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GUIDANCE AND NAVIGATION DESIGN FOR A MARTIAN SAMPLE RETURN ASCENT VEHICLE

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This paper focuses on the work being performed at the NASA Marshall Space Flight Center (MSFC) in support of Mars Ascent Vehicles (MAVs). Specifically, the analysis presented is in support of Martian sample return architectures. In order to assess vehicle sensitivities, a detailed simulation tool, (MAV Analysis Tool in Simscape) MANTIS, was implemented using the MATLAB/Simulink Simscape architecture. High fidelity navigation sensor models and guidance algorithms were included in order to facilitate sensor requirement development and flight algorithm selection. This work focuses on the performance of the integrated system and the coupling of navigation and guidance capabilities. The architecture trades are heavily dependent on the ascent flight profile chosen. This work assesses both open- and closed-loop guidance algorithms to capture their relative performance and the resulting requirements on sensor capability to support preliminary vehicle design. The analysis builds on previous work that focused on navigation performance for initialization and ascent flight of crewed vehicle. The results provide insight into the coupling between sensor requirements and ascent guidance approach. The analysis provides data to support requirements for hardware selection and testing. Additional discussion is also included focusing on other system constraints that affect hardware selection and operational constraints.

INTRODUCTION

Traveling across the Martian surface, autonomous rovers have provided a wealth of information and insight into the geological history of the planet. While this data is invaluable, the scientific return is limited by the specific constraints of the instruments. Having an experienced scientist in the field is expected to provide a much higher level of insight and observation. While human missions to Mars are in development, their expected mission timelines are several years away. Another approach when you can't bring a researcher to Mars is to bring Mars to the research. That is the goal of the Martian Sample Return missions[1]. By taking advantage of the robotic advancements demonstrated by the platforms such as the Mars Science Laboratory[2], engineers are developing autonomous systems capable of collecting samples to be returned to Earth. This paper focuses on one aspect of the missions, the guidance and navigation design of the vehicle being used to transport these samples from the Martian surface into orbit.

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The retrieval of Martian surface samples requires many coordinated projects: a landing vehicle to descend to the Martian surface, a rover to collect samples, a launch pad, an ascent vehicle to deliver the samples to a Martian collection orbit, and a retrieval vehicle to carry the orbiting samples to Earth. The high level architecture of this type of program is given in Figure 1. This investigation focuses only on the ascent vehicle guidance and navigation (the circled element). The current mission profile aims to autonomously launch a vehicle with an approximately 5 kg sample from the Martian surface to a 343 x 343 km, 25 degree inclination, circular orbit for retrieval. Any inaccuracy in the target orbit insertion must be accounted for by the orbital retrieval unit, impacting propulsive performance margin. Additionally, the accuracy of the final orbit insertion must be high enough to allow for the samples to reside in a stable orbit while awaiting retrieval. Given that the ascent vehicle must first be transported to Mars before the ascent launch begins, the ascent vehicle must fall within tight Gross Lift-Off Mass (GLOM) constraints, volumetric footprint constraints, in addition to the ascent performance requirements. These mission design and vehicle constraints emphasize the need for a vehicle with high propulsive efficiency and stable, highly accurate guidance, navigation, and control systems.

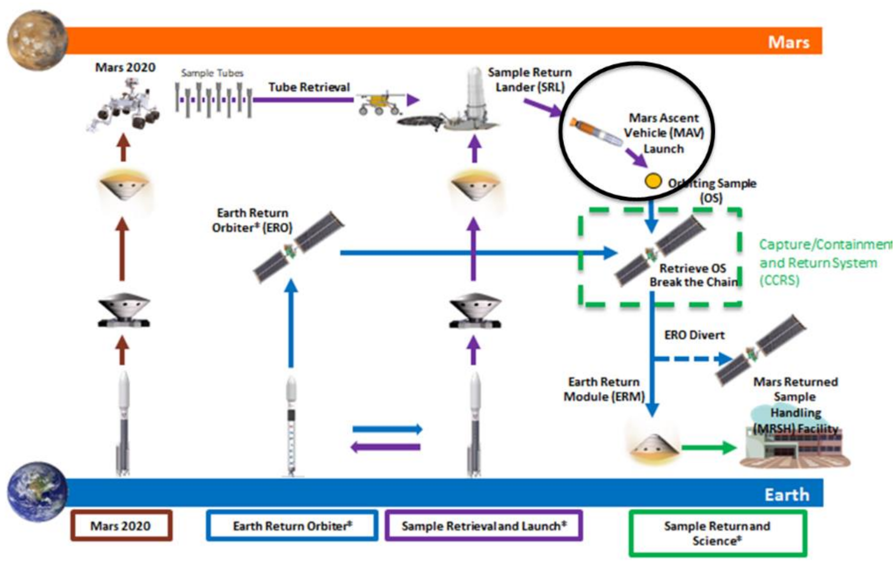

Figure 1: Mission Concept of Operations (NASA/JPL)

