DISPERSION REDUCTION FOR A SOUNDING ROCKET SCRAMJET FLIGHT EXPERIMENT

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A low cost method for reducing the dispersion in the trajectory of an unguided, spin-stabilized, sounding rocket is developed and presented. The method is particularly suited to scramjet flight experimentation because the approach increases the likelihood of meeting Mach number and dynamic pressure objectives. The paper discusses the design and model of the scramjet payload, two-stage launch vehicle, and nominal trajectory as well as a Monte Carlo analysis to quantify the likelihood of a successful scramjet test. Using the results of this analysis, a method is presented for reducing the dispersion in freestream conditions during the scramjet test window. The dispersion reduction is accomplished by modifying the time delay between the burnout of the first stage booster and the ignition of the second stage based on the vehicle state measured during the interstage coast. This method increases the likelihood of a successful test from 71% to 99% without adversely affecting range safety. Since the design and implementation of a vehicle guided control system is not required, this method is relatively inexpensive, making its use highly desirable for low cost scramjet flight experimentation.

Introduction

The development of hypersonic airbreathing engines has traditionally relied heavily on ground-based testing methods. While these tests are very useful, many suffer from test gas vitiation, poor flow quality, poorly matched boundary conditions, or short flow duration. With this in mind, the Short Duration Propulsion Test and Evaluation (SDPTE(Hy-V)) Program[1] has been developed with the aim of examining the influence of ground test facilities on scramjet performance and operation. As part of this program, a flight test of two dual-mode scramjet flowpaths will be conducted in order to generate a database for comparison with ground test data. This comparison is considered particularly valuable because flights performed in atmospheric air do not suffer from the limitations of ground-based facilities described above. Testing in three facilities is planned in order to isolate the effects of test flow duration and facility vitiation. One test article will be tested in the University of Virginia's supersonic combustion tunnel, which is an electrically heated, direct connect facility and can run continuously.[2] A freejet model will be tested in ATK-GASL's Test Bay IV blowdown tunnel. This facility can operate in both vitiated and non-vitiated modes below Mach 5 and has a run-time up to two minutes depending on the flow rate.[3] A second freejet model will also be tested in NASA's HyPULSE facility. This facility is shock heated and gives test times on the order of 10 milliseconds for the test conditions considered here.[4] By comparing the ground test data across facilities and with that of flight, more accurate estimates of flight performance can be made in the future by taking into account the effects of ground test flow duration and vitiation.

The flight experiment for the SDPTE(Hy-V) Program will be conducted in a captive boost mode using an unguided, spin-stabilized, Terrier Improved-Orion sounding rocket that is launched from the NASA Wallops Flight Facility. This vehicle will accelerate the scramjet to the required test conditions at which point the flowpaths will be ignited and combustion data will be recorded. Due to the inherent uncertainties associated with the launch of such a rocket, the actual trajectory that the rocket will follow can only be estimated to within certain degrees of confidence. These uncertainties arise because of differences in modeled and actual day of flight winds, launch dynamics, rocket burn rate and thrust, thrust misalignment, weight, and inaccurate estimates of vehicle drag, amongst others. However, the level of dispersion from the nominal predicted trajectory can be estimated using stochastic computer simulations. This dispersion is very important because the flight Mach number and altitude affect the scramjet operating pressures and temperatures. While dispersion in freestream conditions may not be important for many sounding rocket flights, such as those for astronomical or terrestrial observations, an unsuccessful scramjet test can result if flow quantities deviate significantly from what is expected. It is also imperative that the conditions in flight match those seen in ground testing such that meaningful comparisons can be made.

There have been several scramjet tests in recent history that adopted the use of unguided, spin-stabilized sounding rockets, many of which experienced varying degrees of dispersion in test conditions. For example, HyShot was a flight test program aimed at demonstrating scramjet flight and validating the use of short duration ground test facilities for supersonic combustion studies above a freestream Mach number of 7.5. The maximum test Mach number for the HyShot 2 experiment exceeded the pre-flight prediction by 0.4.[5] Fortunately, the scramjet design was robust enough to accommodate this dispersion and the flight was successful. However, other flight tests have seen greater trajectory dispersion. FASTT was a flight program aimed at demonstrating the operation of a hydrocarbon-fueled scramjet-powered vehicle.[6] Two unpowered surrogate flights (SPV1 and SPV2) were flown prior to a powered scramjet flight (FFV1). The insertion Mach number for the SPV1 surrogate flight, at the beginning of the test time, was 0.81 less than the Mach 5.64 that was expected. Insertion altitude was also 11,300 feet (3,444 m) less than the 63,800 feet (19,446 m) expected which corresponds to a 72% increase in static pressure. For the engine test flight, FFV1, the insertion altitude was 13.5% lower than expected and Mach number was 1.2% higher than expected. This corresponds to a 63% deviation in dynamic pressure from the expected value.[6] Again, this program was successful, partially in this case due to the use of automated on-board fuel control. However, the variation in pressure and Mach number seen in these Terrier Improved-Orion sounding rocket flights is unacceptable for the SDPTE(Hy-V) scramjet design for which strict Mach number and dynamic pressure requirements have been developed.

