Evolution and Impact of Saturn V on Space Launch System from a Guidance, Navigation, and Mission Analysis Perspective

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Abstract

The Saturn V launch vehicle represented a jump in capability for heavy lift launch vehicles, enabling the Lunar Orbit Rendezvous approach to planetary exploration employed by the Apollo program 50 years ago. Following Apollo, and the development of the Space Transportation System, the NASA space exploration program shifted focus from lunar exploration to long-term, sustained, re-usable access to Low Earth Orbit. With the recent focus of NASA on the Artemis program and continued exploration of cislunar space as a precursor to Martian exploration, the shift has swung back to heavy lift capability. To meet this need, NASA has developed the Space Launch System. While the vehicle is a new design, it is heavily influenced by the engineering solutions and approach used on the Saturn V while taking advantage of the state of the art of launch vehicle design. The approach to abort, for example, shares many familiarities with the triggers and concept of operations used on Saturn V. Analysis approaches to dispersed trajectory performance are also very similar, but advances in computing technology have enabled a much more expanded set of inputs that can be modelled and assessed in a rapid manner. Additionally, guided flight algorithms share similar first principles but have expanded to include day of launch wind information. Trajectory optimization has also advanced significantly due to the availability of computing resources, but similar maneuvers and profiles are flown across both vehicles. Also, while the approach of onboard inertial navigation has been maintained between the two programs, the shift from platform to strapdown systems enables reduced complexity in the system design while maintaining required performance. As described, the Space Launch System is the evolution of NASA launch vehicle designs, owing a large heritage to the Saturn vehicle program and incorporating advances in propulsion systems, avionics, computing, and sensor technology over the past 50 years.
Keywords: Saturn V, Space Launch System, Artemis I, GNC, Mission Analysis

Nomenclature
a – variable a (placeholder)

Acronyms/Abbreviations
DOLILU – Day of Launch Integrated I-Load Update
ICPS – Interim Cryogenic Propulsion Stage
IGM – Iterative Guidance Mode
PEG – Powered Explicit Guidance
RCS – Reaction Control System
SRB – Solid Rocket Booster
SLS – Space Launch System
SSME – Space Shuttle Main Engine
TLI – Trans Lunar Injection

1. Introduction

On November 9, 1967, the first launch of a Saturn V took place from Pad 39-A from the John F. Kennedy Space Center. This was the first full flight test of an uncrewed vehicle on the Apollo 4 mission, whose focus was on demonstrating the vehicle's capability with a fully integrated vehicle. The flight test was considered a success and all stages operated nominally. The Saturn V went on to fly an extensive series of missions (13) to put humans on the lunar surface as well as placing the Skylab Space Station into Low Earth Orbit.

An expanded view of the vehicle is given in Figure 1. The vehicle consisted of 3 unique stages, the S-IC, S-II and the S-IV B. The first stage used 5 F-l liquid propulsion engines developed by Rocketdyne, individually producing 1.5 million pounds of thrust at sea level, by burning a liquid oxygen and RP-1 (a refined form of kerosene). The second and third stages used the J-2 engines, developed by Rocketdyne, which are capable of generating 230000 pounds of thrust in a vacuum by combusting Liquid Oxygen and Liquid Hydrogen. The combination of these three stages integrated into a vehicle 360 feet tall, that was capable of placing 100,000 pounds onto a TLI trajectory. This heavy lift capability was the backbone of the Apollo program, enabling the launch of the Service Module, Command Module, and Lunar Excursion Module on a path towards the moon. The last flight of the Saturn V occurred in May of 1973.

After Apollo, NASA's vision shifted to sustainable, re-usable architecture for regular access to space via the Space Transportation System, or Space Shuttle, program. This program operated for 40 years, with 133 successful missions. This vehicle incorporated advances made in solid propulsion, utilizing two solid rocket motors to create large thrust to launch the vehicle off the pad, with 3 RS-25 (aka SSME) Liquid Oxygen/Liquid Hydrogen engines capable of producing 500 thousand pounds of thrust at sea level.

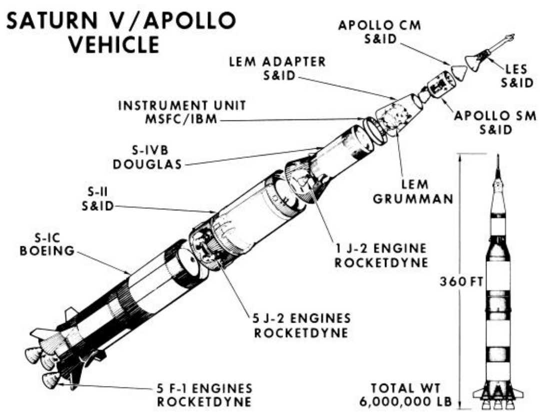

Fig. 1. Saturn V Expanded View

Upon the retirement of the Space Shuttle Program, and the investment in Commercial Space Transportation programs to enable commercial low earth orbit human missions, NASA has shifted launch vehicle development back towards heavy lift to focus on deep space missions, such as missions to the Lunar and Martian Surfaces. This is the goal of the Space Launch System program, returning NASA Human Spaceflight to missions similar to those performed in the 1960s and 1970s.

Figure 2 provides a view of the SLS Block 1 vehicle. It is powered by Space Shuttle heritage technology in a format integrating lessons learned from the Saturn V and STS programs. The core stage is powered by two large five-segment SRBs, an additional segment added from that used on STS. Additionally, 4 SSMEs provide additional thrust to support core stage flight and launch into orbit. An upper stage provided by ULA, the Interim Cyrogenic Propulsion Stage (ICPS), is used to insert the payload onto its desired trajectory whether that is a trans-lunar or deep space mission.

Figure 3 shows key events of the ascent phase. Approximately 130 seconds after liftoff the two SRBs are jettisoned. The core stage continues on alone and jettisons the service module panels which protect the service module from the forces and heating of ascent, followed closely by jettison of the LAS. After approximately 500 seconds of flight, the core stage reaches an orbit with the desired earth parking orbit apogee and perigee well within the atmosphere to ensure ocean disposal of the core stage during first perigee passage.

Figure 4 shows the in-space mission profile. Several seconds after core stage cutoff, the ICPS separates with Orion. After coasting to near apogee, the ICPS performs a perigee raise maneuver to raise perigee out of the atmosphere and into the final Earth parking orbit. Nearly a half revolution later, the ICPS performs the TLI burn to send Orion on its way to the moon. After Orion has separated, ICPS performs a small delta-V burn to modify the trajectory to a lunar swingby and ultimately a heliocentric orbit.