To accomplish the intended mission goal, two vehicle designs are considered and shown in Figure 2: a hybrid motor design and a two-stage solid motor design. The hybrid design is a single stage, two-burn, restartable design employing thrust vectoring and RCS for active control. The solid motor design is a two-stage launch vehicle employing both thrust vectoring and RCS for active control, which jettisons first stage dry mass before the second burn. The vehicle designs are tightly coupled to the flight trajectories, mission constraints, and desired orbit target. Specifics of the vehicle design and sizing, including delta-v splits, burn times, targeted thrusts, pitch rates, and launch azimuth, are integrated into the trajectory design. This approach achieves a vehicle solution and trajectory tailored to each configuration. Maximizing performance margin and spending that margin to improve orbit accuracy and robustness is of particular concern.

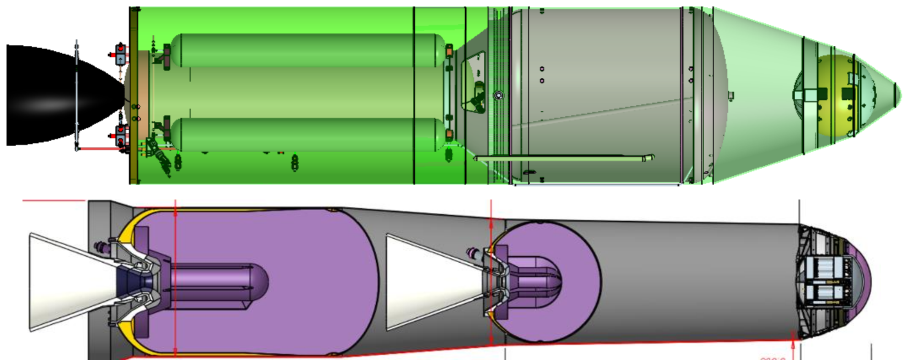

Figure 2: Notional Vehicle Concepts (Top Hybrid, Bottom Solid)

The trajectory design is coupled to vehicle design and optimized to maximize performance margin. Additional performance margin can translate into accurate and robust orbit insertion by allowing excess delta-v to account for uncertainties that drive the executed trajectory off-optimal. Both vehicles employ a standard two-burn profile to achieve orbit: first burn, coast to apoapsis, jettison staging mass (if applicable), then second burn to raise periapsis to the target. The launch azimuth is aligned to achieve the target inclination, minimizing the need for yaw control. Both vehicles employ non-vertical elevation, yielding a two dimensional problem. Initial trajectory work requires a linear, continuous pitch rate per phase within targeted rate limits to enable a realizable control system design. Trajectories were independently simulated and optimized in 3DOF using both NASA Glenn Research Center's OTIS (Optimal Trajectories by Implicit Simulation)[3], and NASA Langley Research Center's POST2 (Program to Optimize Simulated Trajectories)[4]. The simulation results from both OTIS and POST for each vehicle were then compared to support modeling accuracy and optimality conclusions. These 3DOF results were then fed into other tools for further 6DOF vehicle analysis.

To support detailed vehicle requirements development and analysis, the MAV Analysis Tool in Simscape (MAnTiS) was developed. This is created using Simscape Multibody[5]. Simscape Multibody is a visual-based simulation tool that uses a combination of Simulink blocks and MATLAB functions. This creates a very familiar environment for anyone with a control systems engineering background and mild MATLAB coding experience. The Simulink model itself is built into separate modules with a 6DOF kernel at the center. The kernel of the simulation is a rigid body block that simulates a mass element with known inertia. The standard states in the simulation are position and attitude while attitude is parameterized using quaternions. The different submodules built around the kernel are shown in Figure 3: the plant model, sensor dynamics, a navigation and estimation routine, a guidance block, a controller block and actuator dynamics. The plant contains the following models: aerodynamics, thrust, mass, standard atmosphere and gravity models. The second dynamics read in the perfect state vector signal of acceleration and angular velocity. Noise is added to pollute the signal and is then sent off to the navigation and estimation routine to estimate the full state vector from the sensors. The guidance block uses a two stage guidance law based on an open loop chi table to orient the spacecraft and a boundary value solution for the second stage. The commanded signals are then sent to the control block routine to compute Thrust Vector Control (TVC) angles as well as Reaction Control System (RCS) pulse signals. TVC is used to orient the rocket during stage 1 and stage 2 while RCS is used when the main thruster is off. Actuator dynamics are modeled using simple first order filters to mimic servo and hydraulic actuation of the TVC engines as well as any time constant with the pulses from RCS. These signals are then sent to the plant model to compute force applied which includes forces from aerodynamics, gravity and thrust. The simulation requires in-puts in all aspects of the flight including gravity constants, aerodynamic coefficients, mass and inertia properties, sensor noise and many others. The result is over 65 state values including the standard 13 states for a rigid body system. This framework is planned to be used for detailed simulation, enabling assessment of the integrated vehicle with full GNC implementations.

