Modularity as an Enabler for a More Efficient Commercial Small Satellite Program

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Abstract

This paper addresses the key requirements for a commercial small satellite platform, and the specific problems encountered in effectively adapting it to a range of mission types. A modular concept is proposed, which may be reconfigured to more closely match the requirements of each specific mission, whilst minimising the required redesign time and associated cost. This concept enables a programmatic approach where the level of *a priori* design, manufacture, and test of the platform is maximised. The benefits offered by this approach are shown to be the ability of the platform supplier to respond more rapidly and competitively to contract bids, coupled with a significant reduction in spacecraft delivery times.

Introduction

Small satellites are now well-recognised as offering the dual benefits of low cost and reduced schedule times. As a result, the market for small commercial platforms is becoming highly competitive, with these key features of cost and schedule time being critical success factors. In addition, these features must often be incorporated into products suitable for a wide range of mission types and performance levels.

A major obstacle to further cost and schedule reduction in missions based on multipurpose platforms is the degree of tailoring required to match the platform to the mission specifications. Cost and time-to-delivery are minimised by a greater state of readiness, in terms of design, manufacture and test status, of the platform to be used. However, the greater the level of pre-design of a platform, the more difficult it is to make it suitable for a particular mission.

Furthermore, some commercial platforms available are based on a design used for an original "parent" spacecraft and mission. This makes sense, as significant design and development work has already been invested. Subsequent similar platforms can always be produced more quickly and cheaply, due to design heritage and advance knowledge regarding procurement and supply chain issues. However, this approach does have a potential drawback: The design will almost inevitably be optimised to the type of mission flown by that original parent satellite.

Instead, the proposed approach is to invest in design and development of a deliberately non-specific platform. This may be more easily tailored to a wider range of potential missions. The design brief is compiled by examination of the requirements of many missions, and attempting to pre-empt the likely requests of future customers. The philosophy is then to maximise the level of *a priori* design, manufacture, and test of the platform, via the use of a range of modules. The modules may be pre-assembled and tested, and an appropriate platform configuration produced largely off the shelf. This then allows a rapid response time, both to initial invitations to tender for spacecraft contracts, and to the actual spacecraft build itself.

Proposed Approach

As described in the introduction, the proposal is for a modular small satellite platform, offered via a commercial supplier program. Modularity is considered to be a key element for success for any multi-purpose, multi-customer product. 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. These are discussed as follows:

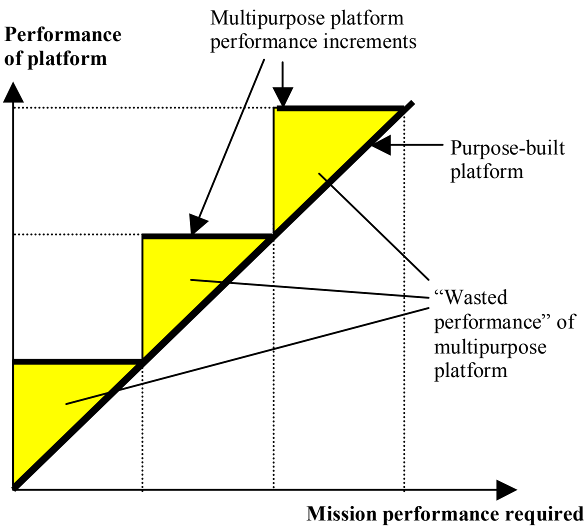
System upgrading
If all missions using a commercial, multipurpose spacecraft had the same set of requirements, there would be little benefit to designing a platform to be modifiable or upgradeable; a single design that met the requirement set would suffice. However, requirements vary widely (as will be shown later), 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 by 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. This is illustrated in Figure 1.

Where the increments are large, there is a great deal of capability or performance that is not necessary, but that still must be paid for. If there is too much wasted performance, it may be cheaper to produce a purpose-built platform that exactly matches the required performance.

Figure 1 Platform Performance vs Required Performance for Purpose-Built Spacecraft and Those Based on Multipurpose Platforms



To reduce this risk, 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. It can 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.

