Abstract
The SPHERES Test Session 13 occurred on 2008-September-27, operated by astronaut Greg Chamitoff. This session has minimal operational difficulties (outside of the SPHERES team control), which were quickly resolved. This resulted in a large number of tests conducted, with several important results. The session provided important data for autonomous assembly of space structures by showing the ability of two satellites to jointly fire thrusters and perform precision maneuvers (previously only coarse maneuvers had been achieved). A JPL led test to demonstrate better use of fuel was successful; the data provided is enough to complete the first stage of this “control allocation” research thread, allowing the tests to move to the next phases of control in future sessions. New advanced controllers (H-infinity and Cyclic Pursuit) were tested during this session with partial results. More tests will be performed for both. Lastly, collision avoidance maneuvers were successfully demonstrated. Multiple formation reconfigurations of three satellites were performed; in each case the satellites “straight” path from one location to another would normally result in a collision. The implemented collision avoidance algorithms successfully modified the trajectories, in real time and autonomously, to prevent collisions during the different reconfiguration maneuvers.

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1 Test Session Objectives

The SPHERES 13th session builds upon objectives from the 12th session. The first two groups of tests run were those that were originally planned for Test Session 12, but that were unable to be run due to time constraints. Overall, the tests include docking, formation flight, and reconfiguration tests. This session adds a wide range of control algorithms for maneuvers previously demonstrated using basic control laws. Modern robust control techniques are combined with path planning and formation flight algorithms to improve the performance of the system. The session also continues to obtain data for control reconfiguration after satellites dock (and their mass properties change).

The test session 13 plan contained the following groups. A mix of one, two, and three satellite tests were included. The objectives are listed for each group. Group E was not run due to lack of time.

a. 12c (MIT): Docking & Reconfiguration
   i. Demonstrate control performance of maneuvering with two active satellites docked
b. 12b (JPL): Control Allocation
   i. Testing of JPL’s “optimal” attitude thrust allocation algorithm
c. 13C (3Sat): Precision & Fractionated Formations
d. 13D (1 Sat): Reconfiguration & Fuel Slosh
   i. Demonstrate maneuverability with large inactive proof mass
   ii. Testing of an H_{\infty} controller to measure performance difference between PID control
e. 13E (2 Sat): Formation Flight & Docking

2 Timeline Summary

This session ran into very few glitches; there were no anomalies or any major obstacles. While several tests still had to be run a few times before a success, 90% of the tests resulted in successful data and all tests obtained at least partial data sets. The chronological list of tests run during Test Session 13 is presented in Table 1. The session occurred on September 27, 2008, starting at 10:30 ET. The average time per test was approximately 9 minutes.

<table>
<thead>
<tr>
<th>Program</th>
<th>Test Description</th>
<th>Start time</th>
<th>Interval</th>
</tr>
</thead>
<tbody>
<tr>
<td>12c: Docking &amp; Reconfiguration</td>
<td>T1 Quick Checkout</td>
<td>17:09:30</td>
<td>03:45</td>
</tr>
<tr>
<td></td>
<td>T1 Quick Checkout</td>
<td>17:13:15</td>
<td>04:32</td>
</tr>
<tr>
<td></td>
<td>T1 Quick Checkout</td>
<td>17:17:47</td>
<td>03:24</td>
</tr>
<tr>
<td></td>
<td>T9 Reconfig: Joint firing high</td>
<td>17:21:11</td>
<td>12:00</td>
</tr>
<tr>
<td></td>
<td>T9 Reconfig: Joint firing high</td>
<td>17:33:11</td>
<td>05:08</td>
</tr>
<tr>
<td></td>
<td>T9 Reconfig: Joint firing high</td>
<td>17:38:19</td>
<td>12:20</td>
</tr>
<tr>
<td>12b: JPL Control Allocation</td>
<td>T1 Attitude Control Allocation</td>
<td>17:50:39</td>
<td>02:35</td>
</tr>
<tr>
<td></td>
<td>T1 Attitude Control Allocation</td>
<td>17:53:14</td>
<td>10:08</td>
</tr>
<tr>
<td></td>
<td>T1 Attitude Control Allocation</td>
<td>18:03:22</td>
<td>29:22</td>
</tr>
<tr>
<td>13d: Reconfiguration &amp; Fuel Slosh</td>
<td>T1 Quick Checkout</td>
<td>18:32:44</td>
<td>01:33</td>
</tr>
<tr>
<td></td>
<td>T2 Fuel Slosh: Sat only</td>
<td>18:34:17</td>
<td>11:27</td>
</tr>
<tr>
<td></td>
<td>T3 Reconfig Maneuvers: Sat only</td>
<td>18:45:44</td>
<td>03:13</td>
</tr>
<tr>
<td></td>
<td>T3 Reconfig Maneuvers: Sat only</td>
<td>18:48:57</td>
<td>08:01</td>
</tr>
<tr>
<td></td>
<td>T5 Fuel Slosh: Batt Proof Mass</td>
<td>18:56:58</td>
<td>07:06</td>
</tr>
</tbody>
</table>
3 Operations

