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# HISTORICAL RETROSPECTIVE ON ORION GNC DESIGN

**Robert S. Gay[*], David Dannemiller[†], Shane Robinson[‡], Greg Holt[§], Chris D'Souza[**], Mark Kane[††], Jeremy Rea[‡‡], John Goodman[§§], Greg Loe[***], Mark Tedesco[†††], Nathan Collins[‡‡‡]**

On November 16, 2022, Artemis I successfully launched and began a nearly 26-day journey returning a human-rated spacecraft to the Moon for the first time in fifty years. The mission was a huge success and once again the world's attention was focused on the Moon. This paper will take a step back in time over the seventeen-plus years of design and development of the Orion Guidance Navigation and Control (GNC) system that carried the spacecraft 1.4 million miles around the Moon and landed safely back on earth off the coast of San Diego California. Key design decisions (good and not so good) and "first-ever" capabilities will be chronicled. This paper will explore such things as the most advanced on-board targeting system ever flown on a spacecraft, never-been-done-before autonomous planetary Optical Navigation, and the first-ever truly skip entry guided to the desired target within a few miles.

## INTRODUCTION

I'm excited to take a step back in time and explore some of the key Orion Guidance Navigation and Control (GNC) design decisions, both good and not-so good, that enabled NASA to send the first human-rate vehicle back to the Moon in fifty years. After eight years working as a contractor in Rendezvous Proximity Operations and Docking (RPOD) for visiting vehicles to the International Space Station (ISS), I transitioned to NASA as a civil servant and began working Orion in November of 2005. I spent the first ten years doing GNC focusing on the Orion navigation system. Shortly after starting on Orion, I became the Entry Descent and Landing (EDL) Navigation Subsystem Manager and ended my tenure with the GNC team as the Orion Navigation Team Lead and System Manager. During that time the team successfully completed the Launch Abort System (LAS) Pad Abort-1 (PA-1) test and the Orion Exploration Flight Test -1 (EFT-1). After Artemis I Critical Design Review (CDR), I moved on as the Orion Program Lead Engineer (PLE)

[*] Orion Spacecraft Chief Engineer, NASA-JSC, 2101 NASA Pkwy, Houston, TX 77058.
[†] Orion Deputy GNC Lead, Lead, NASA-JSC, 2101 NASA Pkwy, Houston, TX 77058.
[‡] Orion Orbit MODE Team Lead, NASA-JSC, 2101 NASA Pkwy, Houston, TX 77058.
[§] Orion Nav MODE Team Lead, NASA-JSC, 2101 NASA Pkwy, Houston, TX 77058.
[**] NESC GNC Tech Fellow, NASA-JSC, 2101 NASA Pkwy, Houston, TX 77058.
[††] Orion Entry MODE Team Lead, NASA-JSC, 2101 NASA Pkwy, Houston, TX 77058.
[‡‡] Guidance Technical Discipline Lead, NASA-JSC, 2101 NASA Pkwy, Houston, TX 77058.
[§§] Engineering Analyst, Odyssey Space Research, 1120 E NASA Pkwy #505, Houston, TX 77058.
[***] Controls Engineer, Lockheed Martin, 2100 Space Park Dr, Houston, TX 77058.
[†††] Orion AAMT MODE Team Lead, NASA-JSC, 2101 NASA Pkwy, Houston, TX 77058.
[‡‡‡] Orion Orbit MODE Team, US Airforce Academy, HQ USAFA/DFAS.

for the Vehicle Integration Office (VIO) and later the Spacecraft and Mission Integration (SMI) PLE covering vehicle Flight Software (FSW), automation, and integrated vehicle performance. During this time, the Orion Program completed the LAS Ascent Abort-2 (AA-2) test and the amazing Artemis I mission. Recently, I was given the privilege to be the Orion Spacecraft Chief Engineer (Deputy Program Chief Engineer), and now I would like to share my incredible journey with a sampling of the significant twists and turns that ended with one of the most sophisticated GNC systems to fly in space.

## IN THE BEGINNING

If you are working GNC, the first thing you need is a simulation to model environments, sensors, effectors (thrusters), and of course the FSW. Most GNC designers are also well versed in modeling and simulation. You always need some sort of simulation to test your design and implementation and Orion GNC was no exception. It all started with a prototype GNC executive architecture that laid the foundation upon which the GNC prototype FSW was built that would later become the actual FSW that flew to the Moon.

### GNC Architecture – "proto code"

For the first three months working Orion I generated the prototype GNC executive architecture. It was all hand-coded C running in the NASA-JSC-developed *trick* simulation environment[1]. The executive code allowed each of the GNC Multi-Organizational Design Environment (MODE) Teams to rapidly develop prototype FSW that could be executed via a data-driven scheme that was highly flexible yet very powerful while being extremely computationally efficient. This executive was data driven with a hierarchy of Phases, Segments, Tasks/Modes, and Sub-modes. Each of these elements allowed the developer to uniquely configure the software through a series of sequences triggered by events based on time or other applicable states. Later the actual FSW executive was modeled after this initial prototype with a similar set of hierarchical data separated into Phases, Segments, Activities, and Modes (PSAM). See Figure 1 and Figure 2 respectively for similarities between the prototype executive and actual FSW design.[2, 3]

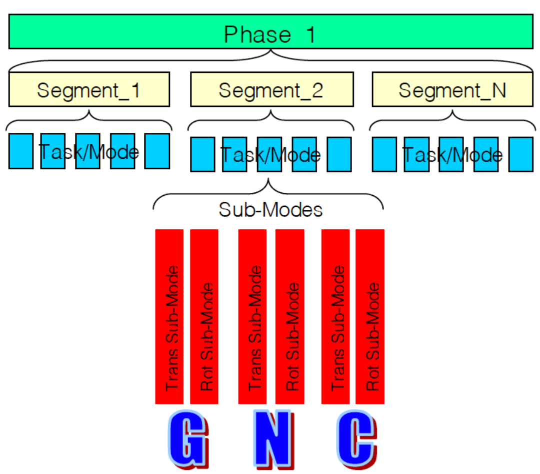


**Figure 1. Prototype GNC FSW Executive Architecture Date-driven Hierarchy**

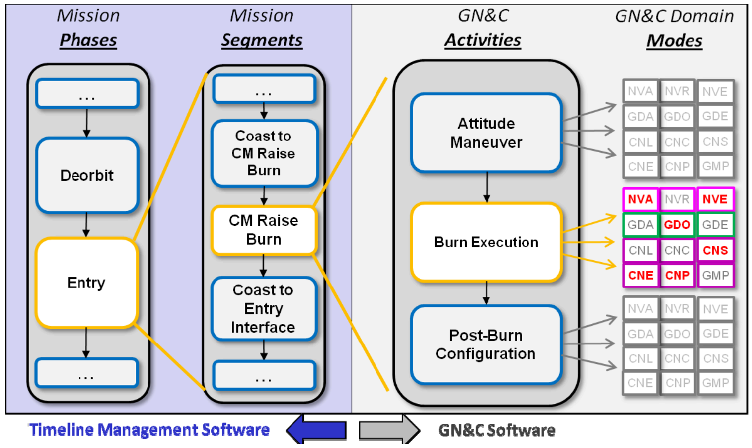


**Figure 2. Actual GNC FSW Executive Architecture Data-driven Hierarchy**

The prototype GNC executive architecture became the backbone of the GNC prototype FSW affectionally known as "proto-code." This prototype simulation was utilized for several years even after the higher-fidelity simulation with the actual FSW came available. The "proto-code" was much faster and produced very similar results. The Entry MODE Team used this version of the FSW and simulation extensively. While it took a couple years to develop all the prototype FSW, it only took 3 months to develop the initial GNC executive architecture with a small upgrade that came along 3 months later. After that, the architecture code was not changed until it was essentially retired. To this day, I look back at that effort as one of the more significant contributions I made to the Orion Program.