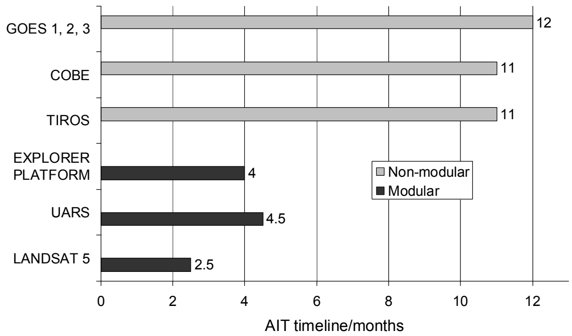
Integration and testing
A spacecraft that is made up of discrete modules can benefit from a greater concurrency in the integration process. Each module may be assembled, and tested at module level, in parallel. Standard interfaces between modules also afford a less complex final integration process,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

Decoupling of the modules, with respect to data and power, reduces the amount of "de-bugging" required when modules are interfaced together[2]. Standard interfaces also mean that test equipment can be much more standardised, and much ground support equipment 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 (EM) for each spacecraft produced using the modular platform is not necessary; an appropriate model can be assembled out of a "test suite" containing an EM 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.

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, and comparable spacecraft using non-modular designs, and a marked timeline benefit was shown. This is illustrated in Figure 2[4].

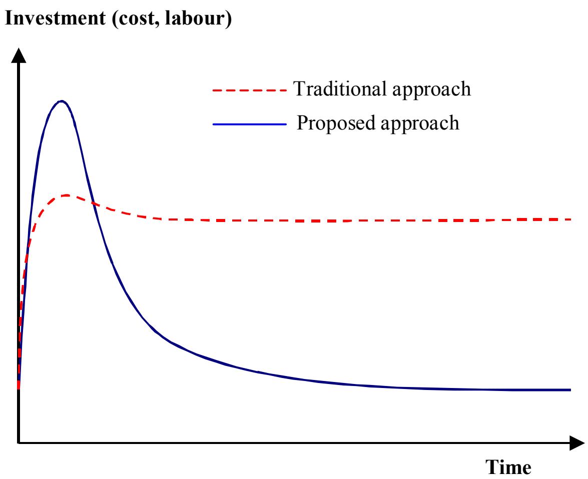
Figure 2 AIT Timelines for Modular vs Non-Modular GSFC Spacecraft



Program Philosophy

The modularity and different possible mission configurations of the proposed platform allows a different strategy to be adopted by the platform supplier. Instead of having one design (perhaps designed around a previous mission), which 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 respond to each new set of mission requirements. The philosophy is illustrated by Figure 3.

Figure 3 Levels of Investment Over Time for a Traditional Spacecraft Production Approach, and the Proposed Approach



This 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.

Target Missions/Customers

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 should also be noted that this strategy would not be suitable for all types of suppliers, due to the heavy initial investment required.

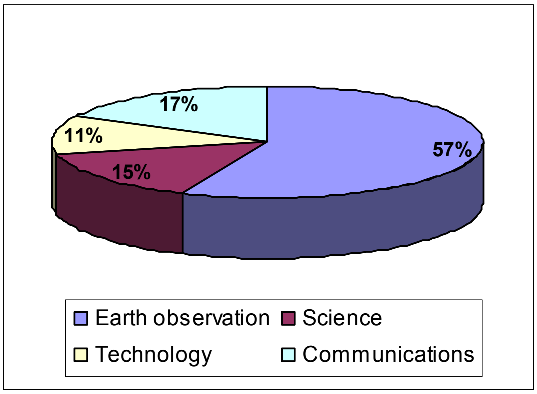
In order to identify likely target missions for the proposed platform, a detailed analysis of the worldwide uses of small spacecraft was performed[8]. This showed that the market for small satellites largely comprises Earth observation, science, technology demonstration and communications missions, with customers being drawn from civil, commercial and military sectors. Examination of these customers allowed the utility areas and cost ranges shown in Table 1 to be obtained.