The only operational issue that arose during TS13 was the inability to start the SPHERES GUI at the start of the session. The problem was traced back to the ISS OpsLAN. After the OC group performed a few steps on the LAN, the T:\SPHERES directory was available and the sessions continued smoothly.

3.1 Consumables Consumed

During Test Session 13 the following consumables were used:

- Batteries:
  - Red Satellite: PSI02014J (new, depleted), PSI02018J (new, depleted), and PSI02083J (new, saved as “used”)
  - Blue Satellite: PSI02094J (new, saved as “used”), PSI02009 (new, saved as “used”)
  - Orange Satellite: PSI02037J (new, saved as “used”), PSI02008J (new, saved as “used”)

- Tanks inserted:
  - Red Satellite: PSI01094J (removed and saved for return PSI01027J)

4 Results Analysis

4.1 Program 216 “12c (MIT): Docking & Reconfiguration”

4.1.1 P216 Test 9: Reconfig - Joint firing high

This test was run 3 times. During the first run, the Red satellite ran out of propellant. The two satellites are attached together for this test, and both are necessary to fire in order to obtain desired movement. The test ran until the second target (4th maneuver), before the crew manually stopped the test (test result 103). The maneuvering for the first target is the expected motion, although it is not the expected performance.

The first maneuver is estimator initialization, during which time there is no actuation. The second maneuver nulls the attitude rates. The third maneuver is the maneuvering to the first target (\( (x,y,z) = [0.4,0,0] \) m). Figure 1 shows the expected behavior for the target 1 maneuvering. First, there should be a large spike in the error indicating the target has been set (occurred at \( t = 18s \)). Then, this large spike should reduce to zero and hold at zero. The speed of reduction and amplitude of oscillation determine the control performance achieved. The goal is to achieve error control to within 5cm. Run 3 was a completely successful test. It completed all required maneuvers and terminated nominally (test result 101). The control performance looks quite good. The target locations were:
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- Target 1: \((x,y,z) = [0.4, 0, 0] \text{ m}, \quad (q_1,q_2,q_3,q_4) = [0,0,0,1]\)
- Target 2: \((x,y,z) = [-0.4,-0.4,-0.4] \text{ m}, \quad (q_1,q_2,q_3,q_4) = [0,0,1,0]\)
- Target 3: \((x,y,z) = [0.4,0.4, 0] \text{ m}, \quad (q_1,q_2,q_3,q_4) = [0,0,0,1]\)

Figure 1 shows the position and attitude of the satellite over time. As seen in the plots, the target positions and attitudes were achieved. The attitude oscillation from 1 to -1 during the second target is acceptable. This is numerical artifact, since the position can be represented by both 1 (+180°) and -1 (-180°) rotations. This does not represent a physical attitude change.

![Figure 1](image1.png)

**Figure 1.** P216 T9 Run3 – Reconfig Joint Firing High – Position and Attitude

To better assess the performance, Figure 2 shows the position and attitude error versus time.

![Figure 2](image2.png)

**Figure 2.** P216 T9 Run3 – Reconfig Joint Firing High – Position and Attitude Error

The attitude performance is very good, since Q4 of the error stays close to 1 (meaning zero total attitude error). The rise and settling times are also quite small, indicating good performance. The position performance is acceptable, and close to the desired performance. The desired control performance was to control the error to +/- 5 cm. Figure 3 shows a zoomed-in plot of the position hold regions. In some cases, such as in the Z direction in Figure 3b, less than +/- 1 cm is achieved.
A view of the thruster firing times depicted in Figure 4, shows that there was little saturation, as compared to previous test sessions. The sharp edges of the peaks, versus a flat rectangular shape, shows that there was no saturation. The maximum duration possible was 800 ms.

