COB686 SIMULATION AND TESTING OF A SATELLITE SOLAR ARRAY GENERATOR WING DEPLOYMENT

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This work studies the solar panels deployment of the Chinese-Brazilian satellite CBERS. The problem solution demands some deployment tests and the development of two different dynamic simulation models: one more complex related to the laboratory and another one related to orbit conditions. This work begins with a description of the deployment mechanisms, of the test device and of the deployment dynamic problem. The problem formulation is discussed and the use of a computer package for dynamic analysis of mechanisms is justified. The dynamic simulation models, the test results, the adjustment of the model parameters to fit the simulation to the test results, the laboratory and the in-orbit simulation results are presented. The importance of the inclusion in the dynamic models of the solar panel synchronizing mechanisms, of the aerodynamic forces of the solar panels due to the laboratory atmosphere, and of the solar panels initial deformations are discussed.

Keywords: *Satellite, solar array deployment, dynamic simulation model, dynamic of multibody systems, model adjustment.*

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Abstract

This work studies the solar panels deployment of the Chinese-Brazilian satellite CBERS. The problem solution demands some deployment tests and the development of two different dynamic simulation models: one more complex related to the laboratory and another one related to orbit conditions. This work begins with a description of the deployment mechanisms, of the test device and of the deployment dynamic problem. The problem formulation is discussed and the use of a computer package for dynamic analysis of mechanisms is justified. The dynamic simulation models, the test results, the adjustment of the model parameters to fit the simulation to the test results, the laboratory and the in-orbit simulation results are presented. The importance of the inclusion in the dynamic models of the solar panel synchronizing mechanisms, of the aerodynamic forces of the solar panels due to the laboratory atmosphere, and of the solar panels initial deformations are discussed.

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1. INTRODUCTION

A satellite to be launched must be assembled in the launcher nose cone which has a geometry designed to minimize the aerodynamic forces. The satellite envelope is determined by the launcher nose cone design. Therefore, in the launching configuration, the satellite must be as compact as possible. Specifications like the power supply, the satellite stabilization, the positioning of sensors far from the satellite main structure and large antennas can be against the envelope constraints requiring structures exceeding the satellite main structure size. This problem is solved by means of foldable structures. These structures are held stowed to the satellite sidewall during launching phase and, after the satellite separation from the launcher, they are moved to a deployed and locked position. The motion from a stowed to a deployed position is provided by a set of deployment mechanisms. In the case of solar arrays, the power to be generated may require a large area to be covered with solar cells. This is supplied by solar panels which can have different designs. In this work we will be concerned with the deployment of a solar array wing with three stiff and flat solar panels. This is the case of the CBERS (China-Brazil Earth Resources Satellite) satellite which has been developed by INPE (Brazilian National Institute for Space Research) and CAST (Chinese Academy of Science and Technology). Figure 1 shows the CBERS' structural model.

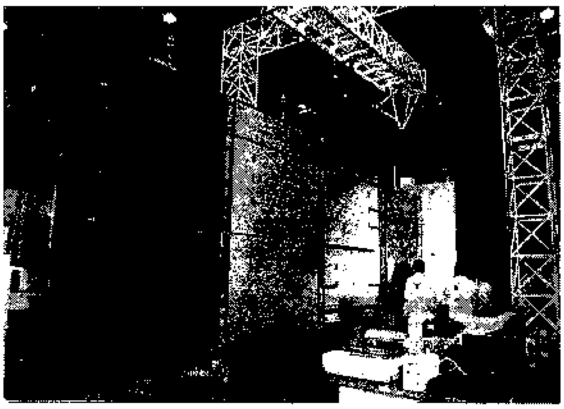

Figure 1: CBERS' structural model.