The success of the single planned SDPTE(Hy-V) flight is critically dependent upon the ability of the launch vehicle to pass near the design test condition. A simple blowdown fuel system designed to deliver a nearly constant fuel flow rate will be used to reduce program costs. For a predetermined fuel mass flow rate, if the air mass capture is greater than expected, the resulting equivalence ratio will be lower than expected and a lean blow-out can occur.[7] Conversely, if the air mass capture is lower than expected, the resulting equivalence ratio will be higher than expected and engine unstart can occur.[8] It is also important that the freestream test conditions are within those able to be simulated by the ground test facilities so that meaningful conclusions regarding the effects of the facility can be made from the resulting data sets.

Multiple options exist for reducing dispersion in the trajectories of sounding rockets. Guided rocket systems utilize thrust vectoring and/or actively controlled aerodynamic surfaces to change the direction of travel. Control surfaces or attitude control jets can also be added to the payload to influence the flight of the vehicle. Any active control of the vehicle's flight, however, is accompanied by a significant increase in project complexity and cost. The method presented here allows the vehicle to satisfy the scramjet test Mach number and dynamic pressure requirements while preserving the economy of utilizing an unguided sounding rocket to accelerate the scramjet to operating conditions. This is achieved by modifying the interstage time delay of the sounding rocket during the flight based on the level of trajectory dispersion experienced up until that point. Such a technique is relatively simple and cost effective to implement and has not been previously reported in the literature.

This paper begins by briefly describing the SDPTE(Hy-V) payload, launch vehicle configuration, the nominal trajectory, and its development using GEM[9], a NASA six degree of freedom trajectory simulation program. A Monte Carlo analysis was performed about the nominal trajectory to quantify the likely dispersion in metrics of interest for a hypersonic airbreathing engine test and to provide a basis for this dispersion reduction technique. For each Monte-Carlo trajectory, the optimal second stage ignition time was found such that the vehicle passes through the target test Mach number and dynamic pressure concurrently. This optimal second stage ignition time and corresponding test time was related to the vehicle's Mach number and altitude at a time during the interstage coast. In order to test the method, a second Monte Carlo analysis was then performed, this time using the relationships derived *a priori* for the optimal second stage ignition time and test time. This resulted in a significant increase in the likelihood of a successful scramjet test. The results of this analysis are presented with a discussion of their implications for a hypersonic airbreathing engine test.

Vehicle Design and Nominal Trajectory

The SDPTE(Hy-V) payload was designed to house two instrumented scramjet flowpaths oriented on opposing sides of a wedge forebody, as well as all supporting subsystems. A model of the launch vehicle was developed by building on aerodynamic data from the FASTT scramjet test, which utilized a similar payload design and suppressed ballistic trajectory. [6] Booster data typically used by the NASA Sounding Rocket Operations Contract (NSROC) was also used.

The vehicle consists of the four sections detailed in Figure 1: the Terrier Mk. 70 first stage booster, the Improved Orion second stage booster, the payload, and the shroud. The latter protects the self-starting inlet from the high aerodynamic and thermal loads of launch. The payload houses two opposing scramjet flowpaths, with slightly varying geometry[1], as well as avionics, telemetry, and a hydrogen fuel delivery system. Since the drag on the shrouded payload was shown to be nearly identical to that on the exposed scramjet inlet, the vehicle model assumes a shrouded payload throughout the entire trajectory.

The nominal trajectory was designed by considering the scramjet inlet such that conditions at the entrance to the scramjet isolator are similar to those seen in ground testing. For this reason, a suppressed ballistic trajectory was chosen with a launch elevation angle near 50 degrees. A more common sounding rocket objective, such as one for astronomical observations or atmospheric measurements, is to propel a payload to a desired altitude. Such a flight would utilize a launch elevation angle much closer to 90 degrees. Figure 2(a) shows Mach number and altitude as functions of time for the nominal trajectory, which satisfies the test conditions of Mach 5 and a dynamic pressure of 1,500 psf (71.82 kPa). This nominal trajectory also satisfies the secondary trajectory design objective which is an approximately constant dynamic pressure with respect to time (dq/dt = 0) when the primary test conditions are met. Figure 2(b) shows the predicted Mach number and dynamic pressure near the test window. After first stage burnout, the first stage booster is separated from the vehicle. The vehicle coasts for approximately 30 seconds while altitude increases and elevation angle decreases. The second stage booster then ignites. During the second-stage burn, the vehicle approaches the required test conditions. The shroud is jettisoned, exposing the scramjet inlets and allowing atmospheric air to flow through the scramjet flowpaths. Combustion is initiated while temperature and pressure data are continuously relayed back to the ground for subsequent analysis. The primary experiment ends at the time of the second stage burnout. A secondary experiment then takes place as both Mach number and dynamic pressure decay. The secondary experiment, which was required to accommodate the lower operating pressures of the University of Virginia's supersonic combustion facility, concludes when the dynamic pressure reaches 1,000 psf (47.88 kPa). The trajectory was chosen with the aim of minimizing dispersion in freestream conditions during the test window as well as integrated thermal loads on the vehicle. Since dispersion in Mach number, altitude, and thus dynamic pressure increases throughout the flight, positioning the test window as early as possible minimizes the dispersion in both Mach number and dynamic pressure during the expected test window. Positioning the test window as early as possible in the flight also minimizes integrated aerodynamic heating, which increases monotonically with time.

