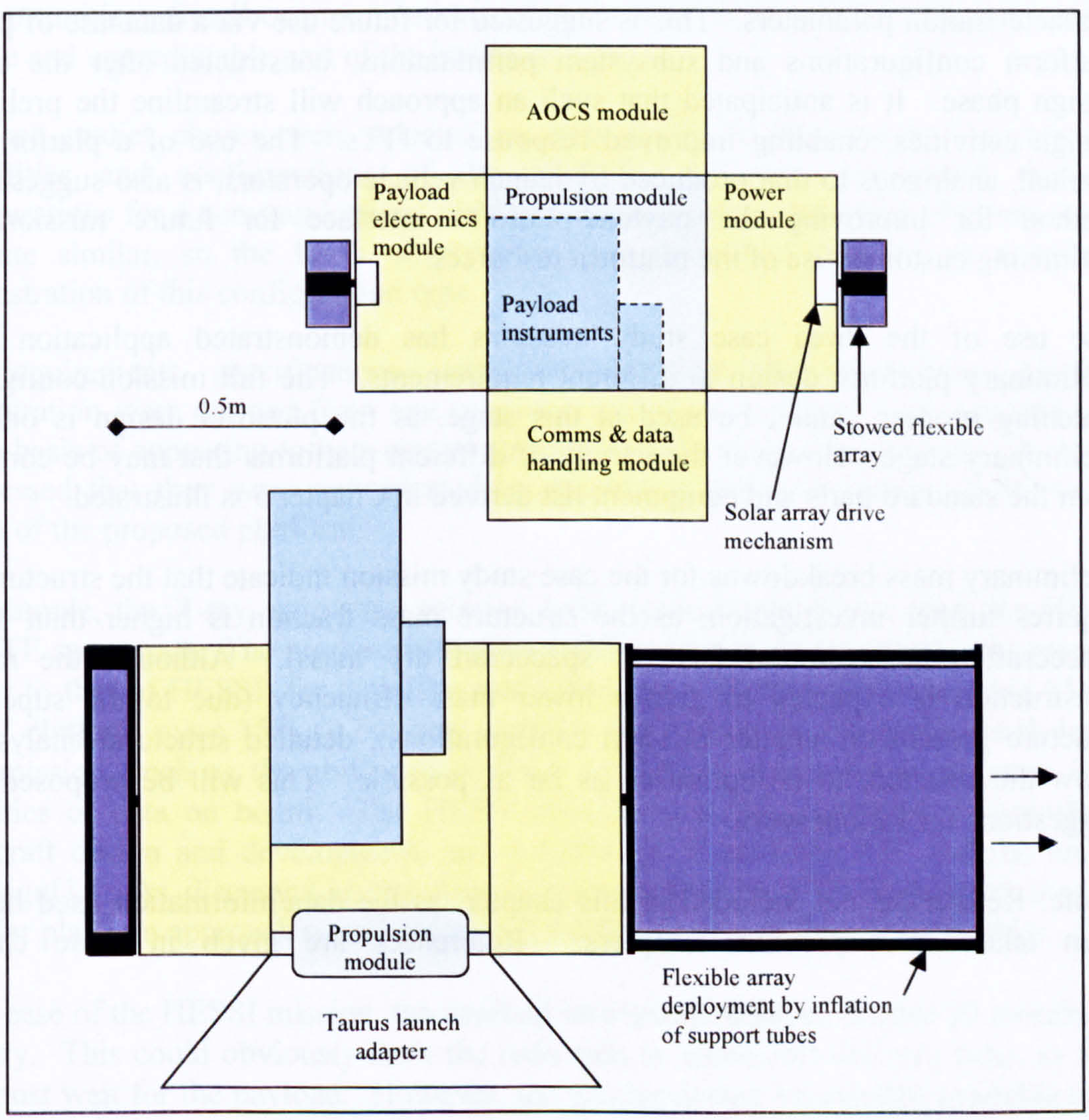

Figure 6-8 Earth observation spacecraft general arrangement

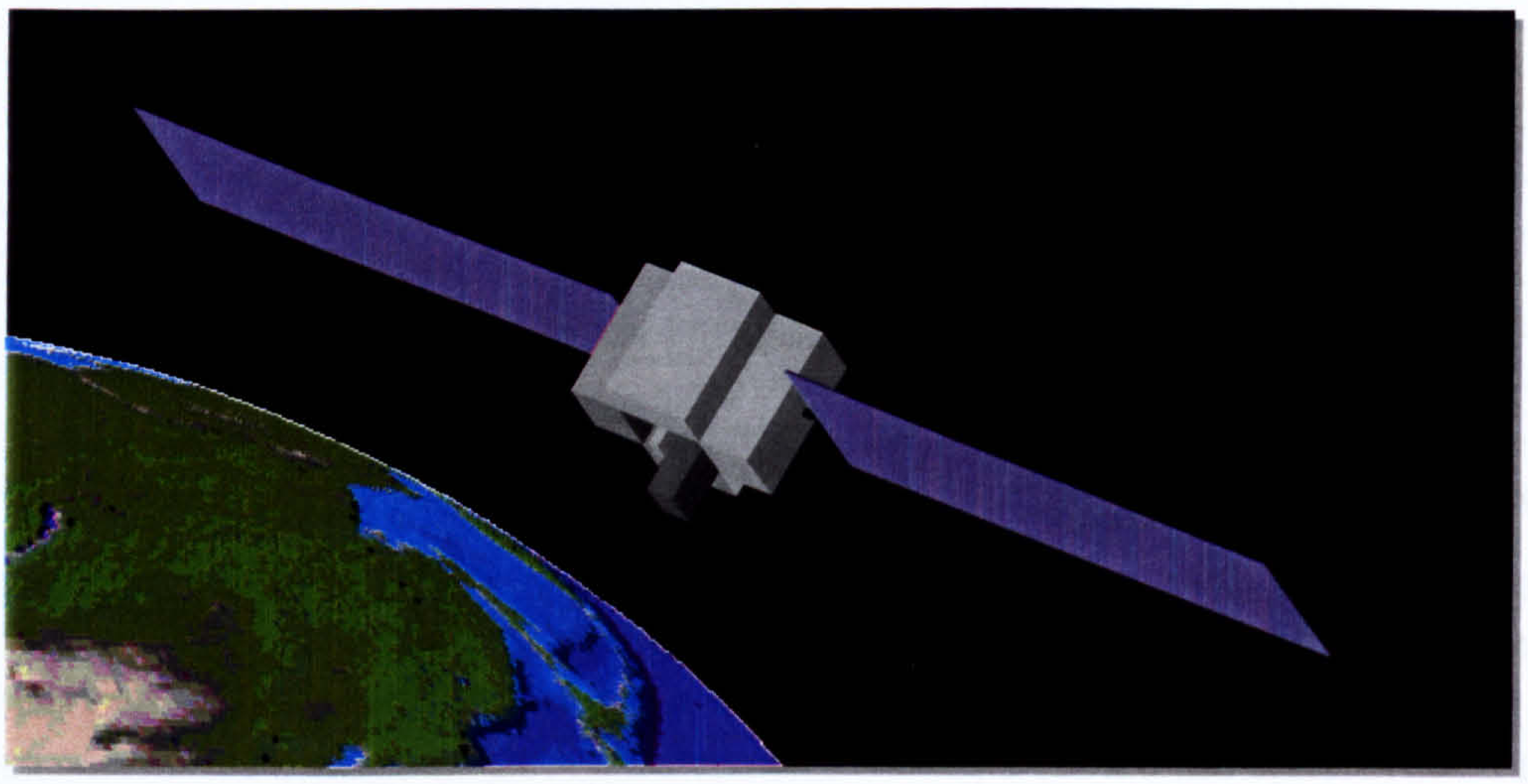

Figure 6-9 Earth observation spacecraft on-orbit configuration

6.4 CHAPTER SUMMARY

A mission-configuration matching process has been proposed, using a set of mission characterisation parameters. This is suggested for future use via a database of possible platform configurations and subsystem permutations, constructed after the detailed design phase. It is anticipated that such an approach will streamline the preliminary design activities, enabling improved response to ITTs. The use of a platform User Manual, analogous to that produced by launch vehicle operators, is also suggested as a method for improving the payload-platform interface for future missions, and optimising customer use of the platform resources.