Table 1 Potential Customers and Markets for a Small Multipurpose Spacecraft Platform

	Civil	**Commercial**	**Military**
Comms	Low-cost domestic comms Government messaging Cost range: \$2-15m	Point-to-point Store-and-forward messaging Asset-tracking Cost range: \$2-15m	Low-cost secure comms Cost range: \$5-20m
Earth obs.	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
Tech.	Demonstration missions to promote domestic industry Cost range: \$10-40m	-	Demonstration missions to test new systems/ equipment Cost range: ~\$15m
Science	Low-cost scientific research Cost range: \$15-40m	-	-

A examination of the industry environment and future trends[5,6,8,11] indicated the approximate distribution across the identified mission applications shown in Figure 4. The analysis also projected that the total number of missions to which the proposed platform may be applicable could be more than 20 per year, worldwide. It is anticipated that the Earth Observation and Communications sectors have the potential to exhibit the most growth, but these markets, though larger than science and technology sectors, are considerably more volatile, giving high associated risks were they to be solely targeted.

Figure 4 Approximate Distribution of Missions Expected Within the Target Market



Generation of Platform Requirements

To allow design of the spacecraft platform to proceed, a detailed requirements specification was produced. This analysed the technical and programmatic requirements that must be fulfilled, and addressed 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 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 equipment from a particular supplier.

The requirements generation process for a typical space project therefore has sets of directed requirements coming from the customer, and requirements derived from the mission objectives. The process then flows from mission objectives to 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. Furthermore, this derivation must come from the anticipated requirements for the range of missions to which the platform is to be targeted, rather than one mission objective.

With the mission and customer types identified, the goal was then to investigate the requirements necessary for the platform to be adaptable to as many of these target missions as possible. The approach used was firstly to investigate the particular requirements that might be encountered from different mission types.

These provided an envelope of mission and payload requirements. From this, the range of requirements for all target missions could be identified, and decisions made as to how broad the scope of the multipurpose platform may be.

To discover the main driving requirements demanded from the target 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).

It was, however, also found that the science category required further sub-dividing into astronomy, space physics, and microgravity, as the requirement sets for these mission types were quite different. The broad mission categories used are defined in Table 2.

Table 2 Mission Categories with Discrete Requirement Sets

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 comms	Store-and-forward messaging and mobile voice communications.	Orbcomm, Iridium, FAIsat
GEO comms	Broadcast services.	
Technology	Demonstration/validation of new technologies and techniques in space.	STRV, MSTI, PROBA

Further investigation of microgravity missions suggested that this type of application would be unlikely to make up a significant portion of the target missions for the small spacecraft. This was 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 were not considered to be key drivers for design of the multipurpose platform.

Similarly, though technology mission requirements were also addressed, but it was decided that these spacecraft could be considered as "special cases" of the other mission types, depending on the technology being demonstrated.

This left the categories of astronomy, space physics, communications, and Earth observation to be addressed. Further study of a range of such missions allowed sets of typical requirements to be generated for each. These mission-specific requirements are summarised in Table 3, at the end of this paper.

The general requirements for the platform as a whole were then examined. These included general configuration, launcher compatibility, cost, and schedule.

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.

Analysis of mechanical design requirements covered launch vehicle drivers, and led to the suggestion of 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

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 were drawn on during the concept design phase.

The spacecraft mass targets for the design section arose from:

- giving comparable payload mass capability to those platforms with which the design is to compete,
- compatibility 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 was 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.

Commercial smallsat 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. These mass boundaries must also fit in with the volumetric accommodation constraints of the launchers, identified previously.

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[1,3,7,10], and the desired ability to co-manifest on a launch at a late stage. 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. It is also highly competitive compared to delivery times for other commercial platforms.

Requirements to Enable Modularity

Examination of previous modular designs indicated that the critical factors enabling 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.

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:

- 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.

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".

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.

Data and Software
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.

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 was achieved via functional breakdown analysis of the spacecraft system. This analysis decomposed all the functions that take place on board into sub-functions, and identified 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.