The CBERS satellite has no active deployment control. At the end of the movement the solar panels latch-up and stops suddenly. This implies in a transfer of kinetic to potential energy which characterizes an impact in the solar panels and in the deployment mechanisms. The dynamic problem consists (1) in guaranteeing that there is enough energy to initiate and conclude the deployment and (2) in determining the angular velocity at panel latch-up. The solution requires some deployment tests and the development of dynamic models to simulate the problem in laboratory and orbital conditions. The laboratory dynamic model must take into account the characteristics of the test device and of the solar panels and deployment mechanisms. The results from the deployment tests are used to adjust the parameters of the laboratory dynamic model. The in-orbit condition can be simulated when the influences of the laboratory are removed from the calibrated laboratory model. Also, the data from the mechanisms friction must be updated to those of orbit environment.

A brief description of the deployment mechanisms and of the test device is done in the next section. The purpose is to give the reader a better understanding on the problem. The description of the problem, based on the simulation parameter, is done in section 3. The problem formulation, including a justification to use a computer package for dynamic analysis of mechanisms and the dynamic simulation models are described in section 4. The test results, the adjustment to fit the simulation to the test results, the laboratory and in-orbit simulation results are present in section 5. Finally the conclusions are in last section.

2. DEPLOYMENT MECHANISMS AND TEST DEVICE DESCRIPTION

The CBERS's solar array is a deployable single wing type configuration with the following characteristics: three solar panels (inner, center and outer) assembled over a flat composite substrate with aluminum honeycomb core and carbon fibber facesheet; one positioning structure (yoke) to support and positioning the solar panels away from the satellite main structure; and a set of deployment mechanisms. The functions of the deployment mechanism are (1) to keep the wing in the stowed position against the side wall of the satellite during the launching phase, (2) to release the solar panels allowing the wing deployment motion in orbit and (3) to fully deploy the wing in a synchronized motion and latch the wing in the deployed position. The deployed wing is about 6.5 meters long and 2.6 meters wide (Vaz and Palerosi, 1992). Figures 2 and 3 show the solar array wing at its stowed and deployed positions, respectively.

The deployment mechanisms are composed by the following parts: (1) Hinge mechanisms.

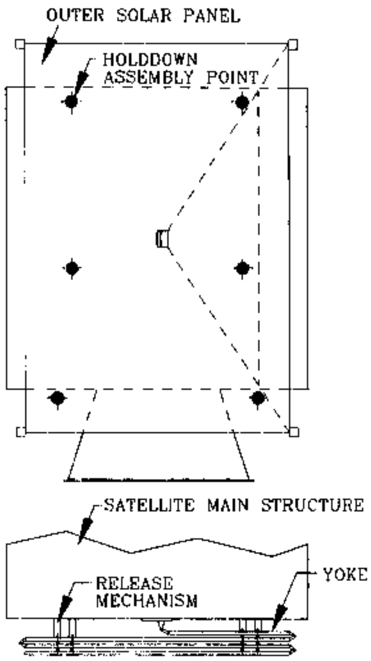

Figure 2: CBERS's solar array at stowed position.

These mechanisms connect one solar panel to another one, the inner panel to the yoke and this to the satellite main structure. At end of deployment a locking lever slides into a slot, latching the wing at full deployed position. They are composed by spherical bearings, each one with one Archimedes spring. These springs provide the energy to deploy the wing; (2) Close cable loop (CCL) mechanisms. They synchronize the deployment angles applying a passive control torque by means of cables and pulleys assembled in each rotation axis; (3) Holddown mechanisms. These are composed by six strengthened pins that hold, by means of solar panel and yoke contact bushings the three solar panels and the yoke in a folded condition against the side wall of the satellite; and (4) Release mechanisms. These cut the holddown pins by means of pyrocutter devices. Figures 2 and 3 show the assembly position in the wing of the deployment mechanisms.

