STARS MDT-II TARGETS MISSION

Brent A. Sims John E. White
Sandia National Laboratories

The Strategic Target System (STARS) was launched successfully on August 31, 1996 from the Kauai Test Facility (KTF) at the Pacific Missile Range Facility (PMRF). The STARS II booster delivered a payload complement of 26 vehicles atop a post boost vehicle. These targets were designed and the mission planning was achieved to provide for a dedicated mission for view by the Midcourse Space Experiment (MSX) Satellite Sensor Suite. Along with the MSX Satellite, other corollary sensors were involved. Included in these were the Airborne Surveillance Test Bed (AST) aircraft, the Cobra Judy sea based radar platform, Kwajalein Missile Range (KMR), and the Kiernan Reentry Measurements Site (KREMS). The launch was a huge success from all aspects. The STARS Booster flew a perfect mission from hardware, software and mission planning respects. The payload complement achieved its desired goals. All sensors (space, air, ship, and ground) attained excellent coverage and data recording.

Introduction

The Strategic Target System (STARS) MSX Dedicated Target mission (MDT-II) was successfully launched from the Kauai Test Facility (KTF) on the Pacific Missile Range Facility (PMRF) on Kauai, Hawaii at 15:41:49 GMT on August 31, 1996 (see figure 1). This target mission included the deployment of 25 test objects from an Operational Deployment Experiment Simulator (ODES) post-boost-vehicle. These test objects were viewed by the Midcourse Space Experiment (MSX) satellite, and a variety of air, sea, and ground based auxiliary sensors. Impact occurred about 400 km northeast of the Kiernan Reentry Measurement System (KREMS) complex on the Kwajalein Missile Range (KMR). Participating auxiliary sensor platforms included the Airborne Surveillance Testbed (AST) aircraft, the Cobra Judy (USNS Observation Island) radar ship, and a variety of sensors on the KMR. The STARS booster and post-boost vehicle operation were nominal, and data in support of mission objectives was collected by all participating sensors.

This paper will begin with a brief description of the MSX satellite. Later sections will detail the target mission senario, describe the MDT-II STARS booster configuration, explain the pre-flight hardware and software testing, and provide a summary of the flight test results.

MSX Satellite [1]

The MSX satellite is the first in-space system demonstration of technology to characterize ballistic missile signatures during the midcourse flight phase between boost and missile reentry. The satellite is designed to detect, track, and discriminate realistic targets against terrestrial, Earth limb, and celestial backgrounds. MSX is capable of observations over a wide range of wavelengths, from the very-long infared to the far-ultraviolet.

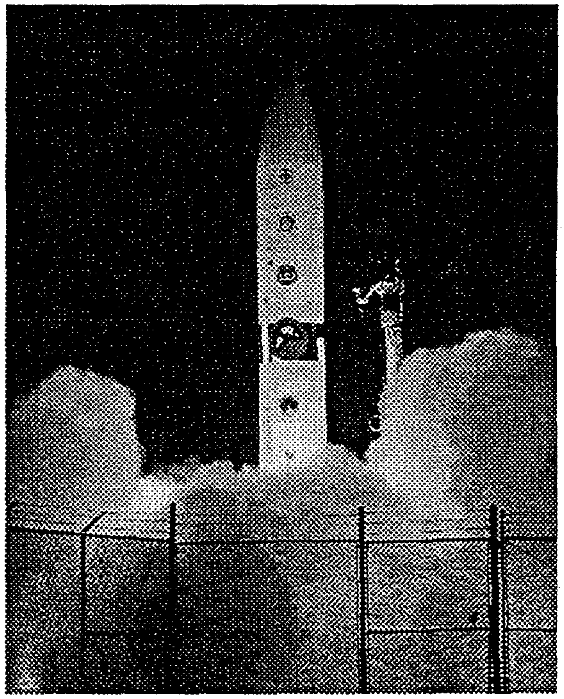

Figure1. STARS MDT-II Launch

The spacecraft has five primary instruments consisting of 11 optical sensors. All sensors are precisely aligned so that simultaneous observations with multiple sensors can be made, which is essential for scenes or targets which change rapidly. These primary instruments include the SPIRIT III (Spatial Infared Imaging Telescope), UVISI (Ultraviolet Visible Imagers and Spectrographic Imagers), SBV (Space Based Visible), OSDP (On-Board Signal and Data Processor), and several contamination sensors. The SPIRIT III was built by the Space Dynamics Laboratory of Utah State University. The UVISI and the contamination sensors were built by the John Hopkins University Applied Physics Laboratory. The SBV sensor was built by the Massachusetts Institute of Technology Lincoln Laboratory, and the OSDP was built by Hughes Aircraft Company.

The MSX satellite was launched on April 24, 1996 from Vandenberg Air Force Base, into a high-inclination, circular, near sun-synchronous orbit at 903.5 km altitude. The satellite was built and operated by the John Hopkins University Applied Physics Laboratory.

MDT-II Target Mission Plan

Planning for the STARS MDT-II targets mission was driven by a complicated set of requirements related to viewing by the MSX satellite on-board sensors, as well as by ground, sea, and air-based supplementary sensors. The MDT-II targets mission included requirements for functional demonstrations of metric and radiometric discrimination, PBV and deployed object acquisition and tracking , as well as cluster track and resolution. Other requirements included collection of target signature phenommmenology on the STARS booster and PBV, deployed RV's and penaids, and clusters of objects. Other practical considerations also resulted in mission constraints. The targets mission launch window was required to be five days long to accommode weather or technical delays, and to reduce range scheduling complications. This requirement allowed for up to five launch attempts per window, since the launch opportunity on each of these days was only a few seconds long. All of the mission objectives were incorporated into the definition of a "feasibility" region, which would ensure satisfaction of mission requirements if the satellite groundtrack was located within this "feasible" geographic region at the time of target launch. The feasibility region was directly related to the target trajectory and deployment timeline, and several iterations on each were used to help ensure the satisfaction of mission objectives.

Another complication in the planning process was the data collection requirements of the auxiliary sensors. Generally, these issues were handled by the placement of the mobile sensor platforms, and via compromises in the target trajectory and mission timeline, particularly in the case of the fixed, ground-based assests at KMR.

Basic target trajectory requirements included a minimum apogee of 914 km to ensure space background tracking during the post-boost phase of the mission, and a maximum range from the KREMS complex of about 550 km to provide best data from ground-based assets on the KMR. All target trajectories were designed to ensure that the 3-sigma low performance case satisfied the minimum apogee altitude requirement.