Development of Platform Design Concept

The approach used to produce a platform design was first to decide on a suitable configuration. This "top-down" method was considered more suitable for this study, as the design was 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 was more important to fit the subsystem equipment into a configuration most suited to supporting a range of payloads, rather than the more usual method of designing the platform around particular subsystem/payload equipment. This approach obviously required later iteration in order to balance the needs of payload, structure, and subsystems.

A range of basic configuration types was identified for analysis. These included thrust-tube/deck, skin-stringer/longeron, box-module, and space-frame/deck concepts. Parameters were then chosen to provide useful metrics for evaluating the applicability of these different configuration concepts. This allowed trade-off studies to be performed and an appropriate concept to be chosen for more detailed development.

After configuration selection, subsystem level design proceeded, with further configuration iterations being performed as required. The subsystem designs were chosen to give phased performance options, which supplied the modularity and reconfigurability required at a functional level. These were then integrated into different platform performance/capability variants.

The key parameters used in evaluating the candidate configuration concepts were adapted from the requirements governing the configuration of the spacecraft as a whole. 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 allowed the definition of the following parameters, which describe how well a particular configuration can meet the driving requirements:

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 considered to be less important in this instance, as it may be acceptable to sacrifice some mass efficiency for the sake of allowing a greater degree of modularity in the design.

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.

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).

Solar array surface area available
As part of the requirements analysis performed, it was determined that the platform is likely to require deployed solar arrays for most missions. However, 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 was considered a reasonably low-priority requirement.

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.

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 sizes, shapes, and masses of payload. This parameter gives a measure of how suitable each design is to be configured into different sizes. A higher rating was given if the design could be re-sized with minimal changes to the parts required. Size adaptability was considered a high priority.

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.

Degree to which platform and payload may be decoupled
Several previous spacecraft that have employed a multipurpose platform largely decoupled the payload from the supporting "service" platform. This gives considerable advantages, as it helps to both enable modularity and allow parallel integration and testing.

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.

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.

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).

Outline of Proposed Design Concept

Analysis of a number of candidate configuration concepts resulted in selection of a "box-module" design. Here, each module takes the form of a rectangular prism box structure, with one "reinforced" side. Different configurations of these box modules may be bolted together to produce the platform, with the reinforced sides forming a central "virtual thrust-tube". The concept is illustrated in Figure 5.

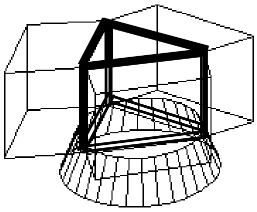


Figure 5 Box-Module Configuration Concept

The key characteristics of this design concept are described as follows, and illustrated in Figure 6, Figure 7, and Figure 8 at the end of this paper:

- The platform subsystems are housed within the modules, which may be integrated and tested independently
- 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
- Payload may be accommodated on top of the modules, within the central void volume, or within additional modules in the larger configurations
- Different size configurations can be formed, by the use of different numbers and orientations of the standard modules
- 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 module dimensions, and configurations within the launcher envelopes, are shown in Figure 6. Choice of dimensions was based on iterative trials of different module dimensions, to achieve a configuration suitable for launch with any of the envelopes identified. 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 module sizes specified are the external dimensions. 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 may 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 previously identified, 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.

Modular Partitioning and 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 therefore uses 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.

Benefits of the Approach

The benefits of the proposed approach may be illustrated by considering the effect on a space project life-cycle. A life-cycle flow of a typical spacecraft programme is as follows:

1. Mission Requirements Specification (ITT, AO, RFP etc)
2. Preliminary Design & Costing ("Phase A")
3. Bid for Contract (Project Proposal Document)
4. Detailed Design ("Phase B")
5. Procurement
6. AIT ("Phase C/D")
7. Delivery, Launch & Commissioning

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.

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.

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.

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. 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. 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 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 4 at the end of this paper. 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.

This schedule is comparable to a microsatellite timeline[12]. This should be valid, as the integration and test process can be considered analogous to the parallel AIT process for several microsatellites (i.e. 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.

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 previously, could be met quite comfortably, especially after the first couple of missions.