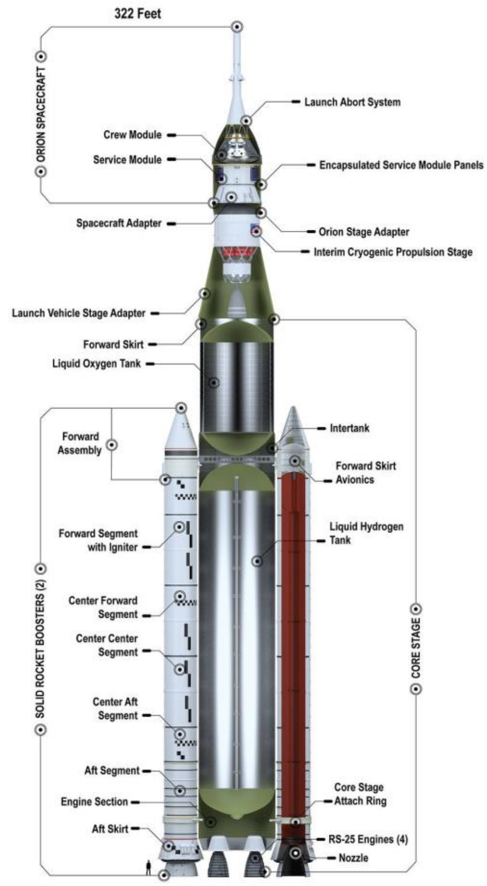

Fig. 2. SLS Block 1 70 Metric Ton Configuration

The Guidance, Navigation, and Mission Analysis (GN&MA) Branch, at the Marshall Space Flight Center (MSFC), is responsible for providing the SLS onboard guidance and navigation algorithms, developing the launch vehicle trajectories and associated flight techniques for the launch vehicle, and providing the onboard loss-of-attitude control detection algorithms. The GN&MA branch is also supporting day-of-launch operations in an independent verification and validation (IV&V) role to verify that the day-of-launch I-Load update (DOLILU) (of the day-of-launch wind-biased first stage steering profile) results in a trajectory that doesn't violate several pre-defined constraints, and that the measured day-of-launch winds are safe to fly though. The present-day GN&MA branch is performing many of the same roles and responsibilities as did its primary predecessor organization at MSFC, the Flight Mechanics Branch, which at the time was in the Aero/Astrodynamics Laboratory. These roles and responsibilities include design verification and flight certification.

The Space Launch Systems owes much of its design and operational heritage to the work performed in support of the Saturn launch vehicle program as well as advancements made during the Shuttle era. This paper provides a discussion of the key differences and similarities between the two programs in terms of its guidance and navigation design, aborts philosophy, trajectory and mission design, and analysis approach. These will each be discussed in the following sections.

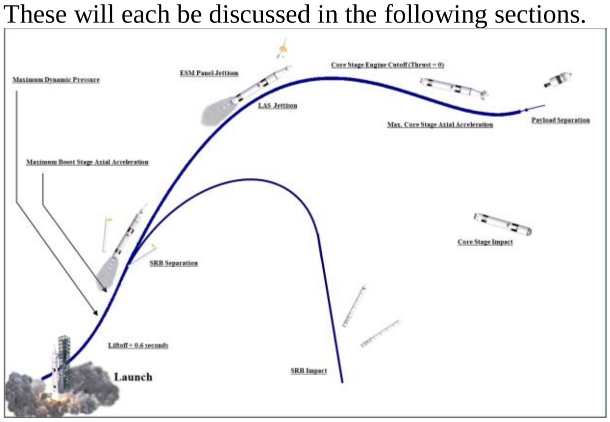

Fig. 3. SLS Ascent Trajectory Profile

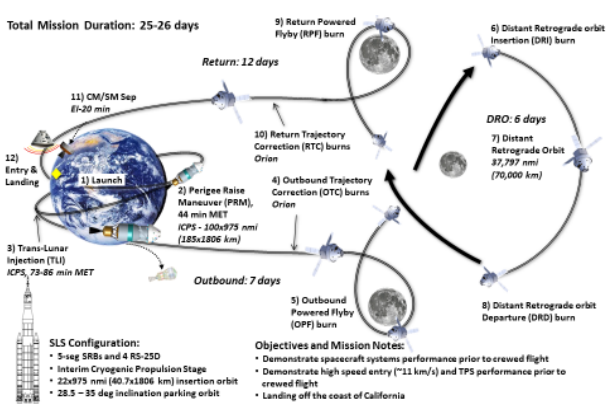

Fig. 4. In-space Mission Profile for Artemis I

2. Flight Mechanics

The flight mechanics discipline in the GN&MA branch assesses integrated vehicle performance, performs dispersed trajectory analysis, provides an independent verification and validation function to support day-of-launch operations, and performs GN&C abort trigger design and analysis. Several of the flight mechanics-related functions being performed for SLS are compared to those of Saturn V in the following subsections.

2.1 Dispersed Trajectory Analysis

The GN&MA branch, along with many of the other SLS and Orion disciplines and subsytems, generally performs Monte Carlo analyses to verify design requirements and perform flight certification. A typical Monte Carlo analysis consists of two thousand dispersed six degree-of-freedom (6-DOF) trajectories, randomized around some nominal operating point, e.g., a fixed launch month. Subsystem (e.g., propulsion system, mass properties, aerodynamics) and Natural Environments subject matter experts provide input parameter uncertainty distributions (e.g., Gaussian or Uniform distributions, with 1-sigma or min/max values) or databases (e.g., historical measured wind pair profiles at the launch pad). The 6-DOF trajectory simulation includes a flight-equivalent GN&C algorithm model, emulation of the day-of-launch first stage steering wind-biasing process, up to 300 of the primary vehicle structural flex modes and its effect on the navigation sensors, and high-fidelity models of the solid and liquid engine thrust vector control systems. Dispersed trajectory parameters are evaluated according to statistical techniques that take into account desired success rate and confidence levels [1] that are required to sufficiently demonstrate requirements (e.g., orbital insertion accuracy) compliance. In the Saturn V time frame, trajectory simulations ran at least an order of magnitude slower than present-day simulations, even accounting for the difference in modeling fidelity. Saturn V-era trajectory simulations would not have included high-order effects such as propellant tank slosh and multiple flexible body modes. A set of 2,000 high-fidelity SLS ascent dispersed trajectories can be run today in less than an hour (using the widespread approach of computer clusters with multiple computing cores). Monte Carlo techniques were not as well known or utilized in the Saturn V era (Ref. Wittenstein). Instead, root-sum-squared (RSS) techniques were a common method of predicting "3-sigma" performance envelopes. This approach involves perturbing critical model input parameters, in one-at-a-time fashion, running a trajectory simulation for each single perturbation, and RSS'ing the results. The perturbations were done at "3-sigma" values or, if additional conservatism was desired, at worst-case values to get a worst-on-worst analysis. These approaches reduced computer run time significantly compared to Monte Carlo techniques, and likely produced answers at least within the neighborhood of Monte Carlo-derived statistical results, although it is generally recognized today that Monte Carlo techniques provide more statistically-accurate results.

2.2 Flight Techniques

The Saturn V maximum acceleration approached 4 gs during S-1C flight. After S-1C separation, the acceleration remained under 2 gs for a more comfortable ride for the rest of ascent which was about 200 seconds longer than that of SLS. Apparently Saturn V didn't have LOX inlet pressure concern like SLS does. SLS maximum acceleration is not quite 3 gs during first stage flight and is approximately 3.3 gs during core stage flight. Throttle is required to maintain RS-25 LOX inlet pressure below limits and for Orion structural limits, and to reduce core-to-booster attach loads at SRB separation. SLS is a more sensitive vehicle than Saturn V in those respects. Saturn V had no throttle capability on any of its stages.

