CALIBRATION OF THE MICRO-NEWTON PROPULSION SYSTEM FOR THE LISA PATHFINDER DRAG-FREE SATELLITE

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OVERVIEW

LISA Pathfinder will test the technologies needed for ESA's future mission LISA, which aims at improving our knowledge of the universe by detecting gravitational waves, a phenomenon predicted by Einstein's General Theory of Relativity in 1916. The mission will fly a European payload, called LISA Technology Package (LTP), developed by European institutes and industry using national funds from seven Member States (Italy, France, Germany, Spain, United Kingdom, The Netherlands and Switzerland) and from ESA.

In general, the LISA Pathfinder mission consists of placing two test-masses in a nearly perfect gravitational free-fall and of controlling and measuring their motion with unprecedented accuracy. This is achieved through advanced technology comprising inertial sensors, a laser metrology system, an ultra-precise micro-propulsion system, and a drag-free control system (DFACS) that is designed in such a way that any non-gravitational forces along the sensitive axis are minimized.

As soon as a drag-free controlled test mass moves away from its nominal position, the DFACS commands the micro-propulsion thrusters such that the spacecraft remains centred on that test mass. The micro-propulsion system contains three sets of four thrusters. The thrust gain, thrust direction will differ from its nominal values. The presented thruster calibration procedure aims at the measurement of the deviation from nominal in thrust direction, thrust magnitude, and effective lever arm. Since the micro-propulsion system is also one of the critical technologies to be tested with LISA Pathfinder, it is necessary to characterize the performance of this novel micro propulsion system, the Field Emission Electric Propulsion (FEEP), and thus, it can be considered as a separate mission goal.

The paper shows the principle of the developed calibration procedure, consisting of an on-board identification experiment and the ground based data processing and estimation algorithms. The two test masses do not provide acceleration measurements directly, which requires new calibration approaches. Performance results are obtained with simulated telemetry data. Furthermore, limitations of the currently available calibration algorithms are presented and future development activities as combined efforts between industry and university are pointed out.

1. LISA PATHFINDER AND THE LISA TECHNOLOGY PACKAGE

1.1. Primary Mission Goal

The technology test mission LISA Pathfinder can be considered as successful if the differential free fall between the two test masses along one axis, the sensitive axis, can be verified with an accuracy of $3 \cdot 10^{-14}\ \mathrm{m/\ s^2/\sqrt{Hz}}$ in the measurement bandwidth between 1 mHz and 30 mHz. This is accomplished by measuring the distance between the test masses along the sensitive axis with a very precise optical metrology system. From this data, the acceleration can be obtained on ground. Figure 1 shows a schematic view of the spacecraft, the two free-flying test masses, and the sensitive axis which is the connection line between the (nominal) centres of mass of the test masses.

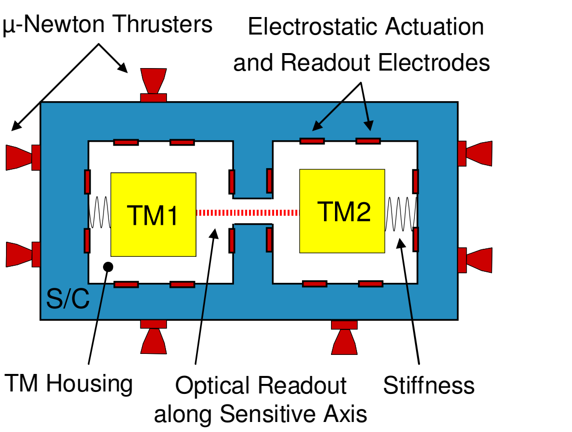

FIG 1. Schematic representation of the LISA Pathfinder spacecraft and the two test masses together with available sensors and actuators.

1.2. Available Actuators and Sensors

Two different types of actuators are implemented on-board LISA Pathfinder, the µ-Newton thrusters and the electrostatic suspension system. The thrusters can apply forces and torques on the spacecraft along 6 degrees of freedom (DoF), whereas the electrostatic suspension system can directly apply forces and torques on each of the two test masses (12 DoF).

