MUSTANG 2001
Summary of the Group Design Project
MSc in Astronautics and Space Engineering
2001/02
Cranfield University

College of Aeronautics Report 0206

ISBN 1 861940 99 8

Dr. Stephen Hobbs, Tom Bowling, Dr Peter Roberts
School of Engineering
Cranfield University
Cranfield
Bedford MK43 0AL
s.e.hobbs@cranfield.ac.uk

18 Sep 2003

Abstract

MUSTANG (Multi-University Space Technology Advanced Nanosatellite Group) was the group design project for students of the MSc in Astronautics and Space Engineering for the Academic Year 2001/02 at Cranfield University. The project also involved students of Southampton University and Astrium (UK) Ltd. and was supported by BNSC. The project involved the initial design of a nanosatellite to be used as a technology demonstrator for microsystem technology (MST) in space.

The project builds on previous work (in 1999/2000 and 2000/01) and is both a critical re-evaluation of the previous work and a development of new design work in specific areas (e.g. electrical subsystem, mechanisms, data handling). By the end of the project, the design has developed to a stage where detailed sub-system design and prototyping / manufacture are the next steps. The goal of launch readiness by 2003/04 is possible, but only achievable with significant extra resources.

Acknowledgements

A project like this depends on input from a wide range of people, who all deserve acknowledgement. The work presented here is primarily that of the Astronautics and Space Engineering students for 2001-02. Research students and staff (Jenny Kingston and others) have helped significantly, as have the many industry contacts (especially at Astrium) who responded to students' questions patiently, and often with enthusiasm; we gratefully acknowledge their input to the project.

Contents

List of Figures

List of Tables

Chapter 1

Introduction

This report summarises the group project of the MSc in Astronautics and Space Engineering at Cranfield University for the year 2001-02. An important part of this report is the compilation of executive summaries from students' individual reports of the project (see Appendix C).

Figure 1.1 shows a general view of the Mustang spacecraft. The mission concept is for two near-identical Mustang spacecraft to be launched and for them to fly in controlled close (100 m) formation while also demonstrating various items of microsystem technology.

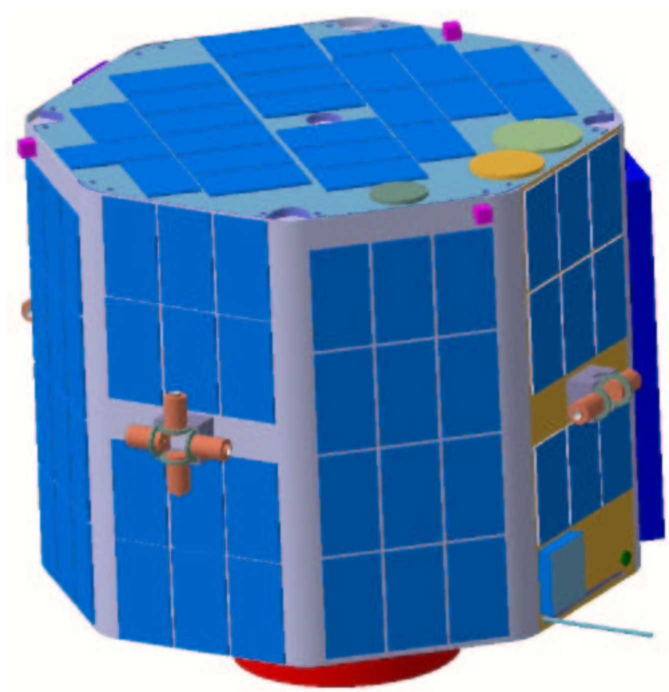

Figure 1.1: An isotropic view of the final external configuration for Mustang 2001 [19]. The spacecraft is 0.35 m high.

1.1 Organisation of the Project

The project runs over the first two terms (October to Easter) of the year long MSc course in Astronautics and Space Engineering at Cranfield University. The students work as one team, organised as several subgroups, and each student contributes about 600 hours' effort to the project; the total resource represented by the project is approximately 15000 hours' work.

Appendix A contains diagrams showing the work packages identified and the corresponding subgroups and their membership. The project was directed by Tom Bowling, Dr Steve Hobbs (Course Director) and Dr Peter Roberts. Several research students helped run the weekly progress meetings and provided support to the MSc students during the project.

1.2 Starting Point of the Project

The project is part of a programme of research into the potential of microsystems technology (MST, also referred to as Micro Electro-Mechanical Systems -MEMS) for space, being carried out in the UK by Cranfield University (Space Research Centre, School of Engineering) and Southampton University (Department of Aeronautics and Astronautics) . The programme of research has been sponsored by the British National Space Centre (BNSC) and involves Astrium (UK) Ltd as the industrial partner.

In previous years, space engineering students at Cranfield have studied two other nanosatellite projects: Custard (1999/2000) and Mustard (2000/01, also students from involving Southampton University). The project for 2001/02 builds on this previous work. MUSTANG (Multi-University Space Technology Advanced Nanosatellite Group) is both the name of the research partnership (Cranfield, Southampton, Astrium) and also the name chosen for the project in 2001/02. For clarity, the current *project* is here generally referred to as Mustang 2001.

The following section presents the requirements that have evolved for Mustang (both the partnership and the project in 2001/02) .

1.2.1 Programme and Project Requirements

The "requirements" identified are presented here as either requirements or constraints, and relate to either the whole programme of research or to this particular project.

<u>Requirements: Project</u>

1. Test MST in space
2. Demonstrate formation flying to accuracy of 1 cm (n.b. need to specify the baseline, duration, axes to be controlled, whether measurement or control is needed, etc.)
3. Demonstrate end of life deorbit (and other space debris mitigation practices - this implies a need for adequate orbit control to perform collision avoidance manoeuvre if necessary)

<u>Requirements: Programme</u>

1. University led, industry supported project; (involving Cranfield and Southampton universities as equal partners; enables technology transfer between universities and industry).
2. Train students in space engineering, with awareness of MST
3. Technology demonstration, not operational

The “customer” (mainly BNSC, but with some input from the project’s leadership) has given the following constraints (for the project).
Constraints: Project

1. Low-cost, COTS approach wherever possible.
2. Design should be suitable for different orbits / launchers (because low-cost, therefore must be able to take advantage of launches of opportunity).
3. Mass $\leq$10 kg (So that the s/c qualifies as a ”nanosatellite”, i.e. for publicity purposes; also a useful engineering challenge. We are prepared to be flexible in defining what has to be within 10 kg - e.g. payload, de-orbit device, etc. could be excluded.)
4. Should be able to accommodate various payloads (some externally provided; widens support for the project).
5. Lifetime of 1 year (must be long enough to be credible).
6. Use Astrium filament wound structure (this technology is also supported by BNSC).
7. Target launch readiness by end of 2003, otherwise technology loses its relevance

1.3 Structure of this Report

Following this introduction there is an overview of the technical work carried out by the team. The overview considers each technical area of the project in turn and concludes with a synthesis based on the system studies. The final chapter (Conclusions) states the projects main findings and outlines areas requiring further study.

Appendices contain all the executive summaries written by the students and summarise the mission and the project organisation.

Chapter 2

Technical Work

This chapter provides an overview of the technical work carried out in the project. Further details are given in Appendix C (containing the Executive Summaries of all the reports) and in the reports themselves [1] - [26] (available for reference in the School of Engineering, Cranfield University). The reports have been examined and any major errors that were identified have been corrected. However, it is not possible to guarantee that no errors remain; users of the summaries and reports should bear this in mind.

Table 2.1 shows the work packages identified for the project. Related student projects were also underway at Southampton University in the areas of GPS and the attitude control system (the previous year's projects at Southampton covered a laser ranger payload and the communications subsystem).

2.1 System

The system group was responsible for the usual system-level tasks of maintaining the budgets, operations, etc., but also studied space debris related issues such as collision avoidance and the end-of-life de-orbit demonstration. Liaison with students at Southampton University was handled by the system group too.

Analysis confirmed that, as expected for a short duration mission, the collision risk was acceptably low. The demonstration of controlled, accelerated de-orbit at the end of the mission using a drag enhancing mechanism was treated as an experiment outside the standard payload budgets. The de-orbit mechanism [2, 20] is based on lightweight deployable surfaces being developed at Cambridge University.

2.2 Payload

Tasks within the payload / mission group fall into three groups: formation flying and related technologies (control laws, FEEP thruster, DGPS, RF ranging), MST demonstration, and the MicroSAR tile. Understanding of the formation flying dynamics has developed significantly [15].

The MicroSAR tile is a radiating element for a radar antenna recently developed by Astrium with BNSC support (and can be used as a small antenna in its own right). It had a significant impact on the spacecraft configuration

Group	Tasks [number of students]		Remarks
Systems (configuration)	Budgets		
	Operations (s/c and p/l)		
	End of life deorbit	[2]	mechanical design system, analysis
	Collision avoidance	[2]	strategy, propulsion system
	ACS*		
	Comms*	[0.2]	
	Assembly, integration, test		
	Orbit, mission	[0.2]	
Payload / Mission	MicroSAR tile		design expts.
	MST	[2]	specific payload proposals
	Formation flying		
	Payload interfacing		
	Differential GPS		
	custom radio location		
	FEEP		
Mechanical	Thermal analysis		Develop structural model
	Thermal design		
	Structural analysis		
	Structural design		
	Mechanisms		release, separation, and LV interface
	Enhancements		deployable arrays
	Enhancements		cross-section change
Electrical (power, OBDH)	Arrays		
	Batteries		
	Power conditioning		
	Harness	[0.2]	
	Simulator (PC based)		
	Software design	[0.2]	Outline only
	Hardware design	[0.2]	Outline only

Table 2.1: Mustang project subgroup responsibilities and principal work packages. Unless otherwise indicated, one student was responsible for each of the tasks listed. Asterisks indicate tasks where the main function was to liase with work at Southampton University.

because of its size (almost filling one of the eight side panels) and its need to view the Earth (but only when operating in certain modes). The ability of the Mustang design to accept a demanding payload such as MicroSAR demonstrates the versatility of the spacecraft for technology demonstration.

2.3 Electrical

Good progress has been made in the design of the electrical power subsystem and data handling. A breadboard prototype of the power subsystem could now be built and would be the basis of any remaining detailed design work necessary. Figure 2.1 shows the power sub-system including solar arrays, Li-ion battery and coarse power regulation.

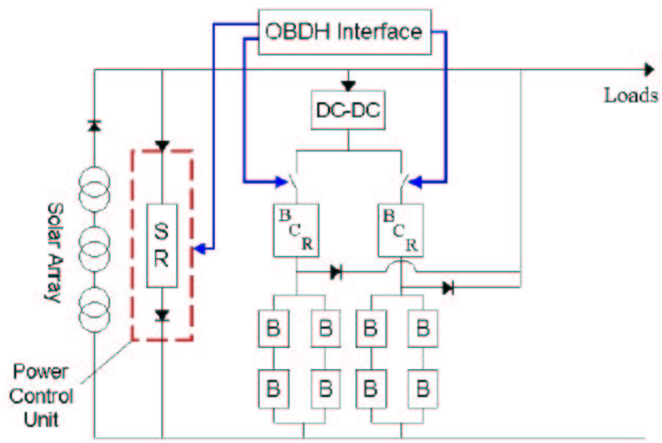

Figure 2.1: Diagram of the proposed power sub-system [13]. The bus voltage is coarsely regulated, using a shunt regulator (SR) to dump excess power from the solar arrays and Li-ion batteries for energy storage.

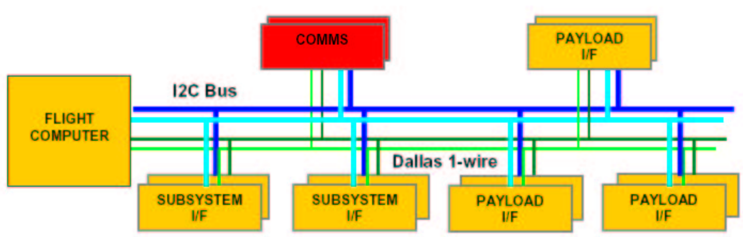

Figure 2.2: Architecture proposed for the data handling system [9]. Two buses (I2C and Dallas 1-wire) are used to give redundancy.

The data handling system has also progressed significantly and a hardware prototype (PC based) has been started based on the architectural design proposed (Figure 2.2). A hybrid (COTS) bus architecture is used to give fault tolerance. The option of buying a complete data handling system is also proposed and is worthy of further investigation.

2.4 Mechanical

Design of the filament wound structure has continued. (Outside of the group project reported here, the mandrel for the filament winding has been designed and manufacture of the spacecraft structure is expected in summer 2003.) Further thermal and structural analysis has been undertaken.

Figure 2.3 shows the basic load-bearing structure including a "vertical" shelf to carry sub-systems. An area in which particular progress has been made is mechanism design; figure 2.4 shows the proposed release mechanism.

Figure 2.3: Mustang's load-bearing structure including the spherical fuel tank at the centre of the "vertical" shelf to carry sub-systems [23].

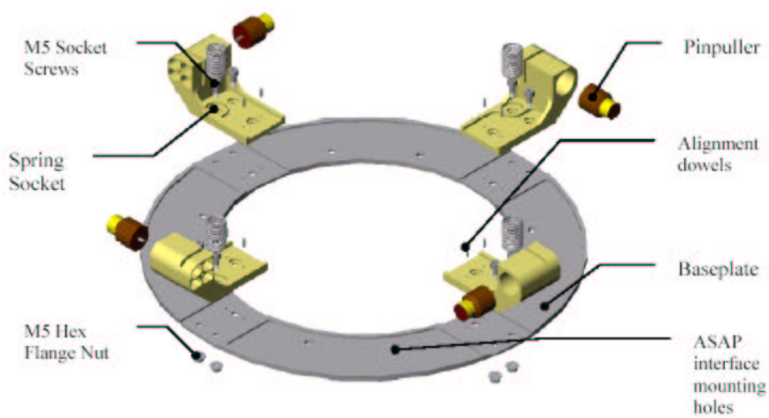

Figure 2.4: View of the proposed release mechanism [3].

Two possible design developments are also discussed: deployable solar arrays (Figure 2.5) to increase the electrical power available, and a change in cross-section from a regular octagon to a square (Figure 2.6).

The cross-section comparison is difficult to make (e.g. should it be based on equal structural masses, equal internal volumes, or equal maximum dimensions?), but it was felt that the square cross-section may usefully simplify internal and external configuration. Results of the comparison so far suggest that the square cross-section may have slight advantages, but not enough to justify a design change at this stage. Deployable solar arrays appear to offer a good

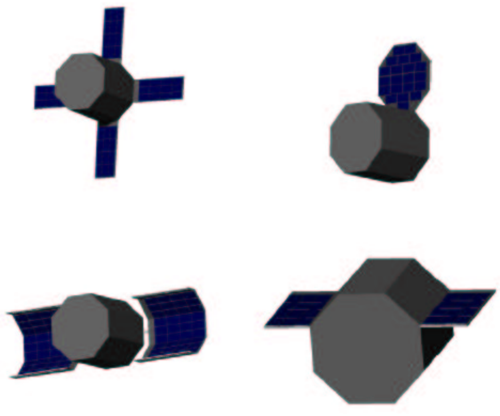

Figure 2.5: Deployable solar array options considered to increase the electrical power raising capability [25].

means of increasing the power available and could be used in later versions of the design.

2.5 Discussion

Progress has been made in practically all areas of the spacecraft design, and in most areas the project is ready to move to hardware prototypes and / or detailed design.

Management of a collaborative student project like Mustang is not easy. Students have many pressures on their time, and their main priority has to be to fulfill their degree requirements, not necessarily to ensure success of all projects within the degree. The skills they bring to the project also vary significantly, and each year a new cohort of students has to be introduced to the project before they are able to contribute. As the project moves to detailed design and manufacture, the project leadership (staff at Cranfield and Southampton universities as well as Astrium) will need to review the work programme to ensure the project objectives can be met.

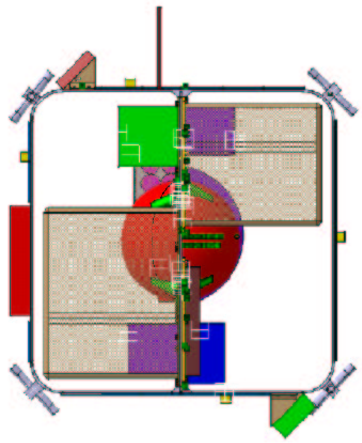

Figure 2.6: Proposed internal configuration for a square cross-section instead of the regular octagon which is the current baseline [22].

Chapter 3

Conclusions

Mustang 2001 is a challenging project for universities to undertake and good progress has been made during the year. All key aspects of the design have progressed and some areas have reached the detailed design stage.

3.1 Achievements during 2001/02

Progress has been made in practically all areas, but several areas in particular have developed significantly during the year 2001/02 at Cranfield : these are highlighted below.

- Structure (filament wound): The mandrel has been designed and ordered; materials (carbon fibre, resin) have been purchased for manufacturing, and it is currently expected that the filament wound structures for Mustang will be manufactured during the summer of 2003.
- Mechanisms: Detailed design of the release mechanisms has been completed.
- Electrical sub-system: The designs of the solar arrays, battery system and power regulation have all progressed significantly and are ready for final detailed design and prototype construction.
- Data handling: The latest design (both hardware and software) is much closer to a feasible solution than previously and breadboarding of a PC-based prototype has begun.
- De-orbit: Further analysis of the de-orbit process has been completed and an outline design for the mechanism exists.
- Formation flying: Several studies have been performed (at both Cranfield and Southampton) of control strategies for formation flying, e.g. [15], [27]. Appropriate sensors and propulsion systems can now be designed in detail.
- MST experiments: More progress has been made identifying suitable experiments and payloads.

It can be seen that in several areas the project is moving into hardware.

3.2 Future Work

There is still some way to go in preparing two satellites for launch. Tasks with a high priority are listed below.

- Begin manufacture of structural models for initial testing
- A breadboard prototype ("flat-sat") for the electrical power and data handling subsystems should be developed, to be used to develop the design and for testing
- Detailed planning of the later project phases (C/D) should be undertaken

The aim of having the spacecraft ready for launch by the end of 2003 is unlikely to be achieved without significant additional resources. To see Mustang through to completion will require further work in all areas, and realistic consideration should be given to ways forward. Options for future work include:

1. Rapid completion of Mustang as described here
2. Use the results obtained to date to develop a simplified design ("Mustang 0") which could be built with existing facilities at Cranfield and Southampton
3. Continue development of elements of technology based on the Mustang concept with a view to being able eventually to build a complete spacecraft opportunistically

The first option requires significant resources in a short period of time. Experience suggests this would be a large step from our current position. The second option requires careful planning to define a useful function for the spacecraft without requiring major additional resources, but has much lower risk associated with it than option 1. The third option is feasible, but can work to no particular user requirements or planned timetable and its utility therefore may be limited as similar space technology could easily be developed more rapidly elsewhere.

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Appendix A

Organisation of the Project

All the course students work as one team on the group project during Terms 1 and 2 (October to the end of March). Formal weekly progress meetings are held with staff present and are minuted with any necessary actions noted. Subgroups meet as necessary between these main weekly meetings.

During the first few weeks a general investigation of the science requirements was carried out by all students. After this initial phase students chose one of the three technical subgroups (payload / mission, electrical, mechanical) and started work on specific areas to allow a baseline mission to be defined. A fourth subgroup (systems) was also formed.

The initial task of the team is to define a baseline mission, and then once this is done the team work to refine aspects of detailed system design. The baseline mission definition was achieved by January. The system group coordinates development of the baseline mission and then is responsible for integration of the detailed technical work of other team members into the mission.

Research was structured around a set of work packages. The following figure (Fig. A.2) shows the work packages defined and their relation to the four subgroups. The individual reports (references [1] to [26]) and their executive summaries (Appendix C) all refer to this common work package structure.

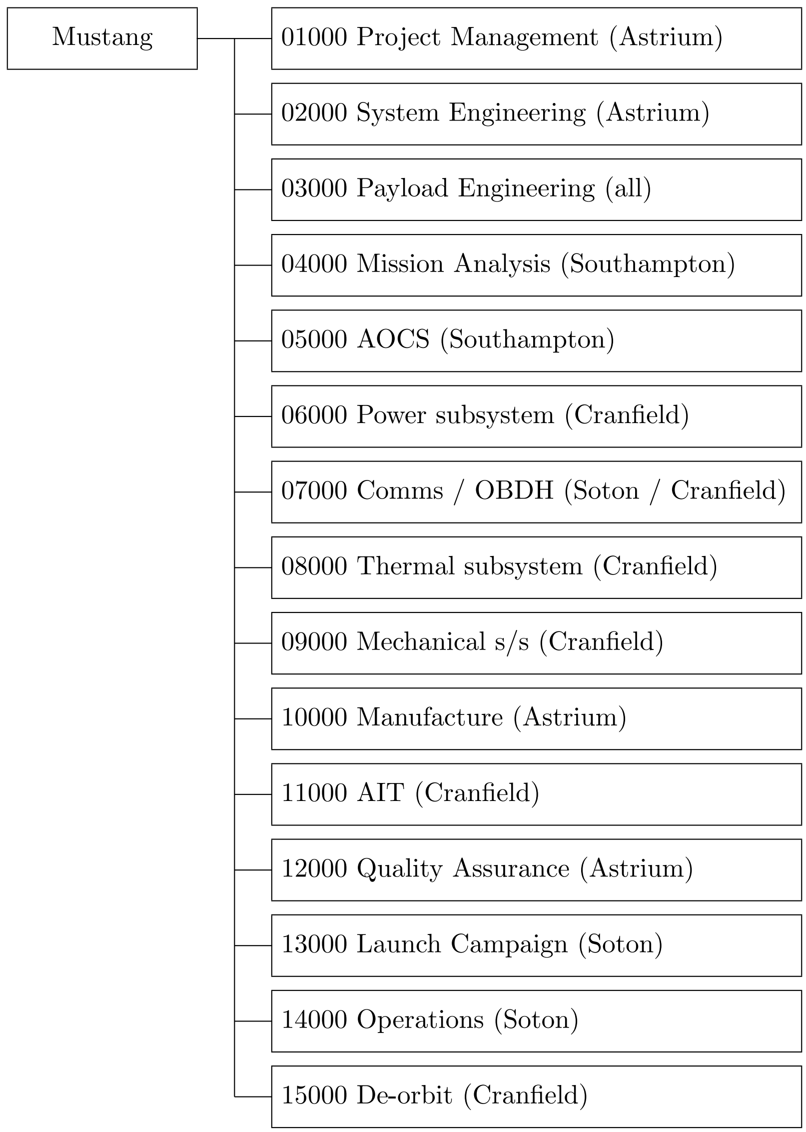

Figure A.1: Distribution of the main work package responsibilities for Mustang between the partners (Cranfield University, Southampton University, Astrium (UK) Ltd.).

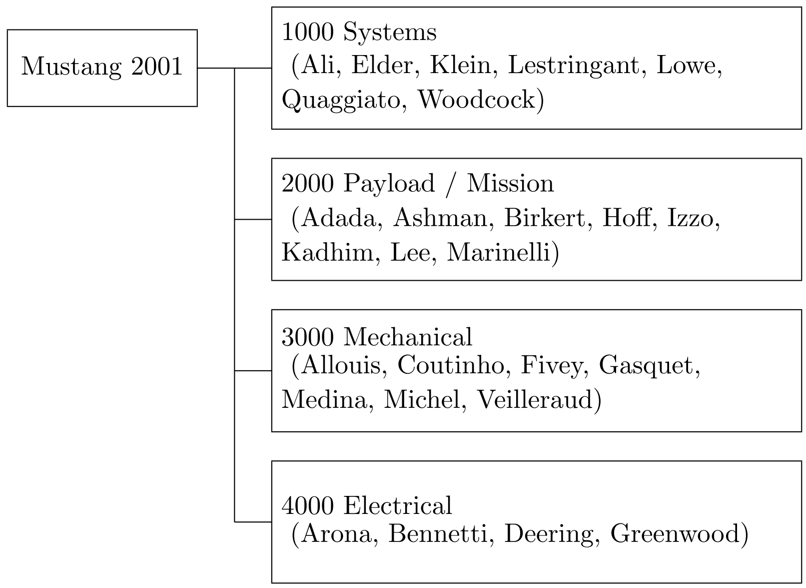

Figure A.2: Project organisation diagram showing the main subgroups and their members.

Appendix B

Mission Summary

The main objective of the mission is described in section 1.2.1. The following table summarises the main characteristics of the baseline mission (as at April 2002).

Mission	Concept	Two nanosatellites flying in formation in LEO
		Orbit: 600 km circular, polar
		3-axis attitude control, no orbit control
	Objectives	MST technology demonstration and related
		technologies (formation flying, end-of-life de-orbit)
	Lifetime	Operational lifetime = 1 yr
Spacecraft	Structure	Filament-wound, octagonal cross-section
		0.3 m dia, 0.3 m high
	Configuration	Body-mounted solar cells, internal payload
		bays and bus subsystems
		Earth facing side has payload apertures
	Electrical	Solar arrays (dual junction, 21.5 % efficiency)
		38 strings of 3, body-mounted, 38.5 mm x 68.9 mm
		Batteries: Li-ion (Sony US18650), 8 cells, 22.5% depth
		of discharge, 1.5 A hr x 3.6 V (each), total of 43.2 W hr
		Quasi-regulated bus, 7 V, 4 A nominal capacity
	Data handling	Hybrid bus (fault tolerant), I2C + Dallas 1-wire
		PIC microcontrollers for payload / subsystem interfaces
		Processor for main data handling
	Communication	S/c to ground (S-band), 500 kbps x 30 min / day
		(shared between the two s/c)
		Ground to s/c (S-band)
		S/c to s/c (UHF), 9600 bps duplex
		Single ground station (mid-latitude)
	Attitude	3 axis control (magnetorquers (3), reaction wheels (4))
		Sensors: magnetometers (3), sun sensors (5), Earth
		sensors (2), rate gyros (3)
Payload	MST demonstration	APS camera, gyros
	Formation flying	DGPS, FEEP, RF ranging
	De-orbit	Separate drag-enhancing device
Operations	Timeline	2 week system check phase
		5 month formation flying (100 m baseline)
		and MST and other experiments
		1 month drift to 30 km separation
		5 month formation flying (30 km baseline)
		and MST and other experiments
		Deliberate trigger of de-orbit device
	Mode	Day-time contacts only (30 min / day)
		Spacecraft autonomous wrt power, ACS, experiment
		management and data downlink

Table B.1: Mustang 2001 baseline summary as defined in April 2002.

Appendix C

Individual Report Executive Summaries

Executive summaries for all the project reports are given in this appendix. Full copies of the reports may be referred to at the College of Aeronautics, Cranfield University, UK.

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

C.1 System Group

The system group responsibilities include

- Budgets (mass, power, data)
- Configuration
- Orbit and mission
- Operations (spacecraft and payload)
- End of life deorbit
- Collision avoidance
- Shadow work packages being studied at Southampton University: Attitude control system, communication subsystem
- Assembly, integration and test

MUSTANG: EOL De-Orbit Device Dynamics

Adnan Ali
Space Research Centre, School of Engineering, Cranfield University

Abstract

De-orbiting the spacecraft at its end of life is a solution to the increasing problem of man made debris around earth orbit. Non-functioning satellites will take up the limited valuable orbital space that can be used for fully functional spacecrafts. A solution is to de-sign a de-orbiting device that utilises aerodynamic forces to reduce the vehicles altitude considerably. This aero-brake will need to satisfy tough end of life design requirements.

The work undertaken in this report is concerned with the impact of the size and shape of the aero-brake on the de-orbiting times of the satellite. Another concern is the spacecrafts ability to stabilise after deployment, without any attitude control and power, to demonstrate end of life autonomy.

Introduction

There is an increasing problem of man made debris in earth orbit, not only taking up valuable space but also increasing the probability of collisions with other satellites. De-orbiting the spacecraft to burn up in the earth's atmosphere is an ideal solution to this problem. This will free up valuable orbital space, to be used by other satellites while reducing the probability of collisions with other functional satellites.

This de-orbiting device will have an autonomous end of life function of the satellite and is required to accomplish its task with no power, communications, thrusters or attitude control. The most suitable method to de-orbit a spacecraft at low earth altitude is to use an aero-brake, since aerodynamic forces are predominant at these altitudes. Also, the aero-brake is required to have inherent passive stability, like a shuttlecock, to point the satellite in the correct direction for effective burn up at de-orbit. A thin film supported by spring like struts are utilized to create a pyramid shaped aero-brake. The trigger device will need to be aware of the conditions to indicate end of life, before the aero-brake can be released.

Stability

Stability can be defined as a reactive force that tends to bring the spacecraft into its equilibrium position after a disturbance of a finite magnitude and also causes the motion to subside. In the case of Mustang, the disturbance will be due to the tumbling motion after the deployment of the de-orbit device.

The prime concern when selecting the shape for the aero-brake is that it must have passive stability to damp down any oscillatory motion. The aim is to get a similar effect as a shuttlecock when it rotates and stabilises in the position where the nose is in front. In Mustangs case it is desired that it is in front of the aero-brake. The pyramid shaped aero-brake will give this desired effect while also being a feasible solution that will meet the minimum weight criteria. It will only require four struts in each corner to hold its shape.

Static Stability[4] [6] [7]
Static stability is concerned with the direction in which the vehicle will tend to move if disturbed from its equilibrium position. A system is considered statically stable when a force / moment on the system, arising from its displacement from its equilibrium position, tends to return the system to its equilibrium state.

Dynamic Stability[4] [6] [7]
Dynamic stability assesses the disturbance in the long term. If the disturbance dies away and that the system returns to its equilibrium state, the motion is considered dynamically stable. A system cannot be dynamically stable if it is not statically stable.

Model Generation

The damped oscillatory motion must obey the 2nd order equation of motion given by equation 1[4][6] [7].

$$I\ddot{q} + A\dot{q} + Bq = 0 \text{ .... } (1)$$

Where, I = Moment of inertia about the axis of rotation
A = Term associated with dynamic stability
B = Term associated with static stability

Important characteristics of any damped oscillatory motion are the time period T, of one oscillation and the time to half amplitude $t_{1/2}$.

Terms A and B need to be found in order to conduct any analysis. Firstly term B will be found as it is associated with static stability of the system and the fact that dynamic stability cannot exist without static stability.

As term B is associated with θ, Newtonian theory can be applied as it incorporates the angle that the surface makes with the molecular flow, equation 2. Thus by further applying equation 3 the forces on each surface can be found. The restorative moment of the system can then be obtained by utilising equation 4.

$$C_{Pi} = 2Sin^2 d \text{ .... } (2)$$

$$F_i = \tfrac{1}{2} r v^2 A_i C_{P_i} \text{ .... } (3)$$

$$M = (F_2 - F_1) l \text{ .... } (4)$$

Knowing the restorative moment, the resulting equation of motion then is given by equation 5.

$$I\ddot{\theta}+\left|\frac{Mf(\theta)}{\theta}\right|\theta=0 \ldots\ldots\ldots\ldots\ldots (5)$$

It should be noted that for static stability, the term associated with θ has to be positive in the equation of motion. The reason behind this is that moment M has to be directed in the opposite direction to the offset angle for the moment to be a restorative motion. For example, if the offset had a positive value, a moment resulting from this offset has to be of negative magnitude for the motion to be statically stable, otherwise the motion will not stabilise.