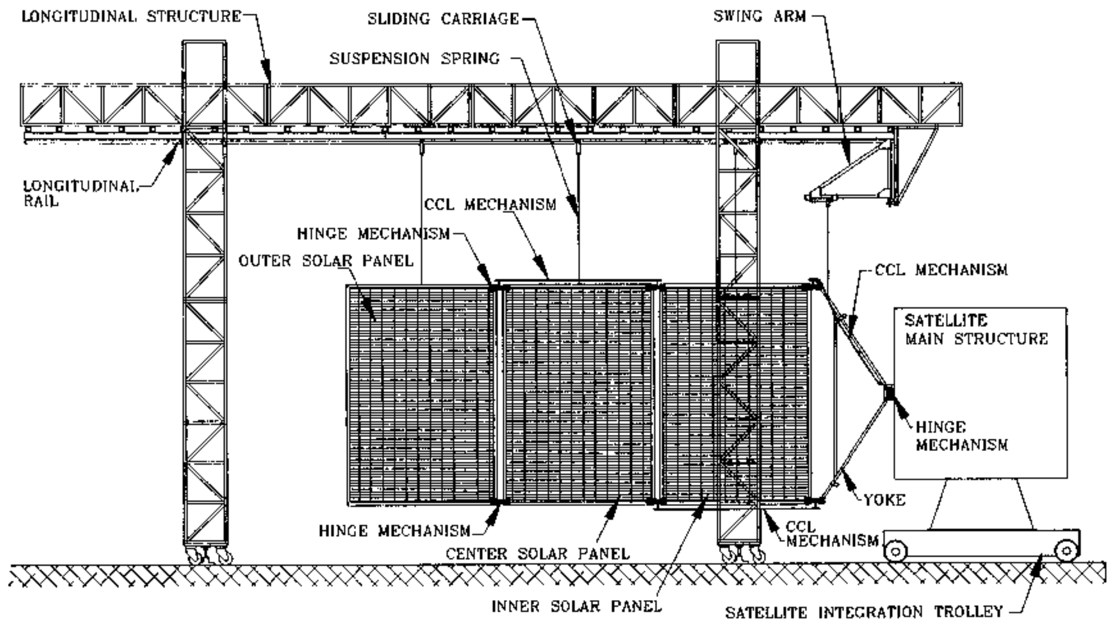

Figure 3: CBERS's solar array at deployed position and test device.

The test device is used to simulate the deployment without the gravity force. This is provided by suspension devices which hang the solar panels and yoke during the deployment. Also, the wing is assembled in this device. The test device is depicted in Figure 3.

The test device has a longitudinal structure which supports a rail. Sliding carriages, one for each solar panels, are assembled in the longitudinal rail. A suspension device is mounted in the sliding carriage holding the solar panels by means of a suspension spring. So, the sliding carriages translate in the rail carrying the solar panels during the deployment. A structure, called swing arm, hangs the yoke. A suspension device similar to that of solar panels is used. The satellite is assembled in a integration trolley as shown in Figure 3.

3. DESCRIPTION OF THE PROBLEM

The problem will be described in terms of the simulation parameters. At end of this section the degrees of freedom and the problem simplifications are presented.

Mass and inertias: The inertias of the problem in orbit condition are: the translational and rotational inertias of the solar panels and hinge mechanisms, and the mass and rotational inertia of the yoke. The translational inertias of the sliding carriages, the mass and rotational inertia of the swing arm are the mass and inertias of the problem in laboratory condition in addition to those of orbit conditions previously listed.

Torques and forces: The energy to deploy the wing comes from the deployment springs assembled in the hinge mechanisms, from the torques applied by the electrical harness which connects one solar panel to another one until the satellite main structure and from the solar panel structural deformations. In the stowed configuration the deployment springs are assembled with a pre-load and at end of deployment the springs have a residual torque due to reliability requirements. The electrical transfer harness apply a motor torque in the beginning of the deployment and, at end, it becomes a resistant torque. This is due to the viscoelastic behaviour of the electrical harness. The solar panels present geometric imperfections due to the manufacturing of the substrate and to the solar cells bonding. As the wing is set in the stowed configuration, the solar panels deform elastically storing potential energy which provide energy in the beginning of the deployment.