The use of the three case study missions has demonstrated application of the preliminary platform design to different requirements. The full mission-configuration matching process cannot be used at this stage, as the platform design is only at a preliminary stage. However the variety of different platforms that may be configured from the standard parts and equipment list derived in Chapter 5 is illustrated.

Preliminary mass breakdowns for the case study mission indicate that the structure mass requires further investigation, as the structure mass fraction is higher than average spacecraft values (around 20% of spacecraft dry mass). Although the modular construction is expected to give a lower mass efficiency (due to the superfluous structure present on smaller mission configurations), detailed structural analysis will allow the structure to be optimised as far as possible. This will be proposed in the suggestions for further work.

[Note: References not included in this chapter, as the data/information used here has been taken from previous chapters. References are given in these chapters.

7 DISCUSSION

This chapter discusses the results obtained in the Chapter 6 case studies, and compares the reconfigurable platform solutions with those using conventional designs. The idea presented in the previous chapter of an expert system for mission-to-configuration matching is also examined with a view to further work. Finally, the cost benefits of adopting the proposed programme, and the benefits of the modular platform for that programme are also discussed.

7.1 CASE STUDY RESULTS

The missions selected for the case studies in the previous chapter formed a representative sample from the target mission scope of the platform. As examined in Chapter 3, the largest proportion of the business is expected to come from the Earth observation sector, with various science missions forming a smaller, but steady, "core" of opportunities. Small communications missions were anticipated as forming a more volatile and unpredictable part of the business.

The case studies chosen were selected on their ability to demonstrate the range of capabilities and configurations of the platform. It was considered that the configurations for a communications platform and an Earth observation platform would be quite similar, so the Earth observation case study was selected as a suitable demonstration of this configuration type.

The requirements specifications used for selecting the appropriate platform configuration were derived from real missions, and these missions were selected only on the basis of appearing to be a representative example of the mission type. It should be stressed that they were not selected by examining their "closeness of fit" to the design of the proposed platform.

For example, the X-ray astronomy mission used requirements drawn from the HESSI and XTE spacecraft. The proposed solution satisfies the key requirements of a mission similar to that of HESSI, for a similar mass (proposed solution platform mass 156kg, HESSI platform mass 154kg), but also includes capabilities found on the much larger XTE mission, such as the ability to perform rapid slew manoeuvres and store large quantities of data on board. The HESSI mission schedule dedicated 23 months to spacecraft design and development, and a further 7 months for AIT and the launch campaign[1]. As discussed in the System Design chapter, it is anticipated that the modular platform approach would reduce total delivery time to around 18 months.

In the case of the HESSI mission, the payload instrument required around 20 months for delivery. This could obviously limit the reduction in spacecraft delivery time, as final AIT must wait for the payload. However, the greater design knowledge available early in the project, due to the "pre-design" approach of the platform, will allow design and production of payload instruments to be started earlier. This would be analogous to designing a spacecraft to known standard launcher interfaces.

The proposed solution for the magnetospheric physics sample mission has a similar platform mass and size to the FAST[2] and SAMPEX[3] missions, again giving confidence in the hardware configuration. The lower spacecraft mass is largely due to lower mass of payloads; the selected configuration could, however, support a much greater payload mass than this. The improved performance in terms of onboard data storage is due to improvements in the technology since the previous missions. The proposed Earth observation spacecraft design solution also has a similar mass, size and capability to the TOMS-EP spacecraft[4].

A comparison of key mission parameters for previous missions and the case study solutions is shown in Table 7-1.

ASTRONOMY	**HESSI[1]**	**WIRE[5,6]**	**Case study**
Spacecraft mass	238kg (payload 130kg)	250kg (payload 93kg)	276kg (payload 110kg)
Spacecraft power	165W	158W	267W EOL
Onboard data storage	2Gbyte	30MByte	100Gbit
Downlink data rate	3.5Mbps	2.25Mbps	100Mbps (X-band)
Stabilisation	spin	3-axis (wheels + magnetorquers)	3-axis, (cold gas thrusters + wheels)
Pointing accuracy	1.5 arcsec knowledge (spin axis)	1 arcminute	3 arcsec knowledge
Propulsion	None	None	None
PHYSICS	**FAST[2]**	**SAMPEX[3]**	**Case study**
Spacecraft mass	191kg (payload 51kg)	160kg (payload 52kg)	127kg (payload 29kg)
Spacecraft power	52W	102W av. EOL	94W EOL
Onboard data storage	1Gbit	30MByte	25Gbit
Downlink data rate	Up to 2.25Mbps	Up to 900kbps	~3Mbps
Stabilisation	Spin	Momentum-bias	Spin
Pointing accuracy	0.1° knowledge	<0.5°	5°, 0.1° knowledge
Propulsion	None	None	None
EARTH OBSERVATION	**TOMS-EP[4]**	**GFO[7,8]**	**Case study**
Spacecraft mass	261kg (payload 35kg)	300kg (payload 47kg)	244kg (payload 67kg)
Spacecraft power	275W	126W	381W EOL
Onboard data storage	16MByte	96MByte	25Gbit
Downlink data rate	202kbps continuous via DSN/GSTDN	4Mbps	100Mbps (X-band)
Stabilisation	3-axis	3-axis	3-axis (wheels + magnetorquers)
Pointing accuracy	0.5°, 0.1° knowledge	Up to 15 arcsec knowledge	0.03° knowledge
Propulsion	Hydrazine	Hydrazine	Hydrazine

Table 7-1 Comparison of past mission parameters with those of case study solutions

The spacecraft hardware and software costs would be expected to be broadly similar between the existing spacecraft and the proposed solutions, or slightly higher where there is some additional hardware to give greater performance. However, the shorter schedule time, with the reduced mission-specific design effort required, may be expected to offset this. It would therefore be expected that a spacecraft of enhanced capability could be produced more rapidly for the same cost as the existing mission, and a spacecraft of equivalent performance would be produced at a lower cost.

It must be noted that, even within the mission types defined as falling within the scope of the platform, there will be certain cases where the platform cannot be configured to give a good match. In these cases, the additional time and cost required to modify the basic platform design may make it more cost-effective to design a bespoke spacecraft. It is expected that possible areas in which the platform may have difficulty satisfying mission requirements may be:

- Total spacecraft mass less than ~100kg. The modular design of the platform makes it a very mass-inefficient solution at the lower end of the small satellite class. The payload fraction is too small compared to other commercial platform solutions.
- Accommodation of excessively large/ unusually-shaped payload instruments. The design has been developed to be able to accommodate a much larger range of payload shapes and sizes than many "standard" designs; however, in extreme cases, an instrument may require a dedicated platform to be designed around it.

However, incompatibilities should be highlighted quite rapidly, during the mission-to-configuration matching process, and so time, effort, and money is not wasted by chasing a project for which the platform is unsuitable. One of the strengths of the proposed process is that the decision whether to bid for a particular project can be made more quickly and objectively.

7.2 MISSION-TO-CONFIGURATION MATCHING

The process by which missions are matched to an optimum configuration has been outlined in the previous chapter. It was proposed that the various configuration permutations be integrated into a searchable database, and, further, that this could flag problems with selected configurations. The suggested methodology is therefore a form of expert system, to be used as a decision-making and design aid.

The creation of this expert system would itself involve a significant amount of design and development, in synergy with the further development of the platform. This work would largely need to be implemented before the platform was commercially offered. However, it would be expected that the process would be refined over the course of successive projects, and could be used to incorporate useful "lessons-learned" information that may otherwise be lost.

