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Ground Based Impact Testing of Orbiter Thermal Protection System Materials in Support of the Columbia Accident Investigation

by

Justin Hamilton Kerr

A THESIS SUBMITTED IN PARTIAL FULFILLMENT OF THE REQUIREMENTS FOR THE DEGREE

Doctor of Philosophy

APPROVED, THESIS COMMITTEE:

[Signatures]

Enrique V. Barrera, Professor, Chair
Mechanical Engineering & Material Science

Yves C. Angel, Adjunct Professor
Mechanical Engineering & Material Science

Andrew J. Meade, Professor
Mechanical Engineering & Material Science

Valery N. Khabashesku, Faculty Fellow
Chemistry

HOUSTON, TEXAS

DECEMBER 2004
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ABSTRACT

Ground Based Impact Testing of Orbiter Thermal Protection System Materials

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Justin Hamilton Kerr

On January 16, 2003, the Space Shuttle Columbia (OV-102) was launched for a nominal 16-day mission of microgravity research. Fifteen days and 20 hours after launch, and just 16 minutes before its scheduled landing, the OV-102 vehicle disintegrated during its descent. The entire crew was lost. Film and video cameras located around the launch complex captured images of the vehicle during its ascent. Of note were data that showed a piece of debris strike the port wing at approximately 82 sec after lift-off (T+82). As resulting analysis would show, the source of the debris was the left bipod ramp of the Shuttle external tank. This foam debris struck the Orbiter leading edge at sufficient velocity to breech the thermal protection system (TPS). During reentry at the end of the mission, the hot plasma impinged inside the Orbiter wing and aerodynamic forces ultimately failed the wing structure.

This thesis documents the activities conducted to evaluate the effects of foam impact on Orbiter TPS. These efforts were focused on, to the greatest extent practical, replicating the impact event during the STS-107 mission ascent. This thesis fully documents the test program development, methodology, results, analysis, and conclusions to the degree that future investigators can reproduce the tests and understand the basis for decisions made during the development of the tests.
In Memoriam

Rick Husband, commander

Willie McCool, pilot

David Brown, mission specialist

Kalpana Chawla, mission specialist

Michael Anderson, payload commander

Laurel Clark, mission specialist

Ilan Ramon, payload specialist, Israel Space Agency
Acknowledgements

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I’ve always had the fortune of knowing wonderful people in my life that have enabled me to take each step forward—primary school, a bachelor’s degree, a master’s degree and finally a Ph.D. First and foremost I thank my family. My parents, brother and extended family have always been there for me through each of these steps. They have always been so supportive of everything I do. And although he hasn’t been there
for the whole journey I thank my son Austin. He may not understand completely what I’ve been up to, but he has certainly shared in the sacrifices. I love you all.

I am blessed with several close friends who have encouraged me over the years of this endeavor. My long time friend Tom Prior has always been a positive influence in my life and has always been there to help me in times of doubt. Jeanne Crews and Eric Christiansen have been the strongest technical influences in my career. Professionally Jeanne was my mentor. She showed me the impact testing field and always encouraged me to broaden my horizons by working with others in and out of the Agency. Personally she’s a great friend and one who—even at great geographical distance—has a way of encouraging me to “get it done!!!” Thanks for your persistence and friendship and I’m looking forward to seeing you in Florida again soon. Like Jeanne, Eric has also had a strong influence on my career (and still does). Technically Dr. Christiansen is the best in the field. I learn something from him every day I work with him and personally he’s a great guy to call friend. Lisa Vidonic is a friend who I was fortunate enough to work with on a lunar curation facility project just after I finished my Ph.D. course work. In Jeanne’s absence she has been needling me to finish my Thesis. Occasional progress checks followed by fresh cookies were a great bribe to keep me moving. On a side note, I think her cookie recipe could be worthy of an equivalent epicurean degree. Thanks to all of you for your technical guidance and most of all your gentle friendly push forward.

My direct management at NASA Johnson Space Center has always supported my continuing education. Having started in the science oriented, lunar “rock jock” division they encouraged us engineers to attain higher degrees. Doug Blanchard and David Thompson helped hire me in to the Agency and continued supporting me by allowing
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In closing, this degree started out as a personal challenge. I knew I would benefit from the additional coursework (as I have), but I almost thought of it as my own personal mountain to climb. Certainly there are individual efforts associated with this degree that
parallel the athlete, but the support and aid of the people mentioned above cannot be overstated. Thanks again to all of you.

Justin Kerr
28 November 2004
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# List of Acronyms

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<th>Acronym</th>
<th>Description</th>
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<tbody>
<tr>
<td>A/D</td>
<td>Analog-to-digital</td>
</tr>
<tr>
<td>ACP</td>
<td>Analysis and control peripheral</td>
</tr>
<tr>
<td>atm</td>
<td>atmosphere (14.7 psi at sea level)</td>
</tr>
<tr>
<td>CAD</td>
<td>Computer aided drafting</td>
</tr>
<tr>
<td>CAIB</td>
<td>Columbia Accident Investigation Board</td>
</tr>
<tr>
<td>CAM</td>
<td>Computer aided manufacturing</td>
</tr>
<tr>
<td>CTE</td>
<td>Coefficient of thermal expansion</td>
</tr>
<tr>
<td>DC</td>
<td>Direct current</td>
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<tr>
<td>ET</td>
<td>External tank</td>
</tr>
<tr>
<td>FEA</td>
<td>Finite element analysis</td>
</tr>
<tr>
<td>FG</td>
<td>Fiberglass</td>
</tr>
<tr>
<td>FOV</td>
<td>Field of view</td>
</tr>
<tr>
<td>ft</td>
<td>feet</td>
</tr>
<tr>
<td>G</td>
<td>Acceleration due to Earth’s gravity</td>
</tr>
<tr>
<td>HB</td>
<td>Huntington Beach</td>
</tr>
<tr>
<td>HRSI</td>
<td>High-temperature reusable surface insulation</td>
</tr>
<tr>
<td>HVI</td>
<td>Hypervelocity impact</td>
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<tr>
<td>Hz</td>
<td>Hertz</td>
</tr>
<tr>
<td>IML</td>
<td>Inner mold line</td>
</tr>
<tr>
<td>IPSS</td>
<td>Impact penetration sensor system</td>
</tr>
<tr>
<td>IRIG</td>
<td>Inter-range instrumentation group</td>
</tr>
<tr>
<td>ISAG</td>
<td>Image Science and Analysis Group</td>
</tr>
<tr>
<td>ISS</td>
<td>International Space Station</td>
</tr>
<tr>
<td>JSC</td>
<td>Johnson Space Center</td>
</tr>
<tr>
<td>KSC</td>
<td>Kennedy Space Center</td>
</tr>
<tr>
<td>L/D</td>
<td>Length to diameter ratio</td>
</tr>
<tr>
<td>lb</td>
<td>Pound</td>
</tr>
<tr>
<td>lbm</td>
<td>Pound, mass</td>
</tr>
<tr>
<td>LC</td>
<td>Load cell</td>
</tr>
<tr>
<td>LCGG</td>
<td>Large compressed gas gun</td>
</tr>
<tr>
<td>LESS</td>
<td>Leading edge structural subsystem</td>
</tr>
<tr>
<td>LRSI</td>
<td>Low-temperature reusable surface insulation</td>
</tr>
<tr>
<td>LVDT</td>
<td>Linear variable differential transformer</td>
</tr>
<tr>
<td>M/OD</td>
<td>Meteoroid/orbital debris</td>
</tr>
<tr>
<td>MADS</td>
<td>Modular auxiliary data system</td>
</tr>
<tr>
<td>MAF</td>
<td>Michoud Assembly Facility</td>
</tr>
<tr>
<td>MCGG</td>
<td>Medium compressed gas gun</td>
</tr>
<tr>
<td>MLGD</td>
<td>Main landing gear door</td>
</tr>
<tr>
<td>mm</td>
<td>Millimeters</td>
</tr>
<tr>
<td>MRT</td>
<td>Mishap response team</td>
</tr>
<tr>
<td>MSFC</td>
<td>Marshall Space Flight Center</td>
</tr>
<tr>
<td>NAIT</td>
<td>NASA Accident Investigation Team</td>
</tr>
<tr>
<td>NDE</td>
<td>Non-destructive evaluation</td>
</tr>
<tr>
<td>Abbreviation</td>
<td>Full Form</td>
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<td>--------------</td>
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<tr>
<td>NIMA</td>
<td>National Imagery and Mapping Agency</td>
</tr>
<tr>
<td>ODIAT</td>
<td>Orbiter Debris Impact Assessment Team</td>
</tr>
<tr>
<td>OML</td>
<td>Outer mold line</td>
</tr>
<tr>
<td>OMS</td>
<td>Orbital maneuvering system</td>
</tr>
<tr>
<td>OV</td>
<td>Orbiter vehicle</td>
</tr>
<tr>
<td>OVEWG</td>
<td>Orbiter Vehicle Engineering Working Group</td>
</tr>
<tr>
<td>psi</td>
<td>Pounds per square inch</td>
</tr>
<tr>
<td>psia</td>
<td>Pounds per square inch, absolute</td>
</tr>
<tr>
<td>psig</td>
<td>Pounds per square inch, gage</td>
</tr>
<tr>
<td>RCC</td>
<td>Reinforced carbon-carbon</td>
</tr>
<tr>
<td>RCG</td>
<td>Reaction cured glass</td>
</tr>
<tr>
<td>RSI</td>
<td>Reusable surface insulation</td>
</tr>
<tr>
<td>RSS</td>
<td>Root sum squared</td>
</tr>
<tr>
<td>RTF</td>
<td>Return to Flight</td>
</tr>
<tr>
<td>RTV</td>
<td>Room temperature vulcanizing</td>
</tr>
<tr>
<td>SCGG</td>
<td>Small compressed gas gun</td>
</tr>
<tr>
<td>SiC</td>
<td>Silicon carbide</td>
</tr>
<tr>
<td>SIP</td>
<td>Strain isolation pad</td>
</tr>
<tr>
<td>SLA</td>
<td>Super light ablator</td>
</tr>
<tr>
<td>SOP</td>
<td>Standard operating procedure</td>
</tr>
<tr>
<td>SRB</td>
<td>Solid Rocket Booster</td>
</tr>
<tr>
<td>STS</td>
<td>Shuttle transportation system</td>
</tr>
<tr>
<td>SwRI</td>
<td>Southwest Research Institute</td>
</tr>
<tr>
<td>TEOS</td>
<td>Tetra ethyl-orthosilicate</td>
</tr>
<tr>
<td>TPS</td>
<td>Thermal protection system</td>
</tr>
<tr>
<td>TTL</td>
<td>Transistor-transistor logic</td>
</tr>
<tr>
<td>TUF1</td>
<td>Toughened unified fibrous insulation</td>
</tr>
<tr>
<td>USA</td>
<td>United Space Alliance</td>
</tr>
<tr>
<td>VCS</td>
<td>Vibration control system</td>
</tr>
<tr>
<td>WSTF</td>
<td>NASA Johnson Space Center White Sands Test Facility</td>
</tr>
</tbody>
</table>
1.0 Introduction

On January 16, 2003, the Space Shuttle Columbia (Orbiter Vehicle (OV)-102) was launched for a nominal 16-day mission of microgravity research. The science mission required the astronaut team to work around the clock to complete over 80 experiments that focused on the effects of microgravity on the human body. This was the twenty-eighth flight of OV-102 and the first flight of the Spacehab Research double module. Fifteen days and 20 hours after launch, and just 16 minutes before its scheduled landing, the OV-102 vehicle disintegrated during its descent. The entire crew was lost.

NASA civil servant and contractor engineers immediately began working to determine the cause of the incident. Initially, these efforts were coordinated through the Mishap Response Team (MRT) and, beginning March 24, through the NASA Accident Investigation Team (NAIT). The investigation followed contingency plans developed in the event of a mishap. Fault tree analyses of every Orbiter system were scrutinized over the coming months to rule out those events that could not be causes for the accident. Following this approach, the Space Shuttle Program systematically identified the most likely failure scenarios. More than one thousand items were ultimately closed during the course of the investigation.

1.1 Test Program Background

Film and video cameras located around the launch complex captured images of the vehicle during its ascent (Figure 1.1-1). Of note were data that showed a piece of debris strike the port wing at approximately 82 sec after lift-off (T+82). The images showed an irregular-shaped object strike the lower surface of the wing, producing a
shower of finer debris (Figure 1.1-3b). Imagery showed that the debris originated from
the –Y bipod area between the external tank (ET) and the Orbiter (Figure 1.1-2a and b).

Figure 1.1-1: Map showing the location of the cameras used to image the debris
strike
Figure 1.1-2: (a) Photo of OV-102 ready for launch. The upper and lower red circles show the –Y bipod ramp and the approximate impact location of the foam debris on the port side leading edge respectively. (b) Close up photo of the bipod ramps.
Figure 1.1-3: (a) Image (frame 4914) from Camera E212. The red circle indicates the debris object in full illumination. (b) Debris impact cloud seen on Camera E-212.

Figure 1.1-4: (a) ET-208 Composite with trajectory of debris (times are in sec after 16:15:40 UTC) and (b) Composite image showing the trajectory of the major piece of debris (Object 1) mapped from camera E-212, frames 4913 through 4922 [1].
Analysis conducted while the vehicle was in orbit found that

- The debris was most likely foam from the ET bipod ramp or flange.
- The longest dimension of the foam was 20±10 in.
- The foam impacted the lower portion of the port wing shown in Figure 1.1-3a and b at a relative velocity between 650 and 730 ft/sec [2].

The foam followed the trajectory shown in the composite images of Figures 1.1-4a and b.

Before and after imagery of the impact area did not indicate damage to the Orbiter surfaces. Engineers evaluated the effects of the foam impact on thermal protection system (TPS) materials: LI-900, FRCI-12, and LI-2200 tile and reinforced carbon-carbon (RCC). These assessments yielded the following conclusions:

- RCC damage would be limited to the silicon-carbon (SiC) coating due to the relative softness of the debris.
- Crater impact analysis indicated the potential for large tile loss.
- “Safe return [was] indicated even with significant tile damage [3].”

Because the impact occurred on the lower port-side wing of OV-102, test articles were envisioned for unique areas on its surface: wing acreage tile and structure, the main landing gear door (MLGD), and the leading edge. The materials in these areas would respond in very different ways to an impact by foam debris; likewise, a breech in these areas would yield varying Orbiter system responses during reentry plume impingement.

February to July was a period of investigation and discovery for the Agency and its contractors. As groups investigating branches of the fault tree learned more about the accident, the knowledge gathered was incorporated into the impact test program to represent the ascent impact as accurately as possible. Thus the events of the investigation
influenced the design of the experiments and their priority. For example, more refined analysis of the mission film and video yielded an improved understanding of the debris parameters such as source, size, velocity, and impact location. Furthermore, forensic evidence strongly suggesting a breach of panel 8L influenced not only a change in the test program schedule, but also the decision to impact panel 8L rather than 9L as originally planned.
1.2 Test Program Scope and Objectives

The T+82 impact was immediately considered a possible initiating event of the accident, and so required further evaluation. The Orbiter Vehicle Engineering Office chartered the Orbiter TPS Impact Test Team to conduct the tests needed to evaluate this fault tree branch. The scope of the task included pretest and posttest analyses of the impact, nondestructive evaluation (NDE) of the test articles both before and after the tests to determine their conditions, design and construction of test article hardware, modification of facilities to conduct the tests, and detailed planning of test parameters. The impact test program was developed and conducted in close coordination with the Columbia Accident Investigation Board (CAIB).

This thesis documents the activities conducted by the Orbiter TPS Impact Test Team for the OVE Office, the NAIT and the CAIB. The thesis is divided into five parts: Introduction, Test Facility Design and Development, Test Article and Projectile Fabrication, Test Program Descriptions, Results and Future Work. This thesis fully documents the test program development, methodology, results, analysis, and conclusions to the degree that future investigators can reproduce the tests and understand the basis for decisions made during the development of the tests.

The primary objective of the Orbiter TPS test program was to replicate the impact event observed on the STS-107 mission ascent to the greatest extent practical. Secondary objectives of the test program were (1) to provide empirical information supporting fault tree and scenario convergence, and (2) to compare the damage data with estimates produced with analytical tools available during the mission. The data generated from
these tests supported the closeout of the STS-107 investigation and fault tree and are available to support the development of new analytical tools for return to flight.

To achieve these objectives, given the uncertainties of impact location, four distinct impact locations were viable failure scenarios at the beginning of the investigation. Figure 1.2-1a shows these locations on the Orbiter's lower left wing. Engineers designed three test articles that accurately represented these areas both in material composition and in structural response. These test articles included the left MLGD (Figure 1.2-1b), panels representative of the general lower wing acreage (Figure 1.2-1c) and the leading edge structural subsystem (LESS) where either the RCC panels or access (carrier) panels could be impacted (Figure 1.2-1d).

To reproduce the debris and its kinematics properly, the Impact Test Team relied on data from the Image Analysis Team and results from forensics being conducted at Kennedy Space Center (KSC). At any given time, these test conditions represented the best knowledge of the NASA community. It was the test team's challenge to design a test facility and articles that allowed enough flexibility to make timely changes to the test parameters as needed.
The Wing Leading Edge, constructed of reinforced Carbon-carbon (RCC) panels

Wing Acreage Area (TPS Tiles)

Main Landing Gear Door (TPS Tiles)

(a)

(b)

(c)
Figure 1.2-1: (a) Drawing of the lower, port wing of OV-102. The arrows point to different structures in the vicinity of the debris impact location. (b) Photograph of the left MLGD. (c) Photograph of a wing acreage test article. (d) Photograph of the LESS test article.
1.2.1 Aside on impact mechanics

1.2.1.1 Definition of impact

The field of impact mechanics is very different from the more classical analysis of rigid body or statically load structures. In particular, impact events are transient in nature and can be defined simply as a mass striking some other body (target) with a relative velocity. The impact creates an impulsive load which, as shown in Figure 1.2-2, is very different from a static load. Damaging impact loads tend to have very high amplitudes (in excess of the material yield) of short duration (on the order of milliseconds to microseconds). Static loads on the other hand demonstrate no or very slight changes in their amplitude during these time scales. The impulse delivered to a body is then the change in momentum before and after the impact.

![Impact Load vs Static Load Diagram]

Figure 1.2-2: Plot demonstrating the difference between impact and static loading.
1.2.1.2 Impact damage mechanism

Consider a solid is comprised of particles. In addition to rigid body motion, impacts induce elastic (and plastic) waves in solids. These wave speeds are properties of a material. Elastic wave speed is a constant property while the plastic wave speed varies as the particle velocity in the solid. Elastic waves are propagated first and for sufficiently high impact velocities plastic waves are propagated through the solid as well (Figure 1.2-3a). The wave(s) propagate initially as compressive wave(s) and as they move through a solid they displace the particles at the velocity of the initial impactor. These displacements give rise to strains in the solid and likewise particle velocities give rise to stresses. These strains (stresses) could be sufficient to locally deform or fail the solid. Wave interactions with free surfaces reduce the stress in the solid to zero and reflect the waves as tensile waves (Figure 1.2-3b). If the tensile wave intensity exceeds the yield strength of the solid it will deform the solid. Even higher achieved stresses can fail the solid in a mode called spallation. As these waves move through the solid, interact with free surfaces and each other, the complicated wave interactions may also give rise to local compressive or tensile deformation/failure.

An elastic wave is faster than the initial plastic wave velocity, however, plastic waves increase in velocity with particle velocity in the solid. At sufficiently high impact velocities (in excess of the sound speed of the material), the plastic wave induced in the solid will overtake the elastic wave creating a shock wave. Identical to plastic waves, shock waves propagate at increasing shock velocities as the particle velocity increases. This relationship between shock velocity and particle velocity is called the Hugoniot relationship. As shocks release pressure at free surfaces, heat is yielded in the solid. This
thermodynamic response can change the phase of the material if the latent heat of melting or vaporization is exceeded in the solid.

Figure 1.2-4 shows the range of damage that can be achieved in solids under impact conditions. For low velocity impacts (<50 ft/s) elastic and limited plastic response is expected. On the other end of the spectrum, hypervelocity impacts can induce very large damages including phase changes in the solid to liquid, vapor or plasma. Appendix A includes an example of hypervelocity impact induced damage on spacecraft shielding due to a simulated meteoroid/orbital debris impact. So in the context of the greater field of impact mechanics, the STS-107 ascent impact would be considered a low velocity impact.

Figure 1.2-3: Diagrams of wave propagation in solids.
Figure 1.2-4: Extent of impact damage and the range of impact velocities and strain rates that yield damage.
1.3 **Orbiter TPS Impact Test Team**

Given the diversity of skills required to accomplish a test program of this scope, the Orbiter Vehicle Engineering Working Group (OVEWG), with the assistance of the Boeing Orbiter TPS Subsystem Manager, assembled a team of NASA civil servants and contractors from across the entire Agency. Figure 1.3-1 is an organization chart of the Orbiter TPS Impact Test Team. The TPS Subsystem Manager led the team and, in addition to providing technical and programmatic oversight, reported status of the test program to the OVEWG, the NAIT, and the CAIB.