Friction torques and forces: These are from the friction torques in the hinge mechanisms bearings (laboratory and orbital condition) and from the friction forces in the sliding carriages (laboratory condition). The friction forces in the sliding carriages are obtained by means of friction tests performed with the rail inclined. The bearings are coated with solid M_oS_2 (molybdenum disulphide) The forces acting on the bearings depend mainly on the wing assembly and they may vary during deployment. So, it is not possible to know in advance the values of these friction torques. This is the parameter to be adjusted to fit the simulation to the test results. It is necessary to perform friction tests in the mechanisms at laboratory and orbital environment to estimate the friction torque at orbital condition. These results are then compared and a mathematical expression relating them can be obtained.

Suspension springs: The forces acting on the hinge bearings change during deployment. The suspension springs avoid an excessive increase of these forces by storing and releasing energy.

Rail levelling and satellite position: The rail levelling and the inclination of the satellite related to the rails must be as small as possible to minimize the effects of the gravity force. These parameters must be measured and included in the laboratory dynamic model.

Iteration of the solar panels with the laboratory air: This is a complex problem due to the unfolding movement of the solar panels. There is turbulent flow around the panels. This makes the determination of the aerodynamic forces very difficult.

Degrees of freedom (DOF) and problem simplification: There are four deployment angles: α_1, between the yoke and the satellite main structure; α_2, between the yoke and the inner panel; α_3, between the inner and the center panels and α_4, between the center and the outer panels. These angles are synchronized by the CCL mechanisms. The angle differences during deployment are due to the flexibility of the CCL cables. So, there are four DOF related to the CCL mechanisms. The laboratory environment adds complexity to the problem. During deployment, each sliding carriage has a different angular position with respect to its solar panel. This adds three more DOF to the problem (β_2, β_3 and β_4). Also, the yoke and the swing arm have different angular position during the deployment, which adds one more DOF (β_1).

The following simplifications are done to define and simplify the problem: (1) the mass of the hinge mechanisms can be considered to be distributed among the solar and yoke; (2) the solar panels and yoke are rigid elements; (3) the angular differences from the flexibility of the CCL cable are very small and do not have influence in the simulation results; (4) the forces due to the panels deformation are assumed to act on the outer panel and (5) the influence of the aerodynamic forces on the outer panel is predominant in relation to the aerodynamics effects on the other panels. The simplification number 3 reduces the four deployment angles to only one, α. Therefore, the problem in laboratory condition has five DOF, β_1, β_2, β_3, β_4 and α. The problem in orbital condition has only one DOF, α.

4. PROBLEM FORMULATION AND MODEL DESCRIPTION

It is possible to find a set of independent generalized coordinates which describes the system configuration. The number of generalized coordinates is equal to the number of degrees of freedom (five in laboratory and one in in-orbit condition). No additional equations are necessary to solve the problem (Meirovitch, 1970). However, the analytical formulation using the Lagrange equations for this problem, which has several elements, constraints and applied forces is cumbersome, difficult to visualize, time consuming and subject to errors. This is mainly due to the time differentiation of those equations which depends on the generalized velocities. Programs of symbolic calculations like Mathematica® are useful, but the handling of equations remains difficult. Moreover, if any change is necessary in the mechanical configuration, significant part of the work must be redone.

A computer package for dynamic analysis of multibody system improves the design activities. The user can be concentrated in modelling the problem, analysing the results and several configurations can be rapidly tested. This kind of computer tool provides, in a graphic-iterative environment, the time-history solution for all displacements, velocities, accelerations and reaction forces in a mechanical system that is driven by a set of applied excitations. The program ADAMS® was used to solve the dynamic problem of CBERS's solar array.