The test mass position with respect to the test mass housing and the spacecraft, respectively can be measured either by means of an optical readout system or by an electrostatic readout system. The optical readout provides the differential test mass displacement along the sensitive axis and selected coordinates of both test masses (in total 6 DoF). On the other hand, the electrostatic readout provides measurements of all six test mass coordinates of the two test masses with respect to their housings (12 DoF).

1.3. Stiffness Coupling

The test masses are dynamically coupled to the spacecraft through stiffness ("virtual spring", compare Figure 1) that is originated mainly by spatial gradients of the disturbance forces. This stiffness affects the non-gravitational acceleration on the test mass, and its dynamic behaviour. In fact, the stiffness may even be negative, which leads to unstable dynamics. This is one of the reasons why the system has to be closed-loop controlled by the DFACS.

1.4. Control Principle

The spacecraft attitude as well as the relative position and attitude of the two test masses are controlled by the Drag-Free, Attitude, and Suspension Control System (DFACS). The DFACS for LISA Pathfinder was developed completely by ASTRIUM GmbH. Real-time test bed activities are currently ongoing. More details about the design of the DFACS are given in [1].

However, two different control principles exist for the twelve test mass coordinates, namely *drag-free-* and *suspension* control. The principle of drag-free control is that the spacecraft has to follow a test mass along the drag-free controlled coordinates. Since this is only possible for 6 coordinates, the other six test mass coordinates must be suspension controlled such that the test mass follows the spacecraft. This is done by applying forces and torques to the respective suspension controlled coordinates by means of the electrostatic suspension system.

2. PROBLEM DESCRIPTION, ESTIMATION PROCEDURE, AND CONSTRAINTS

2.1. Required Parameters

At a certain thrust level each of the 12 FEEP thrusters on-board LISA Pathfinder can be characterized by

- The ratio between the real force and the commanded force k (one unknown parameter)
- The real thrust direction $\vec{u}$ (two unknown parameters, since $\vec{u}$ is a unit vector)
- The effective lever arm $\vec{d}$, which gives the resulting moment with respect to the spacecraft centre of mass (three unknown parameters)

This gives a set of 6 unknown parameters for each thruster. Thus, for all the 12 FEEP thrusters, the problem consists of 72 unknown parameters that are required for a complete characterization of the µ-Newton propulsion system. Figure 2 illustrates the physical meaning of the thruster parameters schematically:

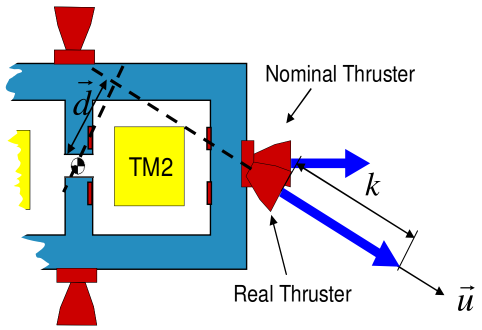

FIG 2. Schematic representation of the unknown FEEP thruster parameters that are required to characterize the µ-Newton propulsion system.

The nominal parameters for each of the twelve thrusters are supposed to be known with certain accuracy. However, as indicated in Figure 2, by reasons of mounting errors, tolerances or imperfections in the production, vibrations during launch, and long-term operational effects during flight, the parameters of the FEEP thrusters might be deviated from the nominal parameters, leading to different parameter values.

Not all of the deviations can be determined on ground, especially those that are caused during launch and operations can not be detected. Thus, the values of the 72 thruster parameters have to be determined in-flight.

2.2. Identification Principles and Experimental Procedure

The whole identification procedure can be subdivided into the following two main steps: the *identification experiment* on-board LISA Pathfinder and the execution of the *identification algorithms* on Earth (see Figure 3).