2.3 DOLILU Process

During S-1C flight, a pitch plane gravity turn ("tilt program") biased to the launch month mean wind profile was flown. For summer months, where month-to-month winds were similar, they would use the same tilt program for the entire quarter [2,3]. A 6-DOF trajectory simulation with day-of-launch measured winds was run on the day of launch in order to evaluate the trajectory and loads, in support of a go/no-go call. An advance in weather balloons, known as "Jimsphere balloons", had been developed specifically for the Saturn program in order to provide more accurate wind speed and direction data in the high dynamic pressure altitude region to support launch operations [4]. These were the precursors to the space shuttle program Day-of-Launch I-Load Update (DOLILU) process, and later the SLS DOLILU process. A key difference between Saturn V launch operations and both the late shuttle program and SLS is that first stage steering profiles were/will be wind-biased to the day-of-launch measured wind profile a few hours prior to launch, in order to increase launch availability due to winds aloft. In the SLS steering design, a gravity turn is enforced in both the pitch and yaw planes. SLS will rely primarily on a standalone Doppler radar system to obtain day-of-launch wind measurements, but historical Jimsphere wind profile databases were used extensively early on in SLS assessments. And wind measurement balloons will be used as backups to the Doppler radar system.

2.4 Stage Disposal

The S-IVB stage performed a "lunar slingshot" [5]maneuver, similar to the planned ICPS disposal maneuver, around the moon to dispose heliocentrically, for the first five manned flights. On Apollo 12 the spent stage flew too high above the moon and failed to achieve a heliocentric trajectory. On the rest of the lunar missions, the stage was intentionally disposed via lunar impact, from which seismic data was collected. In the space shuttle and SLS eras, NASA, as well as the rest of the international space community, has become more conscious of the need to limit orbital and space debris. There is also a strong desire to protect historical lunar landing sites.

2.5 Abort Systems

Saturn V had a Launch Escape System (LES) [6], SLS has a Launch Abort System (LAS), both performing the function of providing a mechanism of quick escape for the crew should there be an "abortable" malfunction on the launch vehicle, such as a thrust vector control hard over or a failure of a critical navigation sensor. The abort motor in both cases would ignite and pull the command module from the malfunctioning stack with a high acceleration, directed so as to outrun, and to laterally maneuver away from (via an abort control motor), the still-accelerating vehicle stack below. The LES was jettisoned approximately 33 seconds after Saturn V second stage (S-II) ignition [5]. The LAS would be jettisoned when the launch vehicle attains a flight condition that provides enough time for crew module reorientation to a heat shield forward orientation prior to atmospheric reentry. The LES and LAS are essential for aborts within the atmosphere when dynamic pressure and/or vehicle acceleration are high, both of which would make it difficult for a spring-jettisoned service module to escape the launch vehicle. It is desirable to jettison such abort systems as soon as they are no longer essential, so as to reduce the payload performance penalty that their high masses incur. LES jettison occurred at ~300kft which is near the middle of the expected range of LAS jettison altitudes.

The onboard emergency detection system (EDS) on Saturn V monitored and directly detected various subsystem failures. The EDS also monitored angular rates as an indirect way to detect failures that could cause the vehicle to lose attitude control and necessitating a crew abort. The Mission and Fault Management System on SLS also monitors several subsystem parameters as well as "GN&C abort triggers" such as attitude rates, attitude errors, attitude rate errors, and attitude control system command saturation metrics. The Saturn V angular rate trigger thresholds were set at 3 deg/s in pitch and yaw rates and 20 deg/s of roll rate. These values are quite a bit lower (more conservative) than are the current SLS attitude rate thresholds. Nominal Saturn V attitude rates are observed to be lower than those of SLS [7], so the lower attitude rate abort thresholds of Saturn V are consistent with that observation. The abort detection system on the upper stage of the SLS, the Interim Cryogenic Propulsion Stage (ICPS), is also deemed the Emergency Detection System (EDS).

3. Trajectory Design and Optimization

Physics has not changed in the 50 years since the Apollo program, but the methods of analysing the physics behind the trajectory problem have evolved over the years as computer power has grown exponentially, resulting in the capability to rapidly analyse aspects of the mission that the Apollo program never could. Published in 1963, the Lunar Flight Handbook [8] describes some of the methodologies used to develop the flight plan, ranging from the Circular Restricted 3 Body Problem (CR3BP) to solving the n-body problem numerically with non-point mass gravity models. Those methodologies are still in use today, but now a trajectory can be calculated in a fraction of a second on a desktop computer compared to multiple minutes on a mainframe [9]. This faster processing time has allowed the SLS and Orion programs to pre-generate many years' worth of mission profiles for the Artemis missions in a matter of hours.

One of the big difference between the Apollo and early Artemis missions will be the Earth parking orbit. The Apollo program designed the Saturn V to insert into a near circular LEO at either 100 or 90 nmi in altitude [5]. The Block 1 SLS configuration requires an elliptical parking orbit due to the delta-V splits of the stages. SLS is ultimately designed for a more capable upper stage, but for the first few flights, will use a variant of the Delta IV Cryogenic Second Stage (DCSS) called the Interim Cryogenic Propulsion Stage (ICPS). The ICPS is slightly larger than an off-the-shelf DCSS, but does not have the capability to push Orion directly to the Moon from LEO. Instead, the SLS Core stage uses as much performance as possible to push the ICPS/Orion stack to the elliptical parking orbit.

Elliptical parking orbits going to the moon offer other complications. A circular orbit, as used in Apollo, offers daily alignments where the geometry favors getting to the Moon. An elliptical orbit reduces that to approximately half a lunar month or less. In order to get to the Moon from Kennedy Space Center, the moon must have a negative declination (i.e be in the southern hemisphere). If the Moon were not in the southern hemisphere, then the TLI burn would occur closer to apogee than perigee negating the benefit of the elliptical parking orbit. This can be partially overcome by expanding the launch azimuth range for ascent, but this really results in an improvement in the launch days that have bad alignment as the Moon's declination is close to zero.

The early lunar Apollo missions flew free return trajectories about the moon [10]. This trajectory would cause the Apollo spacecraft to swing around the moon and return to Earth with effectively no propulsion required. However, a free return trajectory restricts the landing sites on the moon, so later missions moved to a hybrid trajectory that starts on a free return Earth orbit, but once all systems are checked out, the spacecraft would perform a burn to move off the free return trajectory to one that enables landing at the desired site by targeting the lunar encounter directly. After that departure from the original free return trajectory, the Apollo spacecraft is no longer guaranteed to return to Earth without some form of correction burn. Apollo 13 flew a hybrid trajectory and due to the in-flight failure after departing the free return, had to perform a correction burn to get back to an Earth return trajectory [11].

The Artemis missions will differ from the Apollo missions in that only the first crewed Artemis mission will be a free return trajectory where the other missions will be direct lunar flights. Artemis I will be the first uncrewed test flight of the SLS launch vehicle and Orion spacecraft and fly a direct lunar transfer that will enable Orion to insert into a Distant Retrograde Orbit (DRO) about the Moon. This mission profile will allow Orion to check out its performance in deep space while also demonstrating the required reentry speed velocities prior to the crewed Artemis II flight. Artemis II follows a flight profile somewhat similar to the later Apollo missions, but with a twist. Instead of starting on a free return trajectory, Artemis II will start with the Orion spacecraft in a highly elliptical Earth orbit that it will depart from near the first perigee passage to enter a lunar free return trajectory. This mission profile was selected due to relative ease of aborts after the initial Apogee Raise Burn by SLS and to allow the crew sufficient checkout time during the first crewed Orion spacecraft's flight. Mission designs for Artemis III and beyond are still in work, but all variations include a direct lunar transfer and no free returns.