![Organization Chart](image)

**Figure 1.3-1: Orbiter Impact Test Team structure.**

The NASA Test Director was responsible for designing the test plan and ensuring impact facility readiness. Southwest Research Institute in San Antonio, Texas, provided the ballistic facility and personnel needed to operate the range. Their expertise was also
beneficial for the modifications needed to launch foam to high velocities—a task that had never been accomplished at this scale. Boeing-Huntington Beach (Boeing-HB) provided personnel to conduct required pretest and posttest NDE of the test articles. In addition, Boeing-HB provided personnel from their laboratories group who are experts in instrumentation and testing of spacecraft structures. Their skills enabled sensor selection and the design of sensor location and installation that would yield the needed material response data for the tests. Furthermore, the group provided a mobile data acquisition trailer and the personnel to operate the facility, allowing the rapid collection and dissemination of data from the remote test site. The Lockheed Martin Michoud Assembly Facility (MAF) was responsible for delivering all required foam projectiles for the impact test. All of these projectiles were hand-sprayed BX-250 foam similar to that applied to the ET bi-pod ramp. The projectiles were machined to sizes approved by the OVEWG, the NAIT, and the CAIB.

The entire team, including representatives of the CAIB, communicated frequently (once or more per day), sharing the latest data and status of test article and facility readiness. All members of the team participated in defining the details of the test program that was carried forward to the Space Shuttle Program Office. All data collected during testing (instrumentation, still photos, and high-speed video imagery) were distributed to the investigation community via a specially designed website.

Test article design and fabrication was truly a multicenter effort. Tile resources were manufactured specifically for these tests and bonded to test article surfaces at KSC. Instrumentation embedded in the MLGD and wing acreage articles was installed there as well. A team of engineers traveled to the Smithsonian Institution in Washington, D. C.,
and removed a number of components from the Orbiter *Enterprise* vehicle (OV-101), including the left MLGD, six fiberglass and two RCC leading edge panels, and all of their corresponding attach fittings and hardware. The MLGD was refurbished to include embedded strain gages on its outer surface and 120 new LI-900 tiles. In addition, a rigid frame was constructed to fix the door in place during testing in a manner similar to the door on the vehicle. Two honeycomb panels representing the lower wing skin structure were manufactured at JSC and shipped to KSC for instrumentation and tile installation. These wing acreage articles were built to fit on a rigid strongback and through a load cell interface to the test stand. Construction of the LESS structure was a Herculean effort. In less than 5 weeks, a representative section of the spar with its attach fittings was constructed nearly from scratch. The entire assembly was designed and built at JSC, but benefited from the guidance and technical capabilities of KSC engineers and technicians. All of the test articles were designed and constructed by early May to allow a timely investigation.

The last element of the test program was the definition of representative impact conditions. Inputs from the Image Analysis Team helped define the material, size, shape, and rotation rate; the Transport Analysis Team evaluated the trajectory of the debris. These data defined the impact angle and velocity for possible Orbiter impact locations. The analysis teams generated results iteratively—refinements in imagery analysis delivered to the transport team in turn generated improved impact condition fidelity. Forensics would ultimately prove to minimize any remaining uncertainties.
1.4 STS-107 Launch Impact Event Analysis

1.4.1 Ascent Imagery Analysis

Immediately following the accident, the OVEWG quickly organized a team of film and video analysts from three NASA centers—JSC, KSC, and MSFC (Marshall Space Flight Center)—and two “black ops” facilities—Lockheed Martin Data Systems and the National Imagery and Mapping Agency (NIMA). The team was chartered to assess and analyze all available STS-107 mission imagery from ascent, orbit, and entry. The team’s objective was to provide insight into the condition of the Orbiter and the events leading to its breakup through imagery processing and analysis. The following conclusions were drawn in their final report [1]:

- The visual evidence implicated the External Tank -Y bipod ramp as the source of the debris.
- One large piece of debris impacted the underside of the left wing. There was no conclusive evidence of other impacts.
- The size of the debris was approximately \(24 \pm 3\) in. \( \times \) \(15 \pm 3\) in.
- There was no visible evidence of damage to the left wing.
- The debris was observed to tumble, with an estimated rotation rate on the order of 18 cycles/sec.
- Impact was on the underside of the left wing leading edge, in the area of RCC panels 5-9, with most likely impact in the area of panels 6-8.
- Calculations of the debris velocity at impact ranged from 625 ft/sec to 840 ft/sec depending on the various methods and assumptions used, with the most probable velocity estimated to be approximately 700 ft/sec.
• Within the post-impact debris cloud were distinct but unidentifiable objects. The sizes of two of the objects were measurable, estimated to be 12 in. by 11 in. and 7 in. by 7 in., respectively.

Analysis of the imagery acquired on-orbit found no visual indication of damage or anomalies to the Orbiter during the orbit phase of the mission.

1.4.2 Forensic Examination of the Columbia Debris

Meanwhile, as the Image Analysis Team was conducting its work, teams of federal, state and local emergency management and law officials were scouring the states of Louisiana, Texas, New Mexico, Nevada, and California for Columbia debris. Some 84,000 pieces were recovered, cataloged, and reassembled at the Shuttle Landing Facility at KSC. A team of material and process engineers was tasked to “listen to what the hardware is telling us” so that the failure location and cause might be determined [4, 5].

Initially the team focused on the port-side main landing gear and its door. However, examination of the door frame, the landing gear piston, and the landing gear cylinder proved that plasma penetration of the wheel well occurred before the landing gear door opened and the gear itself dropped. A payload bay right-hand sill longeron fitting made of titanium showed melting, which suggested that an oxygen fire may have occurred in the payload area. Analysis of the fitting showed that the damage that occurred upon Orbiter breakup was local. Because what appeared to be a debris cloud was produced during the impact, tiles located on the lower carrier panels or on the acreage near the leading edge were considered a possible failure scenario. Likewise, it was considered that a strike on a lower carrier panel could cause it to mechanically fail at
its attach points, exposing a clear plasma path to the wing spar. All recovered tiles and carrier panels were examined and all mechanical and thermal damage were recorded. These too were ultimately eliminated as possible failure locations.

Initially ten active failure scenarios were identified. By mid-April, eight of the scenarios had been closed. Scenario 2 was entitled “RCC panel damage (panels 5-9).” Work conducted by the forensics team ultimately narrowed the scope of the likely RCC panel damage to panels 8L to 9L. Figure 1.4-1 shows laser scans of the lower surfaces of the reconstructed left side RCC panels.

Figure 1.4-1: Laser scans of the reconstructed left side RCC panels 5L-10L [6].
1.5 Flight Environment

When Columbia was impacted by foam launch debris at T+82 sec, it was clearly not in a static environment. The vehicle was at an altitude of 65,858 ft and was traveling at Mach 2.46. External loads induced on the Orbiter could have influenced the response of the TPS when it was impacted by the foam debris. The temperature and pressure of the foam at altitude could load the TPS in a very different way than foam would at room temperature and sea level hydrostatic pressure. An additional variable to consider was if variations in the foam knit line orientation would change the loading of the TPS. Knit lines are the surfaces between successive applications of spray-on foam insulation. See Section 1.6.6 for a detailed description. In the final analysis it was determined that none of these variables, combined with the debris impact load, would induce significantly different damage in the part than that caused by a foam impactor at ambient conditions. The following sections discuss why these flight environment conditions did not need to be simulated during the impact tests.

1.5.1 Ascent Loading

Boeing structural engineers did a complete assessment of whether to include ascent loading during impact testing early in the test program development [7]. During launch, the vehicle is susceptible to static (accelerations and external and internal pressures) and dynamic loads (vibration and acoustics). At T+81 sec, the vehicle is traveling at Mach 2.46 and the ambient pressure at altitude is 1.1 psi. The Orbiter was designed and certified to withstand a known set of loads including pressure, acoustic, and vibration. The internal (compartment) pressures and external pressures that contribute to
the loads on the vehicle are a function of Mach, dynamic pressure, and entry vehicle angles $\alpha$ and $\beta$ (alpha and beta). The aerodynamics of the vehicle induces acoustics that impinge directly on its surfaces. High frequency vibrations of the wing in turn shake TPS components.

Orbiter thermal tiles are bonded to a strain isolation pad and in turn to the inner mold line skin using RTV (see section 3.1.1.1 for details). The bond and strain isolation system is designed to withstand the accelerations, vibrations, and acoustics encountered during ascent. At T+81 seconds the pressure applied to the tile surfaces is no more than 1 psi, which is well within the capability of the bond and much smaller than the load induced by a foam debris impact. Maximum loading conditions on the tile are produced when aerodynamic shocks impinge on the surface; however, at T+81 seconds these shocks are well aft of the wing. For these reasons ascent loading conditions did not need to be simulated during any thermal tile impact testing (wing acreage or MLGD).

The ascent load analysis for the LESS is a bit more complicated than that for tiles because the RCC composite panels are attached to the LESS by spar fittings and attach hardware. The entire LESS is structurally isolated from the wing itself; thus the internal loads have local, directly applied causes (random vibration is an exception). The critical design cases for the LESS occur earlier on ascent at lower Mach (1.05 for certification case vs. 2.46 at T+81 sec) and at high dynamic pressure (763 lb/ft$^2$ for certification case vs. 490 lb/ft$^2$ at T+81 sec). The thermal and acceleration loads are small because the impact occurred before ascent heating and the mass (37 lb) and the acceleration (approximately 2 g’s) were low. The compartment loads across the RCC panel lower surface (net pressure opposing the impact force) were low compared to critical
certification cases (2-4 psi for certification case vs. 0-0.5 psi at T+81 sec). Analysis of dynamic pressure loads on a panel at T+81 seconds shows the stresses are <10 percent smaller than those generated by a foam debris impact. An analysis of vibration and acoustic loads on a panel show the induced stresses to be orders of magnitude smaller than the typical maximum allowable stress for the RCC material.

By comparing these loads—static, dynamic, and vibro-acoustic—to typical material allowables, one may calculate an effective reduction in material allowables due to the influence of the loads. The results show that the ascent loads are small—1 percent reduction in the panel, 2 percent in the ribs and 5 percent at the attachments. Furthermore, the evaluation proves that the effect of ascent loading is small compared with other uncertainties such as material allowable variation and the impact conditions themselves. Given these arguments and the fact that implementing the ascent loads during testing could compromise the test, it was determined not to simulate their effect.

1.5.2 Temperature and Pressure Effects on Foam Loading

At T+81 sec, the Orbiter was at an altitude of 65,858 ft, where the ambient pressure and temperature were 1.1 psi and -88°F respectively. The bipod ramp foam temperature is a gradient ranging from ambient temperature on the exterior surfaces to approximately -400°F nearest the tank. During test plan development, there was concern about whether temperature and pressure of the foam would influence the impact test results. The need to perform the tests in a vacuum with a chilled projectile would greatly extend the schedule. To determine whether these parameters needed to be included in the test program, a series of tests was conducted at Glenn Research Center. The details and
results of this testing are documented in the final report “Impact Test Results of BX 250 in Support of the Columbia Accident Investigation,” dated July 17, 2003 [8].

BX 250 right circular cylinder foam projectiles were fabricated at the MAF M&P laboratory to a final dimension of 1.25-in. diameter by 3-in. long. The projectiles were launched using a compressed air gun at a 0.5-in.-thick steel plate. Four piezoelectric load cells were attached between the plate and its rigid support. The target mount could be tilted to achieve various impact angles. The following foam impact conditions were varied: impact velocity (700 to 800 ft/sec), impact angle (10, 15, 23 and 90 deg), ambient pressure (1 atm to 1 psia) and foam temperature (<0 and 74°F). The impact event was recorded using high-speed video cameras and the load transferred to the target plate was measured via the load cells. The results showed that the load transferred into the target did not vary with foam temperature and pressure. The foam tended to break up after rebound from the target plate in a vacuum, but not in an atmosphere; however, this phenomenon occurs after projectile/target interaction and so does not affect the target loading. For these reasons, the targets in this test program were impacted in an atmosphere and with foam projectiles at ambient temperature.
1.6 Development of Test Conditions

In order to replicate the STS-107 ascent impact to the greatest extent possible, critical test conditions were established. These included debris size, impact velocity, impact location, local impact angle, debris rotational rate, and clocking angle. Figure 1.6-1 shows how the imagery analysis and forensic evidence matured with time. As planning progressed, the test team responded to changes in test conditions by fabricating new barrels and developing new techniques to launch the projectiles, modifying facilities, and managing the logistics of testing various test article priorities. The timeline shows critical milestones in the testing effort compared with test parameter refinements.

The projectile sizes for MLGD testing were finalized on or near April 1. Of critical note is the recovery of the Modular Auxiliary Data System (MADS) data recorder on March 19. By April 8, data from the recorder sensors were distributed to the fault tree teams. Shortly thereafter, both the forensics and the imagery analysis teams had narrowed down the impact location to the LESS panels 5L through 9L; by May 5, both groups were pointing strongly to panel 8L. This shows that the MLGD testing was conducted after the investigation community was very confident the impact location was the LESS. The reason for conducting the MLGD tests was to fully demonstrate the impact test facility and data acquisition systems. In the interests of time, the NAIT and CAIB decided that testing of the LESS RCC components should have the highest priority and other impact tests would proceed on a basis that would delay the RCC component tests. As testing progressed on the MLGD, it was clear to the investigation teams that the performance of both the facility and the test team had been fully demonstrated. On May 6, the OVEWG decided that the test team and facilities were ready to begin LESS testing.
and that the MLGD test program would end after test 5 was conducted. Furthermore, the wing acreage articles were not tested because they were not in a working scenario.

The LESS tests were conducted during June and early July. Fiberglass panels were impacted at varying locations to evaluate panel strains and establish the best location to impact the RCC panels to achieve failure. In particular, the desire was to establish whether a flow path of the size predicted by aerothermal analyses could be produced in the RCC. The following subsections describe how the critical test parameters were developed for both test programs.
1.6.1 Projectile Material and Size

During the mission, imagery analysis indicated that the source of the debris was the -Y bipod ramp. Additional analysis after the mission proved this to be the case. The ramp is made of a foam material called BX 250 with a nominal density of 2.4 lb/ft³ (see section 3.3 for additional foam details). The bipod fitting, which the ramp covers, is also covered with a higher density (approximately 17 lb/ft³) insulator called SLA (super lightweight ablator). The higher density material could have been shed from the fitting along with the ramp, although MAF personnel have described this event as highly unlikely. The amount and distribution of SLA for such an occurrence is unknown. Clearly, the higher density material would be a more damaging impactor than the BX 250; however, it was decided to use projectiles made only of BX 250. This would yield a "best case" impactor and would allow for the evaluation of BX 250 effects on the TPS. Higher density debris material impact tests are planned for Shuttle Return to Flight (RTF) activities (see section 5).

Figure 1.6-2 is a cutaway drawing of the bipod ramp and its associated fittings and hardware. The NAIT and CAIB determined three projectile sizes while defining test parameters. Each size and shape was based on a different rationale. The debris dimensions were prescribed to have a rectangular cross section with dimensions matching those of the gas gun barrel and a length that was aligned with the launch vector. The length of the projectile was cut to the required projectile mass for each test (due to slight foam density variations). For all tests, the projectiles were sized to fit snugly to the gas gun barrel. Since new barrels had to be constructed for each debris cross section, the achievable debris cross sections were limited to those available in off-the-shelf structural
steel tubing. For this reason, the desired projectile cross section did not match the imagery analysis results exactly. The closest available cross sections were used and the mass was matched exactly by modifying the projectile length.

![Diagram of bipod ramp and associated fittings](image)

**Figure 1.6-2: Cutaway of the bipod ramp and associated fittings [9].**

The smallest projectile size was evaluated by MAF personnel by producing computer-aided design (CAD) replica drawings for each STS mission that exhibited ET bipod foam loss. The CAD program calculated the weight of debris for each debris event. Worst-case bipod foam loss was from STS-50, with a total volume of 700.5 in.\(^3\) and a weight of 0.97 lb based on a foam density of 2.4 lb/ft\(^3\). Based on this analysis and the most likely failure mechanism (confined pressure in or under foam), MAF personnel presented to the OVE and CAIB a probable debris weight of 0.75 to 1.0 lb. Later, the OVEWG requested MAF to supply the maximum reasonable debris size, which MAF determined to be 1.19 lb. Details of these analyses are documented in the *Columbia*
Accident Investigation Board Report, Vol. II, Appendix D.8. The nominal dimensions of the case 1 projectile were 3.5 in. x 11.5 in. x 21.25 in.

The medium-sized projectile had dimensions of 5.5 in. x 11.5 in. x 19 in. and a mass of 1.67 lb. This size is representative of the medium-sized debris assessed during the mission (6 in. x 10 in. x 20 in.).

The largest projectile had a mass of 2.5 lb and dimensions of 5.5 in. x 11.5 in. x 28.5 in. Representing the mean debris size estimated by the imagery analysis team, this size was considered the most representative projectile by the CAIB during the MLGD testing. Based on the results of the transport analysis, the CAIB updated the most representative projectile size to the medium case for the LESS test program.

1.6.2 Impact Location and Angle

Multiple sources of evidence ultimately identified the debris impact location during STS-107 ascent—panel 8L of the LESS. At the beginning of the test program, no finer resolution than the lower left wing was identified as the impact location. Therefore, test articles were designed and constructed for the MLGD, wing acreage, and LESS. For the purposes of the MLGD testing, the specific impact location on the door was unimportant because the failure mode being investigated was tile cratering damage and the variations in the door structure were decoupled from the tile response by the strain isolation pad (SIP). Various impact angles and projectile sizes were tested to examine potential tile damage. The specific locations on the door were selected so that new, unaffected tiles were impacted for each test. The impact angles were limited to those that could be attained during the mission as evaluated by transport analysis.
For the LESS testing, imagery analysis did not precisely determine the impact point on the panel, but it did prove that the impact occurred on the lower surface. The forensic evidence suggested a flow path into the wing on the lower, downrange side of panel 8L. Thermal analyses were conducted for different size holes in panel 8L in an effort to duplicate temperature sensor readings recorded by the MADS. A circular hole of 10 in. diameter produced the most consistent results. To meet the test program primary objective, LESS fiberglass panels were tested to help find the impact location that would best reproduce the damage thought to have been incurred during the mission. The impact angles for LESS testing were predominately a function of the impact point. The local impact angle on a panel is a result of the debris velocity vector for the panel location (determined through transport analysis) and the location on the panel where the projectile leading edge first strikes.

1.6.3 Accounting for Projectile Rotational Effects

Imagery analysis showed variations in color of the debris, suggesting that the debris which impacted the wing was rotating. On April 1, 2003, the Image Analysis Team reported to the OVE that the debris was rotating at a rate of 18 Hz. To account for all of the energy that could be imparted into the TPS by the debris, it was important to include the energy associated with both its translation and its rotation. At the time of this test program, techniques for launching foam accurately on target and at a prescribed rotational rate had not been developed. Dr. James Walker, at the request of the CAIB, performed a study to account for projectile rotational effects by keeping the impact
velocity constant, but increasing the local impact angle [10]. Walker constructed a Lagrangian model of the RCC panel and conducted a number of impact calculations for various conditions. In particular, achieved stresses in the panel and downrange rib were compared for calculations including a projectile rotating at 18 Hz and for nonrotating projectiles with increased local impact angle. The results showed that for a particular impact site, the effect of the 18 Hz rotation could be accounted for by increasing the local impact angle by 5 deg. Similar analysis of tile impacts at an 8-deg impact angle showed that an 8-deg increase in impact angle and a shortening of the projectile by one half would account for a rotational rate of 30 Hz. For the purposes of the test program, no adjustment to the impact angle was made for the MLGD tests; however, a 2.5-deg increase in the impact angle was made to account for projectile rotation for each of the LESS tests. Although a foreshortening of the projectile is required to best approximate the effect of projectile rotation for tile impacts, the same is not required for RCC impacts. The reason for this is that RCC failure is a function of the maximum load applied to the panel, not the duration of the load. Projectile load duration is a function of projectile length. Additional work must be conducted to evaluate the loading contribution for rotating projectiles.

1.6.4 Clocking Angle

Figure 1.6-3 shows a parameter called the clocking angle. This angle describes the orientation of the projectile leading edge surface relative to the target surface. A 0-deg clocking angle is achieved by orienting the 11.5 in. debris leading edge parallel to the Orbiter x-z plane, whereas a 90-deg clocking angle is achieved by orienting the edge
perpendicular to this plane. For test purposes, the Orbiter x-z plane was parallel to the ground.

Figure 1.6-3: The clocking angle of the projectile in relation to the target.

The evaluation of the clocking angle at impact was indeterminate from the imagery analysis. Worst-case clocking angles were established for each TPS. For the tile impact tests, it was anticipated that a 90-deg clocking angle would result in a narrower, longer and potentially deeper cavity than the 0-deg case. Under flight conditions, the resultant cavity would experience higher temperatures due to higher heating factors and a
reduced view to space. The first two tests of the MLGD series were designed to experimentally verify this hypothesis. The test which yielded worst-case damage from a thermal perspective would be used for all remaining MLGD testing. For LESS testing, the clocking angle was varied during fiberglass panel tests to maximize the strains achieved in the panel (and thus the potential damage to the RCC panel tests). Section 4.2 explains the rationale for changing the clocking angle for the LESS tests.

