FFDEM: DEMONSTRATING FORMATION FLYING WITH SMALL SPACECRAFT

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Abstract: Several missions are currently proposed that require very close formation flying. Missions such as IRSI-Darwin and Xeus will require the development of new techniques and equipment to achieve the stringent performances required. Both these missions will be preceded by demonstration missions, aimed at validating concepts of formation flying and generating a European capability in this field. These precursor mis sions will very much be standard missions, using a classical "large project" approach. This has lead ESA to consider the procurement of an innovative mission, which would demonstrate some aspects of Formation Flying, using microsatellite class spacecraft, with a mass around 120kg, based on the Advanced Microsatellite Platform (AMP). The FFDEM mission uses two microsatellites in LEO, to perform advanced formation flying, down to distances of 50m, using a cutting edge set of sensors, actuators and control algorithms. These include a GPS pseudolite, LASER ranging units, optical position estimator, FEEPs, etc. The mission proposed could be implemented in the next few years, although some aspects of it will require innovative solutions.

INTRODUCTION

In the past few years several missions have been proposed that will require that several spacecraft fly in a formation, with specific relative positioning between them. This formation flying is necessary for a very large range of missions, since in many cases represents a large augmentation of the capacity of a mission. From multiple aperture optical systems for planetary exploration (for e.g. the IRSI-Darwin mission) to Digital Elevation Mapping (DEM) of the surface of the Earth, there is a very large range of obje ctives that are suited to formation flying missions.

Although Formation Flying (FF) in itself is nothing new, being current on manned missions, its use on fully automatic missions is rare and considered of high risk. Some of the missions proposed, require that the spacecraft fly for long periods at distances of a few tens of meters, with little margin for error. Currently the equipment and control processes required for Formation Flying are still in their infancy and this hinders the development of the missions. For this reason, the European Space Agency (ESA), has commissioned a study of a simple mission, aimed at testing and qualifying equipment, control algorithms and operational concepts for a FF mission.

The Formation Flying Demonstration Mission (FFDEM) was proposed as a low cost approach to test several technologies and control techniques for FF missions. Based on ESA's "Advanced Microsatellite Platform" (AMP), this mission was intended as a low cost and fast mission, flying two spacecraft equipped with the subsystems required for a FF mission. The definition of the mission was performed by GMV S.A. of Spain and Surrey Satellite Technology Limited (SSTL) of the United Kingdom, and includes a definition of the Guidance, Navigation and Control (GNC) algorithms, the platform design and ground segment definition and the overall definition of the mission. The main aim throughout the study was to define a mission that could be built and fly using the current available systems, but accepting that some units need to be custom designed for the mission (as is the case of the FF specific units). One output of the study was the identification of optimisations possible, if a higher risk approach was used (form both the programmatic and technical point of view).

THE MISSION

ESA selected three case studies that the FFDEM mission should demonstrate (their formation flying aspects only). These were: Digital Elevation Mapping (DEM) mission, High Energy X-ray (HEX) mapping mission (a precursor to the XEUS mission) and some aspects of IRSI-Darwin mission. From the on-set, it was recognised that some of these objectives might not be viable given the constraints on the mission.

The several demonstrations proposed for FFDEM suggested the separation of the mission in different experimental phases, with each one having a set of pre-defined objectives. These are defined by the overall mission objectives, i.e., demonstration of formation flying aspects of the DEM mission, HEX mission and limited IRSI-Darwin mission. Each one of these missions has a set of mission requirements, which were used to derive the formation flying objectives presented on Table 1. In terms of the orbit selection, the HEX orbital requirements were taken as the FFDEM ones, that impose the most difficult formation flying conditions. The chosen orbit was a 561km altitude, 90º inclination one.

The mission timeline follows a pattern of increasing complexity and risk, with each phase more complex and risky than the previous one. The assessment of risk was based on the obvious rule that the closer the spacecraft are flying to each other, the higher the risk. A future task shall be to quantify this risk.

Table 1 - Mission phases and their objectives for FFDEM

Mission Phase	Description	Objectives
1	Formation deployment	- Separation from launcher - Commissioning and early operations - Test of FF payloads
2	DEM Phase	- Formation acquisition - Formation maintenance to within 100m relative position - Test different types of control algorithms and techniques
3	HEX phase	- Formation acquisition and maintenance to within 5cm, at 50m relative distance - Maintain formation for 3 months with relative pointing of 1'
4	IRIS-Darwin phase	- Formation acquisition and maintenance to within 10cm, at 100m relative position - Repeat for 300m and 500m, demonstrating manoeuvre capability
5	Extended mission	- None defined at this stage
6	De-orbiting phase	- Placement of spacecraft at distance that minimises risk of collision until de-orbiting occurs

In Figure 1, a graphical representation of the mission timeline is presented, showing the evolution of the Inter-Spacecraft Distance (ISD) throughout the mission (the vertical lines mark the transition from one mission phase to another).