The next phase incorporates introducing the damping term into equation 5. It can be said that the stability characteristics of Mustang with the deployed aero-brake is similar to a tail of an aircraft. For example, both the tailplane and the aero-brake tend to return the vehicle to their respective equilibrium state after a disturbance. Furthermore, the tailplane also acts to dampen out any oscillatory motion as the restoring moment opposes the disturbance, hence opposes the tendency to deter from its equilibrium state. This damping force depends on the angular velocity of the aero-brake or the tailplane against the on coming flow. Hence, the term associated with the angular velocity in equation 1 can be modelled using the same principles as the tailplane. Adapting the theory stated by R C Nelson in his book titled "Aircraft Stability & Automatic Control" for the aero-brake results in equation 6. This equation finds the damping moment of the system and can be placed in the equation of motion as shown by equation 7.

$$M_d = \tfrac{1}{2}\rho vAl^2C_{n\theta}\,\dot{\theta} \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots (6)$$

$$I\ddot{\theta}+\tfrac{1}{2}\rho vAl^2C_{n\theta}\,\dot{\theta}+\left|\frac{Mf(\theta)}{\theta}\right|\theta=0 \ldots\ldots\ldots\ldots\ldots\ldots (7)$$

Note $C_{n\theta}$ is the gradient of the linear portion from the plot of the C_P difference against θ.

Substituting equations 8, 9 and 10 in to equation 7, dividing by $\theta_0 e^{\lambda t}$, and then utilising the characteristic equation to find its root will yield a complex solution, depicted by 11.

$$\theta=\theta_0 e^{\lambda t} \ldots\ldots (8), \qquad \dot{\theta}=\lambda\theta_0 e^{\lambda t} \ldots\ldots (9), \qquad \ddot{\theta}=\lambda^2\theta_0 e^{\lambda t} \ldots\ldots\ldots (10)$$

$$\lambda=\mu\pm j\omega \ldots\ldots\ldots\ldots\ldots\ldots (11)$$

The imaginary part of the complex roots is the damped natural frequency ω, and the real part governs the damping of the response μ. Thus, time to half amplitude $t_{1/2}$, and the time period T, can be found by utilising equations 12 and 13 respectively[4][6] [7].

$$t_{1/2} = \frac{\ln 2}{m} \text{ ………………….... (12)} \qquad T = \frac{2p}{w} \text{ …………….. (13)}$$

The de-orbiting times of Mustang, using various aero-brake geometries were obtained using STK. These times can directly be compared with time to half amplitude of the corresponding settings to give an indication whether the spacecraft will stabilise before it de-orbits. Furthermore, the aero-brakes strut lengths and semi-apex angle can be varied to analyse the effect this will have on time to half amplitude.

Conclusions

Cross examining the de-orbiting times obtained from STK and the time to half amplitude of the satellites damped oscillatory motion, it can be said that Mustang will not stabilise in time to de-orbit. Even if the satellite is not completely stable at the point of de-orbit, the aero-brake has reduced the oscillations and also increased the spacecraft area to ensure faster de-orbit. Increasing the strut lengths or the semi-apex angle results in a decrease in time to half amplitude but also decreases the de-orbiting times. High settings of semi-apex angle results in the aero-brake losing its inherent stability characteristics. This is due to the fact that high settings for the semi-apex angle will result the shape of the aero-brake to be close to a disc, which has minimum stability characteristics.

It is essential that Mustang burns up effectively in the atmosphere and that the aero-brake does not collapse onto it. A recommendation to avoid this, if no other means of stabilising is found, that the satellite is stabilised as much as possible after the deployment of the aero-brake using reaction wheels. Beyond this, the designed passive stability in the aero-brake will ensure that the spacecraft is stable until de-orbit.

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MUSTANG: Definition of Propulsion Hardware

Kenneth Elder
Space Research Centre, School of Engineering, Cranfield University

Abstract

The MUSTANG mission is a university led, industry directed project to design, manufacture and then launch a pair of nanosatellites by the end of 2003. The objectives of the mission are to demonstrate formation flying, test MST (microsystem technology) components in space and to have a de-orbit capability.

The propulsion system aboard MUSTANG is required only to provide the control necessary for the formation-flying segment of the mission. A three-point trade off study was carried out in order to determine which propulsion infrastructure would best fit the mission requirements and this included. An analysis of all the propulsion options chemical and electrical available was carried out and found that as a cold gas propulsion system would best fit the mission.

However the traditional method of using Nitrogen in the system was not possible due to the mass and volume restrictions imposed on the design. This found that a liquefied gas system and in particular Ammonia was most suitable with an overall system mass of just over 1.2 kg with a tank of under 9cm diameter. A propulsion infrastructure was chosen with most components being supplied by the British based Polyflex company.

The mass and volume savings accomplished by using an Ammonia system are not at the expense of performance however since the system is expected to give an impulse of around 96s and thrust levels around 126mN, which are more than adequate to demonstrate formation flying with a nano-satellite.

Introduction

The design of the propulsion system has been essentially split into four parts: the three point trade off study to decide which propulsion technology should be used followed by a choice of propellant and then a manufacturer capable of delivering the hardware. The performance of the chosen system must then be analysed to find Isp and thrust levels so further planning of the formation flying manoeuvres can be carried out.

Selection of Propulsion system

Initially it was realised that a cold gas propulsion system was going to be the most likely candidate to be flown on the MUSTANG mission but with new technologies emerging all the time it was felt that a system could not be designed in the confidence that it was the most suitable unless all options were analysed. The best individual system relative to MUSTANG from each technology was identified and the results of this analysis is shown below in Table 1:

Table 1: Trade off study of all small satellite propulsion options. These options are representative of the best available system relative to MUSTANG form each technology

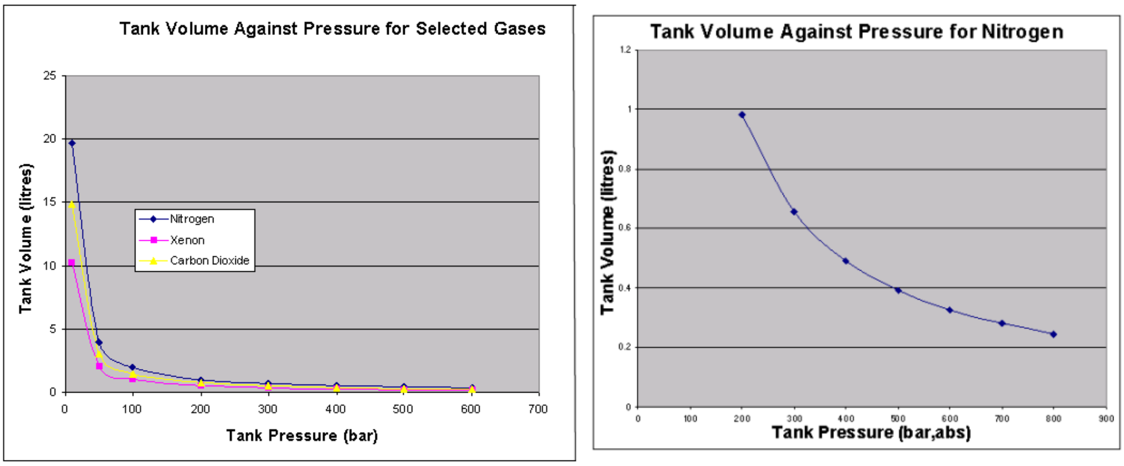
Technology	Manufacturer	Thrust (N)	Isp (s)	weight (kg)
Hydrazine Thruster	Primex	0.9	210	0.33
Micro Bipropellant	Marquardt	10	290	0.55
Hybrid Engine	SSTL	~15	280	0.7
Ion Engine	Hughes	$17x10^{-3}$	2585	6.8
HALL thruster	Busek	$17x10^{-3}$	1400	~4
FEEP	Centrospazio	$40x10^{-6}$	9000	1
Colloidal Thruster	Electro-optical Systems	$7.6x10^{-6}$	700	<2
Pulsed Plasma Thruser	AFRL	0.3	400	0.45

The chemical propulsion systems are clearly unsuitable for use on MUSTANG since each of the quoted weights here are per thruster. The three axis control which is necessary for the MUSTANG mission is not possible using these technologies due to their ‘over performance’. The electric technologies are perfect for small satellite control since they are capable of providing very small impulse bits and most are relatively lightweight. The major downfall of these systems is their extremely large power requirements which MUSTANG simply could not provide for. A cold gas system therefore became the most likely option with its simplicity and reliability lending itself perfectly to the MUSTANG mission.

Selection of Propellant

The first propellant to be investigated was Nitrogen due to its history of usage in such systems. However initial calculations showed that Nitrogen or any other compressed gas simply could not meet the mass and volume restriction imposed by the mission. As figure 1(a) shows even at very high pressures the volume required by these propellants is extremely high. Figure 1(b) shows more detail for nitrogen for the volume of tanks that we can consider.

These figures clearly show that very high pressures would be required to reduce the volume to a satisfactory value. However at such high pressures components become very expensive and MOOG industries of California who are small satellite propulsion market leaders advised that small thrusters which could cope with the pressures above 400 bar would be very difficult to manufacture and if so would be extremely expensive. Liquefied gas propellants were then identified as a possibility since their ease of liquefaction at low pressures would make them suitable. Propellants used in previous small satellite missions include propane, butane and ammonia. Ammonia displays high a high value of density Isp of 610 Ns/litre and due to having a slightly larger vapour pressure than the others it is less prone to liquefaction at the thruster nozzle.



(a) *(b)*

Figure 1 – (a) Tank volume required against tank pressure for the candidate compressed gases, although the volume reduction looks large this is only due to the scale. (b) Smaller scale volume versus pressure plot for Nitrogen. Note the 0.52 litre line which represents the maximum allowable volume of tank.

Following discussions at the Polyflex company an overall propulsion infrastructure was decided upon which is outlined below in figure 3:

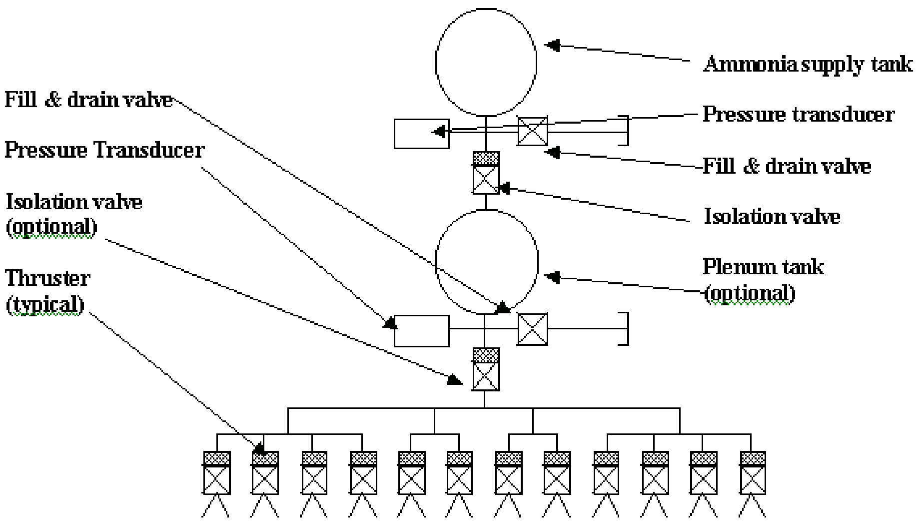


Figure 3 - Schematic diagram of the ammonia cold gas propulsion system to be flown on MUSTANG

Propellant is stored under a pressure sufficient to ensure that all the ammonia is a liquid in the main propellant tank. This propellant is then allowed to flow through a solenoid valve into a plenum volume (which in the case of MUSTANG is satisfied by the inherent volume of the downstream feed lines) where the pressure suddenly increases to the point where the ammonia 'boils-off' to a gas. The plenum is kept below the critical pressure at which there may be any liquid ammonia by means of a pressure switch placed in a feedback loop. For redundancy a temperature sensor can also be placed in this loop to make sure the propellant remains below the critical temperature. This loop therefore ensures that the propellant in the plenum volume is always gaseous.

System Performance

Central to the performance of any propulsion system is the thruster. The thruster chosen for the MUSTANG mission is Polyflex's SV06 small satellite CGPS thruster. This system has a performance which is more than suitable to control formation flying and is outlined below in table 2. The final mass budget of the propulsion system is shown below in table 3.

Table 2- Performance of the SV06 Polyflex thruster with an ammonia propellant.

Derived Thruster Performance		
Final Isp	(s)	106
Nominal Thrust	(mN)	127
Mass Flowrate	(kg/s)	~9.75E-5
Available Burn Time	(minutes)	27
Minimum Impulse Bit	(N.s)	5.04E-10

Table 3- Final definition of all the propulsion components to be flown on MUSTANG

Component	No./S.S	Part Mass	System Mass	Basis
Thrusters	12	32g	384g	Polyflex SV06
Propellant	/	158	158	N_2 at 8.5 bar
Fuel tank	1	62g	62g	8.96 cm Spherical SS Dowty Space Products
Feed Lines	Estimated at 3m	25g/m	75g	1/8" SS
Fill/Drain Valve	1	50g	50g	Polyflex
Plenum Contol Electronics	1	75g	75g	Polyflex
Thruster Mounts	4	30g	120g	'in house'
Structures and Mounts	10% of Dry Mass		93g	In house + Polyflex
Wiring	3% of Dry Mass		28g	In house
Design Margin	15% Dry Mass		157g	
Total			1202g	

Conclusions

This system follows the design philosophy and meets all of the objectives originally set out for this design. The mass is within the 2kg allocation by around 60% and the tank volume below the 10cm original objective. These savings have not however been at the expense of performance since the thrust and Isp levels attainable from the system are more than adequate to control the formation flying demonstration. The author believes that the propulsion hardware for MUSTANG has now been fully described and with the knowledge that all available options were analysed that this is the most suitable infrastructure to be implemented for the propulsion system aboard the MUSTANG mission.

MUSTANG: Orbits, Operations and Mass Budget

Helene Klein
Space Research Centre, School of Engineering, Cranfield University

Abstract

For the MUSTANG mission, the chosen orbit is a 600km circular polar orbit. However other orbits are still possible and the Ariane 5 and Delta IV orbits are presented. The impact of launching the spacecraft into a GTO has been studied. Regarding the operations, the design was limited because payloads are still unknown. A general approach has therefore been made, showing the relevant characteristics of the operations activities. The possible anomalies and their solutions were detailed for each mission phase and a safe mode has been defined. A functional analysis has been made to separate the main functions of the ground segment. The operations sizing using a low-cost philosophy has also been done. Finally the mass budget is presented; the mass of one spacecraft being 10.4kg in total.

Introduction

Regarding the orbit, a trade off must be made to consider the most probable orbits and see what the consequences of a change in the orbit would be for the spacecraft's design. As for the operations design, it should take place early enough in the project, in order for the whole team to lead the overall design in accordance with the operations concept. The operations hierarchy must appear clearly and the size must be defined. Finally, the mass and its repartition in the spacecraft's volume are major concerns for many parts of the design, it is therefore essential to keep the mass budget up to date.

Orbits

The baseline orbit is a 600km circular low Earth orbit with a 90 ° inclination, as shown on figure 1.

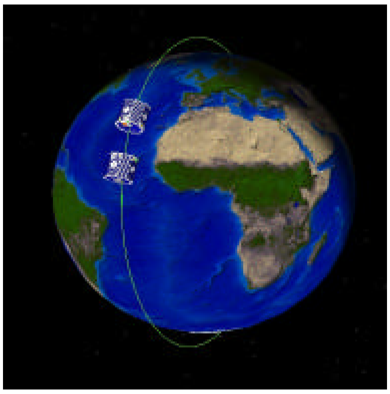


Figure 1 - The orbit is a 600km circular polar LEO.

LEO is the orbit of choice for the MUSTANG project. Firstly, as secondary payloads on large vehicles, nanosatellites have the greatest launch opportunity to go into low Earth near polar orbit. Secondly, LEO offers a lower launch cost and high launch reliability. Thirdly, it allows global coverage for experiments and communications. Lastly, the thermal analysis is also easier because of Earth proximity.

In spite of the choice for a LEO, launch opportunities still exist to Geostationary Transfer Orbits. The spacecraft should therefore be able to operate in a GTO without any important modification. The two most probable launch vehicles for MUSTANG are Ariane 5 and Delta IV. The most common orbit cases for these launchers are detailed in table 1.

Table 1 - Most common orbits achieved by Ariane 5 and Delta IV. These launchers and orbit cases have been used for the subsystems calculations and design.(Arianespace, 2002; Boeing, 2002).

Ariane 5	LEO	circular orbits, inclination from 48 to 86 deg
	GTO	560 km by 35890 km, 7deg inclination
Delta IV	LEO	500 km circular at 90 deg (most common)
	GTO	185 km by 35786 km, 27 deg inclination

These launchers and orbit cases have been used for the subsystems calculations and design (launcher adapter, collision avoidance probability, power raising over an orbit, thermal analysis). The influence of launching the spacecraft to a GTO has been studied for several subsystems. For the payloads and the collision avoidance strategy, the change of orbit is not an issue. On the other hand, the GTO presents a different thermal environment and presents variations in velocity and distance to the Earth over the orbit. The thermal subsystem, the formation flying strategy, the AOCS and the communications system would therefore need to be re-designed.

Operations

Operations activities include: the on-orbit check-out and commissioning of spacecraft, the monitoring and control of spacecraft systems (attitude, power, thermal, telemetry), the monitoring and control of instruments: health, operational mode, data acquisition.

The operations phases are:
1. The check-out phase, the two satellites still being attached (2 weeks)
2. The first collaborative control phase over a short baseline of 100 m (5 months)
3. The separation increase phase (1 month)
4. The second collaborative control phase over a long baseline of 30 km (5 months)
5. The de-orbit phase.

The main anomalies that can arise during these phases are: the power level is too low, the spacecraft fails to stabilise, the satellites lose each other, or a failure occurs in one of the critical subsystem (OBDH, power, thermal, AOCS and Communications). For all these anomalies, the ultimate response (if the fault protection algorithms cannot restore the system) is to place the spacecraft into a safe mode. The safe mode has been defined by the following characteristics: the spacecraft must stop any propulsive manoeuvre in progress, it must assume a predefined attitude (spacecraft end on to the velocity vector, the ground satellite link facing the Earth), protect sensible instruments (close the aperture of optical devices for example), and the subsystems must be turned to a low power state (that must support the anomaly detection and solving process).

Operating with one ground station, the number of passes is limited and the availability of personnel to carry out orbit corrections is limited. A functional hierarchy has been made (table 2) in order to visualize the various tasks that will need to be achieved on ground during the spacecraft's flight.

Table 2 - Functional hierarchy for the MUSTANG operations. (Adapted from Wall and Ledbetter, 1991).

0.0 Spacecraft command and telemetry system				
1.0 Manage data capture	2.0 Manage data products	3.0 Manage platform operations	4.0 Manage payloads operations	5.0 Manage ground system
1.1 Check communications systems	2.1 Calibrate and locate observation	3.1 Perform s/c planning and scheduling	4.1 Perform payloads planning and scheduling	5.1 Manage external interfaces
1.2 Perform data processing	2.2 Process data	3.2 Manage platform subsystems	4.2 Generate payloads sequences	5.2 Manage contact operations
	2.4 Archive data	3.3 Analyse flight results	4.3 Analyse instrument results	5.3 Manage ground resources
	2.5 Distribute data	3.4 Command spacecraft		

The design philosophy for the MUSTANG operations sizing is a low-cost philosophy. To reduce the cost of the operations, communications will only happen during daytime. One ground station will be used: a low cost facility (portable antenna), and Astrium infrastructure will be used for data collection. The spacecraft must be as autonomous from ground control as possible. A campaign mode of operations is likely more efficient than continuous operation, and students will manage operations. The ground segment has been sized following the functional hierarchy: the ideal number for the operations is then 5 persons. Some of the tasks might be done by Astrium, like managing the ground system, as the MUSTANG mission will use Atrium's infrastructure.

Mass Budget

After adding the subsystems mass breakdowns, the total mass of one satellite slightly exceeds the 10 kg set as a baseline at the beginning of the project. However, the design is not finished yet and the mass budget is still subjected to changes, so it was decided not to make any important change in the design because of the mass problem. The exceeding mass is only 4 %; hence the design of the satellite needn't be re-sized for a so small difference. The global mass budget for one satellite is shown in table 3.

Table 3 - Global mass budget for one satellite, end of March 2002

Element	Details	Mass (kg)
Main subsystems:		
Structure	FW + BH + shelf	2.500
	Sep. mechanism (x ½)	0.090
Thermal subsystem		0.400
Power subsystem		1.325
OBDH		0.434
AOCS		2.403
Communications		0.370
(Intermediate total)		**(7.522 + 1.0 margin = 8.522)**
Payloads:		
Payload A	Laser range finder	0.270
	GPS	0.120
Payload B	APS Camera	0.500
Payload C		0.500
Payload D		0.500
Margin (10 %)		1.000
TOTAL		**10.412 kg**
Additional devices:		
De-orbit device		0.500
Launch adapter	stays on launch vehicle	1.090
Total mass on LV	**(for the 2 satellites)**	**22.914**

Conclusions

The objectives of the orbit design have been met: a discussion on the choice of the orbit has been made, and the consequences of a change from a LEO to a GTO have been analysed. The design of the operations has been progressed: the missions operations have been broken down into a functional hierarchy that should help any further design. The low-cost philosophy and the operations sizing have also been detailed. Finally, the mass budget has been presented, showing the repartition of the mass on the various subsystems for one satellite, and the total mass the launch vehicle will need to carry.

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MUSTANG: Satellite External Configuration and Collision Risk Analysis

Franck-Thomas Lestringant
Space Research Centre, School of Engineering, Cranfield University

Abstract

The Multi-University Space Technology Advanced Nano-satellite Group (MUSTANG) comprises of Cranfield University, Southampton University and Astrium Ltd. The aim of the MUSTANG group is to produce two inexpensive nano-satellites capable of being used to accommodate a variety of payloads of opportunity.

The aim of this report is two fold. The first aim was to detail the methods undertaken in order to produce the external configuration required to fulfil the mission objectives. The second aim was to discuss the techniques involved in collision risk analysis, specifically concerning the MUSTANG nano-satellites.

As a result of detailed investigations, the final external configuration has been produced such that all objectives have been met. Evaluating the collision risk to the nano-satellites using Satellite Tool Kit, it was shown that the probability of a collision occurring was minimal.

Introduction

Since the structure, power and payloads will all depend on the exterior of the nano-satellite; the external configuration is a vital part of the investigation. The external configuration inherited from last year had to be changed and the subsequent configuration had to be constantly updated with new components. The payloads, sensors, solar cells, antennas and thrusters had to be drawn and fixed to the surface of the satellites.

The next section of particular interest to the author concerned the possibility of a collision occurring to one of the nano-satellites. The two nano-satellites will probably be placed in Low Earth Orbit (LEO), and due to this being a very populated region of space there is a risk of a collision occurring. The collision risk for the two nano-satellites was analysed and a conclusion made about the risk to the satellites using the Satellite Tool Kit (STK).

External Configuration

Based on the on the external configuration of the 2000/01 GDP, an iterative method was used in order to optimise the space available on the exterior between the various subgroup requirements.

The payloads, communication antennas, solar cells, sensors and thrusters had to be drawn and subsequently placed on the surface of the satellite in such a way that the individual performance of each component was optimised and all mission objectives were met.

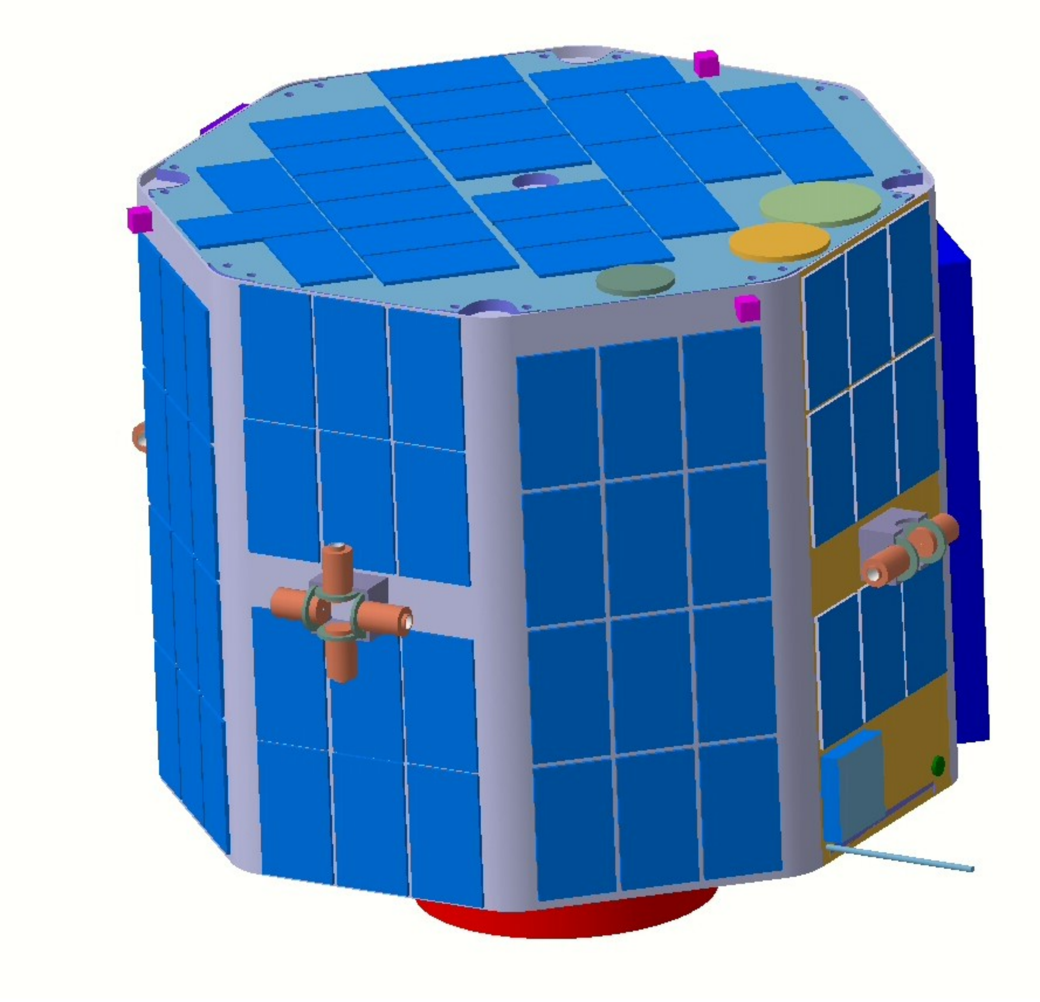

Figure 1 - Isotropic View of the Final 2001/02 GDP External Configuration

Collision Risk Analysis

Due to the increased usage of space for commercial and scientific purposes, the amount of debris in space is increasing and so the possibility of collision is becoming of higher interest to satellite operators. An investigation was undertaken in order to evaluate the risk the MUSTANG nano-satellites may be subjected to.

Three possible MUSTANG orbits were investigated. First of all, a polar, LEO orbit of altitude 600km was evaluated. The operation schedule was split into several sections to analyse the collision risk at each stage of the mission as seen in Table 1.

Table 1 - Details of the satellites relative separation, date of operation at each distance and the argument of perigees required to represent the separation for case 1

Case Number	1.1	1.2	1.3	1.4	1.5	1.6
Separation distance between the two satellites	10 m	100 m	10 km	20 km	30 km	30 km
Date for each Operation	01/01/01 - 15/01/01	15/01/01 - 15/06/01	15/06/01 - 25/06/01	25/06/01 - 05/07/01	05/07/01 - 15/07/01	15/07/01 - 31/12/01
Argument of Perigee Required	8.22×10^{-5} °	8.2×10^{-4} °	0.082°	0.164°	0.246°	0.246°

For each section, a collision simulation was run. For cases 2 and 3, the satellites spent most of their orbital period at high altitudes. This meant that the distance between the satellites relative to their orbit height was very low. Hence, the simulations could be run for a single year without any need of a change in angle of perigee, since the angle measurement would be too small to be of any practical value. Therefore the simulation for Case 1 followed Table 1, whilst for case 2 and case 3 the simulation was simply run for a length of one year. Once the simulations were run, the results were analysed. The comparison of the worst-case scenario for each case is represented in Table 2.

Table 2 - Details of the number of impacts for selected impact particle masses for the worst-case scenario of the 3 simulation cases

		Mass of Impact Particles				
		<u>0.001g</u>	<u>0.01g</u>	<u>0.1g</u>	<u>1g</u>	<u>10g</u>
Number of impacts	**Case 1.4**	5.19×10^{-4}	7.29×10^{-5}	1.08×10^{-5}	1.6×10^{-6}	3.35×10^{-7}
	Case 2	1.25×10^{-3}	1.73×10^{-4}	2.56×10^{-5}	3.76×10^{-6}	7.88×10^{-7}
	Case 3	1.23×10^{-3}	1.71×10^{-4}	2.53×10^{-5}	3.72×10^{-6}	7.78×10^{-7}

The table shows the collision risk of the MUSTANG nano-satellites is very small and therefore the odds of a collision occurring are very small.

Conclusion

The external configuration was an essential part of the GDP. It provided the subgroups with a visual aid to what the satellite would actually look like when the various payloads, solar cells, thrusters, antennas and sensors were placed on the appropriate sides. The subgroups were also able to see where conflicts occurred between certain components. This induced discussions to find an optimum solution. After much iteration, the final external configuration was finalised.

`Using STK, the probability that the MUSTANG nano-satellites would be involved in a collision during its mission lifetime was evaluated. Collision risk assessment is becoming an essential part of any mission analysis, due to the increasing amount of man made space debris. If a relatively high probability of a collision occurring exists, then the satellite operator and manufacturer can take certain steps to reduce the effect the collision will have on the satellite. The satellite can either perform an avoidance manoeuvre or increase its shielding. Both options can be implemented to provide maximum safety.

The results of the collision analysis showed that the nano-satellites are under almost no risk of collision. Therefore MUSTANG does not need to add a collision avoidance manoeuvre to its mission requirements. The nano-satellite should have enough shielding already to prevent any collision with a small particle. The carbon fibre wound octagon tube should provide adequate protection, and the multi-layer insulation adds an extra margin of security.

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MUSTANG: Mechanical Design of De-orbit Device

Steven Lowe
Space Research Centre, School of Engineering, Cranfield University

Abstract

The de-orbit device must function independently from the main satellite and require no assistance from it to deploy. The only connection between the two must be a connection from the satellite that lets the device know the satellite is still functioning so a premature activation is avoided.

The device will use the drag pressure that is dominant in low Earth orbit to decrease the energy of the orbiting satellite and so send it into the Earths atmosphere to burn up. The possibilities for the drag device structure that were considered were inflationary and mechanical. The inflationary structure would be rigidizable so as to maintain its configuration even if punctured. The mechanical structure would use curved tape springs to initiate deployment and support the structure.

The main area projected by the device would be the membrane which must be lightweight but strong enough to survive deployment and the space environment.

The release of the device will be initiated by the watchdog timer which will monitor the satellite and initiate only when the satellite is no longer functioning. It is envisaged that the most mass efficient means of deploying the device would be to use the packaging to hold the device in its folded configuration. When the packaging is opened the device would be released.

The simplest structure for the design of the device to fit into the allotted mass and size would be to use the curved tape springs as struts to support the membrane. The membrane would be made of kapton and coated in silicon oxide to protect against the atomic oxygen that is present.

Introduction

The de-orbit device design was split between two people. Adnan Ali worked on the De-orbit dynamics, while I myself worked on the mechanical design of the de-orbit device.

The need for a de-orbit device comes about from the need to reduce the amount of debris in orbit around the Earth. The most common method for the disposal of satellites at current is to use a controlled burn to either, depending on the orbit, send them into the Earths atmosphere or into a graveyard orbit. The later of these two is also questionable as it is not known if these orbits will one day need to be used, and it does not remove the threat, it mealy moves it temporarily out of the way.