1.6.5 Impact velocity

Impact velocities of the STS-107 ascent debris measured through imagery analysis ranged from 610 to 840 ft/sec. Transport analysis techniques were employed to evaluate the impact velocity on the Orbiter lower surfaces [11]. The impact velocity varies with the impact location and assumed debris density, mass, and ballistic number of the debris. For the investigation a nominal BX 250 foam density of 2.4 lb/ft³ was assumed. For the MLGD, the average impact velocity was determined to be approximately 700 ft/sec. Velocities ranging from 775 to 820 ft/sec were calculated for a leading edge, lower panel impact. Since the reference projectile size of 1.67 lb was selected for LESS testing, a corresponding impact velocity of 775 ft/sec was used.

1.6.6 Knit line orientation

The properties of external tank foam vary with respect to its spray and rise directions [9]. Figure 1.6-4 describes the material axes nomenclature. Since the exact orientation of the foam at the time of impact was indeterminate, every effort was made to fabricate projectiles with knit lines that were oriented in a similar fashion as those on the
bi-pod ramp. Figure 1.6-5a and b show the –Y bi-pod ramp as applied on the ET and a fabricated test projectile respectively. Figure 1.6-5c shows a bi-pod ramp sectioned from the vehicle while the black lines in Figure 1.6-5d mark the location of the knit lines on the ramp. This knit line pattern was reproduced in the test projectiles by spraying using the same MAF process and then hand cutting the specimens to yield a knit line pattern for the proper bi-pod configuration as shown in Figure 1.6-5e. Further details of the projectile fabrication process are documented in section 3.3.

Figure 1.6-4: Projectile cut direction nomenclature [9].
Figure 1.6-5: Photographs and sketches demonstrating the similarity in knit line construction of the -Y bi-pod ramp and the test projectiles. (a) the -Y bi-pod ramp installed on the ET. (b) a typical test projectile marked for impact testing. (c) a bi-pod ramp removed from the ET and (d) the same ramp with knit lines marked with black lines for clarity. (e) a sketch of a typical test projectile showing the knit line orientation from both side and rear views.
2.0 Test Facility Design

2.1 Worksite Description

All of the tests described in this report were conducted at the Southwest Research Institute (SwRI) in San Antonio, Texas. Figure 2.1-1 shows a map of the San Antonio area that includes SwRI. Personnel from the SwRI Mechanical and Materials Engineering Division, Engineering Dynamics Department, were contacted immediately following the STS-107 accident and asked to perform the necessary impact testing. The department’s expertise and unique facilities and launchers were ideally suited for the scale of testing that was envisioned.

The area designated for Orbiter impact testing was site 4 of the ballistics range (see Figure 2.1-2). Figure 2.1-2 shows a map of the range area. A metal building near site 1 was designated for storage of test articles and support equipment. This building provided cover for the test article crates, but was not climate controlled. A covered work area close to test site 4 was used to prepare the test articles. Although not climate controlled, it provided cover from the elements. It had adequate power to operate any equipment that was needed. There is also a small machine shop located in this area. Adjacent to the garage is a climate-controlled bunker with telephone. A high-bay area and office space were designated in building 82 to facilitate pretest preparation and posttest analysis of the test articles.
Figure 2.1-1: Map to SwRI from the San Antonio International Airport.

Figure 2.1-2: SwRI Test Facility and work areas.
2.2 Test Facility and Equipment Description

2.2.1 Test Site 4

The dimensions of test site 4 are given in Figure 2.2-1. The SwRI large compressed-gas gun (LCGG) was utilized for all Orbiter thermal protection system (TPS) impact testing described in this document. Figure 2.2-2 shows a typical test setup of the test site and its supporting equipment. Figure 2.2-3 shows the LCGG raised to its highest vertical position. The LCGG was fitted with a 35-ft-long steel barrel of rectangular cross section. Barrels of various rectangular cross sections were constructed of structural steel tubing to allow for testing of the foam projectiles noted in the test matrices in Sections 4.1.1 and 4.2.1. Inside the barrels, were weld beads about 0.25 in. wide and high that ran the length of the barrel. During the test, the weld beads would scrape a small amount of material off the projectile (less than 1 gram).

The gun operator first loads the projectile into the barrel. A fast-acting valve allows the accumulator tank to be pressurized by temporarily sealing the tank from the barrel. When the operator is ready to fire the gun, the entire area is cleared of personnel and the accumulator is pressurized with nitrogen gas. Approximately 30 psi is required to launch the external tank (ET) foam projectiles to a velocity of 700 ft/sec. Once the pressure in the accumulator has stabilized, the operator counts down from 5 and depresses a fire button that opens a fast-acting valve, allowing the pressurized nitrogen to expand into the barrel and accelerate the projectile downrange.

A secure video teleconferencing (VTC) capability was developed to enable engineering consultation during the practice testing and to allow selected observers to
witness the entire test program. This capability allowed the team to ensure test quality and safety by keeping the number of on-site visitors to a minimum. The NASA test director maintained a list of all personnel necessary to conduct the test series, and the Director of JSC Engineering maintained the list of test site observers, which required concurrence by the NASA Accident Investigation Team (NAIT) and the Columbia Accident Investigation Board (CAIB). The test preparations and the tests themselves were broadcast live via VTC to JSC and other approved NASA centers.

Figure 2.2-1: Test site 4.
Figure 2.2-2: Test pad with typical support equipment.

Figure 2.2-3: The large gas gun raised to its highest vertical position.
2.2.2 Data Acquisition Facilities

To support all of the required data collection, Boeing furnished a mobile trailer (Figure 2.2-4) equipped with data acquisition and instrumentation hardware. The operators worked from this area, approximately 150 ft north of the test pad. Remote signal conditioners and patch panels were located in a bunker just north of the test cell. Cables 50- to 75-ft-long carried the signals from the sensors mounted on the targets to the bunker. Figure 2.2-5 gives a block diagram of the data acquisition system. The system architecture was adjusted several times throughout the test program to eliminate or minimize issues encountered from previous test runs. Further detail of the sensor selection and data acquisition and reporting are documented in Section 2.3.1 and 2.3.2 respectively. A detailed description of the sensor and data acquisition system configuration for each test is documented in Sections 4.1.3.1 and 4.2.1.1.

Figure 2.2-4: The Boeing instrumentation trailer.
Figure 2.2-5. Block diagram of the data acquisition system [12].

2.2.3 Test Stand and Tilt Table

In order to accomplish realistic boundary conditions, the test articles needed to be fixed to a rigid structure (Figure 2.2-6). Furthermore, the impact angle between the shot line and the test article surface needed to be adjustable. To accomplish this, a rigid stand constructed with steel I-beam base and a hinged tilt table top was mounted to a 16,000-lb-capacity scissor lift. The test articles were attached to the tilt table while the entire
assembly was at ground level. Next, the scissor lift was used to raise the assembly to the necessary targeting height. I-beam outriggers were attached to support the assembly at height, so that the load could be relieved from the lift. The tilt table was lifted at one end, pivoting at the hinge line, to achieve the proper impact angle. I-beams were then attached to the tilt table and base to fix the impact angle. The entire assembly was proof-loaded for a maximum operating load of 8000 lb by a local San Antonio company recommended by JSC rigging. This test stand was used for both the main landing gear door (MLGD) and leading edge structural subsystem (LESS) test articles. The design allowed the test team to set the impact angle to an accuracy of ± 0.1 degrees.

Figure 2.2-6: Tilt table stand and scissor lift
2.2.4 Ejecta Catcher

During the impact event, material (target and projectile) could be fragmented and forced downrange. Because it was desirable to collect the materials for later analysis, an ejecta catcher was constructed of plastic mesh netting attached to four 35-ft poles. The bottom of each net was attached to the test stand. A large pillow (several layers of netting attached to a suspended metal beam) was used to capture the largest projectile fragments. All debris was collected, weighed, and stored for potential future analysis. In addition to catching the debris, the catcher also provided a consistent background for high-speed camera imaging of the fragment field. The background was important for successful measurement of the debris size and velocity through photogrammetry analysis.

2.2.5 Lifting Operations

The test program required the lifting of heavy fixtures and test articles, some more than 12,000 lb. To accommodate these lifts, a 40-ton mobile overhead crane was rented and located near test site 4 (Figure 2.2-7). For smaller lifts such as moving test articles to and from crates, or to and from the flatbed truck, a 5-ton forklift was used. Both the crane and forklift had current load and operator certifications. Details regarding these lifts are documented in a critical lift plan, which was approved by JSC Occupational Safety [13].
2.2.6 Trigger System

The instrumentation and camera systems that collected data during the test required a signal to indicate when the gun was fired (time equal zero). Figure 2.2-8 shows a schematic of the trigger system. Make wires located in the gun bore are used to provide this pulse. Two small holes were drilled in the barrel wall near the muzzle. Wires were placed in each hole carefully to avoid touching the sides. When the gun is fired, the projectile presses the wires against the barrel wall and shorts a circuit. The short dumps a capacitor that triggers a Nicolet data acquisition system. The Nicolet then outputs a transistor-transistor logic (TTL) pulse to the instrumentation trailer and one of the SwRI cameras. The other SwRI cameras are triggered based on the short generated by another make wire. During some of the tests, the high-speed cameras were synchronized with the data acquisition system by using an inter-range instrumentation group (IRIG) time-generator pulse.
2.2.7 High-Speed Video Camera Setup

During impact testing it is important to accurately measure a number of properties associated with projectile launch and target impact. In particular, the impact velocity is a critical variable and must be measured as accurately as possible. There are, however, several other aspects of the projectile that must be observed: projectile pitch, yaw, potential deformation, and impact point on the target. Of course, imaging the target response yields target deformation data that is essential to improving existing models and developing new ones. Because the projectile is moving so quickly (approximately 700 ft/sec or 8400 in/sec), it is impossible to view any of these events with ordinary video cameras (30 frames/sec). The test team instead relied on a suite of high-speed video cameras that were capable of capturing images at rates up to 10,000 frames/sec. Software
allowed for the measurement of target and projectile data directly from the digital camera images.

2.2.7.1 High-Speed Video Camera Details

Seven cameras were used to record early tests from the test article exterior. By the end of the test program, the number of cameras had grown to 16, recording from both the interior and the exterior of the test article. Four different types of cameras were used—three different models of Phantom cameras from Vision Research, and Kodak MotionCorder cameras. Their relevant properties are summarized in Table 2.2-1.

**Table 2.2-1 High-Speed Video Camera Inventory**

<table>
<thead>
<tr>
<th>Camera Type</th>
<th>Maximum Image Size (WxH)</th>
<th>Maximum Frame Rate for Maximum Image Size (frames per second)</th>
<th>IRIG-Compatible</th>
<th>Maximum Frame Rate for 256x256 Image</th>
</tr>
</thead>
<tbody>
<tr>
<td>Kodak MotionCorder</td>
<td>512x480</td>
<td>250</td>
<td>no</td>
<td>1000</td>
</tr>
<tr>
<td>Phantom 4</td>
<td>512x512</td>
<td>1000</td>
<td>yes</td>
<td>3731</td>
</tr>
<tr>
<td>Phantom 5</td>
<td>1024x1024</td>
<td>1000</td>
<td>yes</td>
<td>11900</td>
</tr>
<tr>
<td>Phantom 7</td>
<td>800x600</td>
<td>4800</td>
<td>yes</td>
<td>27000</td>
</tr>
</tbody>
</table>
Each camera is capable of recording images smaller than its maximum image size at increased frame rates. A camera is also capable of recording at less than the maximum frame rate for the selected image format. The recording duration depends on the total memory available in the camera, as well as the image size and frame rate.

Cameras record continuously until they receive a trigger signal. Each camera will stop recording at a programmed interval (called the post-trigger) after receipt of the trigger signal. This usually results in a movie that begins before the trigger signal is generated and continues until the camera’s memory is full. Details of the trigger signal are documented in section 2.2.6.

Some cameras had the capability to record a time stamp in each frame. The camera clocks were synchronized by an IRIG timing signal generated from the instrumentation trailer. Due to changing personnel and camera inventories, not all cameras were equipped with the necessary hardware to accept the IRIG signals. Thus, some cameras had incorrect time stamps. The camera vendor, Vision Research, has developed software that will enable recoding of the time stamps in the movies.

2.2.7.2 High Speed Video Camera Layout

Given the differences in the geometry of each test series, the high-speed camera layout was modified to ensure the measurement of necessary data. External cameras were mounted on 7-ft tripods at locations ranging from ground level to platforms elevated as high as 10 ft. In one notable case, a tripod stood on the rear deck of a crane. Cameras were elevated to positions as high as necessary to achieve clear views of the impact surfaces.
For the LESS tests, internal cameras were mounted on three-axis tripod heads attached to shafts. The shafts were mounted to blocks or plates of aluminum that were in turn bonded to the LESS wing spar. A shaft consisted of a threaded rod passing lengthwise through an aluminum pipe. The rod was bolted to the mounting block (or plate) and the tripod head was attached to the rod. As the tripod head was threaded onto the rod it would bear down on the pipe, placing the rod in tension and the pipe in compression, providing a much sturdier support than the rod alone would have. Mounting blocks were aluminum blocks approximately 1.5 in. wide, 3 in. long, and 1 in. thick, with a threaded hole to receive the threaded rod. When use of the mounting blocks became too limiting, with insufficient leeway in adjusting the positions of the cameras, aluminum plates were mounted to the blocks. The camera mounts were bolted to the plates, which were 8 in. square and .25 in. thick, and could have numerous holes drilled in them to provide options in mounting the cameras. Figure 2.2-9 is a photo of a typical interior camera mounting.

![Figure 2.2-9. Typical interior camera mounting for an LESS test.](image)
Test-specific camera layouts for the MLGD and the LESS test articles are described in Sections 4.1.3.2 and 4.2.1.1.2. Typical camera layouts for the tests were as follows. To support the MLGD tests, there were typically six high-speed video cameras located about the Test site 4 area. Figure 2.2-10 is a schematic of the high-speed camera layout and Figure 2.2-11 shows an example of each field of view (FOV). Camera L1 offered a close-up view of the impact point. The FOV was sized to image a tile area including the impact site and 4 feet downrange. Previous tests had indicated this FOV was sufficient to capture the entire projectile/target event. Cameras located at positions L2, L3, and L6 yielded data that allowed the measurement of the projectile velocity and impact orientation in relation to the shot line (pitch and yaw). To measure the size and velocity of material ejected from the impact site, cameras L4 and L5 were installed approximately 90 deg from one another.

Figure 2.2-12 shows a typical camera layout as modified to support the fiberglass and reinforced carbon-carbon (RCC) tests. These changes to the outer test article cameras were needed to collect the same data for this different target geometry. In this example, a total of 13 cameras were used for the test. Six cameras were added internally to the test article to observe any deflections of the leading edge structure or RCC panels.
Figure 2.2-10: Schematic of the high-speed camera setup for a typical main landing gear door test.

L1-L7: Camera locations 1-7
C1: Kodak Motioncorder 2000 frame/sec
C2: Vision Research Phantom 5.0 (2K frame/sec)
C3: Vision Research Phantom 7.0 (7300 frame/sec)
C4: Documentary video (30 frames/sec)
Supplemental lighting added as needed to achieve framing rates

Figure 2.2-11. Example high-speed camera images from each location.
C5: located ~200 ft away

(a) target exterior camera layout

(b) target interior camera layout

L1-L13: Camera locations 1-13
C1: Kodak Motioncorder 1000 frame/sec
C2: Vision Research Phantom 4.0 (1900 frame/sec)
C3: Vision Research Phantom 5.0 (2K frame/sec)
C4: Vision Research Phantom 7.0 (8000 frame/sec)
C5: Documentary video (30 frames/sec)
Supplemental lighting added as needed to achieve framing rates

Figure 2.2-12: Schematic of the high-speed camera setup for a typical fiberglass or RCC test
2.2.7.3 Lighting

Illumination of the exterior of the test articles was primarily from sunlight. During facility calibration testing, however, it was determined that sunlight only was inadequate for the highest speed imaging. Furthermore, the sunlight cast shadows on the test article, obscuring data collection. Theater lighting was used to supplement the fields of views as detailed in Table 2.2-2. The larger lights were positioned for general illumination, and smaller lights were positioned to eliminate shadows.

<table>
<thead>
<tr>
<th>Light Type</th>
<th>Power</th>
<th>Quantity</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cinemills</td>
<td>4 kW</td>
<td>2</td>
</tr>
<tr>
<td>Ianiro Fresnels</td>
<td>4 kW</td>
<td>2</td>
</tr>
<tr>
<td>Cinemills</td>
<td>2.5 kW</td>
<td>2</td>
</tr>
<tr>
<td>Cinemills</td>
<td>1.2 kW</td>
<td>2</td>
</tr>
</tbody>
</table>

2.2.8 Photogrammetry Design

A control point grid was constructed on the bunker behind the test article. Additional reference points were mounted on and around the test article, positioned to allow photogrammetric analysis by the NASA Image Science and Analysis Group (ISAG) at JSC. The positions and orientations of the video cameras could be determined by these reference points; in turn allowing accurate measurement of the position and velocity of the foam projectile and debris. Ultimately, a grid of reference points was
established on the interior of each of the wing leading edge components to enable high
precision photogrammetric analysis of component deformation during the test.

The control point grid was built by dividing the wall into two-foot squares using a
hydraulic level, chalk line, and a plumb line. Four-inch photogrammetric targets were
glued to the wall at the intersections of the horizontal and vertical chalk lines. The
relative positions of the targets were determined by solving a trilateration network
consisting of the distances from each target to its eight nearest neighbors and sixteen
next-nearest neighbors.
2.3 Instrumentation Design

The instrumentation for the thermal protection system (TPS) impact test series was selected to measure the forces and deflections induced in the TPS material and structure when impacted by a foam projectile. Various experts helped define the requirements for strain gages, accelerometers, load cells, and deflection sensors [12]. They ensured that all possible mechanical properties were measured and that sensors with the right range and response characteristics were selected. The instrumentation experts worked closely with other engineering disciplines to enable dynamicists to verify their structural dynamics (such as DYNA) explicit non-linear, and hybrid codes. The exact types, locations and numbers of sensors were tailored and approved by the NAIT and CAIB before each test. Sensor orientations and locations with dimensions are documented in Section 4 of this document. A block diagram of the instrumentation system is located in Section 2.2.2.

2.3.1 Sensor Selection

The impact test instrumentation included several different types of sensors. Accelerations were measured with piezoresistive (DC-type) accelerometers. The strain levels were measured with foil-type strain gages. Forces (or external loads) were measured using load cells. Deflections were measured by deflection sensors called linear variable differential transformers (LVDTs). Gaps were measured using capacitance probes for the first shot and high-speed video equipment for subsequent shots.
2.3.1.1 Accelerometer Selection

Accelerometers were installed to measure accelerations of surfaces and components of the test article. The measurements were used to determine the arrival of the projectile and the quality of the test. Initial accelerometer selection was based on program direction and transducer availability. General purpose, ICP-type accelerometers were selected for the initial "practice plate" testing, supplemented by some large fluid-damped piezoresistive accelerometers capable of frequency response down to 0 Hz, DC. As testing progressed, more of the smaller, higher G level Endevco model 7264 accelerometers became available from the manufacturer. Confidence in the DC accelerometers increased, and the ICP-type and fluid-damped accelerometers were replaced before the RCC-6 fiberglass panel test.

Table 2.3-1 summarizes the general characteristics of the accelerometers used in the test program. For detailed specification sheets refer to the Endevco website [14].

<table>
<thead>
<tr>
<th>Model</th>
<th>Type</th>
<th>Sensitivity mV/G</th>
<th>Frequency Response (Hz)</th>
<th>Maximum Level(G)</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Endevco 2256A-2.5</td>
<td>Piezoelectric</td>
<td>2.5</td>
<td>5-2000</td>
<td>2000</td>
<td>General purpose ICP type</td>
</tr>
<tr>
<td>Endevco 2262A-1000</td>
<td>Piezoresistive</td>
<td>0.5</td>
<td>0-1500</td>
<td>1000</td>
<td>Fluid Damped, full bridge</td>
</tr>
<tr>
<td>Endevco 2262CA-1000</td>
<td>Piezoresistive</td>
<td>0.5</td>
<td>0-1500</td>
<td>1000</td>
<td>Fluid Damped, full bridge</td>
</tr>
<tr>
<td>Endevco 7264-2000</td>
<td>Piezoresistive</td>
<td>0.25</td>
<td>0-4000</td>
<td>2000</td>
<td>Undamped, half bridge</td>
</tr>
<tr>
<td>Endevco 7264-2000TZ</td>
<td>Piezoresistive</td>
<td>0.25</td>
<td>0-4000</td>
<td>2000</td>
<td>Undamped, half bridge</td>
</tr>
<tr>
<td>Endevco 7264B-500T</td>
<td>Piezoresistive</td>
<td>0.8</td>
<td>0-5000</td>
<td>500</td>
<td>Undamped, half bridge</td>
</tr>
</tbody>
</table>

2.3.1.2 Strain Gage Selection

Strain gages were installed to measure the flexure and stress level in sections and components of the test articles. The general-purpose CEA-125UR, CEA-250UR and
FSE-25S strain gages were selected for the test series. This type of strain gage was easy to install and worked well in the ambient temperature environment of San Antonio. The gages were bonded with AE10 adhesive to the aluminum of the main landing gear doors and wing acreage panels. EPY150 adhesive was used to bond the gages to the fiberglass and RCC leading edge panels. All gage installations were covered with M-Coat D coating to protect them from humidity and other environmental hazards.