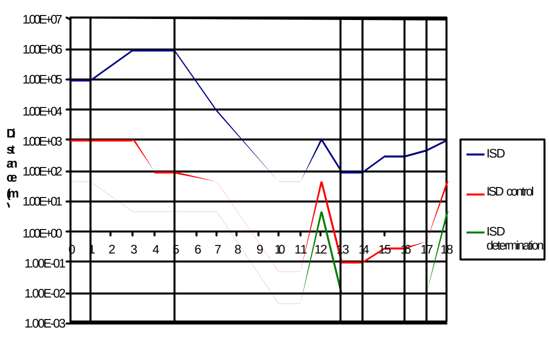

Figure 1 - Intersatellite distances

From the figure it becomes clear that the target duration for the mission is 17+ months, that is the minimum mission duration that allows all the experiments to take place. This duration includes some contingency between the different phases to account for any overrun of the experiments, but this might not be enough if any serious problem is encountered, in which case there is the possibility of extending the mission.

Probably the most critical thing in a mission such as FFDEM is to demonstrate that the mission specific payloads work properly, and also to validate their operation. The strategy to be followed on this mission is to test and validate these subsystems before they become critical to the mission. The timeline includes time to perform such operations, with a similar approach to be used for the control algorithms. The exact processes for test and validation have not yet been defined.

The last phase of the mission consists in the de-orbiting phase, for which the recommendations of ESA[1] were followed. During the study it was estimated that the spacecraft will both decay and re-enter the atmosphere in approximately 18 years after launch[2]. This is below the 25 year limit, above which is recommended that spacecraft have a forced re-entry system.

GUIDANCE, NAVIGATION & CONTROL

In order to carry out the various phases of the mission, a series of different GNC strategies have to be implemented. Highly-coupled GNC algorithms must be on-board the satellites in order to fulfil the mission. A great deal of on-board autonomy is required in GNC by both of the satellites. The two satellites in the formation have a master-slave relationship between the Master and the Flyer.

In examining the GNC requirements, the mission phases described in Table 1 can be subdivided into the following sub-phases, each of which generally requires a somewhat different GNC strategy.

In order to cover the necessities of the various mission phases, a series of attitude and orbital manoeuvres must be included. The orbital manoeuvres include:

- Impulsive Hohmann Rephasing & Transfer
- Impulsive Hopping on V-bar
- Impulsive Transfer between 2 points
- Continuous Tangential Transfer
- Continuous Hopping on V-bar
- Forced Motion, Apply a ΔV & Free Drift
- Collision Avoidance (CAM)

Table 2: Mission Sub-Phases and Objectives for FFDEM

Mission Sub-Phase	Description	Objectives
2.1	DEM Coarse Formation Acquisition	Formation Acquisition at a separation of 1000 km with an error box of 8.5 km.
2.2	DEM Fine Formation Acquisition	Formation Maintenance of 100 m relative position at a distance of 1000 km.
2.3	DEM Experiment	Both satellites acquire a target on Earth at the same time for stereoscopic viewing, and download the images to Earth
2.4	DEM D-control Experiment	Experiment to test D-control algorithm
3.1	HEX Coarse Formation Acquisition	Formation Acquisition and Maintenance from 1000 km to 500 m.
3.2	HEX Fine Formation Acquisition	Formation Acquisition and Maintenance from 500m to 200 m.
3.3	HEX Experiment	Formation acquisition and maintenance at a distance of 50m with an error of 5 cm.
4	IRSI	Fine Formation Acquisition and Maintenance, two rephasings & 3 experiments.

Due to the mission requirements, the following attitude manoeuvres must be implemented as well:

- Fixed & Target Pointing
- Ground Station Pointing for both fixed & moving targets
- Star & Sun Pointing
- Other Satellite Pointing
- Orbital Maneuver Mode
- No control

These different types of manoeuvres require different types of control. LQG and D-control were tested for various aspects of the mission and found satisfactory.

Perhaps the single most difficult GNC aspect is related to navigation. This is due to the fact that in order to fulfil the mission phases in which close formation flying occurs, the satellites must have enormous on-board autonomy and very precise navigation algorithms. The accuracy requirements range from meters to a few centimetres, depending upon the phase. In order to fulfil, these requirements, a series of different absolute and relative GPS-based navigation algorithms have to be implemented. They are:

- State Vector (SV-GPS): This provides an accuracy on the order of meters, generally sufficient for DEM coarse and fine formation acquisition stages of the mission.
- Single and Double Differences (SD-GPS and DD-GPS): These provide accuracies on the

order of tens of centimetres, which is necessary for the HEX coarse formation acquisition and the IRSI phases of the mission.

- Kinematic (K-GPS): This provides an accuracy on the order of 5 to 10 centimetres, and is necessary for the HEX fine formation acquisition phase.
- European Enhanced Formation Flying (E^2F^2) using pseudolites: This provides accuracies on the order of a few centimetres. It is needed for the most demanding and closest formation flying mission phases, such as the HEX experiment.

In addition, it should be noted that as with any close-flying formation, the contingency phase is an important one because of the close proximity of the satellites during some phases of the mission (there may be as little as only 50 meters difference between the 2 satellites). For this reason, a Collision Avoidance Manoeuvre is essential. The CAM must be activated automatically when the Flyer violates the safety sphere of the master.

All of the GNC strategies were tested during the study using FAMOS, a Formation Flying Analysis and Mission Operations Simulator developed by GMV. A couple of results from these simulations are presented in the next figures for the HEX Experiment Sub-phase, one of the most demanding in terms of GNC strategies.