### *CHALLENGER & COLUMBIA* – NEVER AGAIN

The morning of January 28, 1986, I was a freshman in high school and home sick that day. I can still remember walking into the den and turning around to see Dan Rather soberly reporting on the Challenger explosion. I was already sold on being an astronaut one day and this was a devastating thing to see. What would it mean for the US Space Program? Why did it happen? Will the Shuttle ever fly again? How terrible for all the families and why this mission with all the additional attention because of the teacher Christa McAuliffe being onboard? Well, most of those questions were eventually answered in the Rogers Commission Report and other studies that followed.[4] Although the Shuttle did return to flight in September of 1988, the Program would later experience a second tragic accident with the loss of the Columbia in February of 2003.[5] Both of these incidents have had profound impacts on the Space industry.

Orion was originally part of a larger enterprise called Constellation and some of the key objectives of this architecture were to mitigate the inherent risks during the highly dynamic and highly dangerous launch and entry phases of flight. The Constellation architecture went back to the Apollo-style Launch Abort System (LAS) that can pull the Crew Module (CM) off an exploding or out of control launch vehicle, and an aerodynamically stable CM capsule with a robust heat shield for entry. While it is impossible to eliminate all risk, these two fundamental aspects of the Orion design greatly improve safety for the crew.

#### Launch Abort System (LAS) Aborts

Orion has a requirement for full abort coverage from T-5 minutes to lift off through main engine cutoff. In order to achieve this, Orion utilizes three abort options (Modes): Mode 1 LAS aborts, Mode 2 Untargeted Atlantic Splashdown (UAS) aborts, and Mode 4 Abort Once Around (AOA) aborts. Mode 3 Retrograde Transatlantic Aborts (RTAL) are not currently needed through

Artemis III and will be addressed further in the Service Module (SM) Aborts section. See Figure 3 for an illustration of the Orion ascent abort modes.

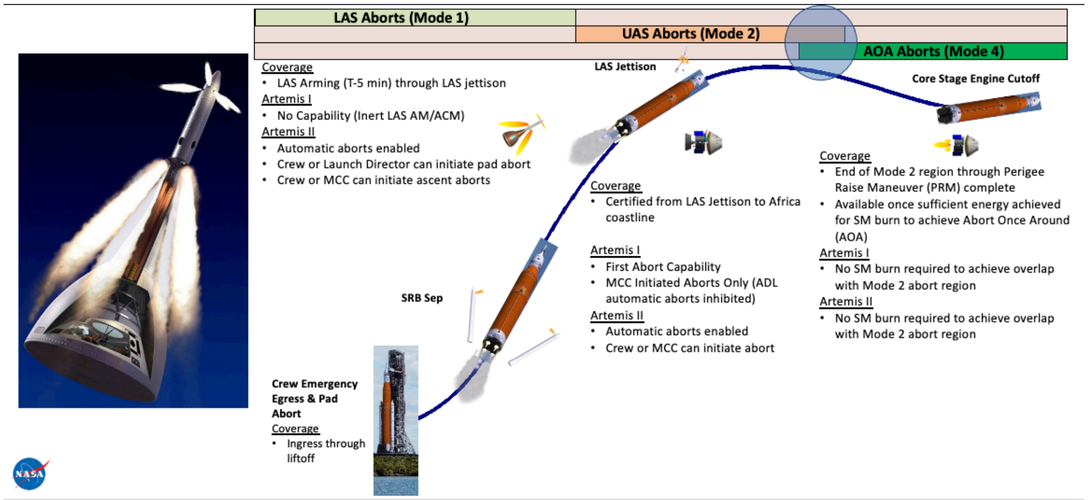


**Figure 3. Ascent Abort Modes**

Although Artemis I did not have a fully functional LAS, a lot of design and testing was completed prior to flight in preparation for the first active LAS that will fly on Artemis II. The LAS is made of three solid rocket motors: the large abort motor that does all the heavy lifting, the attitude control motor (ACM) that uses multiple pintle values to rapidly actuate and direct steering forces near the nose of the rocket, and the jettison motor that fires just enough to pull the LAS safely away from the CM. In May of 2010, Pad Abort-1 (PA-1) test flight successfully launched from the White Sands Missile Range.[6, 7] The pad abort test was designed to demonstrate the capability of the LAS to pull the CM off the launch vehicle while still on the launch pad and get the CM far enough out in the water to achieve a safe landing under parachutes. The trick is fighting the onshore winds that can blow the CM back onshore after chute deploy. Even though some high winds nearly scrubbed the launch, it was a very successful test. Figure. 4 shows the PA-1 abort motor firing as it pulls the CM away from the pad.

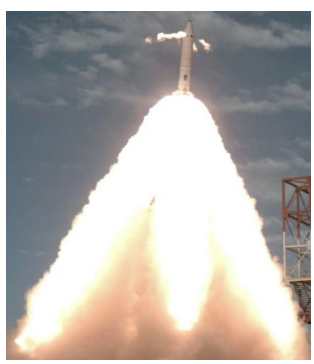

**Figure. 4 PA-1**

Later in July of 2019, another LAS abort test called Ascent Abort – 2 (AA-2) was successfully executed. AA-2 was an in-flight abort test that put the LAS on top of an old Peacekeeper intercontinental ballistic missel and fired it off the coast of Florida from Space Launch Complex 46.[7] The abort was initiated at a point of maximum aerodynamic stress to test LAS capability in this challenging region. Figure 5 shows the LAS atop the Peacekeeper just after liftoff. One interesting decision made to save time and cost was to eliminate the parachute system from this test. PA-1 had already demonstrated this phase of flight, and NASA was already conducting a very thorough stand-alone parachute drop-test campaign. In order to backup the inflight telemetry and ensure that critical data was captured, twelve separate data recorders were ejected from the capsule prior to splashdown. Each data recorder was equipped with a beacon and transmitter to assist in retrieval. The test was very successful and even gave the aerodynamics community great data to better understand the complicated interaction and chemistry of the abort motor plume within the flow field. Orion LAS aerodynamics models were adjusted based on AA-2 data.

**Figure 5. AA-2**

**Service Module Aborts**

Since Orion has a requirement for full abort coverage from launch to main engine cut off, once the LAS is jettisoned, the SM is used to execute the ascent abort.[8] So far only UAS or AOA SM aborts are needed for full coverage. A UAS abort simply separates the Crew and Service Module (CSM) from the stack with springs and a small burn and then the CM separates and employs simplistic guidance to avoid land and manage loads. UAS aborts all land in the Atlantic Ocean. If the abort happens later during ascent and velocity is great enough, Orion can execute an SM burn or no-burn entry that will land the CM in the pacific guided to a target depending on the launch trajectory. Those familiar with the Shuttle may be wondering what about an RTAL abort? At this time analysis shows that this abort option is likely not needed until Artemis IV. There will be a new upper stage for Artemis IV with difference trajectory characteristics that will likely require Orion to add an RTAL abort option. An RTAL requires the vehicle to rotate around and fire the main engine retrograde (opposite to the velocity direction) in order to slow down enough not to land on Africa. The SM UAS and AOA abort capabilities were available and active during the Artemis I mission. Fortunately, neither were needed.

## NAVIGATION – FIRST THINGS FIRST

As a navigator it has always puzzled me why the acronym GNC put guidance before navigation? You must know where you are (Navigation) before you can decide where to go (Guidance) and how to get there (Control) – right? I'll admit I'm a little biased given I'm a "Nav guy." I'm also really blessed to have been a part of history – designing, implementing, and testing the navigation system to fly on the United States' next flagship spacecraft headed back to the moon. The following sections will cover a major navigation upgrade, a big mistake, and ground tracking a human-rated vehicle.

**Navigation Architecture Upgrade – EFT-1 to Artemis I**

I don't have time to go into all the Orion history and cancelation of the Constellation Program, but suffice it to say, we had multiple changes in direction that led to a mad scramble to get Orion ready to fly what came to be known as Exploration Flight Test -1 (EFT-1). It was a relatively simple mission consisting of two orbits with the second one being highly elliptical generating significant entry velocity to test the heat shield. EFT-1 put Orion on top of a ULA Delt-Heavy with an Interim Cryogenic Propulsion Stage (ICPS) as the upper stage. In December 2014, EFT-1 launched and landed in the same day completing a very successful flight test.[9] Figure 6 shows the EFT-1 trajectory.