Another aspect of the mission design is defining when the launch vehicle is capable of completing the mission, or defining the launch period (number of days) and launch window (minutes in a day) for the mission. The basic methodology in defining the launch window has not changed since Apollo. The trajectory profile and launch window requirements of SLS required resurrecting the variable orbit inclination approach used in the Apollo program to target the Moon. The alternate approach of flying to a fixed orbit inclination and then steering to the desired orbit plane requires more performance out of the launch vehicle as can be seen in Figure 5. While either method can work for shorter launch windows, approximately 30-45 minutes, the Apollo program had a desire to have launch windows of at least 2.5 hours and SLS targeted a launch window of up to 2 hours to protect for delays on the day of the launch. To achieve launch windows of that duration, the variable inclination targets were required.

As far as getting out to the moon, Apollo used a target vector with a hypersurface rotated around the target vector to define the TLI burn. To generate the hypersurface, a perigee ring is rotated about the target vector which can be seen in Figure 6. While this approach is not directly used today, the geometry behind the solution can be used to provide initial guesses for the trajectory optimizers. On SLS, the optimal TLI burn must occur near perigee, so a target vector can be approximated using spherical trigonometry for the optimization by knowing the parking orbit inclination (inc) and argument of perigee (argp) and lunar declination (declin) at encounter by Eq. 1.

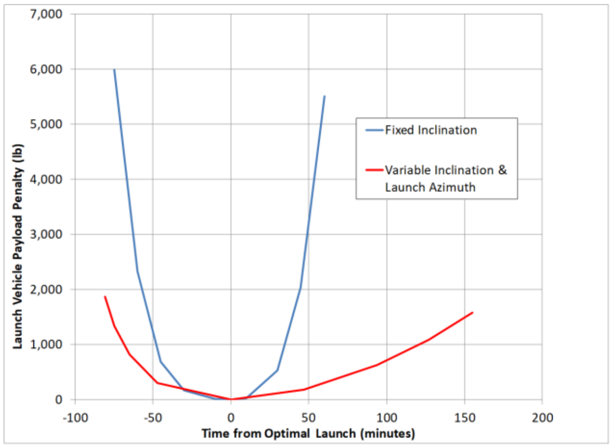

Fig. 5. Launch Vehicle Performance Penalty vs Launch Time.

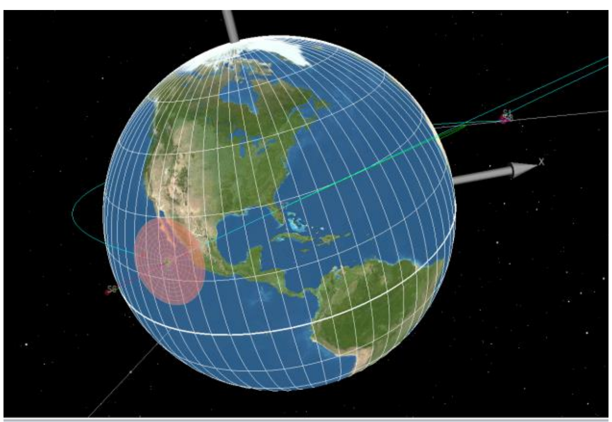

Fig. 6. Target Vector for TLI with Perigee Ring in Red

$$argp = \sin^{-1}\left(\frac{\sin declin}{\sin inc}\right) \qquad (1)$$

5. Navigation Design

Another area of discussion is of the similarities and differences in the navigation system between the Saturn V and the SLS vehicles. Both systems used inertial sensors to provide a measured acceleration in an inertial (non-rotating) reference frame that can be combined with gravity models based on the current estimated state to propagate the state forward and capture current velocity and position. This section focuses on the key areas of the navigation system design, and show how similar baseline approaches have been matured and refined using advances in avionics technologies.

The navigation problem was captured succinctly in the Figure 7 from [12]. It shows how the knowledge of state drives the vehicle insertion performance, and how these errors must be corrected by the payload to enter into its defined state. This mission profile is very similar to that flown for SLS with the largest difference being the state update prior to a trans-lunar injection. In terms of navigation design, no external measurements are used to update the onboard state and this manuever is pre-programmed based on extensive mission simulation and design.

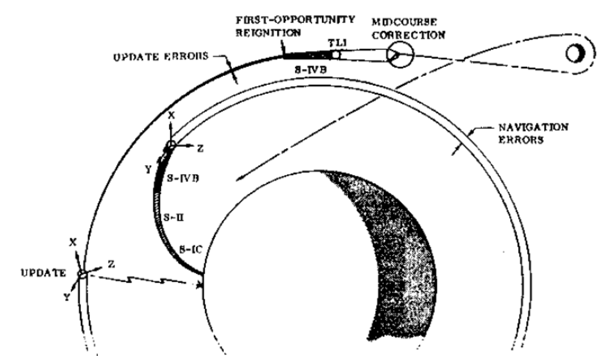

Fig 7. Lunar Geometry [12]

5.1 Platform vs. Strapdown Inertial Navigation

The primary purpose of the navigation system on both vehicles is to provide an estimate of the current position, velocity, and attitude. Both Saturn V and SLS utilized inertial measurement units for tracking attitude and inertial acceleration. The approach to sensor integration, though, was very different. The state of the art at the time of Saturn V (and still is for strategic navigation sensor systems, [13]) was to use an inertial platform. This system uses rate integration gyroscopes in a feedback manner to zero-out an rotation on an internal platform to keep the accelerometers aligned to a fixed inertial frame. From an initial attitude at launch, these rate integrating gyroscopes provide a change in attitude that had to be corrected to keep the platform fixed in inertial space. This provides a measurement of the vehicle's attitude. By maintaining the accelerometers in a fixed inertial frame, the sensors directly measured specific acceleration in that frame, allowing for direct integration into velocity and position given an initial state and gravity calculation approach. The IMU for the Saturn V was the ST-124 stabilized platform. In additional to the platform, 3 pairs of 3 axis rate observation gyroscopes were also integrated into the Control/EDS Rate Gyro package. These were used to feed the vehicle controller and provide rate feedback information for achieving a desired vehicle pointing and insertion state [14]. A cutaway of the ST-124 is given in the following diagram, Figure 8, which shows the individual components of the gimballed platform system [15]. Figure 9 shows an image of the ST124 gimbal systems. Lastly, Figure 10 shows a high level diagram of the guidance and control subsystem. This diagram shows how the navigation and rate sensors independently operate and feed into the primary flight computer.

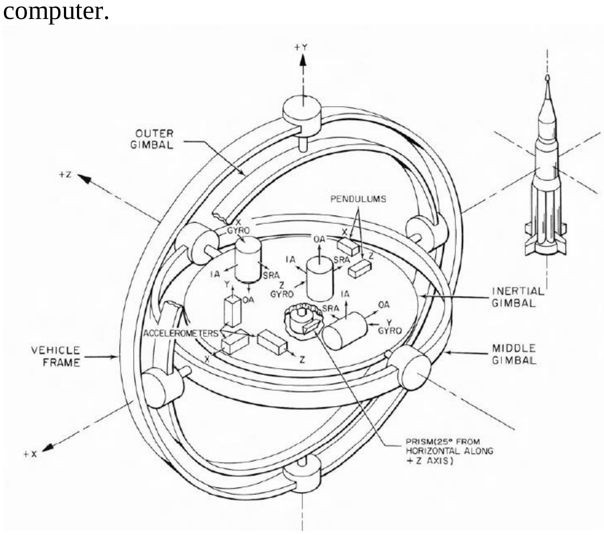

Fig. 8. Platform Assembly Diagram [15]

Fig. 9 ST-124M Inertial Platform (CC BY-SA 3.0/ Edgar Durbin)