For LEO orbits the preferred method is to send the satellite into the Earths atmosphere. The problem with this method is that the satellite must conserve some of its fuel to perform this manoeuvre when it nears the end of its life. The use of this fuel for the de-orbit burn means that the functional life of the satellite is reduced. If more fuel is added to compensate for this burn then the mass of the satellite increases and so does its cost.

Therefore, it is desirable to design a de-orbit device that activates once the satellite is no longer functional meaning that the satellite can be used for longer and this will increase the cost effectiveness in general.

One of the aims of the MUSTANG project is therefore to demonstrate that de-orbit of “dead” satellites is possible by means other than conserving fuel for such a manoeuvre.

The philosophy for the de-orbiting of the MUSTANG satellites is that the de-orbit device must be passive. Therefore it must not require any connection with the satellite except for the satellite to be able to tell the device that it is still functional so that an early activation of the device does not end the mission prematurely.

The de-orbit device must also be very small so that it does not affect the functioning of the satellite and it must also be considered separate from the satellite, in that it should not aid in the functioning of the satellite (e.g. by providing power via solar cells on its surface). The minimum mass for the device must be 560 grams. The options for the type of de-orbit device that could be used were narrowed down to either an inflationary device or a mechanical device.

Investigation of Device

Device Structure

The design of the device is split into six sections. These are:

- Support struts
- Membrane
- Release Mechanism
- Watchdog timer
- Power
- Packaging.

The support struts are used to tension and support the membrane material so that it stays in a certain configuration upon deployment. These will need to be collapsible to fit into the small volume of the device. The two possibilities for this that were investigated are inflationary struts, and curved tape springs.

The membrane material will provide the large area of the device for the drag effect to be more prominent and so will need to have a very small mass.

The release mechanism will hold the device in its packaged state and will be activated by the watchdog timer once the satellite is no longer functional.

The watchdog timer will be used as a monitor for the satellite so that a premature activation of the device is not encountered.

The power will be needed to be contained in the device as none can be drawn from the satellite due to the fact that there is to be not support between the satellite and the de-orbit device.

The packaging will add extra protection to the device while in the stowed configuration.

Investigation

To avoid the loss of structural integrity for an inflationary device it is envisaged that rigidizable structures will need to be used. There are various methods of obtaining inflatable rigidizable structures.

Of all the option the thin-walled aluminium method seems the best choice. In this method the inflatable part is composed of a kapton film and ductile aluminium. The kapton is positioned on both sides of the aluminium. Inflation of the structure proceeds using compressed gas until the wrinkles in the aluminium are eliminated, then the structure is pressurised to the aluminium yield point to permanently eliminate the wrinkles. At this point the structure is rigidized.

Tape springs are also a good option for the strut material due to the fact that their natural configuration is straight. Research by Dr. Pellegrino has indicated that the use of curved tape springs is preferred over that of straight tape springs and have been used in a small-scale “deployment” model of a reflector (Seffen *et al.*, 2000). The material for the tape springs that has been chosen is Copper-Beryllium (Cu-Be). This is due to the fact that Dr. Pellegrinos research was conducted on Cu-Be tape springs and so the properties for this are known.

The membrane material needs to be strong enough to withstand the rigours of stowage and deployment, but also needs to be lightweight. Of various materials that can be used studies have shown (Wooldridge *et al.*) that Kapton is still the best choice to use at this time.

Conclusions

From the previous investigation it seems that the simplest design of the device would be mechanical not inflationary. This is due to the fact that an inflationary device would require a gas canister and the associated valves to initiate deployment, while the tape springs would deploy due to their natural state being straight.

While an inflationary device may not seem like the best method for the MUSTANG satellites, as the size of a satellite becomes larger the inflationary device would be the more appropriate to use. The use of tape springs would become limited in terms of their packaging size and structural efficiency as the device size is increased. But with the MUSTANG satellites the single curved tape spring struts is the most desirable design.

At the current state of design it is recommended that the struts be made from copper-beryllium curved tape springs. The membrane which must be strong enough to survive deployment and the space environment will be a kapton membrane due to its commercial availability and physical properties. In Low Earth Orbit the presence of atomic oxygen would degrade the kapton membrane so a protective coating is needed. This coating will be silicon oxide coated on both sides of the membrane.

The release of the device will be initiated from a watchdog timer which will need to be built. This will signal a pinpuller , that is connected to the packaging, to activate and the packaing will fold open due to the hinges connecting its sides and top. Curved tape springs could also be used for this as they will try to return to their original straight configuration.

Once they are no longer held the tape spring struts will also return to their straight configuration hence deploying the device. For an inflationary device the watchdog timer would also have to send a signal to the gas canister to initiate release of the gas.

Future work

The watchdog timer still needs to be designed and built. The power requirements for the device need to be ascertained in full so that a choice on he type of batteries used can be made. The shape of the device needs to be chosen so that modelling and testing of the deployment can start to be performed.

This project is aiming to prove that the de-orbiting of dead satellites is possible. However, without some form of monitoring it will not be possible to prove this. It may therefore be necessary to design the connection between the satellite and device so that once the satellites mission is over, but the satellite is still functioning, to activate the device then so that the satellite can monitor the activation of the device and de-orbit and communicate this back.

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MUSTANG: Communication System and Attitude Control System

Valentino Quaggiato
Space Research Centre, School of Engineering, Cranfield University

Abstract

The aims of the Multi University Space Technology Advanced Nano-satellite Group, MUSTANG, are to demonstrate formation flying and Micro System Technology (MST) in space. Two nano-satellites, with the same spacecraft bus but potentially with different payloads onboard, should be produced at the end of the project. The major constraint of the design is the mass of each satellite that should not exceed 10 Kg. Another relevant limit in the design is represented by the power available. The report presents the communication system and the attitude control system. The communication system is composed by the Ground Satellite Link, which is used for the communication between the satellite and the earth station, and the Inter-satellite Link, which allows the satellites to exchange data between them. In the report the application of a patch antenna is considered. The attitude control system uses the magnetic control in order to provide a coarse pointing along the three body axis of the satellite. Fine stabilisation is achieved for two axis by using micro-reaction wheels. The report also looks at the way to achieve attitude control through the use of the earth magnetic field. The pointing budget is also defined.

Introduction

The characterising element of the MUSTANG design philosophy is modularity. In fact the satellite aims to offer payload slots, with certain characterises, to potential customers via an “announcement of opportunity” without limiting the satellite configuration to pre-assigned payloads. The mission has a length of one year during which different accuracy experiments will be performed to verify formation-flying capabilities. The orbit of the satellites should be a true polar orbit with an altitude of 600 Km. The spacecraft are 3-axis stabilised and the configuration accounts on body mounted solar cells for power raising. The satellites are based around an octagonal filament wound structure developed by Astrium. The launcher has not yet been decided, the Ariane 5 that has the highest demanding was considered for structural design purposes. The author was a member of the system group, with responsibilities for the communication system and attitude control system. These two tasks were in the previous years of the project responsibilities of the group project of Southampton University; in the current year they have abandoned the system level update to focus the interest on the technology development of some of the devices required for the communication system and attitude control system (differential GPS and reaction wheels). During the project the author has been involved in the continuous development and update of the two systems and the definition of the pointing budget.

Communications Subsystem

The communication system of MUSTANG has two different parts: one is used for the communication between the spacecraft and the earth station that is called Ground Satellite Link (GSL), an other system is in charge to guarantee the communication between the two satellites, (ISL). The general way to design a communication system is to define its characteristic parameters with a link analysis. This is an iterative process performed to achieve the best performance of the system and at the same time to satisfy the requirements. The link budget is a mathematical model that is used to analyse a radio frequency link, and consider the gains and the losses of elements of the system. The link budget equation with all terms express in dB is:

$$P_T = \frac{\mathrm{E_b}}{\mathrm{N_0}} - G_T - \frac{G_R}{T_R} - k - R_b + L_{FS} + L_A + M$$

where: P_T is the transmission power, E_b / N_0 is the energy to noise ratio of the signal, G_T is the gain of the transmitter, G_R / T_R is the figure of merit of the receiver, k is the Boltzman constant equal to –228.6dBW/(Hz*K), R_b is the data rate, L_{FS} is the free space losses, L_A is the additional losses and M is the margin. The losses have to be considered negative in the sum.

This equation can be used for the downlink as well as for the uplink. The most important constraint during the link analysis was to maintain the transmission power P_T of the downlink below 1 W. This figure was chosen for the limited available power that has characterised all the design of the MUSTNG satellites. In order to achieve this target a frequencies selection was performed which identified the S-band as the best frequencies. In particular for licence reasons, the band between 2025 MHz and 2110 MHz for the uplink and the band between 2200 and 2290 MHz for the downlink have been selected. The MUSTANG mission should demonstrate the formation flying of two satellites, to achieve and prove this proposal, the satellites have to be able to communicate to each other. The satellites will exchange data about: initial acquisition of the formation flying, attitude and orbital control system, payloads and data from the GPS and the Differential GPS. The ISL can also work as a redundancy in case one of the Ground Satellite Link systems should fail, in this unlucky situation the data from the satellite with the GSL system failed will be transmitted to the other satellite and subsequently transmitted to the earth station from this satellite. This solution requires that the onboard data handling system should have a memory capacity double the size, because it has to be able to store the data of both satellites. The link analysis of the GSL system has been conducted considering an isotropic antenna that has zero gain. Subsequently in the project it was decided to introduce a patch antenna that has not yet been designed. The possibilities to overcome this problem are two: the first one is to design the patch antenna and the second one is to find on the market the antenna that better satisfies the requirements.

Attitude Control Subsystem

The attitude control system has to be able to stabilise and to orientate the satellite following the requirements of the mission, that usually are defined by the payloads, or achieving the requirements that come from the others systems like the communication system or the propulsion system. The pointing budget is the task of the project where all the requirements for the attitude control system are summarised

Phase of the mission	Requirement	Time period
Checkout	±2°	2 weeks
First separation	Yaw & Pitch ±0.57° Roll ±2°	193 minutes 5 months
Second separation	±2°	~32 hours-1 month 5 months

Pointing budget of MUSTANG: the requirements for the attitude control system are divided for the three phases of the mission. The period of time of each phase is reported; for the first separation a period of time of two orbits (193 minutes) has been considered for the removal of the satellites.

The attitude control system developed for MUSTANG accounts of two level of control, the first is a coarse attitude control acting on the all of the three axis operated by magnetorquers, the second is a fine control system only acting on the yaw and pitch axis operated by reaction wheels. The sensors of the attitude control system that have been selected are: sun sensors, earth sensors, gyroscopes and magnetometers. To achieve the requirements of the pointing budget is necessary that the sum of the errors due to the sensors and the errors due to the actuators is maintained below the pointing value required. The accuracy of the sensors that have been selected is known and the errors due to the actuators can be kept to the desired value by using a closed loop control system that operates interactively. The design of the actuators has to be verified by simulations in order to prove that the actuators can achieve the requirements.

Magnetic Attitude Control

Magnetic control is a favourable way to stabilize spacecraft. Often, the hardware is simple and lightweight, and does not degrade or change mass over time. However, a magnetic control system has some disadvantages and limitations. The control, which is in the form of magnetic moment, can only be applied perpendicularly to the local earth magnetic field. In addition, there is an uncertainty in the earth magnetic field models due to the complicated dynamic nature of the field. Also, the magnetic hardware and the other systems of the spacecraft can interact, causing both to behave in undesirable ways. Finally, the strength of the earth magnetic field decreases strongly with the altitude of the satellite, and in order to compensate to this loss in magnetic field intensity, the maximum obtainable magnetic moment control must be increased accordingly, with an inevitable increase in dimension and power demand. This reason makes the magnetic control convenient typically for spacecraft operating at altitudes up to 1000KmThe reaction wheels are needed because they can achieve pointing requirements with higher accuracy than the magnetorquers, they can perform satellite manoeuvres and react to disturbances at a faster rate than magnetorquers. The reaction wheels have eventually to guarantee the stabilisation of MUSTANG when the satellites fly over the equator and the poles. The simulation of the magnetic attitude control has been divided in two parts: in the first part the open-loop system was considered whilst in the second part it was developed the simulation for the closed-loop system.

Conclusion

The author has tried to create a general overview of the communication system with as much practical information as possible from the huge quantity of data available from the last year report. Particular attention has been given to the downlink analysis because it is a critical task and it characterises the entire communication system. A more accurate analysis is suggested for the uplink of the Ground Satellite Link when more details will be available about the ground station that is been designing in Athens University. The hardware of the GSL is the part of the project that in the future needs more improvements, the architecture of the transmitter and the receiver were chosen and also the electronic components have been selected but the equipments have to be integrated and tested in order to prove the design that has been done. Initially in the design there was also an other GSL system in UHF-band and it was proposed as a redundancy of the one in S-band, during the project development this system has been removed because it did not represent a real redundancy of the GSL system in S-band, that eventually may use the ISL as a redundancy, and it was over the mass and the power available for the system. The design of the inter-satellite link is well defined and the next step should be the validation of the hardware. The antenna remains a critical element of the communication system and further investigations needs to be done in order to minimise the impact of this component on the project. The magnetic attitude control has been analysed with particular attention to the earth magnetic field. The mathematical model of the magnetic attitude control for MUSTANG has been defined. The author performed some simulations with Simulink, but the limited knowledge of the software did not allow the investigation to be conclusive. In future further work should be carried out with respect to the simulations; these should be completed and implemented by integrating the whole of the control system to verify its response and reliability.

MUSTANG: Systems Assembly, Integration and Testing

Andrew Paul Woodcock
Space Research Centre, School of Engineering, Cranfield University

Abstract

Assembly, Integration and Test covers the array of work between the design and operation of a spacecraft. The work includes the planning and provision of hardware for the steps involving manufacture and testing of the spacecraft. Within this report are the results of the continuation of this topic from the point of the Preliminary Design Review that marked the end of the first year of the MUSTANG Project.

A plan and schedule are presented for each stage of AIT along with a treatment of necessary further work for continuation of the project.

One key element of interest is a statistical reliability model produced by the author for use as the electronic hardware to be used in manufacture is further identified. The analysis provided will form a basis for a prediction of the performance of the spacecraft electronics.

Introduction

The MUSTANG Project aims to demonstrate the use of Micro System Technologies in space, rather than be an operational spacecraft fulfilling a specific scientific/ commercial purpose. Bearing in mind the cost and complexity of spacecraft missions it is necessary to prove technologies in operation in space before applying them to active missions.

This report describes the author's activities in the MUSTANG Group Design Project that took place in the academic year 2001-2002. As this is a continuing project, the work from this year shall be referred to in this report as belonging to MUSTANG 01/02, and the preceding year's work as MUSTANG 00/01.

The project is a collaborative effort between the academic entities of Cranfield University and The University of Southampton, and the industrial entity of Astrium Ltd.

The Systems Group

The MUSTANG project team was divided into four groups: **Systems, Payload/ Mission, Mechanical/ Structures,** and **Electrical.**

The author was assigned to the systems group; the general role of this group was to facilitate the flow of information between the various subgroups, and in that way to assist in the effective advancement of the design.

Assembly, Integration And Test- Definition and Breakdown

The definitions relating to AIT are as follows:
Assembly: Basic manufacture, for example populating circuit boards.
Integration: Assembly of complete systems to complete the spacecraft.
Testing: Ongoing procedures testing products from the component to the spacecraft level.

Assembly, Integration and Test encompasses the activities between the design and the operation of the spacecraft. This is not a black and white progression; AIT work begins during the design process, and overlaps with operation.

The work of the Leveque [2000] stemmed from the launcher selection of Ariane V. This is seen to be valid as that vehicle has the most stringent requirements of the available options.

The author provided a work breakdown structure, identifying the following tasks that required attention.

- Define Qualification Philosophy
- Identify AIT Test Regimen
- Set a Timescale for AIT
- Identify
    - Facilities Required
    - Resources Needed
    - Costs
- Lay Out Test Protocols

The required outputs were identified to be

- An AIT Plan
- A Corresponding Schedule for AIT
- Identified Procedures

Assembly, Integration and Test

Greater detail has been achieved in this area than in the Assembly and Integration elements. As planning of the test sequence is to an extent independent of the actual design, there is greater freedom in the progression of the test regimen.

The planned test philosophy is the construction of three classes of model. The initial class is the **Electrical Model [EM].** This will be a bread boarded mock up of the spacecraft systems, testing the basic design.

The next stage is the production of the **Structural Model [SM],** used to test the gross physical characteristics of the structural design. For MUSTANG two models will be produced and tested in a stacked configuration.

The third stage of testing will be the production of **Proto-Flight Models [PFMs].** These models will be effectively complete spacecraft. Where necessary these models will be refurbished, before being utilised as the flight models [FMs].

Bread boarding and testing of the EM will take place within Cranfield University's School of Engineering.

The proposed series of tests to be undertaken on the SMs is as follows:

- Low-level sine sweep
- Random vibration test, to include Shock Testing.
- Repetition of the low-level sine sweep.

For thermal testing, there will first be a thermal vacuum test, checking how well the components perform under space conditions along with the workmanship of the spacecraft. Next comes thermal balance testing, looking at the worst-case hot and cold scenarios. Heaters are installed to represent the operational thermal conditions of the spacecraft

As the PFMs are to be refurbished for flight, a balance needed to be struck between effective testing and overstressing the hardware to the point that operational performance would be affected.

In the finally integrated spacecraft the issue of EMC can be tested by a run through of all of the systems. If a system, or for example the harness, affects the performance of another element of the spacecraft, steps can be taken to fix the problem.

Production of the EM is intended to take place within the College of Aeronautics, providing students with practical experience in assembly of the model, and of test procedures.

Sine testing capabilities are available within the university in the form of Cranfield Aerospace, the associated company mentioned earlier.

For thermal testing, Astrium has a range of thermal test facilities that will be available.

One of the key strengths of a project such as MUSTANG is the latitude provided by its academic nature. If the project were fully industrial, all of the work, for example the AIT would have to meet the stringent regulations of ISO 9000. As an academic project the ruling is less strict, cutting development costs that would, for a mission of this scale, be prohibitively expensive.

To use the correct terminology, AIT for MUSTANG will aim for Qualification level testing rather than Flight Acceptance level, testing to lower levels to prove the design without testing to physical design limits.

The work of scheduling AIT for MUSTANG 01/02 was undertaken using specialist software, containing the capability to form both a precedence network and perform a critical path analysis of the featured project. Further the capability exists to perform a risk analysis, projecting the effect of possible interruptions on the project timeline, combining these and producing a probabilistic prediction of project success.

A decision was made in the early stages of MUSTANG 01/02 to limit concentration on GSE. While a valuable and important area of study, the author that until the design reached a more advanced stage design of GSE would be of limited effectiveness felt it.

The specific toolkit needed for all stages of AIT should be identified and approved. Now that the design has reached a far more advanced stage, these should be the next steps in any continuing work.

Where possible the EGSE should be common with operational equipment, in particular the communications equipment, minimising development costs. It will be necessary to liase with the individual subsystem designers to see what EGSE is required.

Clean room facilities at the university are essential for any student led practical activity. Again in consideration of the less stringent requirements of this project, a class-100000 clean room environment is considered acceptable for any work to be undertaken by the student workforce. This refers to classification given by both US Federal Specifications and ISO regulations, and refers to the number of particles greater in size than 5 microns per cubic foot of air. Bowling [2001].

Conclusions

The Systems group for the next phase of the project should be reduced in size. The suggestion made here is that doing so would allow the next Systems team to perform any logistical tasks more efficiently, without a reduction in the efficacy of the group.

The author recommends an increased number of personnel working on the topic of AIT as the project progresses. If the project is to progress in terms of meeting launch-readiness by the end of 2003, hardware production should be a major goal of any continuing work.

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C.2 Payload / Mission Group

The payload / mission group responsibilities include

- MicroSAR tile proposed experiments
- MST experiments / technology demonstration
- Formation flying (control algorithms and sensing - DGPS and custom radio location)
- Payload interfacing
- Field Effect Electric Propulsion (FEEP) experiment proposal

These topics all relate to experiments which have been proposed for Mustang, although it is not necessary that all these are flown on the first mission.

MUSTANG: Custom Radio Location

Azzedine Abada
Space Research Centre, School of Engineering, Cranfield University

Abstract

The main objective of mustang is to demonstrate collaborative control and the use of the MST (micro satellite technology) where two nanosatellites (10 kg) are to fly in formation at a distance of approximately 100m from each other. A ranging rate and the magnitude direction can define the custom radio location by the proposed receiver SIGTECK in order to prove and find a location using signal phase measurement at different frequencies for two responded channels at frequency of 50MHZ. The use of custom radio location exploration using carrier-phase measurements tend to space like their GPS counterparts, are ambiguous in whole cycles and the position estimation between two MUSTANG where the ambiguous problem by calculation changes to position on the basic of the unambiguous carrier-phase increment with continuous signal tracking accuracy and error by phase measurement.

Introduction

The location of object relative to their environment is a crucial piece of information for asset tracking, this applications may be as a simple as tracking a location, such as GPS, can help to solve this problem as long as the people or objects to be tracked are outdoors, where the signals from 24 orbiting GPS satellites may be received.

Integration with custom radio location is not very cost efficient, but both system are global.

This is a new deployment system study of MUSTANG operation at 5-50MHZ, where the use of UHF signal for phase measurement and comparison system has been carried out the advantage of the use of system is from the use the very low and changeable frequencies enabling the system to cover a sampling rate and vector direction.

The initials GPS have become nearly synonymous with radio location because of the widespread use of the system and sometimes the GPS does not give the right performance from ever-small receiver, however it makes good sense when you explored by another system. The idea we propose navigation radio signal is not a new idea. Like marines and aviators and have used phase signals for radio direction finding for years.

Range Rate Determination

By detecting the reflect radio location where its contain a tuned receiver and transmitter frequency, two antenna and displayed to find and detect the presence of an object (target). The same equation is used here: velocity (or speed) multiplied by the change in time (or the time it took to travel) equals the distance travelled, in this case we know that the velocity of the radio signals travelling towards us is the speed of light, or about 186,000 miles per second.

To find out how long the signal took to reach the target, the answer has to do with the code in the signal that the satellites send out, a binary pattern that repeated periodically. By than the calculation of range will take the following formula as:

$$R= \sqrt[4]{\frac{PG_tG_r\lambda^2\sigma}{S(4\pi)^3}} \quad \textbf{(1)}$$

Where R=range, P=power transmitted, S=received power, G_t=gain of transmitter, G_r=gain of receiver, σ=projected area of target. And time delay will be identify by the equation as follows:

$$T = \frac{2R}{c} \quad \textbf{(2)}$$

(c: speed of light = 3.10^8m/s)

The technique measurement by using two PRF in order to calculate the range (ambiguous and unambiguous). In result, the increase of PRF will result in decrease of max range but the increase of max velocity and vice versa. A modelling mathematical way of calculation in combining the time delay in function PRF, the max range and velocity will show on table 1 followed by the a figure 1 as a result of modelling.

Table 1 - Result data, which shows the relation between max range and max velocity for given PRF to get the delay and the max velocity.

PRF N^{br}	T (micrsec)	PRF (s^{-1})	Rmax (nm)	V_{max} (kt)
1	4196.7	402.82	330	20.62
2	3336.7	527.99	260	26.52
3	2656.7	724.99	205	36.11
4	2270	940.51	174	46.40
5	2090	1098.85	159	50.62

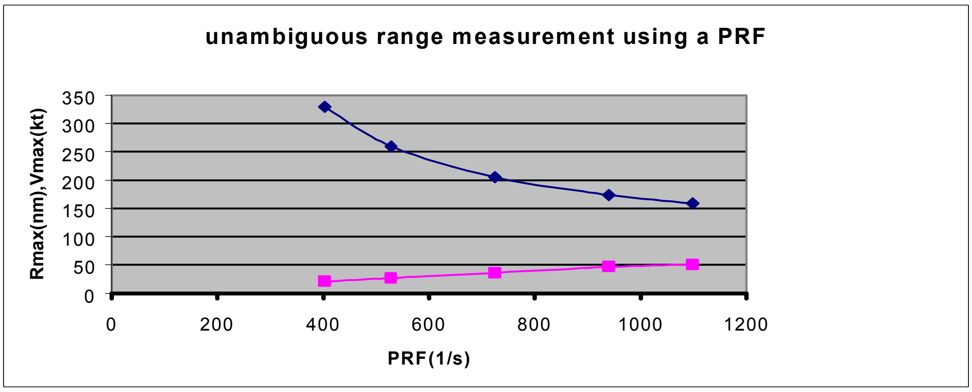

Figure 1 - Variation of max unambiguous range measurement using a set of number of PRF to find the delay (time) and the max velocity where they dependant on each-other.

Phase Measurement using Frequency Measurement

The radio ranging and detecting used one or more frequency modulation, the transmitter radiates radio frequency waves, and the frequency of the RF is continually increase and decrease from a fixed reference frequency. The amount of the difference frequency is determined by the time it took the signal to travel the distance from the transmitter to the target (object).

In order to proceed to a calculation to n value at the phase detector is at 1^0 resolution of an angle of 180^0 (256 level). At the accuracy of frequency of 50 MHZ is 5mm, the period of repetition will be: 1^0 resolution = 0.005 m. Hence 0.005*180 = 0.9m, for a complete cycle of period wave is 360^0 and repletion period 1.8m. By using the equation $1/2\lambda$, (λ=6m), a resolution of 1^0 on phase detector this gives an accuracy of Range accuracy = 1.5/180 = $8.33E10^{-3}$m. Crude frequency is 375 KHZ ($3E10^{-3}$/800). For a resolution of 180^0, the measurement accuracy is 1.11m (only two frequency modulation must be inside).

Table 2 - Summary of frequency modulation to show the accuracy measurement according to the period repetition by two frequencies.

Frequency modulation	Req range accuracy	Resolution	Period repetition	Measurement accuracy
463 KHZ	100m	180^0 360^0	162m 324m	8.33mm
375 KHZ	200m	180^0	800m	1.11m

Measurement accuracy

From the most important of the radio ranging is to determine the accuracy of phase measurement where the signal broken into a real and imaginary components, so we can say or assume that the target of the phase of the signal is just ratio over imaginary components (Real/Imaginary). By using the accuracy radar formula

$$\sigma_{\phi(\deg ree)}(\frac{180}{\pi})\frac{1}{\sqrt{2.10^{SNR(dB)/10}}} \tag{3}$$

Where b: number of bits (8 bits) and α is the ratio between the peak signal power. Let the radio works at 8 bits, and the signal power radio at 20.5dB so SNR = 32.27dB. From the equation (4) $\sigma_{\phi(\deg ree)} = 0.985^0$ and the following graph represent the accuracy result at a 40^0 phase angle measurement. At 1.5(180^0), resolution of 0.831^0, **Error = 6.92 mm.**

Finally from the equation of phase distance $(D = c\varphi / 2\pi f)$, the phase distance (position) can be found by rearranging the equation is 1.17m.

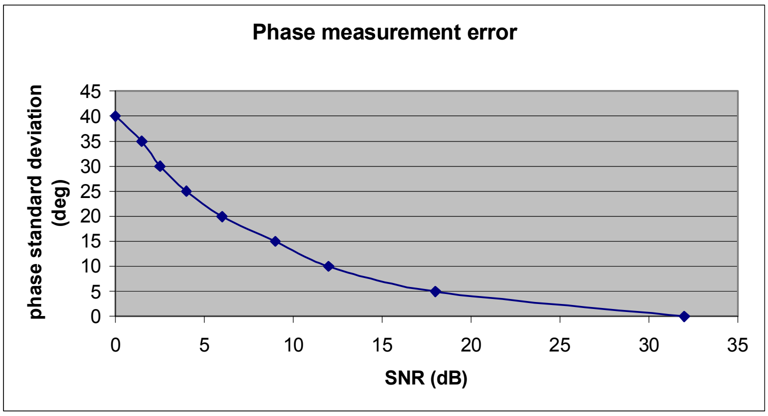

Figure 2 - Phase error is a function of signal noise ratio to identify the phase angle.

If we have more than two-phase measurements (about 3) averaged, we can find the SNR required for each measurement in 1-degree error. The noise power independent measurement of the same process will be the variance by a factor of N, so the standard variation will be reduced by a factor of $\sqrt{N}$, than we will use the formula as follow:

$$\sigma_{\phi(\deg reee)} = \left(\frac{180}{\pi}\right)\frac{1}{\sqrt{2.N.10^{SNR(dB)/10}}} \qquad \textbf{(4)}$$

Where N is number of phase .

Conclusion

The accurate knowledge of the base position directly impacts the accuracy of the position computed by radio station and finding the rate of magnitude vector by using a mathematical model of rectangular coordinate than convert them to a polar coordinate for Q and I. A certain number of sample been found in order to find a differential derivative range of 1 GHZ, position been determined and difference phase measurement accuracy error, so differential mode will remove most of the errors except multipath and receiver errors millimetre of level accuracy could be obtained with two-carrier phase and decimetre level accuracy with the code phase.

On the other hand from grace satellite, type of this radio being flown on a nanosatellite would be proven and more option can be introduced at the level of the result of accuracy been given where the measurement taken between two nanosatellites of a distance 100m.

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MUSTANG: Payload Interface

Michael Ashman
Space Research Centre, School of Engineering, Cranfield University

Abstract

This summary details the work undertaken during the academic year 2001/2002 for the Payload Interface requirement of the MUSTANG Project. The objectives of payload interfacing were to develop a comprehensive, generic payload interface specification to document all criteria relevant to the integration of a selected payload into the MUSTANG spacecraft, and also to progress the integration of current candidate payloads being considered. An interface specification document was produced and this can be found in Appendix A of the main report. It attempts to document all relevant interfacing criteria, with the long-term future of the project in mind. A discussion of current interfacing issues surrounding each candidate payload is included in this summary. It was found that the specification will require continual development in the future so as to ensure its validity. Payload specific issues were largely concerned with exceeding basic requirements such as mass and power. These can be resolved through redesigns or through compromises with certain combinations of payloads. This becomes more complicated however if external customers begin to compete for slots on-board a MUSTANG mission. It is felt that payload interfacing development may benefit from a restructuring so that specific payload engineers interface their own payload. Small working groups with relevant personnel are an alternative to this approach.

Introduction

The role of payload interfacing consisted essentially of three main activities. Firstly, the *Payload Interface Specification* task, which was concerned with the development of a generic document to detail all requirements that any candidate payload must meet in order to be considered for selection into a MUSTANG project. Secondly, *Current Candidate Payload Issues* which dealt with the evaluation of currently proposed candidate payloads against the requirements of the payload interface specification, highlighting particular issues which require resolving, and further work to be done. Thirdly, to provide a *Flow of Information* and a point of contact for all MUSTANG project members, both payload group and non-payload group, to facilitate the flow of payload related information throughout the entire group when requested or appropriate.

Typical Payload Design and Interfacing Issues

The process of designing payload instruments and their subsequent integration into a satellite will often exhibit a recognisable pattern at first glance despite the considerable differences at detailed design and integration level. It is possible in fact, to develop a standard high-level approach to progress a payload design process. The major steps are outlined here. *A Preliminary System-Level Study* focuses on the major objectives of the payload and forms a platform upon which to build the rest of the project. The level of performance can then be determined which in turn will influence the sophistication of the systems on-board. The *Mechanical Design* is the next major step. It is concerned with defining the physical properties of the instrument. The structure must be optimised to endure the expected launch loads and stresses whilst at the same time remaining within specified mass and size limits. The type of structure, materials to be used and the relevant mass properties must also be defined. Next, the *Thermal Design* system can be analysed. Temperature regulation is required to ensure that the systems on-board remain within their specified limits as the thermal range experienced can be considerable. The *Electrical System* of the payload must also be designed to provide power and control for operational commands required to perform the functions of the mission. Power supply, data handling and attitude sensing & control are key areas of the electrical system. Finally, each payload has its own unique *Functional Requirements.* These variations must be taken into account during the design and development stages of a project. A further "standard" step in the payload design and integration process is to consider the operations and tasks which are specific to the payload in question, and how these impinge on other spacecraft systems. This is perhaps deserving of special attention as the functional nature of the payload will vary considerably between projects, thus comprehensive knowledge of all specific aspects is essential in order to avoid overlooking certain parameters.