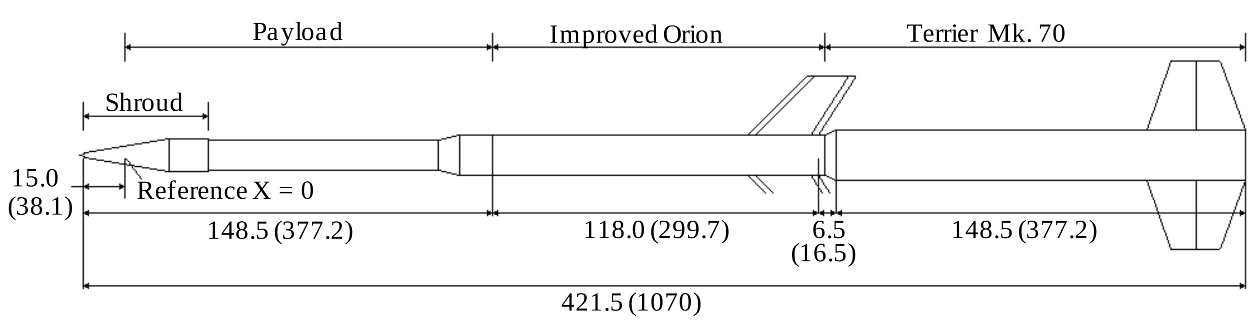
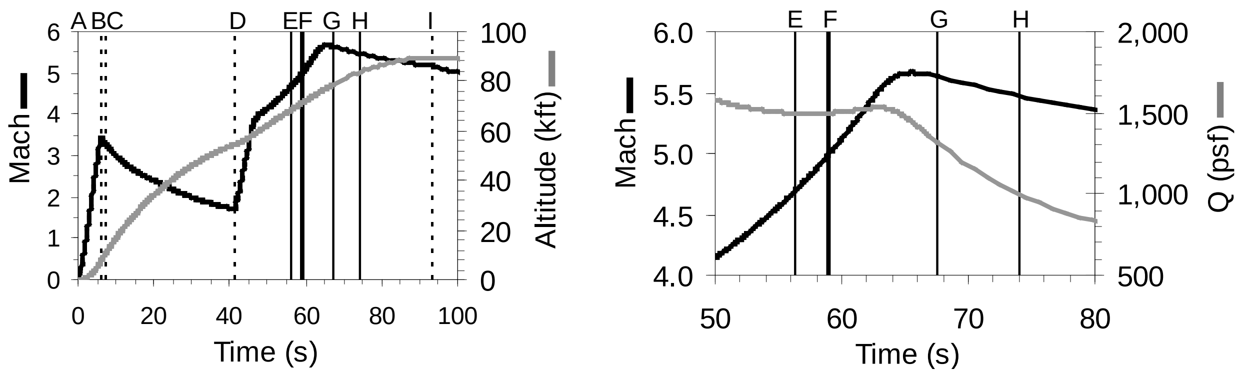


Figure 1. A schematic of the SDPTE(Hy-V) launch vehicle. All dimensions in inches (centimeters).



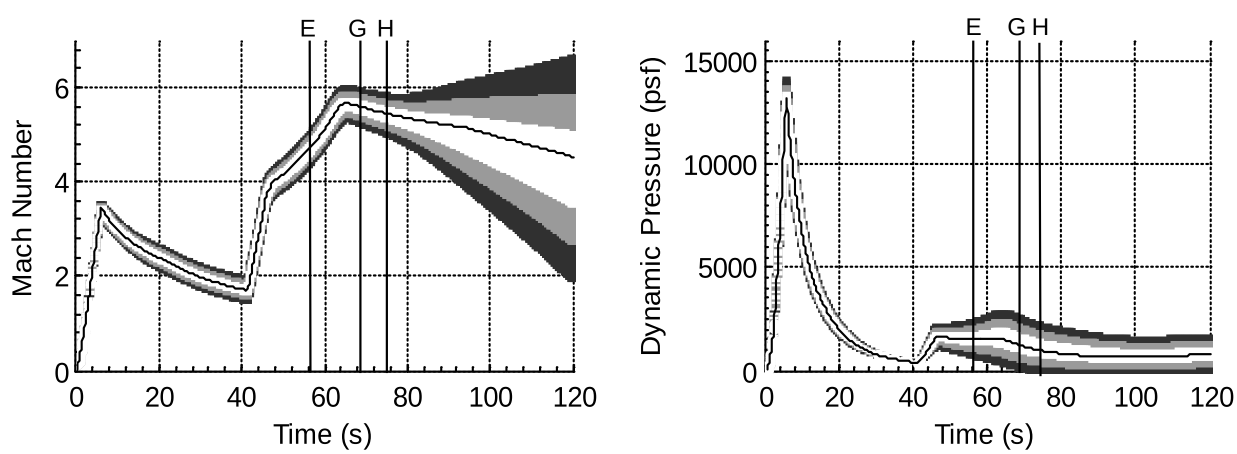
a. b.