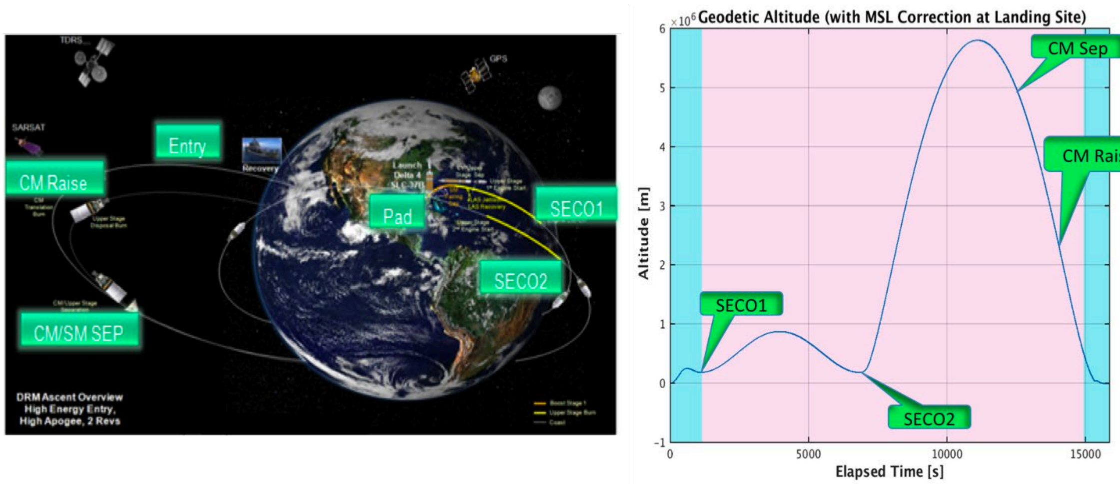


**Figure 6. Orion Exploration Flight Test -1 (EFT-1) Trajectory and Altitude Profile**

As mentioned earlier, we were in quite a rush to get EFT-1 off the ground to avoid any new possibility of being cancelled again. That rush resulted in some, let's just say, "less than optimal" navigation code. Looking forward to Artemis I, it was clear we needed to refactor the code to make it more modular and easier to expand capability as the successive missions would require. We also needed an architecture that would optimally utilize the new navigation hardware coming after EFT-1. Only two Inertial Measurement Units (IMUs), one GPS receiver, and three Barometric Altimeters flew on EFT-1. For Artemis I and future missions, Orion has three IMUs, two GPS receivers, three Barometric Altimeters, and two Star Trackers co-located with an Optical Navigation camera. Notice I didn't say anything about the IMUs being co-located with the star trackers or the IMUs all bring co-located with other IMUs. More to come on that "mistake" in the following section.

In December of 2013, a year before EFT-1 launch, I got a large number of the Orion navigation team members from JSC, Lockheed Martin, and the Flight Operations Directorate (FOD) together in a unique location away from the "usual" work environment to spark creativity and develop a plan for upgrading the Orion navigation architecture for Artemis I and future missions. This was the first of three or more Technical Interchange Meetings (TIMs) that culminated in what I believe was a historical achievement for the US Space Program. Almost a year after that first TIM, the team settled on an architecture that modularized how various Extended Kalman Filters (EKF) were instantiated to support different flight phases keeping standard interfaces and allowing placeholders for future expansion like lunar feature tracking when near the Moon. The architecture is centered around three navigation channels each anchored with an IMU. There is a coupled-attitude EKF for atmospheric flight, independent attitude EKE for on orbit, and separate translational EKFs for near-Earth and Cis-lunar space. In general, when there are two sensors available, each one is on a separate channel, and both are on the third channel. There is also a fourth channel that is inertial propagation only based on the selected IMU.[10] If any of the navigation sensors fail or have off nominal behavior, the complexity of redundancy management is relegated to manual commanding by the ground or crew. Any sensor can be connected to any of the three primary channels and any of the channel states can be transferred to another channel if desired. Ground state updates can update a single channel or multiple channels at the same time. Figure 7 captures the historical moment on November 21st, 2014, when the fundamentals of the Orion navigation architecture were formulated. These simple sketches would later translate into to FSW that would carry Orion to the Moon and safely home to Earth. Figure 8 illustrates the navigation sensor and EKF utilization timeline implemented during the Artemis I mission.

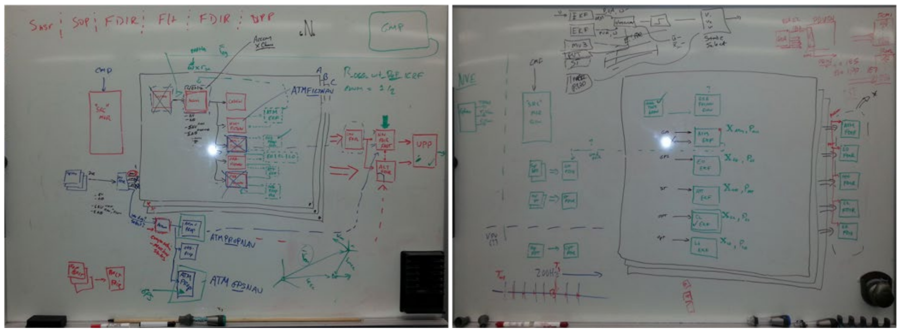

**Figure 7. Orion Navigation Architecture Upgrade – 40Hz & 1Hz Rate Groups**

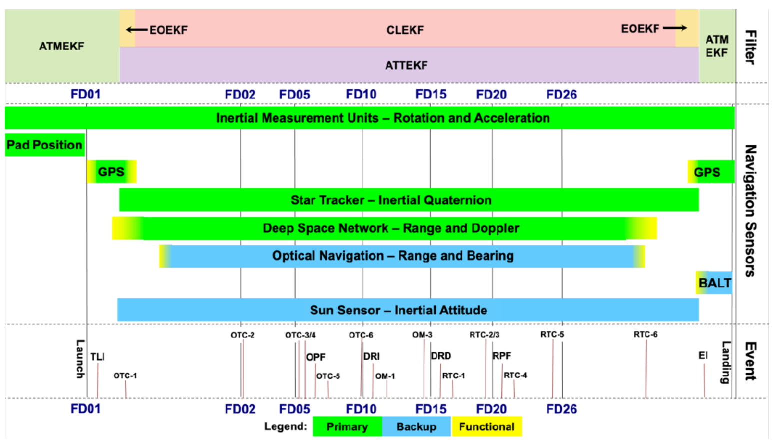


**Figure 8. Artemis I Sensor Utilization Timeline**

## Navigation "Base" – Compromises Hurt

If you know much about navigating a spacecraft, you know the core of most navigation systems is the IMUs and Star Trackers. The IMUs are needed for basic translation and rotation propagation and the Star Trackers provide precise inertial attitude. Of course, ground navigation is also key for missions outside of GPS useful range. We often call the platform for the IMUs and Star Trackers, the "Nav base." It is critical to know the alignment between the IMUs and Star Trackers so you can properly estimate attitude and IMU gyro errors. In addition, Orion has an Optical Navigation (OpNav) Camera whose alignment to the Star Trackers is also very important. More on OpNav a little later in another section of the paper.

The Nav base is often a solid, rigid platform with IMUs and Star Trackers all co-located and relative alignments measured pre-flight to high precision and accuracy. As with many things on Orion, we ended up blazing a new trail thinking things would all work out. During the design phase, it was discovered that if the Star Trackers were to be co-located with the IMUs, the sunshades to protect the optics would have to be made of heavy metal to withstand the heat of entry. This would be a major hit to landed mass that was under constant scrutiny. So, the Program challenged the Navigation Team to make the system work with the Star Trackers mounted on the Crew Module Adapter (CMA) that connects the SM and CM. After studying the problem, we decided it would work so long as we did regular planned attitude maneuvers to estimate the Star Tracker to IMU alignment. It was assumed the structure would be sufficiently stiff to maintain adequate alignment between the preplanned attitude maneuvers. As if this challenge wasn't enough, the configurators told us there was only enough room to put two of the three IMUs in the same general location and the third one would have to be on the opposite side of the spacecraft. If you are a navigator reading this, you probably are sick to your stomach at this point – and you should be.