Summary and Conclusions

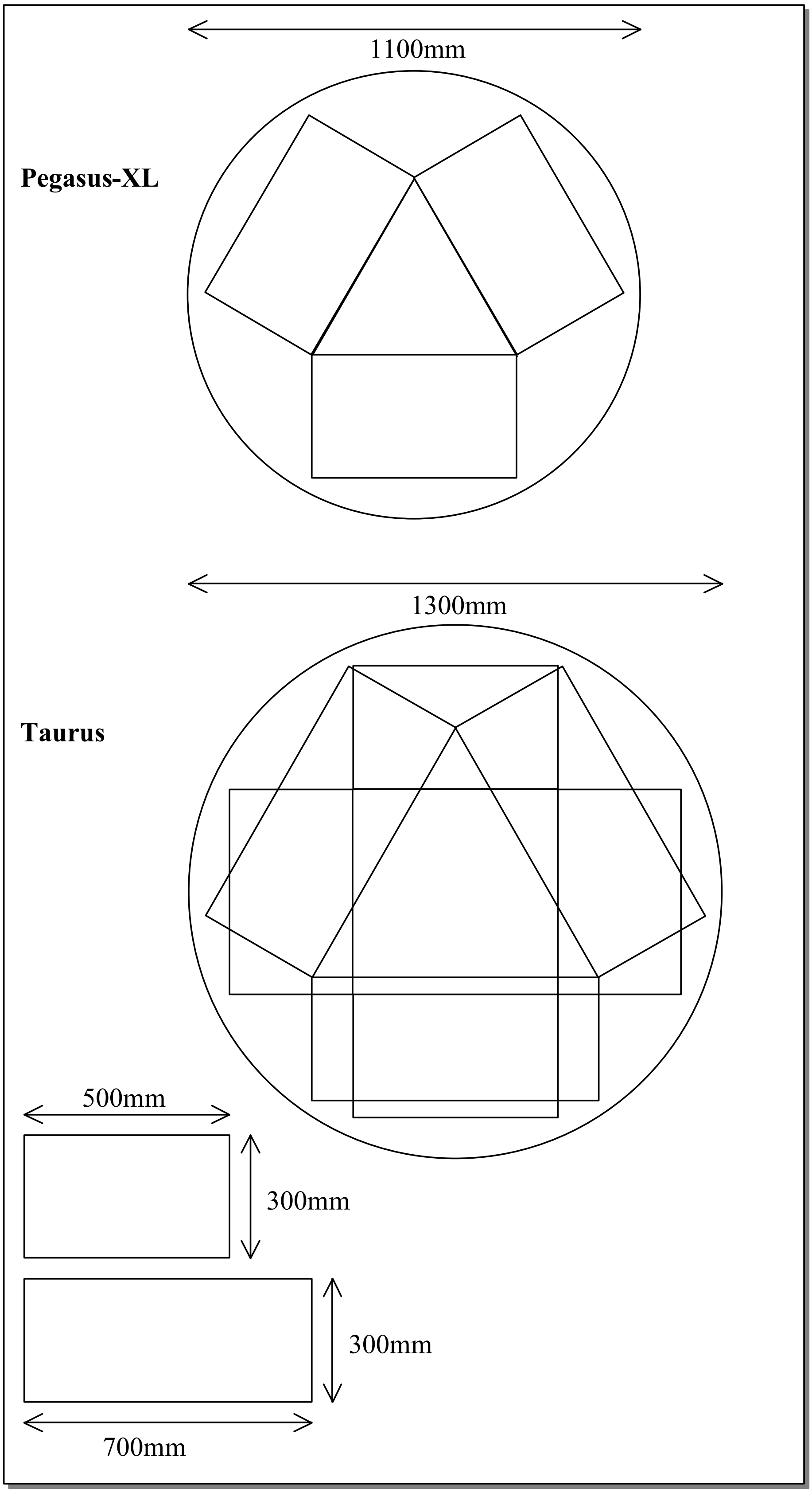
This paper has proposed an approach for a commercial small spacecraft programme, which makes use of a suite of modular platforms, assembled from a set of standard elements. A scheme for satisfying a range of mission requirements has been derived, and an outline concept for a suitable platform configuration suggested. The key benefits offered by the proposed approach may be summarised as follows:

- The philosophy of high initial investment in "pre-emptive design" minimises recurring design effort over the course of successive spacecraft projects.
- A range of different platform "capability variants" minimises wasted performance of the multipurpose platform, and allows "generic" to approach "bespoke".
- Rapid delivery and flexibility of design enables spacecraft based on the proposed platform to take advantage of a variety of launch options, to minimise launch cost.
- The modular design and "standard parts list" of the proposed platform enables streamlining of the AIT process.