Figure 2. a) Mach number and altitude vs. time and b) Mach number and dynamic pressure vs. time within test window. A) First stage ignition, B) First stage burnout, C) Stage separation, D) Second stage ignition, E) Deploy shroud, begin primary experiment, F) Nominal test point, G) Second stage burnout, primary experiment end, H) Dynamic pressure reaches 1,000 psf, secondary experiment end, and I) Apogee.

Dispersion

With a nominal trajectory in hand, dispersion about that trajectory can be determined. Dispersion of sounding rocket trajectories is typically estimated using a Monte Carlo analysis technique. Monte Carlo analysis is a statistical tool that is used to relate predicted independent parameter variations to the performance of the system. Applicable model inputs are varied independently and randomly within estimated uncertainty bounds for each run. When many simulations are performed, each with a unique set of randomly varied contributors, a more realistic model of system performance is developed than if only one input variable, or set of variables, was varied at a time. This is because random variations in input uncertainties can interact with each other in unexpected and potentially detrimental manner. Such an analysis was performed for the Hyper-X Program to simulate the dynamics of the X-43A vehicle. [11]

The contributors used for this Monte Carlo analysis are based on those historically used by the NASA Sounding Rocket Operations Contract (NSROC). The standard contributors are launch elevation angle, weight, thrust, thrust misalignment, center of gravity offset, fin misalignment, wind, drag coefficient error, initial pitch rate and the launch azimuth. For the present analysis, a higher than standard magnitude and uncertainty range for the initial pitching rate and vehicle drag was used as a result of the post-flight trajectory analysis performed for the FASTT Program. The resulting dispersion in the Mach number, and dynamic pressure is shown in Figure 3 for a 5,000 run Monte Carlo simulation. This level of simulation was



a. b.

Figure 3. a) Mach number vs. time +/- 1, 2, and 3 standard deviations, and b) Dynamic pressure vs. time +/- 1, 2, and 3 standard deviations. Primary test window is from 56.4 seconds to 67.5 seconds and secondary test window is from 67.5 to 74.0 seconds, as indicated by vertical bars.

chosen such that the number of runs necessary for statistical convergence could be identified and a lower number of runs could be adopted in latter analyses that were more computationally intensive. Approximately 68% of trajectories fall within one standard deviation, 95% within two standard deviations, and 99.5% within three standard deviations. For both Mach number and dynamic pressure, dispersion increases throughout the trajectory, as can be seen in the figure. This validates the decision to design the nominal trajectory with the test window as early in the trajectory as possible.

The dispersion in the Mach number and dynamic pressure during the test window and at the nominal test time is particularly important because these quantities affect the operation of the scramjet inlet and flowpaths. Given the performance characteristics of the boosters and the nominal trajectory for this flight, it is nearly certain that the vehicle will pass through the design Mach number of 5.0 during the sustained burn phase of the second stage burn as no trajectories in this Monte Carlo analysis failed to do this. However, achieving the design dynamic pressure at this Mach number means that the vehicle must pass through Mach 5 at the proper altitude since the static pressure is determined by the altitude and dynamic pressure is given by $q=(\gamma/2)PM^2$, where q is the dynamic pressure, γ is the ratio of specific heats, P is the static pressure, and M is the Mach number. There is also uncertainty in the time at which the vehicle passes through a given Mach number. This is important because payload events, such as shroud deployment, will be determined by a preprogrammed timer for this program. For instance, if the shroud deployment time is set based on the nominal trajectory and in flight the Mach number is less than expected at this time (or equivalently, the vehicle passes through the expected Mach number at a later time in the trajectory) the inlet may not operate as expected. Computational fluid dynamics studies indicated a lower Mach number limit for which all inlet shocks remain attached and the inlet remains started. For this study, scramjet flowpath geometry and ground test facility limitations determined the upper and lower dynamic pressure and upper Mach number limits for a successful experiment. Based on these considerations, the success criteria were developed of lower and upper Mach number limits of 4.7 and 5.3, respectively. The lower and upper dynamic pressure limits were 1,109 and 1,873 psf (53.10 and 89.68 kPa), respectively. Other success criteria derived from further scramjet operability concerns could be incorporated into this analysis, but for simplicity, they will not be considered here.

Method for Reduction of Dispersion

Given the fact that only one flight test is planned for the SDPTE(Hy-V) program, it is important that every measure is taken to increase the likelihood of a successful test. Without actively controlling the launch vehicle, one of the only ways to change the freestream conditions at the expected test time is to vary the second stage ignition time (SSIT) and alter the expected test time appropriately. A relationship was found here between the state of the vehicle at a time during the interstage coast and the second stage ignition time required to achieve a desired flight Mach number at the altitude consistent with a given dynamic pressure.