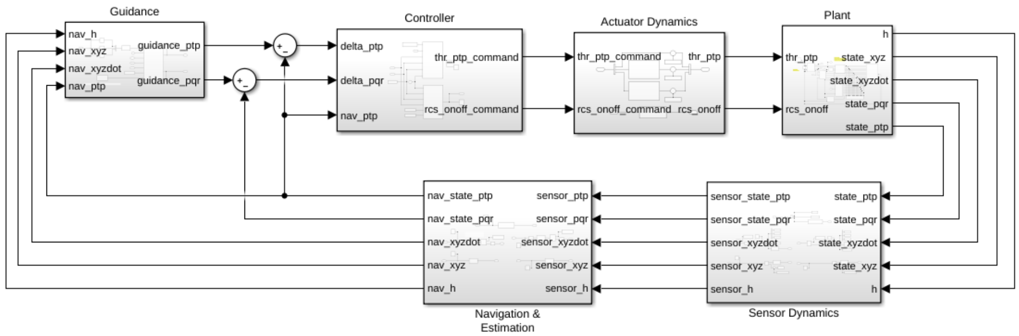

Figure 3: MANTIS Framework

GUIDANCE ALGORITHM SELECTION AND DESIGN

A flexible guidance algorithm design is needed to support both vehicle concepts. The guidance algorithm applies to both a hybrid motor, whose thrust can be terminated on command, and a solid motor, which burns all propellant present and effectively cannot be terminated early. This situation presents a unique challenge. In order to design a vehicle with performance margin available to allow robust, accurate orbit insertion, the guidance algorithm must accommodate the margin in either form: terminated on command, or adjusting the pitch commands to ensure all performance margin is consumed just as the vehicle reaches the guidance target. In addition, the size constraints on the vehicle push toward a simple algorithm capable of being run on low-mass, robust hardware. Two approaches to this problem are open- and close-loop guidance.

Open-loop guidance for both vehicle designs relies on a pre-loaded trajectory. A 3DOF simulation is run in which the 3DOF vehicle achieves the target orbit and provides a desired fly-to trajectory. The roll, pitch, and yaw angles from that simulation are then extracted and placed into lookup tables as a function of either time or altitude. The 6DOF vehicle then pulls the commands directly from these tables. It is preferred to command pitch as a function of altitude, as this provides an inherent feedback mechanism not present in a time-based open-loop system. If the vehicle is overshooting the target trajectory and is too high, an altitude-based open-loop guidance will command a lower pitch angle. This guidance algorithm is simple but cannot adjust for dispersions in launch angle, mass, thrust, winds, etc.

Closed-loop guidance strategies revolve around simplicity and energy management. Two guidance algorithms are presented: Lambert guidance, and General Energy Management(GEM), as explained in Zarchan[6]. Lambert guidance relies on an effective, efficient lambert solver, and quality targets. The current algorithm employs the techniques of Izzo[7], Gooding[8], and others[9] as implemented by Oldenhuis[10], which was tested against known cases in Vallado[11]. The efficacy of the Lambert guidance technique improves as a larger portion of the flight to the target is unpowered. Thus, the first burn targets the apogee of the 3DOF optimal trajectory, while the second burn targets a point far along the final circular orbit, again derived from the 3DOF optimal case. GEM employs an offset to the lambert solution proportional to the excess delta-v capability remaining in the motor beyond that predicted by lambert. This is in effect a non-optimal trajectory that steers off-optimal to waste the excess propulsive energy in the motor. The delta-v capability remaining in the motor is tracked as integrated acceleration and updated with each time step. This solution allows the inherent delta-v margin in the solid vehicle to be spent, while the hybrid vehicle can either burn excess propellant by using GEM, or simply follow Lambert guidance then terminate the burn when the target is reached.