In the HEX Experiment sub-phase, the Flyer and Master are 50 meters apart and this relative position must be maintained with an error of less than 5 cm. The two key elements to maintaining this level of accuracy are the LGQ control algorithm and the E^2F^2 navigation algorithm. Figure 2 shows how the LQG control algorithm is able to maintain the relative position of the two satellites well within the 5 cm control box.

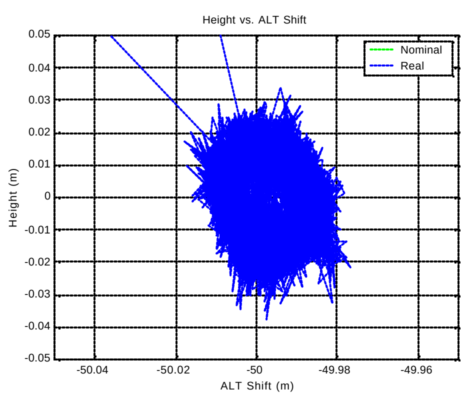

Figure 2: HEX Experiment – LVLH Diagram of Master & Flyer Relative Position

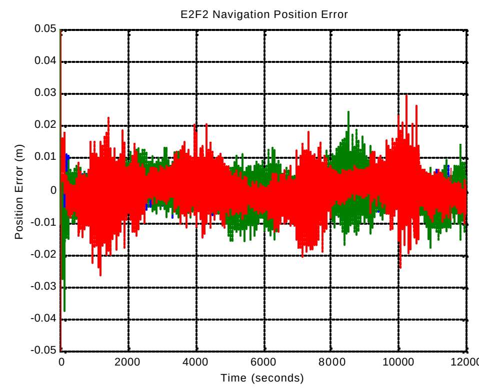

Figure 3: HEX Experiment – Position Error using the E^2F^2 Navigation Algorithm

Figure 3 above shows that the E^2F^2 navigation algorithm accuracy is generally better than 3 cm. This is good enough to provide the LQG controller with accurate enough information in order to maintain the 5 cm control box needed for the relative position of the two satellites.

OPERATIONS AND GROUND SEGMENT

From the GNC strategy it was clear that the maximisation of groundstation accesses was an important feature of the mission. This allows not only more data download opportunities, but also allows better performance monitoring. This is particularly important during the FF phases of the mission, when the spacecraft will fly very close to each other, with an increased risk to the mission.

Form the initial access times study it was clear that during most of the mission there will be a limit of 30 contacts per day with a maximum duration of 8 minutes. This assumes a network of seven groundstations (Kiruna, Villafranca, Maspalomas, Malindi, Perth, Redu and Kourou) selected for there availability to an ESA mission. The use of data relay satellite networks was investigated but they would not provide the necessary data rates, and would impose a severe constraint on the design of the mission. It should be noted that this distribution of groundstations generates a large heterogeneity of accesses, since some orbits will have no contacts while others will have a very long access time, when the spacecraft is flying over Europe.

The spacecraft will have a high level of on-board autonomy, being capable of independently performing all the tasks of the mission. Despite this there are two cases that cannot be handled autonomously:

- General spacecraft reset
- Incorrect functioning of the FF systems

In the first case, at least one of the spacecraft goes into safe mode, in case a serious fault occurs. If only one spacecraft is affected, the other can take the necessary steps to break the formation and to avoid a collision, but if both spacecraft are affected then this is a risk to the mission, and the sooner contact with the controllers is established the better.

Table 3 - Mission phases and contacts per day

Description	Type of operation
Formation deployment	Low intensity. 2 contacts/day required
DEM acquisition (general)	Low intensity. 2 contacts/day required
DEM acquisition (tests)	High intensity. At least 15 contacts/day required
DEM Experiment	High intensity. At least 15 contacts/day required
Extend DEM Phase	High intensity. At least 15 contacts/day required
HEX coarse acquisition (tests and validation)	High intensity. At least 15 contacts/day required
HEX coarse acquisition	Low intensity. 2 contacts/day required
HEX fine acquisition (including tests)	High intensity. At least 30 contacts/day required
HEX station keeping (initial tests and validation)	High intensity. At least 30 contacts/day required
HEX station keeping (tests and validation)	High intensity. At least 15 contacts/day required
IRSI Acquisition (validation)	High intensity. At least 15 contacts/day required
IRSI Acquisition (routine)	Low intensity. 2 contacts/day required
IRSI Experiments	High intensity. At least 15 contacts/day required
De-orbiting	None

The strategy for FFDEM is to launch with only the most basic software on-board and then upload the flight software to the RAM of the spacecraft. This increases the consequences of a major reset (the spacecraft can loose the majority of its flight software), but allows to a much easier correction of such events. In the second case, since the FF systems are experimental, it is likely they will have performance problems, what may risk the mission when flying in very close formation. For this reasons, when flying in very close formation flying, the maximum number of contacts possible will be required. Also, the FF subsystems need to be validated previous to any critical use, what will require large amounts of data to be downloaded.

The number of accesses required per day was optimised to the different phases of the mission, as can be seen in Table 3, where to each one of the mission stages, was attributed a number of contacts per day. Low risk stages, such as the initial phases of commissioning, when both spacecraft are far away from each other, require only two contacts a day, mainly to monitor the health of the spacecraft..