Two different size rosettes were installed, according to the room available at the desired installation location. Large grid (0.250 in²) gages were the preferred size on the leading edge panels and were installed on the large face areas and base flange areas of the fiberglass and real RCC Panels. Smaller (0.125 in²) grid gages were installed in the rib and T-seal areas as well as on the main landing gear doors and wing acreage panels. The small axial gages, FSE-25S models from manufacturer BLH, Inc., were installed near the edge of the rib in the apex areas on later tests.

The sensitivity of the strain gages was calculated based on the bridge excitation voltage, the manufacturer’s gage factor, and the resistances of the strain gage and lead wire. The shunt calibration value was also calculated and used to verify proper calibration of the instrumentation system.

Boeing technicians installed the strain gages on the test articles at KSC and SwRI. JSC technicians installed strain gages on the leading edge spar and helped Boeing install the strain gages on the real and fiberglass RCC panels at SwRI. More than 600 strain gage channels were installed for the test series.
Figure 2.3-1: RCC panel strain gage installation.

Figure 2.3-2: RCC rib strain gage installation.
2.3.1.3 Load Cell Selection

Load cells were installed to measure the force generated by the impact of the foam (Figure 2.3-4). All the load cells measured the axial load, and a few of the load cells were capable of measuring moments in the x and y directions. The wing acreage panel setup required multiaxis load cells. The main landing gear door did not have any load cells because of complex attachment requirements. The leading edge test setup used axial load cells in seven locations to restrain it rigidly. NASA supplied the load cells and their calibration data (Table 2.3-2). The Boeing team was responsible for recording the load cell data.

The load cells were the shear web design to provide the stiffest load link possible. Couplings and rod end bearings were attached to both ends of the load cell to release all moment and shear loads from misalignment. Pins and clevises were sized to minimize slop in the joints.

<table>
<thead>
<tr>
<th>Load Cell Number</th>
<th>NASA JSC ID</th>
<th>Calibration Due Date</th>
</tr>
</thead>
<tbody>
<tr>
<td>LC1</td>
<td>M44347</td>
<td>11/25/2003</td>
</tr>
<tr>
<td>LC2</td>
<td>M44345</td>
<td>11/25/2003</td>
</tr>
<tr>
<td>LC3</td>
<td>M170180</td>
<td>3/16/2008</td>
</tr>
<tr>
<td>LC4</td>
<td>M170233</td>
<td>3/16/2004</td>
</tr>
<tr>
<td>LC5</td>
<td>M170232</td>
<td>11/25/2003</td>
</tr>
<tr>
<td>LC6</td>
<td>M44341</td>
<td>11/25/2003</td>
</tr>
<tr>
<td>LC7</td>
<td>M44353</td>
<td>4/22/2004</td>
</tr>
</tbody>
</table>

Table 2.3-2
Load cell identification and calibration period
Figure 2.3-3: Load cell locations.

Figure 2.3-4: Load cell installation.
2.3.1.4 Deflection Sensor Selection

The deflection sensors were selected to monitor the motion of the test articles. The main landing gear door had sensors mounted at three of the four corners and down the center of the inboard surface. The corner sensors measured the translational and yawing motions. The sensors down the center measured the vertical and pitching motions. The leading edge test article had sensors mounted near the stagnation point. These sensors measured both lateral and vertical motions. Figure 2.3-5 is a top view of the main landing gear door with the four LVDT sensors installed at the corners.

![Main landing gear door with LVDT sensors installed](image)

Figure 2.3-5. Main landing gear door with LVDT sensors installed (the red circles indicate the location of the four sensors).

The sensors selected for the test were the guided ACT series LVDT sensors made by RDP Electronics. The sensors were selected for their mechanical design, accuracy,
and high frequency response. Rod end bearings were attached to both ends of the sensor to allow for out-of-line motion.

The attachment points or clevis plates were bonded directly to the test article with Lord 202 adhesive. The rod end of the sensor was pinned to these clevis plates. The sensor body end was pinned to a rigid Telespar structure. This structure was constructed using 2.25-in² and 2.50-in² Telespar tubes. Extra care was taken to ensure that the whole structure was isolated from the test article. The structure for the MLGD test stood on its own legs with many cross braces weighted down to make it solid. The structure for the leading edge tests was bolted to the base of the strongback with legs extending to the cement slab for additional bracing. Figure 2.3-6 shows the LVDT installation on RCC panel 6. Table 2.3-3 details the LVDT location dimensions as illustrated in the figure.

![Figure 2.3-6. RCC panel 6 with LVDT sensors installed. The letters A through D label critical dimensions of the panel and installed sensors as documented in Table 2.3-3.](image-url)
Table 2.3-3
LVDT Location Dimensions

<table>
<thead>
<tr>
<th>Dimension (Inches)*</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>A</td>
<td>B</td>
</tr>
<tr>
<td>----</td>
<td>----</td>
</tr>
<tr>
<td>RCC Panel 6 Fiberglass</td>
<td>5.6</td>
</tr>
<tr>
<td>RCC Panel 6 Real</td>
<td>5.6</td>
</tr>
<tr>
<td>RCC Panel 8 Fiberglass</td>
<td>6</td>
</tr>
<tr>
<td>RCC Panel 8 Real</td>
<td>6</td>
</tr>
</tbody>
</table>

*Dimensions A & B are measured from the inboard edge of the panel to the LVDT mounting point. Dimensions C & D are measured from the upper edge of each panel, along the surface to the LVDT mounting point.

2.3.1.5 Gap Sensor Selection

Capacitive probes were selected to measure the gap at two locations between the lower flange and the adjacent carrier panel. These sensors were 0.1 in thick and 1.0 in in diameter. They were placed very near the 0.4 in gap on a plate cantilevered from the edge of the lower flange of the leading edge panel. The target was mounted on a phenolic spacer block that was bonded to the box beam of each carrier panel to provide electrical isolation between the aluminum tape target and the carrier panel. The layers of aluminum tape were adjusted to provide a starting gap of 0.4 in. The sensor was connected to the electronics with a coaxial cable and the target was connected to chassis ground by a single wire. The sensors were calibrated with signal conditioner channels. Each channel used a specific set of coefficients to convert the capacitance to gap width.

During the first shot of the leading edge test series on fiberglass panel 6, the high-speed video captured extreme motion of the cantilevered plate. The motion was so harsh
that the plates snapped off the flange and flew through air. The video documented that the flange collided with the carrier panel structure. For all following tests, the capacitance probes were replaced by video instrumentation. Figure 2.3-7 shows a mock-up of the gap sensor installation for the fiberglass panel test.

![Target Plate](image)

**Figure 2.3-7: Gap sensor mock-up.**

2.3.2 Data acquisition and reporting

During each test, excitations of the sensors were recorded to a data recording system. The data recording system was based on the Spectral Dynamics Jaguar systems. Each Jaguar system consisted of a Sun SPARCstation computer, configured as a vibration control system (VCS) with software controlling an analysis and control
peripheral (ACP). An ACP is a modular, multichannel, digital signal processor capable of performing high-speed data acquisition, signal processing, and analysis. Each ACP can be configured for up to 98 channels using 16-bit analog-to-digital (A/D) converters with a maximum “per channel” sample rate of 51.2 kHz.

For this test series, the Jaguar systems were configured with three ACPs working together to acquire signals from up to 260 sensors at 25,600 samples per second, providing a maximum signal bandwidth of 10 kHz. Simultaneous data acquisition was assured by using a common trigger signal to start data acquisition on all ACPs. Additionally, an IRIG-B time code signal converted to “slow code” was recorded on each ACP to resolve any potential timing issues.

For the MLGD testing, the Jaguar was set up using a sample rate of 51,200 samples per second. For the RCC panel 6 and panel 8 testing (both fiberglass and real RCC), the Jaguar was setup with a 25,600 samples per second sample rate. In both cases a 10 kHz anti-alias filter was in effect, and a 640 msec block of data was captured with a 128 msec pretrigger.

Sony model SIR-1000 digital tape recorders were used to provide continuous data recording during testing, ensuring that data would be captured in case an anomaly occurred while preparing the gun for each test run, or in the event that a trigger signal was not initiated during the gun firing. The tape recorders were also set up to provide additional dynamic range above the Jaguar system to allow the possible recovery of signals that over range on the Jaguar. Up to eight SIR-1000 recorders were used in a master-slave configuration, which allowed for synchronous operation of all recorders. All recorders were configured for 32 channels using a SCX-32 expansion unit. Each recorder
was set up to record 32 channels of data at a sample rate of 48,000 samples/sec, providing a 20 kHz signal bandwidth.

The circuit described in Section 2.2.6 triggered the Jaguar, tape backup systems, and high-speed cameras. An IRIG-B time code generator was added to the system during the test series to further help correlate the video frames with the sensor data. The IRIG-B signal was fed to the cameras equipped with readers and output as "slow code" for the sensor data systems. The slow code generator converted the timing pulses into a stream the data systems could record on a typical data channel.

Conventional signal conditioning electronics were selected to provide proper conditioning and normalization of the sensor signals for the high-speed digitizers and the tape backup system. Generally, the electronics supplied excitation to the sensors and amplified the signals to the ± 10-V DC range. Table 2.3-4 documents the manufacturers and model numbers of the signal conditioners.

Data acquired from the Jaguar systems were converted to Matlab-compatible engineering unit data files. A Matlab script was written that read the acquired data and produced data plots in Adobe "PDF" file format. The full 640 msec block of data was formatted in Excel spreadsheets and plotted for the report. A second Matlab script zoomed in on the time interval from 135 to 155 msec starting with the data captured from the fiberglass RCC panel 6 test and produced plots for the report. A third script produced plots of strain gage data with the corresponding check channel plotted below it on the same page for all the RCC test cases. Other scripts were created to filter the data at 100 Hz, 500 Hz, and 2000 Hz cut-off frequencies for diagnostic purposes.
### Table 2.3-4
Signal conditioners summary

<table>
<thead>
<tr>
<th>Sensor type</th>
<th>Manufacturer/model of signal conditioner</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Accelerometers</td>
<td>PCB Piezotronics Model 584</td>
<td>Used for Endevco ICP type accels</td>
</tr>
<tr>
<td></td>
<td>Agilent Model E1529A</td>
<td>Used for DC type</td>
</tr>
<tr>
<td>Strain gages</td>
<td>Agilent Model E1529A</td>
<td>Used early in the test program, but demonstrated cross-talk problems</td>
</tr>
<tr>
<td></td>
<td>R. C. Electronics Model Dynamic 4000</td>
<td>±10V signal output to Jaguar and tape backup with 0.1 mV/µε accuracy</td>
</tr>
<tr>
<td></td>
<td>Vishay Model 2310</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Cybersystems Model 9320</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Trig-Tek</td>
<td></td>
</tr>
<tr>
<td>Load cells</td>
<td>Agilent Model E1529A</td>
<td></td>
</tr>
<tr>
<td>Deflection sensors</td>
<td>RDP Electrosense Model 600</td>
<td>500Hz response frequency, ±10V signal output for full stroke of LVDT (4 in)</td>
</tr>
<tr>
<td>Gap sensors</td>
<td>Capacitec Series 4000</td>
<td>Only used for first Fiberglass panel test. Video cameras used after that.</td>
</tr>
<tr>
<td></td>
<td></td>
<td>±10V signal output for 0.5 in gap measurement; 0.01” accuracy</td>
</tr>
</tbody>
</table>

Immediately following each test, the data were archived and transmitted several ways. Data were put on duplicate CDs for purposes of permanent record and transmittal to the NAIT and CAIB at the test site. In addition, all of the data products were uploaded to the NASA Orbiter Tile Impact Testing website and Boeing’s data server in Huntington Beach.
2.4 Nondestructive Evaluation of Test Articles

2.4.1 Background

Nondestructive Evaluation (NDE) tests were performed on the leading edge Reinforced Carbon-Carbon (RCC) and fiberglass components used in support of the impact testing. NDE support to the impact investigation team was a requirement to 1) baseline existing flaws/defects in the panels pre-test and 2) to evaluate the extent of damage incurred by the impacts post-test. The test materials consisted of fiberglass articles OV-101 Panel 6L and 8L and RCC leading edge components from OV-103 Panel 6L (Figure 2.4-1) and OV-104 Panel 8L. The NDE tests included examinations for both surface and internal damage.

![RCC Panel 6L and T-seal](image)

**Figure 2.4-1: RCC Panel 6L and T-seal**

Although there are mature NDE methods used for the inspection of manufactured materials at the Lockheed Martin facility, those techniques were not specifically
developed to detect impact damage. As a result, a diverse group of developmental NDE methods were used during the impact investigation to evaluate their effectiveness [15].

During leading edge impact testing at SwRI, NDE methods were required to assess the presence and extent of impact damage. Personnel from NASA, USA, Wyle Testing Laboratories, SAIC, SwRI and Boeing provided the resources for NDE methods that would be most applicable for damage assessment of both fiberglass and RCC panels.

The testing was performed by a team of NDE experts under the direction of the Boeing Company. The pre-impact test NDE provided a baseline material characterization that was followed by post-impact evaluations to assess the presence and severity of damage created during impact testing. The post-impact NDE concentrated on the impact site and directly adjacent regions using NDE techniques best suited for detection of flaws expected to occur during impacts: SiC coating loss, and substrate cracking and delamination. Highly stressed regions of the hardware including the lower surface, the slip and lock side ribs, and the attach points were inspected care.

The RCC panels and T-seal test specimens consisted of previously flown hardware. These articles had inherent porosity, varying SiC coating thickness and suspect voids that were likely created by oxidation. Due to the porous nature of RCC materials, ultrasonic signals are attenuated internally and x-rays are dispersed, thereby creating mottled images. This inherent material characteristic required that a baseline inspection be performed on the as-flown pre-test conditions prior to any impact testing. The post-test hardware condition includes as flown anomalies as well as any damage created by impacts. The NDE methods used were not well established for RCC
inspections and the flaw detection capabilities and probability of detection levels had not been established.

2.4.2 NDE Methods for Fiberglass and RCC Impact Testing

Visual inspection was the only NDE routinely performed on the RCC leading edge materials on the Orbiter vehicles during maintenance operations before STS-107. Therefore, additional NDE methods were required to detect and quantify subsurface defects that might occur during impact testing. The NDE methods were required to baseline the condition of the test materials pre-test and post-test to assess impact damage such as delaminations and cracking. Table 2.4-1 lists each of the types of NDE utilized during the test program and indicates their capability to detect impact induced damage. Additional detail on each of these techniques is documented elsewhere [16].

2.4.3 NDE Reference Standards

NDE reference standards are used to calibrate and quantify the capabilities of the NDE test equipment. Reference standards were created from scrap RCC panels. Artificial flaws were created in the specimens by drilling flat bottom holes with diameters of 0.125”, 0.25” and 0.50” to depths of 25%, 50% and 75% of the material thickness (Figure 2.4-2). These samples served as the reference standards for both ultrasonic and thermography NDE.

Typically, reference standards would be fabricated with embedded artificial flaws; however, the flat bottom hole approach was selected in the absence of proper reference standards. In the absence of reference standards, best effort approaches where taken for
the other NDE methods. Standardized testing methods were employed to the greatest extent possible for these NDE techniques.

<table>
<thead>
<tr>
<th>NDE Method</th>
<th>Surface defects</th>
<th>Subsurface defects</th>
<th>Delamination</th>
<th>Porosity/voids</th>
<th>Cracks</th>
<th>Coating thickness</th>
<th>Strain field</th>
</tr>
</thead>
<tbody>
<tr>
<td>Visual inspection</td>
<td>X</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Coin-tap</td>
<td></td>
<td>X (gross)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Thermography</td>
<td></td>
<td>X</td>
<td>X</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultrasonic</td>
<td></td>
<td>X</td>
<td>X</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultrasonic (wet)</td>
<td>X</td>
<td></td>
<td>X (some)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Film radiography</td>
<td></td>
<td></td>
<td>X</td>
<td>X</td>
<td></td>
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</tr>
<tr>
<td>Real-time radiography</td>
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<td></td>
<td></td>
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<tr>
<td>Computed tomography</td>
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<td>Reverse geometry x-ray</td>
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<td></td>
<td>X</td>
<td>X</td>
<td></td>
<td></td>
<td></td>
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<tr>
<td>Eddy current</td>
<td></td>
<td>X</td>
<td>X</td>
<td></td>
<td></td>
<td></td>
<td></td>
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<tr>
<td>Laser shearography</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>X</td>
</tr>
</tbody>
</table>

Table 2.4-1
Applicability of NDE Techniques to detect impact damage on Fiberglass and RCC panels

Figure 2.4-2: Photograph of flat bottomed hole RCC reference standard.
2.5 Test Operations

2.5.1 Procedures

Personnel followed SwRI standard operating procedures for firing the gun and collecting high-speed camera images [17]. An addendum to this procedure, dated March 28, 2003, was made for requirements specific to this test. Instrumentation data was collected using a Boeing Huntington Beach-developed standard operation procedure and calibration. The range manager completed a data sheet during the performance of the test.

On the day of a test, the NASA test director led the test sequence by executing a checklist. The entire test team, in communication with the test director via hand-held radios, followed the checklist. Personnel were allowed to stop the execution of the test at any time before firing the gun.

A detailed job hazard analysis was completed for all tasks related to this test plan. An environmental health plan was also developed.

2.5.2 Test Constraints

Certain weather conditions were required to ensure a good test environment. The test director was responsible for determining that these constraints had been met. No precipitation was acceptable during the test. The skies had to be sunny to partly cloudy, so that lighting conditions are sufficient for the high-speed cameras. Temperatures had to be less than 110°F to ensure that the cameras did not overheat. Wind velocities less than 25 mph were required to limit the applied load on the ejecta
catcher support poles. The test director received weather forecasts from the JSC Meteorology Group.

To account for the fact that data collection devices may fail during the conduct of any test, a set of acceptable data loss was defined. For the purposes of these tests, any individual strain gage loss was acceptable; however, no other sensors may be lost. That is, 10 percent of the accelerometers and LVDTs had to be operating. All six outer high-speed cameras were required. Loss of any of the six inner cameras was acceptable. If these requirements were not met, the test will be put on hold. Adequate spares were available for all sensors (50 percent of quantity used as spares), and one spare Phantom 5 camera was available in the event of camera failure. Orbiter Project Office approval was necessary for any deviation from these requirements. All tests had 100 percent sensor coverage.

2.5.3 Test Success Criteria

An impact test was considered successful if all the following criteria were met:

the impact velocity was within ± 50 ft/sec of the requested velocity,
the projectile struck the impact point intact within a 1-in. radius of the target location,
the high-speed video cameras were triggered and captured images of the impact event and all instrumentation was triggered, and data from the impact event was recorded.

If all of the test criteria had not been met, the results would have been in question, and the Impact Test Team lead could have required a retest. After consulting with the Orbiter Project Office, the team lead could have directed that the article be retested, or
have a new test article installed and tested using the same criteria. However, all tests conducted during the test program were successful, and none required retest.

2.5.4 Gun Alignment

Alignment of the test article with respect to the gun shot line was critical. The test article surface had to be perpendicular to the shot line at the proper vertical and horizontal locations and at the accurate angle of obliquity. An in-bore laser alignment tool was used to sight the target point (Figure 2.5-1). The center of the projectile cross section was used to define the shot line. Mirror and gun sight alignment tools were used to ensure that the target plane was perpendicular to the shot line (± 0.1 deg). The test article was installed on a tilt table top that pivots at the front edge of the target to achieve the proper impact angle (see Section 2.2.3). A calibrated digital protractor was used to verify the impact angle (± 0.1 deg). A large vehicle lift integrated into the gun support frame made it easy to raise or lower the entire gun assembly to sight the target vertically. The gun was likewise easily translated in relation to the target by actuating compressed air bearings at the base of the gun. Two people at each end of the gun could then push it into position with little effort. When the gun was fired, the projectile struck its designated impact location within a horizontal and vertical accuracy of ±1 in.
Figure 2.5-1. The in-bore laser alignment tool.