It is suggested that the expert system be based on those already used in the IT and automotive industries. Within these (and other) industries, there has been considerable research in recent years into design using the *product platform* concept. A product platform is the set of common parameters, features and components that remain constant from product to product, within a given *product family* (related products that share these common features)[9]. For example, in the automotive industry, Audi, Volkswagen, Seat and Skoda cars are all based on only four basic platforms[10]. All the different models available are then built around these common platforms. This is obviously rather similar to the standard modules and sets of subsystems from which the different spacecraft platform configurations can be assembled.

Research into product platform design covers the methods required to model and analyse a common, scalable product platform and its resulting product family. A method has been proposed whereby a mathematical model is used to determine values of design variables that satisfy a set of constraints and achieve potentially conflicting goals. This is known as a compromise Decision Support Problem[9]. This approach would be very applicable to the design and programme proposed here, and is suggested for further research.

7.3 COST SAVINGS AND BENEFITS OF THE MODULAR DESIGN

It has been stated that the modular approach used in the platform design proposed here will give cost benefits to missions using it. The following discussion explores the reasons for this expectation.

It is not suggested that the proposed platform is likely to be lower-cost due to the use of low-cost hardware. Indeed, the actual platform hardware costs are expected to be similar to, or slightly higher than, those of a comparable "one-off" mission (although some cost reductions may be expected as a result of repeat-manufacture of structural parts, and special agreements for repeat-custom with equipment suppliers). It is the programme approach, enabled by the platform design, which allows costs to be reduced.

Ideas of general cost targets for different levels of platform capability were included to give a general guideline for selecting equipment in the detailed design phase. It is expected that, assuming the lowest-cost equipment options are selected for a given performance, the difference in platform hardware cost between the proposed platform and other commercial platforms (or a bespoke platform) will be outweighed by the lower programmatic costs.

Manpower, overheads, and facilities costs often form a significant proportion of the project costs for small spacecraft (often a greater proportion than equipment procurement, for simple missions). As an example, for the Orsted mission, labour costs accounted for 73% of total mission costs, compared to only 27% for procurement[11]. These costs are shown in Figure 7-1 on the following page.

Therefore, a manufacturer using identical hardware to a competitor, but reducing the duration of AIT and the effort required for platform development, would achieve a considerable competitive advantage. Just considering manpower alone, with an average man-hour cost of $30, a team of 20 workers and a working week of 40 hours, a three-month schedule reduction would save nearly $0.3M. Include reduced time required using integration and test facilities, and the cost saving becomes even greater.

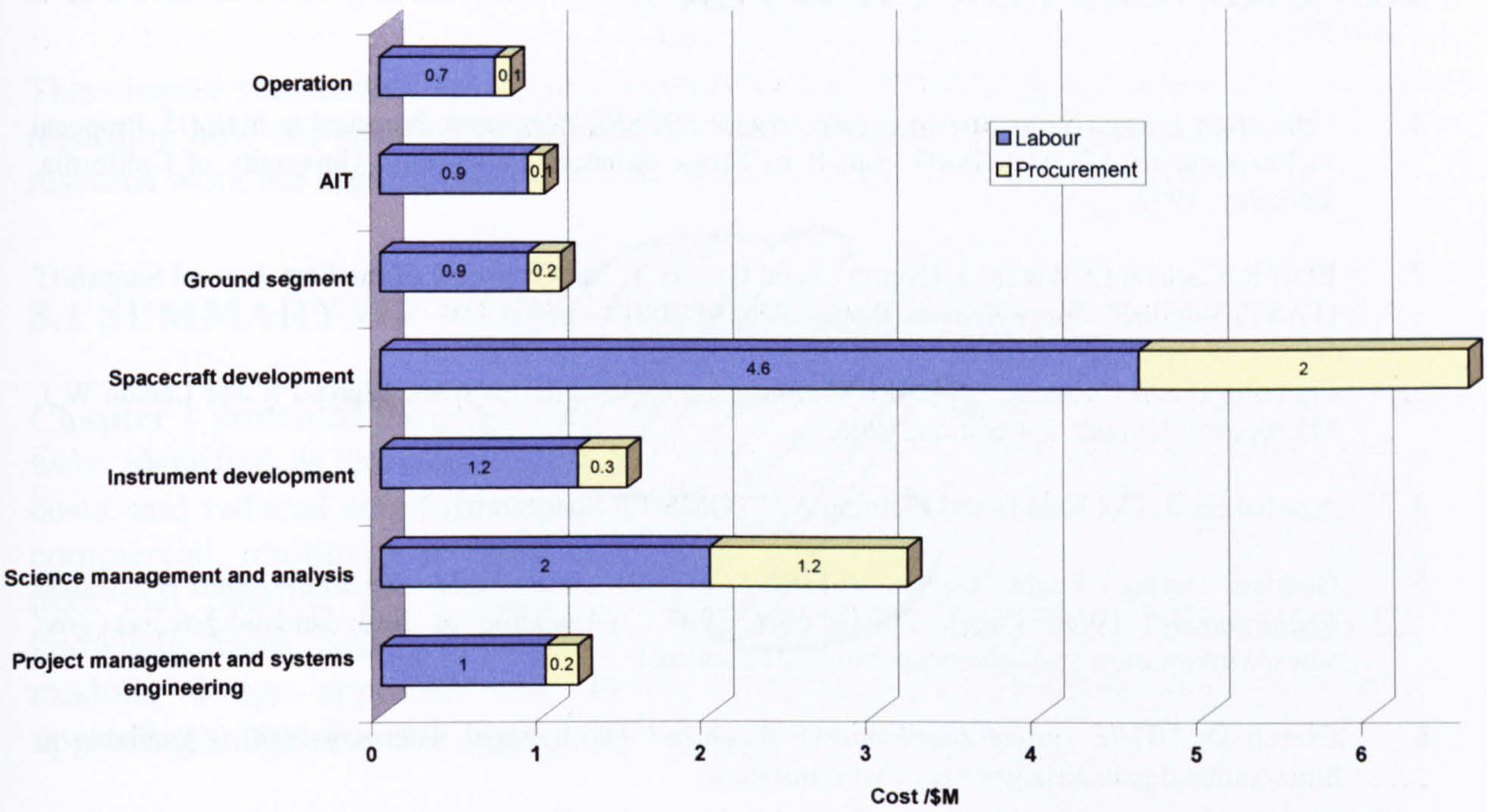

Figure 7-1 Labour and procurement costs for the Orsted mission

The "pre-design" approach also offers manpower savings in the project proposal and mission design and development phases. The costs of the basic platform design are shared across subsequent projects. Continuous lessons-learned knowledge is also accumulated, which may also allow later projects to be produced even more quickly and cheaply.

Of course, the shorter schedules also give a direct benefit, in that advantage can be made of "late-availability" launch slots, which other platforms may not be able to meet. If there is no competition for such a launch, the price is likely to be lower.

The following chapter summarises the work done, and draws together the final conclusions from the research.

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8 SUMMARY AND CONCLUSIONS

This chapter summarises the results obtained in the thesis. Conclusions are then drawn regarding how the stated objectives have been met, and the contributions made by the research work are highlighted.

8.1 SUMMARY OF WORK AND RESULTS

Chapter 1 presented the origin and rationale for the research project. Small spacecraft were identified as providing the key benefits of lower platform costs, lower launch costs, and reduced schedule times, compared to the "traditional" larger spacecraft. A commercial, multipurpose small platform was proposed to give a faster time-to-flight than that possible for a purpose-designed spacecraft, via leveraging the existing knowledge-base, design heritage, and operational/logistics infrastructure. Finally, a modular design approach was shown to offer significant advantages in system upgrading, integration and testing, and configuration adaptability.