As mentioned before, the baseline is to use seven groundstations for TT&C and for data collection. Since the mission requires intensive operations, especially during experimental phases, it is likely that operators will be required to be on attendance for a large percentage of the mission time, or at least will need to be available to monitor each pass. The control will be centralised, although data analysis of experiment results is likely to be performed directly by the users. Several configurations are possible for the ground segment, but for FFDEM they have been reduced to two main options. One, uses ESOC as the central node to collect all the data and then distribute it to the control centre and to the data users. The control centre can be placed at ESOC or at a remote location (Figure 4):

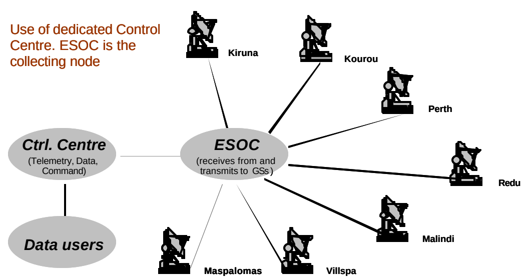

Figure 4 - Centralised ground segment

This option has the disadvantage of all the data going through a central point, what can represent a high cost option, particularly if the control centre is not placed at that central node. The use of ESOC as the central node can be problematic at times, if there is a large usage of its capabilities by other missions. An alternative to the use of a central node, is to use a distributed architecture, based on direct, low cost high speed internet connections between the groundstations, control centre and data users (Figure 5):

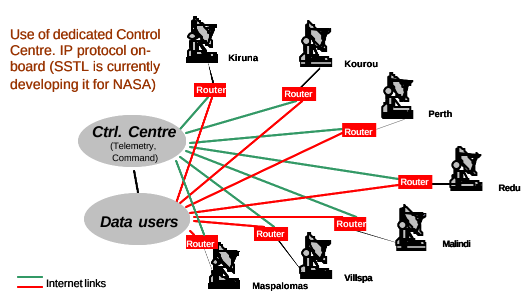

Figure 5 - Distributed architecture for the ground segment

At this stage, no final configuration for the ground segment has been selected.

SYSTEM DESIGN

The objectives for the FFDEM mission definition study clearly stated that the overall system design should be based on a low cost, small spacecraft platform already existing or under development in Europe. Several platforms were initially considered, but it was decided to concentrate on the Advanced Microsatellite Platform (AMP) defined by a consortium of companies[3] working under contract to ESA. This platform is intended as the future European "standard" in small spacecraft missions, and it was designed using state of the art subsystems and concepts. It is a highly optimised design, aimed at increasing as much as possible the mass fraction of the payload.

The requirement to base the design on the AMP platform imposed severe constraints on the FFDEM design, for two main reasons:

- Most of the AMP proposed subsystems do not yet exist in a flight configuration, being still at the development phase
- The AMP characteristics reduce the freedom to select an optimised design for FFDEM

The first point is the most challenging, since it makes it very difficult to establish the basic budgets for the mission, because if a system requirement is not known, it is difficult to budget for that requirement. Besides, programmatic and budget analysis is also very difficult. Given this, it was decided very early in the study, that the defined mission would have to be possible to achieve with currently existing units, with the natural exceptions of units that currently have no direct equivalent and require development anyway. This approach meant that at the end of the mission definition study, the proposed platform could be immediately built, albeit with less optimisation than if many of the subsystems currently proposed for AMP were used.

General

Table 4 presents a list of the specifications for the FFDEM spacecraft, that resulted from the analysis of the requirements of the statement of work[4], coupled with discussions with ESA and the initial analysis performed for the definition of the GNC strategy. The spacecraft were designed to satisfy these specifications, implementing the best technical solutions to achieve them.

Table 4 - Initial specifications for the FFDEM mission

Subsystem	Specification
General	• >17 month orbital life • LEO orbit (561km), at high inclination (90º)
Structure	• AMP or similar platform • <120 kg launch mass • 0.6m x 0.6m x 0.7m volume • Compatible with available LEO launchers
Propulsion	• ΔV: >11.0m/s • µN to mN thrust • Less than 1µN thruster noise • Electrical propulsion preferred.
Power	• 28V power bus • Operation of payloads shall be possible at any point in the orbit
Communications	• 2Mbps downlink • 128kbps uplink • BPSK • Intersatellite link capability
OBDH	• Dimensioned to implement the control algorithms and allow general housekeeping operations • Suitable network for telemetry and data exchange
AODCS	• Suitable set of sensors for attitude determination, including: o Low accuracy o High accuracy • Suitable set of actuators for attitude control: o Coarse control o Fine control • Units for FF demonstration: o E^2F^2 unit o Laser ranging unit o Optical Relative Position/Orientation Estimator (ORPO estimator)
Other	• Demonstration payloads might be flown

Although the specifications are fairly open on most subsystems, they clearly define their performance targets and in some cases this limits the options when selecting subsystem components.

For the FF demonstration, it will be required to include a set of sensors specific to formation flying. The units to be flown on FFDEM for this purpose are the "European Enhanced Formation Flying" (E^2F^2) unit, a Laser ranging unit (both under consideration by ESA) and the Optical Relative Position/Orientation (ORPO) estimator, under development at the Surrey Space Centre. The E^2F^2 unit is of particular interest to the mission, as it includes both a GPS Receiver for orbit determination, and an intersatellite RF link.