The steps outlined above can be applied to the payload development process. The development of a generic payload interface specification for MUSTANG followed these steps and the results can be seen divided into a mechanical specification, an electrical specification, and a functional specification.

Payload Interface Development

The immediate objective for payload interfacing was to identify a set of baseline requirements which the group could work to in order to further the project. These were simply criteria such as mass and power limits as specified by the MUSTANG sub-groups. Conversely, it was also necessary to ensure a flow of information in the opposite direction so that all MUSTANG sub-groups were aware of the current candidate payloads specifications. This enabled the project to move forward whilst more information was added with time.

The major objective however, was to develop a comprehensive payload interface specification detailing all MUSTANG payload interface requirements. This could then be used by candidate payloads for integration purposes, and also by future external customers as a payload integration guide applicable to all Announcement of

Opportunity (AOP) payload proposals. Whilst there was an obvious need to integrate the existing candidate payloads, it was felt that the specification should be a generic document applicable to all proposed payloads. This would ensure the production of a comprehensive specification not unduly influenced by current candidate payloads, and would also be of greater long-term benefit to subsequent MUSTANG missions.

The interface specification document was produced along the lines of the steps outlined above. The document was divided into Mechanical, Electrical and Functional Interfaces as this fitted with the work breakdown of the project and was representative of the criteria established. It should not be considered a completed document however, as criteria will evolve and change as the project progresses. The flexibility of the process was emphasised were possible as it was frequently found that candidate payloads exceeded basic requirements by amounts which were not too excessive. Most candidate payloads exceeded either mass, dimensional or power requirements. One solution to this is to match payloads which compliment each other in the sense of their cumulative specifications. I.e. a payload exceeding the mass limit could be included in a mission with a very light payload therefore giving a cumulative total within the specification. This process becomes more difficult however, if and when external customers begin to compete for payload slots on the spacecraft.

Conclusions

Every effort has been made to produce a comprehensive specification which addresses all payload interfacing issues. It is quite likely however, that previously unforeseen criteria will be identified as the project progresses so the specification document should be viewed as a dynamic tool which is to be continually reviewed and updated as necessary. Criteria which may not immediately be relevant to the project at present, has still been included on the premise that it may become more prominent as the project develops. This will also prompt users to consider these points in future integration activities which in turn, will hopefully spawn further development.

Additional documentation to consider are a modifications record to provide a historical record of changes to the interface specification, and a table, perhaps in the form of a spreadsheet, to act as a quick reference guide when assessing the listed interface criteria for a payload.

The candidate payloads must be developed with their specific interface requirements in mind. At present, analyses have been conducted to a relatively high level indicating whether or not a payload meets the requirements of the specification. The project must build upon this and detail the precise requirements for each payload. Areas to consider are the methods and materials to be used for attachment within the generic payload slot, additional containment and/or protection requirements of a payload, wiring and/or fluid line requirements and design, the consequences of failures within the payload system and payload testing requirements upon integration.

Two approaches to achieving this are suggested. Firstly, each payload group member working on a specific payload could take responsibility for its integration into

MUSTANG. This would facilitate direct communication between the payload engineer and the other MUSTANG groups and effectively place the "expert" within the project directly in the interfacing role. This approach would eliminate the "middle man" which creates the risk of the relevant people not communicating to each other. The second approach is to appoint a dedicated payload interface engineer who takes responsibility for the integration of all payloads. This would be achieved by conducting regular small working groups to include 4 other people – one person from each of the other MUSTANG groups plus the engineer responsible for the payload in question. It is essential to establish and maintain effective communication links when developing an aspect such as the payload interface which, by its nature, encompasses much of the overall MUSTANG project.

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MUSTANG: MST Experiment Design

Felicity Birkert
Space Research Centre, School of Engineering, Cranfield University

Abstract

This report is a study on microsystems technology payload experiments for a pair of nanosatellites. The growth in the interest of using MST in space is mainly due to the need for nanosatellite constellations for telecommunications and earth observation applications. The miniaturisation of spacecraft subsystems is crucial to the future success of nanosatellites, where low mass is a driving factor in design.

The main focus of the report is on microgyroscopes and microaccelerometers, for in orbit attitude sensing experiments. Application, testing and validation phases are considered in experiment design. Important factors in the testing phase are subsystem interfaces, sample times, data storage and download capacity. The validation phase considers pre launch testing, on board validation, environmental factors and post download analysis. Commercial devices suggested for use are the SiRRS01 microgyroscope by Silicon Sensing Systems, and the ASA7000 microaccelerometer in production by Colibrys. Both these devices are suitable for in orbit testing. Appropriate test phases would be during the initial check out phase of the satellites, and during formation flying manoeuvres.

Introduction

This report is a study of possible MST payload experiments for MUSTANG, a pair of nanosatellites. The first question to be answered is why use MST in space? The main advantages for using MST in space are the reduction in mass and power of the core subsystems of the spacecraft - AOCS, thrusters, OBDH. As MST becomes more widely used in space and mass production is introduced, the cost of the instruments will come down. If the mass of core spacecraft systems is reduced then either the payload ratio increases, or a smaller satellite can be used for the same purpose. In either scenario the customer is getting more value for money because launch costs are lowered. With the growth in the interest of nanosatellite constellations for telecommunications and earth observation, MST will be an invaluable resource.

MST Technology

There is a wide range of MST commercially available including actuators, sensors, switches, pumps, valves, thrusters, motors and imagers. After reviewing last year's work done by Andreas Braunwart (2001), I have decided to look at microgyroscopes and microaccelerometers as payload options for MUSTANG. The main suppliers I have found for these technologies are:

For microaccelerometers:	For microgyroscopes:
Colibrys	Murata
Honeywell	Silicon Sensing systems
Silicon Designs	Systron, BEI Technologies Inc.

MST experiment design

In designing MST experiments we need to consider the following factors.

- **Applications**
  - How can the technology be effectively used?
  - When is the most appropriate phase of the mission to demonstrate the technology?

- **Testing**
  - How long does an experiment need to run?
  - What hardware does it require?
  - What electrical connections does it need?
  - What information does it need from other subsystems to run?
  - What data will be produced?
  - Where in the spacecraft does it need to be?

- **Validation**
  - What analysis needs to be done, both pre launch and after downloading?
  - Is information from other on board systems required to validate the results?
  - Does it need a stable environment?

Microaccelerometers and microgyroscopes

The microaccelerometer I have chosen for MUSTANG is the ASA7000 silicon flexure accelerometer, (Figure 1), manufactured by Colibrys. It is an analogue output device, and has resolutions down to less than 100 µg.

Figure 1 – The ASA7000 microaccelerometer by Colibrys

An accelerometer experiment on MUSTANG could be used to sense linear accelerations during manoeuvres. The reason for choosing the ASA7000 is that it is suitable for in orbit manoeuvres where the satellites will be subject microgravity thrusts. Other models examined were designed for high gravity conditions, and would only be suitable for testing during the launch phase.

The microgyroscope I have chosen is the SiRRS01 by Silicon Sensing Systems (Figure 2).

Figure 2 – The SiRRS01 microgyroscope by Silicon Sensing Systems.

A microgyroscope could be best demonstrated during in orbit attitude control manoeuvres to sense angular rotations. For both the accelerometers and the gyroscopes suitable test phases would be during initial separation and formation flying manoeuvres.

The SiRRS01 model is Silicon Sensing's latest model, it has a relatively low mass, ~35 grams, and a good sensitivity compared with other microgyroscopes. The main specifications for the gyroscope and the accelerometer are given in Table 1.

Table 1 - Specifications of the SiRRS01 series gyroscope and the ASA7000 accelerometer.

	ASA7000 accelerometer	**SiRRS01 gyroscope**
Mass	3 grams	<35 grams
Size	15.4 mm dia. x 3.8 mm high	31.6 x 31.6 x 17.3 mm
Power	600 μW @ 3 VDC	0.25 W @ 5 VDC
Rate range	-	±110 °/s
Scale factor	0.5 v/g	18.2 mV/°/s
Bias	10 mg	0.3 °/s
Bias stability	-	3 °/h
Operating temp. range	-40 – 125 °C	-40 – 75 °C
Bandwidth	Up to 600 Hz	50 Hz
Noise	-	0.25 °/s RMS

An important factor in the testing phase is to decide how long each device needs to be operational to have demonstrated MST in space. In each operational period each device will probably run for 100 - 1000 seconds. The controlling factor will be the download time allocated for the experiment. Both devices need to be aligned with the body axes of the spacecraft. They are supplied in sealed packages with integrated electronics and can be mounted using adhesives (or screws for the gyroscope).

The primary validation for a gyroscope experiment will be a comparison with data from the main ACS gyroscope, the ARS-09 tri-axial rate sensor supplied by ATA Sensors.

Validation for both devices could come from independent measuring of the spacecraft attitude. For example, the CMOS APS camera, which is a likely payload, could give attitude information when the separation between the spacecraft is still relatively small. The gyroscope will also need drift correction periodically form the GPS.

Conclusions

My task was to investigate and propose MST experiments for MUSTANG. An initial review of commercial MST and work done last year led to my investigating microgyroscopes and microaccelerometers as payload options for MUSTANG.

The instruments I chose were the SiRRS01 gyroscope and the ASA7000 accelerometer. Both devices are suitable candidate payloads, with feasible operating temperature requirements for the MUSTANG environment. ESA has performed some radiation testing on the gyroscope, and both devices will require gamma radiation protection.

The important issues in the test phases of an experiment were identified as sample times and rates, software requirements, information needed from other subsystems, data storage and download capacity. Primary validation for the gyroscope can be obtained from the main ACS gyroscope. Validation for both experiments can be obtained from independent measuring of the attitude, for example, using the APS camera, which is likely to be on board. The drift of the gyroscope can be periodically corrected using the GPS.

Remaining work in this area of experiment design includes the software requirements. As well as the basic programming of run times, sample rates and data collection, more complex issues need to be addressed, such as the drift updating for the gyroscope. How often will it need updating and how can this be achieved? Hardware requirements such as wiring, and what can be used for radiation protection need to be looked at.

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For information on the ASA7000 accelerometer by Colibrys. www.inertialsensor.com

Silicon Sensing Systems, UK. Website on SiRRS01 series microgyroscopes. www.siliconsensing.com

Website providing links to space industry manufacturers.
www.satellite-links.co.uk/links/

MUSTANG: MST Payloads

Julien Hoff
Space Research Centre, School of Engineering, Cranfield University

Abstract

This report deals with the MST that could go on-board the MUSTANG satellites and their demonstration as it is one of the mission objectives to demonstrate MST in space. In a first step, I have defined a payload top level requirements datasheet to help people in the payload selection. Then I looked in more details at the payloads that I have selected: the Surface Acoustic Wave sensors used for thermal sensing and monitoring and the CMOS Active Pixel Sensor camera used for spacecraft visual telemetry.

Introduction

The interest for nanosatellites has been raised for these last decades as low cost is now a driving requirement for future space missions in general. One of the MUSTANG mission objectives is **to demonstrate Microsystem Technology in space**. In fact, regarding to the size and mass of the satellites, these technologies are very interesting as they have a reduced size, mass, power consumption, a high functionality and an improved reliability and robustness. However, for MUSTANG, the status of MST is experimental. Indeed, MST is only part of the payloads or part of the operational system if there is a full redundancy in the system as the MST, which will be chosen, are not specifically space qualified. But before dealing with the selected MST payloads, we have to understand the payload requirements deriving from the MUSTANG conception.

MUSTANG payload policy

We want the MUSTANG satellites to be versatile. That’s why we want to build or at least to define a standard payload interface. This makes the payload assembly in the satellite simple and this also allows the quick replacement or removal of a faulty payload without affecting the rest of the satellite until the last moment before the spacecraft is put on the launcher. However, we can’t accept all the payloads. That’s why we have defined some rules to allow a payload on-board. These rules concern:

- the size: 110 mm $\times$ 100 mm $\times$ 85 mm
- the mass: $\leq$ 500 g
- the power: 0 – 3 W
- the data: 34 Mbyte/day of downlink
- the software: to be developed by the supplier
- the cost: they have to pay to put their payload on-board…

The Surface Acoustic Wave sensors

A SAW resonator has a piezo-electric crystal (usually quartz) in it which has a resonant frequency that is highly stable in time, but temperature and strain can affect this frequency. This is the principle of a SAW resonator.

The ARGUS system (study done by the AMSTAP members) is designed to monitor the temperature with wireless SAW sensors as the harness is one of the major problem in a satellite nowadays.

Table 1 – ARGUS system top level requirements

Payload Top Level Requirements for MUSTANG	Manufacturer: *Astrium UK*
Field of application: *thermal control & monitoring*	Payload Name: *The ARGUS system*
Payload mission: *This wireless thermal sensing system has to be space demonstrated*	
Payload Standard Specifications: *Mass: 100 g for the interrogator + 2 g/sensor* *Volume: unknown but should fit in a standard payload box and the sensors are tiny* *Power: unknown but has to be < 3 W* *T-range: sensors: -100°C - +100°C minimum as it is their range* *Amount of data generated: should be modest*	
More Specifications: *Operating bandwidth: around 500 - 600 MH* *Number of sensors: 15 to match the number of thermistors*	
Interfaces: *Mountings: Stick the sensor to the surface you want to control; Interrogator unknown* *Packaging: No need for the sensor; Interrogator unknown* *Software: unknown but should be developed by the ARGUS team*	
Demonstration: *Put SAW sensors near the 15 thermistors the thermal group want to put. That way, we could compare and check their response, integrity and calibration in a first time and then demonstrate the continuous or intermittent use, to measure hot or cold cases… All this depends on the power needed by the RF electronics and the amount of data generated*	
Price: *BNSC funding if possible as nobody else could currently give money for this project*	
Contacts: Arnaud LECUYOT arnaud.lecuyot@astrium-space.com Martin SNELLING martin.snelling@astrium-space.com Simon MCCLEMONT simon.mcclemont@astrium-space.com	

The CMOS Active Pixel Camera

To have a visual link with the satellite is a very interesting and useful application as it has not only scientific aims. In fact, for public relation purposes it is a very powerful tool. For the customer and the media, to see the spacecraft evolving in space is something not negligible at all. Therefore, from a scientific point of view, this application helps in many ways to the spacecraft monitoring and control.

Table 2 – VMC top level requirements

Payload Top Level Requirements for MUSTANG	Manufacturer: *OIP sensors systems, Belgium*
Field of application: *Spacecraft visual monitoring*	Payload Name: *Video Monitoring Camera system*
Payload mission: *Taking images of the other satellite for a public relation purpose*	
Payload Standard Specifications: *Mass: 430 g* *Volume: 108 mm × 65 mm × 60 mm* *Power: 3 W in average* *T–range (operational and non-operational): -50°C - +65°C* *Amount of data generated: large but has a local buffer*	 *VMC camera*
More Specifications: *Sensors: FUGA 512 × 512 pixels or IRIRS-1 640 × 480 pixels* *Focal length: 12.2 mm;F–number: 5* *FOV: 29° × 29° (FUGA)/ 40° × 31 ° (IRIS–1)* *Wavelength range: 400–650 nm; colour images*	
Interfaces: *Mountings: Brackets* *Packaging: Yes* *Standard serial digital interfaces TTC–B–01 synchronous* *Software: Developed by OIP*	
Demonstration: *The mechanical and optical parts of the camera are already space-qualified.* *To demonstrate the sensor, we should take images every few seconds after the inter-satellite separation at the beginning and then widen the time between each shot. We should also demonstrate the windowing and sub-sampling of the sensor which will allow us to store more data. To validate this demonstration, we should compare the images with the one of the previous mission*	
Price: *Development cost around 60,000 € (might decrease if other people wants the same camera)*	
Contacts: Tim Baeyens Tim.Baeyens@fillfactory.com Guido Brandt gbr@oip.be	

Conclusion

All my work allows me to set up the requirements of a payload on MUSTANG and to find some valuable MST payloads. In fact, the ARGUS system and the CMOS APS camera that have been selected should be very valuable for future space missions as these two systems are based on a low mass, volume and power policy. Moreover, they match most of the MUSTANG requirements.

Furthermore, the payload top level requirements datasheets have been established as well as some ideas to demonstrate these payloads.

The work I've done so far will have to be carried on as soon as:

- Our policy about the payloads is clearly defined: whether we buy payloads for the demonstrator or we want people to give it for free.
- More details on the payloads are available (the ARGUS system is only a case study).
- Precise data on the satellite are available (power operations schedule…).

With the work I've done, I wanted to pave the way to the selection of the MST payloads and to the hardware. However there is still a long way to go from this point to the final hardware. So now what needs to be done is to contact the people concerned by each payload and deal with the development of the hardware:

- Review the work I've done on the payload top level requirements to see if nothing is missing and to have a critical view on the work done previously.

- Update the top level requirements and the demonstration datasheets when more detailed specifications of the system are given.

- And make an AOP as soon as the satellites up there are working perfectly to carry on with other satellites and payloads.

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Formation Flying for the Mustang Mission

Dario Rocco Izzo
Space Research Centre, School of Engineering, Cranfield University

Abstract

One of the most recent ESA missions, named Cluster II and launched in the year 2000, put four satellites in a tetrahedron formation on a highly elliptic orbit around the Earth. The mission has been so successful in its first two years of life that some of the ESA engineers wrote "*The end of space physics measurements performed by a single spacecraft has come. The future lies in taking coordinated measurements from a formation of spacecraft*". Formation Flying (FF) is indeed a very promising way of designing space missions, it allows for interferometry, for a better exploitation of highly sophisticated imaging techniques and for several others innovative applications. For these reasons it is of a great interest for MUSTANG to demonstrate that precise continuative and autonomous Formation Flying can be achieved in LEO. In this work an innovative linear mathematical approach is proposed and used to gain insight into the MUSTANG mission.

Introduction

Dealing with Formation-Flying issues always arise the question whether it is more convenient to describe the trajectories with a full non-linear model or to use the linear model given by the Clohessey-Wiltshire (CW) equations. Arguments in favour of both approaches can be easily given. The non-linear model has so much of the dynamic into it that the results given are clearly more accurate and precise, on the other hand, due to it complexity, it gives little insight into the physics of the problem and it makes quite difficult to generate control algorithms. The CW equations, on the other hand, are very simple and can be fitted into the well established linear control theory but do not provide, in their classical formulation, any description of non Keplerian forces. A great amount of literature has been written recently on how to overcome this problem. Schweighart and Sedwick [1] wrote the CW equations with an averaged J_2 term, Hughes and Mailhe [2] used the linear CW equation to validate a more complex non-linear model. In the work of Kapila et al. [3] CW equations are used with a non defined general perturbation term that takes into account every possible perturbation. In this work we chose to deal mainly with CW equations to design the MUSTANG Formation Flying strategy. We considered three satellites: the first one, called the Chief satellite defines a reference motion governed by the simple equation:

$$\ddot{\vec{r}}_\Omega = \vec{g}(\vec{r}_\Omega) \tag{1}$$

We then considered a second satellite called Deputy (i.e. one of the two MUSTANG), whose motion obeys to the equations:

$$\ddot{\vec{r}}_P = \vec{g}(\vec{r}_P) + \vec{f}_D(\vec{r}_P) + \vec{J}_2(\vec{r}_P) \qquad (2)$$

Where we have introduced, together with the gravitational acceleration, the acceleration due to Drag and Earth oblateness. The third satellite (the other MUSTANG) undergoes a motion described by the same equations as those of the second satellite. By subtracting the equations of motion of the Chief satellite to that of the other two satellites, linearizing the accelerations with respect to the point occupied by the Chief equations describing the Deputy-Chief relative motion can be obtained. Considering simultaneously the three satellites a detailed linear description of the differential Deputy-Deputy motion (i.e. the real satellites, the two MUSTANG) can be achieved.

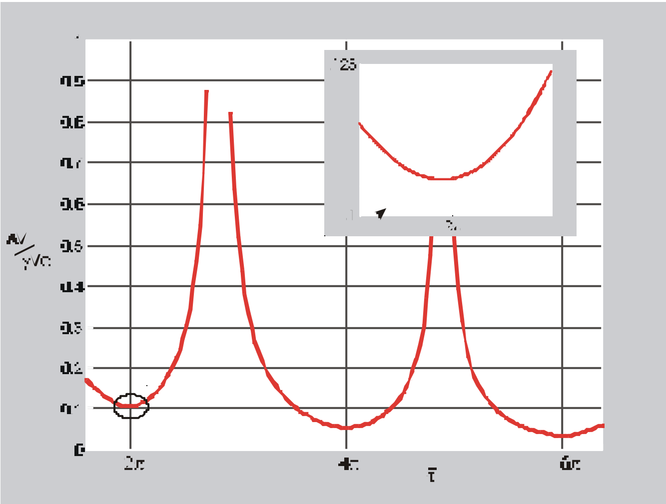

Figure 1 – Non dimensional Chart for the design of the Formation acquisition.

Formation Acquisition

To design the optimal strategy to acquire the formation the standard CW equations were used. The distance between the Leader-Follower formation was adimensionalized with respect to the orbit radius R and was called g. The circular velocity on the orbit is indicated with V_C. The result is shown in Figure 1 where the non dimensional ΔV is plotted against the non dimensional ΔT. The chart allows for a fast study of the acquisition manoeuvre. The optimal strategy is shown to be near the "Walking Orbit" strategy, classically used in Rendezvous and rephasing manoeuvres, but not coincident with it. In Valente-Izzo [4] a non linear study of the rephasing manoeuvre confirms this result.

Formation maintenance

Once acquired the Formation has obviously to be kept, this is done through a control system that gets information from the GPS, the Laser Range Finder and the Custom Radio Location, and sends commands to the on-board propulsion system in order to keep the Formation within the required precision. A first guess on the required ΔV

needed to keep the formation for a whole year was done using a modified CW model in which Drag and J_2 effects were accounted for, the result, plotted in figure 2 for various orbit inclinations, shows, for a polar orbit, a requirement of 15 m/s per satellite.

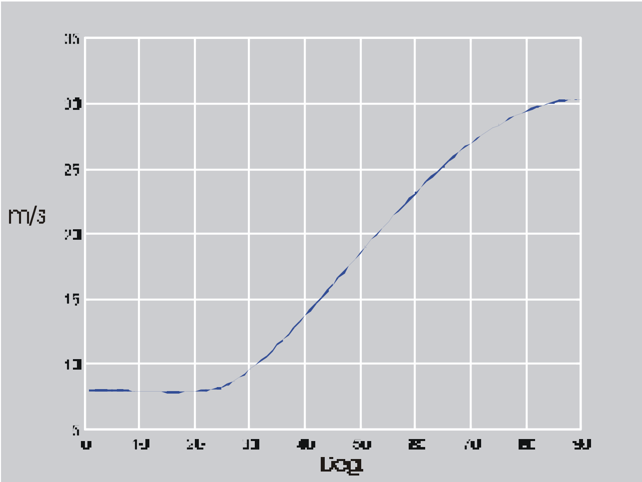

Figure 2 – ΔV required to keep a 100m Leader-Follower formation over one year.

Modelling the relative dynamic with precision

In order to design the control system and to prove that it will capable to guarantee the mission baseline requirements, a detailed description of the relative dynamic is essential. The following linear system was derived and used to gain understanding on MUSTANG mission:

$$
\begin{aligned}
&(1+e\cos\theta)x'' - 2e\sin\theta\, x' - 2(1+e\cos\theta)y' + 2e\sin\theta\, y - (3+e\cos\theta)x - \\
&- 6J_2\frac{R_E^2}{p^2}(1+e\cos\theta)^2(1-3\sin^2 i\sin^2\theta)x - 6J_2\frac{R_E^2}{p^2}(1+e\cos\theta)^2\sin^2 i\sin 2\theta\, y - \\
&- 6J_2\frac{R_E^2}{p^2}(1+e\cos\theta)^2\sin\theta\sin 2i\, y \\
&(1+e\cos\theta)y'' - 2e\sin\theta\, y' + 2(1+e\cos\theta)x' - 2e\sin\theta\, x - e\cos\theta\, y - \\
&- 6J_2\frac{R_E^2}{p^2}(1+e\cos\theta)^2\sin^2 i\sin 2\theta\, x - 6J_2\frac{R_E^2}{p^2}(1+e\cos\theta)^2(-\tfrac{1}{4}+\sin^2 i(\tfrac{7}{4}\sin^2\theta-\tfrac{1}{2}))y + \\
&+ \tfrac{6}{4}J_2\frac{R_E^2}{p^2}(1+e\cos\theta)^2\cos\theta\sin 2i\, z \\
&(1+e\cos\theta)z'' - 2e\sin\theta\, z' + z - 6J_2\frac{R_E^2}{p^2}(1+e\cos\theta)^2\sin 2i\sin\theta\, x + \\
&+ \tfrac{6}{4}J_2\frac{R_E^2}{p^2}(1+e\cos\theta)^2\sin 2i\cos\theta\, y - 6J_2\frac{R_E^2}{p^2}(1+e\cos\theta)^2(-\tfrac{3}{4}+\sin^2 i(\tfrac{1}{2}+\tfrac{5}{4}\sin^2\theta))\, z
\end{aligned}
\tag{3}
$$

An example of the solution of this system, applied to MUSTANG is shown in figure 3.

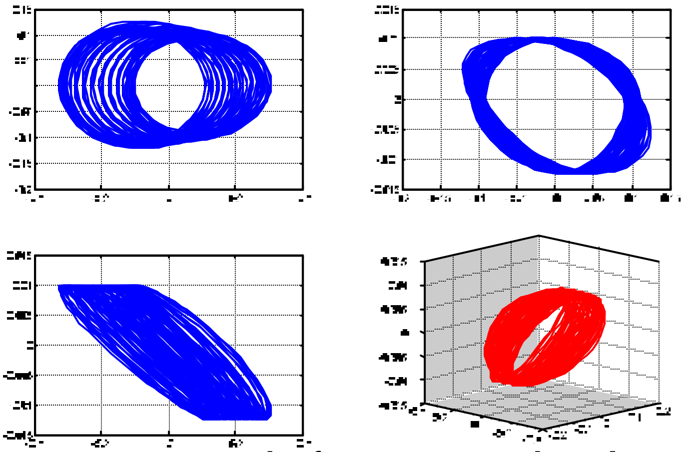

Figure 3 – Relative position vector plot for a 100m initial Leader Follower formation (no control is active). The orbit eccentricity is set to be e=.1

Conclusions

The main result of this work is the development of a new set of equations (3) modelling the relative satellite motion in the most general case. The effect of the J_2 perturbation is enclosed too. The equations are written using the true anomaly as the independent variable. This allows for a straight forward addition of the J_2 term. The drawback is, of course, that to get a time explicit solution we need to solve Kepler's equation. The design and optimisation of MUSTANG control system can take advantage from the linearity of such a system.

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MUSTANG: FEEP Test in Space

Ali Kadhim
Space Research Centre, School of Engineering, Cranfield University

Abstract

This paper presents a summary of the study performed on one of the candidate payloads, to be placed and tested for the first time in space, on the MUSTANG satellites. The report proposes an experiment to test the thruster for the first time in the space environment, to verify the design and performance of this new technology.

Introduction

Field Emission Electric Propulsion (FEEP) thruster development is currently being carried out by Centrospazio in Italy, under ESA and ASI funding, and at the Austrian Research Centres in Seibersdorf, Austria. This activity aims at two application areas for which FEEP is unique, or presents substantial advantages with respect to other propulsion technologies: drag-free scientific missions and small satellites attitude control and orbit maintenance. This report looks to build confidence amongst potential commercial and scientific users in a fairly attractive electric propulsion concept

Electric Propulsion

A description of existing and emerging types of EP devices, with their potential applications on future scientific and commercial space missions, is described in detail. The ones described are: PPT (Pulsed Plasma Thrusters), Colloid Thrusters, FEEP (Field Emission Electric Propulsion), Hall thrusters, and Ion engines.

The FEEP Thruster

The field emission electric propulsion device operates in a manner similar to the colloid thruster in that it directly extracts charged particles from a liquid propellant. The main differences lie in the propellant used and the voltage operating regime. Instead of using an electrolytic fluid, the FEEP uses a liquid phase metal, like cesium or indium, which are particularly attractive because of their low ionisation potential, high atomic weight and low melting point (see *Fig.1* below) (Reichbach, 2001).

FEU is the Field Electrical Emission Propulsion Electronic (FEEP) Unit and includes all the electronics for operating the thrusters. Each FEU may drive up to 4 thrusters and contains all the electronics for the management of the thruster operations, that is: control of the thrusters cover release aperture, thruster thermal control, thruster firing failure detection.

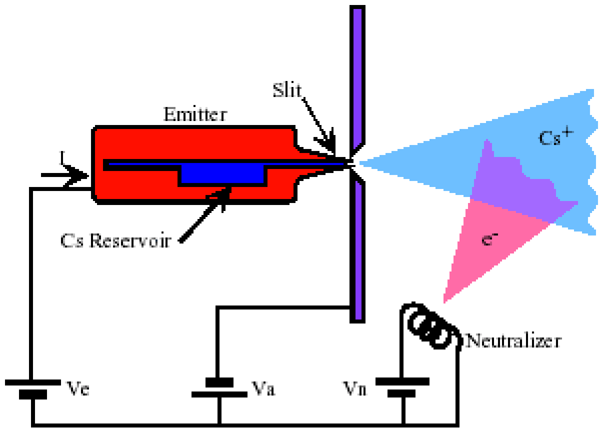

Fig.1 – Schematic for a FEEP thruster assembly (Martinez-Sanchez, 1998)

The very low and highly controllable thrust levels provided by some electric propulsion technologies enable a new category of missions to be contemplated which would otherwise not be possible due to their demanding fine-pointing and positioning requirements. This is where Field Emission Electric Propulsion (FEEP) thrusters are highly suited, providing micro-Newton to milli-Newton thrust levels with high precision control capabilities. The report also describes future missions that could possibly employ FEEP thrusters, like Galileo Galilei (GG), devoted to testing the Equivalence Principle (EP) of Galileo, Newton and Einstein to 1 part in 10^{17}, and to validate the technology of FEEP Thrusters for fine drag-free control; GAIA, which aims to measure the positions of an extremely large number of stars with unprecedented accuracy; and LISA, a six spacecraft mission devoted to gravitational wave detection in the frame of the ESA Horizon 2000+ programme (Marcuccio, 1997).

FEEP Test Objectives

The FEEP thrusters flight demonstration on MUSTANG is intended to verify the design and performance of this new technology in the space environment, for the first time. Successful completion of this test would increase potential commercial and scientific users' confidence to use this kind of electric propulsion on future satellites for attitude and orbit control, drag-free control and fine pointing of scientific spacecraft.

The FEEP test in space on board the MUSTANG nanosatellite has several aims, the *primary* one being:

- Firing the thruster in space for the first time, and measuring the thrust produced (see Fig.2 below for thrusters opening).

The *secondary* aims can be broken down into 3 phases:

- Initial Phase – survival and operation of FEEP system immediately after launch

- Operational Phase – This phase of the mission will study the operation of the FEEP thruster by firing the thruster for a particular period of time (TBD), and will evaluate the performance of the system by means of electric parameters recorded through the test.
- Technology Demonstration Phase – The final phase will involve firing the thruster in an attempt to change the spacecraft attitude, namely the rate of spin of the spacecraft. The purpose of this criterion is to determine the applicability and suitability of FEEP Thrusters for applications on satellite constellations, etc.