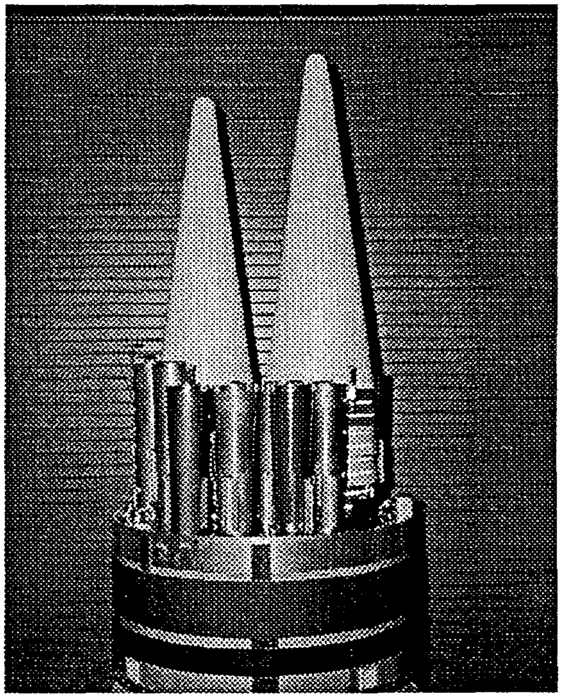

Figure 2. MDT-II Targets and PBV

Off-nominal performance during the STARS boost phase was one determinant of the feasibility region. boundary. None of the STARS motors is equipped with a thrust cut-off capability (except for the flight termination system, which would be used only to ensure range safety), and off-nominal performance has a direct bearing on the resulting target trajectory. Several third stage targeting options were evaluated during the mission planning process to produce a target trajectory with the least restriction on the feasibility region. Two basic third stage guidance options are available with the STARS system. Both are based on solutions to Lambert's problem, which provides for a target point constraint. The first option is an energy wasting procedure designed to constrain time-of-flight. In this case all trajectories resemble the 3-sigma low case, with significant maneuvering during third stage burn, and nearly constant time-of-flight. The second option provides for a nearly constant attitude third stage burn, a variable time-of-flight, and no energy wasting. In this case the resulting trajectory shape will vary with booster performance, but the desired target point will be achieved at a variable time. Target points off of the 3-sigma low trajectory baseline at pre-apogee, apogee, post-apogee, and impact were evaluated. After extensive analysis by the mission planning team, the second guidance option was chosen with a pre-apogee target point. This senario provided for acceptable trajectory variations during the pre-apogee deployment phase, while directing all available energy into delivering the targets near to the KREMS complex on the KMR.

The MDT-II mission can be briefly described as follows. First stage ignition was initiated at a time determined daily based on the latest orbit data of the MSX satellite. The launch window was only a few seconds long on any given day, so a fixed time associated with the middle of the feasibility region was chosen as the launch time for that day. Any problems encountered late in the countdown generally required a slip to the next day. The first stage initial fly-out azimuth was also set late in the countdown, based on launch day winds, to ensure adequate range safety margins as the instantaneous impact point (IIP) traversed north of the inhabited island of Niihau west of Kauai. The launch day selection of launch azimuth was required due to the transition to open-loop guidance after the initial pitch-over was completed at 20 seconds into the mission. At this time an angle-of-attack control logic was enabled which attempted to fly zero angle-of-attack through the staging event. This controller used inputs of launch day winds from balloon flights to fly near true zero angle-of-attack so as to improve control margins for the extended length STARS II booster. Closed-loop guidance was reestablished 10 seconds after 2nd stage ignition, and the turn towards the KMR was initiated. After second stage burn-out, a minimum separation coast of 10 seconds was employed prior to third stage ignition. The short coast period provided for the earliest possible third stage ignition, which resulted in the best overall trajectory range and earliest payload deployment times. During this coast the spent 2nd stage was retroed away, the initial guidance solution for the third stage was completed, and an attitude control system (ACS) reorientation maneuver was executed to setup for the third stage burn. Approximately 5 seconds into the coast phase the clamshell shroud separation was initiated. The ACS system was disabled just prior to and then re-enabled just after the shroud separation event. At the end of the 10 second coast period, the third stage motor was ignited. After a nearly constant attitude burn, the PBV and payload complex were essentially on the required path to the target point. At third stage burn-out a 12 second coast was initiated to prepare the PBV for the separation event. During this time, the ACS was active to maintian control of the vehicle as the third stage tailed-off. Once the PBV fuel and electrical systems were activated, the separation of the PBV and third stage was initiated. The PBV propulsion system was used to provide a separation velocity, and was also used to eliminate any off-nominal performance of the third stage. After a series of maneuvers, the PBV was situated to begin payload deployments. At the end of the payload deployment sequence, several PBV maneuvers and engine burns were executed for viewing by the MSX satellite. A final satellite experiment was conducted late in the mission as the PBV engines were turned-on again to be viewed with an earth limb and upper-atmosphere interactions.

A variety of targets were deployed during the MDT-II mission. All targets were designed and built by SNL. There were 26 test objects to be deployed during the MDT-II mission, of which 10 were instrumented with the SNL Light Weight Instrumentation System (LWIS). These included an instrumented MSX Aeroshell (MAS), 2 ultra-light replicas (ULR), 6 Rigid Light Replicas (RLR) (3 instrumented), 7 Canisterized Light Replicas (CLR) (4 instrumented), 4 Canisterized Traffic Balloons (CTB) (2 instrumented), an Emissive Reference Sphere (ERS), and a Multi-Balloon Canister (MBC). Telemetry data from the instrumented targets was relayed to the ground via a re-rad system onboard the PBV. A SNL designed video system was also located on the PBV to provide real-time confirmation of test object deployment, and reference data for post-flight analysis. The integrated PBV and payload can be seen in figure 2.

Because of the quantity of targets involved, and the requirement by auxiliary sensors to switch from object to object in real time during the mission, GPS corrected IMU state vectors were generated onboard the STARS PBV and telemetered to the KTF.