Originally, one spacecraft was referred to as "master" or "target" and the other was referred to as "flyer". In order to reduce costs, it seems logical to have one spacecraft less capable, which is a reduced version of the other one. In such an approach, the master would have a simplified propulsion system, but would include most of the FF units, while the flyer would be a "dumb" unit with more sophisticated propulsion but dependent on the master for control during FF phases. This approach would require spacecraft more reliable, following a classical approach of including full dual redundancy on the platforms (otherwise one subsystem failure would loose the mission), what would defeat the attempt to cut costs. Following in the SSTL tradition of small spacecraft design, a more robust approach is preferred, with "decentralised" redundancy being preferred. In this approach, either spacecraft can take the role of flyer or master, including each one all the necessary FF subsystems. These are implemented as single string versions, although the bus systems (such as OBCs, RF and power) are partially redundant, and will operate in graceful degradation mode (what means that failures might reduce the capability of the mission but will not mean its loss). This approach has the advantage that both spacecraft are the same with the reduction in design and build costs, and in the case one spacecraft looses all its FF units (sensors and propulsion), the other can still perform the mission. The obvious disadvantage is that the platforms will be more complex. Despite this, in order to correctly simulate the HEX demonstration conditions, it might be necessary to implement an artificial shape difference on the spacecraft (this is still under analysis), since the HEX mission will be implemented using two different platforms.

Attitude and Orbit Determination and Control Subsystem (AODCS)

Table 1 specifies the attitude and orbit performance required from the AODCS throughout the mission, which can be divided in:

- Low accuracy control and determination (attitude and orbit) phases
- High accuracy control and determination (attitude and orbit) phases

Both have very different requirements and each spacecraft will fly a set of different subsystems to be used on each type of phases, since this optimises the design of the AODCS, reducing usage of power.

For low accuracy attitude control and determination, a set of low power, simple sensors and actuators was selected. The sensors are magnetometers and analogue Sun-sensors, capable of attitude determination to within 0.1º. These are complemented by a set of redundant magnetic torquers, which will be used to control the attitude to within 0.3º. These will be used during non-formation flying phases, such as LEOP and early transition from the DEM experiment phase to the HEX experiment phase. For low accuracy orbit determination, no specific sensors will be employed, although the GPS receiver on the E^2F^2 unit might be used, but for control, a nitrogen propulsion system is the baseline. This is a relatively high thrust propulsion system, to be used on major impulsive manoeuvres, when high precision is not required.

High accuracy attitude determination will be achieved by dual redundant star cameras, coupled with an inertial measurement unit, while high precision attitude control will be achieved by four momentum wheels mounted in tetrahedral arrangement, to produce zero momentum bias control. This configuration provides inherent redundancy to one wheel failure.

Figure 6 presents a block diagram of the AODCS implementation. Although included on the diagram, the SGR-20 GPS receiver will only be flown if it is considered important to have a redundant replacement to the receiver of the E^2F^2 unit.

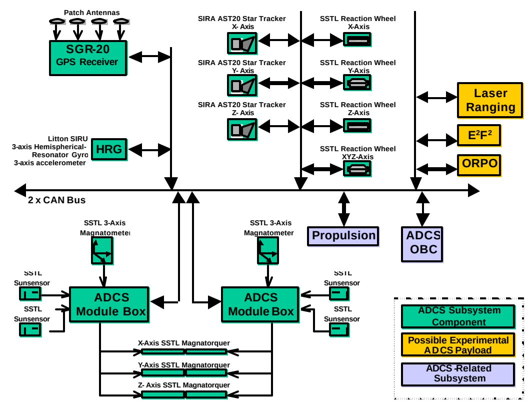

Figure 6 - AODCS block diagram

Most of the processing will take place at the OBDH subsystem, with one of the On-Board Computers dedicated to AODCS tasks.

For high precision orbit control, the baseline is a Field Emission Electrical Propulsion (FEEP) unit, with six thrusters. This will be used for high precision orbital manoeuvres, during the close FF phases when propulsion firings in all directions will be required. This option is still under review as the power requirements and the mass of the system impose a severe penalty on the mission budgets.

The dual type propulsion system optimises its design, since a system capable of doing both high thrust manoeuvres and very low thrust manoeuvres, is not viable in the limited mass and volume available. The ΔV requirements as a function of the mission phases (see Table 1) are presented in Table 5. To achieve these values, the mass of the N_2 system was estimated at 6.7kg (including propellant) while the mass of the FEEP was estimated at 17.2kg (including power supply and propellant). It should be noted that the FEEP configuration chosen for FFDEM is not redundant, and as such, a failure of a thruster will have an implication on the manoeuvring performance.

Table 5 - DeltaV requirements

Phase	Dv required from N_2 system (m/s)	Dv required from FEEPs system (m/s)	Total Dv (m/s)
1	2.101	0.000	2.101
2	3.920	0.000	3.920
3	2.101	0.171	2.272
4	0.000	0.569	0.569
5	TBD	TBD	10.50
6	0.00	0.00	0.00
Totals	**8.122**	**0.740**	**8.862**

On-Board Data Handling (OBDH) Subsystem

The AMP documentation[3] largely defines the configuration of the OBDH subsystem for the platform, but most of the units proposed for the subsystem are currently under development or are only now reaching prototype phase. This is the case for instance, with the Spacewire bus or the Leon processor. The use of these as the basis for the OBDH subsystem of FFDEM is not viable in the envisaged timescales, and also makes it very difficult to have reasonable estimates of the characteristics of the subsystem. For this reason, alternative solutions were selected fro FFDEM.