A 1,000 run Monte Carlo simulation was performed using the vehicle model and dispersion contributors discussed above in order to establish this relationship. This number of runs was deemed optimal as statistical convergence was obtained and incorporating more runs would have been computationally prohibitive as each trajectory is later iterated upon in order to determine the best second stage ignition time. Each run was comprised of one trajectory which incorporated its own distinct set of randomly varied input contributors. For each trajectory, the second stage ignition time was iterated upon until the trajectory passed within 3 psf (0.143 kPa) of the design dynamic pressure at a Mach number of 5.0. For 0.2% of the trajectories, altering the second stage ignition time to attain the design dynamic pressure precluded the vehicle passing through a Mach number of 5.0. For these trajectories, the second stage ignition time was adjusted to bring the dynamic pressure as close to 1,500 psf (71.82 kPa) as possible without preventing the vehicle from achieving the design Mach number. Since second stage ignition time was adjusted to achieve the desired test conditions, the time at which we expect these conditions to occur must also be adjusted. Therefore the time at which the test conditions were achieved was extracted in addition to the second stage ignition time required to achieve the proper test conditions.

For each trajectory, the vehicle's state was observed at a flight time of 27.0 seconds. This time was chosen to be as late as was feasible such that the vehicle has as much time as possible to stray from the nominal trajectory giving the largest possible variation in measured Mach number and altitude. This observation time, however, cannot be made so late that it is after the earliest optimal second stage ignition time. The earliest optimal second stage ignition time was found to be 30.0 seconds. Three seconds was deemed a conservative estimate for the time required to take the requisite measurements, calculate the vehicle's state, determine the optimal second stage ignition time, and ignite the second stage booster.

To determine the relationship between the optimal second stage ignition times and the Mach numbers and altitudes observed at 27.0 seconds, a third order polynomial surface was linearly regressed to fit the extracted optimal second stage ignition time data. This curve represents the statistical relationship between the optimal second stage ignition times and the altitudes and Mach numbers observed at 27.0 seconds. Since the second stage ignition time was adjusted to achieve the desired test conditions, the time at which we expect these test conditions to occur must also be adjusted. Similar to the relationship for the optimal second stage ignition time, a third order polynomial surface was fit to the optimal test time data. The R^2 values for the optimal second stage ignition time and optimal test time surfaces were 0.961 and 0.653 respectively. Figure 4 represents the second stage ignition time (SSIT) relationship for various altitude ranges. The solid lines are lines of constant altitude corresponding to the upper and lower limits in altitude measured at t = 27.0 seconds from which the data points were selected. A similar series of curves can be developed for the optimal test time.

Once the relationship between optimal second state ignition time and test time with Mach number and altitude was determined, a new 1,000 run Monte Carlo analysis was performed to test the dispersion reduction approach. This analysis used a new set of contributors for each trajectory but adopted the determined relationships to calculate the second stage ignition time and the expected test time based on the observed altitude and Mach number at 27.0 seconds. A new analysis was required because the original Monte Carlo runs were used as the basis for the relationships and it would not be an independent test of the effectiveness of the dispersion reduction approach to reuse the original trajectories. While the base values and 3-sigma ranges for the contributors were the same, the actual random values that the contributors took on for each Monte Carlo trajectory were different. Since these new trajectories utilize the optimal second stage ignition and test time maps, which were determined *a priori*, they give an accurate prediction of the dispersion that results when utilizing the dispersion reduction method.

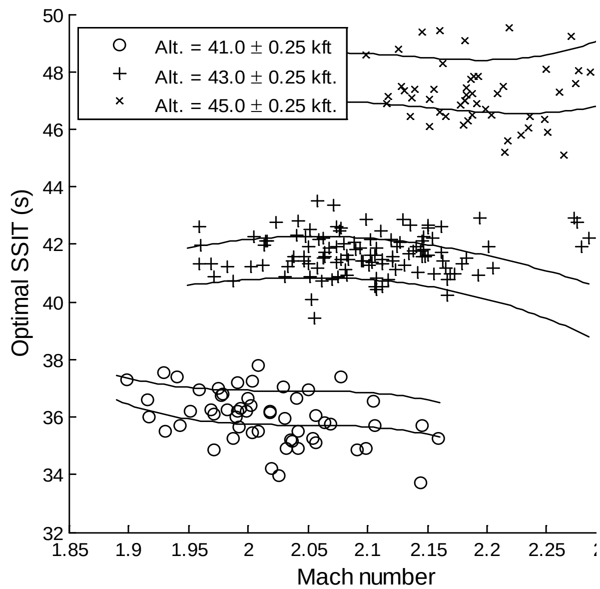
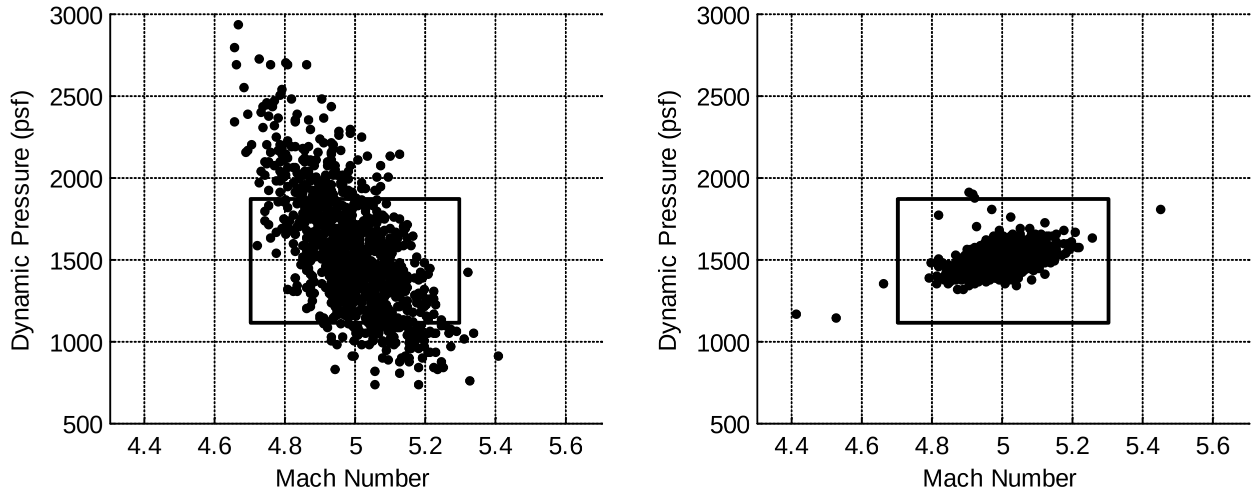