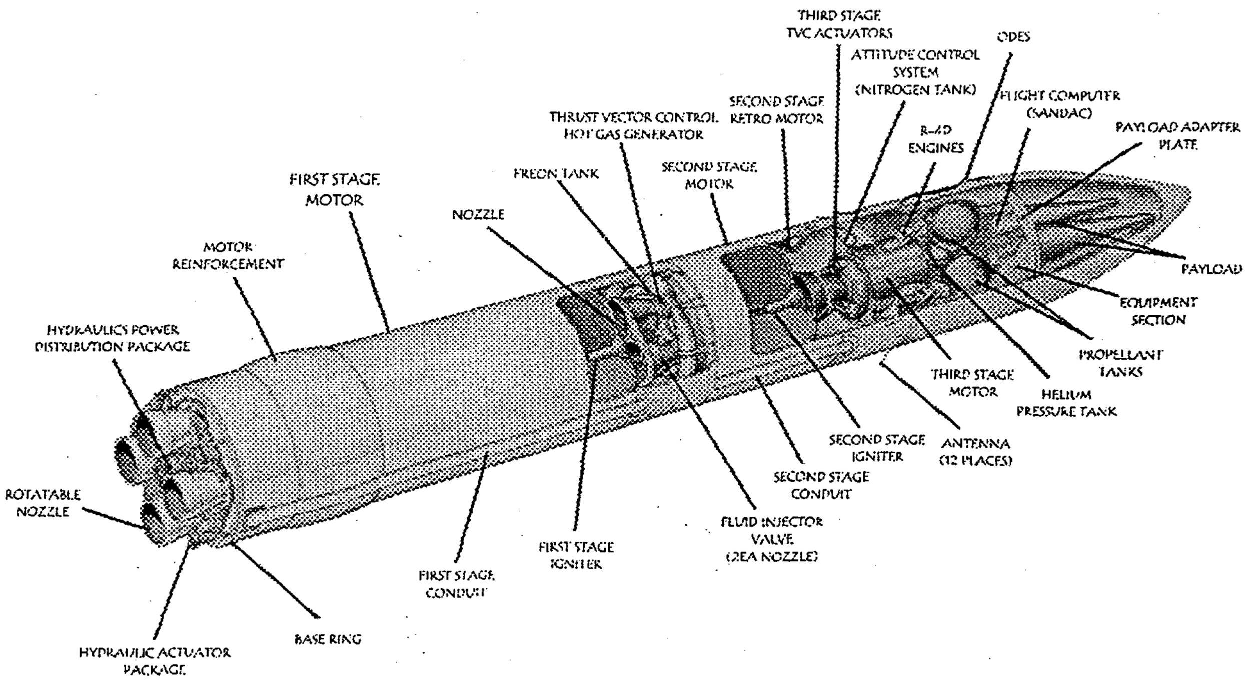

Figure 3. STARS II Booster Cutaway

Subsequently, these state vectors were transmitted to the auxiliary sensors to be used as aids to improve real-time radar acquistion within the complicated target complex. This capability has now been successfully demonstrated on the last two STARS missions and provides increased reliability in achieving mission data objectives for complicated target missions.

STARS II Booster [2]

The STARS II is a three stage solid rocket motor booster with a post boost vehicle (PBV). This vehicle can be seen in figure 3.

The STARS II first stage (FS) rocket motor is a refurbished Polaris A3P FS motor. The stage is 182 inches (15.2 feet) long, 54 inches in diameter, and contains approximately 20,800 pounds of solid propellant (high explosive equivalency of 10,500 1bs TNT). The STARS II FS motor assembly consists of an externally modified motor casing, an igniter assembly, four rotatable nozzle assemblies with hydraulic actuation for thrust vector control (TVC), and onboard flight termination system (FTS). The motor chamber is a fiberglass filament wound pressure vessel with skirts and bosses, filled with solid propellant. An external casing modification has been made to the motor case that consists of a 24 inch-wide composite overwrap, centered 31 inches forward of the aft skirt. The composite overwrap is a hot gas seal consisting of a casing bonded insulation blanket under fiberglass hoop wraps. Motors that were selected for flight had passed a full body radiographic inspection. The igniter assembly consists of a pyrotechnic igniter chamber and a new 28 volt hot bridge-wire initiator. Four equally spaced rotatable nozzle assemblies are bolted to the nozzle bosses in the motor aft dome. The nozzles are controlled by individual hydraulic actuator packages(HAPs) that are powered by a hydraulic power distribution package (HPDP) centered on the aft dome. Prior to flight, the nozzles are removed from the candidate flight motor, recertified by leak and torque testing, and reinstalled. The HAPs and HPDPs are certified for flight by acceptance testing of the individual components and functional testing of the system.

The FS Flight Termination System (FTS) is located on the forward dome of the motor. The FTS consists of flexible linear shape charge (FLSC) bonded to the dome, two MC3644 detonators with attachment hardware, and the MA170 FTS electronics package with auto-destruct capability. The motor thrust can be terminated by cutting the fiberglass forward dome with the FLSC and venting the internal motor pressure.

The STARS II interstage section (IS) is a modified Polaris A3 IS. It is a cylindrical shell made of a magnesium-thorium sheet, HK31A-H24, which is 54 inches in diameter and 34.3 inches long. The first stage motor ignition umbilical interface is on the IS. A mild detonating fuse (MDF) is installed around the forward circumference of the section to separate the first stage and interstage from the rest of the missile after first stage burnout. The MDF is initiated by two MC3644 detonators. The interstage section contains an A3 Hydraulics Battery for powering the FS TVC. The original A3 interlocks have been removed.

The STARS II second stage (SS) rocket motor is a refurbished Polaris A3P SS motor. The stage is 89 inches (7.4 feet) long, 54 inches in diameter, and contains approximately 8,800 pounds of solid propellant (high explosive equivalency of 8712 lbs TNT). The STARS II SS motor assembly consists of a modified motor chamber, an igniter assembly, four fixed nozzle assemblies with liquid injection for TVC, and an onboard FTS. The motor chamber is a fiberglass filament-wound pressure vessel with skirts and bosses that is filled with solid propellant. The chamber modification is internal in the forward dome region. The refurbishment modifications involve draining the liquefied potting out of the gap between the chamber insulation and the propellant shrinkage liner; repotting the gap with a silicone material; and replacing the rigid potting containment device with a flexible potting containment baggie. The baggie collects residual liquefied potting and prevents the potting from contacting the propellant. The igniter assembly consists of an igniter chamber and a new 28 volt hot bridge-wire initiator. The main charge of propellant in the igniter has the basic chemical properties of the SS motor propellant. The SDI initiator also ignites the hot gas generator (HGG) in the SS TVC system. Four equally spaced, fixed nozzles are attached to the nozzle ports by retaining rings. The flight motor candidates are received and x-rayed without nozzles installed. The nozzles are inspected separately, refurbished (if required), and certified for flight prior to installation. The SS liquid TVC system consists of four injector valve assemblies, a manifold assembly, a toroidal fluid filled tank, a HGG, and a hot gas relief valve. Second stage thrust deflections are accomplished by injecting fluid into one side of a fixed nozzle to create a shock wave. Nozzles 1 and 3 are used for pitch control while nozzles 2 and 4 are used for yaw control. Roll control is accomplished using all nozzles. All TVC components are certified for flight by inspection and acceptance testing. The manifold and injectors are functionally tested with nitrogen during assembly and final systems checkout.

The SS FTS is located on the forward dome of the motor. The FTS consists of FLSC bonded to the dome, two MC3644 detonators with attachment hardware, and the MA170 FTS electronics package with auto-destruct capability. The motor thrust can be terminated by cutting the fiberglass forward dome with the FLSC and venting the internal motor pressure.

The STARS II third stage (TS) extension assembly is a new design replacing the old Polaris A3 equipment section (ES). The TS consists of a structural extension which houses an Orbus 1 motor and an interstage between the second and third stages. The new TS is 62.8 inches long and 54 inches in diameter. The skin consists of a 0.160 inch magnesium-aluminum alloy sheet, AZ31B-H24. A 0.030 inch thick coating of Dow Corning 92-009 applied to the exterior surface of the TS protects the magnesium skin from aerodynamic heating during the ascent through the atmosphere.