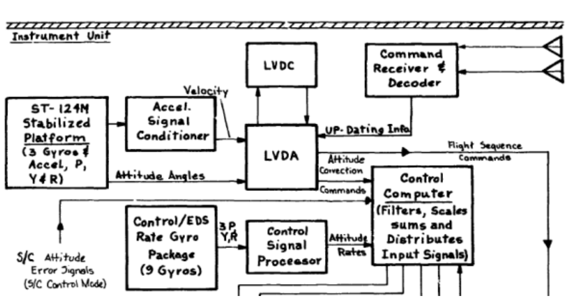

Fig. 10. Saturn V Guidance and Control Block Diagram [14]

In contrast to these systems, SLS utilizes advanced inertial technology, primarily the design of strapdown inertial systems [16]. With advances in the state of art of gyroscopes in devices such as a ring laser [17] and fiber optic gyro [18], it is possible to directly measure the inertial angular rate of a body in motion with incredibly high precision. Along with the advent of modern accelerometers [19], strapdown systems provide direct measurement of the angular rate and inertial acceleration in vehicle's body axis without the mechanical complexity of a gimballed platform system. The caveat is that the system must now track the vehicle's attitude and transform the body-frame accelerations into an inertial frame for state integration. The SLS vehicle utilizes the Redundant Inertial Navigation System (RINU) developed by Honeywell International, an internally redundant navigation guide strapdown unit. This unit, mounted in the MSFC 6 Degree of Freedom facility, is shown in Figure 11.

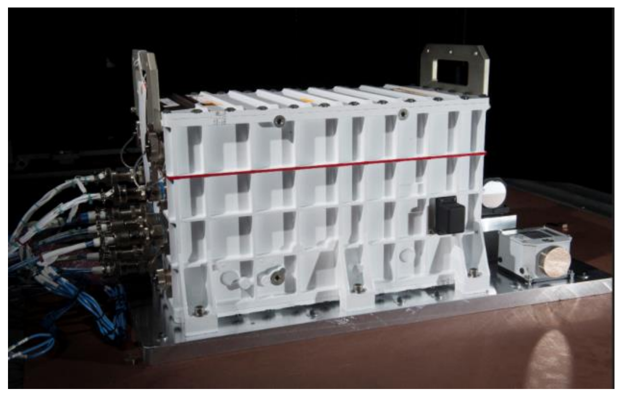

Fig. 11. RINU mounted on a 6DOF Motion Table [NASA/MSFC/Emmet Given]

5.2 Initial State Determination on the Pad

Sensitivity analysis results from launch vehicle navigation simulations identify that the largest source of uncertainty in insertion accuracy for a purely inertial system is due to initial uncertainty in the vehicle's knowledge of attitude. Both systems required initialization to a known attitude, but used different approaches.

The Saturn V design approach was to align the inertial platform such that the in-plane accelerometer channel is aligned with the desired launch azimuth, based on pre-flight analysis. In order to point the sensor in the correct direction, the Saturn V used a theodolite to provide an external reference to guide the inertial system [20]. Figure 12 shows an overview of the systems involved. Internal to the ST-124, there are several prisms mounted which enable observation of the current platform alignment. Through a ground alignment algorithm similar to that described in [21], the system can be calibrated on the pad prior to flight.

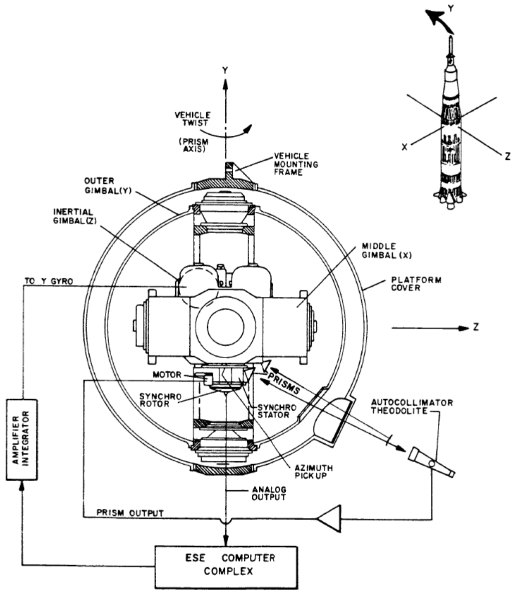

Fig. 12. Inertial Alignment Process for Saturn V

In comparison to platform-based approaches, strapdown systems take an alternate approach to initial attitude determination. As opposed to pointing the sensors in a specific direction, the focus is to determine to the orientation of the sensor about some datum that can be transferred to inertial-relative frame at the time of go-inertial. This is conducted through a sequence called Gyrocompassing [22], in which observations of local gravity provide a measurement of the local level orientation of the IMU, and observation of the inertial rotation rate of the Earth provides insight into the IMU's azimuth orientation. Multiple approaches can be used to estimate this orientation. The primary limitations to this accuracy is based on accelerometer bias uncertainty, gyroscope bias uncertainty, and long term noise behaviour of the gyroscope (angular random walk).

For both sets of systems, Kalman Filters and other approaches and extension simulation are used to assess predicted performance under expected environments. The key environment here is the Twist and Sway of the stacked up vehicle on the launch pad. Effects due to wind gusts and flexible body dynamics must be estimated to provide a truth model for simulation. Historically, sawtooth modes like those in [21] provide worst case dynamic behaviour to verify filter behaviour. With the increased fidelity of modelling tools and pre-flight analysis of SLS, analysts can generate predicted motion profiles of the vehicle on the pad, allowing for simulation of gusts events and steady winds during on-pad alignment. For SLS pre-flight testing was performed to captured the performance of the RINU under expected and worst case pad dynamics. A summary of this work is given in [23]. For this testing, the RINU was mounted to a 6DOF Table in MSFC's 6DOF Motion Facility to characterize the performance of a flight-like unit under potential motion. An image of the RINU on the table is shown in Figure 13 below.

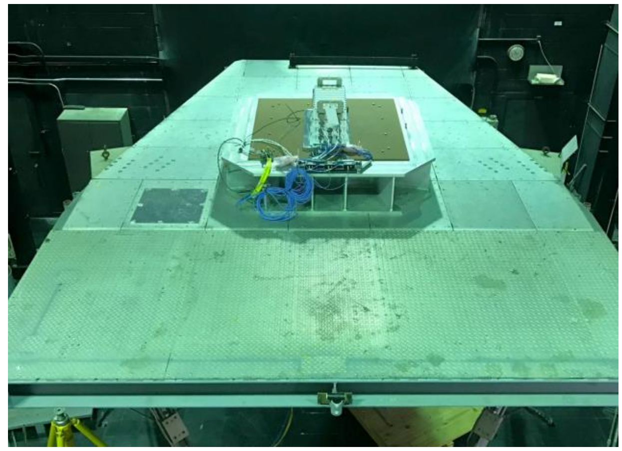
Fig. 13, RINU mounted on 6DOF Table

5.3 Range Tracking

To provide capability for knowledge of the vehicle during initial ascent, both vehicles are required to include systems to support range ground tracking. This is typically implemented in the form of a C-Band transponder that allows for radar tracking. For range safety, multiple independent sources of tracking data are required. The primary update to these systems is in the integration of GPS range tracking as part of the loop. Certification of these devices for range usage has been performed and they have been used. An example of a component in this area is the Vehicle Based Independent Tracking System by Space Information Laboratories which has supported multiple DoD and NASA missions [24,25]. This approach helps to provide additional high accuracy data to the range support assets while limiting reliance on ground radar trackers.