Given the curved surfaces of the LESS test article, a second spatial reference point was required to define the velocity vector. A model of the foam impactor and its velocity vector were plotted using a computer-aided design (CAD) application (IDEAS) and a detailed model of the LESS test article. The block was aligned in CAD so that the appropriate points on the block and the RCC or fiberglass panel were in initial contact at a place called the initial impact point. The aim point is where the centerline velocity vector intercepts the panel when the block is aligned. For gun alignment purposes, a second aim point is required. A tool was created from extruded aluminum to attach rigidly to the glove spar. At the top of the tool, a plate was attached with a through hole. This plate could be raised and lowered to a range of heights. The second aim point was
modeled using the CAD application, and the appropriate height for the plate was determined for each test. Section 4.2.1 describes the targeting data required for each LESS test.

To sight the target, the alignment tool was attached to the glove spar at the appropriate height, and the aim and target points were marked with indelible marker on the target panel. The center bore laser was installed in the gun and turned on. Next, the gun was adjusted (raised, lowered, or translated) to ensure that the laser light passed through the hole in the alignment tool. Finally, the gun was yawed with respect to the target until the beam struck the aim point. This targeting process resulted in impact results, which were consistently within the 1-in. tolerance.

2.5.5 Test Reporting

The data distribution lead was responsible for collecting all still images, high-speed video, and instrumentation files, and placing them on the impact test website. The link for the website is http://hitf.jsc.nasa.gov/hitfpub/archive. During the test program, all data files related to the program were archived here, and they will remain available until the data are archived by the Space Shuttle Program.
2.6 Facility Calibration Tests

Before any Orbiter TPS impact testing, the entire testing team participated in a total of twelve tests together. Various targets were impacted, including a ¼-in. aluminum plate, a ½-in. instrumented aluminum plate, and a plywood/fiberglass RCC mockup. To facilitate the design of the test setup, a total of 22 tests were conducted on these targets. In addition, five tests were conducted on the left main landing gear door of OV-101, and seven tests were conducted on the LESS. The achieved velocity was within 20 ft/sec of that desired for these tests.
3.0 Test Article and Projectile Fabrication

3.1 Main Landing Gear Door (MLGD) Test Article

MLGD testing was initiated to evaluate the influence of the STS-107 impact if it had hit the MLDG. Two potential issues exist with such an impact. First, an impact to the center of the door could damage the LI-900/RCG tile, allowing the door structure to be exposed to high heating; second, an impact to the gear door perimeter tile could cause damage resulting in a flow path past the MLDG thermal barrier. Each case could result in hot plasma flow penetrating the MLDG cavity.

During the progression of tests for the STS-107 investigation, the MLDG was eliminated from the failure scenario. As a result, only the acreage (center of the door) tiles were tested during the investigation. The susceptibility of the perimeter tiles and thermal barriers was not explored, but such a study is planned as a follow-on effort for Orbiter return to flight (RTF).

3.1.1 Thermal Protection System (TPS)

The TPS of the Space Shuttle Orbiter is unique among atmospheric reentry vehicles in that it, along with other Orbiter subsystems, is reusable [18]. During a typical reentry heating cycle, the Orbiter is subjected to temperatures in excess of 2300°F as shown in Figure 3.1-1. The mostly ceramic-based TPS protects the Orbiter aluminum and payload bay door graphite epoxy structure from reaching temperatures over 350°F, and the orbital maneuvering system (OMS) pod graphite epoxy structure from exceeding 250°F. The Shuttle TPS team is responsible for integration of all of the materials, development, design concepts, fabrication techniques, installation processes, and
refurbishment procedures necessary to protect a vehicle from the severe heating environment of atmospheric reentry.

In addition to protecting the structure from heat loads up to 66,000 Btu/ft², the outer mold line (OML) of the TPS serves as the aerodynamic shape of the vehicle. This shape is maintained by tight control of the step and gap between installed TPS components. Excessive steps and/or gaps between parts can result in early transition of the boundary layer from laminar to turbulent, which would induce higher heat loads. Minor steps and/or gaps can result in local overheating, which could slump (i.e., melt and deform) tiles or permit subsurface plasma flow that, in turn, could degrade the TPS bond line or underlying structure.

More than 25,500 reusable surface insulation (RSI) tiles were installed on OV-102. The tiles are made from one of four substrate materials (LI-900, FRCI-12, LI-2200 and AETB-8). They are coated with a white or black glass coating, except for AETB-8, which is impregnated with a toughened unified fibrous insulation (TUF1) coating, followed by the standard black coating. The use of the TUF1 is qualified for use only in the high impact areas such as the base heat shield, due to the thermal conductivity limitations of the AETB-8 substrate. White-coated RSI tiles are referred to as low-temperature reusable surface insulation (LRSI) and black-coated tiles are known as high-temperature reusable surface insulation (HRSI). LRSI is used in areas where the peak temperatures do not exceed 1200°F, and HRSI is used in regions less than 2300°F.
Figure 3.1-1: Surface temperatures during reentry [18].

After the tile substrate has been machined and coated, the inner mold line (IML) of the tiles is densified by applying a material made of colloidal silica and silica powder with an emittance agent to the tile substrate. This process allows the stress loads at the tile interface to be uniformly distributed, thus achieving a tile system with the ultimate
strength of the tile substrate. The interface between the tiles and the structure has to accommodate strain induced from flight loads and the coefficient of thermal expansion (CTE) differences between the tiles and structure experienced during the mission. To achieve this, a strain isolator pad (SIP) fabricated with felted Nomex fibers is bonded between the tile and structure using RTV adhesive. The SIP is a Nomex felt pad, available in 0.090-in., 0.115-in. and 0.160-in. thickness. The 0.160-in. SIP is used in the acreage areas where LI-900 tiles are typically used, whereas thinner SIP is used in higher load areas, including the MLGD perimeter tiles. An additional felted Nomex material called filler bar is bonded to the structure in a lattice pattern, providing a compression seal around IML perimeter of the tile where the SIP was not bonded. This protects the structure between the tile gaps by creating a physical barrier to the structure and a pressure barrier to the tile IML. If heating is detected in a gap in a critical location, a gap filler may be bonded between tiles to further prevent heat intrusion. Figure 3.1-2 depicts a typical tile installation.

![Diagram of a typical tile installation](image)

Figure 3.1-2: Typical tile installation [18].
3.1.2 Main Landing Gear Door Test Article

3.1.2.1 Structure

The MLGD is a complex aluminum skin and stringer build-up with features to accommodate stowing of the landing gear during flight. Early analytical predictions expected major damage to the tile system. This damage, if sufficiently large and located on the door or thermal barrier area, could yield a catastrophic vehicle failure either due to local structural damage or at the door attach points. As a result the fidelity of the test article structure became important for accurate prediction of tile damage response. To achieve the most representative structure and to support the aggressive investigation schedule, the doors from OV-101 (*Enterprise*) were selected for use as the test structure. *Enterprise* (Figure 3.1-3) was ferried to Dulles Airport, Washington, D.C., and became the property of the Smithsonian Institution in 1985. Arrangements were made with the Smithsonian to have a United Space Alliance (USA) team travel to Washington, D.C., remove the MLGDs, and transport them to KSC to have tiles installed for testing. Figure 3.1.4 shows the USA ground operations crew removing the port MLGD from OV-102.

![Space Shuttle Orbiter Enterprise](image_url)

*Figure 3.1-3: Space Shuttle Orbiter Enterprise*
Figure 3.1-4: USA ground operations crew in the process of removing the left (port) MLGD.

The doors have discrete attach points to the surrounding structure. For proper structure and TPS response, it was important that a holding fixture be designed to represent the vehicle interface as closely as possible. The frame geometry was designed to support the door at its three-hinge line, and at four latch and four up-stop points. The door hinges were attached to the test fixture clevises using flight hardware. A threaded rod and clamp arrangement attached the four door latch rollers to the support fixture. The four up-stops of the test fixture consisted of a threaded rod to which a swivel head was welded. These stops impinge upon the design striker plates on the MLGD. The threaded rod concept allowed for locating the MLGD to its vehicle-installed orientation (approximately 2 degrees fore to aft and 5 degrees inboard to outboard), while maintaining the rigidity of the total MLGD/support fixture assembly. The door was attached to its support fixture at KSC in flight orientation and shipped as a unit. A lift
sling (proof-loaded at KSC to a 4000-lb working load) was constructed to allow for easy movement of the target system at SwRI. The support fixture, shown in Figure 3.1-5, is approximately 22 in. tall, 156 in. long, and 95 in. wide.

![Figure 3.1-5. An isometric drawing of the MLGD strongback.](image)

3.1.2.2 TPS Installation

The surface of the Enterprise MLGDs had a simulated TPS that was installed to provide a vehicle mold line for its flight tests. This material was removed and the structure cleaned to allow the installation of strain gages and TPS (Figure 3.1-6). Prior to installation of the tiles, strain gages were affixed to the outer skin of the door so that the structural response could be measured during impact testing. Discussion of their installation is in Section 3.1.2.3.

Part of the TPS function is to create the OML surface of the vehicle; thus, the tiles on the surface of the MLGD vary in thickness from 1.4 in. on the inboard edge to 2.0 in. on the outboard edge as shown in Figure 3.1-7. To minimize the complexities in the tile
configuration, a standard tile thickness of 1.7 in. was chosen. This represented the thickest tile on an MLGD and would allow for the evaluation of the deepest tile penetration. The plan form of the tile was also standardized so that all tiles in the test would have the same configuration. The configuration selected was 6” x 6” with the sidewalls having a 15-degree taper to the outboard edge of the vehicle. This sidewall configuration is representative of tiles on the MLGDs. A basic layout of the tile configuration is shown in Figure 3.1-8.

Figure 3.1-6: Photograph of ground operations crews removing the simulated TPS and preparing the door for tile TPS installation.

Figure 3.1-7: Contour plot of tile thickness on the Orbiter lower surface. The values shown are tile thickness in inches [18].
Figure 3.1-8: A schematic of the left MLGD skin showing the attachment locations for the 120 LI-900 ceramic tiles.
All tiles installed on the MLGD were LI-900. The installations were conducted per standard Orbiter processes [14]. The gap requirement between tiles was established at 0.045 inches ± 0.015 inches. All gaps between tiles were filled with ceramic AMES gap fillers (Figures 3.1-9 through 3.1-10).

3.1.2.3 Instrumentation

The instrumentation package for the MLGD articles was designed to enable dynamicists to verify their structural dynamics (such as DYNA) and explicit nonlinear and hybrid codes. Accelerometers, strain gages, and deflection gages were selected to measure the required data.

Figure 3.1-11 shows the instrumentation layout for the MLGD test articles. Three 2-axis accelerometer sets capable of measuring vibrations from 5 Hz to 5000 Hz were attached at the locations labeled GX 1-3 and GZ 1-3. They were bonded to the inner skin along the center stiffener of the MLGD. In addition, three single-axis accelerometers capable of measuring accelerations up to 1500 Hz were attached at the locations labeled AZ 1-3. The single axis accelerometers were also bonded to the inner skin of the MLGD along the center stiffener.

Figure 3.1-9: Photograph of LI-900 tiles installed on the OV-101 port MLGD. Note the black material in between the tiles are Ames gap fillers. The tiles are all 1.7” thick with a 15 degree sidewall taper.
Eight rosette strain gages with a response of up to 25,000 Hz were bonded to the outside skin of the MLGD before tile installation. Furthermore, all of the wiring was run in such a way as not to interfere with the tile bonding surfaces. The locations of the strain gages are noted in the figure as S1-8.

Four deflection gages with responses of up to 500 Hz were mounted to the door interior skin along the MLGD center stiffener. The locations of the gages are noted in the figure as D1-4.
Figure 3.1-11: Schematic of the MLGD instrumentation layout.
3.2 Leading Edge Structural Subsystem (LESS) Test Article

The LESS test article was designed and constructed to provide a representative wing leading edge structure for impact testing. It was determined early on in the Investigation that testing an individual panel supported in some non-flight representative fashion would not provide sufficient fidelity for the test program. The test article was designed to represent the left wing leading edge of OV-102 in the area of panels five through ten (known also as panels 5L to 10L). This area is shown in Figure 3.2-1.

![Figure 3.2-1: Test article location relative to Orbiter [16].](image)

A computer generated model of the left wing structure in this region is shown in Figure 3.2-2. For the purposes of the LESS impact testing the wing structure was analyzed to determine ways of simplifying the test article that would still yield an accurate response.

Many simplifications were made to reduce the complexity of manufacturing since schedule was also an important consideration [19]. Most of these compromises were
material substitutions to simplify machining and material availability. One important consideration was just how much of the wing structure was needed to mimic an accurate response. It was decided to test RCC panels 6L and 9L since the launch film/video analysis and forensics pointed to a likely impact location on one of these panels. Fiberglass panels 5L, 7L, 8L, and 10L were used to provide adequate boundary conditions for the testing of panels 6L and 9L (ultimately panel 8L was tested instead of 9L). Furthermore, the test article wing structure ends seven inches aft of the spars and is attached to a test fixture. This minimized the overall size of the test article yet still allowed for the appropriate boundary conditions and load path necessary for the spars.

![Figure 3.2-2: Close-up of Orbiter wing structure near Panel 5L to 10L area [16].](image-url)
A photo of the LESS test article is shown in Figure 3.2-3. It includes RCC and Fiberglass panel and T-seal assemblies, spar fittings that attach the panels to the spar, wing leading edge spar structure, simulated wing structure that approximates the boundary conditions for the spar and a test support fixture.

Figure 3.2-3: Photograph of the LESS test article [16].

Manufacturing the LESS test article was a great challenge due to the short schedule. The schedule allocated was just over four weeks. Over 600 individual components had to be manufactured, metal finished, inspected and assembled. Over 12,000 purchased parts had to be procured. A total of five NASA centers and a number of private vendors were enlisted to help distribute the manufacturing workload. Computer Aided Manufacturing (CAM) was used extensively to reduce assembly time. Complex parts were manufactured directly from their Computer Aided Design (CAD) files. Most of the metallic components were manufactured at the Johnson Space Center,
while other components were taken from Enterprise, Discovery, Atlantis, and spares. The test article was assembled at the Johnson Space Center.

3.2.1 Fiberglass Panels

Only two RCC panels were available for impact testing; however, a complete set of fiberglass panels 5L through 10L were removed from the OV-101. Impact tests on the fiberglass panels were used to verify gun targeting accuracy and to evaluate instrumentation prior to the RCC tests. They were also used as boundary conditions for the RCC panels and to gather engineering data applicable to the RCC panel tests.

The OV-101 fiberglass panels were built with the same OML as the RCC parts and were physically interchangeable with the OV-102 panels. There were, however, three major differences between the OV-101 and 102 panels. The most obvious difference is that the OV-101 panels were made of fiberglass while the OV-102 panels were made of RCC. In addition, the OV-101 panels did not have internal stiffeners known as spanner beams. Finally, the attachment fittings on OV-101’s panels were made out of a different material than OV-102’s panels.

The fiberglass panels were manufactured with a ply for ply substitution of fiberglass for RCC. Each ply of fiberglass is 0.010 ±0.001 inches. This makes the fiberglass panel face sheets 0.190±0.019 inches thick. The fiberglass panels are thinner than the RCC panels (which are 0.233±0.020 inches in total thickness). The RCC panels are about 40% stiffer than the fiberglass panels and would be expected to deflect about 29% less under impact loading [16]. At the same time, the fiberglass panels are significantly stronger than the RCC panels and would be expected to survive higher
impact loads. It was expected, however, that the impact loads for a foam impactor would be approximately the same for the fiberglass and RCC panels. If the same load is applied to both the fiberglass and RCC panels, then the stresses on the RCC panels could be inferred from the response of impact testing of the fiberglass panels. The RCC panel would see approximately 66% of the stresses and 90% of the strains achieved on the fiberglass panels. These estimates were suggested for the time period where the foam makes direct contact with the panel face sheet. The dynamic response of the fiberglass panels, however, would be quite different than that of the RCC panels due to their differences in both mass and stiffness.

The second difference between the fiberglass and RCC panels was the presence of spanner beams. The spanner beams, also called moment ties, are shown in Figure 3.2-4. Their purpose is to help redistribute loads on the RCC panel ribs. Therefore spanner beams were added to the impacted fiberglass panels to help ensure that the impact loads would not break the fiberglass panel ribs. Fiberglass panels that only provided boundary conditions (5L, 7L, and 10L) did not have spanner beams added except for fiberglass panel 10L, which had spanner beams added for Fiberglass 8L Tests #1 and #2 only.

The metallic parts used to attach the fiberglass and RCC panels to the wing are known as shear and clevis fittings (shown in Figure 3.2-5). On OV-102, the shear and clevis fittings were made from Inconel 718 heat-treated to a tensile strength of 180 ksi. On OV-101, the shear and clevis fittings were made from 4340-alloy steel in Condition F1, with a tensile strength of approximately 150 ksi. Hardness tests were performed on the actual OV-101 shear and clevis fittings to confirm the strength. The results of the hardness tests were consistent with a tensile strength of 140-150 ksi.
Figure 3.2-4: Panel spanner beams [16].

Figure 3.2-5: Panel shear and clevis fittings [16].
Since the modulus of elasticity of both materials is virtually identical, and the fittings on OV-101 and OV-102 are dimensionally identical, the Enterprise parts have the same stiffness as the Columbia parts; however, the lower strength of the Enterprise parts was a concern.

An effort was made to determine whether or not the lower strength of the OV-101 fittings would cause them to undermine the impact tests. The impact loads on the test article were not known, making a stress analysis of each fitting under impact loads difficult. Based on available flight certification calculations, any part that had a margin of 0.29 or less (rounded up to 0.30) was replaced in order to match the strength of the OV-102 parts. As a result, some additional Inconel 718 fittings were manufactured for the fiberglass panels to replace the original OV-101 4340 fittings. New upper and lower shear fittings were added to fiberglass Panels 6L, 7L, 8L and 9L. A new lower outboard clevis fitting was added to fiberglass Panel 9L. The remaining clevis fittings were considered acceptable for use in the impact tests.

3.2.2 RCC Panels

3.2.2.1 RCC Material

RCC is a structural composite whose development was targeted for the high temperature re-entry environments of reusable space vehicles. This material had successfully demonstrated that capability on the Space Shuttle Orbiter. Through 112 flights, the RCC leading edge and nose cap had performed flawlessly and the LESS was considered one of the model systems on the vehicle for which all the design and performance parameters had been met.
3.2.2.2 Fabrication of RCC Material

Fabrication of Shuttle RCC begins with a rayon cloth that is converted into a graphite cloth and impregnated with a phenolic resin. This impregnated cloth is layed-up as a laminate of 19 to 38 plies and cured in an autoclave. After cure, the laminate is pyrolyzed at high temperature which converts the resin to carbon. To increase density and strength, the part is then impregnated with furfuryl alcohol in a vacuum chamber, cured, and pyrolyzed again to convert the furfuryl alcohol to carbon. The impregnation and pyrolysis processing is repeated three times to achieve the desired strength after which the part is trimmed, drilled and sanded to obtain a very close tolerance fit-up to the final assembly fixture. On steps and gaps in the stagnation area, machined part tolerances are held on the large molded RCC parts. Holding these tolerances has been one key of the program challenges and is an attention area for future application of carbon-carbon on leading edges and control surfaces. To obtain an oxidation-resistant coating, the material is packed in a retort with a mixture of alumina, silicon, and silicon-carbide, and placed in a furnace with an argon atmosphere at a temperature of 1650°C (3000°F). A diffusion reaction occurs that converts the outer carbon-carbon layers to silicon-carbide with no thickness increase. Oxidation resistance is then enhanced by impregnating the coated RCC part with tetraethyl-orthosilicate (TEOS) and overcoating with a sodium silicate sealer (Type A). Together, the silicon-carbide, TEOS, and Type A protect the internal layers of carbon-carbon from oxidation during entry into the Earth’s atmosphere. The advantages of RCC are its good high-temperature mechanical properties and high resistance to fatigue loading. The primary disadvantage of RCC is associated with loss of
strength due to subsurface oxidation, which is greatest at an intermediate operating
temperature around 870°C (1600°F).

3.2.2.3 Installation of RCC to the Orbiter

The maximum size of individual RCC components is limited, from a practical
sense, by the deflections of the structure and the linear growth that can be accommodated
conveniently in the design as the material expands at high temperature. There are 99
unique pieces of carbon-carbon on the Orbiter. The wing leading edge is made up of 44
RCC panels (22 for each wing) and the nose cap is a 9 piece assembly (one large nose
cap with 8 seals designed to accommodate the thermal growth of the assembly and
provide interface with the adjacent thermal protection system tiles). The RCC wing
panels are mechanically attached to the wing with a series of floating joints to reduce
loading on the panels caused by wing deflections and differential thermal expansion. The
RCC seals between wing leading edge panels are referred to as “T-seals” and allow
lateral motion for thermal expansion between the RCC leading edge cavity during entry.
An additional two pieces of RCC form the “arrowhead” and are bolted onto the vehicle
structure at the Orbiter external forward attach strut just aft of the nose wheel door. The
RCC design was developed because tiles originally attached in this area were blown off
during separation tests. The RCC design was expedited and installed prior to STS-1 and
survives the explosive separation forces very well.