The TS extension section houses the STARS II downstage missile electronics. The forward end of this section is defined by the mounting ring for adaption of the post boost vehicle (PBV), and the aft end by the lower MDF separation system. The four missile umbilical interfaces, including the payload systems umbilical, are located here. The electronic systems housed in this section include; 1^{st} 2^{nd} and 3^{rd} stage TVC packages, Arm and Fire (A&F) system.. Power and signal distribution for the missile electronics, including batteries and switches. Attitude Control System (ACS) with pneumatics. FTS command receivers and command destruct packages. The Orbus 1 motor is a new motor with TVC designed for STARS to SNL specifications by United Technologies Corporation, Chemical Systems Division (UTC/CSD). The motor is 49.2 inches long and 27.2 inches in diameter. It contains approximately 910 pounds of UTP-19687A solid propellant (high explosive equivalency of 920 lbs TNT) with an HTPB binder. The Orbus 1 motor assembly consists of a graphite composite case, an igniter assembly, a nozzle assembly with a flexseal joint, a TVC system with thermal battery, and an onboard FTS. The igniter assembly is a toroidal pyrogen igniter initiated by dual Teledyne McCormick Selph electric low voltage detonators, part number 817447. During third stage burn, vehicle pitch and yaw are controlled with the motor TVC system. Roll attitude is controlled during burn using the cold gas ACS. The Orbus 1 FTS is located on the aft dome of the motor. The FTS consists of FLSC bonded to the dome and two MC3644 detonators with attachment hardware. The motor thrust can be terminated by cutting the graphite aft dome with the FLSC to separate it along with the nozzle assembly from the rest of the motor.

The Post Boost Vehicle (PBV) is mounted to the adapter ring on the front of the TS extension. The PBV is separated from this structure in flight by a debris free MDF separation joint just below the adapter ring. The PBV is constructed with a composite monocoque design utilizing balsa wood, carbon fiber and aluminum as required. This design allows for increased strength with a lightweight structure. The PBV is divided into two discernible sections; the Component Section (CS) and the Propulsion Section (PS). The PS contains all hypergolic fuel tanks and fuel delivery system along with FTS related hardware. The PS also contains the R4D rocket motors and their actuator mounting systems or electro-mechanical actuators (EMA). The CS section contains all of the flight electronics which is required for fullfilling the mission requirements after all stages are separated. These systems include the Guidance Navigation and Control System (GN&C) which includes; Ring Laser Gyro Assembly (RLGA) inertial measurement unit (IMU), Sandia National Laboratories Digital Airborne Flight Computer (SANDAC), GPS receiver and supporting equipment. Also, included are the PBV TVC packages, Arm and Fire (A&F) system with programmable sequencer, telemetry support hardware, antennas and all system batteries.

The MDT-II STARS II utilized a newly designed composite clamshell shroud which was jointly developed by McDonnell Douglas and Sandia National Laboratories. This shroud split into two halves upon separation. The shroud provided a RF transparent cylindrical section to allow all PBV mounted antennas to radiate through the shroud. The shroud also contained a GPS antenna in the nose tip for pad and early ascent GPS viewing. The fully stacked missile with shroud is 37.7 feet tall which is 5 feet longer than the STARS I configuration.

Development And Testing (Albuquerque)

The following describes the development and testing philosophy followed by Sandia National Laboratories. Once the MDT-II payload delivery requirements were determined the requirements which drove modifications to the existing STARS II design were recognized. As typical with most launch vehicles, reducing weight i.e. increasing performance was critical. Also deploying the ascent shroud prior to a relatively immediate 3rd stage ignition was desired.

Design and Development

The existing system as flown on STARS M2, ODES Development Flight (ODF) was evaluated and two major mechanical modifications were required. First, the PBV was targeted for weight reduction primarily through lightening of the structure and payload plate. Also, all components which could reasonably be modified for weight reduction were repackaged and in some cases redesigned. Second, the existing STARS shroud and ejection method was too slow to meet requirements, therefore, a clamshell shroud was designed to reduce the latency between 2nd stage burn out and 3rd stage ignition. From performance analysis each second earlier the 3rd stage was ignited an increase of approximately 1.4 pounds of payload weight could be gained to the target. This modification allowed the growth of the payload by around 50 pounds, the single most significant performance gain of all the performance modifications.

Based on previous lessons learned, MDT-II requirements, and lightening issues, a number of components were targeted for redesign or repackaging. The AF&F Sequencer was redesigned for capacitive discharge. This design tends to limit impact on flight batteries during ordnance firing operations, and significant channel capability, up to 108 firing lines. The PBV TVC control electronics was redesigned based on lightening and improved servo control functionality. The Sandia Digital Airborne Flight Computer (Sandac) flight computer was upgraded from Motorola 68020 to Motorola 68040 processors. This modification reduced the number of processors required and hence weight reduction. The system junction box was redesigned and packaged for a significant reduction in weight. Each new or repackaged component was passed through a qualification test which is more strenuous than the typical flight acceptance testing which is required of all flight components before being deemed flight worthy. Both qual and acceptance include shock, vibration and in some cases resonant bar (pyro shock) and temperature and vacuum cycling.

Once the components were designed and connectors defined, mock ups were fabricated and each component (actual or mock) was mounted physically on the structure (PBV or 3rd stage). This placement was determined either by mass properties or location dependence. After these components were placed and the pinouts and connectivity requirements known, the cable routing and lengths could be determined. The design allowed for component replacement and reduction of conductors in a given cable. This reduced cost and impact for redesign of certain cables if a problem was found or design modified during testing. There are about 180 cables in the PBV, and about another 50 on the remaining portions of the missile. These cables go through pin to pin testing, hipot testing and in all cases go through system level environmental testing while components are functioning.

Modifications to components and mission requirements drove modifications to ground support equipment and ground test simulators. Ground Launch Computer (GLC), telemetry decoms, missile simulators and other mission specific programmable or electrically modified equipment were modified and checked out. Once the support equipment is checked out the flight hardware is electrically safe to connect to.

Test & Evaluation (Albuquerque)

The acceptance tested components are mounted into the PBV and cable connections made. Initial electrical continuity checks are performed and then a tedious procedure is performed to connect up the integrated system to tested ground support equipment. At this point the system can be powered up and down as prescribed by the GLC software or as required for testing. The 3rd stage components are also installed and the cable connections made. A similar procedure for checkout is also run on this part of the vehicle.

The basic setup of the PBV for the following autopilot testing is described next. The PBV is mounted on the system test stand which is a controllable moving base which provides independent roll, pitch and yaw motion for IMU input. This input drives the GN&C system in the current flight sequencer mode and tests the hardware-in-the-loop system with live input and response.

First, the PBV autopilot test is run. This test requires the PBV mounted on the test stand for motion input. The GN&C and PBV TVC system are brought up and put into the correct autopilot mode. Certain tests are performed to confirm the motion is passed correctly through the inertial sensors, software paths and processed correctly throughout the autopilot code. The actuators are also tested independently for range, accuracy of motion and bandwidth.

The attitude control system test follows (see figure 4.). This test is different from the other control system tests since the system actually controls the vehicle through a single axis maneuver (roll, pitch and yaw are all tested). The stacked PBV and 3rd stage (with mass mock 3rd stage motor) are hung below a crane hoisted air bearing for relatively frictionless rotational motion about the vertical axis. The handling fixture allows the vehicle to be put in all cardinal orientations. This test again confirms correct attitude motion passing through sensors and software. Also, the attitude control code is exercised and control loop gains are confirmed (based on test moments of inertia). Solenoid action and pressure delivery system are tested and confirmed.