5.4 Fault Approach

Another area where the two designs differ is in terms of approach to faulted sensors. For the Saturn V platform, loss of accelerometer must be detected in flight and the internal gimbal be rotated such that the system maintains an active sensor in the in-plane direction [12]. Due to the gyroscope's ability to track attitude, this shift allowed the vehicle to maintain acceleration observation of the thrust axis. As part of the shift to strapdown systems, launch vehicle inertial measurements units have also switched to an approach using redundant sensors such as described by [26]. With the expanded onboard computational capability, it is possible to compare the sensors together and with enough sensors be able to fly through a sensor failure with minimal impact to navigation knowledge.

Additionally, fault redundancy requirements levied across flight hardware include first fault detection and isolation with second fault detection. This capability provides levels of redundancy and reliability needed for human spaceflight. Additionally, computational advances and expanded vehicle simulation capability allows for tuning of fault detection algorithms using modern optimization techniques such as Genetic Algorithms and assessment of capability within full 6DOF simulations [27]

5.5 Upper-Stage Enhancements

As mentioned, the Saturn V utilized a purely inertial navigation system for the ascent phase of flight. This approach has been de rigor for launch vehicles ever since. With the high reliability and performance of navigation systems, they have offered a proven, robust, and trusted approach to launch payloads into space. Due to the short mission durations from launch to initial orbit insertion, the errors (while not small) are well constrained and understood. Advances in navigation systems have changed this and provide an alternate path to low cost accurate ascent flight. This is primarily through the integration and usage of Global Navigation Satellite Systems, which can provide an external state observations and tightly bound navigation errors. These satellite-based systems allow for shifting away from high-grade navigation sensors due to the ability to correct for inertial errors during flight at a high rate.

The primary caveat to this approach, though, is on security and robustness of the solution. By coupling in another system, accuracy can be greatly improved but risks of interference due to jamming or spoofing must be considered [28]. While solutions such as encrypted channels do offer protection, a powerful approach is to combine a high accuracy navigation grade IMU with a high quality GPS. This is especially import in the development of upper stages for extended missions being planned for SLS.

One application of GPS that has enabled a new mission for the Space Launch System is in integration and usage of these systems onto the upper stage design. Coupling this capability has helped to increase the launch window for potential flight times and helped to reduce the system sensitivity on Earth-Moon geometry. This is particularly important for upper stage disposal and secondary payload deployment. A unique capability of SLS vs. Saturn V is the inclusion to host cubesats on the upper stage and place these satellites on a lunar flyby trajectory, providing access to deep space. To enable this, a very accurate navigation solution is required at high altitudes in Earth orbit. This is difficult due to the large of potential trajectory correction maneuvers or control of the stage post initial launch. As such, the stage must be placed on its lunar flyby trajectory several days ahead of time with sufficient accuracy to hit a tight window about the lunar surface. This analysis and approach is described in detailed in [29]. This is in contrast to the stage disposal approach of Saturn V, where the upper stages were aimed for a lunar impact.

Global Navigation Satellite Systems have had a profound impact on improving state knowledge around the Earth, both in orbit and on the surface. Capabilities coming online and being demonstrated now, such as the Navigator GPS Receiver, have shown the potential for this technology to enable greatly enhanced and autonomous navigation knowledge out to and beyond the moon [30,31]. Similarly to the way this technology has enabled a breadth of improved autonomy in ground-based systems, so too has space been changed. Similarly, small spacecraft in orbit are now able to track their state within a km, using electronics that can fit inside of a chip. As the Space Launch System continued to evolve in future designs, GPS will play a large part in allowing increased capability as described in [32]. Additionally, recent advancements in inertial technology will also have a large effect on the sensors being used to support this flight, not only in reducing size and weight [quote MEMS navigation grade papers], but also increases in capability [33].

6. Guidance Design

Guidance, a sub-discipline within GN&C, is responsible for commanding optimal steering profile to get a multi-stage spacecraft into a desired targeted orbit. Guidance system depends on pre-designed set of targets, generally derived by trajectory optimization, and Navigation, which provides Guidance system with current state estimate. Fundamental Guidance design has been preserved since Saturn V/Apollo program with addition of efficient numerical techniques as result of improved technology and computational power. The current SLS design for Artemis I shares many similar feature with did Saturn V. Both of these programs consist of multi stage rocket flight through atmosphere and vacuum of space. Both of these designs consist of utilization iterative guidance schemes, extended launch window availability of up to 4.5 hours, usage of polynomial design to vary targeted orbit and ability to handle single engine failure. Guidance specific flight modes are also similar. For example, Saturn V has 5 distinct modes which consisted of PRE-IGM, IGM, COAST GUIDANCE, IGM and COAST [34] as shown in Figure 14. Analogous to Saturn V’s PRE-IGM, SLS consists of Open Loop Guidance, Powered Explicit Guidance, coast, Perigee Raise burn and Trans-Lunar Injection (TLI) burn phases. Trajectory correction, post TLI, is also performed via a RCS settling motor burn in order to correct stage dispersions caused by hydrazine blowdown.

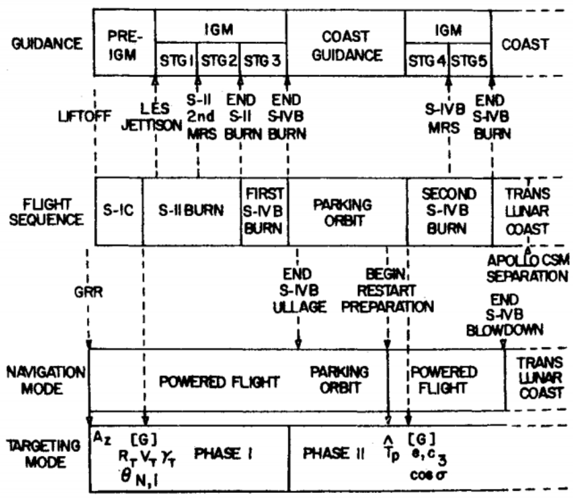

Fig. 14. Saturn V GN&C Flight Modes.

6.1 Open Loop First Stage Guidance

For Saturn V, Open loop guidance operated from Liftoff until Launch Escape Tower (LET) jettison. To support 4.5 hour long duration, pre-computed quartic polynomials for targeted azimuth and quantic polynomials of inclination and node were utilized. Open loop guidance had no yaw maneuver (except for an initial small tower avoidance maneuver) at all [7,3,35], unlike SLS which does day-of-launch wind-biasing which builds in yaw maneuvers to design a rigid gravity turn maneuver with respect to the measured design wind. That, combined with monthly (instead of day of launch) wind-biasing on Saturn V, would have resulted in higher sideslip angles during high Q conditions.