These results very closely match the simulation results. The simulation was run for the same initial starting point as in P216 T9 Run3. As seen in Figure 5, the performance is slightly better in the simulation. This is because the thruster integration is enabled. In the simulation, the estimator accounts for the updated mass of two satellites connected, while the hardware test does not.
The objective of the SPHERES ISS test sessions is to validate a simultaneous six degree-of-freedom thrust allocator. Because of the current limited memory of a SPHERE, the allocator is to be validated during three separate tests, this being the first of the three tests. Here, only the thrust allocator’s ability to command torques is examined. To this end, eight maneuvers were performed: three rotations about the x-, y- and z-body axes at 15% thrust capacity, followed by three x-, y- and z-rotations at 85% thrust capacity, and finally, two maneuvers about an arbitrary axis at 15% and 85% thrust capacity. For all maneuvers, the SPHERE is supposed to rotate in a positive (right-handed) direction for approximately 360 degrees about an axis and then rotate in the negative direction back to its initial pose. For this test, the thrust allocator has been configured for ideal 0.13 Newton force thrusters with no misalignments.

This section presents the analysis of the data obtained to meet the different session objectives. For each of the maneuvers (shown in Table 2) a simulation has been performed to act as a basis for comparison. The axes x, y, and z form the body frame of a SPHERE.

### Table 2. Maneuver Timeline for P312 Test 1

<table>
<thead>
<tr>
<th>Maneuver Number</th>
<th>Start Time</th>
<th>Duration</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>12 s</td>
<td>43 s</td>
<td>x-rotation at 15% thrust capacity</td>
</tr>
<tr>
<td>2</td>
<td>56 s</td>
<td>43 s</td>
<td>y-rotation at 15% thrust capacity</td>
</tr>
<tr>
<td>3</td>
<td>100 s</td>
<td>43 s</td>
<td>z-rotation at 15% thrust capacity</td>
</tr>
<tr>
<td>4</td>
<td>144 s</td>
<td>19 s</td>
<td>x-rotation at 85% thrust capacity</td>
</tr>
<tr>
<td>5</td>
<td>164 s</td>
<td>19 s</td>
<td>y-rotation at 85% thrust capacity</td>
</tr>
<tr>
<td>6</td>
<td>184 s</td>
<td>19 s</td>
<td>z-rotation at 85% thrust capacity</td>
</tr>
<tr>
<td>7</td>
<td>204 s</td>
<td>43 s</td>
<td>Off-axis’ rotation at 15% thrust capacity</td>
</tr>
<tr>
<td>8</td>
<td>248 s</td>
<td>19 s</td>
<td>Off-axis’ rotation at 85% thrust capacity</td>
</tr>
</tbody>
</table>

* The unit axis of rotation while not performing simple body axis rotations was \([x \ y \ z]^T = [0.568, -0.821, -0.053]^T\).

Each of the following figures has four subplots. The first three subplots provide the rotational rate about the x-, y-, and z-axes in units of degrees-per-second. The solid line is the expected result based on simulations and the dotted line is the estimated rate produced by the SPHERES-core state estimator. The fourth subplot gives the error of the
rotational rate. The error is defined to the difference between the experiment result and the simulation result. A positive error occurs when the experimental rotational rate exceeds the simulated rate. For clarity, the residual rate from the previous maneuver has been subtracted out. Figure 6 shows the rotational rates and error for the entire test. Each of the following sections looks in detail at each of the maneuvers.

![Figure 6: Plot of rotational rates and error during entire test.](image)

### 4.2.1.1 Maneuver 1: Rotation about x-body axis at 15% thrust capacity

The results of maneuver one can be seen in Figure 7. As seen, the phasing and direction of the rotations agree and minimal off-axis rotation has been produced. However, at the end of the maneuver, the simulated and experimental rotation rates differ by three degrees-per-second. Examining the error plot shows that for positive rotations, the error grows by approximately 0.08 deg/s/s and during the negative rotations at -0.24 deg/s/s. Checking the durations commanded by the thrust allocator provides that during both positive and negative rotations, the active thrusters were to be opened for only 23 milliseconds. Given that, the residual rotational rate is attributed to the error between the ideal 0.13 Newton thrust and the actual force produced by each thrusters.
Figure 7. Plot of rotational rates and error during maneuver one.