In this program the mechanical system modelling is based on the mathematical description of the behaviour of each element (parts, constraints and forces) and Cartesian coordinates are used instead of generalized coordinates. The software adds constraint equations to the problem set of equations. The resulting equations of motion are greater in number but are simpler compared to the approach of using generalized coordinates. The efficiency of the software is enhanced by taking advantages of sparse matrices numerical techniques. The resulting formulation is a set of equations including the state variables and the mathematical equations relating these variables. The formulation used in this program is described by Wielenga (1994).

A dynamic simulation model for laboratory and another one for orbital environment were prepared. The in-orbit is a simplification of the laboratory model. The model construction in ADAMS® was performed using existing standard types of parts, constraints and applied forces. Simplified geometries were used to model the system components. The actual inertia properties of each component was either measured or computed and imposed on the dynamic model.

The wing synchronization was modelled using an artifice. During deployment the solar panels rotate and translate. A point located at the panel top (synchronization point), follows a straight path along the deployment axis. The synchronization was modelled by an additional massless block assembled on the solar panel synchronization point. This block allows the solar panel rotation and drives the panel according to the defined path. The rail levelling and the satellite inclination related to the rail were modelled by changing the direction of the gravity force. The vertical component was decreased and a horizontal component was added to the model.

5. TEST AND SIMULATION RESULTS

Seven deployment tests were performed. Angular transducers were assembled in each rotating axis. The software Labview® was used to release the wing from the stowed position and to acquire the angular position during deployment. A data acquisition calibration was previously done with the angular transducers assembled in a precision table device. A numerical operator, representing the calibration curves, was included into Labview®. Figure 4 shows the results from the test number 1.

The angular differences among the wing elements (yoke and solar panels) are due to the compliance of the synchronizing mechanism cable and the force variations on this system. The center panel shows a more uniform behaviour and is used as the reference axis.

Seven deployment tests were performed to characterize the system repeatability. For each test there was a different set up for the wing in the stowed configuration. So, the forces due to the solar panels deformations differs from one test to another. This causes a small difference among the resulting curves of angular position as a function of time. Test number seven, which has an average behaviour, was used as test reference to adjust the simulation parameters to fit the simulation to the test results. Figure 5 shows the test result and simulations using different values for the friction torques.

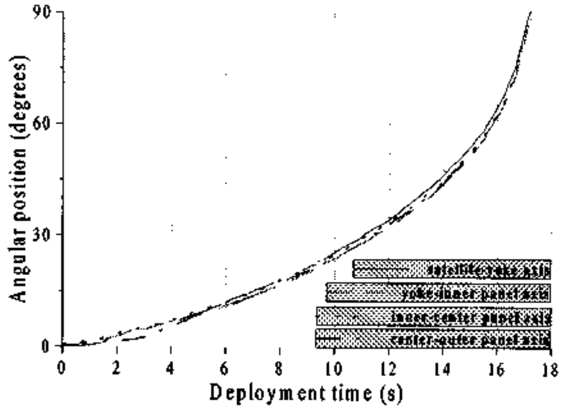

Figure 4: Angular position as a function of time for the deployment test 1.

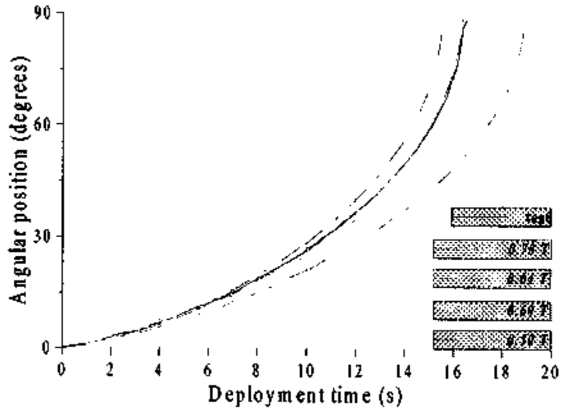

Figure 5: Test results and simulation with different values of friction torques.