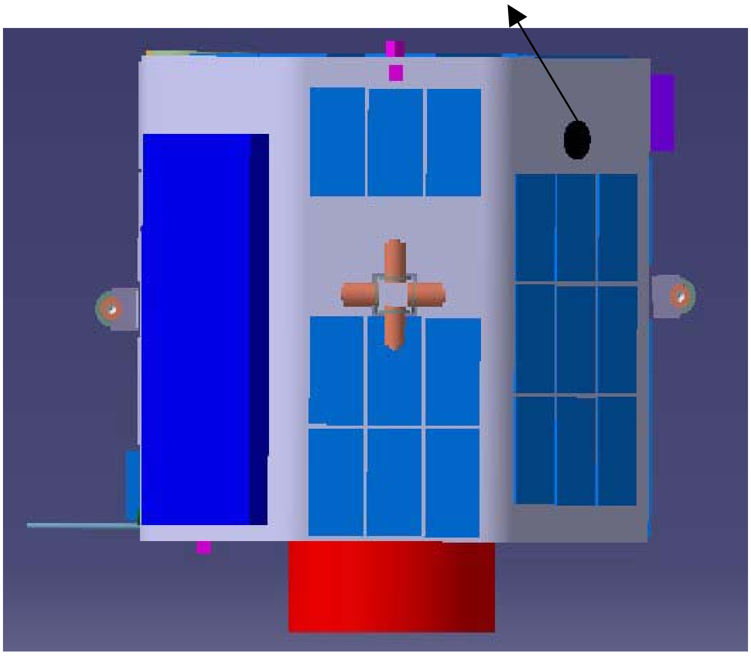

Fig.2 – MUSTANG satellite showing extra opening for FEEP Test (Elder, 2002)

Further Work

- Miniaturisation of the FEEP Electronics Unit (FEU)

- Feasibility study of employing other FEEP designs (ARCS In-FEEP)

- Investigation of 'Technology Demonstration Phase', which involves firing the thruster in an attempt to change the spacecraft attitude, namely the rate of spin of the spacecraft. The purpose of this criterion is to determine the applicability and suitability of FEEP Thrusters for applications on satellite constellations, etc.

- Look into major failure modes of FEEP thrusters: (1) clogging of the caesium feed by CsOH, produced by the interaction of caesium with ambient laboratory water, and (2) sparking damage to the knife edge of the emitter when the FEEP is run at high voltages and high currents. Water vapour will not be a problem in orbit or when the thrusters are firing continuously, but the process of bringing the FEEP into space avoiding water vapour during pre-launch, launch and orbit transfer must be carefully designed.

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MUSTANG: MicroSAR Experimental Payload Proposals

Steven Lee
Space Research Centre, School of Engineering, Cranfield University

Abstract

The Multi University Space Technology Advanced Nano-satellite Group (MUSTANG) is committed to the design and manufacture of a cost effective nano-satellite platform designed for the testing of micro systems technologies (MST). A major application of such a system is in the field of synthetic aperture radar (SAR) interferometry, where micro systems fly in formation, forming very large phased arrays. These arrays can provide extremely accurate topographical maps for earth observation and environmental science.

As a first mission it was agreed that MUSTANG should be used to demonstrate such a system. The Astrium built MicroSAR tile was suggested as a candidate experimental payload on the first MUSTANG flight. This report introduces the reader to the MUSTANG project and the MicroSAR tile before presenting a series of experiments. These experiments were designed to test critical sub-systems of the MicroSAR tile and MUSTANG's ability to control a SAR system during space operations.

Introduction

The MicroSAR tile is a synthetic aperture radar tile designed and built by Astrium Ltd as a radar transmitter capable of flying on micro and nano-satellites. Its small dimensions and in-built power supply make it an ideal candidate for use and experimentation on the MUSTANG mission. This study proposes a set of experiments designed to firstly test the MicroSAR tile sub-systems and secondly test MUSTANG's ability to control an experimental payload in space flight and provide a customer with convincing results for their package.

MicroSAR was an obvious choice for an experiment on MUSTANG both in its nature as an untested MST but also because formation flying micro-satellites like MUSTANG will be employed in the future for exactly the same type of role as MicroSAR was designed for. The first MUSTANG could pave the way for further MUSTANG spacecraft that would eventually incorporate a full SAR imaging package.

The MicroSAR Experiments

Because it is a new component and as yet has not had all its systems tested in space, these experiments were designed to test its ability to operate in the various modes available for SAR imaging.

Proposed experiment1 : Radar Altimeter

Radar altimeters were developed for use on airborne platforms where exact knowledge of altitude is vital to the craft's systems. When applied to satellites they can be used in reverse to perform a number of tasks.

The radar illuminates a circular surface on the Earth, the return echo is dispersed and a 'small window' returns back to the transmitter. An analytical model known as the Brown model, gives the average distance to the surface by extracting the amplitude and slope of the return echo. This distance gives the altitude of the satellite.

To be able to fit the Brown model with reasonable accuracy the antenna has to be set to give a small enough range resolution to suit the experiment. For a range resolution of the order of 10s of centimetres the bandwidth has to be extremely large; to the order of hundreds of MHz.

By using MicroSAR's 300Mhz bandwidth fully a range resolution of this magnitude is possible, however this would be costly in terms of power usage. By implementing some simple circuitry and a technique called 'deramping', the bandwidth can be cut down to more appropriate levels.

Figure 1 shows a typical circuit that could be designed for use in the experiment.

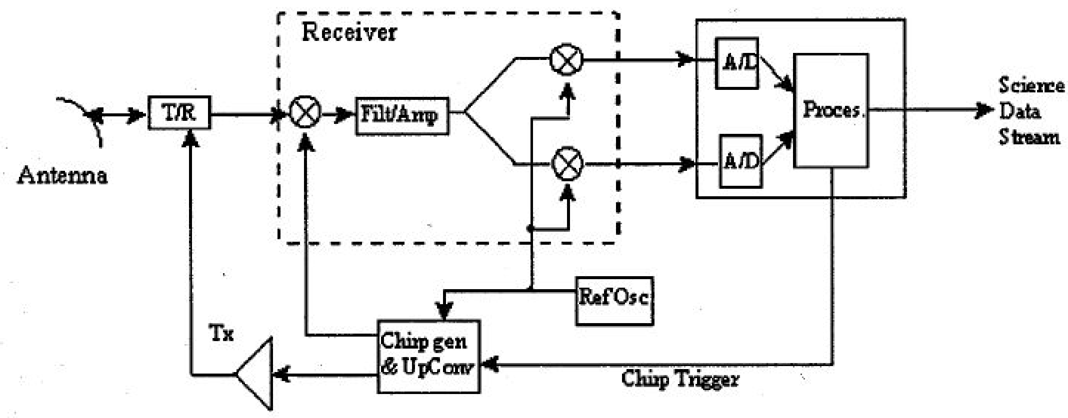

Figure 1 – Example of back end electronics for use in Altimeter experiment.

With this circuit and some simple signal processing, a number of MicroSAR's systems can be tested in one experiment. Firstly, the tile will have to be switched by logic circuits to Tile enable. Then Tx enable must be floated high whilst the chirp signal is transmitted. Within one tenth of a second (plus circuit hysteresis time) the tile must be switched to Rx enable. The tile should receive a signal and pass it through to the back-end electronics. Successful reception of a signal proves that all the above enable circuits work effectively. Secondly, by analysing the return signal when tested over calm seas, the resolution and therefore gain of the tile can be measured.

Proposed experiment 2 : Inter-satellite antenna tests

MicroSAR has a number of uses in the nano-satellite technologies proposed for test by MUSTANG. However in many of these applications basic data on the tile has to be gathered. The next experiment tests MicroSAR's use as an inter satellite ranger, for this precise knowledge of the tiles abilities to work as an antenna are vital. This following experiment proposes a set of simple experiments that calculate the gain and precise power level of MicroSAR and its RF properties. To measure the gain and power, MicroSAR will be pointed at the other satellite accurately using the MUSTANG AOCS. The tile will be fired at a given RF power, supplied by back end electronics and at the tile's carrier frequency 9.85 GHz. The other satellite will have a simple X-band receiver in the form of a gunn diode which will measure the received power density. Because MUSTANG will know the exact distance of the other space-craft through radio location or a laser ranger, as well as knowledge of the receiver's gain and size; the gain of the transmitter at 9.85 GHz can be calculated (received power density / transmitted power density.) The antenna's receiver gain can be calculated by doing the experiment in reverse with the MicroSAR tile receiving a signal from the other satellite of known power and using the same calculations.

Proposed experiment 3 : Intersatellite Radar for formation flying

This experiment is an extension of experiment 2 and relies on its results for the antenna gain of MicroSAR. For formation flying and collaborative control precise knowledge of the inter-satellite distance is vital for accuracy in SAR operation and interferometry. With the MUSTANG mission the spacecraft will fly in collaborative control up to 100m apart and with a drift of no more than 10s of centimetres. This accuracy will require precise measurements from the ranging system.

The MicroSAR tile will be used as a radar transceiver; a necessary step in the testing of its abilities as a SAR antenna. To range with antenna is relatively simple in theory, as with the altimeter the echo of a variety of signals; continuous, pulsed or frequency modulated, is analysed along with a measure of the signal strength returned from an object.

The experiment relies on the fact that on transmission the radar pulse will be diluted by the inverse square of the distance it travels. The signal is then reflected off the other satellite and reduced in strength due to the reflectivity of the MUSTANG spacecraft. A measure of this is the radar cross-section s. The return signal is then received by the first satellite and measured.

The effect of all these stages in the radar signal's passage between the two MUSTANG satellites is that the return signal has a weaker amplitude given, in theory, by equation 1.

$$P = \frac{1}{4\pi}\frac{P_{avg} G \sigma A_e t}{R^4} \qquad \textbf{(1)}$$

Where P is the return signal power, P_{avg} is the average transmitted power, G is the antenna gain (known from previous space/ground experiments), t is the time on target per pulse (effectively the pulse duration) which is, as mentioned before, dependent on the type of signal transmitted. R is the inter-satellite range and s is the radar cross section.

Equation 1 can be rearranged in terms of R. Measurements in space during the experiment along with ground tests on the reflectivity of the illuminated face of MUSTANG and knowledge of the signal pulse widths and strengths, can then provide the values of all the parameters needed to evaluate R. This would give the control systems for formation flying a value of the inter-satellite range to a certain accuracy and also test MicroSAR's ability to work with MUSTANG as a simple radar system.

Proposed experiment 4 : Downlink/ Communications antenna

This experiment is not so much a test of specific systems, but more of a test on the tile's use as a sub-system within MUSTANG. The justification is that beyond any of the other experimental aims, MicroSAR will be a redundant system with tremendous capabilities for use as a communications antenna. Its high frequency, high power and large bandwidth make it perfect for use as a downlink or inter-satellite communications antenna.

The MicroSAR design allows for any RF signal to be transmitted around the 9.85GHz carrier frequency with a bandwidth maximum of 300MHz. MicroSAR could be used either as the main earth/satellite link, replacing the chosen S-band systems.

Conclusions

A major aim in the design of these experiments was to perform simple tests that examine the MicroSAR tile's future use as a SAR platform. This could be as an individual radar as in the altimeter or as a complex collaborative control system for formation flying and multi-satellite SAR imaging.

In every case vital information fundamental to future missions and experiments is gathered. The demand on the MUSTANG mission and other payloads is minimal due to the small time required for the tests. Indeed, once testing is complete, MicroSAR can be left alone or even benefit the mission further as a sub-system. MicroSAR is launched untested and returns space worthy with minimal danger to the mission.

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MUSTANG: Differential GPS Techniques for Formation Flight

Alessio Marinelli
Space Research Centre, School of Engineering, Cranfield University

Abstract

The report will start with some theory and example that helped the author to understand fully the principal behind GPS. It then describes the errors affecting the GPS measurements and position estimates.

These principals are then apply to the MUSTANG mission and some techniques to improve the accuracy of the GPS systems are analysed, highlighting the best ones for our case. According to the chosen hardware and the techniques mentioned above, a general GPS mission outline is drawn.

Introduction

The report will take the reader through the basic principals of GPS describing with the aid of theory and example the processes involved with position estimate. All the errors affecting GPS readings will be described and the ones affecting this particular mission highlighted.

The reader will then go through a series of techniques involving DGPS and carrier phase, and even in this case the most useful one to this mission will be selected. This report requires a good level of mathematics to be fully understood.

The final part will deal with a brief re-cap of the characteristics of the chosen hardware and the integration of this and the chosen techniques to the mission outline.

The mathematics behind GPS and Differential GPS

GPS receivers measure the time delay from when the message is sent to when the message is received, since the message propagate at the speed of light, by multiplying the time delay with the speed of light, a range is obtained, the range is affected by errors and is therefore called pseudo-range.

$$\tilde{\rho}^{(1)} = \left[\left(X^{(1)}-x\right)^2+\left(Y^{(1)}-y\right)^2+\left(Z^{(1)}-z\right)^2\right]^{0.5}+c\Delta t_r+c\Delta t^{(1)}{}_{sv}+c\Delta t^{(1)}{}_a+SA^{(1)}+E^{(1)}+MP^{(1)}+\eta^{(1)} \quad \textbf{(1)}$$

$$\tilde{\rho}^{(2)} = \left[\left(X^{(2)}-x\right)^2+\left(Y^{(2)}-y\right)^2+\left(Z^{(2)}-z\right)^2\right]^{0.5}+c\Delta t_r+c\Delta t^{(2)}{}_{sv}+c\Delta t^{(2)}{}_a+SA^{(2)}+E^{(2)}+MP^{(2)}+\eta^{(2)} \quad \textbf{(2)}$$

$$\tilde{\rho}^{(3)} = \left[\left(X^{(3)}-x\right)^2+\left(Y^{(3)}-y\right)^2+\left(Z^{(3)}-z\right)^2\right]^{0.5}+c\Delta t_r+c\Delta t^{(3)}{}_{sv}+c\Delta t^{(3)}{}_a+SA^{(3)}+E^{(3)}+MP^{(3)}+\eta^{(3)} \quad \textbf{(3)}$$

$$\tilde{\rho}^{(4)} = \left[\left(X^{(4)}-x\right)^2+\left(Y^{(4)}-y\right)^2+\left(Z^{(4)}-z\right)^2\right]^{0.5}+c\Delta t_r+c\Delta t^{(4)}{}_{sv}+c\Delta t^{(4)}{}_a+SA^{(4)}+E^{(4)}+MP^{(4)}+\eta^{(4)} \quad \textbf{(4)}$$

The standard GPS positioning problems involves four unknowns that are determined with the following system of equations:

The measured pseudo-range linearised equations can be written as:

$$\begin{bmatrix} \tilde{\rho}^{(1)}(x) \\ \tilde{\rho}^{(2)}(x) \\ \tilde{\rho}^{(3)}(x) \\ \tilde{\rho}^{(4)}(x) \end{bmatrix} = \begin{bmatrix} \hat{\rho}^{(1)}(x_0) \\ \hat{\rho}^{(2)}(x_0) \\ \hat{\rho}^{(3)}(x_0) \\ \hat{\rho}^{(4)}(x_0) \end{bmatrix} + H \begin{bmatrix} (x-x_0) \\ (y-y_0) \\ (z-z_0) \\ c\Delta t_r \end{bmatrix} + \begin{bmatrix} \chi^{(1)} \\ \chi^{(2)} \\ \chi^{(3)} \\ \chi^{(4)} \end{bmatrix} + O(2) \qquad \textbf{(5)}$$

$$H = \begin{bmatrix} \frac{\delta\rho^{(1)}}{\delta x} & \frac{\delta\rho^{(1)}}{\delta y} & \frac{\delta\rho^{(1)}}{\delta z} & 1 \\ \frac{\delta\rho^{(2)}}{\delta x} & \frac{\delta\rho^{(2)}}{\delta y} & \frac{\delta\rho^{(2)}}{\delta z} & 1 \\ \frac{\delta\rho^{(3)}}{\delta x} & \frac{\delta\rho^{(3)}}{\delta y} & \frac{\delta\rho^{(3)}}{\delta z} & 1 \\ \frac{\delta\rho^{(4)}}{\delta x} & \frac{\delta\rho^{(4)}}{\delta y} & \frac{\delta\rho^{(4)}}{\delta z} & 1 \end{bmatrix} \qquad \textbf{(6)}$$

An example to prove this theory was carried out using MATLAB some of the results are shown in figure1.

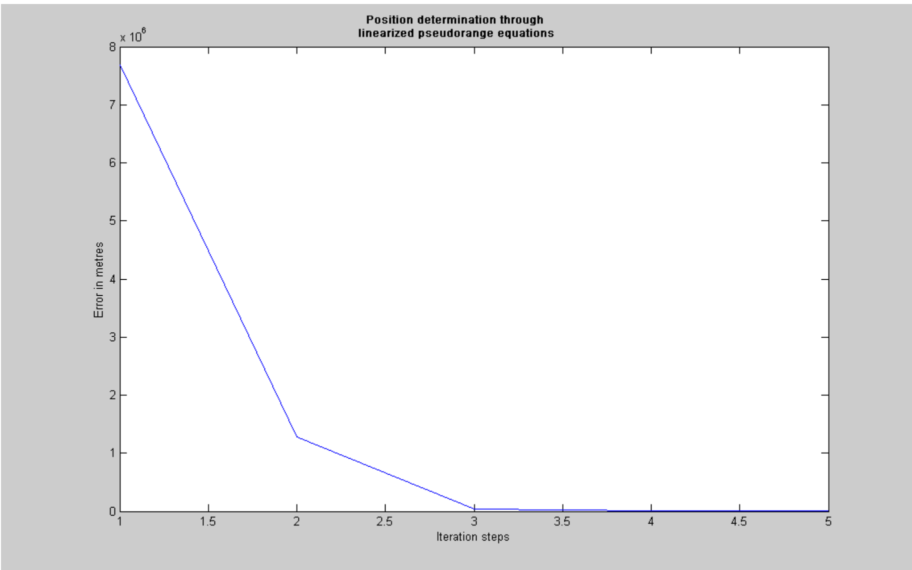

Figure 1 - Decrease in GPS receiver error with iteration steps

Figure 1 shows the rapid decrement of the error with the iteration steps. From the graph it is possible to notice how after only five iteration steps the position of the receiver is been determined.

The differential GPS and carrier-phase techniques analysed for this mission are summarised in the table below.

Table 1 - Summary of the propose GPS technique applicable to MUSTANG.

Technique	**Characteristic Equations**	**Possible accuracy**
Position-Space DGPS	$\delta\hat{x}_r = \delta x_r + (H^T H)^{-1} H^T (\delta\eta_r - \delta\eta_0)$	>10
Range-Space DGPS	$\tilde{\rho}_r - \tilde{\rho}_0 = r_r - r_0 + \eta$	~ 1-10 m
Double-Differencing	$\nabla\Delta\rho^{(i,j)} = (h^{(i)} - h^{(j)})(x - x_0)$	N/A
Phase Carrier	$\Delta\phi^{(i)}\lambda = (\tilde{\phi}^{(i)} - \tilde{\phi}^{(i)}{}_0)\lambda$	~0.1 or less

For the formation flight phase, it clearly appears that the optimal solution would be to use carrier phase techniques to achieve a precision down to the centimeter range, although the 1 cm range will not be ensured. However other differential GPS techniques may be tested to analyze the degree of precision obtainable. Finally a technique applied to a very recent mission has been integrated to the procedures summarized in the table above.

Operation

In the first part of the mission the two satellites will fly in a stacked configuration for few weeks after the launch.

From this disposition Position-Space, Range-Space and carrier phase DGPS techniques can be used to for attitude determination and the data obtained compared with the actual reading obtained from the AOCS.

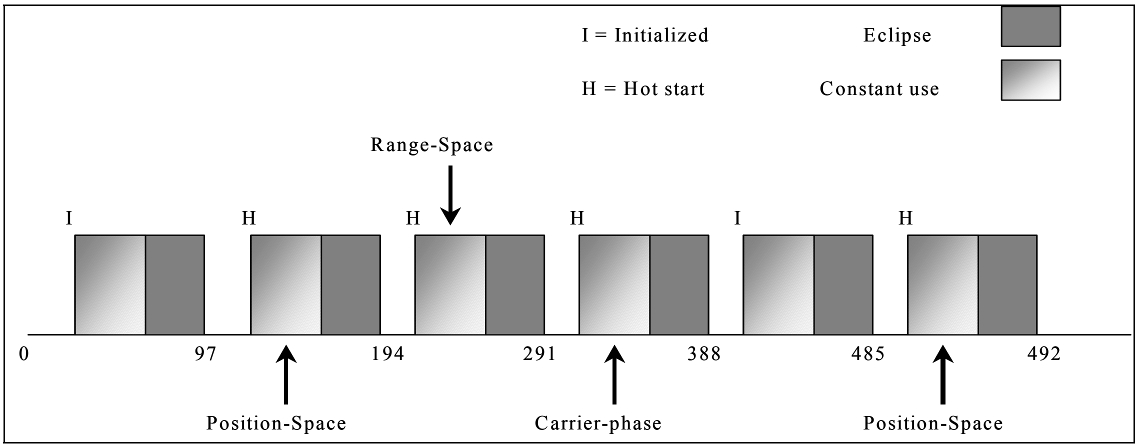

Figure 3 - Phase 1 mission breakdown.

In the second phase the two satellites will separate and reach a distance from one another of about 100 m. The only two techniques that will be used in this part of the mission are: range-space and carrier-phase. Now although the range-space technique can nowhere near achieve the accuracy of the carrier-phase, it would be interesting compare the actual deviation values obtained with the two techniques in space.

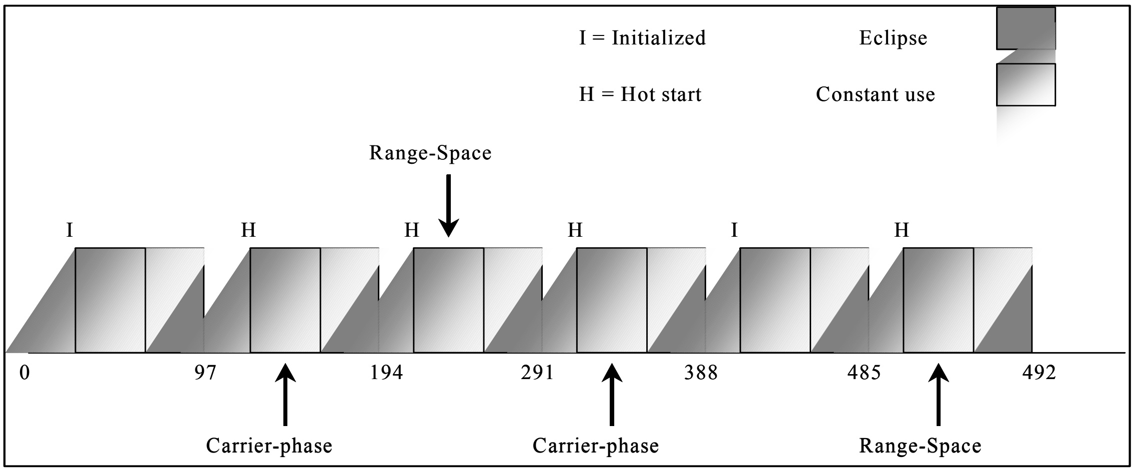

Figure 4 - Phase 2 mission breakdown .

The only technique used for phase 3, when the separation of the 2 spacecraft will reach 30 km, is going to be carrier-phase, because a part from being the most precise one, the other DGPS techniques are affected by larger errors when the distance between the two receivers is bigger then 10-15 miles.

Conclusions

The scope of this report was to move forward from what was done with regard to GPS on MUSTANG in the past years.

After a literature survey on the latest reports from Southampton and Cranfield the author decide to work on the mathematical side of the project to find a valid technique that would allow to achieve the highest accuracy possible.

The Orion-Emerald technique seems to be the most appropriate, both for the similarity of the two missions and for the level of accuracy achieved applying this method. However since the GPS is now considered to be a payload, it would be useful and interesting where possible, to apply all of the DGPS techniques described in this report, so to establish a good landmark of the size of the errors affecting the measurements for these types of missions.

C.3 Electrical Group

The electrical group's responsibilities include the power and data handling subsystems.

- Batteries (primary and secondary)
- Solar arrays
- Power conditioning
- Harness
- Data handling system: outline hardware and software design

MUSTANG: Battery System

Lorenzo Arona
Space Research Centre, School of Engineering, Cranfield University

Abstract

This report concerns the choice of batteries and related circuits, besides a first analysis of the electrical parameters involved. The analysis is forced to be only a introductory one both because the final configuration of the satellites is still likely to change and because the existing literature now by now is still fragmentary and quite often not available to people outside industries.

Introduction

Given that MUSTANG is supposed to be a technology demonstrator, Lithium-ion batteries -the newest technology nowadays available in the battery field- had been chosen to stock the energy collected from the solar arrays and supply it to subsystems and payloads during the eclipse time. This has, as usual, positive and negative aspects: sure it's true that the overpowering performances of this new technology make it definitely preferable to the older ones such as Ni-Cd and Ni-MH batteries, but on the other hand Li-ion technology is so new that a true and proper literature on the subject not even exists.

Rechargeable Li-ion batteries offer significant advantages compared to Ni-Cd and Ni-MH cells. Li-ion cells are lighter, about half the weight of Ni-Cd batteries, and 30÷50% smaller in size, yet have a volumetric energy density nearly twice that of competing technologies and, furthermore, do not suffer from many of the problems of Ni-Cd cells. For example, Ni-Cd batteries show a self-discharge of 20÷30% over a month, compared to the 5÷10% of a Li-ion and, most essentially, Li-ion batteries have no memory effect, the lack of a full charge based on earlier incomplete charge/recharge cycles. In addition, Li-ion batteries store three times the voltage of competing cells and, by weight, provide roughly twice the energy density of Ni-Cd cells. As a result, to supply the same output as a Nickel-Cadmiumd or Nickel-Metal Hydride battery, a Li-ion battery will weigh about half as much. By volume, Li-ion cells provide respectively about 25% and 40% more energy than Ni-MH or Ni-Cd cells.

Choice of the Cells

The first industry to study, develop and commercialise Lithium-ion technology has been the Japanese colossus Sony, that is still nowadays leading the market, thanks also to a good number of patents (actually Li-ion technology itself is licensed to Sony). Other factories such as AGM had only improved what Sony had begun, engineering cells with a greater capacity, using Sony cells as a development tool, but that are based on exactly the same working principle.

Other Li-ion batteries manufacturers are the usual giants in the electronic market, such as Hitachi, Sanyo, Motorola and Panasonic, even if other much smaller companies are quickly rising, conquering a non-negligible slice of the market. Unfortunately, none of the products of these manufacturers -that are studied mainly for the use in laptops and mobile phones- is space-tested, and so the choice is restricted to those companies that are somehow involved in space application, as to say AGM and Sony.

From every point of view, AGM cells seem to be the best candidate for MUSTANG battery system: thank to their outstanding energy density -up to 150 Wh/Kg- and to their very high capacity -up to 14.4 Wh. They not only are capable to storage a quite noticeable amount of energy in small volumes and weights, but also allow a very low complexity of the system (only a couple of these batteries are enough in order to feed all the subsystems and payloads during the whole eclipse time) lowering the probability of failures. Most of the batteries charge and discharge regulators that we can find nowadays are suitable only for a very small number of batteries (from 1 to 4), and so having a number of cells as low as possible gives less problems in the choice of the monitor circuit. On the other hand, unfortunately, these batteries at this stage aren't still formally space-tested, even if they are supposed to take the patent before the date fixed for MUSTANG launch. Moreover, it seems that Astrium UK -one of MUSTANG project main partners- is going to assembly a Li-ion battery regulator on purpose for the use of Sony Us-18650 cells; hopefully, this regulator, that will not be able to fly for at least another year, will be space-tested.

These are the main reasons for which Sony cells have been chosen, even if their performances are lower than those of AGM ones. Among Sony batteries, the US-18650 model has been chosen because it's the one with the highest energy density and because the literature concerning this particular type of cell is slightly wider than the others.

The Analysis

From the data collected on the net appears obvious that Li-ion batteries are meant to face in nominal conditions something like four or five hundreds full charge/discharge cycles, each of which is supposed to last roughly three or four hours. Given that MUSTANG is supposed to orbit the Earth on a 600 Kilometres high circular orbit, some quick calculations tell us that the orbital period is more or less 5800 seconds, which means a little bit less than 5500 orbits completed in one year expected lifetime of the spacecraft. This brings to two important considerations.

- We do not have time for a full charge/discharge cycle in one orbit.
- We must somehow bring the battery system in a situation that is able to face something like 5500 charge/discharge cycles.

Luckily, given that those two objectives do not exclude each other, there seems to be a unique solution: in fact changing the depth of discharge brings to both shorter charging periods and to an increase of the cycle life. From a simple analysis carried out with MATLAB, it can be shown that a reasonable DOD for MUSTANG batteries is 35%.

Once decided the DOD, we can properly size the battery pack. The restraint imposed is that the power required from all the subsystems and payloads is not supposed to change in sunlight or eclipse: this is both why there could be some payloads, such as MicroSAR, which, in order to be useful, are supposed to be continuously running and because the up-link and down-link could happen without distinction during sunlight and eclipse, according to what is the position of the ground station.

The analysis was carried out with the help of an excel spreadsheet in which the input were the energy collected by the solar arrays in one orbit and the payloads and subsystems power requirement: some of them are supposed to be always switched on while others have only an high power demand for short period of time. An average of the whole power requirement in the whole orbit has been done, in order to have a number representative of how much power is required each moment: this power has to be somehow achieved both during eclipse and during sunlight. Once we have the average power demand and the total energy furnished by the solar arrays, it's possible to calculate how much of this energy is going to feed all the subsystem and payloads in the sunlight period and how much is going to recharge the batteries. It goes without saying that this energy itself stored in the Li-ion cells is going to feed the same subsystems and payloads in the eclipse period.

At last, with some simple calculations, it's possible to foresee what's the total capacity needed in order to feed all the subsystem and so do a proper sizing of the battery system: as a computer-based simulation run by Dr. Rob Spurret of AEA Technology confirmed, six US-18650 cells are sufficient in order to face all the requirements both in LEO and GTO, in nominal or emergency working conditions. Another couple of cells were added for redundancy only.

The Battery Regulator

The battery monitor circuit has to manage some functions which are of the greatest importance for the survival of the satellite. In fact it not only has to feed the battery using that particular CCCV procedure but has also to avoid overcharge and overdischarge (the former for safety, the latter for performance), and to make sure that voltage and temperature of each cell are going to stay in the acceptable range. A failure in the regulator could bring to the loss of the battery system itself, and probably of the whole spacecraft. Given that it is such a capital component in the EPS, the most logical thing to do is searching for a circuit which has been qualified for surviving the hostile space environment.

We have in fact to consider that outside the atmosphere not only the temperature range is extremely high, according to the satellite position in relation to the Sun and the Earth, but also that the changing magnetic fields in LEO can bring to the presence of induced parasite currents in the circuit. Besides high-energy particle bombardment in higher orbits can decrease performance of PCSs, BCRs and BDRs until threatening the whole mission; furthermore, the lack of pressure brings to a faster ageing of all the materials and alloys the spacecraft is made up with.

Unfortunately, the literature concerning Lithium-ion batteries regulators is, if possible, even thinner than that concerning batteries themselves, and space-tested pieces of hardware are quite difficult to find: this probably happens because qualifying circuits for being able to survive the space environment is so expansive that satellite industries prefer to build them on purpose for each spacecraft.