As it turned out the structure was sufficiently stiff between the attitude maneuvers and the alignment estimation worked very well during Artemis I. However, prior to Artemis I, it was discovered that the change in external pressure as the spacecraft travels from the pad to orbit did in fact deflect the Orion pressure vessel enough to cause problems, especially with the IMU mounted on the other side of the vehicle. Figure 9 illustrates the pressure vessel ballooning at vacuum, and Figure 10 shows the general mounting locations of the IMUs. While the preplanned attitude maneuvers on orbit did accurately estimate the altered IMU to Star Tracker alignments, the IMU-to-IMU Fault Detection and Isolation and Recovery (FDIR) was negatively impacted during ascent and entry when pressure changes occur. It was also found that the aerodynamic loading during entry deflected the structure. These issues forced the Navigation Team to adjust IMU FDIR thresholds during ascent and entry and other computations related to Star Tracker FDIR to compensate for the structural changes. Fortunately, the structure was very stable on orbit even in various thermal conditions and the alignment maneuvers worked very well. The ascent and entry threshold adjustments also performed as expected. Additional adjustments are planned to improve alignment estimation and FDIR on future missions.

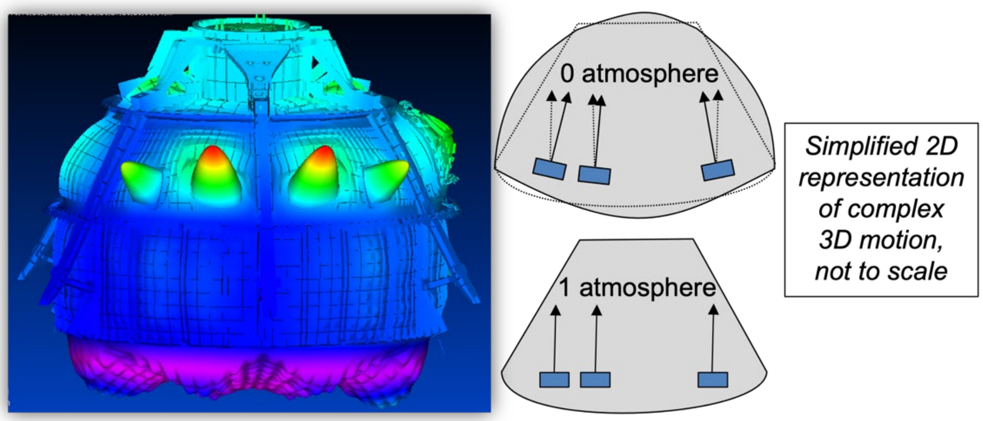


**Figure 9. Pressure Vessel Ballooning at Vacuum**


**Figure 10. General IMU Locations**

**Ground Navigation Tracking Stations – How Many are Needed?**

Back in 2009, the Orion Navigation Team presented data during the GNC Preliminary Design Review (PDR) that demonstrated the need for 9 ground tracking stations to adequately support human-rated missions to the moon. Apollo used closer to 13 stations to guarantee good and timely tracking solutions.[11] While it is possible to use only the 3 Deep Space Network (DSN) sites (Goldstone, Madrid, and Canberra) for a "quiet" vehicle with few trajectory perturbations if all goes as planned; however, if anomalies occur more tracking is desired to aid in resolving the problems. For example, a failed burn during a lunar flyby that is done without comm as the spacecraft goes behind the moon, may require a rapid tracking solution to ensure that a recovery burn could be attempted while it is still small enough to be executed with the remaining propellent.

Fast forward to late 2015, the Orion Navigation Team convinced the Program to acquire the use of 3 additional tracking sites beyond the 3 primary DSN sites to support the Artemis I mission. This was a compromise down from the desired 9 total sites, but given it was an un-manned mission, the risk acceptance was appropriate. These 3 additional sites are receive-only 3-way doppler sites that relay the signal back to the associated primary site. The 3 sites chosen were: Swedish Space Corporation (SSC) Santiago, Chile and Hartebeesthoek, South Africa, and JAXA in Japan. See Figure 11 for the Ground Tracking Stations used for Artemis I. While agreements were made and compatibility testing was completed, only the JAXA site demonstrated actual tracking prior to Artemis I and this led to formatting issues for the SSC sites during the mission. The SSC data was not usable during the mission, but some of the data is now useful after working through the issues. A key lesson learned is to spend the time and money to fully demonstrate tracking functionality prior to flight. These 3 "additional" sites were not mandatory (but highly desired) for Artemis I, thus some reductions in testing were made. Going forward to Artemis II more emphasis on tracking demonstration will be applied.

Although this paper is focused on Artemis I, it is interesting to note that relatively new information on the Artemis II version of Orion is showing that the Environmental Control and Life Support System (ECLSS) venting for the carbon dioxide scrubbers is imparting a significant trajectory perturbation that is likely to require more than the 3 DSN sites just to do the nominal mission. The Navigation Team is in negotiations now on what is needed for the total number of tracking stations and proper pre-flight testing for Artemis II.

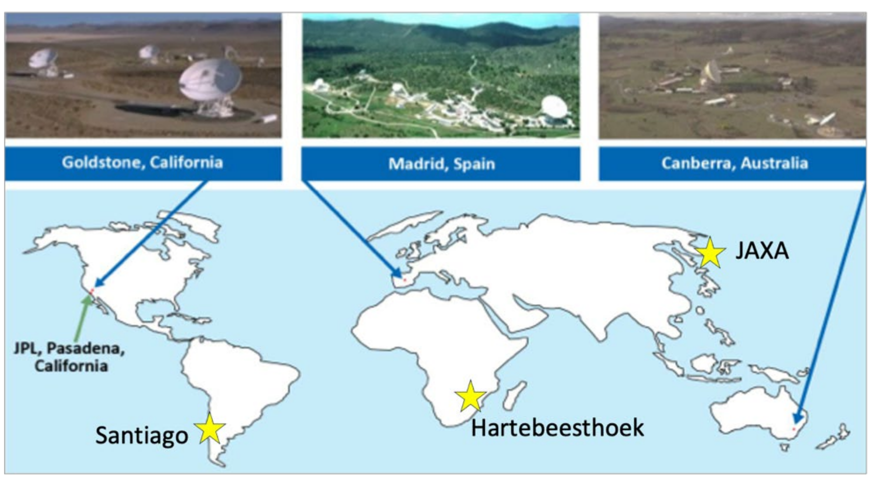


**Figure 11. Artemis I Ground Tracking Stations**

## RETURN WITH PERMANENT COMM LOSS – WHAT?

One of the key driving requirements for Orion is the ability to return the crew safely to earth without communications with the ground. This means no telemetry, no voice, no commanding, and no ground tracking. While this requires multiple failures, NASA wanted this core capability to improve overall safety. This ability may sound not so hard given all the autonomy we have in the world today, but building in true autonomy to a Cislunar human-rated spacecraft is a monumental task. Almost all spacecraft heavily utilize resources on the ground to accomplish their mission. Using the ground allows designers to simplify onboard systems reducing costs and often improving reliability and robustness by having more computing power and keeping the human in the loop. This requirement which eventually became known as Permanent Comm Loss (PCL) drove some very unique capabilities into the Orion design. The following sections will summarize Orion's autonomous Optical Navigation and Onboard Targeting and burn execution functions. Capabilities that have never been flown before.

### Autonomous Optical Navigation – First Time Ever

As mentioned above, Orion must be able to return the crew safely to earth in a loss of comm situation. Since ground tracking is no longer possible and ranges are much too far for standard GPS usage, Orion navigators had to design a system that could function autonomously and well enough to support the burns necessary to get back home. As this capability matured it came to be called Optical Navigation or "OpNav" for short. The OpNav system utilizes a technique that was demonstrated on Apollo and has been done in various forms on other un-manned missions. However, in all cases the computing was done on the ground or there was a human-in-the-loop. Orion OpNav was not only autonomous it was also fully automated.

The basic idea is to calculate the range and bearing to a planetary body (or moon) by using a special camera and image processing to identify the location and size of the object within the field of view. Finding the centroid of the object within the image along with the precise inertial attitude of the camera allows you to compute the bearing to the object. Determining the size of the object within the image along with known radius of the object allows you to determine the range. Figure 12 illustrates the basic geometry of a camera image and how to determine the range to a body of know radius.