Chapter 2 identified the key application areas for small satellites as science, Earth observation, LEO communications, and technology demonstration. An examination of recent missions and international programmes provided a background to the small satellite area, including general design issues. Available commercial small platform characteristics were summarised, and found to range quite widely, with costs between $1m and over $100m, and payload mass capabilities between 20kg and over 500kg.

The chapter went on to address modularity. Investigation of previous modular approaches identified the key issues as:

- Standard interfaces
- Interchangeability of modules
- Separate integration and testing of modules
- De-coupling of payload from platform

The crucial area was found to be interface standardisation. This led to definition of the parameters to be standardised for mechanical, thermal, power, data, and software interfaces. Possible interface positions were then defined, to divide the platform into the desired modular units.

Finally, a number of new technologies were investigated to assess their potential as small mission enablers. The maturities and applications of these technologies were examined, and a number of the technologies suggested as suitable for use on the proposed platform. These were:

- Inflatably-deployed flexible solar arrays
- SMA actuators
- Body-mounted concentrator arrays

- Lithium-ion batteries
- GPS for attitude and orbit determination
- Pulsed-plasma thrusters
- Low-power resistojets
- Fibre-optic data bus

Wireless onboard communications and multifunctional structures were also suggested, for consideration in the near future.

Chapter 3 addressed the commercial aspects of producing a spacecraft platform. The demand side of the industry was first analysed, to identify the likely customer base for the proposed platform, and their requirements. Customers were found to be divided across the civil, military, and commercial sectors. The potential customers and markets identified are summarised in Table 8-1.

	Civil	**Commercial**	**Military**
Communications	Low-cost domestic communications Government messaging Cost range: $2-15m	Point-to-point Store-and-forward messaging Asset-tracking Cost range: $2-15m	Low-cost secure communications Cost range: $5-20m
Earth observation	Low-cost weather satellites Disaster monitoring Resources Cost range: $15-40m	Images for fishing, agriculture industries Cost range: ~$15m	Surveillance Operations support Cost range: $15-50m
Technology	Small demonstration missions to promote domestic industry Cost range: $10-40m		Small demonstration missions to test new systems Cost range: ~$15m
Science	Low-cost scientific research Cost range: $15-40m		

Table 8-1 Summary of potential customers and markets

The chapter continued with an analysis of the characteristics of the space industry itself. This included examination of trends, and estimates for levels of space platform procurement in the different market sectors. Based on these estimates, it was predicted that there would be over twenty missions per year to which a small commercial space platform may be applicable. The anticipated split by mission type is shown in Figure 8-1.

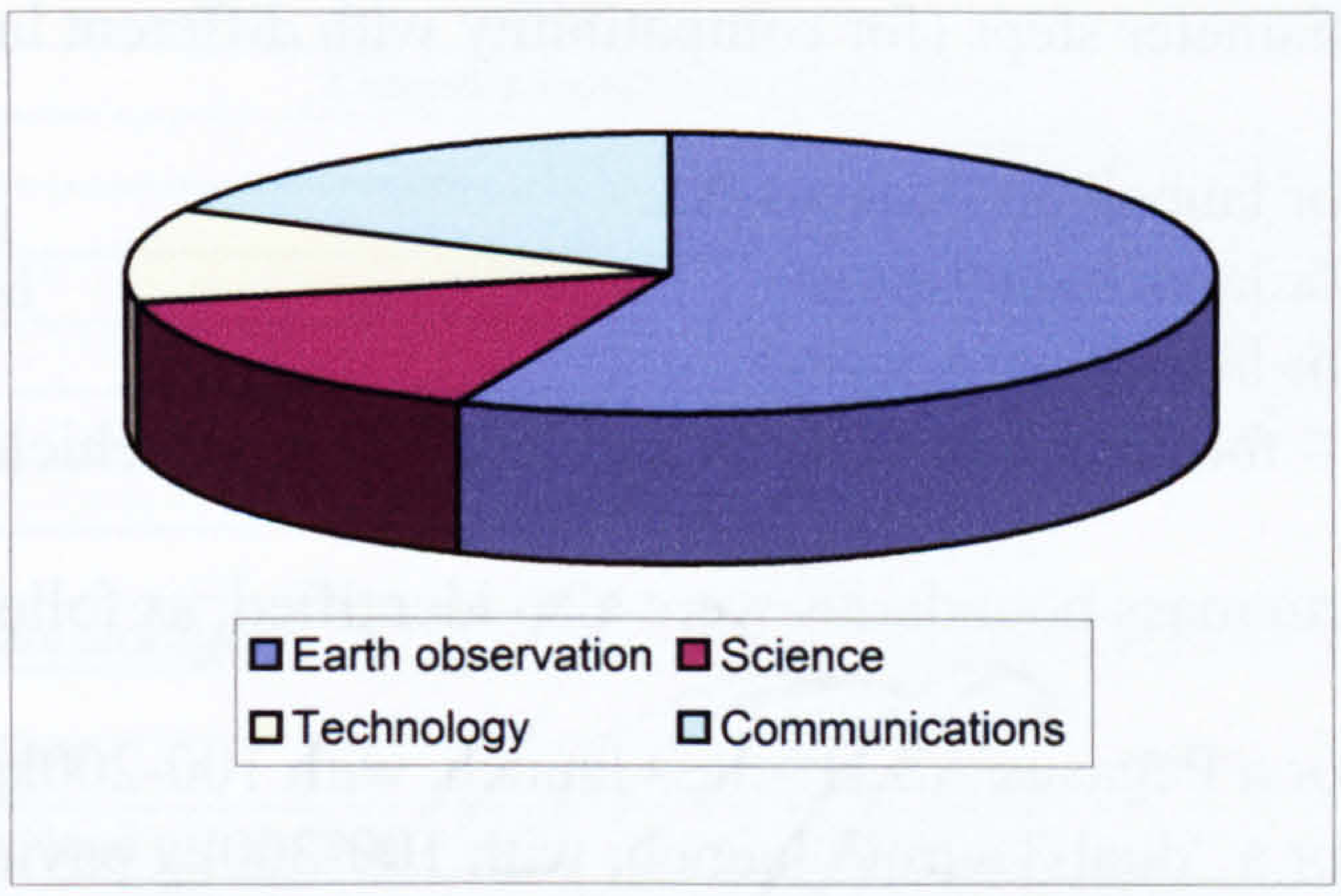

Figure 8-1 Approximate distribution of applications expected within target missions

It was also predicted that Earth observation and communications would show the greatest growth potential, although these markets were identified as rather volatile. The science and technology missions were shown to be at lower levels, but levels that remained fairly consistent.

Finally, the chapter examined the supply side of the industry, and characterised the different spacecraft manufacturer types in terms of their suitability to adopt the proposed platform and programme approach. The types identified were:

- Large multidisciplinary aerospace companies, e.g. EADS
- Large space subsidiaries, e.g. Alcatel Space
- Large-to-medium-sized specialists, e.g. Spectrum Astro
- Small space subsidiaries, e.g. Verhaert
- Small specialists, e.g. SSTL

It was suggested that the "large-to-medium-sized specialist" manufacturer group is most suitable for adoption of the programme, as these are capable of the required up-front programme investment, but still small enough to be able to manage smallsat projects cost-effectively.

Chapter 4 performed a requirements analysis for the multipurpose platform. The specific requirements of different mission types were first addressed, with missions being separated out into the categories of astronomy, space physics, communications, and Earth observation.

Microgravity mission requirements were examined, but it was decided that these were likely to form such a small proportion of target missions, and had such a specific requirement set, that they would be disregarded. Technology missions were also examined, but it was decided that these could be considered as special cases of the four mission types outlined above, and so did not have a separate requirement set.