4.2.1.2 Maneuver 2: Rotation about y-body axis at 15% thrust capacity

The results of maneuver two can be seen in Figure 8. Here again the simulated result and the experimental result agree in phase and direction. However, instead of an accumulating error as seen in the first maneuver, the error at the end of this maneuver is less than one degree-per-second. For this maneuver, the error between the simulated results and the experimental results grow at a rate of ±0.30 deg/second for positive and negative rotations. Again, the thrust allocates that the active thrusters open for 23 milliseconds. Given that and the symmetry of the rate error, the difference between the simulated result and the experimental result is attributed to the error between the principal moment of inertia about the y-axis used in simulation and its actual hardware value.
4.2.1.3 Maneuver 3: Rotation about z-body axis at 15% thrust capacity

The results of maneuver three can be seen in Figure 9. In this maneuver, the thrust allocator commanded the active thrusters to open for 23 milliseconds for both positive and negative rotations. However, the positive rotations produced an error growth of 0.26 deg/s/s and the negative rotations produced an error growth of -0.37 deg/s/s causing a residual error of -2.2 deg/s. Also, the error about the z-axis has a signature similar to that of maneuver two. Unlike the first two maneuvers, a significant error was observed in an off-axis. Error about the y-axis grew to over a degree-per-second by maneuver’s end. This error is due to the accumulation of residual rates. After the first maneuver there remained a three degree-per-second rotation. As time progresses, this will result in a precession of the body about its major principal axis. Thruster force mismatch and either thruster misalignments or larger than modeled off-axis inertia terms are the most likely causes.
4.2.1.4 Maneuvers 4, 5 and 6: Rotations about the x-, y- and z-body axes at 85% thrust capacity

The results of maneuvers four, five and six can be seen in Figure 10, Figure 11 and Figure 12, respectively. The phenomena observed during the first three maneuvers are repeated. The force produced by thrusters about the x-axis is different than the ideal 0.13 N, the moment of inertia about the y-axis differs from that used in simulation and the z-axis suffers from a combination of the two. In addition to this, a large jump in the angular rate estimate is observed 159 seconds into the test. This jump in the x-axis alone is 21 deg/s in only 0.6 seconds. Given this jump, the moment of inertia about the x-axis of $2.84 \times 10^{-2}$ kg\(\cdot\)m\(^2\), and the thruster lever arm of 5 centimeters, a thruster would have to produce 0.35 N of force to generate the jump. Given that only two 0.13N-thrusters are available to produce torques, this jump is not realizable by the SHERES hardware. Instead it appears to be an issue with the SPHERES-core state estimator and the high rotation rate about the x-axis.
**Figure 10.** Plot of rotational rates and error during maneuver four.

**Figure 11.** Plot of rotational rates and error during maneuver five.
4.2.1.5 Maneuvers 7 and 8: Rotations about an arbitrary axis at 15% and 85% thrust capacity

The results of maneuver seven and eight can be seen in Figure 13 and Figure 14, respectively. For the seventh maneuver, the jumps in the rate estimate make it difficult to determine how well the hardware experiment matched the simulations. However, by maneuver 8 the estimator seems to have recovered. Here we see that the dominant effect that differentiates the hardware motion from that of the simulation is the precession of the SPHERE.
Figure 13. Plot of rotational rates and error during maneuver seven.

Figure 14. Plot of rotational rates and error during maneuver eight.
4.3 Program 221 “13D (1 Sat): Reconfiguration & Fuel Slosh”

4.3.1 P221 Test 2 & 5: Fuel Slosh – Sat only & Batt Proof Mass

The purpose of these tests is to observe nutation frequency and amplitude growth in the non-spin axes of a spinning SPHERE due to fluid slosh. The Test Session 13 tests are baseline tests of the SPHERE both without attachments and with a battery proof mass. Fluid slosh bottles will be attached to the exterior of the SPHERE in future tests.

Both tests began with a short estimator convergence period, followed by a position and attitude hold (to the position and attitude in following estimator convergence).

For Test 2: Fuel Slosh: Sat only, the satellite obtained a steady rotation about the Z-axis via closed loop control. There was no position hold during this period. The satellite then went into a “drift” period about 65 seconds into the test. About a third of the way into this “drift” period (91 sec), a perturbation was programmed in. This was a 30 ms X-axis torque, achieved by open control of individual thrusters. Other than the short perturbation, all control was turned off during the “drift” period. Following the “drift” period, there is a short attitude stopping maneuver (134 sec).

Test 5: Fuel Slosh: Batt Proof Mass varies from Test 2 in two ways: a battery proof mass is attached to the X-(Velcro) face of the SPHERE and the steady rotation is about the Y-axis. The battery proof mass alters the inertia properties of the SPHERE, making the inertia matrix less like that of a geometric sphere. This decreases the nutation period, so that more nutation cycles can be observed for same period of time. The control gains are also changed slightly, to account for the changes in the moments of inertia.