Figure 4 and Figure 5 below show theoretical applications of the closed-loop guidance. The right shows the GEM guidance calculating new pitch angles during the first burn as the vehicle trajectory advances. The left plot shows the GEM guidance commanding the off-optimal guidance path to consume excess delta-v capability. Once the excess delta-v is consumed, the simulation switches to lambert guidance, which mirrors the nominal trajectory.

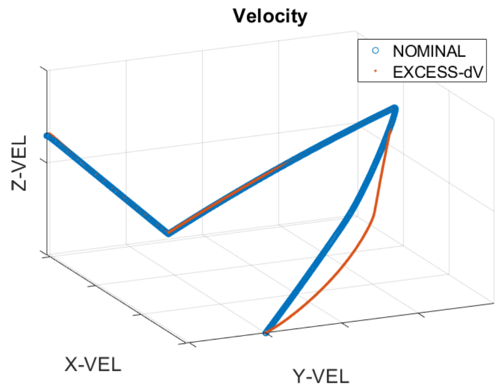

Figure 4: Velocity Profile

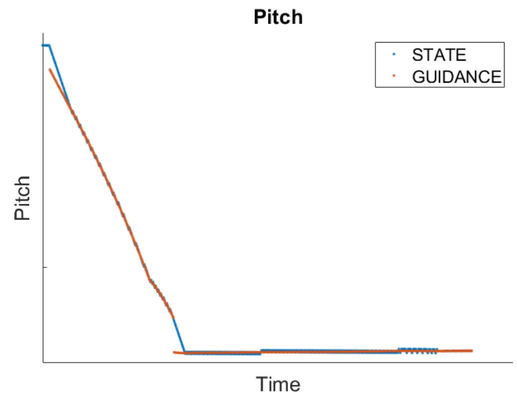

Figure 5: Pitch Profile

NAVIGATION ARCHITECTURE

Regardless of the Guidance algorithm chosen, vehicle state knowledge is required in order to ensure the vehicle matches the desired motion. For example, the vehicle can only maintain an attitude profile within its own knowledge of attitude. Due to the operational constraints, this mission places strongly conflicting requirements on the vehicle's navigation system above and beyond performance alone. To support the capture of the sample canister, the vehicle must insert the payload as accurately as possible into the desired orbit in order to help support coordination with the capture and return spacecraft. Any errors on insertion orbit parameters will widely increase the search area and the possible location of the payload will continue to grow as the expected perturbing forces will cause the insertion orbit to decay, due to effects such as drag, gravity gradient, and higher order gravity effects. Similarly, due to the autonomous operation, the system must be extremely reliable. These requirements would typically push for a navigation grade internally (or externally redundant) inertial measurement unit, such as that used for human launch vehicles. This grade of instrument is also needed due to the lack of supporting infrastructure that could be used for external aiding.

The push to use a highly accurate IMU directly comes into conflict with another mission constraint: the entire launch vehicle and supporting launch structure must all be transported to Mars. Due to the limited volume and mass available and the complexities in landing large masses on the Martian surface, the design tends towards a small, limited mass vehicle. For this case, the solution tends towards smaller MEMS sensors with near-tactical levels of performance. While these are accurate enough to support vehicle control, their navigation perspective is marginal. This directly conflicts with the need for a highly accurate navigation solution. As such, the two must be traded off to select a sensor suite that is able to minimize insertion errors while also minimizing volume and mass.

The approach to guidance chosen also influences sensor selection as the tradeoff between knowledge and robustness. This trade is in having confidence that a vehicle will fly a pre-determined trajectory and demonstrate that the trajectory is insensitive to expected environmental effects such as winds and gust dynamics and uncertainties in launch initial conditions, i.e. angle of the launch rail and pointing of the vehicle. The choice of command and matching look-up index for open-loop control can also constrain the sensor specifications. For example, if the system designer selects time and commanded attitude, the architecture must include high grade attitude sensors and have accurate initial attitude knowledge in order to properly execute the commands. Using other metrics such as commanded angular rate as a function of time increase the effect that uncertainties in the launch azimuth and heading angles have on insertion states. Other internal studies have shown altitude to be a preferred look-up for commanded attitude. In this case, both high accuracy altitude and attitude knowledge is required to match the design trajectory.

The other option is to design the system to be able to accurately observe the as-flown trajectory in real-time and use closed-loop guidance algorithms to guide the vehicle to meet the desired insertion state. Here, the navigation architecture includes the capability to determine its attitude prior to launch and operates much more like a conventional Earth-based rocket. Both options and their effects on sensor requirements are studied in detail below. The following sections also provide a description of the specific sensors under consideration, modeling assumptions, performance, and identification of key sensitivities.