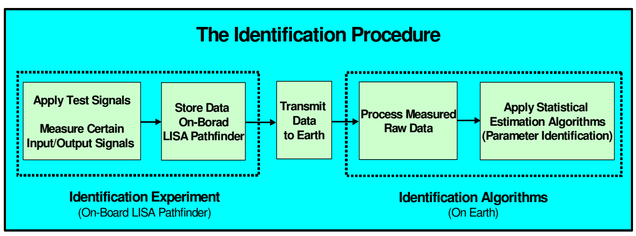

FIG 3. Overview of the operational procedure for thruster identification.

2.2.1. Identification Experiment

The first step is the execution of the identification experiment in-flight. This includes the application of the test signals (directly commanded thrust-signals to the FEEP thrusters) and the recording of the corresponding system response (resulting test mass displacement or suspension controller command). Since the considered system is unstable and thus closed-loop controlled, the identification experiment must be accomplished within the given closed-loop control configuration (see Figure 4).

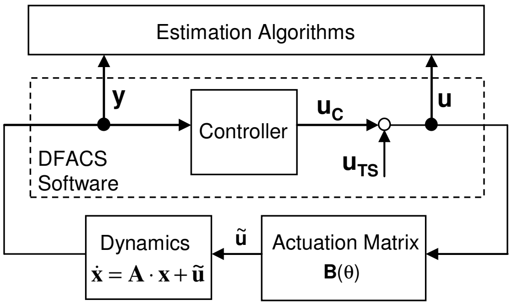

FIG 4. Principle of the thruster calibration approach.

The commanded thrust levels are composed of the test signals and an additional controller signal that stabilizes the system. As both signals are known, the sum of both signals will be recorded. This principle allows for the *direct identification* of the open-loop plant parameters without considering the controller transfer functions. Moreover, this method does not rely on steady state conditions, i.e. one can immediately use the data after switching on the test signals [2], [4].

2.2.2. Identification Algorithms

The second step is the execution of the identification algorithms on ground. This includes processing of the transmitted raw data and the application of statistical estimation algorithms to the processed data. The required parameters are finally obtained using standard parameter identification methods [2].

Both the excitation signals and the system response are processed by means of statistical estimation algorithms such that the unknown 72 thruster parameters of a predefined mathematical model are obtained. The mathematical model is linear and includes spacecraft and test mass dynamics. Furthermore, all of the 72 desired parameters are contained in the actuation matrix of the model (compare Figure 4).

While an originally derived mathematical model usually is given in continuous-time, the actual parameter estimation process is implemented in discrete-time (the on-board sampled telemetry data is only available in discrete-time and the numerical implementation on a digital computer is more adequate). Since the parameter estimation algorithms are based on discrete-time model descriptions, they estimate the corresponding discrete-time parameters (in a statistical sense). A second step is required in order to recover the physically meaningful thruster parameters of the original continuous-time model. The recovery can be considered as an inversion of the continuous-time to discrete-time conversion and makes use of the known interrelation between the two model descriptions. Recovery algorithms are available in [4].

The used statistical estimation methods are least-squares and instrumental variables (IV) parameter estimation algorithms. The IV method is required because the standard least-squares method is not always sufficient in order to obtain accurate results. However, it requires instrumental variables that are generated by “noise-free” simulations with true parameters. Of course, these are not known and therefore an iteration (”*bootstrap-iteration*”) must be performed whereas the initially used parameters are obtained by the least-squares method. The simulation for the IV generation also contains a model of the feedback controllers, which may not be necessarily identical with the on-board controllers [3], [4].

2.3. Constraints

The major constraint for the determination of the 72 thruster parameters is the duration of the on-board identification experiment, which shall be minimized. Therefore the required parameters of all twelve thrusters shall be determined with one experiment.

3. APPROACHES FOR IDENTIFICATION OF THRUSTER PARAMETERS

Two methods for the identification of the µ-Newton propulsion system parameters are presented:

1) The first method utilizes the commanded thrust signals and measurements of the relative displacement between test mass coordinates and the spacecraft.