3.2.2.4 Flight Hardware Used for Impact Testing

Table 3.2-1 lists the part numbers and the source of the parts for flight hardware
used for the Investigation.
Table 3.2-1
Flight Hardware Listing

<table>
<thead>
<tr>
<th>Part Number</th>
<th>Description</th>
<th>Source</th>
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<tbody>
<tr>
<td>V070-199806-031</td>
<td>RCC Panel 6</td>
<td>OV-103</td>
</tr>
<tr>
<td>V070-199806-041</td>
<td>RCC T-Seal 6</td>
<td>Spares</td>
</tr>
<tr>
<td>V070-199710-071</td>
<td>Lower Carrier Panel 6</td>
<td>OV-103</td>
</tr>
<tr>
<td>V070-199711-055</td>
<td>Upper Carrier Panel 6</td>
<td>OV-103</td>
</tr>
<tr>
<td>V070-199808-029</td>
<td>RCC Panel &amp; T-seal 8</td>
<td>OV-104</td>
</tr>
<tr>
<td>V070-199714-085</td>
<td>Lower Carrier Panel 8</td>
<td>OV-103</td>
</tr>
<tr>
<td>V070-199715-072</td>
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<tr>
<td>V070-199810-034</td>
<td>RCC Panel &amp; T-seal 10</td>
<td>OV-103</td>
</tr>
</tbody>
</table>

3.2.3 Carrier Panels

The carrier panels, also known as access panels, are used to bridge the gap between the RCC panels and the wing acreage tile. There were concerns that foam impacting on the RCC panels could cause collateral damage on the carrier panels, and so it was deemed important to include carrier panels in the LESS impact test article.

The carrier panel structure consists of a special extruded aluminum box beam with a 0.160 in. thick plate riveted to the top. The assembly is attached to the spar fittings using two bolts through spacers in the box beam. The upper access panel insulation consists of a single row of HRSI tile and a thermal barrier that seals against the RCC component. There is also a single row of interior tiles with the OML facing the RCC and
the sidewall of the tile to the OML. The lower access panel insulation consists of a single row of HRSI tiles and an additional row of small HRSI tiles that face the interior RCC cavity as shown in Figure 3.2-6. A horse collar gap filler assembly surrounds the tiles on the entire panel. The entire assembly is installed with an interference fit to all adjacent components.

![Diagram of carrier panel assembly with labels: Outer Mold Line Tiles, Tiles Facing RCC panels, Aluminum Box Beam.]

**Figure 3.2-6: Photograph of a carrier panel assembly [16].**

Time constraints did not allow for the fabrication of new carrier panels, but carrier panels from the shuttle OV-103 were available. Each RCC panel (6L, 8L, and 9L) that was impacted utilized these upper and lower carrier panels from OV-103. The carrier panels removed from OV-101 were used for all other test article panel locations. Though the OV-101 carrier panels were structurally similar they did not have real tile or gap fillers.

The OV-102 access panels consisted of LI-2200 with reaction cured glass (RCG) coating. Vehicles built after Columbia utilized the lighter FRCI-12 substrate with the same coating. Other than the substrate difference, all other features were the same. The
interface of the tiles with the panel for both vehicles is through a 0.090-inch thick strain isolation pad (SIP). The interface with the tiles was made through the horse collar gap filler. Because no direct interface existed between the tile and substrate, it was accepted to utilize the OV-103 access panels without modification of the tiles.

3.2.4 Spar Fittings

The spar fittings are metallic components that interface the panel shear and clevis fittings to the wing spar structure. OV-101 and OV-102 both have separate upper and lower fittings that are made from A-286 heat resistant alloy, while later vehicles have one-piece titanium fittings. The spar fittings for the test article were obtained from OV-101, and are shown in Figure 3.2-7. The OV-101 spar fittings were in very good condition and required only cleaning and the replacement of a few floating nut plate elements that had seized.

Figure 3.2-7: Photograph of the LESS test article showing the upper and lower spar fittings [16].
3.2.5 Wing Leading Edge Spar and Simulated Wing Structure

A complete transition spar, a partial leading edge spar and a simulated glove spar were constructed for the LESS test article as shown in Figure 3.2-8. This structure forms the wing leading edge mounting planes for the spar fittings and subsequently panels 5L through 10L. The components that make up the LESS test article wing leading edge spar structure are depicted in Figure 3.2-9.

The simulated wing structure simulates the X<sub>w</sub>1009.75 frame, X<sub>w</sub>1040 spar, Y<sub>w</sub>198 rib, Y<sub>w</sub>226 rib, Y<sub>w</sub>254 rib and upper and lower skin panels (Figure 3.2-10). Each of the ribs and the frame are represented by rib caps, strongback rib cap angle clips, truss tubes and strongback truss tube fittings. All of this structure ties the spar structure to the test article strongback. These components represent the Orbiter wing structure but are not intended to be duplications. Figure 3.2-9 and 3.2-10 show the general differences in the Orbiter wing structure and the LESS test article. The simulated wing structure is a good stiffness and strength representation of the vehicle structure and approximates the boundary conditions on the spar from the wing structure.
Figure 3.2-8: Photo of the LESS test article wing leading edge spar structure [16].

Figure 3.2-9: Wing Leading Edge Spar Structure (strongback and simulated wing structure removed for clarity) [16].
Figure 3.2-10: CAD assembly model showing simulated wing structure (strongback removed for clarity) [16].

3.2.6 Test Support Hardware

3.2.6.1 Strongback Test Fixture

The strongback test fixture was designed to provide a rigid attach structure for the simulated wing structure. The strongback was also designed to interface with the tilt table at SwRI (Figure 3.2-11). By orienting the LESS test article such that the RCC panels point upward, the tilt top table could provide the adjustment necessary to achieve the correct foam impact angle for the test. The tilt top table can rotate the test article in the wing z-axis and y-axis, and can translate in the wing x-axis, y-axis and z-axis (Figure 3.2-12). Additionally, the gun can rotate in the wing x-axis and y-axis. The strongback is attached to the tilt table through load cell brackets.
In order to mount the strongback to the SwRI Tilt Top Table, an arrangement of seven uni-axial load cells were used. Four support the test article in the Z direction, two in the Y direction, and one in the X direction. Figure 3.2-13 depicts the locations where the load cells attach to the strongback. Although only six load cells were needed for a statically determinate connection, seven were chosen for added stability. Only three were necessary on the base of the strongback, but due to its large size and beam orientation, four load cells (LC1-LC4) gave the structure more stability and potentially eliminated additional dynamic modes of the strongback (though this was never formally investigated).

Figure 3.2-11: Photograph of the RCC Test Article Strongback attached to the SwRI Tilt Table [16].
Figure 3.2-12: Side view drawing of the LESS test article strongback and tilt table [16].

Figure 3.2-13: RCC Test Article Load Cell Connection Location (Exploded View) [16].
Each connection consists of a load cell, a turnbuckle, and two rod ends that connect to clevises attached to the strongback and tilt table (Figure 3.2-14). In addition, brackets were used to attach the load cells to the tilt table at three of the seven locations due to the orientation of the load cell connection (Figure 3.2-15).

To accommodate transportation, swivel eye lifting hooks were attached to the strongback and used as tie-down points (Figure 3.2-16). This allowed the article to be tied down in a variety of positions depending on the attachment points on the delivery truck.

Figure 3.2-14: Photograph of an LESS test article typical load cell connection [16].
Figure 3.2-15: Photograph of an LESS test article typical load cell bracket [16].

Figure 3.2-16: Photograph showing the Swivel Hook and Strap Use Locations for Transportation [16].
3.2.6.2 Spreader Bar

A spreader bar was the lifting fixture for the LESS test article. It interfaces the strongback with the lower lugs, and a crane hoist with the upper lugs (Figure 3.2-17). It enabled the test article to be lifted in an orientation with the transition spar horizontal with respect to the ground. The spreader bar was made entirely of wide flange I-beams and welded at all joints. It was designed so that the test article could be lifted with or without the RCC panels on the test article.

Figure 3.2-17: Photograph of the Spreader Bar Lifting the LESS Test Article [16].
3.2.6.3  Debris Shield

The debris shield was used to provide protection to the inner exposed cavity of Panel 5L. Though it would have been nearly impossible for the foam to miss the intended target by that kind of margin, the debris shield also provided protection to minimize the amount of air surrounding the foam that could enter the Panel 5L cavity and potentially disrupt gauges and sensors (Figure 3.2-18).

The shield was designed for a Panel 6 impact and its orientation was dictated by the direction and velocity of the foam determined several months prior to the actual test. The impact angle changed several times leading up to the test so the orientation of the debris shield was not positioned exactly as intended; however, it still served its purpose.

The shield was made of steel plates and beams and it was welded at all joints (Figure 3.2-19). The debris shield assembly bolted to the strongback using the lower plate and two support brackets. The debris shield also served the purpose of a camera mount for the foam impact shot at Panel 6L. A camera was desired to provide a view inside the cavity looking at Panel 6L. Extra protection was given to shield the camera from unwanted light and the air blast that accompanies the foam.
Figure 3.2-18: Schematic showing the debris shield location [16].

Figure 3.2-19: Schematic of the debris shield [16].
3.3 Test projectile fabrication

3.3.1 Bipod Ramp TPS Application

BX-250 is a closed-cell urethane foam manufactured by Stepan Chemical Company. This foam is utilized in selected external tank (ET) thermal protection system closeout areas primarily to meet pre-launch requirements for protecting against ice, frost and air liquefaction. BX-250 foam material is typically used in areas of low heating rates and low shear environments.

The bipod spindle (shown in Figure 3.3-1) enables the structural attachment of the Orbiter nose to the ET via two forward attach struts. Each strut is fastened at one end to the Orbiter and at the other to the ET bipod clevis. The bipod spindle clevis is designed to allow some movement of the strut at this connection. Since the ET is filled with cryogenic fluids prior to launch, a heater element is incorporated in the design to enable nominal movement of the clevis attachment. Finally, to minimize the loss of cryogenic propellant, the bipods are closed out with a combination of ablator and foam TPS called the bipod ramp. A ramped shape is utilized for aerodynamic considerations. Figure 3.3-2 shows a completed bipod ramp. The fabrication of the bipod ramp TPS is shown in the production sequence of Figures 3.3-3 to 3.3-5.

Thermal protection materials are applied at both the component level and closeout level during the build process. Prior to application of the TPS, a surface preparation may include the application of a primer or an adhesive to the surface. These applications are all performed within specified environmental conditions and typically require curing at specified conditions. Environmental conditions such as temperature and humidity are
monitored and verified using calibrated devices at the start of each process and maintained throughout the process.

Figure 3.3-1: Photograph of the bipod spindle.

Figure 3.3-2: Photograph of a completed BX-250 bipod ramp.
The following manufacturing spray process describes the application of BX-250 after ablator, Super Light Ablator (SLA) and PDL-1034-2.5LNO resin (PDL) have been applied. The BX-250 spray starts at the aft side of the bipod web or pocket (Figure 3.3-3, Step 1) then migrates to the aft side of the clevis (Figure 3.3-3, Step 2) where the PDL and NCFI edge interface with one another. The spray then continues onto the inboard side of the spindle area and body of the bipod working until sufficient foam is built up at which time it continues towards the outboard side of the spindle (Figure 3.3-3, Step 3). The outboard side is then built up similarly to the inboard side until sufficient foam is in place. The areas beneath the flange bolts and the forward pocket (Figure 3.3-4, Step 4) is then filled with foam until the foam is approximately flush with the BX flange foam at which time the spray shifts to the forward end of the ramp starting on the outboard side. Spray continues from the front of the ramp towards the bipod spindle making sure the intertank valleys to sidewall are tied into the bipod fitting (Figure 3.3-5, Step 5). Spray then moves back to the front of the ramp and the second shingle spray pass is performed making sure that the overlap time between consecutive foam application passes does not exceed 45 seconds maximum. This is continued until the forward portion of the ramp is complete. Sprayed foam is allowed to cure for 24 hours prior to machining.

3.3.2 Projectile Fabrication

Planks of BX-250 foam measuring 2 feet wide, 6 feet long and 10 inches thick were shingle sprayed in the same orientation as the front portion of the bipod ramp in order to obtain appropriate knitlines between successive sprayed layers. All test projectiles were cut and then fabricated from these BX-250 planks. All planks were
Figure 3.3-3: Steps 1 through 3 of the ramp application process are depicted by the red arrows. Foam sprays are completed in the direction of the arrows building up the foam in a single-like pattern.

Figure 3.3-4: Photograph of the bipod and bipod pocket. Step 4 of the ramp application process is depicted by the red arrows. The areas beneath the flange bolts and forward pocket are filled with foam until flush with the BX flange foam.
Figure 3.3-5: Photograph of the completed bipod. Step 5 of the ramp application process is depicted by the red arrow. The ramp is sprayed to essentially fill the box denoted by the white lines. Hand machining of the foam block achieves the final ramp shape.

manually shingle sprayed using a type “D” spray gun with a 55L776 spray tip and an HII proportioner. The spray booth, material temperature, substrate temperature and relative humidity were within the spray parameters stipulated in STP-1503 which is the governing document for the application of BX-250 [20]. This spray was intended to replicate the forward section of the bipod ramp which is a manual shingle sprayed configuration. No machining of foam planks was allowed until two weeks after spray application in order for the material to fully stabilize at which time projectiles were fabricated to the final gun barrel dimensions. Projectiles were cut from the plank where the knitlines were in the same plane orientation as those seen on actual flight configured bipods. Due to the slight range in density from projectile to projectile, the final length of each projectile exhibited some variability in order to meet the final weight requirement stipulated by the test plan.
3.3.3 Material properties

Table 3.3-1 lists the individual test numbers for the corresponding projectile identification numbers. After the fabrication of each BX-250 projectile plank, the material properties of compression strength, tensile strength, number of knitlines and density for each foam sample were analyzed in accordance with ASTM procedures. All projectiles were within the acceptable allowable limits for foam production.

Table 3.3-1
Material property data for BX-250 foam projectiles

<table>
<thead>
<tr>
<th>Test No.</th>
<th>Projectile ID</th>
<th>Spray Number</th>
<th>BX-250 Panel ID</th>
<th>Density pcf</th>
<th>Number Knitlines</th>
<th>Compression Strength psi</th>
<th>Flatwise Tension psi</th>
</tr>
</thead>
<tbody>
<tr>
<td>MLGD-1</td>
<td>BX-250-4</td>
<td>464</td>
<td>BX-250-2</td>
<td>2.113</td>
<td>1.51</td>
<td>29.08</td>
<td>80.62</td>
</tr>
<tr>
<td>MLGD-2</td>
<td>BX-250-2</td>
<td>464</td>
<td>BX-250-2</td>
<td>2.113</td>
<td>1.51</td>
<td>29.08</td>
<td>80.62</td>
</tr>
<tr>
<td>MLGD-3</td>
<td>BX-250-9</td>
<td>464</td>
<td>BX-250-2</td>
<td>2.113</td>
<td>1.51</td>
<td>29.08</td>
<td>80.62</td>
</tr>
<tr>
<td>MLGD-4</td>
<td>BX-250-5</td>
<td>464</td>
<td>BX-250-2</td>
<td>2.113</td>
<td>1.51</td>
<td>29.08</td>
<td>80.62</td>
</tr>
<tr>
<td>MLGD-5</td>
<td>BX-250-7</td>
<td>464</td>
<td>BX-250-2</td>
<td>2.113</td>
<td>1.51</td>
<td>29.08</td>
<td>80.62</td>
</tr>
<tr>
<td>FG #6 – 1</td>
<td>BX-250-15</td>
<td>487</td>
<td>BX-250-3</td>
<td>2.1432</td>
<td>1.15</td>
<td>27.14</td>
<td>67.28</td>
</tr>
<tr>
<td>RCC #6 – 1</td>
<td>BX-250-13</td>
<td>487</td>
<td>BX-250-3</td>
<td>2.1432</td>
<td>1.15</td>
<td>27.14</td>
<td>67.28</td>
</tr>
<tr>
<td>FG #6 – 2</td>
<td>BX-250-4-2</td>
<td>487</td>
<td>BX-250-4</td>
<td>2.1184</td>
<td>1.42</td>
<td>38.46</td>
<td>80.98</td>
</tr>
<tr>
<td>FG #6 – 3</td>
<td>BX-250-4-1</td>
<td>487</td>
<td>BX-250-4</td>
<td>2.1184</td>
<td>1.42</td>
<td>38.46</td>
<td>80.98</td>
</tr>
<tr>
<td>FG #8 – 1</td>
<td>BX-250-4-4</td>
<td>487</td>
<td>BX-250-4</td>
<td>2.1184</td>
<td>1.42</td>
<td>38.46</td>
<td>80.98</td>
</tr>
<tr>
<td>FG #8 – 2</td>
<td>BX-250-4-3</td>
<td>487</td>
<td>BX-250-4</td>
<td>2.1184</td>
<td>1.42</td>
<td>38.46</td>
<td>80.98</td>
</tr>
<tr>
<td>RCC #8 – 1</td>
<td>BX-250-4-5</td>
<td>487</td>
<td>BX-250-4</td>
<td>2.1184</td>
<td>1.42</td>
<td>38.46</td>
<td>80.98</td>
</tr>
</tbody>
</table>
4.0 Results

4.1 Main landing gear door test program

4.1.1 Test matrix

A total of 4 tests were originally planned for the left MLGD. Table 4.1-1 lists the parameters for each impact test. Note that the smallest debris sizes (1.19 lbm) were tested first in order to minimize the potential for structural damage to the door. The projectile sizes for tests 1 and 2 were established based on the MAF CAD analysis as discussed previously. The velocity and impact angle for these two tests were based on imagery and transport analysis at the time that indicated the average impact velocity on the MLGD was 700 ft/s and the likely impact angle was 8 degrees or less [21]. Since it was unclear which projectile clocking angle would yield worst case thermal performance damage, clocking angles of 90 and 0 degrees were used for tests 1 and 2 respectively. Comparison of the damage from these two tests then yielded the worst case clocking angle that was used in the remaining 2 tests.

Upon completion of MLGD test 3, the NAIT and CAIB suggested increasing the impact velocity to 775 ft/s to reflect improved transport analyses. The reference projectile mass was also changed from the 2.5 lbm to 1.67 lbm in light of these new analyses. It was determined that this size was more representative of the ascent debris. Given this change, it was further decided by the Investigation team to add a test 5 to determine if the tile could be damaged at higher impact angles. These improved transport analyses showed that 13 degrees was the maximum possible impact angle attainable for bipod ramp foam striking the MLGD [21].
Table 4.1-1
MLGD Test Matrix

<table>
<thead>
<tr>
<th>Test Number</th>
<th>Projectile volume (in$^3$/mass (lb))</th>
<th>Projectile dimensions* (inches)</th>
<th>Velocity (ft/s)</th>
<th>Local angle of impact (degrees)</th>
<th>Clocking angle (degrees)</th>
</tr>
</thead>
<tbody>
<tr>
<td>MLGD 1</td>
<td>855/1.19</td>
<td>3.5x11.5x21.25</td>
<td>700</td>
<td>5</td>
<td>90</td>
</tr>
<tr>
<td>MLGD 2</td>
<td>855/1.19</td>
<td>3.5x11.5x21.25</td>
<td>700</td>
<td>5</td>
<td>0</td>
</tr>
<tr>
<td>MLGD 3</td>
<td>1800/2.5</td>
<td>5.5x11.5x28.5</td>
<td>700</td>
<td>5</td>
<td>90</td>
</tr>
<tr>
<td>MLGD 4</td>
<td>1201/1.67</td>
<td>5.5x11.5x19</td>
<td>775</td>
<td>8</td>
<td>90</td>
</tr>
<tr>
<td>MLGD 5</td>
<td>1201/1.67</td>
<td>5.5x11.5x19</td>
<td>775</td>
<td>13</td>
<td>90</td>
</tr>
</tbody>
</table>

*The length of each projectile was varied to match the mass described in the test matrix.