While the integrated GNC simulation was under development, additional tools enabled the analysis of navigation-only performance along the reference trajectory. Previously developed tools for Martian human ascent vehicles[12] served as the baseline for original analysis and sensitivity assessments. These tools include full inertial sensor error models, inertial navigation algorithms, and mathematical utilities. For this analysis, the baseline trajectories provided reference position, velocity and attitude information. With these inputs, the simulation tools were used to develop a full 6 degree of freedom trajectory. This allows for demonstration of attitude integration and traceability from attitude to final insertion errors. While this toolbase does not include the ability to disperse environment factors or run a close-loop simulation, it does enable assessment of the contribution of navigation errors and system sensitivity to inform sensor selections and trade studies. This simulation framework was used in the following analysis scenarios.

Sensitivity to Initial Errors

Regardless of the guidance approach chosen, the uncertainties at launch in both position and velocity can have a very large effect on the state. In order to focus on this sensitivity, a parametric study was performed assessing insertion accuracy as a function of initial position and attitude errors, assuming a perfect sensor suite. This allows for assessment of only the initial errors and navigation algorithms, decoupling sensor-specific terms. This data is also used to define requirements on initialization systems, affecting vehicle mounting accuracy, original landing site determination, and attitude initialization approach (whether through gyrocompassing[13], transfer alignment[14], or pre-flight measurement[15]). For this study, the combination of all possible starting conditions was each modeled as a 1-sigma normal distribution and applied to the starting navigation state. The results of a series of 500 case Monte Carlo using the Solid Trajectory provide insight into the uncertainties at orbital insertion. For each simulation set, the initial positon and attitude errors were dispersed based on the input parameters in a 2D parametric exploration. Uncertainties at insertion conditions was captured for each simulation and integration into an integrated data set for post-processing.

The results of the sensitivity analysis are shown below in Figure 6. The contours in the plots below represent 1-sigma uncertainty values and are plotted using a logarithmic scale in order to show the orders of magnitude increase in position and velocity errors as a function of initial position (x) and attitude (y) errors. As expected from previous studies[12], the initial attitude uncertainty has a large effect on position and velocity errors. Initialization errors above .1 degrees are the primary drivers of insertion uncertainty. Only for smaller attitude errors, do the larger initial position errors begin to have an effect. This effect is clearer on position than velocity errors. Errors in gravity estimation based on position uncertainties are the primary driver of the position on velocity states, whereas attitude errors directly cause thrust to be integrated into the wrong direction affecting both position and velocity through inertial navigation.

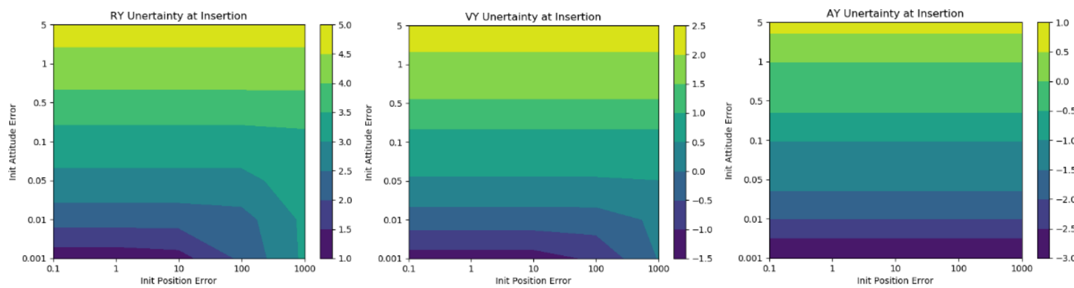

Figure 6. Sensitivity of Initial Errors on Inertial Y Position (L), Velocity (M), and Attitude (R)

Sensitivity Analysis to Sensor Terms

With the variations in performance between the potential sensor options, the team performed a sensitivity analysis to identify the key sensor metrics driving performance. In order to provide insight into the differences among quality, the assessment focused on the MEMS device in comparisons to a high-nav grade device (such as ring laser gyroscopes). The analysis utilized a Monte Carlo-based statistical analysis to determine key metrics in performance, defined here as the uncertainty in each parameter at insertion. This approach assumes that the total variance due to all sensor terms can be captured as the linear combination of variance for each individual parameter. The results of a series of Monte Carlos allows for calculation of primary drivers. Each set consisted of between 10-15 250 case simulations, based on the number of inputs. While the limited number of runs does cause an increase in the overall statistical errors[16], it still provides useful insight into the primary sensor error terms. For these cases, the initial errors are assumed to be 0 in terms of position, velocity, and attitude. This assumption is used to decouple pre-flight dynamics (such as gyrocompassing or transfer alignments) from errors induced over the ascent flight. The sensitivities shown here would couple into and magnify the sensitivities to initial position and velocity errors given above.