The lack of available technical information about BCRs and BDRs reached sometimes frustrating levels but it is now possible to do the point of the situation: a research was carried on in order to find a monitor circuit suitable for being mounted in the nanosat PCS. This research didn't give good results: it seems that a COTS space-tested Li-ion battery charge/discharge regulator able to handle a power demand with an average value of just 13 Watts (and with a peak power demand that is not supposed to reach 20 Watts) at this stage does not yet exist. If it will be possible to find a space-tested regulator before MUSTANG launch date, it immediately should be mount in the EPS but, until that day, we have to go for an alternative solution: the suggestion is to use two or more regulators but dissociated by each other. Each of them could monitor only 4 cells in a 2p-2s junction, bringing to the constitution of two different and non-interconnected battery systems. The redundancy level of the system is good, in fact we can say that:

- If a string fails, the remaining six batteries will be more than enough to feed all the subsystems and the payloads.

- If a regulator fails, we lose also the four batteries connected: this will compel us to redefine the power budget, but the satellite will survive. Appendix E shows the allowable power management with only 4 cells running: the problem would be thermal, because almost a half of the energy collected from the solar arrays will have to be destroyed in the form of heat in the shunt circuit.

The total number of cells is eight (six plus two redundant), so two regulators able to monitor four batteries each will be mounted: the model chosen is the Maxim/Dallas Semiconductor MAX1737: the reasons for this choice are the high number of terminator methods of this regulator and its wide operating temperature range.

Primary Batteries

At this point no words have been spent on the primary battery system, which is made up with non-rechargeable batteries: the idea is to get rid of it and using only the secondary one. One of the many advantages of Li-ion cells over Ni-Cd and Ni-MH is in fact the very long shelf-life and the low self-discharge: this means that if we have the possibility to charge all the batteries about -say- three months before the launch, at the beginning of its life MUSTANG will have an available energy of roughly 30Wh.

This energy is more than enough to keep the satellite alive in the very first days of its mission, and however enough to run attitude control system, on board data handling, thermal control and communications for almost five hours, period in which the solar arrays will have already started to collect energy from the sun, even if the spacecraft is still not in his final orbit. So the decision to mount eight cells rather than six will make us save weight, because will allow us not to mount a non-rechargeable battery.

MUSTANG: Solar cells and power raising

Andrea Bennetti
Space Research Centre, School of Engineering, Cranfield University

Abstract

The objective of this paper is to describe the photovoltaic power system developed for the Mustang satellite. Mustang is a University led space project that aims to demonstrate “formation flying “ and Micro System-technology (MST) application in space. The report analyses the inherited configuration for the power raising system and describes the development steps performed to reach the final configuration. The primary source of power during sunlight operation is provided by the body mounted dual junction $GalnP_2$/GaAs/Ge solar cells. The solar panel layout accounts on 114 solar cells divided in 38 strings of 3 solar cells. The power system developed is analysed and proven sustainable through a series of numerical simulations. A sensitivity study is also performed to evaluate the efficiency of the solar cells spatial arrangement in the satellite external configuration. Protection schemes and testing methods for evaluation of solar cells mismatching problems are. An investigation for different temperature regimes is also performed to establish the dependency of the solar cell electrical parameter for different operating temperatures

Introduction

The author was involved in the continuous design review of the power system for the Mustang satellite, particular attention was paid to the power generation throughout the use of solar arrays. The design review took into consideration the interfaces between the power system and the rest of the satellite subsystems in an iterative fashion with weekly updates. The tasks the author attempted to complete during the project are here briefly listed:

1.Perform a critical review of the inherited power generation system
2.Power generation for LEO orbits (different possible cases)
3.Power sustainability
4.Components identification, selection and ordering
5.Evaluation of solar cells performances
6.Evaluation of solar cells shadowing problems
7.Design of an electrical test rig
8.Components assembly & testing

The inherited configuration & design review

Body mounted solar arrays were chosen because of their relative mechanical simplicity compared to deployable ones, and because of their obvious mass benefits. The solar cell type chosen for the Mustang solar panels was a $GalnP_2/GaAs/Ge$ dual junction solar cell. The power raising system originally developed counted on 402 cells connected in 67 chains of 6 cells. The desire to make Mustang a much more attractive satellite platform for candidate payloads was the main driving requirement for improving the Mustang power system. On top of this commercial reason, an analytical review of the results obtained in the previous year, revealed these to be slightly too optimistic. Using an Excel spreadsheet developed by Astrium Ltd. the author re-performed the simulation to obtain the Mustang power profile for the worst-case scenario with some updates (introduction of new payloads) to find out that the system was not sustainable.

Design Update & final configuration

During the design update, it was decided to change the dimensions of the solar cells to be used. The dimensions of the solar cells previously proposed were 20 by 40 mm whilst the dimensions of the newly considered solar cells were 39.5 by 68.9 mm. The decision to use these new cells was driven by two main considerations:

- The new dimensions proposed for the solar cells are the standard dimensions in which the solar cells are provided by Spectrolab. By using these standard cells additional cost for custom-made solar cells could be avoided. Further more during the test phase of the project it could be possible to obtain some scrap cells (of nearly identical specification) from Spectrolab for testing.
- The number of cells is drastically reduced, hence reducing harnessing problems, and reducing the number of electrical connections, which are, always the cause of electrical dispersion.

The power raising system for the reviewed Mustang configuration (fig.1) is based around dual junction $GalnP_2/GaAs/Ge$ solar cells. These provide an energy conversion efficiency of 21.5%. Mustang final solar panel accounts on 114 cells arranged in 38 strings of three cells (fig.1). Each cell is provided with its own bypass diode and each string is protected with a blocking diode.

Simulations results & solar cell analysis

An investigation on the power raising system performances for Mustang was performed The analysis was conducted trough mean of simulation, starting from the most significant condition, the worst-case scenario, and then moving to analyse the satellite performances throughout a complete 1-year mission. In parallel to the power analysis, a sensitivity study was also performed on the current solar cell layout, identifying strengths and weakness of the developed configuration. In the solar cell analysis section are reported results of an analytical investigation on the electrical parameters behaviour, for the chosen cells at different temperatures [Kreifels], [Lorenzo]. The analysis performed demonstrates that energy conversion efficiency, current for max power point and voltage for max power point for the dual junction GalnP$_2$/GaAs/Ge solar cell chosen, vary linearly with the temperature.

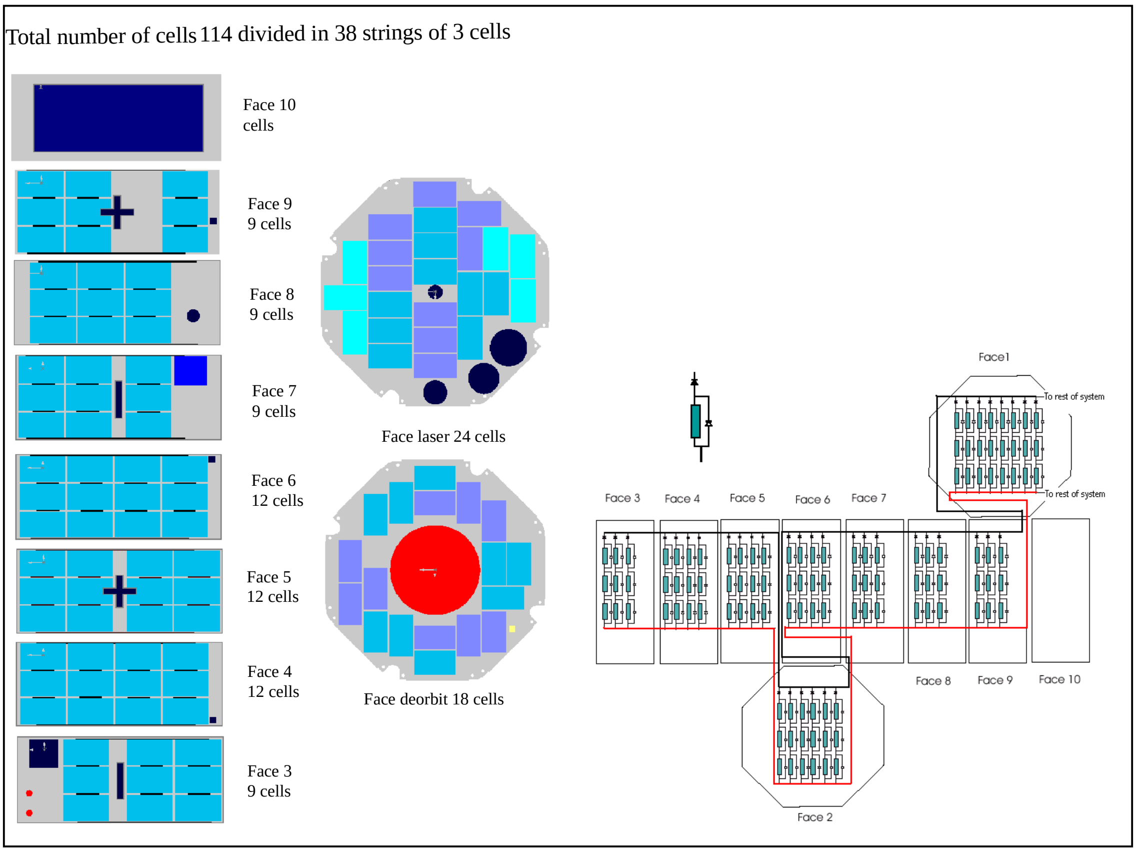

Fig 1-Final solar cells layout and circuitry arrangements

Conclusions

A critical review of the inherited configuration was performed demonstrating the power system unsustainable. The necessary changes were implemented, these include: The use of a different type of dual Junction GalnP$_2$/GaAs/Ge solar cell, with increased dimensions (39.5 by 68.9 mm) and increased energy efficiency (21.5%), the re-arrangement of the external satellite layout, (solar cells spatial disposition in particular) to accommodate for the solar cells dimensions change. Power generation for LEO orbits and systems sustainability were analysed through means of simulation, and system sustainability for the updated configuration, was achieved for the suggested LEO polar orbit. The new solar cell type to be used was identified and an order was placed to Spectrolab for their acquisition. A sensitivity study was performed to establish the different contribution from the solar panels to the total power raised in the worst-case scenario and during the entire length of the mission. Results indicate that to improve Mustang power performance efficiently, for the worst case scenario condition, changes should be made to increase the solar cells covered area of the two octagonal faces, whilst to increase performances during other mission periods the solar cells covered area should be increased on faces 5 and 9. The solar cell performances and the influence that temperature exerts on them have been investigated, highlighting the existence of a linear relationship between the solar cell electrical parameters and the cell operating temperature. Solar cells shadowing problems have been considered in the design of the power raising system, and protection schemes have been suggested. The 114 cells arranged in 38 strings of three cells are provided with a bypass diode each, and every sting is protected by a blocking diode. In conclusion, all of the objectives have been meet with the exception of the components assembly and testing. Failure to complete this task was due to the long lead-time required by the manufacturer to provide the ordered solar cells.

Further work

The author suggests that further work should look at improving the power analysis for the complete mission length, possibly by introducing a detailed payload schedule. The electrical circuitry design should be finalised with the identification of opportune blocking diodes for the strings protection. Solar panels assembly procedures should also be identified. The final electrical testing of the Mustang power system should be performed as described by Abete [Abete[2]]. Two typical experimental tests should be effectuated on the solar panel, one with totally irradiated cells and the other with one or more shaded cells. Other procedures [Abete] reveal the possibility to determine the short circuit current and the open circuit voltage of each cell and, what is more interesting, a practical factor which can characterize the quality of the solar panel.

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MUSTANG: On-Board Data Handling

David Deering
Space Research Centre, School of Engineering, Cranfield University

Abstract

Within this paper is presented a new high-level OBDH topology for Mustang built around the three key words: COTS, Fault-Tolerance and Modularity. The main objective of this work has been to investigate the initiation of an OBDH breadboard. Within this context, an investigation into the fault tolerance of COTS products and the techniques to compensate for the fault tolerance weakness in COTS standards formed the main focus of this study. Testing the most appropriate methods for Mustang will be the purpose of the breadboard. In the main, the original Mustang I^2C data bus design was found to be susceptible to single point failure. Therefore, a Hybrid, Layered & Distributed Architecture concept built around COTS data bus fault tolerance and distributed computing modularity has been presented, wherein the fault-tolerance of COTS standards has been realized.

Introduction

A breadboard is a functional laboratory layout of the flight and test OBDH system implemented in hardware. So, why want to build an OBDH breadboard? The rationale is simple. Progress. What a breadboard would provide is the framework to perform basic data transfer demonstrations, for building and performing trade-offs, leading eventually to the final test model for the Mustang OBDH flight system. Additionally, most problems only come to light when trying to put into practice a concept. For the purpose of this work, this means the previous high-level OBDH design of Medrano, (2001). Undeniably, while studying techniques to implement the first stages of basic data transfer, a number of problems have arisen. These primarily concern the fault tolerance of the chosen Philips I^2C data bus. Therefore, the majority of this work is built around the implementation of COTS products, i.e. the I^2C bus among others.

Data Bus

Two data buses have been primarily chosen. From Medrano, (2001), the I^2C bus, a synchronous serial protocol that uses only 2 wires, data (SDA) and clock (SCL) It has very low power consumption (0.01W/Node), an adequate data rate (100kbps and 400kbps), a simple protocol that supports multi-master capability, and strong commercial support. In addition there are advantages of using the Dallas 1-wire bus both for basic telemetry and to complement the I^2C in fault recovery. It provides a simple signalling scheme that performs two-way communications with peripheral devices over a single connection.

Fault Tolerance

Data Bus Fault Protection

In the first stages towards the implementation of an OBDH simulation, an overall COTS strategy towards I^2C fault tolerance has arisen as a possible concern. This quickly became a prime objective of the project. A COTS fault-tolerant strategy based on the X2000 experiences has been realized, wherein four primary levels of fault tolerance enhancement are being envisaged: Native Fault Tolerance; Enhanced Fault Protection; Fault protection by Design Diversity and Fault Protection by System Level Redundancy, (Chau et al, 1999). First use the native fault-tolerant features of the I^2C bus to detect fault occurrences. This corresponds to the Acknowledgement bit, ACK, of the I^2C protocol. Enhanced Fault Protection follows where an additional layer of hardware or software protocol enhances the fault detection and recovery capability of the bus. An added layer of protocol provides the simplest enhancement. The basic layer could include a byte count, a cyclic redundancy check (CRC), or critical commands sent followed by their complement. A primary concern of the I^2C bus is a stuck-at-low fault on the SCL signal line. To recover from this failure mode every node needs to include a byte timeout timer to monitor the duration of the SCL signal. Design Diversity addresses the fundamental fault tolerance weaknesses that usually relate to single points of failure. The Solution is to use a double-bus arrangement: One bus to assist in the isolation & recovery of the other bus, i.e. Dallas 1-wire to perform basic I^2C node and health interrogation. There is one failure-mode here that is of particular concern to the Mustang COTS methodology, known as Conflict of Node Addresses. This mode can have unpredictable consequences and is difficult to recover from by a single bus itself. Finally, the set of busses is duplicated at system level to provide redundancy for fault recovery. In this case, if a simple procedure cannot correct the problem then the backup set of buses is activated and the system operations are transferred to the backup bus. This leaves more time to diagnose the failed bus-set.

COTS Microcontroller Fault Tolerance

The PIC16C73 from Micron, or a close relative, is primarily chosen for distributed computing on Mustang. Under radiation tests (Bezerra et al, 2000), the PIC16C73 SEU sensitivity was quite low with good latch-up behaviour. Under heavy ions no corruption of the code was noticed. For total does calculations on unpowered parts, no parametric drift or functional failure was observed. A simple method for fault protection of the PICmicro against SEU and SEL, is an anti-latch-up system that consists of monitoring the Idd current, and switching the PIC off when a threshold is passed (3 to 4 times its standard value) and resetting the board.

Hybrid, Layered & Distributed

A Hybrid, Layered & Distributed Architecture approach to COTS products, Fault Tolerance and Modularity, combining the distributed data bus lessons of the JPL X2000 programme with the distributed computing lessons of the Stanford Emerald programme, has been realized. Figure 1, illustrates the high-level data bus topology wherein Fault Protection by Design Diversity and System Level Redundancy have been implemented. In addition to Design Diversity, Caldwell, (1997a), has proposed the Dallas 1-wire system as a solution to distributed temperature sensing on small satellites. This has been considered as a very simple and attractive approach option to Mustang development. One interesting additional point to note is the possibility of using the communications system to route the high-level commands to the subsystems in the event of CPU failure. This can easily be accomplished by using the inter-satellite link to forward the scheduled commands from the fully operational satellite to the partly failed satellite.

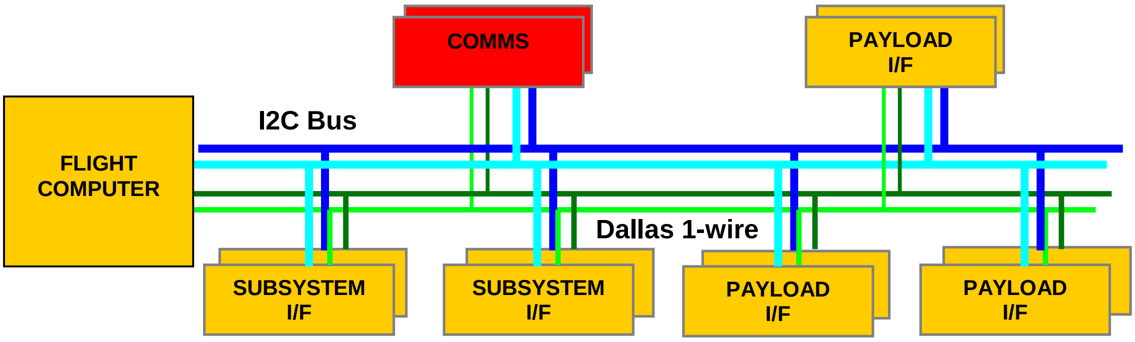

Figure 1 – Hybrid Layered Distributed OBDH Architecture. A central flight computer controls a network of smart subsystems over an I^2C bus. Dallas 1-wire bus is used for fault recovery by design diversity and for basic telemetry, such as temperature sensing.

In addition to the Distributed I^2C and Dallas data-bus topology, smart subsystems are considered. This enables the central flight computer to interact with the subsystems via high-level commanding, with each subsystem node performing its own application specific computing. This offloads the central flight computer from having to be concerned with the more menial low-level tasks. As such the central CPU takes on the role of a co-ordinator for a network of intelligent bus nodes. A command such as "c(1)" sent from the central CPU to a subsystem could indicate the subsystem to collect data every second and store it in local memory until issued a Stop condition, (Townsend, 2000). Hence, if a payload development is running behind schedule, a payload needs to be reconfigured in the last stages, an extension of capability is desired, or even a late bug fix is required, in the case of distributed computing it actually has no effect on the rest of the spacecraft or even, more importantly, on the flight computer.

Implementing a Breadboard

Two PCI cards, by Calibre UK, which interface a computer with the I^2C bus, have been purchased, and the Microchip, MPLAB IDE, Integrated Development Environment, is available for writing software for PICmicros. Tools that have not yet been purchased but are key to development are: A complete "hands-on" CD C-language Programming & Development course for PICmicros, complete with a PIC Tutor development board and PICmicros. PICmicros and a PICmicro chip programmer, such as the EPIC or WARP-13, which is required to get the compiled HEX files onto the chip's memory.

Immediate future work is recognised in the form of three phases, for which high-level tools have been identified previously. Firstly, the set-up of an I^2C bus circuit over which desktop PCs can communicate and demonstrate the basic features of I^2C data transfer and fault enhancement. Secondly, an understanding of programming microcontrollers needs to be realized, along with demonstration microcontroller communications over an I^2C bus, and the Dallas 1-wire distributed temperature sensing system. Thirdly, demonstrating distributed computing and combining the I^2C with the Dallas-1 wire for fault recovery.

Conclusions

A Hybrid, Layered & Distributed Data-Bus Architecture has been identified as a possible solution to the most common or critical failure modes of the I^2C bus, wherein COTS products, modularity and fault tolerance have been realised. Distributed Computing can modularise Mustang to the extent that subsystems can be swapped and updated without affecting the central bus. Combining the three learning phases associated with the breadboard tools shall evolve into the OBDH breadboard, followed by the electrical and flight-test model. However, the question must be raised as to whether it is plausible for Cranfield to design, develop, build and test the Mustang OBDH system within the Mustang credibility timescale: launch readiness by end of 2003. Careful thought should be given to purchasing a ready-built small spacecraft modular system such as the AeroAstro “Bitsy Spacecraft Kernel”, (McDermott, 1999).

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MUSTANG: Power Management & Harness

Paul Greenwood
Space Research Centre, School of Engineering, Cranfield University

Abstract

A 7V, 4A direct energy transfer, quasi-regulated bus was chosen for the MUSTANG electrical power system (EPS) and its functions were defined. The power control unit (PCU) incorporates a wire wound resistor and a power transistor along with various electronics for sensing and switching, its specific functions were also defined. The harness for the power, OBDH (onboard data handling) and telemetry were identified along with connectors for the harness network.

Introduction

The main aims of the "Power Management & Harness" work package were to design an EPS for MUSTANG along with a PCU, identify possible components/harness and investigate the physical layout of the harness. MUSTANG accommodates many payloads (experimental payloads included) as well as subsystems that all have very varied power requirements ranging from 3.3V to 12V and 38mA to 0.9A. The highly variable power demand was one of the main driving forces behind the design of the MUSTANG power system.

The Electrical Power System

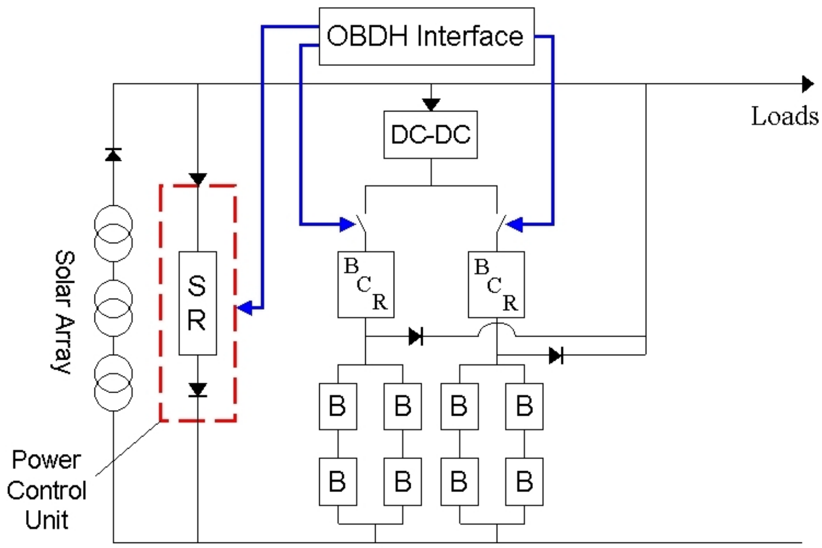

Figure 1 – Power bus schematic showing a DC-DC converter, two battery charge regulators (BCRs), eight batteries (B) and a power control unit containing a shunt regulator (SR). Diodes are included to prevent reverse charge flow.

The power system (figure 1) chosen for MUSTANG was a direct energy transfer system, quasi-regulated bus incorporating shunt regulation. The power bus is a 7V, 4A (28W) power bus, the voltage variation from the solar arrays was calculated to range from 6.75V to 7.08V at most and the battery voltage supplied to the bus was on average 7.2V. The total current drawn from MUSTANG's systems and payloads was estimated to be no more than 4A as the full power details of the subsystems and payloads were unknown. Two BCRs were incorporated into the EPS for redundancy reasons chosen by Arona, 2002 and are operated (i.e. switched on or off) by the OBDH subsystem. The electrical power system's functions were as follows:

- Generate sufficient energy from solar arrays.
    - 114 Spectrolab $GaInP_2/GaAs/Ge$ dual junction cells with an efficiency of 21.5% delivering a maximum voltage of 7.08V.
- Store the energy in batteries.
    - 8 (3.6V) Sony US18650 cells with a total capacity of 12A-hrs (1.5A-hrs each) and total energy storage of 43.2W-hrs (5.4W-hrs each).
    - 2 Maxim Max1737 BCRs to charge the batteries.
    - Redundancy (i.e. switching) of the BCRs is controlled by the OBDH.
- The EPS has to be robust, reliable and provide power during sunlight and eclipse.
    - During sunlight periods, power will be distributed to the subsystems and payloads directly from the solar arrays. Excess power will be used to charge the batteries to full capacity within the 61.2 min sunlight period.
    - During eclipse, power will be drawn solely from the batteries where the total depth of discharge (DOD) will be ~22.5% (Arona, 2002). There will be no shunting during eclipse, as this will be a waste of battery power.
- Control the distribution of power to all subsystems and payloads.
    - Power is taken from one power bus with switching at each subsystem and payload power bus interface by the OBDH system.
    - Latch-up circuitry is located at the power bus interface.
- Regulate the power appropriate for all subsystems and payloads.
    - Local DC-DC power conversion is used.
    - Excess power is "shunted" by a shunt regulator.
- Communicate power system status/health to the OBDH system.
    - The OBDH system is continually monitoring the state of the power bus, BCR's, subsystems and payloads.

Power Control Unit

The power control unit pictured in figure 1 is further illustrated in more detail in figure 2 below.

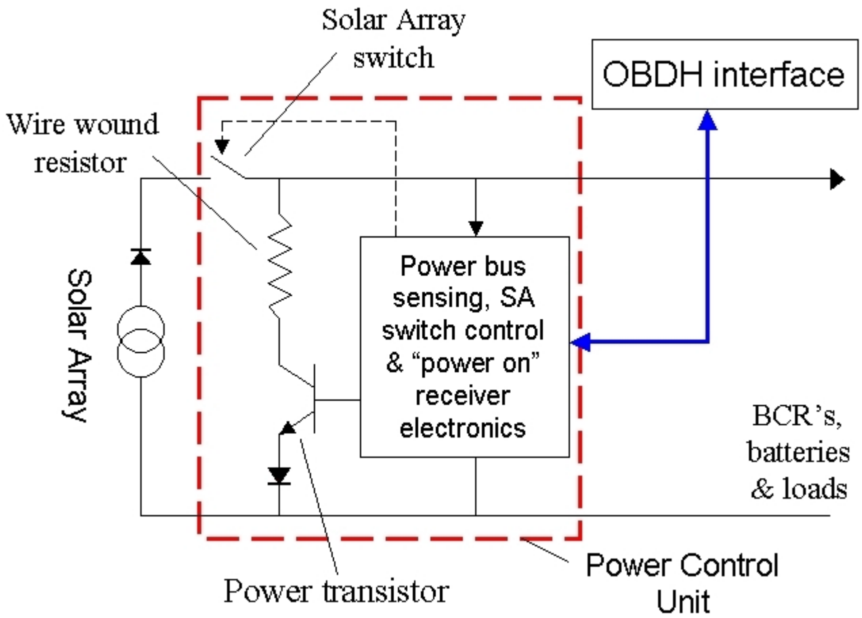

Figure 2 – Power control unit containing a wire wound resistor, a power transistor and power bus sensing, solar array (SA) switch control and power on receiver electronics.

The functions of the conventional PCU were decentralised for MUSTANG i.e. power conversion, latch-up, power switching and battery regulation takes place outside of the MUSTANG power control unit. The highly variable power demand was one of the main reasons for choosing a decentralised power system. The functions of the power control unit were defined as follows:

- Switch the solar arrays off during launch and during idle times before launch.
    - To prevent any power flow to the power bus for safety when not in use.
- Switch on the solar arrays and power on the satellites after separation from the launch vehicle.
    - Power on would involve the PCU to be on constantly during launch. Post launch & separation when the PCU receives a signal from the ground for example, the PCU would switch on the OBDH subsystem that will in turn power on the other subsystems and payloads (Smith, 2002). The OBDH switching can be done over the OBDH-PCU connection.
- Sense power on the power bus for shunt regulation.
    - Pre-programmed power level above which to signal shunting.
    - The shunt would not have to shunt the total power of the satellite, as the solar arrays can just be switched off if the need should arise.
- Shunt any excess power when the power level rises above the pre-programmed level.

- The shunt has to shunt between 0.4W – 10W taken from Bennetti (2002).
- The OBDH system can override the pre-programmed power level to shunt power at any time or act as a backup for redundancy (for the pre-programmed power level) and be done over the OBDH-PCU connection.

Harness

The chosen harness for the power was the Tyco Electronics Raychem Spec 44 wire with an American wire gauge (AWG) rating of 18. It has one twisted pair inside a braided screen. The OBDH harness was based on the Belden 8103 cable, 3 twisted pairs inside a Beldfoil (aluminised polyester) shield and a braided screen with each wire having a rating of AWG 24. One possible candidate for the telemetry bus was chosen although the sizing requirements of this bus are unknown. The Belden 9841 with the same characteristics as the OBDH harness but with one twisted pair was chosen for telemetry.

Screw terminating, 9 and 15-pin, male D-connectors were chosen for the cable endings on the power and OBDH harnesses respectively. 9 and 15-pin female, 90^{o} PCB terminating D-connectors were chosen for the sockets on the interfacing PCBs. They were chosen for their good mechanical interlocking to withstand launch vibrations.

The current layout of the harness was done by Michel (2002), however the layout of the power harness was revised to allow for latch-up circuitry and power switching to be located close to the ODBH network so that the OBDH subsystem can do the switching. It was also suggested that the latch-up and power switching circuitry could be integrated with the OBDH interfacing boards so that the space on the vertical shelf is more efficiently utilised. The telemetry bus is currently omitted from the internal configuration, as its routing is unknown.

Conclusion

An EPS and PCU allowing for EPs were designed and whose functions met the requirements of the subsystems and payloads. Harness and components within the PCU were identified according to their required function. Future work includes bread boarding, breadboard testing and an investigation into the telemetry bus routing as well as the harness routing for the externally mounted systems/payloads which currently has not yet been investigated.

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C.4 Mechanical Group

The mechanical group's responsibilities include

- Structural design
- Structural analysis
- Thermal analysis
- Mechanisms
- Alternative design (change the spacecraft cross-section from octagonal to square)
- Enhancements to the baseline (extra solar arrays to raise additional power)

MUSTANG: Separation Mechanisms

Elie Allouis
Space Research Centre, School of Engineering, Cranfield University

Abstract

The MUSTANG Separation Mechanism task provides all the necessary mechanisms to perform the launcher and intersatellite separation. Since the launcher for the mission hasn't been identified, MUSTANG must be able to be cope with different launcher interfaces, and a versatile launcher adapter is therefore required. This MUSTANG Launcher Adapter or MLA provides the separation mechanisms to achieve the baselined separation velocity, tilting and actuation speed at low mass, cost and power. The design of the MLA is based on a ring mounted onto the launcher interface. It houses 4 pinpullers that maintain the spacecraft in place by a clevis release configuration combined with 4 vertical springs to provide the kick-off impulse. Specially designed brackets provide structural points on the spacecraft, the clevis for the separation mechanism and mounting for the bulkhead of the spacecraft. The MUSTANG Intersatellite Separation System or MISS is based on a single point release on the main axis of the spacecraft where lay the two centre of gravity. The separation device called the Ejector, a SMA device mounted on the bottom satellite, releases a component fitted on the other spacecraft, releasing the spacecraft as well. An arrangement of cup/cone and button/post devices combined with a set of 4 springs provides the alignment of the spacecraft and the required separation velocity. This solution meets all the baselined constraints in term of velocity, tilting and mass, but also at a low power and cost.

Introduction

MUSTANG is not launcher specific and must be versatile enough to be accommodated on several launchers depending on the launch opportunities. To give the spacecraft this relative versatility, a launcher adapter must be designed to provide the spacecraft the required separation velocity, tilting and actuation speed at low mass, cost and power. An intersatellite system is also required to separate the two spacecraft in flight while optimising as well the separation velocity, tilting angle and actuation speed for a low mass, power and cost.