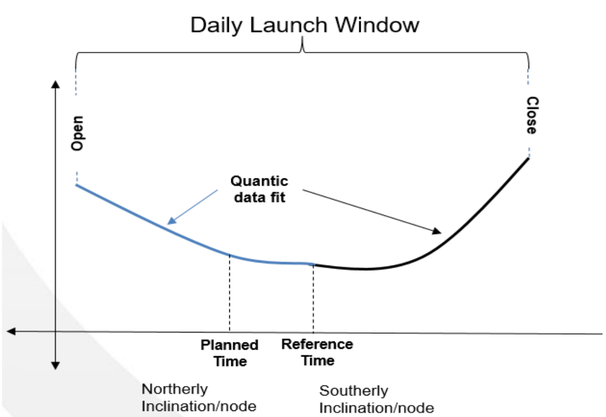

Fig. 15. Launch Window Polynomial Curves

SLS, on the other hand, uses two sided 5[th] order polynomials for inclination and node curvatures, and also supports up to 4.5 launch window. In contrast with Saturn V's usage of a Azimuth Polynomial, SLS utilizes a quantic polynomial for open loop guidance yaw command bias, also known as del-psi ($\Delta\Psi$). $\Delta\Psi$ is an additional bias added to the optimized set of Euler yaw commands after day of launch wind biasing has been performed. Figure 14 briefly shows SLS launch window polynomial design. For each node and inclination, there are two 5[th] order sets of coefficient determined using data from the trajectory optimization process. Division between two curves, Northerly and Southerly polynomials, is determined by reference launch time. Reference launch time is a time at which polynomial error is minimized as a result of curve fitting process. Planned time is chosen such that additional $\Delta\Psi$ bias due to launching at open is the same as for launch at close. This helps with load relief during max Q region.

SLS differs from Saturn V as the SLS Flight Computer (FC) does not have ability to vary azimuth targeting. Instead, SLS FC relies on an external Day of Launch tool call CHANGO [36]. CHANGO is developed by the SLS Guidance team to generate input consisting of vehicle attitude and throttle as a function of change in altitude since launch on day of launch. For each, there is a plan to have unique input files that would be uploaded to the FC on each potential day of launch. This way, SLS is able to take advantage of incorporating day to day wind variation in open loop design.

6.2 Closed Loop Guidance

Both Saturn V and SLS utilized a closed loop iteration form of guidance, which periodically updates the steering solution necessary to achieve a desired orbit. Saturn V used Iterative Guidance Mode (IGM), which is an explicit guidance scheme that targeted parameters specified explicitly (i.e., targeted position, velocity and flight path angle). IGM does not use calculus of variation, however, performance is near optimal, that is performance difference is on order of magnitude smaller, due to sub-optimality, than variations due to uncertainties. IGM is able to adapt itself in case of engine failure scenarios via means of pre-specified inputs. IGM generates steering commands specified in the guidance frame, which is similar to a rotational frame with axes in the direction of position vector, angular momentum vector, specified at predicted engine cutoff. In its formation, IGM utilized a flat earth model and used an average gravity between current stage and cutoff state, updated periodically. IGM used results of a linear tangent steering law derived from calculus of variation solution that assumes downrange direction is free. Linear tangent steering commands consist of pitch and yaw [34], forming a system of non-linear equations. These equations are simultaneously solved to compute current attitude commands.

On other hand, SLS uses Powered Explicit Guidance (PEG), a highly evolved version of IGM. Development for PEG started during Apollo program. During that time, it was referred as Guidance theory and was baselined from IGM. After the Apollo program, PEG was referred to as Linear Tangent Guidance. During the shuttle program, it was renamed as PEG. PEG benefitted in its advancement through its usage in the Shuttle program, Ares program, and further advanced during SLS Block-1 and Block-1B development work [37]. The PEG solution is semi analytical predictor-corrector algorithm. It uses velocity to be gained by thrust as an independent variable (analogous to Newton-Raphson technique). With the assumption that co-states are orthogonal, it analytically solves for co-state at current time that arise from optimal control theory (i.e., λ and rate of change of λ). The algorithm then propagates state and co-state to predict velocity at cutoff. The difference between desired and predicted velocity at cutoff is used to form a miss velocity. PEG's corrector updates the velocity to be gained by miss velocity for the next iteration. Figure 16 shows a representative Monte Carlo distribution of PEG's predicted MECO time to demonstrate its robustness. Plot shows that PEG's predictor-corrector algorithm is able to predict cutoff time earlier in flight.

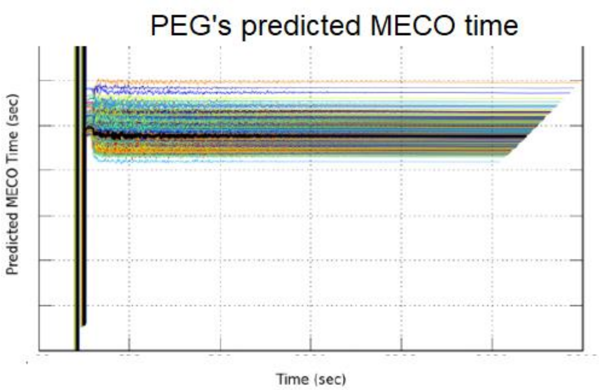

Fig. 16, PEG's predicted cutoff time for Block-1B

6.3 PEG Enhancement

As stated in previous section. Saturn V's IGM has paved a way for PEG algorithm development and enhancements. PEG allows for multiple phase modelling which allows for reliable time of flight even for burns with long burn arcs (up to 20 minutes). A lofting parameter was introduced which implicitly optimized the trajectory by adding additional pitch to comply with a free fall time requirement. Appropriate input were selected which maximized mass to orbit. Similar to Saturn V's engine out logic, PEG's engine out logic is able to do a combination of adjusting constant pitch and/or selecting Alternate MECO Target. Artemis I 1 Flight Software is able to load up to nine alternate targets. Thrust factor, ratio of estimated thrust to sensed thrust, provides more accurate acceleration to predictor. Flatness in Figure 14 curves are direct result of this algorithm. It also implicitly protects for stuck throttle and unknown engine failure cases. More targeting routine has been added that includes Shuttle derived Linear Terminal Velocity Terminal Constraint and targeting hyperbolic orbits in support of the Europa Clipper mission.

7. Conclusions

As described in this paper, many of the underlying mechanisms and engineering developments that went into the Saturn V launch vehicle are still used today. A common thread across this discipline is how increased computational capability, both onboard and on the ground, has allowed the current design team to expand analysis to include more effects, increase knowledge of the pre-flight environment, and be able to refine the capability. With this increased capability, the team has been able to make a more efficient vehicle that can account for and fly under more dispersed conditions to improve its capability. Additionally, the high level of fidelity in simulation and analysis provides confidence in the vehicle design and continued comparison to hardware and software testing reinforce this. With continued development and vehicle performance updates such an updated upper stage and booster improvements, the capability of this vehicle will continue to improve and provide enhanced cargo capacity to LEO and beyond.

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References

[1] J. M. Hanson, B. B. Beard, Applying Monte Carlo Simulation to Launch Vehicle Design and Requirements Verification, Journal of Spacecraft and Rockets, Vol. 49, No. 1, January-February.

[2] A.C. Mackey, R.D. Schwartz, "Apollo Experience Report - The Development Of Design-Loads Criteria, Methods, And Operational And Midboost Conditions Procedures For Prelaunch, Lift-Off, And Midboost Conditions", NASA TN D-7373, Washington, DC, August 1973.

[3] G. Wittenstein, J. Lowery, J. Urquhart, A Brief View of Rocket Science, Systems Engineering and Integration, The Flight Mechanics Perspective of Project Apollo, Unpublished Personal Notes.

[4] Anonymous, NASA Spinoff", NASA Technology Transfer Program, NASA, Washington, DC, 1996.

[5] R.W. Orloff, Apollo by the Numbers: A Statistical Reference, NASA SP-2000-4029, NASA History Division, Office of Policy and Plans, NASA Headquarters, Washington, DC, Oct. 2, 2000.