Figure 4. Relationship between optimal second stage ignition time and the Mach number and altitude measured at t = 27.0 seconds. The curves represent the polynomial regression at the limits of the +/- 0.25 kft range for each nominal altitude.

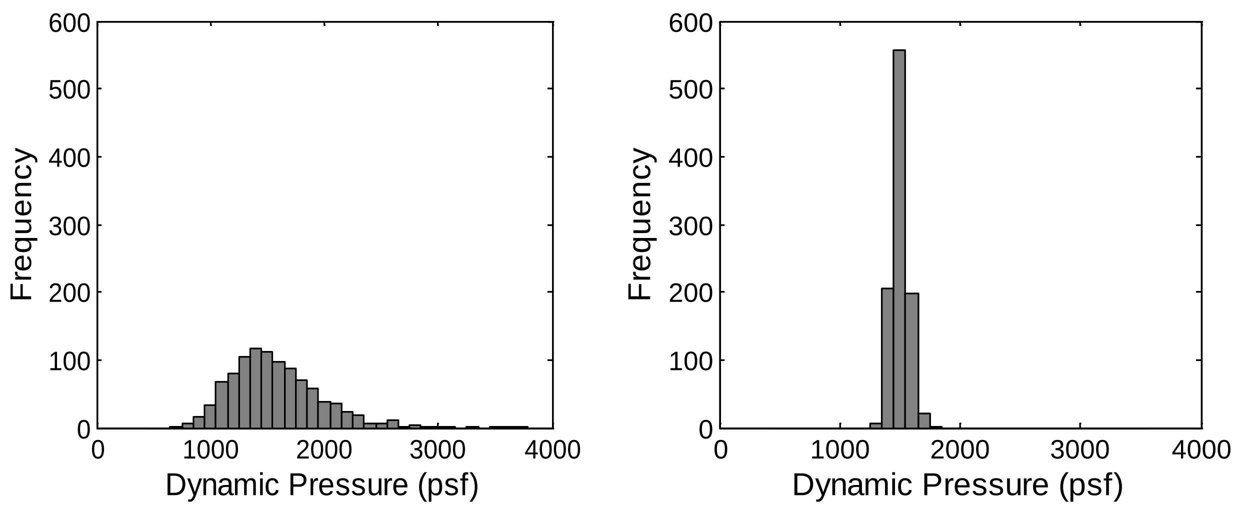
Results

The results from the Monte Carlo analysis, which utilized the computed maps to determine the second stage ignition and test times, are presented here. The likelihood of a successful test was significantly increased by use of this method. It is useful to plot the dynamic pressure and Mach number at the expected test time for each trajectory and compare against the program specific success criteria discussed above. For the uncorrected trajectories, the expected test time is the time when the nominal trajectory passes through Mach 5.0. This is equivalent to initiating the test time in flight via a timer that was originally set using the nominal trajectory as a guide. For the corrected trajectories, the expected test time is that determined from the optimal test time relationship described above. Figure 5 shows this plot with success criteria cast into dynamic pressure and Mach number limits, indicated by the black box. Without dispersion reduction, 71.0% of trajectories fall within the success criteria. Using the method for reducing dispersion, 99.3% of trajectories fall within the success criteria at the anticipated test time. This represents a significant increase in the likelihood of a successful scramjet test. Figure 6 shows a histogram of dynamic pressures when the vehicle passes through a Mach number of 5.0. The standard deviation of dynamic pressures without and with dispersion reduction is 389 and 66.0 psf (18.63 and 3.16 kPa), respectively. Again, this is a significant improvement.



a. b.

Figure 5. a) Dynamic pressure and Mach number for each trajectory without and b) with the application of dispersion reduction method, at the expected test time. Boxes represent success criteria.



a. b.