The links between subsystems are of two types:

- Point-to-point links for high speed data transfer
- Bus network links for telemetry, telecomand and low speed data transfer

Following SSTL's highly successful use of Control Area Network (CAN) for general TT&C and low speed data transfer, this was selected as the bus of the spacecraft. The CAN controllers in each subsystem are the TT&C subsystem of the spacecraft, being responsible for command decoding and telemetry gathering and formatting. To complement the CAN, a set of point-to-point Low Voltage Differential Signal (LVDS) links will be used between subsystems that require transfers of high volumes of data.

Concerning on-board computers (OBC), the choice was for the SIL DHS-S32 unit, based on the ERC-32 processor, and currently flying on the PROBA mission. Three units are to be flown, with one allocated to AODCS tasks, one for housekeeping duties and a third unit that will take over from any of the other two in case of failure. This arrangement provides a graceful degradation operation, since in case of failure of two units, the mission can still progress, although just one OBC operating will need to be shared for AODCS and housekeeping tasks, what will reduce the performance of the platform. The option of flying four units represented a large mass penalty on the mission.

The presence of several experimental payloads requires the capacity to store the data generated, since that will be necessary to validate their operation. It is estimated that under nominal operation, up to 630Mbytes might need to be stored on-board for downloading. This requires mass memory storage, provided on FFDEM by two Solid State Data Recorders (SSDR), provided with point-to-point links to some of the payloads and to the transmitters. These are "intelligent" units, capable of processing the data while it is stored, thus allowing its compressing or selection, what reduces the data download requirements. Figure 7 presents a simplified block diagram of the OBDH subsystem.

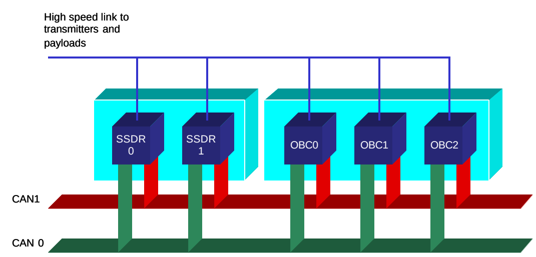

Figure 7 - Simplified block diagram of OBDH subsystem

The chosen OBCs are heavy, over-specified to the requirements and power hungry. Despite this, they were deemed the best available option and were thus selected. In case a better OBC becomes available and is ready to be included on the spacecraft on time, then the DHS-S32 can be replaced.

RF Subsystem

In order to maximise the number of groundstation accesses in one orbit, the spacecraft are equipped with a S-band communications system on both uplink and downlink. A 2Mbps, BPSK downlink was selected as this is compatible with most groundstations, while still allowing the download of all the necessary data. On the uplink, a 128kbps BPSK link was selected, although the S-band receiver will be used mainly at the lower rate of 4kbps, only switching to 128kbps during software uploads or other high volume data uploads. The protocols used will be compatible with ESA groundstations, as this is the preferred network for the FFDEM mission.

The type of operation of the spacecraft on the FFDEM mission means that both spacecraft can be in any orientation towards Earth, even during accesses to the groundstations. For this reason both receiving and transmitting antennas have to provide hemispherical coverage. This reduces the gain of the antennas, and would require an increase in the power of the spacecraft transmitters. The antennas selected, two receiving patch antennas and two transmitter quadrifilar helixes, provide almost hemispherical coverage, with slight "blind" angles, where it will not be possible to transmit or receive at the higher data rates, but low rate TT&C communications can still take place. In Figure 8, we have the link margin for the downlink, when the output power of the transmitter is 7W:

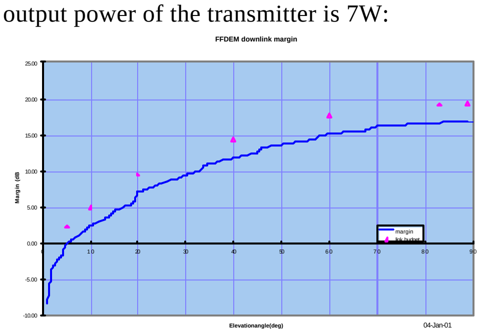

Figure 8 - Downlink margin (blue line)

Figure 9 presents a block diagram of the RF subsystem:

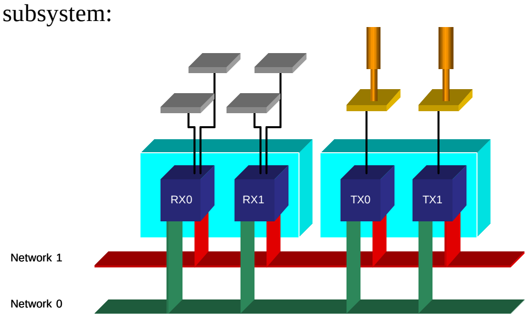

Figure 9 - RF system block diagram

Each receiver has two antennas attached to it for providing hemispherical coverage in case of loss of attitude control (the receivers operate in hot redundant mode and their power lines are non-switchable) while each transmitter antenna is placed on opposite sides of the spacecraft, and the one with the best line of sight to the groundstation will be selected.