[6] J.F McCarthy, Jr., J.I. Dodds, R.S. Crowder, Development of the Apollo Launch Escape System, Journal of Spacecraft, Vol. 5, No. 8, August 1968.

[7] Anonymous, Saturn V Launch Vehicle Flight Evaluation Report-AS-506: Apollo 11 Mission, George C. Marshall Space Flight Center, Saturn Flight Evaluation Working Group, Sept. 20, 1969.

[8] National Aeronautics and Space Administration, NASA SP-34, Space Flight Handbooks, Vol. 2. "Lunar Flight Handbook, 1963.

[9] F.D. Cooper, A Method of Implementing Cutoff Conditions for Saturn V Lunar Mission out of Earth Parking Orbit Asuuming a Continuous Launch Window, NASA Technical Memorandum, N65-30650, May 25, 1965.

[10] R. Wheller, Apollo Lunar Landing Launch Window: The controlling factors and constraints, https://www.history.nasa.gov/afj/launchwindow/lw1.html, 2009.

[11] National Aeronautics and Space Administration, MSC-02680, Apollo 13 Mission Report, Sept 1970.

[12] R.M. O'Brien, J.P. Sheats, Saturn V Navigation Update, AIAA Guidance, Control, and Flight Mechanics Conference, Princeton, New Jersey, 1969 August 18-20, AIAA69-883.

[13] Smithsonian Institution, Collections Search Center, http://collections.si.edu/search/detail/edanmdm:nasm_A20090019000 , assessed 2019.09.16.

[14] NASA/MSFC,Technical Information Summary AS-501, Apollo Saturn V Flight Vehicle, September 14, 1967.

[15] B.J. O'Connor, A Descriptoin of the ST-124M Inertial Stabilized Platform and its Applcation to the Saturn V Launch Vehicle, Bendix Corporation, June 26, 1964.

[16] P.D. Groves, Principles of GNSS, inertial, and multisensor integrated navigation systems. Artech house, 2013.

[17] W.W. Chow, et al, The Ring Laser Gyro, Reviews of Modern Physics 57.1 (1985): 61.

[18] G.A Pavlath, Closed-loop Fiber Optic Gyros, Fiber Optic Gyros: 20th Anniversary Conference, Vol. 2837, International Society for Optics and Photonics, 1996.

[19] A. Lawrence, Modern inertial technology: navigation, guidance, and control, Springer Science & Business Media, 2012.

[20] H.E. Thomason, A General Description of the ST124-M Inertial Platform System, NASA/MSFC, NASA TN D-2983.

[21] J. Bellantoni, E. Koenke, Launch Pad Alignment of a Strapdown Navigator by the Kalman Filter, AIAA Control and Flight Dynamics Conference, 1968.

[22] N.S. Reddy, J. Murray, Transfer orbit stage gyrocompass alignment algorithm in a twist and sway environment for mars observer mission on commercial titan, 9th IEEE/AIAA/NASA Conference on Digital Avionics Systems,IEEE, 1990.

[23] K.W. Geohagan, W.P. Bernard, T.E. Oliver, et al, 6DOF Testing of the SLS Inertial Navigation Unit, Proceedings of the AAS Guidance and Control Conference, Breckenridge, CO., 2018.

[24] E. Burke, E. Rutkowski, E. Rutkowski, Vehicle based independent tracking system (VBITS): a small, modular, avionics suite for responsive launch vehicle and satellite applications, Proceedings of the 6th Responsive Space Conference, 2006.

[25] T.C. Moore, M. A. Carr, H. D. Friesen, Launch vehicle mission capability enhancement through global positioning system metric tracking, 2011 IEEE Aerospace Conference, 2011.

[26] J.E. Potter and M.C. Sumen, Thresholdless Redundancy Management with Arrays of Skewed Sensors, Integrity in Electronic Flight Control Systems. AGARD-AG-224. Jan. 1977.

[27] E. Anzalone, Optimization of Second Fault Detection Thresholds to Maximize Mission Probability of Success, Proceedings of the AAS Guidance and Control Conference, Breckenridge, CO., 2018.

[28] J. Bellantoni, E. Koenke, Launch Pad Alignment of a Strapdown Navigator by the Kalman Filter, AIAA Control and Flight Dynamics Conference, 1968.

[29] E.J. Anzalone, T.E. Oliver, Application of GPS to Enable Launch Vehicle Upper Stage Heliocentric Disposal, Proceedings of the 30th International Technical Meeting of The Satellite Division of the Institute of Navigation (ION GNSS+ 2017), Portland, Oregon, September 2017, pp. 1722-1734.

[30] L.B. Winternitz, W.A. Bamford, S.R. Price, New High-Altitude GPS Navigation Results from the Magnetospheric Multiscale Spacecraft and Simulations at Lunar Distances, Proceedings of the 30th International Technical Meeting of the Satellite Division of The Institute of Navigation (ION GNSS+ 2017), Portland, Oregon, September 2017, pp. 1114-1126.

[31] L. Winternitz et al., GPS-Based Autonomous Navigation Study for the Lunar Gateway, AAS 19-096, Proceedings of the AAS 42nd Annual Guidance and Control Conference, February 2019. [32] J. Bellantoni, E. Koenke, Launch Pad Alignment of a Strapdown Navigator by the Kalman Filter, AIAA Control and Flight Dynamics Conference, 1968.

[32] T.E. Oliver, T. Park, et al, Space Launch System Block 1B Preliminary Navigation System Design, Proceedings of the AAS Guidance and Control Conference, Breckenridge, CO., 2018.

[33] F. Sorrentino, et al., A compact atom interferometer for future space missions, Microgravity Science and Technology 22.4 (2010): 551-561.

[34] D. Martin and R. Sievers, Saturn V Guidance Navigation and Targeting, Journal of Spacecraft, VOL. 4, NO.7

[35] Anonymous, LVDC Equation Defining Document for the Saturn V Flight Programs, Part I and Part II, Revision E, IBM Space Systems Center, Huntsville, Alabama, Feb. 12, 1970.

[36] N. Ahmad, M. Hawkins and P. Von der Porten, CHANGO: A Software Tool for Boost Stage Guidance of the Space Launch System Exploration Mission 1, 2019 AAS/AIAA Astrodynamics Specialist Conference, August 11, 2019 - August 15, 2019, Portland, ME, AAS 18-270.

[37] P. Von der Porten, N. Ahmad, M. Hawkins, and T. Fill, Powered Explicit Guidance Modifications and Enhancements for Space Launch System Block-1 and Block-1B Vehicles, Proceedings of the AAS Guidance and Control Conference, Breckenridge, CO., AAS 18-136, 2018.

Appendix A. Additional References

Kabe, A.M., Spiekermann, C.E., Kim, M.C., and Lee, S.S., Refined Day-of-Launch Atmospheric Flight Loads Analysis Approach, Journal of Spacecraft and Rockets, Vol. 37, No. 4, July-August 2000.

Martin, Daniel T., and Sievers, Russell F., Saturn V Guidance, Navigation, and Targeting, Vol. 4, No. 7, July 1967.

Chandler, Doris C., and Smith, Isaac E., Development of the Iterative Guidance Mode with Its Application to Various Vehicles and Missions, Vol. 4, No. 7, July 1967.

Harrington, Brian E., Space Shuttle Day-of-Launch Trajectory Design Operations, United Space Alliance, Houston, TX, 2011.

Frank, M.P., Detailed Mission Planning Considerations and Constraints, MSC Internal Note No. 66-FM-134, Nov 1966.