The general requirements for the platform as a whole were also addressed, resulting in proposed platform diameter steps (for compatibility with different launchers) of:

- 1100mm – for launch on Pegasus-XL
- 1300mm – for launch on Taurus
- 1500mm – for launch on ASAP-5
- 1900mm(+) – for launch on Athena and larger-fairing vehicles

Approximate platform mass boundaries were also identified, as follows:

- 100-150kg for a Pegasus/ASAP-class launch, with 100-200kg payload
- 200-300kg for a "dual-Taurus" launch, with 100-200kg payload
- Around 400kg for a dedicated Taurus launch, with up to a 500kg payload

Some approximate cost ranges for the platform were also proposed, based on previous missions, and civil budgets:

- "Basic" platform, lowest performance level: $5-10m
- Higher capability, larger payload mass, higher power: $10-20m
- Advanced platform, with "mission tailoring": $20-25m

However, it was not considered very meaningful to design the platform around cost, as its hardware costs will be similar to an equivalent "bespoke" spacecraft, as similar equipment will be used. The cost savings mainly arise from the different programmatic approach, which uses more efficiency to design, assemble, and test the spacecraft.

Finally, a delivery schedule target of 18 months was proposed, based on launcher mission cycle times, and the desired ability to co-manifest on a launch at a late stage. This schedule is also highly competitive compared to other commercial platforms.

Chapter 5 described the system design process for the proposed platform. A configuration concept was selected, from a number of possible candidates and design iteration. The configuration uses a number of box-modules, which attach to form an effective thrust tube structure at the centre. Payload may be mounted on the top of the platform, within the central "thrust-tube" volume, or within the modules themselves. The modules can be assembled into different configurations, to fit different launch envelopes, and accommodate different payloads.

The chapter then covered the platform subsystems, assessing the options available, and selecting a set of subsystem equipment to give different platform "capability variants". The levels at which the equipment should be divided to form separate modules was also addresses. Summaries of the mass and power budgets for the different subsystem variants are given in Table 8-2.

Variant	Subsystem mass and power estimates
Communications subsystem	
Basic	2.5kg, 10W (30W when transmitting)
Basic with ranging	3kg, 10W (30W when transmitting)
High-power S-band	3.2kg, 10W (50W when transmitting)
High-gain S-band	7.2kg, 10W (30W when transmitting)
High-performance	8.3kg, 15W (35W when tranmsitting)
Data handling subsystem	
Basic	8.5kg, 15W
Basic with extra data storage	9.25kg, 45W (for 100Gbit storage)
Redundant	14.5kg, 20W
Payload processing	13kg, 18W
Redundant with paylaod processing	19kg, 23W
Attitude determination and control subsystem	
Baseline	22kg (8kg)*, 175W (43W)
High accuracy	23.5kg (8.5kg), 196W (62W)
High manoeuvrability	28kg (16kg), 188W (58W)
Spin stabilised	10kg, <5W
Momentum bias	11.5kg, 34W
Thermal control subsystem	
Baseline passive plus heaters	2kg, <5W
Power subsystem	
Basic (1.5m^2 body-mounted)	12kg
Basic w. 1m^2 deployed-fixed panel arrays	18.25kg
Basic high power deployed-fixed flex-arrays 5m^2	31.5kg
Complex high power articulated panel 5m^2	~33kg
Propulsion subsystem (dry mass)	
Cold gas	4kg
Hydrazine	10kg
Resistojet	1.25kg
Ion thruster	25kg

*Lower mass/power option in brackets – for smaller mission configurations, using smaller wheels/rods

Table 8-2 Summary of mass and power estimates for the different subsystem capability variants

The accommodation of onboard equipment, and sizing of the platform modules for launcher compatibility was also addressed. The modules were sized such that each module could be used in a "short" or "tall" configuration, with a tall and narrow three-module platform sized to fit the Pegasus-XL fairing. For a Taurus launch, a four-module configuration, with the modules in a tall, narrow orientation, could be accommodated. Alternatively, a three-module configuration with the modules in a "short, wide" orientation gives a larger volume available to the payload. The modules could also be configured to fit an ASAP-5, or larger-fairing launchers. It was suggested that all the platform subsystems could be accommodated within three modules, leaving a "spare" module on the larger configurations, which could then be used for payload.

Finally, programmatic issues were examined. The effect of the proposed platform and programme on schedule were addressed, and significant schedule reductions were projected. A baseline project schedule of less than 20 months from approval to spacecraft delivery was estimated, for a "first time" use of a particular platform configuration. This schedule was expected to reduce with increasing lessons-learned and familiarity of the project team.

Chapter 6 addressed the process of matching a proposed mission to a suitable platform configuration. A process was proposed, by which a candidate mission is characterised via a set of standard parameters. These parameters are then compared to an equivalent set of parameters, produced for each of the possible platform configurations. A database system is suggested for this process. It is anticipated that such a process would allow improved quality and rapidity for response to ITTs. The use of a User Manual for the platform, analogous to those supplied by launch vehicle providers, is also suggested.

Finally, the chapter used three case study missions, x-ray astronomy, magnetospheric physics, and Earth observation, to illustrate application of the platform to different mission requirements.

Chapter 7 discussed the results obtained in applying the platform to the case study missions, and addressed the areas in which expected cost savings are expected to be made. It also expanded further on the application of an expert system to handle the complexity of matching specific missions to appropriate platform configurations.

8.2 CONCLUSIONS

The stated objectives of this thesis were:

- **To analyse the technical and commercial requirements for a modular, multipurpose small satellite platform**

- **To produce a technically and commercially valid system-level design for such a platform**

The technical requirements included the following:

- General requirements for a small spacecraft
- Requirements for multi-use, i.e. requirements for a range of different mission types
- Requirements for modularity

The research work has covered all of these areas, culminating in the detailed set of requirements described in Chapter 4. These requirements were sufficient to allow a system-level design study to be performed for the platform. While it is not possible to anticipate the exact requirements of future missions, it is believed that the requirements envelopes derived would encompass enough missions to be a valid basis for the design work.

The commercial requirements analysis has produced a detailed study of the small satellite market, with projections for the anticipated numbers of missions per year for which the proposed platform could compete. These missions have been identified by mission type and by customer type (military, civil, or commercial). The characteristics of the different markets have been identified, in terms of their volatility (i.e. potential risk were they made the sole focus of the platform), and in terms of their likely growth. The work has shown a continuing requirement for small spacecraft platforms, which can be met by the type of platform proposed here.

The thesis also provides an outline process for tackling the problem of designing a generic product. It proposes the identification of all the possible customers, all of their likely requirements, and all of the other competitors in the market. It then narrows down the area of concern by:

- Defining an envelope of target customers, and assessing the size of this market
- Attempting to pre-empt the requirements that will be demanded from these customers
- Identifying a key area or areas with which to gain advantage over the competitors

For the proposed programme, the key areas where advantage can be gained are in schedule reduction, and adaptability to a range of different requirements

The proposed system design for the platform meets the objectives of being modular and multipurpose, and also meets the derived technical and commercial requirements. The key benefits offered by the proposed design are:

- Adaptability to different payload sizes and configurations
- Adaptability to different launch vehicles
- Rapid response to mission requirements
- Easy conversion to different capability levels
- Ability to perform much of the AIT phase with each module in parallel
- Ability to pre-qualify the configurations using a standard "mass dummy kit"
- Suitability for future upgrade and incorporation of new technologies
- Ability to make use of lessons learned for improvements on future projects, due to commonalities between platform configurations and equipment

The primary commercial advantages given by these benefits are:

- Improved detail, accuracy, and response time to customer ITTS (to make it more likely that a bid is successful)
- Design effort shared across many projects
- Reduced project schedule times

Cost reductions then occur as a consequence of the schedule reductions, as discussed in Chapter 7.

In summary, it is believed that the platform and programme approach presented in this thesis provides a good baseline for a commercial product, and meets the objectives set out at the beginning of the research project.