Power Subsystem

The power budget for FFDEM revealed itself difficult to derive, as the variety of operational scenarios and the uncertainty on the power requirements of many subsystems made it difficult to obtain an accurate answer. In order to correctly analyse the power problem, the mission was divided in "power phases" that group the mission phases that share similar power requirements. These are loosely connected to the mission phases presented on Table 1, as can be seen on Table 6. To each power phase it was assigned an Orbit Average Power (OAP), that represents the amount of power that on average will need to be supplied to the platform.

Table 6 - Power phases

Power Phase	A	B	C	D
Description	Formation deployment	DEM and De-orbiting	HEX Acquisition (coarse and fine)	HEX Experiment and IRSI-Darwin
Power (W)	**29.7**	**44.4**	**76.8**	**90.3**

Logically the power requirement tends to go up with the complexity of the phase, particularly due to increased power necessary for the FF subsystems.

From the power generation side, it was clear that it was necessary to accommodate all the power phases, at least during some time in the mission timeline. Although there was not a limit imposed on the size of the solar arrays for the mission, it is natural that they should not be much bigger than the dimensions of the structure. On the other hand, the full attitude agility necessary on this mission, means that both spacecraft can have any orientation towards the Sun, at any time. This has a serious impact on the mission as it will either constrain the manoeuvring capability of the platforms or reduce the amount of available power, unless the solar array configuration can accommodate this. For this reason, a configuration featuring four body mounted solar panels and two deployable, double sided panels was chosen, as can be seen on Figure 10

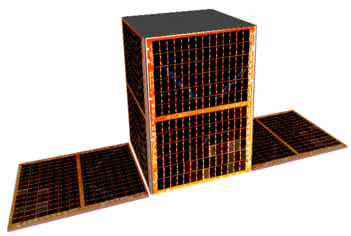

Figure 10 - Solar array configuration

Although this configuration is more expensive and heavier, the deployable arrays significantly reduce the dependence of the power budget on the spacecraft orientation. The overall orbit average power margin can be seen in Figure 11, as a function of time and for the different power phases. This show that for power phases A and B, the spacecraft power systems are largely over-engineered, but for power phase D there are periods of the time where the power budget is negative (this arises from the variation of the β angle and consequent variation of the eclipse duration). This means that the mission phases associated with those power phases cannot take place at that time. The mission timeline takes this into account,

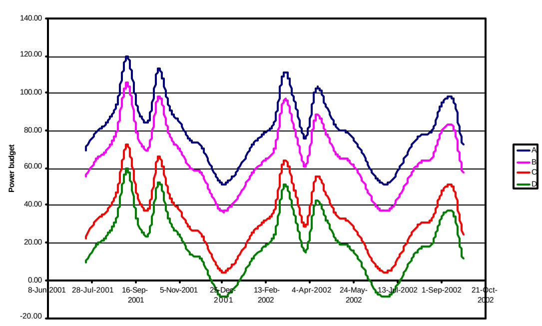

Figure 11 - Power Margin for different power phases

The power subsystem architecture is similar to the one proposed to the AMP platform[3], using a centralised power distribution unit, with redundant power switches for operating other subsystems. The batteries are Lithium Ion batteries, with 10 A.h capacity, on a voltage of +28V.

Structure

The structure selected for the FFDEM platform is the same as the one for the AMP[3], based on a carbon fibre thrust tube, to which subsystems are bolted on. This solution is far from ideal from the point of view of assembly and integration, but it allows for a very optimised structure. This option is volume constrained, but it is possible to accommodate the mission in the available volume, as can be seen in Figure 12

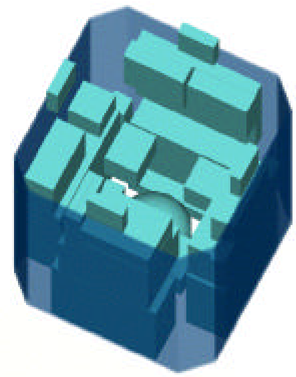

Figure 12 - Accommodation of FFDEM on the AMP structure

The main body of the spacecraft will fit in a volume of 0.6m x 0.6m x 0.71m, before deployment of solar arrays, but this does not include the antennas, and E^2F^2 and the Laser ranging units, that are still under development and whose final dimensions are at this stage unknown. The Laser ranging unit and the ORPO estimator will need to have line of sight to the other spacecraft while operating, and as such will need to be placed where they can have an unobstructed view. Also, the placement of the FEEP thrusters needs to be carefully chosen as there have been several questions raised on plume impingement.

The mass budget currently shows that the spacecraft is heavier than the target mass of 120kg, currently standing at 140kg (including contingency), although several mass saving measures have been identified. Figure 13 presents the mass distribution by subsystem:

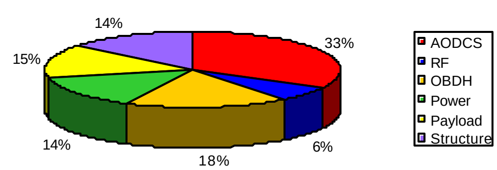

Figure 13 - Mass distribution

Current mass reduction measures include replacing the OBCs with lighter units, or in case that is not possible, to use a lighter, less powerful OBC instead of the third DHS-S32, that would be responsible for housekeeping tasks in case of failure of one of the DHS-S32. For example, if three SSTL OBC695 were flown, it should be possible to reduce the total mass of the OBCs from 21kg to 6kg. There are also significant mass savings to be made on the FF subsystems, namely the E^2F^2 that currently has an estimated mass of 12kg. Furthermore, the propulsion system can be significantly optimised although that will require significant development work, particularly on the FEEPs.