The switch in axis of rotation is due to the changes in the moments of inertia. The minor and major axes of rotation are commonly used for spacecraft to spin about; the intermediate axis is unstable and not used as a spin stabilized axis of rotation. Without the battery proof mass, the Z-axis is the minor axis of rotation. Once the battery proof mass was added, the Z-axis becomes the intermediate moment of inertia axis. Therefore, the Y-axis (the major axis in this configuration) is used for the steady axis of rotation. The Z-axis is used as the axis of rotation for the Test 2 test because at the time of test delivery, it was believed that the Z-axis would be used as the axis of rotation when the fluid slosh bottles were attached, in order for them to be aligned with the CO2 tank inside of SPHERES.

The single attempt of Test 2 performed nominally. There were three attempts of Test 5, all of which the satellite reset during the test and the translatonal drift of the satellite was an issue. This reset was never observed on the ground and could not be reproduced. The first two attempts have incomplete test data and the third attempt data shows the wall touch 115 sec into the test, during the drift period.

The Test 2 case in Figure 15 illustrates the satellite attaining a Z rotation rate within 0.01 rad/s of targeted 0.75 rad/s. Over a cycle of nutation is observable during the drift period, and the increase in nutation amplitude due to the X torque. The processed Inertial Measurement Unit (IMU) data in Figure 16 matches rotation rates and nutation values in the gyroscopic data with that given by the telemetry. The accelerometer data shows the forces for the end of the controlled rotation, the long drift period only broken up by the two spikes for the perturbation, and the forces for the attitude hold maneuver at the end of the test. The perturbation has two spikes due to an error in the code that repeats the torque.

The nutation period for the satellite only case is estimated to be 53 sec +/- 5 sec, based on the Y rotation rate from Figure 16. The nutation period expected was 97 sec, given the inertia matrix of the SPHERE and the Z rotation rate.

The Test 5 cases in Figure 17, Figure 18 and Figure 19 each illustrate the satellite retaining a Y rotation rate within 0.01 rad/s of targeted 0.75 rad/s. The Test 5 cases did not given clear enough data to accurately estimate the nutation period. Test 5 Attempts 1 and 3 have full telemetry data, but the test was reset or disturbed by 115 sec into test (50 sec into drift). Test 5 Attempt 2 has data for the entire drift period, but the reset affected the data recording and the frequency of the data is drastically lower (0.5-2 Hz compared to normal 5 Hz). The figures show telemetry from each attempt.

The nutation period for the satellite plus battery proof mass case is estimated to be at 115 sec +/- 40 sec, based on the X and Z rotation rates in the three figures. This indicates that the nutation period is greater than the expected value of 56 sec, given the inertia matrix of the SPHERE and battery pack combined and the Y rotation rate.

Since the SPHERES have an inertia matrix similar to that of a geometric sphere (principle inertia values are close in value), small differences in the values can change the nutation period dramatically. The differences in the expected
and experimental nutation periods can be accounted for if the principle inertia values are off by as little as 4% each. The amount of fluid and distribution within the CO₂ tanks also would also accounts for some of the differences in expected and actual inertia matrix, and pass those differences on to the expected and experimental nutation periods. The inertia matrices for the SPHERE at the time of the test are being looked into.

The tests showed the SPHERES capable of reaching and maintaining a set rotation rate in a single axis. They also show that nutation period and amplitude can be observed in the non-spin axes of a spinning SPHERE.

Future tests will fix the error with the perturbation so that there is only a single torque. Also, the perturbation will be moved to the beginning of the drift period, instead of in the middle. Though the period of the nutation is not disturbed by the perturbation, the effect on amplitude makes it hard to determine the period when there’s a perturbation in the middle of the data.

The amount of IMU data will be reduced, to allow for longer drift period in the active portion of the test without increasing overall test time. The rotation rate will be also increased. The increase rotation rate and longer drift will allow for more nutation cycles to be observed. The controlled rotation portion of the test will also have a position hold, since much of the translational drift seen originated during that maneuver.