8.3 SUMMARY OF CONTRIBUTION OF THE THESIS

The main contributions arising from this research thesis may be summarised as:

- Detailed characterisation of typical requirement sets for different small satellite mission applications
- Identification of applicable markets, market sizes and growth scenarios for the proposed platform
- Identification of commercial customer requirements for the target markets
- System-level design of a modular, multipurpose spacecraft platform, which can be applied to the range of requirements identified
- Proposal of a programme approach that will reduce schedule times and provide competitive advantage

The specific requirements derivations, particularly for enabling modularity and multi-use, should also be useful as discrete studies in their own right. They may be applied to different categories of spacecraft, as they are not necessarily specific to smallsats.

8.4 SUGGESTIONS FOR FURTHER WORK

The baseline system-level design of the platform needs to be extended into a more detailed design. A detailed structural analysis and optimisation will be particularly useful, as it is believed that the current design over-estimates the required structural mass. The inter-module fastening scheme requires investigation.

After more detailed design specifications are produced, the mission-to-configuration matching methodology can be addressed in more detail. As discussed in Chapter 7, the application of product platform concepts to the spacecraft would offer an interesting extension to this work. It is believed that this method has not yet been formally used in spacecraft platform design (although the underlying philosophy is largely that used in the research presented here). Further research in this area would allow these design engineering techniques from other fields to be applied to spacecraft, and make them accessible as tools within the space industry.

APPENDIX A: World small satellite missions 1999-2002 [Spacecraft listed in launch order][1,5]

Spacecraft	Mission/ payloads	Mass	Cost	Launcher	Orbit (km)	Power	Description	Country	Manufacturer	Delivery
					1999					
ROCSAT	Space science, plus oceanography, Ka-band comms & ionospheric experiments	410kg	US$83m	Athena	588x601, 35 deg		Hydrazine propulsion, deployed solar panels	Taiwan	TRW (USA)	
Globalstar	Global communications	450kg	US$47.5m	Soyuz & Delta	1414	1100		USA	Space Systems Loral	18 months
WIRE	IR telescope for galaxy evolution study	259kg	US$45m	Pegasus-XL	539x593km, 97.5deg	158		USA	NASA GSFC	36 months
UoSAT-12	Earth imaging & digital comms technology demonstration	312kg	US$9m (platform)	DNEPR	638x654, 64.55deg	300W		UK	SSTL	
Abrixas	All-sky X-ray survey	470kg	DM40m	Cosmos-3M	544x603, 48.4deg	200W		Germany	OHB, ZARM	
Shi-Jian-V	Attitude control technology demonstration	350kg		Long March 4B	870km sunsynch.			China	Shanghai Bureau of Astronautics	
TERRIERS	EUV & radio tomography	121kg	US$12.3m	Pegasus-XL	537x552, 97.8deg, 9am-9pm	31W		USA	Bus AeroAstro, instruments Boston & Illinois Univ's.	
KitSat-3	Multispectral camera, science experiments	107kg		PSLV-C2				Korea	KAIST	
KOMPSAT	Ocean colour monitoring, ionospheres & particle physics	510kg		Taurus	685km, sunsynch.	630W	Based on TRW Eagle-class-2 bus	S. Korea	KARI (based on $75m TRW test model)	
ACRIMSAT	Solar irradiance measurement	115kg	US$26m + US$6m towards launch (platform US$8.3m)	Taurus (shared with KOMPSAT)	683x724, 98.3deg, 10:50am	62W	Based on MiniStar bus	USA	Orbital, instrument JPL	24 months
					2000					
JAWSAT	Deployment of 4 independent microsatellites, plus imaging & plasma experiment	191kg		OSPSLV		80W		USA	Orbital	
Globalstar	See 1999									

TSX-5	Military vibration-suppression & IR aircraft imaging technology demonstration	250kg	US$85m	Pegasus-XL	406x1706, 69deg	183W	Uses LEOStar bus	USA	Orbital	
Champ	Atmospheric, ionospheric physics; gravitational, magnetic field research	522kg		Cosmos-3M	460x460, 87.3deg			Germany	Jena-Optronik	
Mita-0	Technology demonstration	170kg		Cosmos-3M (shared with Champ)	460x460, 87.3deg	75-95W	Based on SAGE bus	Italy	Carlo Gavvazio Space	
MightySat-2.1	Technology demonstration, inc. concentrator cells, ultra-light comms, multifunctional structures and bimodal composites	130kg	US$23.5m for 5 buses	OSP Minotaur	548x585, 97.8deg	330W		USA	Spectrum Astro	24 months
HETE-II	X-ray astronomy, gamma ray bursts	150kg	US$18.4m + US$14m for launch	Pegasus-XL	637x650, 1.8deg		Partly uses systems left over from HETE	USA	MIT	
AMSAT Phase-3D	Amateur Radio	~400kg		Ariane 5	4000x47000, 63.4deg			International	AMSAT	
STRV-1c, 1d	Technology demonstration	100kg each		Ariane 5 (ASAP)	600x35786, 7deg, (GTO)			UK	DERA (QinetiQ)	
EO-1	Land imager, multispectral imager, plus technology demonstrators including PPT, x-band antenna, lightweight flexible solar array	529kg	US$193m (inc. development, launch, 1yr operations)	Delta-2	705km, 98deg	300-600W	Aluminium, Hexagonal Right Prism - 1.25m across flats, 0.73 m high	USA	Swales Aerospace	
SAC-C	Multispectral camera, panchromatic camera, GPS occultation experiment	467kg	US$30m (spacecraft) US$45m (inc. instruments?)	Delta-2 (with EO-1)	698x1800,	450W		Argentina, + collaboration from Brazil, Denmark, France, Italy, USA	INVAP (for CONAE)	
2001										
ODIN	Astronomical research, search for interstellar water & oxygen, study ozone depletion	250kg	US$40m	START-1	606x630, 98.73deg	340W		Sweden, + collaboration from Canada, Finland, France	Swedish Space Corp.	24-36 months
ORBView-4	Earth imaging	368kg		Taurus	Failed			USA	Orbital	
QuikTOMS	Study of ozone depletion	168kg		Taurus (shared with ORBView-4)	Failed			USA	Orbital	

JASON-1	Sea-surface altimetry, atmospheric radiometry	500kg		Delta-2	1320x1320, 66deg		Based on Proteus bus	France/USA	Alcatel	
COSMOS 2384-6	Military data messaging & reconnaissance			Tsyklon SL-14	1415x1447, 82.5deg		Based on Strela-3 bus	Russian Federation		
GONETS 12, 13, 14	LEO data comms, disaster monitoring & alerts	231kg each		Tsyklon SL-14 (shared with Cosmos, above)	1415x1447, 82.5deg		Based on Strela-3 bus	Russian Federation (SMOLSAT)	NPO Applied Mechanics	
2002										
MDS-1	Technology demonstration of commercial components in space	449kg		H-2A	500x35696, 28.5deg		Box structure, 3.3x1.6m with deployed solar arrays	Japan	NEC	
RHESSI	Study of solar flares via gamma ray & neutron imaging	304kg	US$12m (spacecraft)	Pegasus-XL	600x600, 38deg	110W	Based on SA-200S bus	USA	Spectrum Astro	36 months
GRACE	2 spacecraft, to map Earth's gravitational field.	432kg each	US$120m (NASA cost cap) Platform US$9m each?	Rokot	500x500, 89deg		Based on Dornier Flexbus	USA/ Germany	Dornier (Now Astrium Germany)	36-48 months
Haiyang-1	Ocean colour scanning	340kg					Similar bus to Shi-Jian-5	Republic of China	Chinese Academy of Space Technology	
Ofek-5	Reconnaissance	300kg		Shavit	370x600, 143deg		2.3mx1.2m diameter	Israel	Israel Aircraft Industries	

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APPENDIX B: Spacecraft system functional breakdown