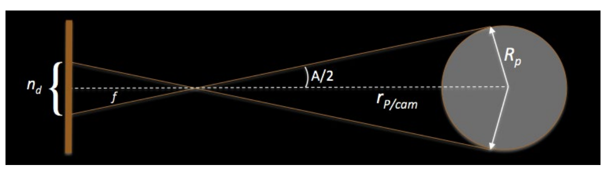


**Figure 12. OpNav Basics - Pinhole Camera Model**

You might be thinking, "What's so hard about that?" Well, there are many details that must be worked out to make this all function automatically and autonomously. Originally, the OpNav design was to be similar to what they did on Apollo where they took angular measurements from multiple stars to the limb of the Moon or Earth. On Apollo, the astronauts manually used a sextant to site in the measurement and the system was configured to ingest that "mark" into the onboard computer upon command. Apollo had a sextant integrated into the navigation system not only for these OpNav measurements, but it was also used for establishing the inertial attitude to align the gyros. In addition, Apollo had less stringent entry-corridor requirements and just getting bearing to the object was good enough to get them home if they loss comm. For Orion the entry-corridor requirement is much tighter (due to optimized heat shield sizing to save mass) so bearing and range are necessary. After going through a camera trade study, it was determined that while the human eye can see a dim star and a bright moon or earth at the same time, a camera could not (at least not one that wasn't Classified for military use). So, instead of multiple star-to-limb measurements, the bearing is computed by finding the centroid of the object in the camera image plane along with the inertial attitude of the camera. For Artemis I, the OpNav camera inertial attitude was computed by using the Star Trackers that are mounted on either side of the camera on an "Optical Bench." This Optical Bench is rigid so the relative alignment of the Star Trackers and OpNav camera remain well known from pre-flight measurements. Unlike the IMUs, the alignment between Star Trackers and the OpNav camera remains sufficiently constant to support the OpNav function.

Another difficult challenge to making OpNav a success was lens calibration. Since the measurements in the camera image must be very precise given the ranges involved, any lens distortion will cause large errors. Therefore, a sophisticated calibration technique had to be developed. OpNav lens calibration involves rotating the vehicle to generate multiple images of various stars in the field of view. The technique uses Star Tracker-like algorithms and a star catalog to estimate the lens distortion coefficients that are then used to correct the lens effects on the OpNav measurements.[12] In addition, prior to Artemis I flight, the Navigation Team developed a method to use the OpNav images processed on the ground as a backup Star Tracker.[13] Later, these same techniques were used to provide a more onboard integrated Star Tracker backup, and OpNav camera inertial attitude is now computed just with OpNav star images before and after range and bearing measurements removing the dependence on Star Trackers.

Finally, one of the biggest efforts for OpNav was flight certification. How much could be done on the ground pre-flight vs. what must be done during the mission to test in-flight conditions? The Navigation Team did an amazing job in doing everything possible prior to flight to prove out the system. Error models were generated based on camara specifications, post-calibration lens distortion residuals, atmospheric and spectral reflection effects on earth measurements, terrain and albedo effects on lunar measurements, camera exposure settings, and camera boresight angle to the Sun. Linear Covariance and Monte Carlo simulations were used to assess performance. Ground testing using camera hardware was conducted to validate the lens calibration technique. Live Sky testing with two OpNav cameras and Star Tracker hardware was also performed. Lastly, integrated hardware-in-the-loop simulation testing was conducted using a high-resolution synthetic image displayed on a mobile phone in front of an OpNav camera with a collimator to make the image appear as if it were at an infinite distance. The geometry and appearance of synthetic images were driven by the Orion simulation using high-fidelity image-rendering graphic software.[14] Figure 13 shows the OpNav camera hardware-in-the-loop integrated testing setup.

Even after all the pre-flight testing, OpNav could not be fully certified until after demonstrated performance on the Artemis I outbound leg to the Moon. Once performance was verified and the events que needed to execute the remainder of the mission autonomously was large enough to handle all the events, then OpNav could be setup for use to bring Orion home after PCL. Until then, Orion would have to execute some form of disposal to protect from any catastrophic consequence on Earth or in Cis-lunar space. Thankfully, all the hard work of the Navigation Team paid off and the OpNav system performed even better than expected and was fully certified to complete the Artemis I mission.[15] Fortunately, we never had a PCL event on Artemis I, but OpNav was ready if we did.

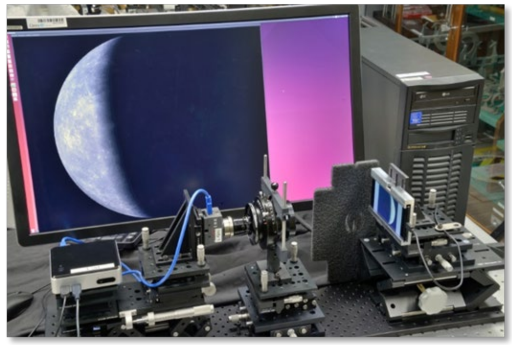

**Figure 13. OpNav Integrated Hardware-in-the-Loop Testing**

### Autonomous Onboard Targeting and Burn Execution – Most Sophisticated Ever

Returning crew safely to Earth without comm drove much more autonomy and automation than just OpNav. In order to execute the necessary burns to bring Orion back home without comm, a sophisticated data-driven architecture was developed. The design and implementation evolved into a Burn Plan Manager (BPM) operating on Burn Plan (BP) data and executing the Two-Level-Targeter (TLT) to generate targets and other directives to orbit guidance (OrbGuid) to in turn generate steering commands for SM control (CNS) and Solar Array Wing (SAW) articulation control (CAR). The BPM also provides backup data in case TLT or OrbBuid does not converge, and data used by the Timeline Vehicle Manager (TVM) which controls when things happen, how the FSW is configured, and other vehicle configuration information not provided by the BPM. Between the Trans-Lunar Injection (TLI) burn and Entry Interface (EI), mission sequencing cycles between three main segments: Coast, Burn Configuration, and Burn. TVM uses Burn Plan data supplied by BPM to determine when and in what order to execute activities during a burn.[16] Figure 14 depicts the BPM interfaces and functional flow for burn targeting and execution.

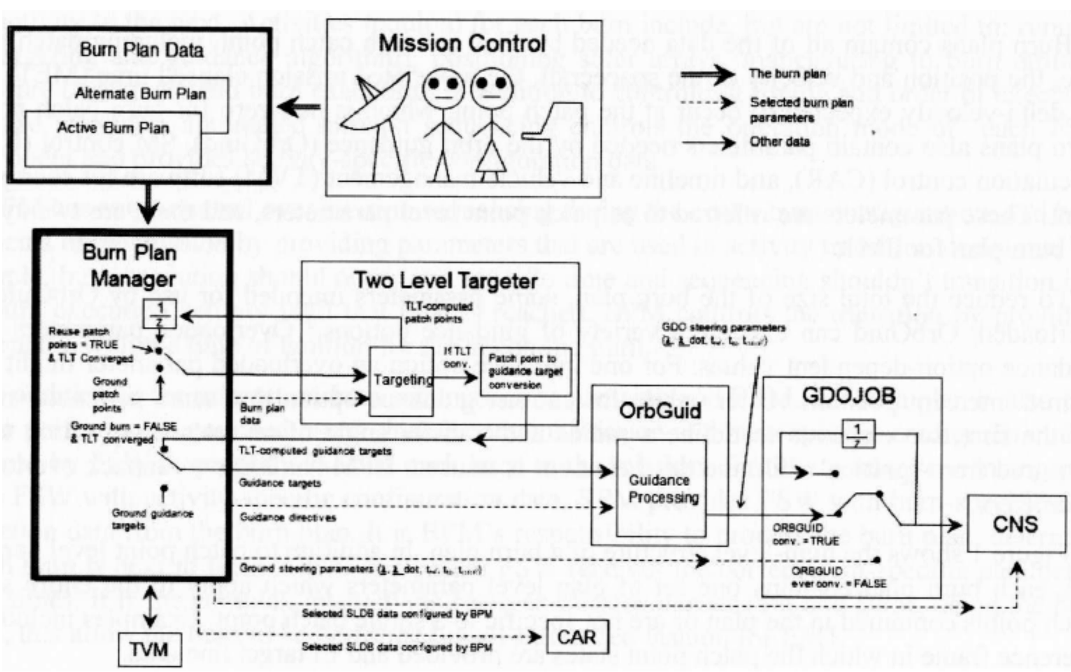


**Figure 14. Burn Plan Manager Interfaces and Functional Flow**

For Artemis I, the FSW can store two Burn Plans and has room for a BP buffer to temporarily store BP data while it is uploaded to the vehicle via one or several Date Exchange Message (DEM) commands. This ensures that all desired parameters are updated and validated prior to overwriting data in an existing BP. There is only one active BP at a time and which one is active can be controlled via DEM command, including determining which BP is active during PCL (achieved by DEM commands triggered by events).