Environment and Thermal

Simulations were performed to establish type of environment that can be expected to the FFDEM mission. The main interest was to establish the total dose to which the subsystems will be exposed during the mission lifetime. With a general shielding of 2mm, for a two year mission, the total dose will be well below 10krad (Si), in worst case conditions, although in normal conditions and given that the effective shielding is higher than 2mm for almost all subsystems, the expected value is actually smaller than 5krad (Si). This is still quite above what is seen by the SSTL missions currently in orbit, but is low enough to justify the use of COTS components.

A basic thermal analysis was performed to evaluate the temperature distribution on the spacecraft. The Thermal Mathematical Model created was solved for nominal conditions and showed that a fully passive thermal control subsystem is enough to maintain the spacecraft temperatures, as can be seen in Figure 14:

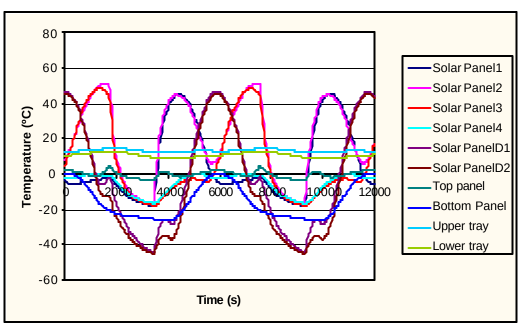

Figure 14 - Temperature distributions for maximum eclipse season.

LAUNCH

Several launch possibilities were identified, although no selection took place. The main constraint on the launch is that it must be possible to launch both spacecraft from the same launcher, since it was demonstrated [2] that an eventual separate launch would require at least 15m/s of ΔV to bring the spacecraft together, what would require a much larger propulsion system, with the consequent problems of mass and accommodation. This was judged too much of a risk and so, a joint launch is the baseline.

Five launchers were considered on the analysis for their accommodation capabilities:

- Ariane-5 (ASAP)
- Cosmos-3M
- Rockot
- DNEPR
- Start-1

Only European launchers have been considered in this analysis mainly because of easier access to them. Nevertheless, the most likely American launcher would be Pegasus, but given that the dynamic loads on the structure are above the ones to which the AMP structure has been designed, the choice of this launcher would require further structural qualification, with the increased cost associated with it.

A launch on Ariane-5 should be possible in terms of accommodation, although there is some confusion about the exact dimensions to be allowed on the launcher. The main problem with such a launch is the availability to the required orbit, since a dedicated launch is highly unlikely.

Cosmos-3M is a viable option as the launcher can accommodate the two spacecraft on a side-by-side configuration. A similar configuration could be selected on both DNEPR (albeit not currently flying into polar orbits this might change in the future) and Rockot launchers. In any of these cases, a dedicated launch would probably be required, with the consequent high cost.

Start-1 faring is not big enough to accommodate the two spacecraft.

RISKY TECHNOLOGIES

Throughout the study, there were several technologies that were identified as critical for the mission. Most of these are new and untried technologies, which carry a certain degree of risk in both programmatic and operational terms. In general, all the FF technologies are risky, imposing severe risks to the mission in case of failure, with a collision being the biggest of them. On the program side, they are a schedule and cost risk, as they are not currently available (except ORPO), and this can generate serious delays to the program.

SCHEDULE

One of the outputs of the study was a tentative schedule for the mission design, build and test. Keeping with the spirit of the project, a short schedule is in itself an objective, and as such, an effort was made to minimise the duration of the different steps. The overall duration of the project from start to launch is estimated at 24 months, but this assumes assembly and testing practices similar to the ones used by SSTL:

Table 7 - FFDEM Schedule

Project Phase	Duration (months)	Start	End
Project start	-	T0	
Mission Definition	1.0	T0	T0+1.0
Preliminary Design	6.0	T0+1.0	T0+7.0
Spacecraft Design	3.0	T0+7.0	T0+10.0
Flight Build	5.0	T0+10.0	T0+15.0
Assembly Integration and Test	3.0	T0+15.0	T0+18.0
Environmental Tests	3.0	T0+18.0	T0+21.0
Launch campaign	3.0	T0+21.0	T0+24.0

Although some of these durations might be a bit optimistic, they are feasible if no major delays occur in the project.

CONCLUSION

The study has demonstrated the viability of accomplishing the FFDEM mission using small low cost platforms. It was demonstrated that this could be achieved for a cost under €25,000,000.00 (cost to ESA under their standard conditions), in a reasonable schedule, and making as much use as possible of existing facilities. This would be a cost effective way of flight qualifying and demonstrate formation flying techniques and systems for future missions.

As new, better subsystems become available, it is possible to significantly increase the performance of the mission, optimising it in terms of mass and power. These optimisations could be used to increase the spacecraft payloads, by allowing some other, non-FF payloads to be flown, thus increasing the usefulness of the mission beyond the strict demonstration and validation role.

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