Table 3 Summary of Typical Mission-Specific Requirements

Requirement area	Required performance/ characteristics	Importance
Astronomy missions		
Payload accommodation	Often large volumes, 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 (often limited by supply of cryogenic coolant)	Medium
Other	Detector cooling often required. Cleanliness for optics.	
Space physics		
Payload accommodation	Multiple, smaller instruments. May require mounting on long booms. Deployable booms may require quite large volumes.	Medium
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	Often spin-stabilised. Spin axis usually inertially-fixed.	Med-high
Pointing knowledge	Up to arcsecond accuracy.	Med-high
Pointing accuracy	Few degrees.	Low
Manoeuvring	Not often required. (But note that spin-stabilised spacecraft will require higher torques for manoeuvring).	Low
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, and require deployment.	Med-high
Data rate	Low (Kbps – little or no payload data, only housekeeping)	Low
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 in LEO, required for station-keeping in GEO.	High in GEO
Attitude	Nadir pointing	High
Pointing knowledge	Few tenths of a degree	Medium
Pointing accuracy	Up to a few tenths of a degree – dependent on payload antenna beamwidth	Medium
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, and accurate orbit insertion.	Med-high
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 (longer lifetimes likely to be an advantage)	Med-high

Figure 6 Module Dimensions and Layout Within Different Launcher Fairings



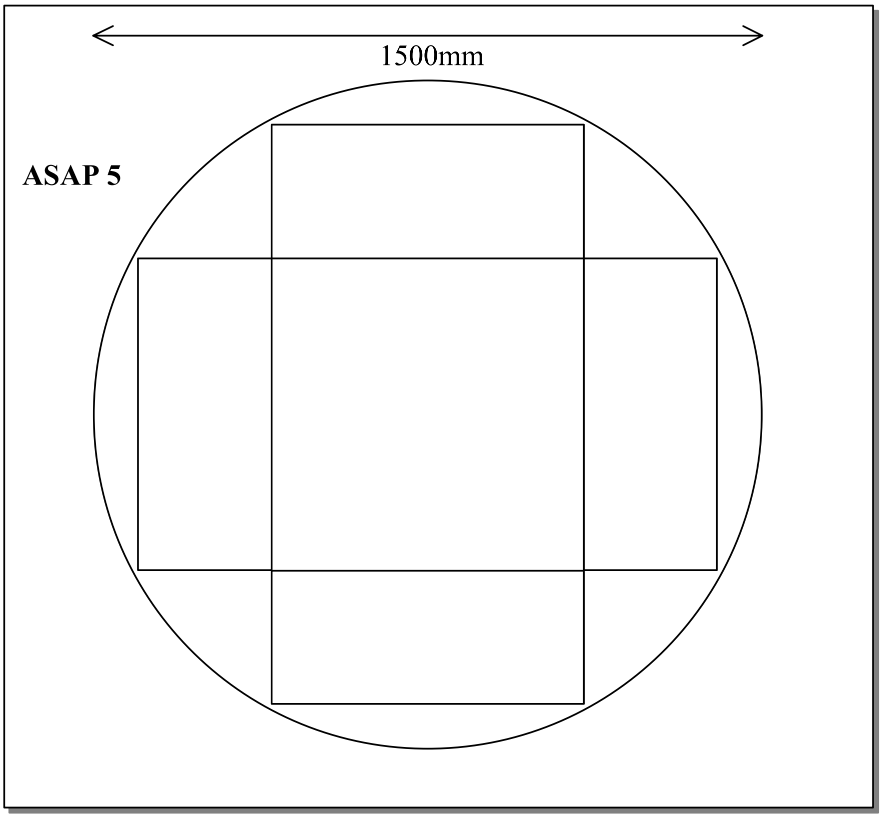


Figure 7 Module Dimensions and Orientation Options