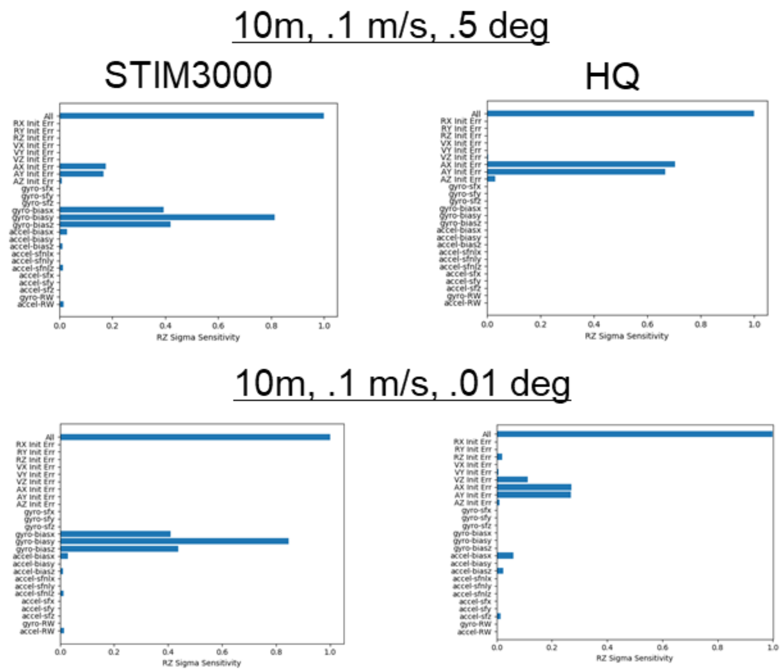

Figure 7: Sensitivity of Inertial Z Position Errors

As seen above, for the lower grade sensor, the noise terms dominate the sensor performance, while for the higher-grade instruments, individual terms such as random walk (a form of noise) are dominant, as well as scale factor errors due to the high dynamics of the ascent trajectory. This provides insight that once the inherent noise levels in the sensor are reduced other terms start to have a larger impact, and can provide focus for technology development efforts.

Navigation Performance for Baseline Trajectories

In order to assess the navigation performance over the ascent mission, the team used a Monte Carlo approach to simulation and modeling to capture knowledge uncertainty at the insertion point. For each of the sensors, the results of 250 simulations with dispersed error terms was used. For these cases, a conservative set of initial conditions was used to help inform requirements on position and velocity state knowledge. The initial position error is 10 m, velocity is .1 m/s, and .1 degrees on attitude (Note: these are applied per-axis, not total).

The results here are captured in terms of orbital elements. The variances on these are given below in Figure 8. The plots show uncertainty in terms of eccentricity, inclination, and semi-major axis. These were selected to provide insight into in- and out-of-plane errors in regards to the target orbit. As can be seen, the differentiation between the sensors grades is clear. Moving from a tactical MEMS device to a tactical non-MEMS device shows a dramatic improvement. Further gains are seen in moving from tactical to navigation grade devices. In order to provide a sensor selection, analysis and negotiation between partners will be needed to understand mission-wide operational constraints.