Design of Structural points for the mechanisms

Some structural points have been defined on the spacecraft to provide safe load paths for the separation mechanisms. Shaped as brackets, these are bonded and riveted to CFRP tube at the corners (Figure 1). Their role is also to provides an efficient way of mounting the bulkheads allowing easy removal if needed. Few designs have been produced to answer specific needs: Simple bracket for bulkhead mounting, Intersatellite bracket with alignment device mounting, and Launcher separation bracket featuring a clevis.

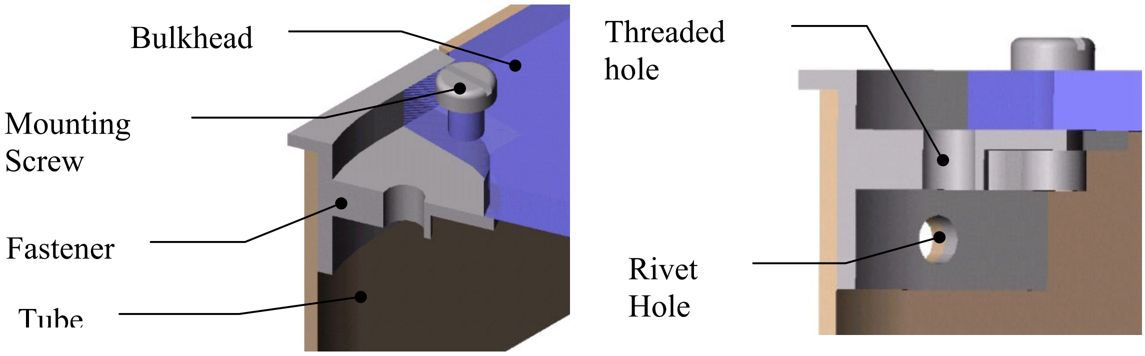

Figure 1-Example of a bracket mounting,(here an intersatellite bracket). Section view through a corner of the spacecraft showing a bracket fixed on the tube and the bulkhead mounted on the top of it (shaped to accommodate the spring and alignment device).

Design of the MUSTANG Launcher Adapter or MLA

Using proprietary launcher interface such as the Ariane ASAP 5 is not an option taking into account the high impact on the geometry of the spacecraft, mass, shock at separation and tilting angle. Moreover, the spacecraft lacks the ability to be easily mounted and need heavy redesign. Therefore, a launcher adapter is needed to fulfil all these requirements and the specific constraints of the mission such as separation velocity. The MLA design is based on a ring that can be mounted on the launcher payload interface. Four pinpullers and their housing are mounted onto this ring to provide the clevis release system as shown in figure 2:

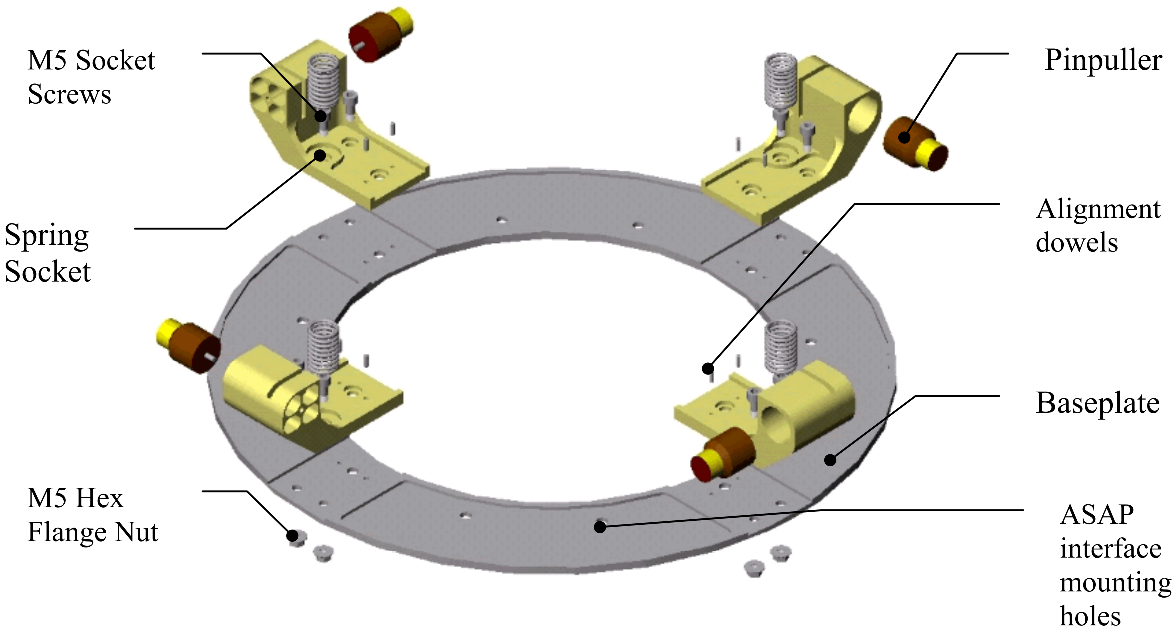

Figure 2- MLA Assembly showing the base, the pinpullers and their housing

The base is an aluminium plate, 5 mm thick pocketed to 3 mm on most of its surface. Each separation mechanisms are aligned on the baseplate by two dowels and bolted

with 2 M5 socket screws onto it. Alignment of the separation mechanism with respect to the baseplate is a critical issue as it influences directly the alignment of the pinpuller with respect to the clevis. The springs, dimensioned to provide 13 mm/s velocity at separation are then bonded in the dedicated pocket on the pinpuller housings. Polymer damper in conjunction with the springs may be used to damp the shocks during launch.
The operation principle of the clevis release system is simple. The pin of the pinpuller maintains in place the clevis. When the pinpuller is triggered, the pin retracts and let the clevis disengaging as shown in *figure 3*.

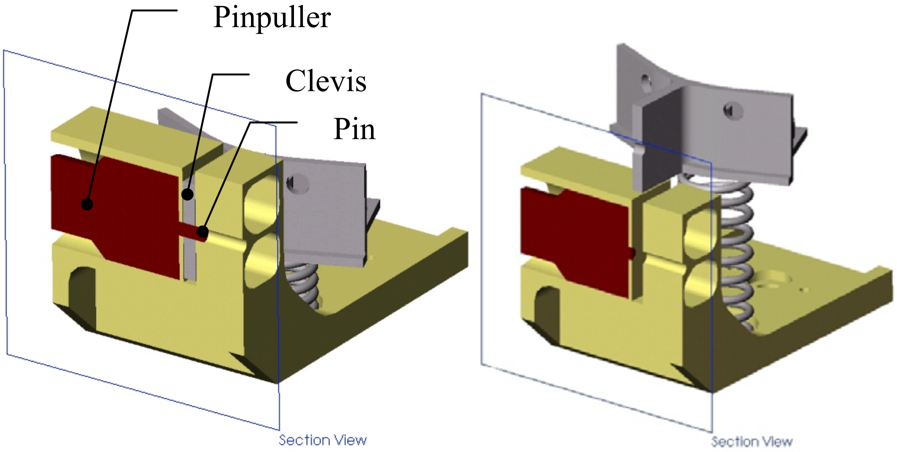

Figure 3 - Clevis release system operation. The system is set (left), when the pin retracts, the clevis disengage (right)

Design of the MUSTANG Intersatellite Separation System or MISS

The MISS is based on a single release point on the main axis of the spacecraft where lay the two centre of gravity. The separation device called the Ejector, a SMA device mounted on the bottom satellite, releases a component fitted on the other spacecraft, releasing the spacecraft as well. An arrangement of cup/cone and button/post devices combined with a set of 4 springs provides the alignment of the spacecraft and the separation velocity of 7mm/s, (figure 3). The Ejector is preloaded when the two satellites are stacked and assembled with a turnbuckle with a left and right thread (figure 4).

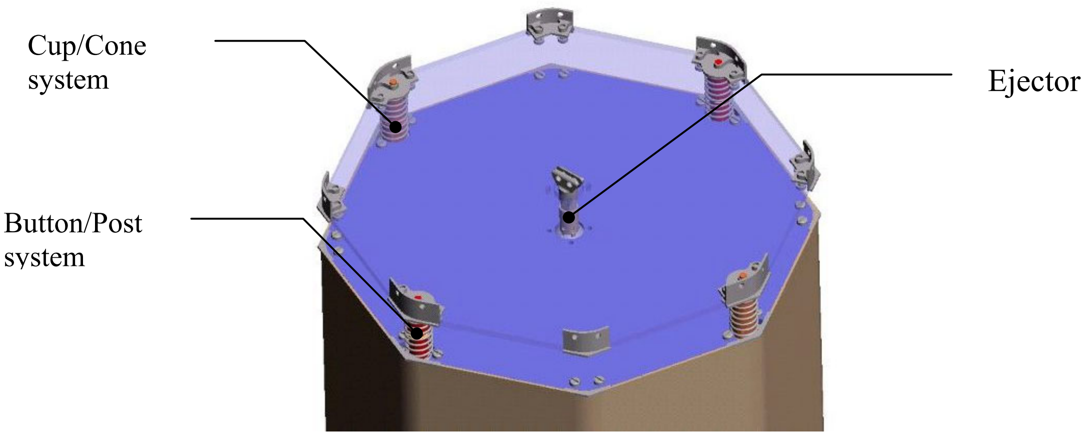

Figure 3 – MISS configuration with the Ejector, cup/cone and button/post devices

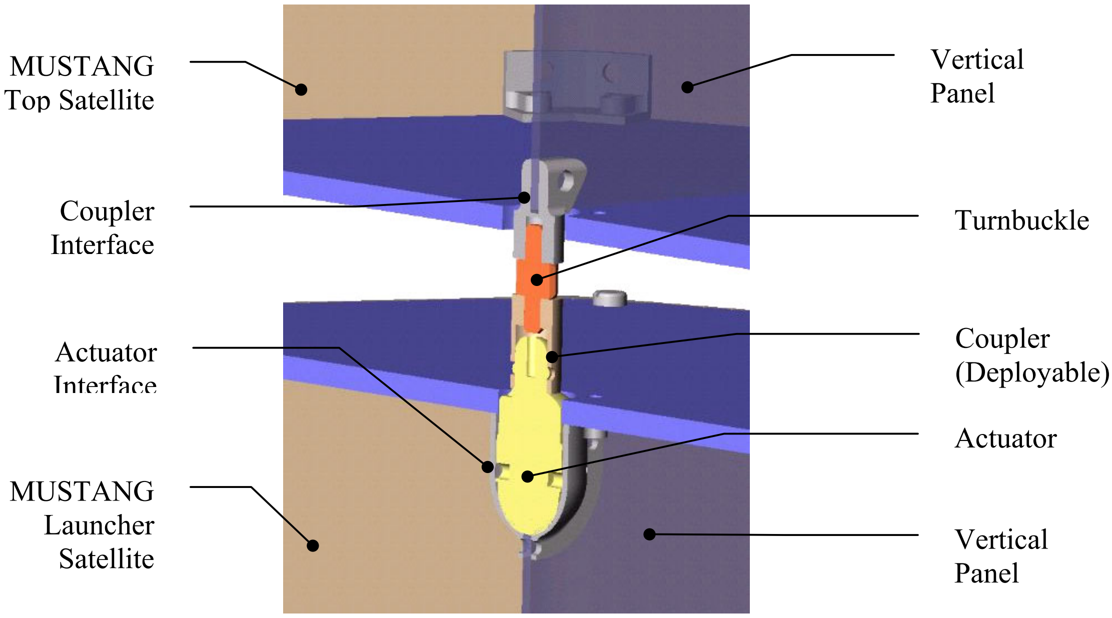

Figure 4 – Ejector mounted on the vertical panel by interfaces and preloaded by a turnbuckle

Conclusions

The MLA and MISS fulfil all the baseline requirements as shown in table 1. Moreover, an analysis of the failure modes indicates that the designs are sound and a critical failure remote. Sets of detail drawings are available for most of the MLA and MISS parts, emphasising the need of hardware soon if MUSTANG is to be on schedule. A detailed FE and vibration analysis of the critical component is still needed, as well as the testing of the various devices proposed.

Table 1: Summary of the separation mechanism task in term of mass, velocity, power and cost.

	MLA			Intersats		
Mass	Pinpuller	4 x 30	120g	Ejector	85	85g
	Baseplate	550	550g	Cup/Cone	2 x 20	40g
	Fixtures	4 x 100	400g	Button/Post	2 x 20	40g
	Bolts & Springs	20	20g	Springs	15	15g
	Total (Staying on Launcher)		1090g	Total		180g
Velocity	4 Springs		13mm/s	4 Springs		11.5mm/s
Power	Per Pinpuller		6 amps x 50 msec	Ejector		6 amps x 50 msec
Cost	Pinpuller	4 x 3700	14800$	Ejector	1 Unit	4000$
	Springs	4 x 4.2405	16.962$	Springs	4 x 4.2405	16.962$
	Total interface Cost		14817$	Total Interface Cost		4016.962$
	Total Sep Mech Cost		18833.9$			

MUSTANG: Thermal Analysis

Paul Coutinho
Space Research Centre, School of Engineering, Cranfield University

Abstract

The focus of this paper is to present the thermal analysis for the satellites which were part of the MUSTANG project. The objectives of the thermal analysis are to create a thermal model to predict temperatures on the spacecraft and to use the thermal model to validate a thermal control strategy. The thermal model used to analyse the satellite was created using a software package known as IDEAS-TMG. This thermal model showed that without a thermal control system the temperature limits of the various equipment onboard the satellite would be exceeded. The thermal model was then used to evaluate different thermal control strategies. The most appropriate strategy relied on isolating the inside of the spacecraft from the outside. Conduction paths were then used to control the heat transfer. This particular strategy when incorporated into the thermal model was shown to improve the onboard temperatures. Although this strategy in its current form goes a long way at improving the temperatures it does not satisfy all temperature criteria and therefore further refinement is required.

Introduction

A *thermal control system* is an essential subsystem for any satellites. Its primary purpose is to maintain all equipment on the satellite within their operating temperatures. Before a thermal control system can be designed an estimate of the temperatures to be encountered by the satellites must be determined. It is the purpose of the thermal analysis to provide these temperature estimates. This report carries out the thermal analysis for the MUSTANG satellites.

Aim

There were two aims of the thermal analysis. These were

1. To develop a model which will simulate the thermal conditions of the MUSTANG satellites while in orbit
2. To incorporate a thermal control strategy into the model to verify that the temperatures of all subsystems are within their prescribed limits

Thermal Model

The thermal model was created using the software package IDEAS-TMG. This software used a finite difference method to solve the temperature distribution in the model. The model created was a simplified geometric representation of the actual MUSTANG satellite. Included in the model was the internal structure. Furthermore, IDEAS-TMG allowed the temperature of individual subsystems to be measured. An illustration of the model can be found in Figure 1.

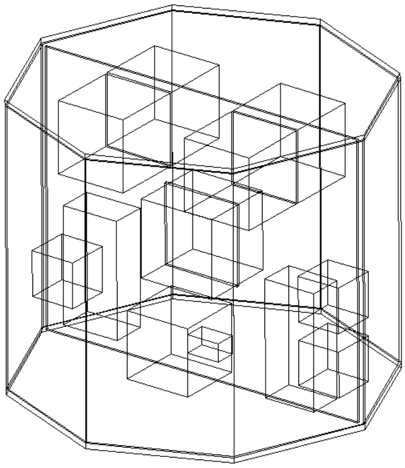

Figure 1 – Schematic of the model of the MUSTANG satellite generated using IDEAS-TMG

Of interest were the transient solutions. The transient solutions determine the resulting temperatures onboard the spacecraft as it orbits the Earth. The orbit which was considered was the Noon-Midnight orbit (where the satellite passes directly between the Sun and the Earth). Two cases were tested. These were the worst hot case and the worst cold case. The relevant parameters for each case may be found in Table 1. If the thermal control system could handle these two cases, then it could be concluded that the system would be able to handle any operating conditions the satellite is likely to experience in orbit.

Table 1 - Design parameters for the worst operating cases for the satellite in orbit

Parameter	Hot operating conditions	Cold Operating Conditions
Solar Radiation	1418W/m^2	1316W/m^2
Albedo Factor	0.35	0.25
IR Earth	237W/m^2	216W/m^2
Power Dissipation	30W	16W

Thermal Control Strategy

In order to satisfy the temperature criteria a thermal control strategy was devised. The most appropriate strategy relied on isolating the inside of the spacecraft from the outside. Conduction paths were then used to control the heat transfer. The required modifications to the satellite are as follows

- Heat transfer by radiation is minimised by insulating all the internal surfaces.
- The conductivity between the subsystem boxes and the internal panel is enhanced allowing heat to flow out of and into the subsystem boxes.

- The internal panel is used to distribute heat evenly over the subsystem boxes ensuring a stable temperature inside the satellite. To accommodate this task the internal panel chosen was to be made from aluminium
- The conductivity between the internal panel and the outer case is carefully chosen to allow heat to leak out of the satellite at an appropriate rate to ensure that the spacecraft does not overheat

Evaluation of Thermal Control Strategy

In this section the results of the thermal model are presented. Table 2 compares the temperature variations of the equipment with and without the thermal control system during the worst hot case. The shaded boxes indicate where the temperature limits have been exceeded. Without the thermal control system it is seen that most of the temperature limits are not being met. When the thermal control system is incorporated into the thermal model most of the temperature limits are satisfied with only two violations. This is seen as a substantial improvement.

Table 2 – Comparison of the temperatures of the equipment with and without a thermal control system. These were predicted by the thermal model during the worst hot case. The shaded boxes indicate where the temperature limits have been exceeded. (All temperatures in degrees Celsius)

Component	**Temperature Limits**		**Without Thermal Control**		**With Thermal Control**	
	Lower	**Upper**	**Lower**	**Upper**	**Lower**	**Upper**
Payloads	0	40	**-19.5**	**54.6**	0	**47.4**
Fuel Tank	0	55	**-23.2**	40.5	0	47.3
CPU	-10	55	**-28.8**	42.4	0	44.0
Battery	0	40	**-8.4**	**40.1**	0	**49.6**
Power Unit	0	75	**0**	**82.4**	0	47.7

Table 3 – Comparison of the temperatures of the equipment with and without a thermal control system. These were predicted by the thermal model during the worst cold case. The shaded boxes indicate where the temperature limits have been exceeded. (All temperatures are in degrees Celsius)

Component	**Temperature Limits**		**Without Thermal Control**		**With Thermal Control**	
	Lower	**Upper**	**Lower**	**Upper**	**Lower**	**Upper**
Payloads	0	40	**-33.9**	37.6	**-0.3**	15.1
Fuel Tank	0	55	**-34.1**	25.3	0	15.3
CPU	-10	55	**-37.0**	29.4	-0.8	14.0
Battery	0	40	**-24.7**	20.2	0	15.5
Power Unit	0	75	0	70.8	0	16.6

In Table 3 the temperature variations for the equipment with and without the thermal control system are compared for the worst cold case. It is revealed that under these cold conditions, without any thermal control system most of the lower temperature limits are not satisfied. Again when the thermal control system is applied there is a large improvement with only one violation of the temperature criteria. It must be pointed out that this violation is by less then a degree. From this one can infer that the thermal control system can adequately handle the worst cold case.

From the results in the worst hot case and the worst cold case, the thermal control system can be seen as being successful at improving the temperatures inside the satellite.

One of the important features of the thermal control system was isolating the inside of the spacecraft from the outside. This enabled the satellite to retain heat inside during the part of the orbit where the satellite was in the Earth’s shadow. This was evident by the lower temperatures for both the hot case and the cold case being significantly higher.

The objective of the thermal design must now be to modify the thermal control strategy so that all temperature limits can be satisfied. Suggestions for improving the thermal control strategy are outlined below.

- Altering the internal layout. By moving sensitive equipment to more thermal stable locations on the satellite would improve the temperatures experienced by this equipment.
- Relaxing the upper temperature limit of the payloads to 50^{o}C. However this would eliminate many of the candidate payloads that are being considered for MUSTANG and would only be suggested if all other avenues are exhausted.
- Increase the thickness of the payload boxes. This would increase the heat capacity of the box meaning that the boxes themselves will be able to absorb more heat before the temperature would increase. This will have the effect of lowering the upper temperature.

Conclusions

The thermal model showed that without a thermal control system many of the temperature limits of the equipment on the satellite would not be satisfied. The thermal model was used to assess the performance of potential thermal control strategies. The most appropriate strategy consisted of insulating all internal surfaces in order to reduce heat transfer by radiation. Conduction is then promoted by improving the conductivity between the contact surfaces inside the spacecraft. Finally, the conductivity of the joint between the internal panel and the outer case is carefully selected to ensure heat can leak out of the system so that the inside of the spacecraft does not overheat. This strategy was incorporated into the thermal model and was shown to substantially improve the temperatures of the equipment. Despite this improvement there were still cases of where the temperature limits of various equipment were not being satisfied. It is therefore required that this chosen scheme be modified so that all temperature criteria can be met.

MUSTANG: Structural Analysis

Carl Fivey
Space Research Centre, School of Engineering, Cranfield University

Abstract

As a member of the Astronautics and Space Engineering course for the academic year 2001/2002 the requirement was to take active involvement in the Group Design Project MUSTANG. This project was a continuation of the preliminary study already carried out by the previous year's students. The document, which this summary supports, presents the steps taken in the structural analysis of the MUSTANG. The structural analysis involved both static and dynamic principles whereby a model with the appropriate boundary conditions and load cases can be meshed and submitted to the finite element solver MSC/NASTRAN. The considerations in the structural design included; mass, payload accommodation and power raising capabilities.

Introduction

The summary presented here attempts to explain in a concise yet informative manner the procedure by which the structural analysis was performed. At present the structure of the MUSTANG possesses an octagonal cross section, for reasons such as average power raising capabilities (Toomey, 2001), the material of high strength Carbon was selected since it provided the group with the best combination of low mass, high strength characteristics. The main drivers in the design were proved to be associated with power raising, since enough energy is needed to operate the payloads. The main concern was the dimensions of the solar cells; it was therefore necessary to increase the flat side of the MUSTANG since a fillet does not lend itself to solar cell mounting. This in turn had implications in terms of mass, although during the period of the project the mass has been gradually reduced and presently stands below the 10kg threshold for nano-satellite definition. The utilization of fillets was a constraint imposed on the structural design in the light of the capabilities of the filament winding manufacturing process kindly being offered by Astrium UK for the project. This process also automatically selects the material, limiting the choice to Carbon Fibre Reinforced Plastics (CFRP). The analysis starts with the considerations of the fundamental structural principles. This involved the use of simple static loading equations and the equations of motion for a simple system for modal analysis. An assessment was made on how to use previous work and how to progress with this material, new configurations were developed and the analysis of one such configuration can be found in the main body of the full report.

Structural Principles

The basis of structural analysis stems from the fundamental equations involved in engineering. The following equations can be related to a vast array of engineering problems and are in the main displayed in Wertz and Larson (1992).

The static analysis involves the stresses and strain imposed on the body when a load is exerted on it. Stress is given in terms of load per unit area, and the strain is a ratio of the elongation of material to its original length. Other properties of importance are Young's modulus and Poisson's Ratio. These and the others can be determined using the equations below.

Stress
$$\sigma \equiv \frac{Load}{Area} \equiv \frac{P}{A} \quad \textbf{(1)}$$

Strain
$$\varepsilon \equiv \frac{\Delta L}{L} \quad \textbf{(2)}$$

Poisson's Ration
$$\upsilon \equiv \frac{\varepsilon_{lateral}}{\varepsilon_{axial}} \quad \textbf{(3)}$$

Young's Modulus
$$E \equiv \frac{\sigma}{\varepsilon} \quad \textbf{(4)}$$

The dynamic analysis involves the manipulation of the equations of motion when applied to a particular body. There are defined in terms of displacement, velocity and acceleration. The natural frequency is of utmost importance and can be found using the natural circular frequency $\sqrt{k/\ m}$ and the period T, given below;

$$T = \frac{2\pi}{\omega_n} = \frac{2\pi}{\sqrt{k/m}} \quad \textbf{(5)}$$

To give;

$$f_n = \frac{1}{T} = \frac{1}{2\pi}\sqrt{\frac{k}{m}} = \frac{\omega_n}{2\pi} \quad \textbf{(6)}$$

Spacecraft Configuration

The configuration has been improved on from last year (Toomey and Sarchiapone, 2001) to incorporate a vertical shelf construction, this provides a medium on which to centrally locate the fuel tank and minimise any pitching moments caused by movement of the centre of mass of the satellite.
Figure 1 shows the panel and figure 2 shows the panel inside the MUSTANG satellite.

Figure 1 - Vertical shelf with location for the fuel tank

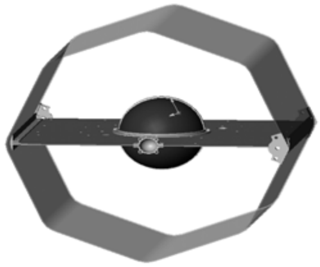

Figure 2 - The MUSTANG shell with showing the vertical partition and central fuel tank configuration

The external configuration has only changed in order to accommodate new candidate payloads. One slight change which became one of the main drivers in design was the requirement of the filament winding manufacturing process to have a minimum radius of 30mm, this in effect change the dimension of the satellite and lead to an increase in mass. The masses of the structural components are given below.
Mass of the core structure:

- Tube: 1275 g
- Vertical panel: 528 g
- Bulkheads (x2): 300 g

Fasteners**:**

- Rails (x2): 200 g
- Bolts (x40): 160 g

FE Model

The model was initially developed by drawing a 2D octagon and meshing with bar elements, upon extrusion of the octagon to the required length, these bar elements became quads. The use of shell elements created a much simpler solution. The mid-plane elements can be seen below (Scarchiapone, 2001);

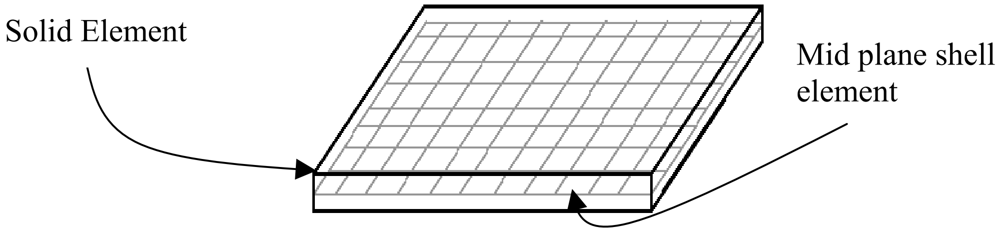

Figure 3 - Shell element displayed as a mid plane element

Launch Configuration

The spacecraft must be able to withstand the loads outlined by the launch vehicle manufactures, in this case Arainespace. The first mode frequencies of 90 Hz longitudinal and 45Hz lateral must be exceeded in order for the spacecraft to withstand the vibrations experienced at launch. The satellites are to be stacked in the manner outlined in Toomey (2001) in a manner so the payloads are 'facing' each other from each spacecraft.

Conclusion

It is obvious that several more iterations are needed to complete the design process of the MUSTANG satellites. Particular emphasis must be made to the modelling of the payloads to create a more accurate solution. The decision on the final design can only be made after a full review and an in depth trade off study

Further Work

Particular emphasis must be made to the modelling of the payloads to create a more accurate solution. The ply lay up of the laminate is another factor that must be addressed. The modelling of the launcher adapter ring and the satellite interface must also be improved. Verification tests on the satellite would eliminate any ambiguity when the spacecraft integration process begins (Sarafin, 1995)

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MUSTANG: Thermal Design

Matthieu Gasquet
Space Research Centre, School of Engineering, Cranfield University

Abstract

The main objective of this year's Group Design Project was to freeze the external and internal configuration of the MUSTANG satellite. Based on the preliminary design developed last year, a large part of the spacecraft design is now almost finished, particularly the internal design. From a thermal point of view, a large part of the hardware has been chosen within the mass and power budget limits and a thermal strategy has been developed.

Introduction

This paper summarizes the report which covers the thermal design of the satellite from the user requirements to the final selection process of the thermal control hardware. After an overview of the different available technologies used to control the temperature, the thermal strategy applied to MUSTANG and its results are presented.

User requirements

The main user requirements concerning the thermal design of MUSTANG are about mass, power and temperature. The mass budget is limited to 400g and the power allocated to the thermal control system should not exceed 1 W. Moreover, all components have to be kept within allowable limits (as specified in table 1) during all mission phases.

Table 1 - Temperature requirements of each systems and sub-systems of MUSTANG.

Equipment	Operating Temperatures (^{0}C)
Microprocessor	-10 ~ 55
Gyroscope	0 ~ 60
Magnetometer	-55 ~ 85
Reaction Wheels	-5 ~ 45
Magnetorquers	-55 ~ 85
Li-ion Battery	0 ~ 40
Payloads	0 ~ 40
Laser Range Finder	-10 ~ 40
GPS Receiver	-10 ~ 70
Microcontroller	-10 ~ 55
Amplifier	-10 ~ 55

Equipment	Operating Temperatures (^{0}C)
Communication	-10 ~ 50
Fuel Tank	0 ~ 55
Data Handling	-10 ~ 55
Solar Cells	-100 ~100

Available technological solutions

There are two different types of thermal control systems (TCS) : passive and active. A passive system relies on conductive and radiative heat paths (no convection in space) and has no moving parts and no electrical power inputs. An active system is used in addition to the passive system when passive system is not adequate. An active system relies on pumps, thermostats, and heaters, use moving parts and need electrical power.

The most common components of a passive TCS are:

- Multi Layer Insulation (MLI) which are made with several layers of insulation foil alternated with low conductance spacer. This type of components is quite light and very efficient to limit any radiative and conductive heat transfer.
- Thermal control coatings, which are generally paint with special optical properties. By fixing the emissivity and absorptivity of such a device it is possible to modify its skin equilibrium temperature and then its radiated heat.
- Conductive tapes and joint which increase considerably the contact conductance between two surfaces.

An active TCS is generally made with:

- Heaters, which produce heat by Joule's effect.
- Heat pipes and pumped-loop system, which is used to transfer heat from one location to another. The fluid inside the tube operates by changing phases during heating and cooling phases.
- Louvers, which are mechanical devices that, in effect, regulate the area of a radiator in response to its temperature.
- Thermal switches, which are devices that provide a direct conduction path between the heat source and the equipment mounting plate when the contacts are closed (like an electrical switch).

An active TCS is generally very efficient, but it often needs an external source of power and is relatively heavy and bulky compared to a passive TCS.

Thermal design

Because of the low mass and power budget, MUSTANG has to use as much as possible a passive thermal control system. After several simulations, it appears that the best thermal strategy would be to minimise heat transfer by radiation, increase the conduction between the boxes, and play with the conductivity between the vertical shelf and the casing. By doing that, the heat produced by some boxes can be shared between the others and the excess of heat is then transferred to the casing and expulsed to the space.

To do so, and after having checked the results with another simulation, the following thermal hardware will be applied to MUSTANG:

- Conductive tapes will be applied between solar cells and the body to ensure a homogeneous temperature on the outer side of the satellite.
- An insulation foil will be added on the inner face of the body to limit heat transfer between external environment and internal.
- Multi Layer Insulation will be stuck on each box of the spacecraft to limit any radiated heat transfer between the boxes and the rest of the spacecraft.
- The vertical shelf will be made in aluminium to ensure a good heat path between boxes and avoid hot or cold point.
- High conductive contact joints will be added between boxes and vertical shelf to increase the heat path between boxes and the shelf.
- Corners will be painted in white to limit sun absorptivity and to radiate heat from the internal.
- Heat sensors will be placed on each box to measure temperature and switch off the system in case of temperature control failure.
- There will be a thermal switch between the vertical shelf and the case to control the conductivity between the two devices.

Moreover, some box locations have to be chosen carefully. For instance, the batteries box, which is very sensitive to temperature gradients, has to be on the Earth facing face because it is the most stable from a thermal point of view.