Figure 6. Histogram of dynamic pressures at the time the vehicle passes through Mach 5.0 for trajectories a) without dispersion reduction and b) with dispersion reduction.

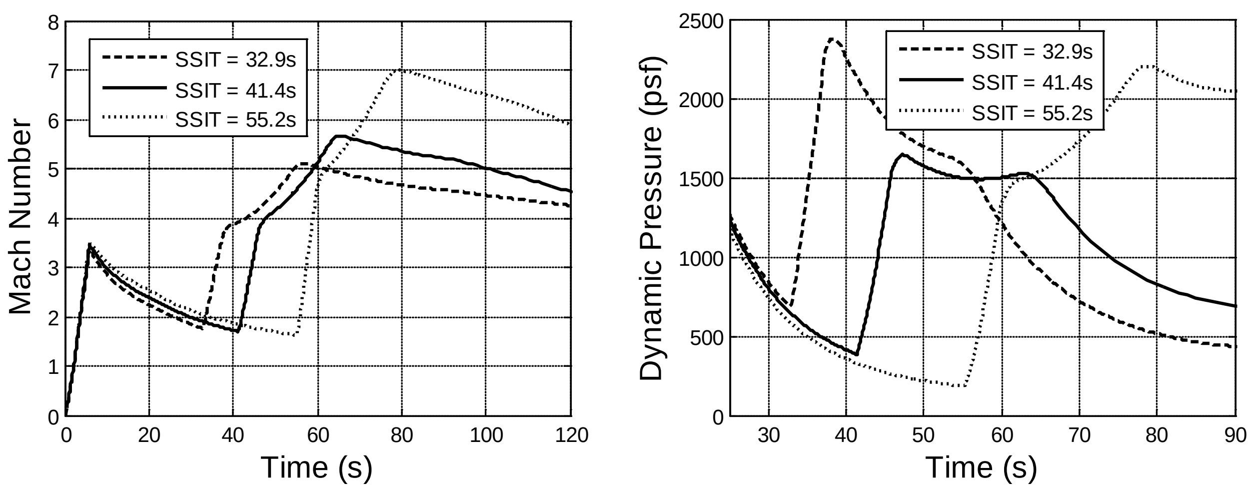
Discussion

The method described here was conceived with the aim of decreasing the dispersion in freestream conditions that the payload and vehicle experience at and near the design test point. While successful, utilizing this method has other implications for practical scramjet flight testing. How this method affects the trajectory off the design test point must be also considered as this affects ignition flow conditions of the scramjet and equivalence ratios seen by the scramjet combustor during the test. Range safety must also be considered, as the eventual splashdown location of the second stage and payload may change. Finally, practical implementation of the dispersion reduction technique should also be considered.

As discussed above, the nominal trajectory incorporates a nearly constant dynamic pressure near the test point. This is possible because the increasing Mach number during the sustained burn phase of the second stage Improved Orion booster is almost exactly balanced by the decreasing static pressure with increasing altitude. Figure 7 shows both Mach number and dynamic pressure vs. time for the nominal trajectory as well as trajectories which initially undershoot and overshoot the nominal altitude. These two latter trajectories have been corrected by decreasing and increasing the second stage ignition time, respectively. Since both these trajectories are outliers, it is likely that the correction required, and thus the effects of this correction, will be less than is shown here. In fact, the two sample trajectories in Figure 7 both had optimal second stage ignition times over 1.7 standard deviations away from that of the nominal trajectory. Trajectories that initially undershoot the nominal altitude are corrected back to pass through M = 5.0 and q = 1,500 psf by decreasing the second stage ignition time. For these trajectories, the dynamic pressure is decreasing in time near the test point, and the test point occurs near the end of the sustained burn phase of the second stage booster (test time at t = 56.0 sec. for SSIT = 32.9s.). Since the second stage burn takes place at a lower altitude where total drag is higher, the peak Mach number attained is lower than for the nominal trajectory. Conversely, when trajectories that initially overshoot altitude are corrected, second stage ignition time must be delayed. This results in an increasing dynamic pressure around the test point, which occurs closer to the beginning of the sustained burn phase of the second stage booster than for the nominal trajectory (test time at t= 62.5 sec. for SSIT = 55.2s). Since the second stage burn takes place at a higher altitude where there is less total drag, a higher peak Mach number is attained.

It is important to consider the effect of trajectory correction on scramjet operation. For a trajectory which initially undershoots the desired altitude and is corrected back, a decreasing dynamic pressure in the vicinity of the test point means that before the test point, dynamic pressure will be higher than expected. If the fuel system is designed to provide a nearly constant and predetermined fuel flow rate, as is the case for the SDPTE(Hy-V) flight, the equivalence ratio when the scramjet is ignited will be lower than expected and the scramjet may not ignite. For a trajectory which initially overshoots the desired altitude and is corrected back, an increasing dynamic pressure means that the equivalence ratio at light-off will be higher than expected and the inlet could unstart. If it is determinate that light-off equivalence ratios are outside predetermined limits for a particular scramjet, then maps of optimal shroud deployment and ignition times as a function of Mach number and altitude measured during the interstage coast may be required. The procedure for determining this relationship would be exactly the same as that for finding the appropriate second stage ignition time and test time. Operability of the scramjet following ignition should also be considered and expected equivalence ratio limits compared against preflight ground test results.