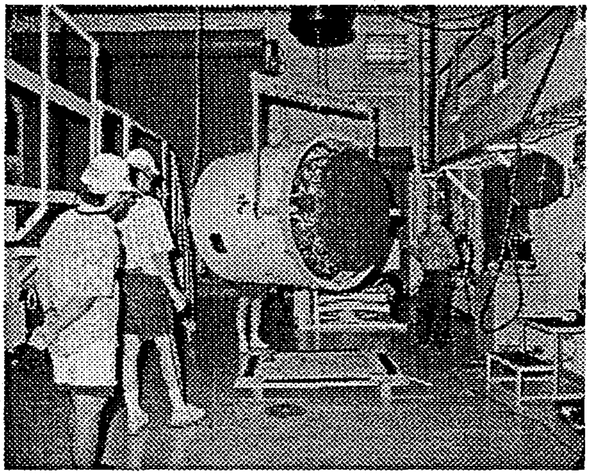

Figure 4. Air Bearing Testing of the ACS

Next, the 3rd stage autopilot is tested. The setup for this test is basically the same as the PBV autopilot except that extension umbilical cables are connected from the base of the PBV to the 3rd stage structure. Also, an HSTM 3rd stage motor (actual motor with inert motor grain) is loaded into the 3rd stage structure. The correct GN&C mode is switched to and the motion is converted into correct pitch and yaw actuator deflections and roll ACS commands with nozzle firings. Actuator performance is also determined during this sequence of testing to evaluate TVC flight acceptance or to evaluate acceptance procedures.

The 2nd stage autopilot is tested next. This system is configured with the PBV on the system test stand and the umbilical to the 3rd stage and then passed through to the 2nd stage. This test produces servo motion of the fluid injectors with gaseous nitrogen at about 180 psi.. This test proves functionality but does not test servo gains since the flight system utilizes freon (liquid) at about 750 psi. However, the test does confirm all flight hardware/software and telemetry hardware is functioning correctly.

The 1st stage autopilot is now tested. This system continues the hookup from 2nd stage to 1st stage. The testing is very similar but tests the gains of the 1st stage autopilot code and also the servo loops and hydraulic actuators of the 1st stage nozzles. This system works very well, however, the actual flight environment operates under a considerably different nozzle load due to engine thrust passing through the nozzles. The actuator bandwidth is evaluated and compared to data derived from loaded actuator tests.

Next the preliminary system test (hardware-in-the-loop flight sequence checkout) is run (see figure 5.). This test is the culmination of all the testing so far. The system is configured with the PBV on the test stand with all stages (i.e. TVC) connected. The system is now fully TVC and ACS functional. Next, the main reason for the test, the flight sequence, is confirmed by placing bridge wire simulators in all connectors to squibs or initiators. The test is run by the GLC, the onboard flight computer and A&F sequencer. The GLC sends correct navigator modes and the initial FIDU (launch discrete) to start the on board sequence. The GLC also initiates motion of the system test stand at prescribed times in the flight sequence to provide IMU input and times to test TVC modes and functionality. The corresponding bridge wire simulator firings at the correct time confirm all systems are functional and working per mission requirements.

At this point the missile system is checked out and prepared for payload integration.

The payload interfaces are completely checked for

Figure 5. STARS Vehicle Testing

continuity and electrical signals. The payload interface hardware is then mounted to the payload plate on the PBV. The payloads or simulators are mounted next. Note that this is not the final payload integration. The payloads are then checked out through the use of payload groundstation hardware and powered by the GLC. Once all payloads or simulators are checked the full system is ready for testing.

Another system test is run which effectively is the same as the previous except that the payload sequencing and firing are also checked per the mission timeline and requirements. This concludes the pre environment system testing of the integrated vehicle.

Independent of the Albuquerque testing, the first & second stage motors are refurbished at their respective manufacturers and delivered to China Lake NAWC for Betatron on the 1st and 2nd stage motors and Linatron or computed tomography (CT) scan on the 1st stage for motor grain, insulation and bond information. Components for the TVC systems of the 1st and 2nd stage are certified by a number of methods and shipped to Hill AFB for integration. Once the motors are delivered to Hill AFB, the thrust vector control (TVC) system actuators and related TVC and FTS components are mounted and preliminary checkouts run. This is the final test of the flight motors performed prior to shipment to the Kauai Test Facility (KTF).

Environmental Testing

At this point the system is checked out and working as designed and required. Now the system is ready for vibration & shock testing of the integrated 3rd stage and PBV with Payloads. The stacked 3rd stage, PBV and payload are installed onto the shaker at the Sandia Environmental Test Facility. This system is configured to test independently roll, pitch and yaw axes. Each axis will be tested at varying levels of the vibration specification and also, a shaker shock applied to each axis. The system is functioned electrically through vibration but typically only before and after shock due to the nature of the environment (survival test).

The Pyro-shock test of a mock 3rd stage, PBV and Payload is now performed. This test stacks the PBV and Payload onto a mock 3rd stage with an actual 3rd stage-PBV separation joint installed. The system is functioned electrically prior to and after the test. The test fires the separation joint and actually cuts the metal and transmits a flight like shock through the system.

Software Validation

Flight Software validation is required to provide quality control, test and evaluation of the flight code. In some cases this environment provides the only flight like test of the software. The flight software is functioned in an identical Sandac Flight Computer with a flight like A&F sequencer, PCM encoder, GPS receiver, 1553 bus and other components which are then hooked into the simulation computer through a custom interface. This computer simulates the missile dynamics and all TVC actuators and other system models. The software is then run through a significant number of parametric runs to test robustness of flight control and other flight sequence software. For the software to be flight certified it must be run through this process after the last software modification is made.

Once the environmental testing is completed and the system is deemed flight worthy, a final system test in Albuquerque is run. This test effectively repeats the test prior to environments to confirm all systems are still in a flight worthy working order.

After the hardware is deemed flight worthy and ready to ship to the launch facility a preshipment review or Vehicle Readiness Review (VRR) is held. This meeting allows the presentation of vehicle status, anomalies and waivers. At this time all interested parties evaluate the status and determine readiness to commit to the field. The missile, payload and all support equipment are now shipped to KTF/PMRF.

Testing and Integration (Kauai)

The entire shipment of flight motors, ordnance, booster hardware, payload and support equipment are shipped via Military Airlift Command (MAC) to the Pacific Missile Range Facility (PMRF). The day of shipment all items are offloaded and put into the respective buildings for assembly, test and storage at the Kauai Test Facility (KTF) (see figure 6.).

Initial Checkout and Preparation

Basically, the Albuquerque checkout procedures are performed again to confirm all systems are ready for launch. Also, some components are tested for the first time due to the ordnance installed and other factors.

The PBV and 3rd stage are checked out electrically to confirm all components and cables are mounted and connected per procedure. Also, serial number and component identification. are recorded in the Missile Build Book for completeness and acceptance confirmation. This is a non powered test configuration.

The flight Inertial Measurement Unit (IMU) is gyrocompassed and evaluated independent of the flight vehicle in the GN&C lab of the Missile Assembly Building (MAB). This testing is performed on a surveyed granite table and evaluated prior to installation into the PBV.