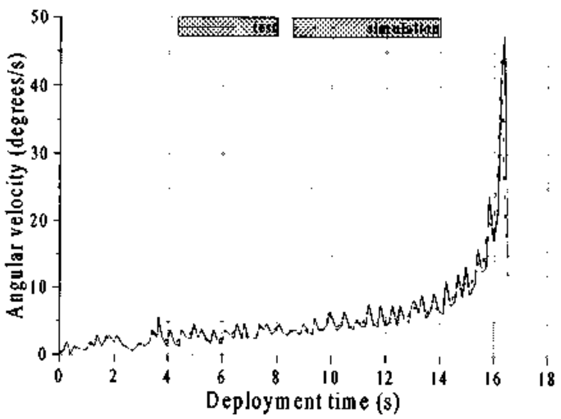

Figure 6: Comparison between test and simulation angular velocity.

In Figure 5 the total motor torque are multiplied by constants which represent the friction torques in the hinge bearings at laboratory environment. The better approximation is obtained using the torque *0.64 T*. This implies that the friction torque represents *36%* of the total available torque calculated. The simulation and test angular velocities are compared in Figure 6. The test angular velocity was calculated using the central difference differentiation method.

At end of deployment the angular velocity presents a sudden change and there is a discrepancy between the simulation and test results. Better agreement would be obtained if a higher data acquisition frequency was used. Also, it can be seen that the test curve presents *saw tooth* oscillations about the simulation curve during deployment. These small oscillations in the angular displacements can be attributed to the compliance of the CCL mechanism cables, which were not included in the simulation model, and measurement errors.

There was no available friction test results from the hinge bearing in the laboratory and orbital environment at the time this work was prepared. So, it was decided to simulate the in-orbit deployment according to two limit situations: (1) in-orbit friction torque equal to the laboratory and (2) in-orbit friction torque equal to zero. Under vaccum condition, the dry film of M_oS_2 applied to the hinge bearings has a better performance than laboratory condition (Roberts, 1989). So, in the first situation is possible to predict, in a conservative way, if the torque margin is large enough to guarantee the deployment. In situation two, the angular velocity is greater than the actual one and the latch-up analysis can be done based on this conservative value. Figures 7 and 8 show the results for both situations for the in-orbit condition.

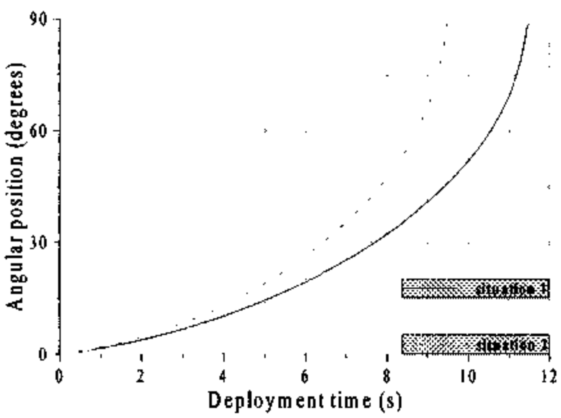

Figure 7: Angular position as function of time for the in-orbit simulation.

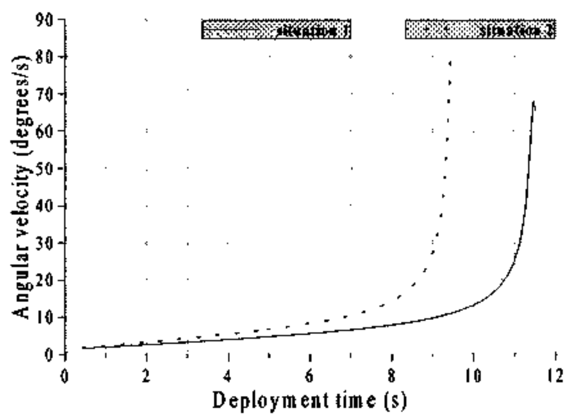

Figure 8: Angular velocity as function of time for the in-orbit simulation.