Adjusting the second stage ignition time mid-flight also has implications on the splashdown location. While splashdown location is of little consequence for the success of the scramjet test, it is very important for range safety. The effect of this dispersion reduction technique on splashdown location is non-intuitive. Use of this technique actually decreases the dispersion in splashdown location and creates a bimodal distribution



a. **b.**

Figure 7. a) Mach number vs. time for the nominal trajectory and two with adjusted second stage ignition and test times, and b) Dynamic pressure vs. time near the test window for the nominal trajectory and two with adjusted second stage ignition and test times. Approximate test points are where trajectories cross M = 5.0 and q = 1500 psf as indicated by horizontal dashed lines.

of splashdown locations. Figure 8 shows the splashdown locations for all trajectories with and without dispersion reduction. Trajectories which are initially low on altitude and are corrected by decreasing the interstage time delay tend to be clustered near the nominal trajectory and with trajectories which require little correction. Trajectories which are initially high on altitude and are corrected by increasing the interstage time delay are clustered down-range. Total downrange dispersion is decreased by approximately 30%. As expected, cross-range dispersion is unaffected by use of this technique. As such, it is unlikely that the implementation of this technique will adversely affect range safety.

For this dispersion reduction technique to be implemented, an onboard processing unit is required. Two-dimensional lookup tables generated from the maps calculated above could be used. In this case, the processing unit would read the Mach number and altitude at the predetermined measurement time (t = 27.0 seconds here) and interpolate the values for the times of various events. Alternatively, the fitted equations for the required maps could be coded and the values for the times of various events calculated directly. Both methods for implementing this dispersion reduction scheme would be possible using any onboard programmable processing unit that can calculate the vehicle's state from available instrumentation and output a signal to ignite the boosters. An example of such a device is the GLN-MAC, a roll-stabilized inertial measurement unit developed by Sandia National Laboratories.[10] While its primary utility is post-flight trajectory reconstruction, it also contains a programmable processing unit which is capable of performing the calculations described above. Simpler and lower cost units could also be used in place of the GLN-MAC if inertial measurement capabilities are not required for a particular mission.

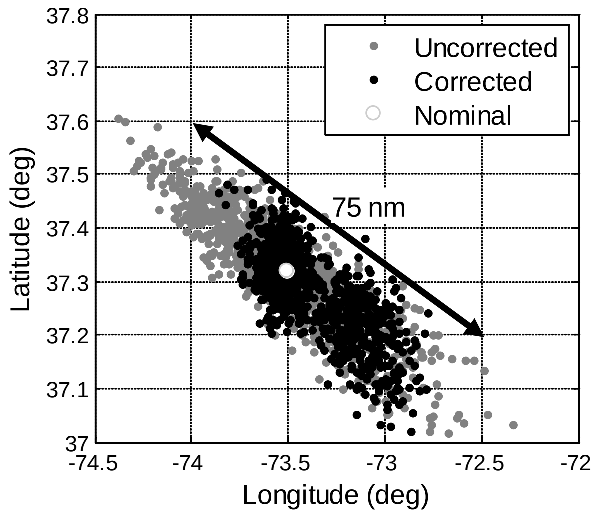


Figure 8. Splashdown location for all trajectories with (corrected) and without (uncorrected) dispersion reduction. Vehicle enters figure from the upper left.

Conclusion

A novel and inexpensive method for decreasing dispersion for a spin-stabilized, unguided sounding rocket scramjet flight experiment is presented here. Using a Monte Carlo simulation, the confidence with which the nominal trajectory is simulated has been quantified and presented. While not all resulting trajectories are acceptable, it is possible to reduce dispersion in the test conditions by altering the delay between the first stage booster burnout and the second stage booster ignition. This is accomplished by using a statistical relationship between Mach number and altitude, measured during the interstage coast, and the second stage ignition time which allows the vehicle to pass through the desired dynamic pressure at the target Mach number. Altering the second stage ignition time necessitates adjusting the times at which other payload events take place. Since the relationship between the vehicle state during the interstage coast and the optimum timing for payload events is generated through trajectory simulations, the fidelity of this relationship is dependent upon the accuracy of the input parameters and modeling of the contributors. While active control of the vehicle may help a test article achieve desired freestream conditions, such systems carry with them additional risk of failure and are expensive to develop and implement. The method for test condition dispersion reduction presented here is an inexpensive way to increase the probability of a successful scramjet test aboard unguided, spin-stabilized sounding rockets without adding significantly to the complexity or cost of the program.

In this analysis, dispersion in the dynamic pressure at a given Mach number was decreased at the expense of dispersion in the time rate of change of the dynamic pressure near the test point. In general, decreasing dispersion in any given parameter will almost certainly be accompanied by an increase in dispersion in other parameters. The implications of this must be carefully considered for any mission. It is also conceivable that this technique can be extended to a three stage sounding rocket, although the utility and implications of this application are yet to be explored.

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