![Figure 15. Test 2 Satellite Only – Position and Rotation](image1)

![Figure 16. Test 2 Satellite Only – Inertial Measurements](image2)
Figure 17. Test 5, Attempt 1: Satellite + Battery Proof Mass – Position and Rotation

Figure 18. Test 5, Attempt 2: Satellite + Battery Proof Mass – Position and Rotation

Figure 19. Test 5, Attempt 3: Satellite + Battery Proof Mass – Position and Rotation
4.3.2 P221 Test 3 and 4: Reconfig Maneuvers - Sat only and Sat Proof Mass

The objective of T3 and T4 was to test a method of maneuvering that would not use thrusters along the X body axis of the satellite. In cases, where two satellites are attached via the Velcro, the thrusters on the X faces of the satellite are completely blocked, and therefore, have little to zero thrust ability. The difference between T3 and T4 is that T4 has a satellite proof mass attached, and the appropriate gains for a 2 satellite configuration. Both T3 and T4 were each run 2 times. The basic maneuvering of the satellite was to point its –Y face towards the position target \((x,y,z) = [0.3,0.3, 0] m\) and move towards the target. If it overshot, it was to point its +Y face and go back to the position target. The satellite will not perform position control unless the attitude is within approximately 5° of the proper orientation.

For T3, Run1 was terminated early by the crew in the 4th out of 5 maneuvers. T3 Run 2 completed all of its maneuvers successfully. However, these maneuvers did not perform as expected. The satellite was unable to point towards the target, instead kept spinning around the X and Z body axis. This is reflected in Figure 20 depicting the attitude error. The attitude control is very important, because without successful pointing at the target, the satellite cannot move towards the position target.

![Figure 20. P221 T3 – Reconfig Maneuvers Sat Only – Attitude Error](image)

The oscillations indicate that the variable that determines which satellite face (-Y or +Y) to point towards the target is not being set correctly. Figure 21 confirms this assumption as the face to point keeps changing, even though the positioning has not changed sufficiently to warrant a change in face. The value of +1 refers to setting the +Y face, while a value of -1 refers to setting the –Y face.

![Figure 21. P221 T3 – Reconfig Maneuvers Sat Only – Side to Point to Target](image)

Since the attitude was not properly set, there was little position control, which caused the satellite to drift off in its initial directions. A sharp direction change in Run1 indicates that it impacted the wall at this time, seen in Figure 22.
Similar results were obtained for T4. T4 Run 1 only executed 4 out of 5 maneuvers, before it was manually terminated by the crew. Run 2 completed all maneuvers successfully and terminated nominally (test result 1). Figure 23 and Figure 24 show the results for Run 2. Figure 23 depicts the attitude error shows the same oscillation as T3, only slower. This is expected since the only difference between T3 and T4 is the satellite proof mass, with consequently higher mass and lower bandwidth. Figure 24 shows the position versus time for T4 Run 2. There is more movement towards the target in this run. Most likely, this is due to the fact that since the oscillations are slower, there is more time when the attitude is sufficiently controlled to enable position control.

The performance of P221 T3 was verified in simulation. Figure 24a is a plot of the attitude error of a simulated run, starting at the actual T3 Run1 starting state. Figure 24b shows the actual data, though it is truncated since the crew manually stopped the test. Similar results were obtained for the satellite plus proof mass case.
4.3.3 P221 Test 7: Hinf position controller

The objective of this test was to compare the Nominal Performances and Robust Performances of the current controllers available for the SPHERES satellites. The controllers tested were PID, Phase-Plane LQR and Hinfinity Control. It was known before the test session but after the code submission that the Phase-Plane LQR had a code glitch. Therefore the real performance of this controller cannot be evaluated. The goal of this test is to demonstrate the stability of the Hinfinity controller and gathering data of its Nominal and Robust Performance. The principal specification in performance is zero overshoot, which is useful for docking maneuvers. At the same time, this controller must show a quick response displayed in the form of a rise time similar to the PID/Phase-Plane LQR. It is known from the theory that the Phase-Plane LQR must have the better dynamical performances such as rise time and settling time because it is an optimal $H_2$ controller while the Hinfinity controller is composed by this optimal $H_2$ controller and an internal state filter.

The test performed a square trajectory. The idea behind this was to test different controllers with step inputs with a magnitude of 80 cm under the same initial conditions. The step input test is considered a very useful tool to compare the dynamical performance of linear systems. The controllers were tested during the X-axis displacement while the Y-axis displacement was run to position the satellite in the new initial position and to test the PID controller and gather data for comparison. The maneuvers are as follows:

- First of all, the satellite performs position and hold at the release point for 10 seconds. After that, it moves from the release position towards the position (0,4,0,4,0) in the ISS test volume frame.

- The satellite moves from its deployment position towards (-0.4, 0.4) using the Hinfinity controller, then moves to (-0.4,-0.4,0) with the PID controller and then to (0.4,-0.4,0) with the Phase-Plane LQR controller and back to (0.4,0.4,0) with the PID controller to return to the initial conditions.