2) The second method is based on acceleration measurements, i.e. the test mass suspension control axes are treated as accelerometers. The suspension controller commands are such that no relative displacement between the spacecraft and the test mass occurs.

3.1. Method 1: Identification Using Inertial Sensor Measurements

3.1.1. Basic Concept and Used Coordinates

Basic Concept. The parameters of the FEEP thrusters are obtained by applying test signals on the thrusters and observing the corresponding test mass displacement measurements of the drag-free coordinates, as provided by the capacitive readout system. Figure 5 shows the principle of the thruster identification using inertial sensor measurements schematically.

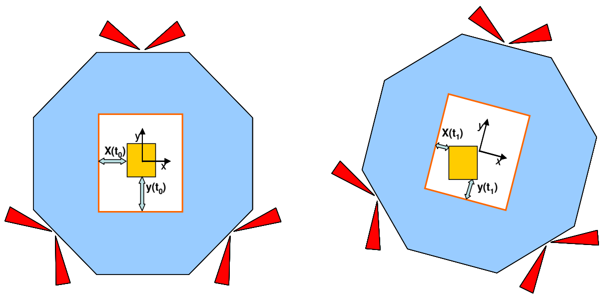

FIG 5. Principle of the thruster calibration approach: Commanded test signals on the thrusters cause certain displacement between test mass and spacecraft.

There is a mathematical connection between forces and torques on the spacecraft and the corresponding motion of the drag-free coordinates relative to the spacecraft [5]. The desired thruster parameters are part of the actuation matrix "B" of that mathematical description.

The plant dynamics has a $1/s^2$ behaviour for frequencies above the test mass stiffness [6]. Six test mass displacement signals and 12 thrust command are the input to the identification algorithm (see Figure 6).

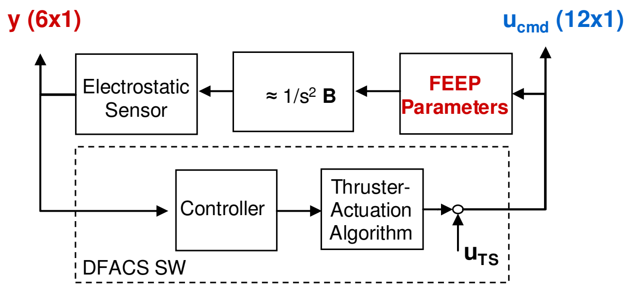

FIG 6. Illustration of commanded thrust test signals and recorded test mass displacements.

Used Coordinates. An important issue for the presented approach is the choice of suitable test mass coordinates. In order to achieve good results, ideally no other influence on the measured coordinates other than the one caused by the commanded thrust levels should be present. For this reason the drag-free coordinates have been selected. Notice that all six independent coordinates are required in order to extract the desired thruster parameters out of the B-matrix [5]. Thus, the displacement measurements of all six drag-free coordinates are required.

3.1.2. Setup of Identification Experiment

Test Signals. The test signals are applied as direct sinusoidal thrust commands with amplitude of 8 µN. Notice that the test signals have to be permanently biased in order to:

- Ensure positive thrust (negative thrust is not possible for an individual thruster)
- Cover different thrust levels (possibility to identify static nonlinearities of individual thrusters)

The DFACS controllers used for attitude, suspension, and drag-free control are separated in bandwidth, where $\omega_{ATT} < \omega_{SUS} < \omega_{DF}$. From a certain frequency on, the attitude and suspension controllers do not influence the drag-free coordinates any more.

By choosing test signals with a higher frequency than the bandwidth of the suspension-controllers, both the suspension controller and the stiffness do not influence the test mass dynamics of the drag-free coordinates. The lower frequency limit for the test signals is marked in Figure 7.