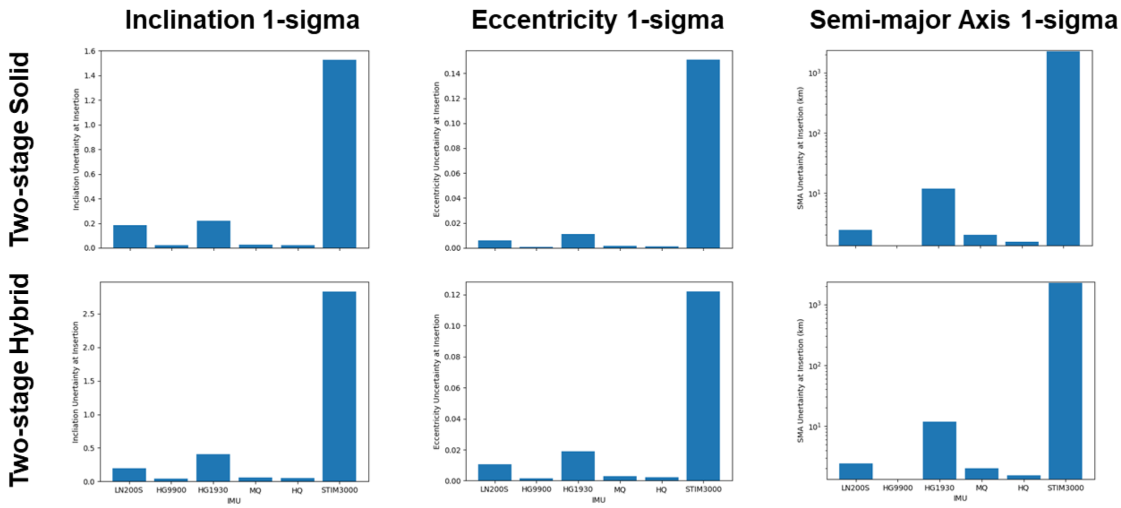

Figure 8: Orbital Insertion Performance

INTEGRATED PERFORMANCE

The coupling of Guidance algorithm and navigation sensor selection can be seen in clear detail through assessment of the Open-loop guidance implementation for the solid motor vehicle design. For this design, the optimal trajectory was used as a look-up table for commanding attitude as a function of time. The initial attitude and position errors were chosen based on estimates of what could be provided by an external system, i.e. the landers inertial measurement unit. This provides a baseline for best case insertion accuracy possible. The results of the study are summarized in Table 1. Each row captures the uncertainty in orbital elements for a given IMU in terms of insertion altitude, total velocity, apogee and perigee altitudes of the final orbit, and inclination. The results below show the coupling to attitude uncertainty. As the sensor grade decreases, the errors quickly grow. These orbital errors are taken at the end of the insertion burn. It is quite likely that upon propagation of the achieved states, some of the cases do not actually achieve orbit. With the solid burn, the actual maneuver are of minimal time duration, but the initial pointing drives the accuracy. Most of these cases showed similar behavior over initial descent (independent of navigation errors). Buildup of attitude errors over the long coast to second motor ignition drive errors in insertion by pointing the engine off optimal. Additionally, the benefit of a higher accuracy initial attitude is shown in the last row. For this case, the initial attitude is determined by the capability of the unit to gyrocompass rather than external alignment. In this scenario, the improvement in initial attitude knowledge greatly reduces uncertainty at insertion. Integrated results of closed loop guidance and navigation were not available at the time of publication, but for those scenarios, the navigation requirements become even more stringent and the ability of lower grade sensors to complete the mission will be greatly reduced.

Table 1. 1-sigma Insertion Uncertainty with 0.1 Degree Initial Attitude Uncertainty

	ALT (m)	Vmag (m/s)	Ha (m)	Hp (m)	Inc (deg)
Init. Error Only	940	2.4	1500	1900	0.08
HQ	930	2.25	1500	1800	0.08
LN200	1300	3.1	2000	2500	0.11
STIM3000	2000	4.7	3100	3800	0.17

HQ w/ .01 Deg	93	0.22	150	180	0.01

CONCLUSIONS AND FORWARD WORK

This research focuses on the development of the Guidance and Navigation subsystem for a Martian Ascent Vehicle for a sample return mission. The operational constraints of the system create a challenging engineering problem with strongly competing performance metrics. In addition to high performance, low mass, and high reliability, the system must also be able to execute a fully autonomous launch sequence with minimal human interaction. This platform will be a demonstration ground of a bevy of technology developments across many areas including the Guidance and Navigation domains. The analysis here demonstrates the applications of launch vehicle approaches to this mission to capture notional performance and inform hardware selection and flight operation. This research does not provide a clear, easy answer due to the constraints, with the clear trade between mass and performance. Discussions and interactions between design engineers, program and project staff, and research scientists is necessary to inform these design decisions in ensuring a successful mission.

Technology Pull for Small Navigation Grade Sensors

The current availability of off the shelf inertial measurements units limits the options for this mission and reinforces the trade between SWAP and performance, forcing the vehicle to accept large insertion uncertainty in order to meet the desired integration constraints. This will directly impact and complicate the mission of the rendezvous mission being developed to retrieve the payload canister and return it to Earth. Navigation systems do exist that will help to reduce this uncertainty, but do not fit within the available mass or volume. Figure 9 shows the insertion accuracy as a function of system statistics from published documentation.