These maneuvers are presented in Figure 25.
Figure 25. P221 Test 7 Test Maneuvers

Two cycles of this square maneuver are performed. The first cycle tests the nominal performance (NP) of the controllers, while the second cycle adds a perturbation to test the robust performance (RP) of the controllers.

This perturbation is represented in the form of a firing in the perpendicular direction, this means in the +Y direction. The magnitude of this firing was considered to be half of the maximal nominal value, which means 0.05 N. Figure 26 shows the results obtained from the first cycle of this experiment. The first test objective was accomplished during the position hold.

There was a glitch in the code, because the time allowed for each of the steps was 30 seconds, which was somewhat optimistic for a 80 cm. displacement with the SPHERES satellites. Therefore this issue has been solved and updated for Test Session 14. The data is not conclusive but shows some guidelines in the performance of this Hinfinity controller compared with the PID and the Phase-Plane LQR.

Figure 26 shows the results of the first cycle (NP). Although the Hinfinity controller fails to meet the primary specification (zero overshoot) it shows interesting performances, it shows the same rise time than the PID and LQR controllers. The Hinfinity controller had to displace from approximately 1 m. instead of the expected 80 cm and it had the smaller overshoot (11%). The LQR overshoot by 14% and the PID did not meet the target on the specified time (30 seconds).
It has to be considered that the Phase-Plane LQR was not working when the range to the target position was less than 5cm. This explains in part the fact that it did not reach the target in the specified time. Future testing will include the corrected version of this controller.

Figure 27 shows the results for the Robust Performance (RP) cycle. Once again, the timeout was 30 seconds and the same behavior was seen. Given the different initial conditions in position and velocity are different the data is inconclusive. Yet, it can be said that the proposed Hinfinity controller does not go into instability in the presence of a perturbation and it is able to reject it. The lateral perturbation happened early in the maneuver, at T=5 seconds. The PID and the Phase-Plane LQR did not reach the target position in the specified time. Other than that, the dynamic performance of this controller is interesting because it was not designed to be highly performing. Future work will consider the input of the present test and improvements in the robustness of this controller taking into account thruster dynamics and reducing the overshoot while improving further the tracking capabilities.
4.4 Program 225 “13C (3 Sat): Precision & Fractionated Formations”

4.4.1 P225 Test 2: Collision Avoidance

This test demonstrated the use of a simple behavior-based collision avoidance maneuver for multiple-satellite formations. The maneuver is based on a steering law that attempts to maximize the distance between the satellites at their Closest Point of Approach (CPA). The avoidance algorithm checks to see if a collision is imminent by projecting its path forward in a straight line and finding the CPA distance for each of the other satellites. If the CPA distance is below a specified threshold and the event is in the future, the satellite activates the avoidance routine. The avoidance steering law commands a velocity change that corresponds to climbing the gradient of the CPA distance with respect to relative velocity. This maneuver has the effect of increasing the CPA distance and decreasing the CPA time.\(^1\)

Two runs of T2 were performed during TS13, the first of which resulted in a reset of the primary satellite. Data from the primary satellite indicates that it reset within a few seconds of the start of the test, but data collected from the other two satellites and video of session show all satellites continuing to move through the first two maneuvers before the secondary satellite also experienced a reset. Video and data show performance of the two functioning satellites comparable to the results discussed next.

The second test completed successfully with all three satellites demonstrating avoidance behaviors. Figure 28 examines a collision avoidance event between the secondary satellite and the primary satellite from 148 seconds to 162 seconds. In this part of the test, the primary satellite maintained position at the center of the test volume and acted as an obstacle while the secondary satellite moved to avoid it. Figure 28a displays the estimated distance at closest approach as calculated by propagating the background telemetry of the primary and secondary satellite. Figure 28b displays the estimated time until the closest point of approach at each time step.

At \(t=150\), the secondary satellite activated the collision avoidance maneuver as shown by a vertical line in the distance plot. As indicated in Figure 28a, this was the first time the CPA distance was below the avoidance trigger distance of 26.5 cm at the beginning of a control cycle. The algorithm continued to function as expected with decreasing CPA times and increasing CPA distances after each avoidance event. It is also interesting to note that near the end of this event, the avoidance maneuver was activated every other control cycle. Here it is possible that the PD controller kept driving the satellite back toward a collision on every other cycle.