TLT is a targeting function that starts with Patch Points along a reference trajectory and executes a two-level iteration scheme to connect the points and adhere to the desired constraints. The Patch Points are often where burns are placed, special constraints are needed (like EI), or they are simply used to break up long coasting arcs along the trajectory. The initial Patch Points are loaded in the Burn Plan and are generated using a high-fidelity trajectory optimizer called Copernicus. It would be too computationally expensive and too complex to implement Copernicus within the onboard FSW, so TLT combined with initial data from Copernicus provides the desire functionality needed for autonomous onboard targeting. An initial version of TLT was used in the design of the GENESIS solar wind sample return trajectory, but Orion is the first to use TLT as an onboard targeting routine.

As its name implies there are two levels of iteration within TLT that converge on the desired trajectory target points to be passed along to OrbGuid. Level-1 propagates sequentially through the Patch Points iterating using discrete delta-velocities (delta-V) or burn arcs to enforce position continuity along the trajectory. Once Level-1 has converged, Level-2 applies additional controls or constraints to reduce delta-Vs and impose other trajectory constraints such Flight Path Angle (FPA) at EI. The Level-2 process alters the position of the Patch Points generated by Level-1, so Level-1 is repeated and then Level-2 and so on until the trajectory target points (updated Patch Points) have stabilized within a desired tolerance. See Figure 15 for an overview of the TLT function. Once TLT has converged, BPM will send the updated target information and other directives for the next burn to OrbGuid. If TLT did not converge, the BPM will pass along targets already stored in the Burn Plan. If TLT did converge, the current target points may be used as initial Patch Points for the next burn (unless disabled by DEM command).[17]

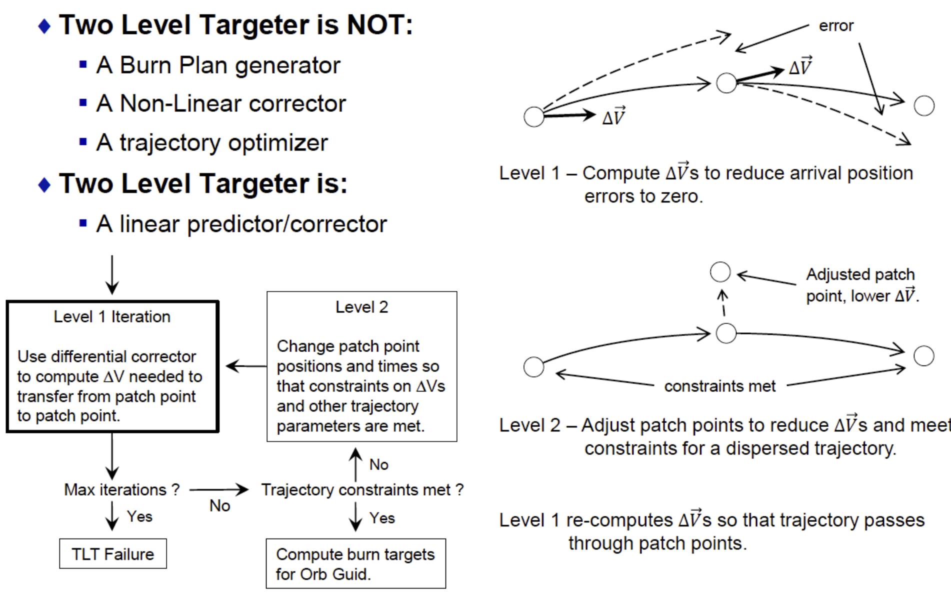


**Figure 15. TLT Overview**

OrbGuid is used to take the targets generated by TLT and other directives from BPM and compute the necessary burn to achieve the desired target. The core of OrbGuid is the Power Explicit Guidance (PEG) routine developed for the Shuttle but wrapped by an executive function that allows it to be used for a variety of desired burn solution types. PEG is a predictor-corrector used to solve for steering law information and burn duration. The data-driven flexibility of the OrbGuid architecture allows it to cover a wide variety of burn guidance needs and reduces the number of separate guidance routines required to support all the planned Artemis missions. There are five available options for desired velocity: External Delta Velocity (simply burn out the desired delta-V), Linear Terminal Velocity Constraint (LTVC – primarily used for Low Earth Orbit (LEO) orbit insertion and de-orbit burns), Free Range LTVC (variation of LTVC that removes the transfer angle constraint to minimize delt-V for lunar return CM-raise burns), Transit Guidance Option (solves the classic Lambert time-of-flight point-to-point transfer), and Constrained Intermediate Terminal Intercept (CITI) mode (applies a FPA constraint on an intermediate point).[18]

Steering outputs and burn duration data from OrbGuid are passed to CNS to execute the desired burn with the requested SM engine (OME, Aux, or RCS) and any secondary attitude constraints (like roll for antenna pointing to maintain comm). The BPM also provides data for CAR to direct the SAWs to the proper burn position to adhere to loads and pluming constraints while maximizing power generation when possible. Lastly, the BPM provides data necessary to carry out any available downmode option during a burn (OME to Aux, 8 Aux to 6 Aux, or Aux to RCS).

Clearly, in order to bring the crew home safely with no comm, Orion needs a very robust, autonomous, automated, data-driven targeting and burn execution system. I believe that Orion BPM, TLT, and OrbGuid collectively comprise the most sophisticated onboard FSW of its kind flown to date.

## TIME TO COME HOME – RETURNING TO EARTH

At some point, all Artemis missions must return the crew home to Earth. Burns are executed to place Orion on a trajectory that intersects the atmosphere at just the right location and orientation to allow the CM to complete a successful entry and land at the desired location off the coast of San Diego, California. The following sections will describe the key elements of Orion GNC that make that possible.

### GPS Fast Acquisition – Goddard IP on Honeywell's Boeing 787 Receiver

As always, first things first, we need to know where we are before we can decide where to go and how to get there. Early in the Orion Program the Navigation Team spent a lot of effort on what type of GPS receiver (GPSR) would be used for near-Earth and atmosphere flight. Given the proven robust and accurate capability of GPS for near-Earth operation, it was a logical choice to make this a key part of Orion Navigation for this flight regime. One of the primary areas of concern was the ability of the receiver to acquire GPS satellites quickly to minimize measurement outage time following plasma comm blackouts and first acquisition post LAS jettison for an ascent abort. During entry the heating produced as the capsule slows down generates a plasma wake surrounding the vehicle and blocking all comm signals including GPS. When Orion does a skip entry there are two pulses that produce such plasma events, and it is desired to reacquire GPS signals as quickly as possible. More on skip entry in a coming section of this paper. In addition, until the LAS is jettisoned, ogive panels cover the CM and the GPS antennas blocking any signals from reaching the receivers. Depending on altitude, a LAS abort has only a small window between LAS jettison and chute deploy, so it is important for GPS to rapidly acquire satellites.

While less critical, acquiring GPS post-LAS jettison is also useful for any SM aborts, especially an AOA that required a targeted burn (not needed for Artemis I however).