Input	Input Source	Function	Output	Output Destination
		Attitude determination & control		
Photons	Sun, stars, Earth	Identification of target and target direction	Target vector(s)	AOCS processor
Magnetic field	Earth	Identification of field lines and field line direction	Magnetic field vector	AOCS processor
Target vector data Attitude drift measurements	Attitude sensors Inertial reference sensors	Calculation of spacecraft attitude by relation of sensor vector data to spacecraft co-ordinate axes, or propagation of inertial changes	Attitude data	AOCS processor HK telemetry Payload instruments, if required
Attitude data	AOCS processor	Storage of time-labelled attitude data	Time-history of spacecraft attitude	AOCS processor As required
Attitude data Requests for desired attitude	AOCS processor Telecommands Pre-defined attitude requirements	Comparison of actual attitude with desired attitude	Attitude error measurement Required attitude change	HK telemetry Payload instruments, if required AOCS processor
Required attitude change	AOCS processor	Calculation of required actuator operations, using spacecraft control laws	Actuator control	Actuators
Actuator control	AOCS processor	Generation of attitude correction torques	Change in spacecraft attitude	-
Response to actuator input	Attitude sensors	Comparison of actual response with expected response. Identification of possible anomalies	Error flag	HK telemetry AOCS processor
Wheel speed data	Reaction/ momentum wheels	Comparison of current speed with maximum speed. Identification of whether momentum dumping is required	Instruction to perform momentum dumping, if required	Gas thrusters, magnetorquers
Indication of imminent deployment operation/ planned motion	Telecommands	Identification of expected compensation torques required (by calculation or look-up) Confirmation that compensation can be achieved (e.g. if momentum dumping must be performed first)	Actuator control Status confirmed as ready	Actuators HK telemetry

Orbit determination & control				
Ranging data (range, azimuth, elevation vs time)	Ground Station (data taken during passes)	Calculation of spacecraft orbit Propagation from updated orbital elements Update of orbit ephemeris tables	New orbit ephemeris	AOCS processor HK telemetry Ground station
Orbit ephemeris Required orbital elements	AOCS processor/ Ground station	Comparison of actual orbit to required orbit If not within acceptable boundaries, calculate delta-v required for correction	Delta-v requirements	Actuators (thrusters)
Orbit ephemeris	AOCS processor/ Ground station	Calculation of ground pass timings and geometries Calculation of antenna pointing/tracking requirements	Ground pass predictions Antenna pointing instructions	Ground station and spacecraft
Orbit ephemeris	AOCS processor	Calculation of sun vector Calculation of solar array pointing requirements	Pointing instructions to solar arrays, as required	Solar array pointing mechanisms
Orbit ephemeris	AOCS processor	Calculation of magnetic field vector (predicted from field models), if magnetometers are used for attitude determination	Magnetic field vector data	AOCS processor
End of life indication	Telecommand	Prepare for de-orbit/graveyard orbit insertion manoeuvre	Actuator control	Thrusters
Data handling & onboard communications				
Analogue data	Sensors	Analogue to digital conversion	Digital data	DHS
Raw digital source data	Payloads, sensors, subsystems	Data sampling, as per bandwidth allocation strategy	Source packets	DHS buffers
Source data	DHS buffers	Addressing of temporally- and type-labelled data to particular memory locations	Data	Memory
Source packets	DHS buffers	Application of compression algorithms Error detection and correction Screening of bad data Addressing compressed data to particular memory locations	Compressed source data packets	Memory

Source packets Other data, such as telecommand acceptance reports	DHS buffers/ memory	Combination of source packets and data into defined order Inclusion of frame headers, trailers, identifiers, synchronisation bits as required Addition of error-detection codes Encoding e.g. Reed-Solomon, as required	Telemetry frames	Communications subsystem
Attitude/ orbit data	AOCS	Switch to different data rates as required (e.g. due to loss of attitude, link considerations at apogee)	Data rate specification	DHS
Clock pulse	Oscillator	Calculation and tracking of Mission Elapsed Time (MET)	MET Clock pulse	Subsystems, payloads, HK telemetry On-board systems
		Spacecraft command & control		
Telecommands	Communications subsystem	Verification and acceptance of commands	Command acceptance report	HK telemetry
Telecommands	Communications subsystem	Interpretation of command type (pulse commands, proportional commands, time-tagged commands) Transfer of command data to required locations	Command data	Subsystems/ payloads
"OK" input	Processor	Watchdog countdown timer re-set	No output (nominal case)	
No input (anomaly)		Countdown timer times out On-board decision made from stored IF-THEN contingencies	Remedial actions as required e.g. processor re-boot	As required
Housekeeping data	DHS buffers/ memory	Autonomous verification and control made from stored IF-THEN contingencies	Remedial actions as required e.g. go to safe mode	Subsystems/ payloads as required
		Electrical power		
Photons	Sun	Conversion of incident solar radiation into electrical power, via photovoltaic solar cells	Electrical power	Power distribution/ conditioning system Shunt regulator Battery charge regulator
Electrical power	Battery charge regulator	Regulated charging of batteries, prevention of over-charging	Stored electrical power for use during eclipse or periods of high power consumption	As required

Electrical power	Batteries	Regulation of battery discharge, prevention of over-discharge		
Battery voltage data	Voltage sensors	Monitoring of battery voltages	Battery housekeeping data	DHS, HK telemetry
Battery current data	Current sensors	Monitoring of battery currents	Battery housekeeping data	DHS, HK telemetry
Battery temperature data	Thermistors	Monitoring of battery temperatures	Battery housekeeping data	DHS, HK telemetry
Pointing instructions	Solar array pointing/ deployment mechanisms	Alignment of solar arrays to face the sun direction/ provide desired power generation level	Nominal power level	Power distribution/ conditioning system
Electrical power	Solar arrays	Dissipation of excess power by means of voltage/ current shunts (required particularly at BOL)	Nominal power level	Power distribution/ conditioning system
Main bus voltage	Main power bus from power distribution system	Power conversion to provide required voltages to individual loads. May also provide protection against inrush currents.	Regulated power supply	Individual loads as required
Main bus voltage	Main power bus	DC-DC conversion to remove noise present on main bus, where loads are susceptible to noise	Regulated power supply	Individual loads as required
On-off commands	DHS	Switch power on or off to loads, such as payloads or subsystems	Relay switching	
		Thermal control		
Temperature data	Thermistors situated around the spacecraft	Compare actual temperatures with required temperature boundaries for each region Identify temperature trends	Raise error flags if temperatures out-of-limit, or trend is tending towards out-of-limit	DHS, HK telemetry
Requirement for temperature control identified	DHS, telecommand	Application of control laws to determine necessary actions (if active thermal control is used)	Commands	Thermal actuators (heat transport pipes, heaters, refrigerators, louvres, sun-shades etc)
Indication of predicted change in thermal balance (e.g. if spacecraft is about to enter eclipse)	DHS, telecommand	Required pre-emptive actions determined	Commands	Actuators (as above)

Communications				
Telemetry frames	DHS	Carrier generation Subcarrier generation from telemetry data Modulation of carrier with subcarrier	Transmitter signal	Power amplifier
Transmitter signal	Transmitter	Amplification of signal to required levels	Amplified signal	Waveguides
Amplified signal	Power amplifier	Transmission of signal via waveguides Selection of antenna(s) e.g. high-gain, low-gain, by commanding waveguide switches	Amplified signal	Antenna
Amplified signal	Waveguides	Production of radiated RF signal to required direction, with required gain	RF signal	Ground station receiver
RF uplink	Ground station transmitter	Detection by receiver	Receiver locked on to uplink	
Uplink	Receiver	Down-conversion	Lower-frequency signal	Demodulator
Signal	Down-converter	Demodulation	Uplink data	DHS
Indication of loss of spacecraft attitude	AOCS, via DHS	Switch to omnidirectional antenna to maintain communications	Switch controls	Waveguide switches
Ground segment				
RF signal	Spacecraft	Acquisition of spacecraft, and locking-on to signal Tracking of spacecraft	Pointing instructions	Ground station antenna
Laser ranging pulse	Tracking station, reflected by spacecraft	High-accuracy orbit determination	Accurate orbit ephemeris	Spacecraft, ground station
Telemetry data downlink	Spacecraft	Reception by ground station receivers Downconversion Demodulation	Mission data for decoding and analysis	Data recovery systems
Mission data	Reception systems	Extraction and separation of different data types (payload and housekeeping/ engineering data)	Payload data Housekeeping data	Payload operations centre Spacecraft operations centre