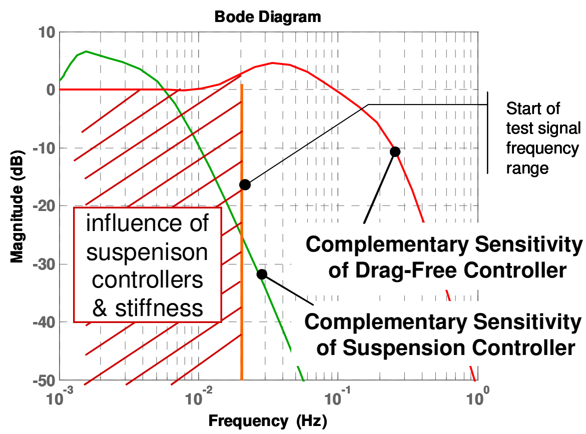

FIG 7. Frequency Separation of the suspension and drag-free control loops. Remark: attitude controllers have even lower bandwidth than suspension controllers.

Finally, the test signals for each of the 12 thrusters have to be separated in frequency, whereas the frequency range has been chosen between 20-31 mHz.

Required Data. Six electrostatic displacement measurements (one per drag-free coordinate) and twelve commanded FEEP thrust levels (sum of test signal and controller signal) have to be recorded on-board during the experiment. In total 18 signals must be recorded; the maximum sampling rate is 10 Hz.

Experiment Time. It has been demonstrated that after 10000 seconds of integration time the results are not improving any more. Therefore the experiment time is chosen to be 10000 seconds (with a sampling rate of 10 Hz). However, after 5000 seconds the improvement of the performance results is marginal. Thus, it can be stated that 5000 sec. of integration time is sufficient for the presented parameter estimation method.

3.1.3. Performance Results and Limitations

Performance Results. Results are obtained with simulated telemetry data. A detailed LISA Pathfinder performance simulator has been developed by ASTRIUM GmbH [7]. The simulation includes the 18 DoF non-linear dynamics of the spacecraft and the test masses and features, amongst others, the following relevant models: a GRS model on voltage level for electrostatic actuation and sensing, a FEEP thruster model, a detailed optical metrology model, an environment as well as a disturbances model. The flight-software with the Drag-Free and Attitude Control System (DFACS) and the Charge Management System (CMS) is also included.

Table 1 shows a summary of the performance results after an integration time of 10000 seconds. The data have been taken at 10 Hz. Note that not all 72 parameter estimates are listed, but only the worst estimation results and the average estimation errors.

Parameter	Worst case	Average
k error in %	1.3	0.8
$\vec{u}$ error in deg	1.3	0.6
$\vec{d}$ error in %	4.6	2.0

TAB 1. Performance results for the thruster identification using inertial sensor measurements.

Notice that the information about the effective lever arm is of minor importance since it does not represent the micro-propulsion characteristics itself. All results are based on perfect knowledge of the test mass and spacecraft mass properties.

Limitations. The performance limitation on the presented method depends on

- The delay in the control loop (total delay is composed of the FEEP thruster hardware delay, on-board computer delay, and inertial sensor hardware delay)
- Knowledge of spacecraft mass and geometrical properties
- Cross-talk of electrostatic readout system

The main performance driver is the parasitic electrostatic readout cross-talk. For the delay in the control loop, knowledge of 100 msec is required, which is practically no constraint. The specification values [8] have been used for simulations.

3.2. Method 2: Identification Using the Test Mass as Accelerometer

3.2.1. Basic Concept and Used Coordinates

Basic Concept. The thruster parameters are identified by using the test masses as accelerometers. The signals used for identification are the commanded thrust signals and the output signals of the suspension controllers. The experiment is accomplished in a special accelerometer mode in which all twelve test mass coordinates are suspension controlled (this requires a special control mode of operation with high-bandwidth suspension controllers - no drag-free control is present).

The basic idea of the method is depicted in Figure 8. Notice that in the previous method, the test mass is not moved with respect to the inertial frame (but displaced w.r.t the test mass housing). Here, the test mass is moved in the inertial frame but kept fixed w.r.t. the test mass housing.