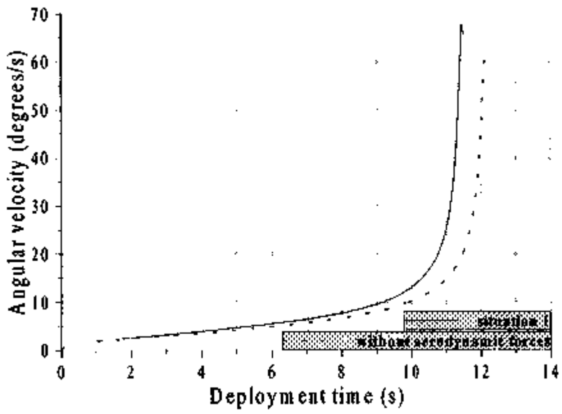

Figure 9: Comparison between the in-orbit simulated angular velocity for the friction torque adjustment with and without the aerodynamic forces

If the aerodynamic forces are not taken into account in the laboratory model, the adjustment of the friction torque will be biased. This error affects the in-orbit simulation results. A new adjustment of the friction torque to fit the simulation to the test results without the aerodynamic force was performed. A comparison between the in-orbit simulation for the friction torque adjustment with and without the aerodynamic forces is presented in Figure 9. The friction torque situation 1 is used in this comparision. The figure shows that the error is non-conservative.

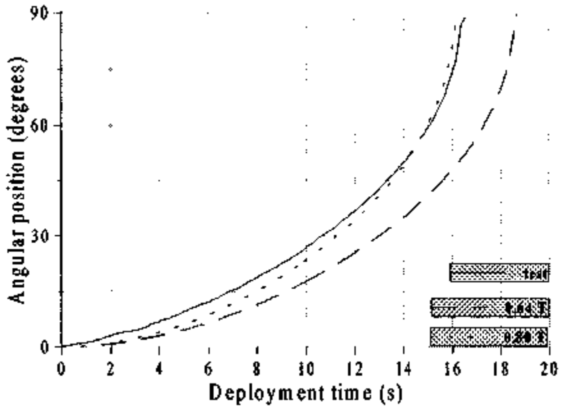

Figure 10: Tentatives to adjust the simulation to the test results without considering the solar panels deformations.

The importance of modelling the forces due to the solar panels deformations is indicated in Figure 10. Tentatives to adjust the simulation to the test results are presented. It can be seen that adjustment is not possible. The best approximation is obtained with the curve *0.80 T*, which is slow in the beginning of the deployment because the energy from the solar panels deformations is not included in the analysis.

6. CONCLUSIONS

Excellent agreement was obtained between the adjusted simulation model and the test results. A higher data acquisition frequency is recommended during the deployment test of the satellite flight model. This could show the effect of the compliance of the CCL mechanism cables. The small differences among the deployment angles do not have influence in the model adjustment. This is a very important simplification because the deployment angles are reduced from four to only one. The specification of the CCL mechanism cables and its initial tension can be done based on an analysis of the test data.

If the aerodynamic forces are not included in the simulation model, an error of about *8%* in the in-orbit angular velocity at panel latch-up occurs. The angular velocity obtained in this case is smaller than the one obtained when aerodynamic forces are included in the model adjustment procedure. This represents an error against safety conditions.

It is not possible to adjust the simulation model if the solar panel initial deformations are not included in the model. Measurements of these deformations and corresponding forces during the set-up of the solar panels in the stowed position must be carefully done.

The problem must be analyzed under a general view. A structural transient analysis must be performed and stress safety margins due to the solar panels latch-up must be calculated. Once these models are available, the influence of each parameter in the in-orbit angular velocity at latch-up can be better analyzed using the dynamic simulation models developed. Moreover, it will be possible to determine tolerances to each model parameters.

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