Housekeeping data	Data interface	Evaluation of engineering parameters Health and status monitoring Fault detection Decisions taken regarding actions to be taken Archiving of data	Telecommands as required	Telecommand uplink equipment
Command requests	Spacecraft operators Payload PIs	Verification of suitability of commands/ operations strategy Generation of command sequence Verification of command sequence	Telecommands	Telecommand uplink equipment
Payload data	Data interface	Dissemination of data to payload operations centre, end users, data archiving	Payload data	All payload data users
Payload status reports	Payload operations centre	Spacecraft operations centre kept informed of status of payloads, anomaly events, requests to change nominal operations plans	Changes to nominal operations plan if required	
Uplink status report	Spacecraft	Re-sending of commands if they have not been correctly received on board	Telecommands	Telecommand uplink equipment
Spacecraft ephemeris	Tracking systems	Analysis and planning of spacecraft passes over ground station, AOS/LOS times, predicted paths	Ground pass operations schedules	Ground station systems and staff
Spacecraft anomaly situation	Detected in housekeeping data	Reference to spacecraft manuals, contingency plans, spacecraft simulators Consultation with subsystem experts Generation of remedial operations plans and commands	Telecommands	Telecommand uplink equipment
Ground station anomaly situation		Reference to ground station manuals, expert consultation, avoidance of risk to spacecraft	Remedial actions	Ground station systems, as required

APPENDIX C: Spacecraft manufacturers

Company	Country/ countries	Product range	Size	Heritage	Remarks
AeroAstro	USA	Microsatellites, nanosatellites, miniature subsystems & components	40 staff (1995)	Founded 1988. 3 microsatellites built, plus many subsystems	Independent
Alcatel Space	France	Small satellites, large GEO spacecraft, military spacecraft, payloads, subsystems, ground systems	6000 employees, revenues (2000) EURO1.4bn	Over 30 years experience, many spacecraft produced, contracts for many large organisations such as ESA, CNES, Eutelsat, Intelsat. Producing small satellite platform in association with CNES.	Part of Alcatel: 130, 000 employees, sales EURO31bn
Alenia Aerospazio	Italy	Small satellites, GEO comsats, subsystems, instruments, science spacecraft, space station components	2800 employees	30 years' experience, prime contractor for Italian Space Agency missions & many ESA missions	
Astrium (EADS)	UK, France, Germany, Spain	Small and microsatellites, Earth observation, science and communications spacecraft, military programmes, launch vehicles, ground systems, space station components	8000+ employees, turnover $1.5b	Prime contractor for many ESA missions, 50+ comsats, several small spacecraft	Part of EADS (European Aeronautic Defence & Space Company)
Ball Aerospace & Technologies Corp	USA	Spacecraft, subsystems, cryogenics, payloads, remote sensing services, tracking systems	2200 employees, sales $363m	Founded 1956, subsidiary of Ball Corporation. Many instruments flown on NASA missions, several spacecraft flown including one using the commercial platform BCP-3000.	Part of Ball Corporation Smallest spacecraft platforms are 700kg+
Boeing	USA	Large spacecraft, space station, space shuttle, launchers, aircraft, defence systems, communications, missiles.	Largest aerospace company in the world. 198,000 employees. Revenues $51bn (2000)	Has produced space systems since the start of the space age.	Does not commercially offer small satellites, but is involved in the integration of a microsatellite for the Air Force Research Lab.

Company	Country/ countries	Product range	Size	Heritage	Remarks
Carlo Gavvazio Space	Italy	Small satellites, payloads, ground stations	130 engineers	20+ years experience in space technology, first small satellite launched in 2000, various payloads produced including for space station.	
Dornier (EADS)	Germany	Small satellites, Earth observation & science spacecraft, space technology, defence & civil systems, telecommunications, aviation	4000 employees	Dornier System established 1962	Part of EADS (European Aeronautic Defence & Space Company)
INTA (Instituto Nacional de Tecnica Aerospacial)	Spain			1 Minisat flown	National institute – works with space agency, CASA.
Kayser-Threde	Germany	Small spacecraft, subsystems & components. Also aerospace, scientific & industrial systems	300 employees, DM100m	Founded 1967, various systems flown	
Lockheed Martin Space Systems	USA	Small spacecraft, GEO commsats, military spacecraft, large science missions, subsystems, ballistic missiles, space station components	130,000 employees, turnover $25b (in all of Lockheed Martin)	750 successful spacecraft (heritage from founding companies)	
MegSat	Italy	Small satellites, solar arrays, subsystems, ground stations		1 MegSat test satellite flown.	
Mitsubishi Electric Corporation	Japan	Comms & Earth observation satellites, subsystems, ground systems, payloads	Large	Prime contractor for several spacecraft, including first Japanese domestic commsat in 1977. Worked with Loral, Aerospatiale & Lockheed Martin.	
NEC	Japan	Satellites, ground systems, subsystems, payloads	Large	43 satellites integrated by 1996	Integrated Japan's engineering test minisatellites (ETS series)
OHB System	Germany	Small satellites, payloads, re-entry technology, subsystems & components, ground equipment	120 employees	Equipment flown on Shuttle & Mir, several small spacecraft contracts.	

Company	Country/ countries	Product range	Size	Heritage	Remarks
Orbital Sciences Corp	USA	Small satellites, launch vehicles, transport management systems	$600m (1997) sales, several thousand employees	Founded 1982, nearly 90 satellites built & launched.	Joint ventures ORBCOMM (data communications) & ORBIMAGE (Earth imaging)
Space Systems/Loral	USA	GEO comms & weather satellites, Globalstar small comms satellite constellation, payloads	3100 employees, sales $1bn	Joined Loral Space & Communications in 1990. Built spacecraft since 1960, more than 200 spacecraft designed, built, or in progress.	Mainly more involved in larger spacecraft.
Spectrum Astro	USA	Small satellites, subsystems & components, ground systems	420 employees, revenues $120m (2000) Employee-owned	Founded 1988. Spacecraft produced for US military & NASA. Further spacecraft studies on-going for US military.	
Swales Aerospace	USA	Small spacecraft, science spacecraft, support to shuttle program, payloads	900+ employees, revenues $95m (2000) Employee-owned	Founded 1978. Prime contractor for NASA New Millennium EO-1 spacecraft.	
Swedish Space Corp	Sweden	Small satellites, microsatellites, subsystems & components, experiments, payloads, sounding rockets, ground systems, airborne remote sensing systems	300 employees, government-owned, turnover SK340m (2000)	Established 1972. Prime contractor for 4 satellites, produced many payloads.	
SSTL	UK	Microsatellites, small satellites, nanosatellites, subsystems, ground systems, technology transfer	102 employees, turnover £5.7m (1997/98)	Formed 1985. 14 missions launched over 20 years	
TRW	USA	Small satellites, Earth observation & science spacecraft, military defence & intelligence systems, IT, laser systems, telecommunications, automotive systems	102,878 employees, sales $17.2bn (2000)	Nearly 200 spacecraft built, since launch of TRW's Pioneer 1 in 1958.	
Verhaert	Belgium	Small satellites, payloads, subsystems, space instruments, IT terminals, process monitoring & automation systems, new materials	150+ employees	Formed 1969. Producing the ESA PROBA spacecraft.	