During the test, the crew noted several small grazing collisions between the satellites. It is expected that this can be corrected by increasing the minimum CPA distance in the collision detection algorithm. Due to the behavior-based nature of the maneuver, it is difficult to develop a definitive guarantee that collisions will avoided be avoided, but it is possible to use Monte Carlo simulations to develop an estimate of the probability of collision for a given maneuver. Running the avoidance routines inside of the 5Hz global metrology update may also provide a faster response to impending collisions.

\(^1\) Note that in the limit, if the satellite had enough control authority, the CPA time could be moved to the present time, and the current separation would be the CPA distance.
4.4.2 P225 Test 3: Cyclic Pursuit

The primary objective of these tests is to demonstrate a decentralized control algorithm based on a cyclic pursuit. This decentralized algorithm uses only the relative information to one other vehicle in the configuration. The test consists of a series of four maneuvers that intent to demonstrate the capabilities of the algorithm in different scenarios. After achieving initial positioning, two of the satellites start a rotation maneuver in the x-z plane, following each other in a circular pattern, demonstrating the basic capability of the algorithm to achieve a circular formation. In the second maneuver, a change in radius is commanded and the vehicles achieve a similar maneuver but with a larger radius. In the third maneuver, the third vehicle joins the formation and they reconfigure into a three-satellites circular formation, aiming to demonstrate satellite addition and reconfiguration capabilities. As a last maneuver the pattern becomes elliptical to demonstrate the ability of the algorithm to achieve non-circular formations. The test was run unsuccessfully 2 times. The first time the primary satellite reset because of low batteries. After changing the batteries a second attempt was performed, but this time the secondary satellite had run out of propellant, being unable to track the desired state. From the results however, a few aspects were inferred. The satellites achieve their initial position, and satellite 3 maintains its position until maneuver 4 when it joins the formation. The target state attempts to follow a circular trajectory around the partner satellite as expected. Further results are expected when the test is completed on a future test session. These features are shown in Figure 29.
5 Conclusions

Test Session 13 was a successful test session. With minimal operational difficulties, the crew was able to perform tests in four different groups, completing tests in each of the main three research areas: docking, formation flight, and common areas.

- Reconfiguration was demonstrated to have sufficient ability for docking tests. These results overcome the difficulties from previous attempts to perform assembly demonstrations, enabling the team to perform these integrated tests in future sessions.

- The control allocation experiment from JPL demonstrated the ability for a simultaneous six-degree-of-freedom thrust allocation to command torque has been evaluated. To determine how well the allocator performed, simulations results were used as a basis of comparison. All discrepancies between the simulated motion and the hardware experiment were the result of the simulation parameters differing from the true hardware values and the SPHERES-core state estimator. This experiment verifies and validates the simultaneous six-degree-of-freedom thrust allocator ability to accurately command torques. Having achieved this, further experiments will be performed to test force-only commanding and simultaneous force-plus-torque commanding.

- The fuel slosh tests demonstrated that SPHERES has the ability to maneuver with enough precision that future tests to excite fuel slosh in external tanks will be able to cause the disturbances required. Further, the tests show that the SPHERES sensors should be able to detect these disturbances.

- The collision avoidance tests demonstrated a high level of autonomy. The satellites successfully avoided collision with each other without any a-priori knowledge of their paths. All the avoidance calculates were performed in real time based on communication of the states between the satellites.

- The data for Hinf and Cyclic Pursuit tests, the first run for either of these types of controllers with SPHERES, showed promising results. Hinf will need further development; the test was not complete for cyclic pursuit. In both cases future sessions will include further testing.
6 Lessons learned

There were no major operational difficulties. The only lesson learned during this session is:

- It is important to divide the test in as many “atomic” parts as possible, so that, if a long test does not run to completion, the amount of science lost is minimized. For example, during the Hinf tests, it would be best to have had four tests that performed one of the maneuvers at a time (ie, one to do Hinf, one to do PID, and one for LQR). In this way each test is not dependent on the previous one working. Only put together long tests after the individual parts have been proven to work and if and only if the longer test provides substantial science return.

7 Actions

There were no action items resulting from this session.

8 SPHERES Team

The SPHERES team members who played a direct role in the preparation, operation, and data analysis part of Test Session 13 are identified in Table 3. This group is in addition to the support of the SPHERES sponsor at JSC, the DoD Space Test Program.

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<tr>
<th>Principal Investigator</th>
<th>Prof. David W Miller ScD ’88</th>
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<td>MIT Professors</td>
<td>Prof. Jonathan How PhD ’93</td>
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<td>Safe Docking (MPC)</td>
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<td>Gurkirpal Singh</td>
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9 Revision History

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