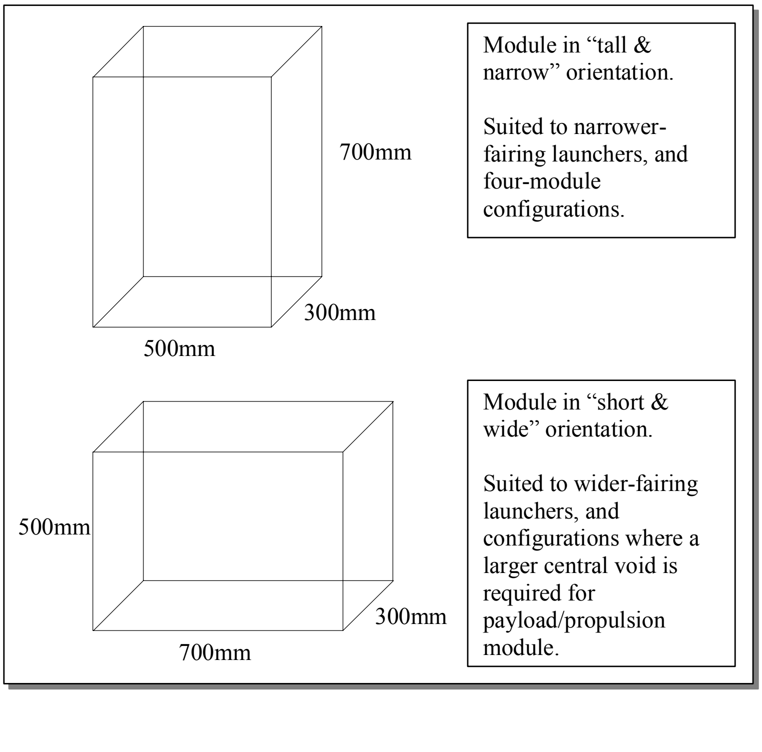
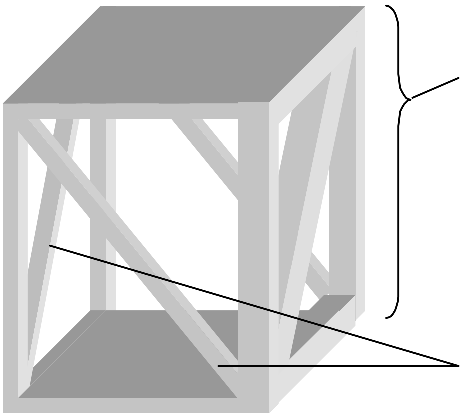


Figure 8 Module General Description



Reinforced side – forms supporting frame
This attaches to the similar frames on the other modules, and forms a strong, rigid central tube structure. It attaches to the launch adapter at the base, and can support large, heavy payloads at the top, or in the central void

Remaining sides of the box module may be closed with honeycomb or machined isogrid panels (if equipment is to be panel-mounted), or constructed from lighter frame-truss assemblies (particularly suitable if equipment requires apertures or must protrude from the module.)
Top and bottom panels are similarly interchangeable.

The module can be constructed in such a way that it may be rotated through 90° about an axis normal to the reinforced side. If the module is non-cubic, this gives an option for variation in side-length of the central void, and hence overall geometry of the configuration.

Table 4 Outline AIT Schedule for a Spacecraft Based on the Proposed Platform

Activity	**Months**									
	1	**2**	**3**	**4**	**5**	**6**	**7**	**8**	**9**	**10**
Structural model design & assembly										
Structural model testing										
Harness manufacture										
RF & thermal balance testing										
Equipment acceptance testing										
Module-level AIT – bench-level										
integrated level										
Payload AIT										
Platform integration & functional test										
Platform mechanical test										
Deployment tests										
Thermal-vacuum test										
Ground system test										
Delivery										

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