4.1.2 Pre-test damage predictions

Prior to the MLGD testing, a team of analysts used existing modeling tools to predict the expected damage for each test. The team generated predictions for the extent of tile cratering and the structural response of the MLGD [22]. Tables 4.1-2 through 4.1-4 document the extent of crater damage expected for tests 1-5.
### Table 4.1-2

Pre-test predictions for MLGD test 1 and 2

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Sandia</td>
<td>CTH</td>
<td>Test 1: 11.5&quot;x3.5&quot;x21&quot;</td>
<td>5</td>
<td>none</td>
<td>30</td>
<td>5.5</td>
<td>0</td>
<td>Coating damage likely, possible gouge to 1cm if compressive strength 70ps</td>
</tr>
<tr>
<td>MSFC/Stellingwerf</td>
<td>SPHC</td>
<td>Test 1: 11.5&quot;x3.5&quot;x21&quot;</td>
<td>5</td>
<td>0.4</td>
<td>12</td>
<td></td>
<td>3.5</td>
<td></td>
</tr>
<tr>
<td>UT/ Fahrenhold</td>
<td>EXOS</td>
<td>Test 1: 11.5&quot;x3.5&quot;x21&quot;</td>
<td>5</td>
<td>1.1</td>
<td>15</td>
<td>4</td>
<td>19.6</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>Test 2: 3.5&quot;x11.5&quot;x21&quot;</td>
<td>5</td>
<td>0.9</td>
<td>6</td>
<td>6</td>
<td>12.0</td>
<td></td>
</tr>
<tr>
<td>Boeing</td>
<td>Classic Crater</td>
<td>Test 1: 11.5&quot;x3.5&quot;x21&quot;</td>
<td>5</td>
<td>1.87</td>
<td>25.8</td>
<td>4.2</td>
<td>70</td>
<td>Range in predictions based on range of tile strengths</td>
</tr>
<tr>
<td></td>
<td>Revised Crater</td>
<td>Test 1: 11.5&quot;x3.5&quot;x21&quot;</td>
<td>5</td>
<td>0.99</td>
<td>11.3</td>
<td>8.3</td>
<td>16</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>Test 2: 3.5&quot;x11.5&quot;x21&quot;</td>
<td>5</td>
<td>0.72</td>
<td>8.4</td>
<td>4.2</td>
<td>8.4</td>
<td></td>
</tr>
<tr>
<td>JSC/SX</td>
<td>Semi-Empirical</td>
<td>Test 1: 11.5&quot;x3.5&quot;x21&quot;</td>
<td>5</td>
<td>0.54</td>
<td></td>
<td></td>
<td></td>
<td>Original estimate of ricochet without penetration was changed</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Test 2: 3.5&quot;x11.5&quot;x21&quot;</td>
<td>5</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>Ricochet, No damage to target</td>
</tr>
</tbody>
</table>

### Table 4.1-3

Pre-test predictions for MLGD test 3 and 4

<table>
<thead>
<tr>
<th></th>
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<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Sandia</td>
<td>CTH</td>
<td>Test 3: 11.5&quot;x5.5&quot;x28.5&quot;</td>
<td>5</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>MSFC/Stellingwerf</td>
<td>SPHC</td>
<td>Test 3: 11.5&quot;x5.5&quot;x28.5&quot;</td>
<td>5</td>
<td>1.7</td>
<td>15</td>
<td>5 to 6</td>
<td>100</td>
<td>SIP is penetrated in places. Back of projectile does most of damage</td>
</tr>
<tr>
<td>UT/ Fahrenhold</td>
<td>EXOS</td>
<td>Test 3: 11.5&quot;x5.5&quot;x28.5&quot;</td>
<td>5</td>
<td>1.25</td>
<td>18</td>
<td>5.5</td>
<td>48.2</td>
<td>Final results of simulation to 6000 micro-sec</td>
</tr>
<tr>
<td>Boeing</td>
<td>Classic Crater</td>
<td>Test 3: 11.5&quot;x5.5&quot;x28.5&quot;</td>
<td>5</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>Range in predictions based on range of tile strengths</td>
</tr>
<tr>
<td></td>
<td>Revised Crater</td>
<td>Test 3: 11.5&quot;x5.5&quot;x28.5&quot;</td>
<td>5</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>JSC/SX</td>
<td>Semi-Empirical</td>
<td>Test 3: 11.5&quot;x5.5&quot;x28.5&quot;</td>
<td>5</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>Ricochet, No damage to target</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Test 4: 11.5&quot;x5.5&quot;x19&quot;</td>
<td>8</td>
<td>0.75</td>
<td>6.2</td>
<td>6</td>
<td>14</td>
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</tbody>
</table>
### Table 4.1-4
Pre-test predictions for MLGD test 5

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<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Sandia/ Crawford</td>
<td>CTH</td>
<td>Test 5: 11.5&quot;x5.5&quot;x19&quot;</td>
<td>13</td>
<td>.6</td>
<td>31</td>
<td>5.5</td>
<td>51</td>
<td></td>
<td>Volume estimated using half of maximum depth</td>
</tr>
<tr>
<td>MSFC/ Stellingwerf</td>
<td>SPHC</td>
<td>Test 5: 11.5&quot;x5.5&quot;x19&quot;</td>
<td>13</td>
<td>.78</td>
<td>25</td>
<td>5.5</td>
<td>55</td>
<td></td>
<td>Results at 3.5 msec</td>
</tr>
<tr>
<td>UT/ Fahrenheit</td>
<td>EXOS</td>
<td>Test 5: 11.5&quot;x5.5&quot;x19&quot;</td>
<td>13</td>
<td>to SIP</td>
<td></td>
<td></td>
<td>78.6</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Boeing/ Parker Crater V1.1</td>
<td></td>
<td>Test 4: 11.5&quot;x5.5&quot;x19&quot;</td>
<td>8</td>
<td>1.9</td>
<td>16.6</td>
<td>6.6</td>
<td>70</td>
<td></td>
<td>Baseline</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td>0.1</td>
<td>1.0</td>
<td>6.6</td>
<td>0.2</td>
<td></td>
<td>Adjusted for typical TPS strength and 85% increase in threshold velocity</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Test 5: 11.5&quot;x5.5&quot;x19&quot;</td>
<td>13</td>
<td>3.4</td>
<td>18.4</td>
<td>6.6</td>
<td>136</td>
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<td>Baseline</td>
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<td></td>
<td></td>
<td></td>
<td></td>
<td>1.8</td>
<td>9.9</td>
<td>6.6</td>
<td>39.5</td>
<td></td>
<td>Adjusted for typical TPS strength and 85% increase in threshold velocity</td>
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<td>JSC-SX/ Christiansen</td>
<td>Semi-Empirical</td>
<td>Test 4: 11.5&quot;x5.5&quot;x19&quot;</td>
<td>8</td>
<td>0.8</td>
<td></td>
<td>6</td>
<td>6</td>
<td>27</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>Test 5: (2.0 lb/ft²) 11.5&quot;x5.5&quot;x22.8&quot;</td>
<td>13</td>
<td>1.0</td>
<td>8.5</td>
<td>6</td>
<td>27</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

### 4.1.3 Instrumentation and high speed video set up

#### 4.1.3.1 Sensor placement and data acquisition configuration

For the MLGD tests, a variety of rosette strain gages, deflection gages and accelerometers were installed on the article to measure its response. Figure 4.1-1 shows the precise layout of these sensors for all of the tests. Table 4.1-5 documents the manufacturers and model numbers of the sensors used. In addition, the table describes the quantity and types of data acquisition equipment to support the MLGD tests.
Proposed Test Panel Instrumentation Layout:
8 x S = Rosette Strain gages bonded to outboard side of outer skin
4 x Dz = Deflection gages mounted to inboard side of inner skin (for deflections normal to MLGD)
2 x Dx = Deflection gages mounted to forward corners of door (for longitudinal deflections of MLGD)
2 x Dy = Deflection gages mounted to outboard corners of door (for lateral deflections of MLGD)
2 x B = Bi-axial accelerometers mounted to inboard side of inner skin (for out of panel & longitudinal accelerations)
16 Uni-axial Load cells (not shown) - 1 at each of the 4 up stops (up z loads) and 4 latches (z loads), 2 at the forward hinge and 2 at the aft hinge (y and z loads) and 3 at the center hinge (x, y, and z loads)

Figure 4.1-1: Schematic of the sensor layout for the MLGD test program.

4.1.3.2 High speed video camera configuration

For the MLGD tests a total of six high speed cameras enabled the measurement of projectile launch and target impact. Table 4.1-6 documents the camera type and settings used for the tests. Figure 4.1-2 shows the camera layout for the MLGD tests. Camera L1 imaged a close-up of the target impact area from an oblique view (Figure 4.1-3). A Phantom 7 camera was selected for this view to enable the largest viewing area at the highest possible resolution and fastest frame rate. The remaining cameras were all Phantom 5s. Views L2 (Figure 4.1-4) and L3 (Figure 4.1-5) imaged the projectile from the side of its flight path. The L2 view captured images of the projectile just after leaving the barrel until it just reached the target, while the L3 view imaged the projectile motion across the target to the ejecta catcher. The L3 camera images were primarily used for determination the projectile velocity.
Table 4.1-5
Sensor selection and data acquisition system configuration for the MLGD tests 1 - 5

<table>
<thead>
<tr>
<th>DESCRIPTION</th>
<th>QTY</th>
<th>MANUFACTURER</th>
<th>MODEL NO.</th>
<th>NOTES</th>
</tr>
</thead>
<tbody>
<tr>
<td>Data Acquisition System</td>
<td>2</td>
<td>Spectral Dynamic</td>
<td>Sun Ultra 10</td>
<td>Jaguar data acquisition systems</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td>Spectral Dynamic</td>
<td>SD 2570</td>
<td>Analysis &amp; control peripheral (ACP)</td>
</tr>
<tr>
<td>Tape Recorder</td>
<td>4</td>
<td>Sony</td>
<td>SIR-1000</td>
<td>All recorders are synchronized</td>
</tr>
<tr>
<td>Strain Gage</td>
<td>8</td>
<td>Measurement Group</td>
<td>CEA Type</td>
<td>Rosettes</td>
</tr>
<tr>
<td>Strain Gage Signal Conditioner</td>
<td>2</td>
<td>Hewlett-Packard</td>
<td>E1529A</td>
<td>Agilent remote signal conditioner</td>
</tr>
<tr>
<td>Displacement Transducer</td>
<td>8</td>
<td>RDP Group</td>
<td>ACT1000C</td>
<td>LVDT</td>
</tr>
<tr>
<td>LVDT Signal Conditioner</td>
<td>8</td>
<td>RDP Group</td>
<td></td>
<td></td>
</tr>
<tr>
<td>DC Accelerometer</td>
<td>6</td>
<td>Endevco</td>
<td>7264B-500T</td>
<td>500G bridge type accelerometer</td>
</tr>
<tr>
<td>DC Accelerometer Sig. Conditioner</td>
<td>1</td>
<td>Hewlett-Packard</td>
<td>E1529A</td>
<td>Agilent remote signal conditioner</td>
</tr>
<tr>
<td>AC Accelerometer</td>
<td>6</td>
<td>Endevco</td>
<td>2256A-2.5</td>
<td>ICP type accelerometers</td>
</tr>
<tr>
<td>AC Accelerometer Sig. Conditioner</td>
<td>1</td>
<td>PCB Piezotronics</td>
<td>584A</td>
<td></td>
</tr>
<tr>
<td>IRIG Signal Generator/Slow Code</td>
<td>1</td>
<td>Datachron</td>
<td>3070</td>
<td>Channel 31 of each tape recorder</td>
</tr>
<tr>
<td>Trigger Signal (SWRI provided)</td>
<td>1</td>
<td>Nicolet</td>
<td></td>
<td>Channel 32 of each tape recorder</td>
</tr>
<tr>
<td>Power Supply</td>
<td>3</td>
<td>Hewlett-Packard</td>
<td>E3632A</td>
<td>Used to supply power to the HP E1592A signal conditioner</td>
</tr>
<tr>
<td></td>
<td>6</td>
<td>Hewlett-Packard</td>
<td>E3610A</td>
<td></td>
</tr>
<tr>
<td>Printer</td>
<td>1</td>
<td>Hewlett-Packard</td>
<td>8500</td>
<td>Color Printer</td>
</tr>
</tbody>
</table>

Figure 4.1-6 shows typical images from views L4 and L5. These oblique cameras imaged the projectile flight and target pre and post impact. The cameras were setup orthogonal to each other so as to make the measurement error equal in each dimension. Views L4 and L5 enabled a multi-camera analysis of the projectile motion and any ejecta produced by its impact with the target. The overhead camera L6 (Figure 4.1-7) was used in combination with L2 and L3 to measure the projectile pitch and yaw just prior to impact. It also provided a redundant view for the measurement of projectile velocity.
Figure 4.1-2: Camera layout for MLGD tests 1-5

<table>
<thead>
<tr>
<th>Camera Location</th>
<th>Camera Type</th>
<th>Lens Focal Length (mm)</th>
<th>Image Format (WxH)</th>
<th>Frame Rate (fps)</th>
<th>Scene Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>L1</td>
<td>Phantom 7</td>
<td>50</td>
<td>640x480</td>
<td>7000</td>
<td>Close-up of impact zone</td>
</tr>
<tr>
<td>L2</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile prior to impact</td>
</tr>
<tr>
<td>L3</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile after impact</td>
</tr>
<tr>
<td>L4</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile and debris</td>
</tr>
<tr>
<td>L5</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile and debris</td>
</tr>
<tr>
<td>L6</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x1024 or</td>
<td>2000 or 1000*</td>
<td>Overhead close-up of impact zone</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>1024x512**</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

* Test 1 only
** Test 3 only
Figure 4.1-3: Camera view from L1 location. A close-up of the foam impact area.

Figure 4.1-4: Camera view from L2 location.

Figure 4.1-5: Camera view from L3 location.
Figure 4.1-6: Camera view from L4 (top) and L5 (bottom) locations. These cameras are mounted perpendicular to each other and have oblique views of the impact area and post-impact foam debris scatter field. These views are intended for three-dimensional analysis of the pre-impact foam projectile and post-impact foam and TPS debris.
4.1.4 Results of main landing gear door testing

4.1.4.1 MLGD test 1 results

MLGD Test #1 had an actual projectile impact velocity of 723 ft/sec and a pitch and yaw at very close to zero. The projectile used for this test was made of BX 250 foam material having a projectile identification number of BX250-4. Its dimensions were 3.58” x 11.47” x 24.97” and its mass was 541 grams. The projectile leading edge struck 0.9-inch lower than the desired target, resulting in the projectile striking the side edge of the tile 0.16-inch below the top surface. There was no measurable offset between the
impact and targeted location left to right. The projectile impact caused one crater in each of the following tiles: tile 9, tile 16, and tile 26. Table 4.1-7 shows the dimensions of the craters. In order to completely characterize the cratering for this an all of the MLGD tests, SwRI performed non-destructive evaluations NDEs of the tiles using a structured light technique. The z-axis measurement recorded is at its deepest point. Figure 4.1-8 shows a photograph (Figure 4.1-8a) and a contour plot depiction (Figure 4.1-8b) of the largest crater.

<table>
<thead>
<tr>
<th>Table 4.1-7</th>
<th>Tile crater measurements for MLGD test 1</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Tile Number</strong></td>
<td><strong>Crater Size (mm)</strong></td>
</tr>
<tr>
<td></td>
<td>x-axis</td>
</tr>
<tr>
<td>9</td>
<td>5.6</td>
</tr>
<tr>
<td>16</td>
<td>20.9</td>
</tr>
<tr>
<td>26</td>
<td>16.4</td>
</tr>
</tbody>
</table>
4.1.4.2 MLGD test 2 results

MLGD Test #2 had an actual test projectile impact velocity of 717 ft/sec, with the pitch and yaw at near zero. The projectile used for this test was made of BX 250 foam material and had a projectile identification number of BX250-2. The foam had dimensions of 3.53" x 11.53" x 24.99" and a mass of 548 grams. The projectile leading
edge struck 0.9-inch lower than targeted and there was a 0.2-inch offset between the impact and targeted location left to right. The projectile impact caused no crater damage and no other tile damage was observed.

4.1.4.3 MLGD test 3 results

MLGD Test #3 was performed with a measured projectile impact velocity of 725 ft/sec, with the pitch and yaw at near zero. The projectile used for this test was made of BX 250 foam material and had a projectile identification number of BX250-9. The dimensional measurements were 5.63” x 11.62” x 33.44” and the mass was 1157 grams. The projectile leading edge struck vertically on target since corrections were made for the 0.9 inch drop observed in the previous tests. There was no measurable offset between the impact and targeted location left to right. The projectile impact caused one crater in tile 20 and one in tile 38 (as shown in Figure 4.1-9). Table 4.1-8 describes the specific crater dimensions.

<table>
<thead>
<tr>
<th>Tile Number</th>
<th>Crater Size* (inches)</th>
</tr>
</thead>
<tbody>
<tr>
<td>20</td>
<td>0.276 x 0.394</td>
</tr>
<tr>
<td>38</td>
<td>0.591 x 0.157</td>
</tr>
</tbody>
</table>

* Crater size data comes from hand measurement
4.1.4.4 MLGD test 4 results

MLGD test 4 had an actual projectile impact velocity that measured 827 ft/sec, a pitch of less than 1 degree and a yaw of 5 degrees. The projectile used for this test was made of BX 250 foam material having a projectile identification number of BX250-5, with dimensional measurements of 5.5” x 11.5” x 22.63”, and a mass of 755 grams. The projectile leading edge struck the target 1.7 inches higher than the aim point. There was no measurable offset between the impact and targeted location left to right. The projectile impact caused one crater in each of tiles 94, 101, 103, and 111. Table 4.1-9 states the dimensions of each of the craters. Figure 4.1-10 shows a photograph (Figure 4.1-10a) and a contour plot depiction (Figure 4.1-10b) of the tile 94 crater. Figure 4.1-11 shows a photograph (Figure 4.1-11a) and a contour plot depiction (Figure 4.1-11b) of the tile 103 crater.
Table 4.1-9
Tile crater measurements for MLGD test 4

<table>
<thead>
<tr>
<th>Tile Number</th>
<th>Crater Size (mm)</th>
<th>Crater Volume (mm³)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>x-axis</td>
<td>y-axis</td>
</tr>
<tr>
<td>94</td>
<td>21.7</td>
<td>84.4</td>
</tr>
<tr>
<td>101</td>
<td>28.7</td>
<td>15.8</td>
</tr>
<tr>
<td>103</td>
<td>27.7</td>
<td>82.3</td>
</tr>
<tr>
<td>111</td>
<td>80.0</td>
<td>12.1</td>
</tr>
</tbody>
</table>

Figure 4.1-10: Photograph and structured light measurement of MLGD test 4 tile 94 crater damage.
Figure 4.1-11: Photograph and structured light measurement of MLGD test 4 tile 103 crater damage.
4.1.4.5 MLGD test 5 results

MLGD test 5 had a measured projectile impact velocity of 787 ft/sec, with the pitch and yaw at near zero. The projectile used for this test was made of BX 250 foam material and had a projectile identification number of BX250-7. The dimensions were 5.56” x 11.5” x 22.13” and the mass was 754 grams. The projectile leading edge struck the target less than 0.5 inches higher than the aim point. There was no measurable offset between the impact and targeted location left to right. The projectile impact caused one crater in each of tiles 82, 88, 90, 100, and 107. Table 4.1-10 states the dimensions of each of the craters. Figure 4.1-12 shows a photograph (Figure 4.1-12a) and a contour plot depiction (Figure 4.1-12b) of the tile 88 crater.

Table 4.1-10
Tile crater measurements for MLGD test 5

<table>
<thead>
<tr>
<th>Tile Number</th>
<th>Crater Size (mm)</th>
<th>Crater Volume (mm³)</th>
</tr>
</thead>
<tbody>
<tr>
<td>82</td>
<td>x-axis</td>
<td>y-axis</td>
</tr>
<tr>
<td>88</td>
<td>15.9</td>
<td>38.9</td>
</tr>
<tr>
<td>90</td>
<td>40.1</td>
<td>7.0</td>
</tr>
<tr>
<td>100</td>
<td>6.5</td>
<td>22.8</td>
</tr>
<tr>
<td>107</td>
<td>8.5</td>
<td>24.5</td>
</tr>
<tr>
<td>107</td>
<td>22.8</td>
<td>7.0</td>
</tr>
</tbody>
</table>
4.1.5 MLGD testing discussion

During the flight of STS-107, the only analytical model available for estimating impact damage to tile was the semi-empirical "crater" code. This model was developed to provide conservative day of launch evaluations for potential impact damage and was developed using 3 in³ foam and was being extrapolated to the estimated STS-107 flight
condition of 1920 cu in. Crater predictions during flight and pre-test indicated that the tile impacted would be removed to its full thickness.

The results of the MLGD impact testing provided insight into the STS-107 Investigation and a look at the way in which the tile system reacts to foam debris of a larger size than had previously been evaluated. Impact testing of the MLGD tiles showed that a bipod ramp foam hit on acreage tile would not have resulted in a loss of vehicle. On the contrary, the damage that resulted from the impact test would have been repaired using conventional methods before the subsequent flight.

The resiliency of the tile system to withstand large foam can be attributed to multiple factors. The first is the low view factor the tile presents to the debris sources simply due to their location on the vehicle. Excluding some upper surface locations, debris traveling along the vehicle flow field would impact tile at an angle of less than 15 degrees. This low angle dramatically reduces the normal energy transferred into the tile. Additional variables such as the coefficient of friction between the debris source and the tile surface become increasingly important at low angles and will be evaluated in follow-on testing.

The second factor is the compliance of BX-250 debris source, even under high strain rate conditions. High-speed cameras showed the compression of the foam on the surface of the tile minimizing point loads/stress risers that could concentrate the impact energy. This loading and subsequent unloading of the foam on the tile surface carries away additional energy not transferred into the tile system. Impact testing conducted at NASA Glenn in support of foam characterization showed greater than 80% compression of the foam samples when impacted against a load cell.
The third factor is provided by the tile system construction. The tiles are coated with a high temperature fired ceramic coating. This coating, though fragile, presents a very hard surface. This hardness increases the failure threshold of the tile system. Another tile system feature not previously accounted for is the use of the Strain Isolation Pad (SIP). The only failures noted on the tile surface as a result of impact were located on the forward facing edge of the tile surface. This failure location indicates that the foam impacts compressed the impacted tiles exposing the forward edge of the trailing tiles. Previous impact testing on tiles was conducted on single tiles that were not bonded with SIP. Interaction of the foam with tile edges yields damage in some of the tests because locally the impact is at a near normal incidence.