Table 2 shows the simulation results of such a system and almost all the temperature are in the allowable range. More over a first evaluation of the mass and power budget shows that this Thermal Control System can be fitted into MUSTANG (mass budget: 400g, power budget: only few mW for the sensors).

Table 2 – Evaluation of the thermal control stategy applied to MUSTANG.

EQUIPMENT	TEMPERATURE LIMITS	RESULTS NO THERMAL CONTROL	RESULTS THERMAL CONTROL
Payloads	0 ~ 40	-28.7 ~ 48.4	0 ~ 47.4
CPU	-10 ~ 55	-37.0 ~ 42.4	0 ~ 44.0
Battery	0 ~ 40	-8.4 ~ 40.1	0 ~ 49.6
Fuel Tank	0 ~ 55	-23.2 ~ 40.5	0 ~ 47.3
Reaction Wheels	-5 ~ 45	-18.6 ~ 38.3	0 ~ 46.8
Ext. Surface	-100 ~ 100	-23.5 ~ 46.4	-64.5 ~ 68.4

Conclusions

During this year Group Design Project, the detailed transient thermal analysis which has been done shows that the satellite and its systems need a thermal control strategy to stay in the allowable temperature range. After several simulations, a first hardware selection has been made and its suppliers are identified. There is still some work to do. It could be useful to refine the mass budget and to check that the thermal strategy is compatible with the launch conditions.

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MUSTANG: Preliminary Design of a Cubic Satellite

Ardani Medina
Space Research Centre, School of Engineering, Cranfield University

Abstract

Efficiency is the word used on space; in particular satellites must be designed to provide an efficient task due to the cost of sending an object to orbit. Different shapes of satellite were studied, analysing the possible contributions and penalties that this shapes can have on the MUSTANG project (Multi-University Space Technology Advanced Nano-satellite Group). The aim of this report is to demonstrate the feasibility, and comparison of the octagonal shape against the cubic shape for this particular MUSTANG mission. This has been done by analysing the best cross-sectional area, mass performance, how suitable was each face for its payload and power obtained. By using CAD design tools, the tasks were simplified to a minimum. Also with the help of simulation tools, it has been able to prove different aspects of the structure. It was found that a cubic shape satellite fits perfectly with the aim of the mission, being able to provide more power, more space available, and a longer lifetime compared with the octagonal shape. These pay-offs from the cubic shape satellite can be combined, using all the advantages for a common solution. The major problem or inconvenient that MUSTANG satellite has is its performance at the worst-case scenario where power is not enough to run the satellite to a sufficient level in the eclipse time. This problem could be solved by using this new cubic shape satellite.

Introduction

One aspect on the design of a satellite is the fact of which shape should it has. The external shape of it, and the internal cross-sectional area can determine, lifetime of the mission, power achieved without deployable arrays, weight savings etc…

These particular areas become even more critical when the satellite in question is a Nano-satellite. The mass limitations are very strict and because this kind of satellites are very small, the power is also limited without deployable arrays, so a it is necessary to use its external and internal shape in a way that it is possible to obtain the maximum efficiency. This report will produce a preliminary comparison about the two main shapes used on the MUSTANG project, the cubic shape and the octagonal shape. Paying attention to the thermal, structural and functional areas of it.

Objectives, Work done

- *Weight/Analysis performance* (area effective): This area was the most time consuming because changes were very often through the project, and also because no mistakes were allow due to the importance of this topic. All the work of this report was done in basis of this area. This task was achieved, by finding a improved solution.

- *Structure analysis and proposed structure design*: This area was perform as preliminary, no detail work was provided, however the task increased when the octagonal shape was also analysed to be able to produce any comparison between the two shapes. External configuration was provided, deflections but not resonant frequencies.
- *Thermal analysis*: This task was performed by the thermal specialists due the deep knowledge necessary to generate this information. However this task was achieved as a team, and not by the author himself.
- *Mechanisms related to the cubic shape*: No mechanisms were studied, however the cubic shape had extra or different components which had to be analysed and designed. The time assigned for this task was used on this the study of extra components. Also the central panel for both satellites were analysed.
- *Proposed space ability*: Internal configuration was done, studying the possible extra space available for the payloads.
- *Preliminary power budget*: Analysis of the power obtained in different cases was obtained. Also different structures were analysed, however with negative results. This task was achieved for the final cubic shape and compared to the octagonal, obtaining positive results

Results

The results are very positive, much more than expected. The main points obtained from this research can be summarized as follows:

- Weight saving: Taking the cubic satellite, as a final shape of the satellite will provide with a weight savings of 3% of the overall mass budget, which represents around 300 grams.
- Stiffer structure: The cubic structure, under preliminary analysis shows a stiffer structure which can give some advantages at the launch stage, providing to the components more safety conditions. Also resonant frequencies have not been calculated, however the preliminary deflection figures give a better response than the octagonal one.
- External configuration: All the proposed external subsystems and payload can be fitted into the satellite without inconvenient and major changes.
- Internal configuration: Thanks to the cross-sectional area provided by the square shape, 30% more internal payload can be assembled in to the satellite.
- Power: The cubic satellite provides more power on the worst-case scenario, by 7%, which is the time where more power is needed to be able to run the systems normally.
- More efficient dynamics: Because the thrusters are situated at the corners on the cubic shape, the distance between the centre of gravity and the corners is bigger than in the octagonal case where the thrusters are on the sides. Therefore an increment of 15% is achieved on rotational thrusters moment.

- Thermal environment: the octagonal shape seems to be more situated for this area, however the differences are not significant and not generates any restrictions.

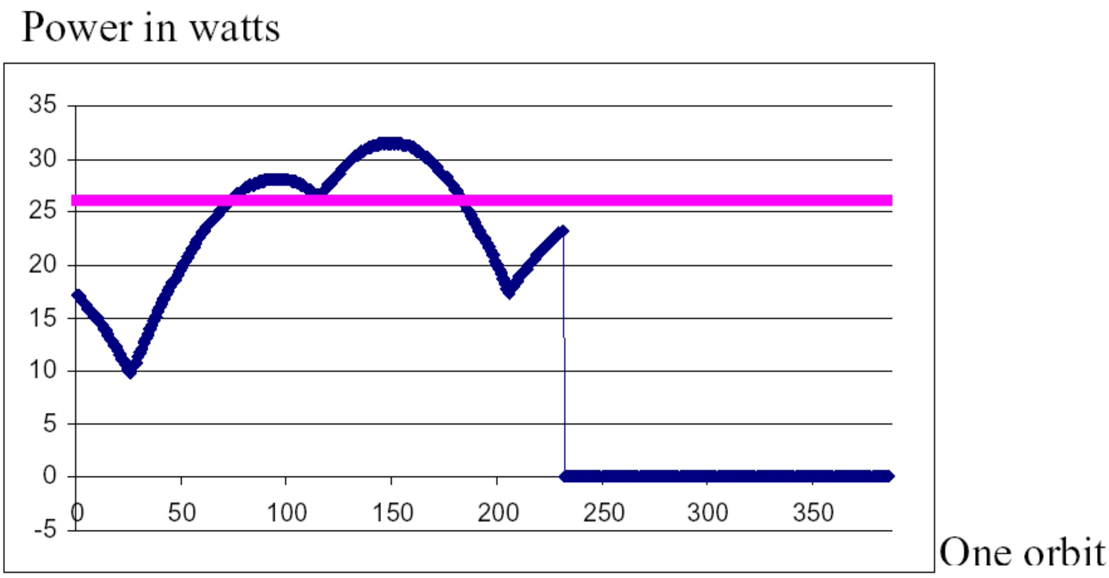

Figure 1 - Power data of two different cases, the best-case scenario is shown on pink, and the worst-case scenario in blue.

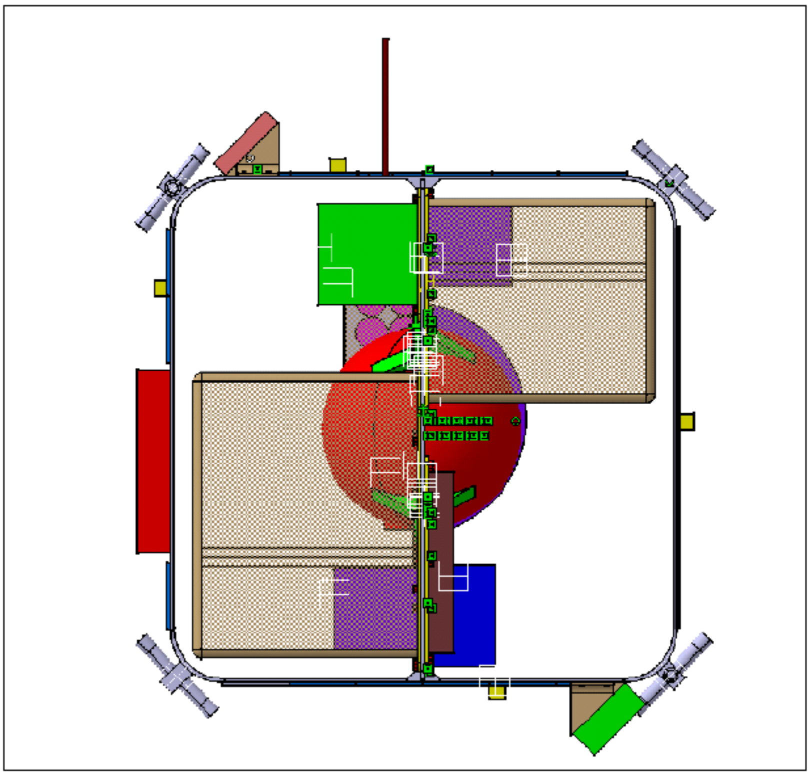

Figure2 - Assembly the satellite showing clearance between the external configuration and the internal configuration.

Conclusion

The conclusion from this research is that the savings in weight provided by the lighter cubic structure can be used to place more batteries on the extra internal payload space, which then will absorb the extra power obtained from the solar cells in the worst case scenario. Doing this, the satellite will be able to provide an improved performance of the systems on the worst-case scenario, which when the eclipse time can reach up to 40% of the whole orbit.

Also providing that the moment generated by the thrusters is improved, the lifetime of the satellite can be extended for some time.

The cubic shape satellite has been designed keeping in mind the fact that if at anytime the team leaders decide to change to this shape, minimum time will be necessary to produce this change.

At this stage of the project it is very risky to change to anew shape, and may be do not meet the deadlines, and therefore take this project to an early end. The previous team did this comparison, last year, but it was so superficial and so mediocre that the team was leaded to the wrong path. This is usually a typical problem which many project faces. Not enough time spend at the beginning can result on a failure of the project at a late stage. This research can be a guideline for the future satellites, with similar tasks or size.

MUSTANG: Structural Design

Yann Michel
Space Research Centre, School of Engineering, Cranfield University

Abstract

The main objective of 2001/2002 Group Design Project realised by the ASE MSc students was to carry on the 2000/2001 study of the MUSTANG nanosatellite and to define a detailed configuration of this satellite. This aim is especially important for the structural design of the spacecraft because it needs to be frozen from its structural concept to its internal configuration There is no perfect single structural configuration but the use of a filament wound tubular structure allows the design of an adaptable concept at low cost. Starting from a preliminary analysis of various possible concepts the mechanical group has selected one configuration that meet many market and mechanical requirements. Another challenge was to collect all the data on internal components of the satellite and to integrate it into the design process.

Introduction

This paper summarises the report which covers the structural design of the satellite from the concept selection to the final structural and internal configuration. The results and the main arguments explaining the decisions are presented in this document. This summary show the process we used in the design, the concept selection and the main requirements need for the structure are presented, the second step is the design of the internal configuration and assembly and integration processes are also discussed.

Concept selection

Many concepts were studied this year from an inner tube structure to the classical Tube with two internal bulkheads. All these concepts were evaluated in accordance with their different advantages and a relative coefficient showing the weight of each factor in the total project. The characteristics evaluated in this part were:

- The mass of the main structure (less than 2.5 kg) is the main constraint we have to face for the design of the satellite.
- The stiffness and strength of the structure was taken into account because the spacecraft shall be able to support the loads avoiding any failure especially during the launch which is the critical phase of the satellite life.
- The integration of the payloads and other subsystems as well as the access to them was also evaluated because of their necessity during the pre-launch phase.
- The integration of the fuel tank(s) will be one of the most difficult problems we will have to face in the design because of its shape and its location constraints.

- The concept chosen shall allow the thermal control system to maintain all the items of a spacecraft within their allowed temperature limits during the mission.
- Some other factors have influenced our choice. The manufacturing capability and mounting aspect are also important and we tried to take into account the work done before.

This selection method led us to choose the vertical partition structure which seems to be the best one. This concept was also proposed and studied by Adrian Russel (Astrium Ltd) and has already been used on a nanosatellite (Dawgstar project, USA).

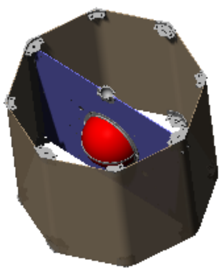

Figure 1 – *2001/2002 Chosen concept: Apex-to-apex vertical partition inside the octagon.*

The main advantages of this concept are its low mass compared to other solutions with two or three structural bulkheads. In this case, the bulkheads will not play an important mechanical role (just for external components which are light). The mechanical and thermal performances of this structure are also very good in the longitudinal axis and in term of conductivity of heat in the satellite. Moreover the use of the vertical panel is in complete accordance with the constraint of the filament wound structure. Finally this configuration allows us to integrate the release mechanisms designed by Elie Allouis with a central ejector and allows for easy access for integration and equipment replacement.

Internal configuration of the satellite

The design of the internal configuration was the main task of the structural design this year. We had indeed to freeze the size and other characteristics of the main structure elements, inventory all the sub-systems to be mounted on the satellite and allocate slots for all these items respecting particular constraints. As we want the satellite to be as reliable as possible especially to accommodate various payloads we decided to allocate all the top of the satellite to the Payload bay (four slots) and the bottom to the Systems Bay. With this configuration the MUSTANG platform can be used for various missions and payloads. The constraints for the systems are presented below:

The Shunt Regulator and power Management units are gathered together in a common box. As it radiates all the non-used power of the satellite the box must quite far from other sensitive to heat boxes (Batteries, Fuel tank…) but it must also be in the middle of the spacecraft to provide power to all the subsystems.

The batteries and its electronics unit (grey on the drawings) are very sensitive to the temperature so it has been decided to put them on the most thermally stable slot which is the closest to the earth pointing face.

The Reaction Wheels box (in dark blue on the drawings) must be on the main axis of the satellite.

The CPU and Data Handling unit (in yellow on the drawings) needs to have an access to all the items of the satellite so it must be located at the interface between payloads and systems.

The Gyroscope and Magnetotorquer (grey boxes on the drawings) have been gathered together because of their similar field and to avoid cutting to many holes in the vertical shelf.

The **fuel regulator** (in red on the drawings) must be near the fuel tank. It must also provide fuel to the thrusters using four different pipes. As these thrusters are located in the middle of the panels the unit will be located in the middle of the shelf (in height).

The **Communications** unit (in purple on the drawings) has been added to the internal configuration near the end of the GDP when all the slots were already allocated the other subsystems. This element hasn't special requirements.

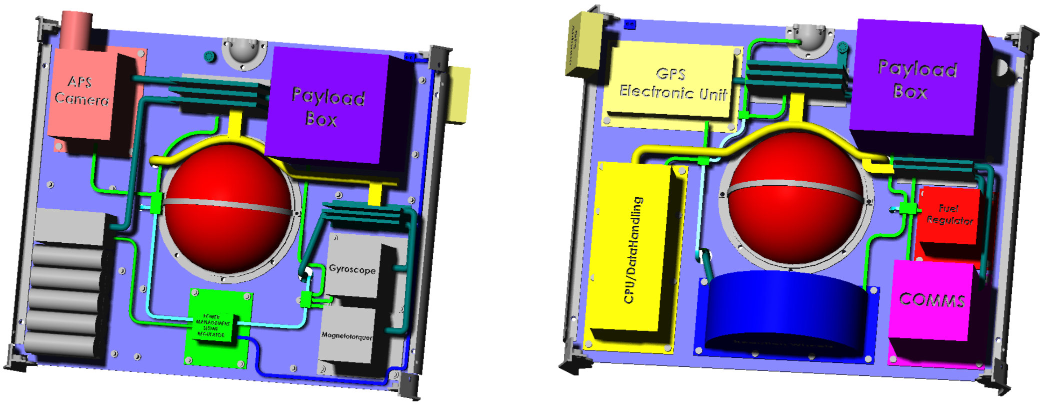

Figure 2 – *Final internal configuration of the spacecraft (both sides of the partition). The two wires network (Data and Power networks) were also taken into account in our CAD model.*

Attachment method and Assembly process

To allow a simple attachment process we also decided to add an aluminium plate under every box. Another issue needs to be resolved; by bolting through the panel on both sides of it we may encounter problems with conflicting boxes footprints. With this box concept the size of the plate (located under every box) can be extended thus we can use common bolts for two items located on the same slot but on opposite sides of the partition (cf. fig 3).

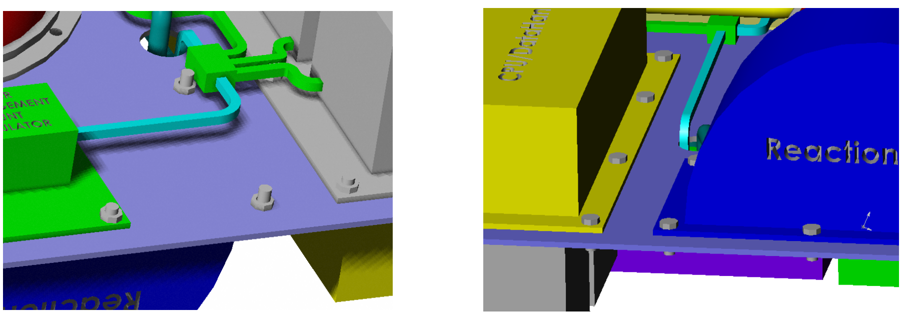

Figure 3 – *Detailed view showing the method used to solve the problem of conflicting footprints on both sides of the partition.*

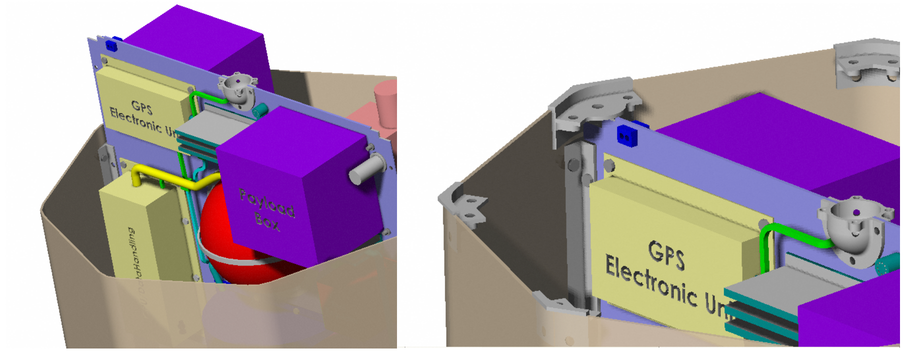

Figure 4 – *Assembly of the vertical partition into the tube and the mounting of the brackets on the tube once the shelf is assembled.*

The most difficult task in the design of fasteners between the vertical panel and the tube was to find the concept. With two simple rails we just have to slip the vertical panel into the rail and to insert bolts through the rail and the panel. This solution is simple one and an aluminium rail provides also a better conductivity of heat between the panel and the tube and finally it has excellent mechanical performances in the lateral axis. The fixation method of the rails to the tube has not been defined yet but we can bolt them through the tube or glue them on the tube using adhesives epoxies.

Conclusions

In conclusion, during the *Group Design Project* the selected concept and the internal design of MUSTANG has proved its reliability in term of mechanical performances and it respects the mass budget assigned to the mechanical group. Its simplicity has also been demonstrated during the assembly process study and the internal configuration with the vertical partition allows for easy access for integration and equipment replacement which is particularly important during the pre-launch phase. Even if the internal configuration seems to be frozen there is still a lot of work to achieve in the structural design of the satellite especially in term of integration of the external configuration in term of compatibility with the internal needs a further analysis. A complete finite element analysis of the structure and its fasteners also needs to be done.

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MUSTANG: Deployable arrays overview

Frédéric Veilleraud
Space Research Centre, School of Engineering, Cranfield University

Abstract

The Mustang project satellite is currently using body mounted cells as only source of power. However an increase either in the need in power or in the need of free surface on the body may lead to the use of deployable solar arrays. The purpose of this study is an overview of that possibility. Therefore a comparison of five various shapes for the arrays, selected for their simplicity, is done; calculating the average cross-section intercepting the solar flow on the orbit, and taking into account the precession of the nodes. The highest powers are given by the "octagonal array", when it is used as extra to the current system, and by the "length array" when used as only power source. Their efficiency relatively to their mass is also evaluated and the "length" possibility gives indisputably the best optimisation. The possibility of using adjustable arrays is briefly tackled and shows that except for very large increases of power, there is no point to go in that direction. Finally, a few concepts for simple deploying and positioning mechanisms are proposed.

Introduction

The use of deployable arrays on a nanosatellite might seem to be a bit ambitious. But as the Mustang satellite is a non spinning spacecraft, there is basically not much difference with a "normal" satellite. The challenges in it are mostly the respects of simplicity and lightness particular to the project. This, added to the fact that the arrays have to fit the current configuration in order to minimise modifications if they become necessary further in the project, dictates the general shapes of the array. The overview consists then in a comparison of the possibilities including simple adjustable panels and in a basic proposal of simple mechanisms for deployment and positioning.

Arrays description

There are five principal fixed shapes and one adjustable. Before deployment, they all keep the basic octagonal aspect of the body. There are shown deployed in figure 1. To simplify the study, we will use a descriptive name for each of them. The "flower" type of array is composed of 4 rectangular opposite faces erected perpendicularly to the velocity vector. The "trailing" type is an octagonal panel also perpendicular to the motion. The "length" array is made of 5 rectangular panels placed in the length of the body on the trailing side. The "octagonal" type is composed of 2 half of the octagonal tube of the body. They deploy letting their concave side facing the motion. The "wing" arrays are 2 rectangular faces on the sides of the satellite and in its horizontal plane. Adjustable arrays are made of 4 octagonal faces aligned on the vertical above the satellite. A second panel is placed symmetrically. All the rectangular faces coincide with the small sides of the octagonal tube and lean again them before deployment. The octagonal panel coincide to the bulkheads faces.

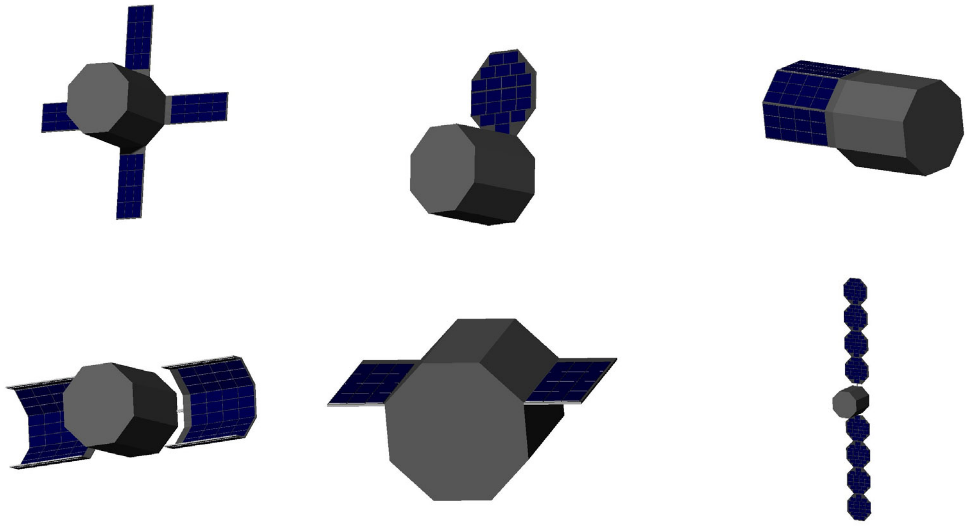

Figure 1 – The definition of the various shapes studied for the deployable arrays on Mustang project. On the first line, from left to right, the "flower", the "trailing" and the "length" types. On the second line, the "octagonal", the "wing" and the adjustable types. Before deployment they all respect the basic octagonal shape of the body.

Concerning the material, there are a few possibilities but in a first approach we are considering honeycomb aluminium panels of mass 4.73 kg/m² (including solar cells).

Comparison of the various shapes

One of the aims of the study is to issue the best candidate array in case of an eventual modification of Mustang satellite. For that we need to compare the various shapes selected in the first section. The method consists in evaluating the average cross-section intercepting the solar flow which is proportional to the average power provided by the system. The average is calculated on one orbit and for the precession of nodes around the Earth. The comparison in term of the gross increase of power relatively to the current one or in term of increase per kilogram of array is then possible.

In a first step we are studying the various arrays as a way to increase the power by adding them to the current system. The "length" and the "octagonal" shapes give the highest values with respectively 43% and 51%. The second type has a creditable increase. We can then normalise that increase relatively to the mass in order to get rid of the difference of size for the various shapes. The "length" type is undoubtedly the most optimised with 45% of increase per kilogram of array. All that is shown in figure 2. Therefore the best candidate depends on what we are looking for. If the aim is the highest increase, the "octagonal" shape is fine, but if we are looking for a compromise, the "length" array is better.

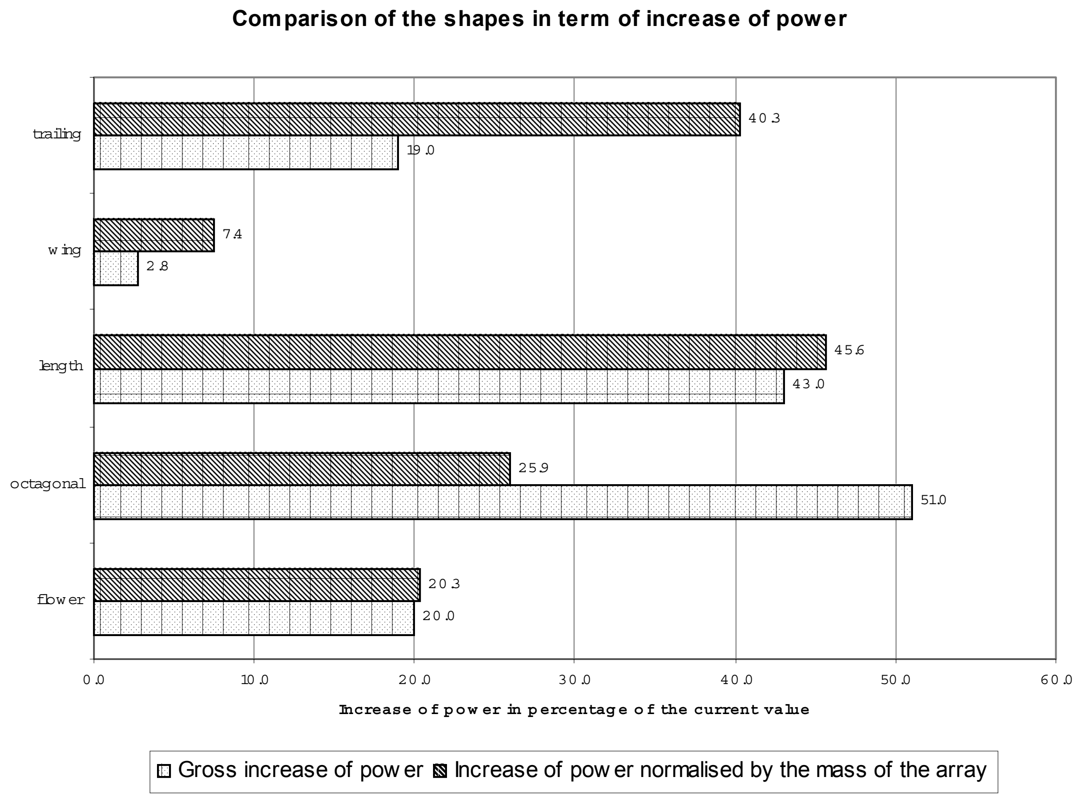

Figure 2 – Comparison of the increase of power for various shapes when used as an extra. In striped are the increases relatively to the mass and in plain the gross values. "Length" and "Octagonal" are the best candidates.

The second step consists in looking at the case where the deployable arrays are the only source of power. Actually, we calculate the increase of surface they require, relatively to the basic one defined in the first section, in order to provide the same power than the current one. The best results are for the "octagonal" and the "length" arrays which surfaces have to be less than doubled. But if we keep an eye on the best optimisation relatively to the mass, the "length" type is obviously the best candidate with 62% of the current power provided per kilogram of array.

Briefly looking at adjustable arrays, we can calculate that in order to produce twice the current power, two panels of 4 times the surface of the bulkhead each are necessary if we limit the motion to a rotation around the length axis of the array. This can be optimised using only one panel of 6 times the surface of the bulkhead above the satellite. Nevertheless, the production of power per kilogram of array is not much higher than the one for the "length" candidate; therefore this solution is only suitable when the other is not possible.

Mechanisms proposal

Three concepts of mechanisms are proposed. The first one is a simple deployment system for the array. This one is linked to the body by a hinge. A spring is pre-strained

in the close position and the system is maintained by two pin-pullers. When the pins are retracted, it frees the panel which is deployed by the spring. A viscous damper may be included to the hinge.
The second proposal is a deployment mechanism with the possibility to retract the arrays to their primary position. It is based on the simple mechanism above. To this one, we add a shape memory alloy cable (probably a Nickel Titanium alloy) which is attached to the body on one side and to the array on the other one. It is free everywhere else but has to run around the hinge as shown on figure 3. When heated, it shortens, pulling the array in the close position.

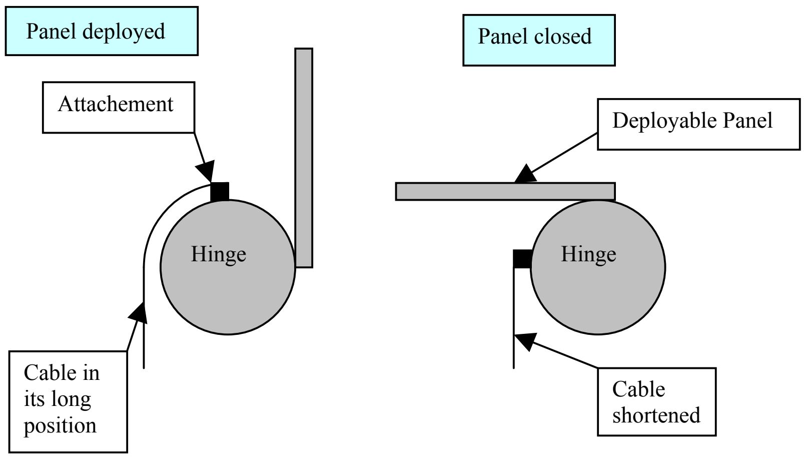

Figure 3 – Reversible mechanism to move the deployable arrays using a shape memory alloy cable. Heated, this one shortens, pulling the hinge in the close position.

Finally, the idea for the positioning mechanism is to use a very small step by step motor, transmitting the motion by contact to an axis which aim is to carry the load.

Conclusions

As we can see, it is not yet possible to determine the array that has to be use for Mustang satellite if necessary. Nevertheless, the study has shown that the "octagonal" shape is the best candidate in term of increase of power when added to the current system. The "length" type is undoubtedly the best compromise between power increase and mass, either as an extra to the current satellite or as only source of power. Adjustable arrays should not be considered except if there is no other solution. Anyway, the shadowing problem will have to be seen as soon as the payload will be known and a detailed study will be necessary on the array when it will be selected as well as a real analysis on mechanisms that where here just mentioned.

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