After a make-or-buy trade study, Honeywell decided to take the receiver they had recently developed for the new Boeing 787 aircraft and modify it to meet Orion requirements. Around the same time, Goddard Space Flight Center (GSFC) was developing their Navigator GPSR that had a unique process for simultaneously performing the autocorrelations across all the GPS satellite frequencies. This algorithm was implemented within a Field Programmable Gate Array (FPGA) for improved efficiency along with flexibility. Honeywell took the Goddard fast acquisition design and implemented it on an application-specific integrated circuit (ASIC) to make it even more efficient (but less flexible). Honeywell also made other modifications necessary for a receiver that flies outside of Earth's atmosphere and even beyond the GPS satellites. Overall Orion's GPSR has performed well for Exploration Fight Test-1 (EFT-1) and Artemis I. One unfortunate miss in the implementation of the Goddard design resulted in having to reset the GPSR to go back to fast acquisition mode after being set to mixed or weak signal tracking. Mixed or weak signal tracking mode helps the receiver track GPS signals at a greater distance. Since we must reset the box to get the receiver back into fast acquisition mode for entry, we have chosen to leave it in strong signal mode (fast acquisition) which reduces the ability to track GPS signals at longer ranges. While this is disappointing, GPS navigation is not needed at those longer ranges to complete the mission. If we had a high-gain antenna (which we don't) and could track GPS prior to the final correction burn on the return leg, that would be a different story, and something worth pursuing. As for now we are fine with this limitation.

### Barometric Altimeter and other Backups for Chute Deploy

Orion had a barometric altimeter (BALT) in the initial design concepts, but along the way it was removed, and others and I fought hard to bring it back. Some of the arguments for not needing it was it took two GPS failures to need it. While unlikely, there are many ways that GPS may be limited or not available and it seemed crazy to me not to at least have the same capability as Apollo. The primary need for the BALT is chute deploy and other critical altitude-dependent triggers during entry descent and landing (EDL). If GPS is not available, all you have is inertial propagation which is very poor in the altitude direction due to errors in gravitational modeling which cannot be sensed by the IMU. Attitude errors also greatly impact inertial-only propagation. Fortunately, a simple but robust BALT is accurate enough to do the job for chute triggers. Thus, no entry vehicle reliant on parachutes, should ever go without a barometric altimeter. The Orion BALT is made up of three barometric altimeters for redundancy and utilizes a mid-value select approach for output (or average if one is failed). Since Orion has multiple altitude sources, it uses altitude selection logic along with FDIR health checks to select from the following altitudes in this priority order: filtered altitude from selected EKF, independent GPSR-derived altitude from selected GPSR, BALT, and inertial-only propagation.

### Skip Entry Guidance – First Ever "real" Skip Flown

Have you ever tried to skip a rock across a pond? I've seen it done by others on videos and TV but it's a lot harder than it looks and very dependent on the type and size of rock, the wind, and water conditions. One of the many amazing things accomplished during Artemis I was the perfect execution of a true skip entry guided to the desired target within a few nautical miles. While some other spacecraft may have done some "lofted" entries, no other spacecraft has completed a true skip entry. Orion entry guidance uses the lift and drag forces generated by the capsule shape to slow the vehicle from lunar-return speeds down to subsonic drogue deploy conditions and steer the trajectory to the designated landing site. The guidance does this by rolling the lift vector to go left or right and up or down. Although Orion does not have a large lift-to-drag ratio (L/D), it has enough to do the job, primarily in the along-track range control. Since the L/D is relatively small, Orion has very little cross range capability. Thus, the orbital burns controlling the geometry of the trajectory at EI are very important. Using the "skip" technique and high-speed entry velocity however, Orion can use what L/D it has to fly short or long ranges relative to EI. This allows Orion to target a standard landing site even with variable Earth-Moon geometries that drive different return trajectories. Orion targets a special EI target line (see Figure 16) to enable landing off the coast of San Diego, California.[19]

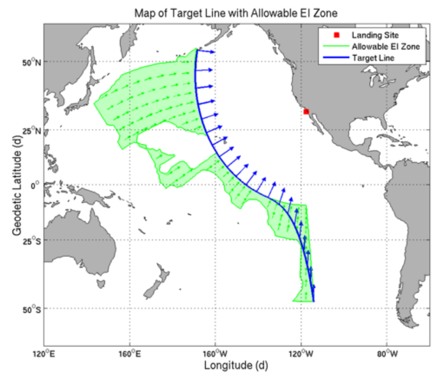


**Figure 16. Horizontal EI Target Line**

Early in the Orion Program, there were three entry guidance algorithms considered: NASA's Numerical Skip Entry Guidance (NSEG), Draper's PredGuid, and LM's LMGuid. The LMGuid did not continue to be pursued, but the other two were compared in a multi-phase fly-off to pick which algorithm to use. After phase-1, algorithms were shared between the two and the fly-off really became more of a merger. Phase-2 of the fly-off showed that NSEG and PredGuid had similar performance with some attributes favoring PredGuid and others favoring NSEG.[20] Figure 17 shows one of many comparison plots between NSEG and PreGuid. In the end, PredGuid was chosen in part because it utilized more of the already flown Apollo entry guidance algorithms. Unofficially, going with the Draper version also helped to ensure that the much-needed expertise of the Draper guidance team would remain to help continue development.

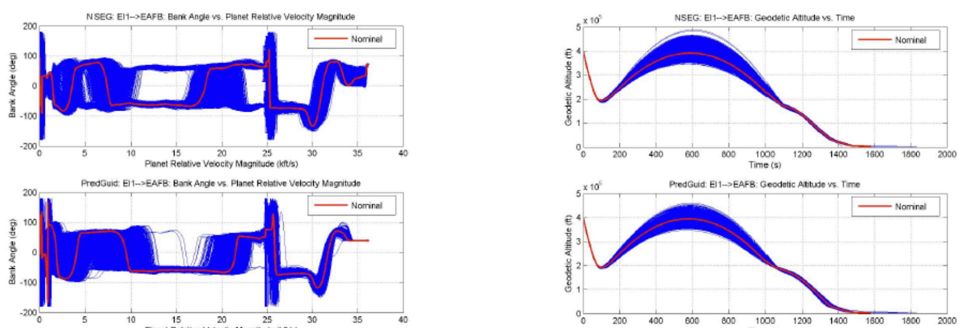


**Figure 17. NSEG vs PredGuid: Bank Angle and Geodetic Altitude vs Range**

During the NSEG vs PredGuid flyoff back in 2007, I remember working with the guidance team on the performance sensitivity relative to navigation errors. The details are in the reference cited,[20] but one aspect both algorithms really rely on is IMU acceleration data to estimate the combination of L/D and atmosphere dispersions. The individual estimates of drag coefficient, lift coefficient, and density may not be accurate, but the IMU aids the guidance routines in estimating the total effect of these on vehicle capability and greatly improves overall performance.

Another interesting piece of history is the evolution of the CM raise burn. This is a burn executed with CM RCS jets shortly after CM/SM separation and before EI. It is used to tweak the trajectory and make small adjustments, primarily to the EI flight path angle. Originally, the CM raise burn was needed to allow proper disposal of the SM for ascending approach entries returning from ISS. Orion was supposed to go to ISS in the beginning, but all that changed after Constellation was cancelled. Even though Orion would no longer return from an ISS orbit, the CM raise burn capability was an enabler for the Program to save mass on the heat shield and save mass by removing the battery from the SM. The SM battery allowed for an SM disposal burn that would no longer exist, so in order not to impact L/D, entry corridor sizing, or CM mass, the CM raise burn is used to adjust the FPA at EI. This capability allowes for the initial trajectory with the SM attached to be steepened thereby placing the SM where it needs to be and then executes a CM raise burn to put the CM back within the required corridor. The entry corridor is critical to the heat shield thermal protection system (TPS) sizing. The larger the corridor, the larger the thermal dispersions which drive more TPS mass required. Keeping the CM raise burn also allows for larger initial EI FPA dispersions for a permanent loss of comm (PCL) scenario. Thus, although the original reason for the CM raise burn no longer exists, the desire to reduce mass has kept the need for this capability. Along with the perfect execution of the first-ever skip entry, Artemis I also successfully executed Orion's first closed-loop guided CM raise burn (EFT-1 CM raise burn was open-loop).