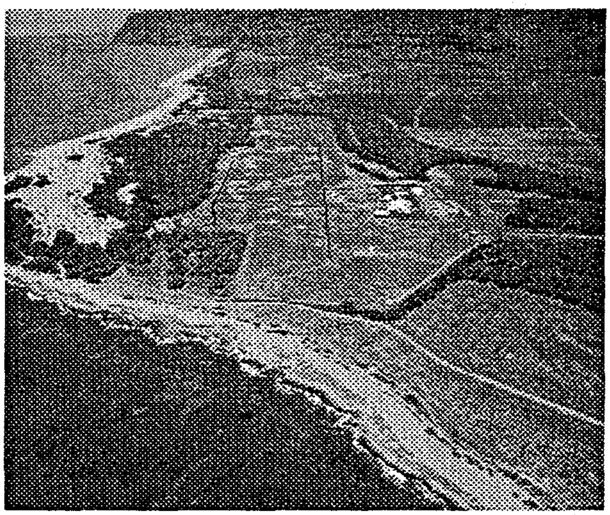

Figure 6. The Kauai Test Facility (KTF)

In parallel, the MAB ground support equipment is checked out and any modifications are made prior to connection to the flight vehicle.

Also, the Launch Operations Building (LOB) systems are configured for the flight vehicle. The missile personnel configure the STARS racks which contain the GLC, Booster TM, Flight Termination System (FTS), GN&C and support PC's and control consoles. These consoles communicate and control operation of the missile and support hardware in the MAB and Missile Service Tower (MST)/Auxiliary Equipment Building (AEB). The payload personnel configure payload TM, control and battery charging hardware and other support equipment in their respective position in the LOB. The range personnel continue with range support and range configuration which includes FTS transmitter, TM receiving and distribution along with all range support functions.

Testing & Evaluation (Kauai)

The PBV EMA's are calibrated on a test stand with independent measurements corresponding to the data acquired by the flight computer. This forms the calibration utilized by the flight software. Next, the PBV is mounted on the system test stand and the PBV autopilot procedure is performed as in Albuquerque.

Next, The ACS testing will be performed. The PBV is mounted to the 3rd stage structure (which has a 3rd stage mass mockup installed). And the ACS system is prepared by filling the high pressure bottles with Nitrogen to the prescribed pressure. This testing confirms the polarity of the nozzles (solenoids) and the operability of the system. Then the integrated vehicle (PBV and 3rd stage) is hung from the crane hoisted air bearing and single axis maneuvers are performed. This confirms solenoid delays, control loop gains and basic functionality of the complete system, including power, TM, control electronics, battery operation, etc.

The 3rd stage motor, Orbus I, is prepared and installed into the 3rd stage structure. Thermal insulation is checked and modified if required on the underside of the 3rd stage structure or around the base of the Orbus motor. The PBV is then remated to the system test stand. Umbilicals are connected from the PBV to the 3rd stage and the system is ready for 3rd stage autopilot testing. From now on all powered tests will be run remotely from the LOB, due to the hazardous nature of powering electronics in close proximity to explosives. Since the test is remote, video cameras with fiber optic feeds to the LOB are utilized for visual verification of TVC polarity along with telemetry. From now on video cameras will be used extensively.

The 2nd stage TVC and autopilot testing is now performed. This test requires umbilicals from the 3rd stage to the 2nd (since the booster is not being stacked yet). A custom designed kite stick with pressure switches is installed in each nozzle and connected to a strip chart recorder. This provides independent verification of the nozzle TVC polarity along with the TM data. The custom kite sticks are removed and the 2nd stage autopilot procedure from Albuquerque is performed remotely.

The 1st stage autopilot testing is done next. An umbilical is installed from the 2nd to the 1st stage. This completes the full hook up of the system downstage (all TVC's connected). Video cameras are set up looking back at the nozzles of the 1st stage.

This video will provide the independent verification of the actuator polarity along with TM. Next, the 1st stage autopilot test is run as before, remotely.

With the system completely connected electrically, a preliminary system test is run to confirm flight and ground hardware and software are working correctly. Next, the payload interface hardware and payloads or simulators are installed and the system is readied for the full up system test. A dry run system test is run to confirm all systems are functioning correctly and all personnel are aware of their respective duties. The bridgewire simulators are loaded and readied for tests. The system test procedure is then run as before, except remote, and all flight systems are verified with the flight timeline.

Now that the system has been verified it is ready to be integrated for flight.

Missile Final Integration

All remaining ordnance needs to be installed at this time, typically prior to integration. This include mild detonating fuze (MDF) and initiators for skin separations, initiators for motor ingiters, etc.. these operations are performed with a minimum of required, authorized personnel. Also, during this period, no other operations are performed in the MAB.

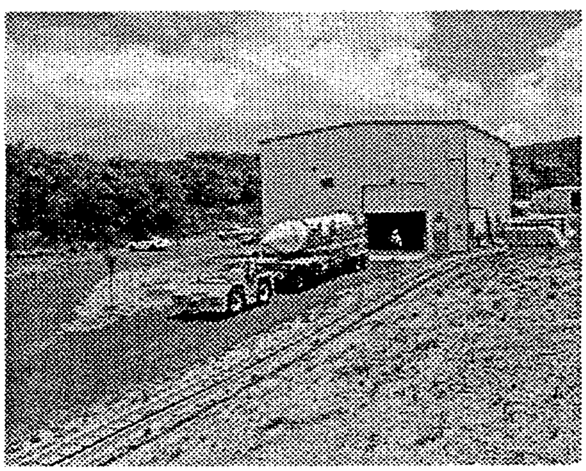

Figure 7. Missile leaving the MAB

Once the ordnance is installed, the 1st and 2nd stages are aligned and mated per procedure on the transporter erector.

Next, the 3rd stage is mated to the front of the 2nd stage while on the transporter erector. At this point the STARS lifting shroud is installed and the booster is completed and ready to be moved to the pad and erected on the launch stool.

Payload integration is now performed. The PBV is moved to a test stand which is better suited for complete access to the payload plate. The payload personnel then install all flight payloads with ordnance for the final time. At the completion of this stage the integrated PBV and payload are ready to be moved to the launch pad (MST) for hypergolic fueling of the PBV.

Pad Operation Procedures

The integrated PBV and payload are moved to the launch pad and installed under the fueling tent which is connected to the MST. This operation is performed per procedure with a minimum number of personnel present due to the hazardous nature of the propellant and oxidizer. Once the fueling is completed, the fueled PBV is moved to the environmentally controlled shelter connected to the AEB. The fueling tent is taken down and the pad area prepared.

The MST is removed from the launch stool and the pad area prepared for the missile to be erected. The integrated 1st/2nd/3rd stage with the lifting shroud is moved to the pad on the transporter/erector (see figure 7.). This vehicle is moved to a prealigned location along with a crane and man lift. A coordinated lift is performed once the missile is erected with the transporter/erector and the crane is attached to the missile shroud. The missile, once free of the erector, is placed on the launch stool within a degree of the desired roll angle orientation (see figure 8.). At this point the missile has been placed and the

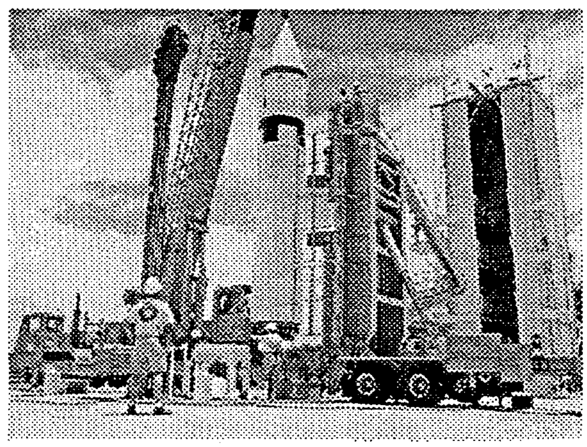

Figure 8. Missile being erected on pad.