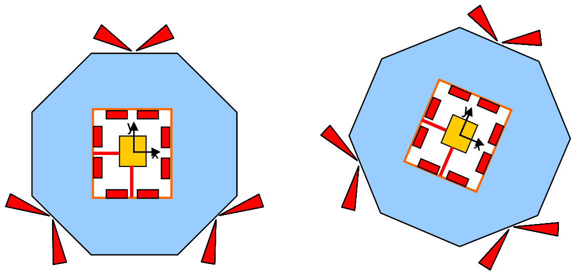

FIG 8. Principle of the thruster calibration approach: Suspension controllers keep test mass centred in the housing.

Via direct thrust commands, test signals apply forces and torques on the satellite. However, with ‘perfect’ suspension controllers the test mass can be kept fixed in the housing frame. Thus, the suspension controllers apply exactly the same forces and torques on the test mass as the thrusters (with negative sign). The outputs of the suspension controllers can be considered as a measure of the forces and torques that are applied to the test mass by means of the thrusters.

Similar as for Method 1, a mathematical model describes the connection between forces and torques on the spacecraft and the corresponding suspension forces and torques on the used test mass coordinates. As before, the desired thruster parameters are part of the actuation matrix “B“ of that mathematical model [5].

Motivation for Alternative Approach. The motivation for such an alternative method is the fact that in the previous approach the thrust levels of some FEEP thrusters is already about 15-20 µN due to the drag-free control (constant disturbances like the solar pressure have to be compensated). When test signals are applied, the average thrust level is even higher. In a dedicated “accelerometer” mode of operation (i.e. w/o drag-free control), the average thrust level is lower. An estimation method using such an accelerometer mode is capable to identify thruster parameters at a lower thrust level. This is of interest since the thrusters are normally operated at such thrust levels.

Used Coordinates. The choice of the used test mass coordinates for thruster identification is not as crucial as case of the previous method since all coordinates are suspension controlled. Again, six independent coordinates have to be chosen. In order to re-use some of the statistical estimation algorithms, the same coordinates as in Method 1 have been used.

3.2.2. Setup of Identification Experiment

Test Signals. Compared with the commanded thrust levels, the method relies on negligible spacecraft disturbances. The main disturbance on the spacecraft is the solar radiation pressure which has a dominating peek between $2{\cdot}10^{-3}$ Hz and $4{\cdot}10^{-3}$ Hz. In order to assure that the assumption on negligible disturbances is valid, this peek has to be avoided and thus the test signals should be at frequencies higher than $6{\cdot}10^{-3}$ Hz. Figure 9 depicts the limitations of the test signal bandwidth.

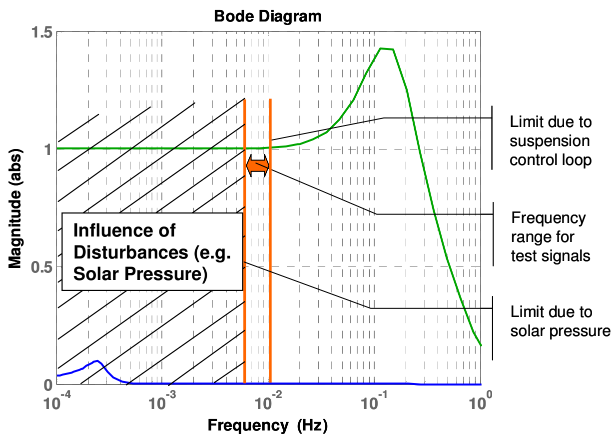

FIG 9. Thruster identification using test mass as accelerometer: Frequency range for test signals.

As illustrated above, the test signal range is limited to a rather small bandwidth between $6 \cdot 10^{-3}$ Hz and $1 \cdot 10^{-2}$ Hz. Notice that the upper limit is imposed by the suspension controller bandwidth. The test signals are sinusoidal thrust commands with amplitude of 15 µN. The thrust commands have to be biased because of the same reasons as reported for Method 1.