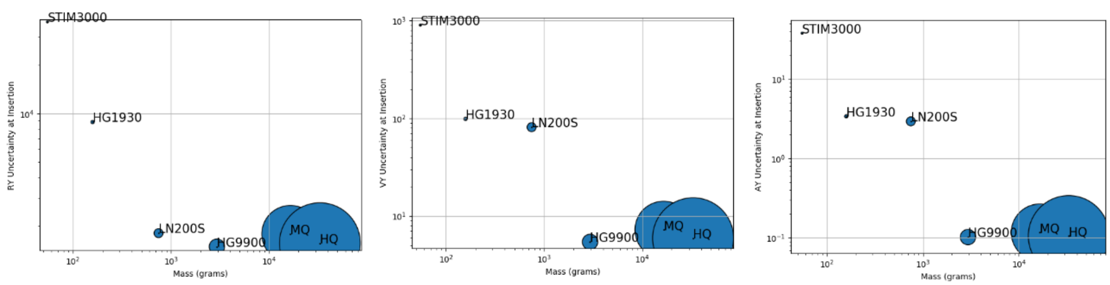

Figure 9: Sensor Performance as a Function of Sensor Mass (Size = Relative Volume) (10 m, 1 m/s, 0.1 degree initial uncertainty in position, velocity, attitude)

These results also identify a large gap between small SWAP, low grade sensors, and higher grade but large SWAP. The continued development and demonstration of sensors platforms will fill this gap and enable a drastic jump in performance both in terms of navigation accuracy as well as size. The remaining lead times, large cost, and development risks associated with projects make them infeasible for integration into the existing design to meet design schedule. Future missions, including human landers[17], will be poised to take advantage of these advances and be able to integrate high performing robust solutions. The continued development of these platforms is an enabling technology for continued autonomous exploration of the solar system.

Initialization Options

This paper lays out the importance of the initial state determination in regards to the ability to correctly insert into the desired orbit. It does not provide very much detail into how that can actually be accomplished. Similar efforts have laid out the ability of these grades of sensors to gyrocompass. Initial attitude errors of .1 degrees are used in the performance assessment, but some of the tactical grade units are not able to achieve this through gyrcompassing. Thus, external sensor systems will be needed to support initialization processing. This requires coordination and collaboration between the vehicle and lander designers. Options exist for using the lander's higher grade IMU for a transfer alignment, but the algorithm has not been tested yet. Future work will focus on assessing this capability, as the probability of using an existing navigation-grade IMU is low due to vehicle mass and volume constraints. This is primarily a driver for attitude knowledge, existing methods can be utilized to determine the landed position, and the results above show a low sensitivity to this.

Additional Potential Measurement Sources

Another option for improving the initial state knowledge is to include a star tracker on the lander or vehicle. This would allow the reduction of attitude-induced errors to a minimal level. Star trackers able to measure attitude on the level of multiple arcseconds are widely available and in use on other landers. In this case, the primary uncertainty factor would be in the mounting alignment between the star tracker and the launch vehicle's IMU. This would put stringent requirements on mounting knowledge and stability of both the launch vehicle to the lander and from the launch vehicle to the IMU. Continuing analysis will assess the potential for using this sensor and understanding the level of structural stability over the course of flight from manufacturing, transport, ascent, cruise, and Martian descent to understand the feasibility of transferring the alignment between frames or accepting a lower quality gyrocompassing or other external measurement of initial attitude.

Similarly, external sensors can also be used to support in-flight navigation. This is primarily to improve the navigated position knowledge. For example, other architectures have proposed using in-orbit assets to perform radiometric ranging to support navigation support for an incoming lander[18]. This same approach can be utilized on ascent from the surface using an orbital or ground navigation beacon. Multiple approaches exist for implementing a ranging capability between assets. By ranging to a known asset, the vehicle will be able to utilize lower grade navigation instruments and have an improved insertion capability. The primary caveat to this is maintaining visibility over the trajectory. While a lander-based beacon, would provide support early in the mission limiting error growth during initial ascent, as the vehicle flies over the horizon, it is expected the vehicle would lose signal. Similarly, an orbiting asset could be used to provide ranging support, but this would require accurate ephemeris knowledge (measured near to the ascent vehicle launch time) and coordination between projects in order to ensure operational coverage. The use of beacons is being considered and assessed as a possible path forward and risk reduction activity to improve the performance of the ascent vehicle and to understand interface and integration requirements.

Forward Work

As the design continues to mature, the team will be heavily involved in sensor selection and algorithm trades. Immediate forward work for the team includes the integration of the two baseline vehicle concepts into the high fidelity 6DOF simulation tools. This work will support upcoming reviews in 2019 and selection of flight hardware elements. The team will continue to assess guidance and navigation options as part of continued work to ensure a robust capable system.

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