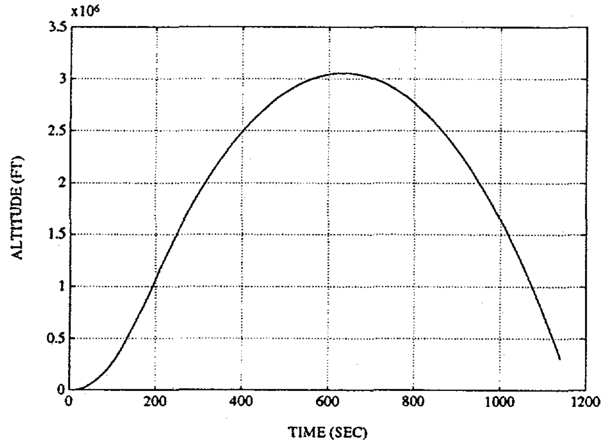

Figure 9. Altitude Versus Time

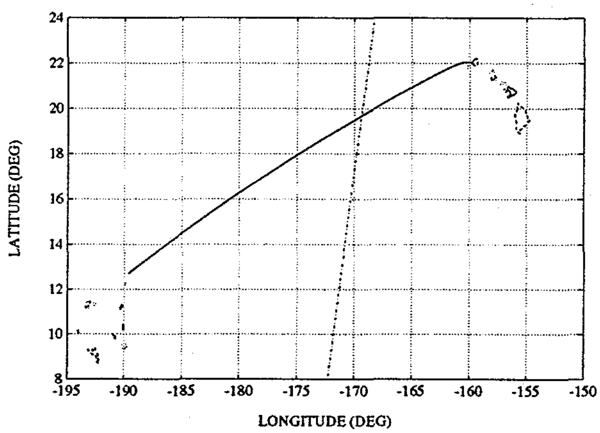

Figure 10. Ground Track

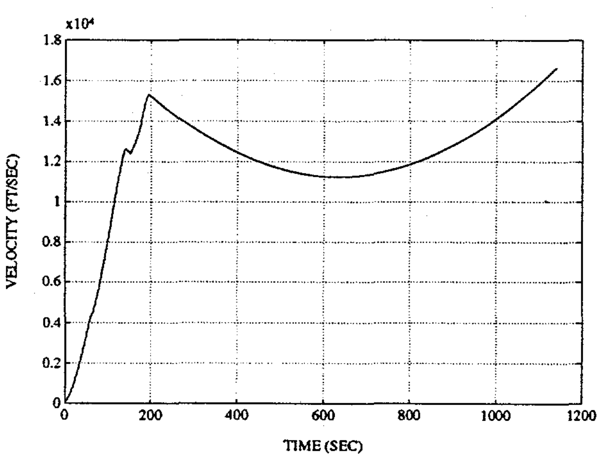

Figure 11. Total Velocity Versus Time

MST is placed back around the missile for environmental control and to provide access to work levels.

Once the missile is erected and the MST is in place, the integrated and fueled PBV with payload can be lifted and mated to the missile. First, the lifting shroud is removed which exposes the top of the 3rd stage. The PBV is then lifted with the correct handling gear utilizing the MST crane. The PBV is then mated (umbilicals and in-flight disconnects (IFD)) and all attach hardware installed. Now the system is fully functional.

After the payload personnel are satisfied with the final installation of the payload the bottom cylinder and clamshell shroud half are installed. This allows access to the payload plate and payloads and also provides some protection to the payloads. Also, the shroud is nearly completely installed.

With the system fully integrated and all umbilicals attached, the system is ready for post mate integration testing. These tests will exercise all systems which are feasible and determine that all systems are flight worthy. This test may span a number of days and actually be a set of independent tests, however, they all have the same goal, confirm flight worthiness and the ability to proceed toward launch.

Flight worthiness determined to be satisfactory, the decision to perform the final ordnance system hook ups is made. This does not necessarily mean arming the missile, but it does move the safing point to the arm & safe plugs. The system is ready to begin dry run count downs at this time.

Count Down Preparation

All personnel at KTF involved in the actual launch now begin practice counts and preparations for launch. These counts tie all missile, payload and range systems and personnel to a common goal of the launch of the missile safely and reliably in the required launch window. These dry runs bring out rough areas which require procedural modifications and also give personnel practice to provide good confidence that all operations can be performed on time to meet the desired launch window.

The next step is to perform count downs which include the PMRF personnel and hardware/software as well. This provides the same function as the previous dry runs but checks out more required systems and personnel. This enables the confidence factor to be higher for meeting the launch window.

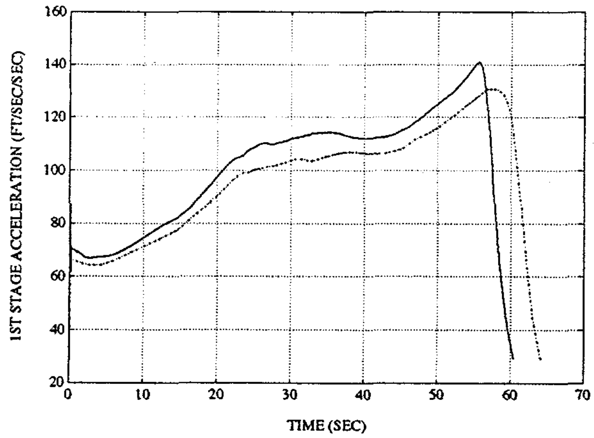

Figure 12. First Stage Acceleration

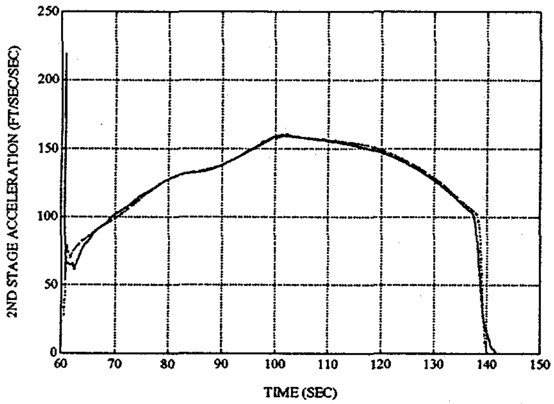

Figure 13. Second Stage Acceleration

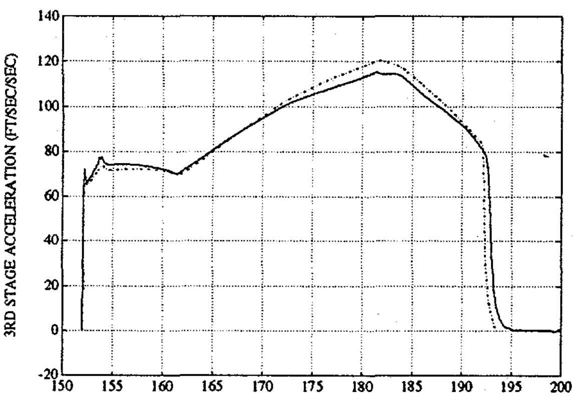

Figure 14. Third Stage Acceleration

Finally, the Mission Readiness Test (MRT), a dry run with all participants by the clock, is run. This includes KTF, PMRF, KMR, corollary sensors and others. This test is basically a launch attempt without arming the missile. All pre launch procedures are run just like the actual launch. The countdown proceeds through T-0 and simulated data is passed on to all ranges and sensors as will be passed during the actual flight. Typically, only one MRT will be run due to the significant cost. However, if problems occur, sub system tests between ranges will be performed to provide high confidence in success.