Required Data. Six suspension controller output signals (namely the controller outputs for the used coordinates) and the 12 commanded thrust levels (sum of test signal and controller signal) have to be recorded on-board during the experiment. As in for the method using measurements of the test mass displacements, in total 18 signals must be recorded; the maximum sampling rate is 10 Hz.

***Experiment Time*:** Simulations have shown that the experiment time for this method has to be 20000 seconds in order to obtain satisfactory results.

3.2.3. Performance Results and Limitations

Performance Results. Results are obtained with simulated telemetry data using the detailed LISA Pathfinder performance simulator.

Table 2 shows a summary of the performance results after an integration time of 20000 seconds. The data have been taken at 10 Hz. Note that not all 72 parameter estimates are listed, but only the worst case estimation results as well as the average estimation errors.

Parameter	Worst case	Average
k error in %	4.0	1.4
$\vec{u}$ error in deg	1.9	0.9
$\vec{d}$ error in %	9.1	3.3

TAB 2. Performance results for the thruster identification using the test mass as accelerometer.

Remember that the information about the effective lever arm is of minor importance since it does not represent the micro-propulsion characteristics itself.

Limitations. The performance limitation of the presented method depends on:

- Knowledge of spacecraft mass and geometrical properties
- Actuation cross-coupling of the electrostatic suspension system
- Cross-talk of electrostatic readout system

The main performance driver is the parasitic electrostatic actuation cross-talk. For the simulations, the specification values have been used [8].

3.3. Summary and Comparison Between the two Approaches

The advantages and disadvantages of the two methods are summarised in Table 3.

Method 1 (Displacement Measurements)	**Method 2** (Accelerometer Principle)
Advantages: • Short integration time • Science 1 mode is used	**Advantages:** • Lower average thrust level (no DFC (approx. 15 µN) • Estimation of single thrusters
Disadvantage: • High thrust level due to DFC (approx. 25-30 µN)	**Disadvantages:** • New control mode required • Less performance • Longer integration time • Sensitive to external disturbances
Performance limited by: • Knowledge of delay • **Sensor Cross Coupling**	**Performance limited by:** • **Actuation Cross Coupling** • Sensor Cross Coupling

TAB 3. Comparison of the two presented thruster calibration methods.

4. FUTURE ACTIVITIES

Future activities between EADS Astrium GmbH and the Institute of Flight Mechanics and Control at University of Stuttgart are intended in order to enhance the existent thruster calibration algorithms. Among the planned activities are:

Identify Nonlinear Characteristic Curves. So far the thrusters have been assumed to be linear. It might be possible that the thrusters show a certain nonlinear behavior. Especially over the whole thrust range, linearity might not be guaranteed.

Identify Thruster Bias. Nonlinearity could be caused by a constant thruster bias. In the FEEP specification, the bias is limited to 2µN. A nonlinear behavior of the thrust vector, depending on the thrust level could be possible (but does not seem very likely).

Optimise Test Signals. Computation of optimal test signals.

Dynamics of Thrust Control. Estimate the dynamic behaviour of the thrust control actuation (i.e. consider thrusters as dynamic actuator).

5. SUMMARY AND CONCLUSOIN

Two methods for the calibration of the novel µ-Newton propulsion system on-board LISA Pathfinder have been developed and tested by using simulated telemetry data. The achievable performance results for the important parameters (thrust magnitude, thrust direction) is typically around 1 % for Method 1 and in the range of about 1-2 % for Method 2.

For all activities in the field of thruster calibration several contributors are involved. Initial algorithms have been developed by students; further refinements and work in order to assure the compatibility with the DFACS drag-free system have been done by PhD candidates and industry engineers. Eventually, flight data analysis will be performed in a joint effort by university and industry. The results are relevant to the European Space Agency (ESA) for further applications and development in the field of µ-Newton propulsion technology.

6. REFERENCES

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