**Touchdown Roll Control**

As if skipping Orion off the Earth's atmosphere and landing just off the coast of California wasn't hard enough, pressure vessel and crew seat stroke limitations drove the need to "knife" the capsule in the water at a particular orientation (CM +Z body frame). There is only so much room in the CM to allow the seats to stroke in a specific direction, and the structure is optimized to save mass. Thus, the Orion GNC team worked many years to develop a system that would reliably orient the vehicle as desired prior to splashdown. This technique is called Touchdown Roll Control (or Touchdown Heading Control). The parachute riser line attach point is adjusted off the center of gravity to establish a hang angle allowing the CM to slice into the water and minimize loads. Navigated knowledge of inertial attitude and the Earth-relative Nort-East-Down (NED) reference frame is used to control the vehicle attitude relative to the horizontal velocity direction of travel. The necessary performance to limit impact loads requires that GPS be available to aid the navigation solution, otherwise the knowledge of the horizontal velocity direction is not accurate enough. Therefore, a backup heading based on predicted winds in the vicinity of the expected landing zone is used if GPS is not available. Figure 18 shows the desired splashdown orientation.

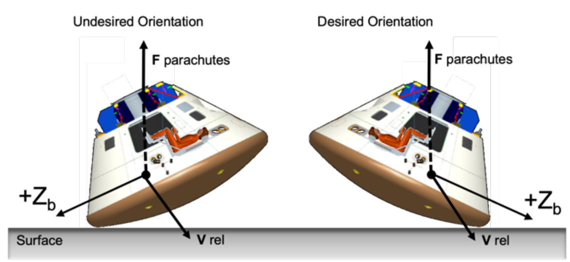


**Figure 18. Desired Orientation at Splashdown**

Shortly before EFT-1 flight, it was discovered through parachute drop testing that a pendulum motion could be excited if one of the three main chutes failed leaving the CM to descend under two mains. While it is still safe for Orion to only have two main chutes, a large pendulum motion would pose a threat to the success of Touchdown Roll Control. There was no time to implement any changes for EFT-1, but the GNC Team went to work right away to address the issue and get ready for Artemis I.

A large amount of work was poured into solving the pendulum problem and is well documented in the cited reference.[21] A pendulum "observer" was created utilizing the navigated NED velocity to estimate the plane and amplitude of the pendulum oscillations. This allows the system to track the overall direction of travel and not chase the oscillating velocity shifts due to pendulum motion. Studies showed that Orion did not have enough control authority or prop to damp out pendulum motion if it were to manifest, so efforts were made to see what else could be done to minimize the effects on impact. Unfortunately, depending on the CM orientation relative to the swing plane and the direction of travel at splashdown, the location of impact on the heat shield could end up failing the structure. After studying the problem, it was determined that if the CM +Z body axis was oriented perpendicular to the pendulum plane of motion, this would move the impact point further away from the center of the heat shield reducing loads on the structure. Since there are two sides of the plane, the direction producing the smaller angle relative to the horizontal velocity vector is chosen. This pendulum logic is only executed if the horizontal velocity is below a given threshold and pendulum energy is above a certain amount. Otherwise, the standard heading control is engaged. This new technique was validated using reconstructed parachute drop test data. More details on the final implementation and performance on Artemis I can be found in the cited reference.[22] There were no chute failures during Artemis I and the nominal Touchdown Roll Control performed as expected.

**Touchdown Detection**

After controlling the CM to the proper orientation for splashdown there is one more key item for the GNC system to perform: touchdown detection. About half of the Apollo missions ended with the CM capsule upside down in the water. Orion calls this stable-2 position and upright is stable-1, and if it is somewhere in between, that is stable-3. Like Apollo, Orion has airbags that inflate post-splashdown and are designed to upright the CM if needed. However, the desire is to avoid going into stable-2 if possible, and cutting the parachute lines after splashdown is done to keep the parachutes from dragging the CM through the water and into the stable-2 position. Consequently, the chute lines must be cut after splashdown driving the need for a robust method of touchdown detection. Touchdown detection is also the segment transition trigger within the FSW that initiates the Crew Module Up-righting System (CMUS) airbags. Thus, detection must be timely enough to have the CM upright before any ill effects are incurred from the crew hanging upside down. In addition to cutting the chutes, there is a desire to terminate the RCS jets that are actively doing touchdown roll control before they are submerged by water. Early in the Orion Program it was thought that firing the thrusters underwater could be a catastrophic hazard. Further testing showed this was likely not a significant risk, but it is still a good idea not to fire jets underwater, so the trigger remains.

Since EFT-1 and Artemis I were both uncrewed, touchdown detection needed to be automated and autonomous so as not to rely on comm with a vehicle whose antennas could be under water or splashed with water. Starting in early 2010 I spent nearly five years designing, testing, and implementing the Orion touchdown detection algorithm. Of course, I had lots of help along the way from a graduate co-op, other NASA and LM engineers, and an LM GNC team member who implemented the FSW. In addition to the functional requirements desired, the driving design factor was safety. In no way was I going to be the one to lose the vehicle or crew by cutting the parachutes at the very end of a successful mission. This edict resulted in a simplistic, but robust design tuned to bias toward safety over missing a soft landing.

After running through a design trade study that looked at detection method, frequency of data evaluation, persistence counter, and filtering rate data for lever-arm corrections, the Orion touchdown detection was formulated.[23] Detection is initiated after a fixed time from main deploy and an altitude less than a given threshold (altitude only checked if GPS or BALT data is available). Touchdown detection method is magnitude of filtered acceleration measured by the selected IMU with a lever-arm correction (computed with filtered gyro rates) applied for vehicle rotation (IMUs are not at the center of gravity). This allows touchdown detection to work even if the navigation state is in error or badly corrupted, and three IMUs provide redundancy to hardware failures. It also simplifies the logic relative to other methods that may use GPS or barometric altimeter data. Using other such data can be useful, but failures of those data types must be considered, and additional logic added to maintain robustness. Two thresholds are employed to separate the emphasis on terminating RCS jets prior to water submersion vs cutting the chutes to prevent stable-2. By using two thresholds safety relative to cutting the chutes can be maximized while providing a more sensitive and faster reaction to shutting down the jets. The algorithm utilizes the high-rate 200Hz IMU data along with a tuned persistence counter to maximize sensitivity of the trigger. Figure 19 illustrates the impact of persistence on detection performance.

Once touchdown is detected, the RCS jets are terminated, and a delay timer starts that will give the crew (if present) the ability to be prime for cutting chutes. If crew are not present, the delay timer is set to 1 or 2 seconds. I strongly believe, that if crew are present, they should be the ones to initiate cutting the parachutes. There is no reason to risk lives for something that is easily detectable by the crew. In addition to the delay timer, there is a backup timer that starts when touchdown detection is activated and goes long enough to bound all possible main chute deploy variations. The backup timer covers for undetected soft landings. This algorithm flew on both EFT-1 and Artemis I and performed perfectly on both missions.

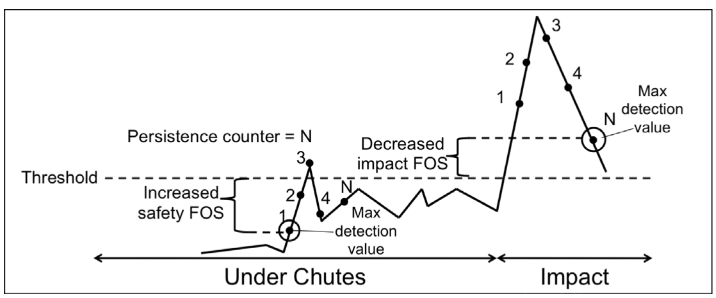


**Figure 19. Touchdown Detection Persistence Counter Assessment**

## CONCLUSION

I hope you enjoyed this brief sampling of what was an incredible journey designing, developing, testing, and flying the GNC system on the first human-rated spacecraft to go back to the Moon in fifty years. It has been truly remarkable to be a part of the Orion Program from the beginning, and to see all the hard work and creativity transformed into an amazing vehicle that will no doubt be a flagship in the United States return to the Moon and beyond for decades to come.

## ACKNOWLEDGMENTS

I would like to thank all the thousands of men and women who contributed to the historical Artemis I mission, and I would particularly like to congratulate the great Orion GNC team for their amazing achievement.

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