Prior to flight a Flight Readiness Review (FRR) will be held. This meeting includes all interested parties and provides a forum to present the status of all flight hardware and any anomalies. Each system is discussed and any deviation from procedure. The outcome of this meeting is a GO or NO GO for launch.

Flight Test Results

STARS Booster Performance

A simplified MDT-II flight timeline is shown in Table 1. This timeline shows most of the major events of the target mission. Most of the timing variations relative to nominal preflight predictions were due to burn time variations in the boost phase rocket motors. Mission sequencing was designed to ensure that test object deployment times were fixed relative to launch to facilitate sensor acquisition. All deployed objects were released at the prescribed nominal times, with the exception of test object #17 (MAS), which did not deploy. Apogee and reentry times were somewhat earlier (9 and 15 seconds respectively) than nominal prediction due to a slightly low boost phase performance.

The boost phase flight trajectory is shown in Figures 9-10. Figure 9 shows altitude versus time, while Figure 10 shows the STARS groundtrack for the entire mission, as well as the MSX satellite groundtrack (dashed line) for mission times between 582 and 858 seconds. The trajectory apogee was about 7.6 km less than the nominal prediction, but well above the 914 km minimum requirement. The third stage vacuum impact point was about 44 km short of the nominal preflight prediction. Figure 11 is a plot of total velocity versus time. Total velocity was approximately 29 m/s lower than predicted. These variations in the trajectory were the result of a lower than nominal net boost phase performance, but were well within the bounds established for mission success. Overall boost phase performance was within 1-sigma of nominal. Target miss for the third stage through the desired pre-apogee point was about 0.55 km.

Performance variations for each of the individual stages can be observed from acceleration data. Thrust acceleration vs time plots are shown in Figures 12-14 for stages 1-3, respectively. These figures show flight data (solid line) compared with the preflight predictions (dashed line). First stage integrated acceleration was about 1.5% lower than preflight predictions, but the burn time was 3.7 seconds shorter than nominal. The second stage integrated acceleration was approximately 0.7% low with a burn time that was 1.5 seconds long. Third stage integrated acceleration was only 0.1% low with a burn time that was 1.2 seconds longer than predicted. Each of these individual variations was within the expected performance parameters for the respective motor.

The ODES PBV performed as expected. PBV propulsion temperatures and pressures were nominal. PBV thrust acceleration was generally somewhat less than predicted preflight, although fuel useage was near nominal.

MISSION EVENT	TIME	MISSION EVENT	TIME
IGNITION COMMAND	0.0	DEPLOY TO13	371.5
BEGIN PITCH-OVER	2.2	DEPLOY TO14	373.5
BEGIN AOA CONTROL	20.0	DEPLOY TO15	376.0
1ST/2ND STAGING	60.5	DEPLOY TO16	378.0
BEGIN NIIHAU TURN	70.5	DEPLOY TO17	(380.0)
2ND/3RD STAGING	141.6	DEPLOY TO18	382.0
ACS MANEUVER	142.0	DEPLOY TO19	383.5
SHROUD JETTISON	147.0	DEPLOY TO20	385.0
3RD STAGE IGNITION	152.0	DEPLOY TO21	386.5
3RD STAGE BURN-OUT	194.5	DEPLOY TO22-TO26	396.5
PBV SEPARATION	206.5	TURN FOR BURN #1	459.0
TURN TO DEPLOY	250.0	BEGIN BURN #1	475.0
WAIT TO DEPLOY	268.0	END BURN #1	488.0
DEPLOY TO1	300.0	TURN FOR BURN #2	503.0
DEPLOY TO2	302.0	BEGIN BURN #2	515.0
DEPLOY TO3	304.0	END BURN #2	530.0
DEPLOY TO4	306.0	TURN FOR BURN #3	545.0
DEPLOY TO5	307.5	BEGIN BURN #3	557.0
DEPLOY TO6	309.0	END BURN #3	572.0
DEPLOY TO7	311.0	TURN FOR BURN #4	592.0
DEPLOY TO8	312.0	BEGIN BURN #4	618.5
DEPLOY TO9	313.0	END BURN #4	625.0
DEPLOY TO10	314.0	BEGIN COAST	625.0
DEPLOY TO11	364.0	TURN FOR REENTRY	1082.7
DEPLOY TO12	369.5	BEGIN BURN SEQUENCE	1102.7

Table 1. MDT-II Mission Timeline

<u>Mission Data Objectives [3]</u>

Much data was collected during the MDT-II target mission. Data collection objectives included:
(1) Acquire, track, and observe the ODES PBV and all deployed objects in sunlight
(2) Observe deployed objects with high signal-to-noise in sunlight with SPIRIT III radiometer sensor in selected IR bands
(3) Obtain precision tracking on ODES and deployed objects in support of functional demonstration of metric discrimination
(4) Observe unresolved target clusters as they evolve into separated individual point source images associated with each test object
(5) Test the efficacy of various deployment related techniques

(6) Observe individual deployed objects with SPIRIT III radiometer during and after solar terminator crossing
(7) Observe ODES vehicle with engines on/off at various aspect angles during and after target deployment
(8) Observe STARS II booster and upper stages during powered flight with an earth limb background and after burnout in medium and long wavelength IR
(9) Observe individual targets and penaids as they begin to reenter the atmosphere
(10) Observe the emissiive reference sphere to aid in calibration of the SPIRIT III

MSX satellite data collection was outstanding and all program objectives were achieved. Data acquistion by auxiliary sensors was also excellent. AST observed about 900 seconds of the mission and recorded IR signature data as well as tracking data. Cobra Judy acquired several hundred seconds of track time on the target complex with its X-band radar. Data reduction is in progress, and is likely to require years of detailed analysis.

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[1] "MSX Successfully Observes Dedicated Ballistic Missile Flight", Press Release, Ballistic Missile Defense Organization (BMDO) Office Of External Affairs, Sept. 13, 1996.
[2] "Strategic Target System (STARS), STARS I Handbook for Payload Designers", SM-PH-01, Revision A, Sandia National Laboratories, April 1994.
[3] "Strategic Target System (STARS M-3)/MSX Dedicated Target Mission (MDT-II) Quick Look Report", U.S. Army Space And Strategic Defense Command (USASSDC), Sept. 1, 1996.

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