Although the impact testing for the investigation showed the tile system to tolerate large foam impacts, post-investigation impact testing will be conducted to further understand and accurately model the impact threat from all debris sources and vehicle locations. Multiple tile test panels will be employed to characterize the damage caused by available debris sources on various tile types including new tiles with “toughened” coatings. Specific test panel configurations identified to be sensitive to impacts, including the MLGD parameter and the leading edge access panels, will be impacted to determine the robustness of the design. The damages created will allow the on-orbit repair team to plan for realistic damage levels that may require attention. This data will allow the program to eliminate threats, increase robustness and prepare contingencies for any future flight anomalies.
4.2 Leading Edge Structural Subsystem (LESS) Test Program

The LESS tests were conducted during the months of June and July. Fiberglass panels were impacted at various locations to evaluate panel strains and to establish the best location to impact the RCC panels to achieve failure. In particular, at the time of testing the forensic evidence indicated there was a breech in the LESS in the vicinity of panel 8L. Thermal analysis demonstrated that a 10 inch diameter hole in the lower surface of panel 8L would yield the sort of damage and MADS sensor response exhibited on OV-102.

A total of 2 tests were planned for the leading edge with RCC material installed. That is, one impact test was to be conducted on RCC panel 6L and one on panel 9L. As the forensic evidence matured, a test on panel 8L was substituted for the panel 9L test. Analysts evaluated varied impact points for these tests and determined that the results would differ widely based on where the panel was struck. An aiming error of 1 inch was established to insure that the article was struck sufficiently near the analyzed location. Given that aiming the gun was a critical function for these tests and RCC panels are of limited availability, a number of tests were conducted on fiberglass panels. These fiberglass panel tests ensured that gun aiming was accurate and instrumentation systems were collecting the appropriate data. The following sections describe the results of each test and critical knowledge gained to enable the final successful test of panel 8L.

4.2.1 LESS fiberglass and RCC panel 6L testing

Once the LESS test article was assembled on the tilt table, fiberglass panels 5-10 were installed. Table 4.2-1 lists the parameters for the fiberglass impact tests as designed
at the beginning of the test program while Table 4.2-2 shows the actual tests conducted. Figures 4.2-1 through 4.2-11 show the aim points for the four impact cases analyzed for panel 6L. Case 1 is an impact of a foam projectile corner on panel 6L while case 2 represents the impact of the projectile leading edge with the panel surface. Following these two tests the analysis team reviewed the instrumentation data and advised the test team which case should be used for RCC testing. Initially, the test team was going to conduct three tests on the fiberglass panel 6L to target the proper area. Once the gun was demonstrated to be properly aligned, a second test was going to be conducted to demonstrate impact location repeatability. Without moving the gun or the test article, technicians would then replace the fiberglass panel 6L with RCC panel 6L. The RCC test was then to be conducted using the demonstrated impact location. However, once the first test was conducted on fiberglass panel 6L the team redesigned the test matrix. The primary reason for this change was concern that additional tests on the panel would critically damage the spar hardware (of which there were no spares).

<table>
<thead>
<tr>
<th>Test Number</th>
<th>Projectile Volume (in³)/mass (lb)</th>
<th>Projectile dimensions (inches)</th>
<th>Test case</th>
<th>Velocity (ft/s)</th>
<th>Clocking angle (degrees)</th>
</tr>
</thead>
<tbody>
<tr>
<td>FG6-1</td>
<td>1200/1.67</td>
<td>5.5x11.5x22</td>
<td>Case 1</td>
<td>775</td>
<td>0</td>
</tr>
<tr>
<td>FG6-2</td>
<td>1200/1.67</td>
<td>5.5x11.5x22</td>
<td>Case 2</td>
<td>775</td>
<td>0</td>
</tr>
<tr>
<td>FG6-3</td>
<td>1200/1.67</td>
<td>5.5x11.5x22</td>
<td>Case 2, modified</td>
<td>775</td>
<td>30</td>
</tr>
<tr>
<td>RCC6</td>
<td>1200/1.67</td>
<td>5.5x11.5x22</td>
<td>TBD, Case 1, 2 or 2 modified</td>
<td>775</td>
<td>TBD</td>
</tr>
</tbody>
</table>

*FG6-x: fiberglass panel 6, test number x, RCCx: RCC panel x*
**Table 4.2-2**

Final panel 6 fiberglass/RCC test matrix

<table>
<thead>
<tr>
<th>Test Number</th>
<th>Projectile Volume (in³)/mass (lb)</th>
<th>Projectile dimensions (inches)</th>
<th>Test case</th>
<th>Desired Velocity (ft/s)</th>
<th>Clocking angle (degrees)</th>
</tr>
</thead>
<tbody>
<tr>
<td>FG6-1</td>
<td>1200/1.67</td>
<td>5.5x11.5x22</td>
<td>Case 1</td>
<td>775</td>
<td>0</td>
</tr>
<tr>
<td>RCC6</td>
<td>1200/1.67</td>
<td>5.5x11.5x22</td>
<td>Case 1</td>
<td>775</td>
<td>0</td>
</tr>
<tr>
<td>FG6-2</td>
<td>1200/1.67</td>
<td>5.5x11.5x22</td>
<td>Case 3</td>
<td>775</td>
<td>0</td>
</tr>
<tr>
<td>FG6-3</td>
<td>1200/1.67</td>
<td>5.5x11.5x22</td>
<td>Case 3 (modified)</td>
<td>775</td>
<td>30</td>
</tr>
</tbody>
</table>

*FG6-x: fiberglass panel 6, test number x, RCCx: RCC panel x

---

**Figure 4.2-1:** Drawing showing the case 1 corner impact (shot line view)
Figure 4.2-2: Drawing showing the case 1 corner impact (oblique view of the article)

Figure 4.2-3: Drawing showing the case 1 corner impact (side view of the article)
Figure 4.2-4: Drawing showing the case 2 edge impact (shot line view of the article)

Figure 4.2-5: Drawing showing the case 2 edge impact (oblique view of the article)
Figure 4.2-6: Drawing showing the case 2 edge impact (side view of the article)

Figure 4.2-7: Drawing showing the case 3 point impact (shot line view of the article)
Figure 4.2-8: Drawing showing the case 3 point impact (oblique view of the article)

Figure 4.2-9: Drawing showing the case 3 point impact (side view of the article)
Figure 4.2-10: Oblique view of Case 3 modified.

Figure 4.2-11: Target and aim points for Case 3 modified.
4.2.1.1 Instrumentation and high speed video setup

4.2.1.1.1 Sensor placement

Figures 4.2-12 through 4.2-16 show the instrumentation layout for the leading edge test article panel 6L, 6/7 T-seal and fiberglass panel 7L [12]. A total of 199 sensor channels were installed on the leading edge test article. The sensors included a total of 37 accelerometers. Bi-axial and tri-axial sensors were comprised of two to three Endevco 7264B-500T accelerometers. These accelerometers are capable of measuring vibrations up to 3,000 Hz and have a 500 g range. One tri-axial accelerometer and two bi-axial accelerometers were attached at the locations labeled T and B (7 sensors). They were bonded to the inner surface of panel 6L along the stagnation point. Each spar fitting had a tri-axial sensor attached to it (total of 6 fittings, 18 sensors) as did the lower carrier panels for 6L and 7L (6 sensors). In addition, the strongback had two tri-axial sensors attached at the forward and aft, left and right sides.

Five rosette strain gages with a response of up to 25,000 Hz were bonded to the inside skin only of the panel 6 (fiberglass and RCC). Two rosette strain gages were installed back to back on the panel 6L lower flange. Furthermore, a total of 12 rosette strain gages were installed back to back on the panel 6L inboard and outboard ribs. In addition, the panel 6/7 T-seal had a total of 4 rosette strain gages installed on the outboard side of the rib.

Instrumentation technicians installed four deflection gages to the panel 6L exterior surface along the stagnation line. The locations of the gages are noted in the figures as Dx and Dz. The gages had a response of up to 500 Hz. Two Capacitec sensors
were used to measure the spacing between the panel flange and the carrier panel. The sensor had a 500 mil range to a 10 mil accuracy. The frequency response is 3.1 KHz. The Capacitec sensors were used during the fiberglass panel 6L test number 1. During the test the sensors were dislodged from their mounting locations by the impact load. For the RCC panel 6L tests, Kodak motioncorders viewed the RCC flange/carrier panel gap.
**Fiberglass Panel**

**Proposed Test Panel Strain Gage Layout:**

- 5 x S = Rosette strain gages bonded to interior only on lower section of exposed RCC panel
- 2 x S = Rosette strain gages bonded to lower flange (back to back)
- 2 x Dx = Deflection gages mounted to forward face of lower flange (for longitudinal deflections of RCC)
- 2 x Dz = Deflection gages mounted to leading edge (for z deflections of RCC)
- 1 x T = Tri-axial accelerometer
  - Accels mounted on inside surface of panel, 22.0 (1XZ), 22.5 (2XYZ), 23.0 (3XZ) inches up from inside surface of aft rib
- 2 x B = Bi-axial accelerometer

---

**RCC Panel**

**Proposed Test Panel Strain Gage Layout:**

- 5 x S = Rosette strain gages bonded to interior only on lower section of exposed RCC panel
- 2 x S = Rosette strain gages bonded to lower flange (back to back)
- 2 x Dx = Deflection gages mounted to forward face of lower flange (for longitudinal deflections of RCC)
- 2 x Dz = Deflection gages mounted to leading edge (for z deflections of RCC)
- 1 x T = Tri-axial accelerometer
  - Accels mounted on inside surface of panel, 22.0 (1XZ), 22.5 (2XYZ), 23.0 (3XZ) inches up from inside surface of aft rib
- 2 x B = Bi-axial accelerometer

---

**Figure 4.2-12:** Schematic of instrumentation locations on panel 6 fiberglass (upper) and RCC (lower)
Fiberglass Panel

Proposed Test Panel Strain Gage Layout:

1 x S = Rosette strain gage bonded on inside of outbnd rib just below stagnation point
2 x S = Rosette strain gages bonded back to back on inbnd rib just below stagnation point
2 x S = Rosette strain gages bonded back to back on outbnd rib just forward of lug attach
2 x S = Rosette strain gages bonded back to back on inbnd rib just below upper lug attach
2 x S = Rosette strain gages bonded back to back on outbnd rib just below upper lug attach

RCC

Proposed Test Panel Strain Gage Layout:

1 x S = Rosette strain gage bonded on inside of outbnd rib just below stagnation point
2 x S = Rosette strain gages bonded back to back on inbnd rib just below stagnation point
2 x S = Rosette strain gages bonded back to back on outbnd rib just forward of lug attach
2 x S = Rosette strain gages bonded back to back on inbnd rib just below upper lug attach
2 x S = Rosette strain gages bonded back to back on outbnd rib just below upper lug attach

Figure 4.2-13: Schematic of instrumentation locations on panel 6L ribs fiberglass (upper) and RCC (lower)
Fiberglass Panel

**Proposed Test Panel Strain Gage Layout:**

$4 \times S$ = Rosette strain gages bonded on the outbd side of the rib only (not back to back)

Note: T-seal 7 is the T-seal attached to the outbd side of panel 6 and slips on the inbd side of panel 7

**RCC panel**

**Proposed Test Panel Strain Gage Layout:**

$4 \times S$ = Rosette strain gages bonded on the outbd side of the rib only (not back to back)

Note: T-seal 7 is the T-seal attached to the outbd side of panel 6 and slips on the inbd side of panel 7

Figure 4.2-14: Schematic of instrumentation locations on T-seal 6/7 fiberglass (upper) and RCC (lower)
Fiberglass Panel

Proposed Test Panel Strain Gage Layout:
This panel 7 instrumentation is for the impact on panel 6
2 x S = Rosette strain gages bonded to interior only on lower section of exposed RCC panel
2 x S = Rosette strain gages bonded to lower flange (back to back)

Fiberglass Panel

Proposed Test Panel Strain Gage Layout:
2 x S = Rosette strain gages bonded on Inbd rib just below stagnation point and just forward of lug attach (lower rib)
2 x S = Rosette strain gages bonded on Outbd rib just below stagnation point and just forward of lug attach (lower rib)

These rosettes are not back to back.
Dimensions for gages just forward of lugs are TBD

Figure 4.2-15: Schematic of instrumentation locations on fiberglass panel 7
Proposed Test Panel Strain Gage Layout:

This panel 8 instrumentation is for the impact on fiberglass panel 6

1 x S = Axial strain gage bonded on inside of outbd rib just below stagnation point
1 x S = Axial strain gage bonded on inside of inbd rib just below stagnation point

Figure 4.2-16: Schematic of instrumentation locations on fiberglass panel 8 (for Fiberglass panel 6, tests 2 and 3)

4.2.1.1.2 High speed video camera configuration

For the LESS panel 6L tests a total of up to 13 high speed cameras enabled the measurement of projectile launch and target impact. Figure 4.2-17 shows the camera layout for the panel 6L tests while Table 4.2-3 documents the camera type and settings used for the tests. Cameras L1 and L13 (Figures 4.2-18) imaged a close-up of the target impact area from angles that allow imaging of the front and underside of the projectile respectively. Phantom 7 cameras were selected for this view to enable the largest viewing area at the highest possible resolution and fastest frame rate. This allowed for careful measurement of the initial impact point. Lower resolution cameras were used for the remaining views. Views L2 and L3 (Figure 4.2-19) imaged the projectile from the side of its flight path. The L2 view captured images of the projectile just after leaving the barrel until it just reached the target, while the L3 view imaged the projectile motion across the target to the ejecta catcher. The L3 camera images were primarily used for
determination the projectile velocity. Figure 4.2-20 shows typical images from views L4 and L5. These oblique cameras imaged the projectile flight and target pre and post impact. The cameras were setup orthogonal to each other so as to make the measurement error equal in each direction. Views L4 and L5 enabled a multi-camera analysis of the projectile motion and any ejecta produced by its impact with the target. The overhead camera L6 (Figure 4.2-21) was used in combination with L2 and L3 to measure the projectile pitch and yaw just prior to impact. It also provided a redundant view for the measurement of projectile velocity.

Figure 4.1-17: Camera layout for LESS panel 6L impact tests. (a) Schematic of the external camera layout for all tests. (b) Schematic of the internal camera layout for panel 6L tests.
## Table 4.2-3
Camera settings for LESS panel 6L tests

<table>
<thead>
<tr>
<th>Camera Location</th>
<th>Camera Type</th>
<th>Lens Focal Length (mm)</th>
<th>Image Format (WxH)</th>
<th>Frame Rate (fps)</th>
<th>Scene Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>L1</td>
<td>Phantom 7</td>
<td>50</td>
<td>512x512</td>
<td>8000</td>
<td>Close-up of impact zone from ahead</td>
</tr>
<tr>
<td>L2</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile prior to impact</td>
</tr>
<tr>
<td>L3</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile after impact</td>
</tr>
<tr>
<td>L4</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile and debris</td>
</tr>
<tr>
<td>L5</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile and debris</td>
</tr>
<tr>
<td>L6</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Overhead close-up of impact zone</td>
</tr>
<tr>
<td>L7</td>
<td>Phantom 4</td>
<td>60</td>
<td>512x512^1 256x256^2 256x512^3,4</td>
<td>3802^1 3704^2 1900^3,4</td>
<td>Close-up of panel 6 alignment pin</td>
</tr>
<tr>
<td>L8</td>
<td>Phantom 5</td>
<td>14</td>
<td>512x1024</td>
<td>1900</td>
<td>Panel 6 interior surface</td>
</tr>
<tr>
<td>L9</td>
<td>Motioncorder</td>
<td>25</td>
<td>256x240</td>
<td>500^4</td>
<td>Gap between panel 6 and carrier panel</td>
</tr>
<tr>
<td>L10</td>
<td>MotionCorder</td>
<td>6</td>
<td>256x240</td>
<td>500^4</td>
<td>Gap between panel 7 and carrier panel</td>
</tr>
<tr>
<td>L11</td>
<td>Phantom 5</td>
<td>15</td>
<td>512x1024</td>
<td>1900</td>
<td>Panel 6 interior surface</td>
</tr>
<tr>
<td>L12</td>
<td>Phantom 7</td>
<td>20</td>
<td>800x600 256x512^2</td>
<td>4800 1901^2</td>
<td>Panel 6 outboard rib</td>
</tr>
<tr>
<td>L12</td>
<td>Phantom 4^2</td>
<td>25^2</td>
<td>512x512^1 256x512^2 256x256^4</td>
<td>1000 15873^2 2141^4</td>
<td>Close-up of impact zone from below</td>
</tr>
</tbody>
</table>

* Only cameras L1 through L7 were used for fiberglass panel 6L, test 1

1 Setting for fiberglass panel 6L test 1 only
2 Setting for RCC panel 6L test only
3 Setting for fiberglass panel 6L test 2 only
4 Setting for fiberglass panel 6L test 3 only
Figure 4.2-18: Camera views from L1 (a) and L13 (b) locations. A close up of the foam impact area.

Figure 4.2-19: Camera views from L2 (a) and L3 (b) locations. These views show the foam trajectory from the gun muzzle through the impact area. The views are orthogonal to the trajectory and allow for measurement of the projectile velocity.
Figure 4.2-20: Typical images from views L4 (a) and L5 (b). These views enabled a multi-camera analysis of the projectile motion and any ejecta produced by its impact with the target.

Figure 4.2-21: Camera view from the overhead camera L6. It was used in combination with L2 and L3 to measure the projectile pitch and yaw just prior to impact.
Analyses of panel 6L fiberglass and RCC impacts were conducted prior to any testing. Of concern was that sufficient deflections could be achieved in the panel that could result in deformation or failure of the panel attach hardware. In particular, the analyses showed that the lower panel 6L shear pin would likely fail due to the impact event. Since the pin could not be instrumented without affecting the test result, a camera was attached to the debris shield and focused on the shear pin and fitting. Figure 4.2-22 shows a typical view from the L7 camera. During the first fiberglass test exterior views showed a great deal of motion induced in the target. As a result five additional cameras were attached to the spar inside the LESS. Section 2.2.7.2 discusses the details of the interior camera installations. Cameras L8 and L11 were added to measure the deflection of the interior panel surface (Figure 4.2-23). The cameras were installed orthogonal to one another to balance the measurement errors. Measurement of the gap between the panel lower flange and carrier panel was desired. Unfortunately, the gap sensors designed for this purpose did not work properly (as demonstrated in the first fiberglass panel 6L test). Cameras L9 and L10 were added for all subsequent panel tests to replace the gap sensors. Figure 4.2-24 shows typical views from the L9 and L10 cameras. Finally, measured strains on the first fiberglass impact test were sufficiently high that analysts expected the RCC panel rib could fail. Camera L12 (Figure 4.2-25) was added to view the lock side (outboard) rib in the expectation the failure would be observed during the test. In fact, during the panel 6L test the failure was imaged quite nicely. The results of this imagery will be discussed later.
Figure 4.2-22: Camera view L7 showing the dynamic motion of the panel 6L lower shear pin

Figure 4.2-23: Camera views L8 and L11. The cameras have oblique views of the target grid on the inside surface of panel 6L enabling three-dimensional dynamic analysis of the deformation of the target panel.
Figure 4.2-24: Camera views L9 and L10. These interior views allow the measurement of the gap between the lower flange of panels 6 and 7 and their carrier panels.
Figure 4.2-25: L12 camera view of the panel 6L lock side (outboard) rib.
4.2.1.2 Results of LESS panel 6L impact testing

4.2.1.2.1 Results of LESS fiberglass panel 6L test 1 (FG6-1)

The first LESS impact test was conducted on fiberglass panel 6L (FG6-1) on May 29, 2003. A fiberglass panel of similar geometry as the RCC panel was used to verify aiming accuracy and check out instrumentation performance. Due to the difference in material properties and manufacturing lay-up, the fiberglass panel was more flexible and stronger than its RCC counterpart. A BX 250 foam projectile of dimension 21.5"x11.5"x5.6" and mass 1.68 pounds was launched to an impact velocity of 784 ft/sec. The foam projectile impacted the fiberglass panel within 0.5 inches of the target. Ink lines on the foam projectile left a long black streak or skid mark across panel 6L, its T-seal and panel 7L as shown in Figure 4.2-26. Test FG6-1 was conducted successfully since the impact velocity was within the desired tolerance, the projectile impacted the target within the 1" targeting requirement and there was no measurable pitch or yaw of the projectile. All instrumentation data were successfully collected. Fiberglass panel 6L and its T-seal each showed signs of damage post test. A significant amount of foam was wedged in between panel 6L and its T-seal as shown in Figure 4.2-27. No holes or cracks were readily apparent to the naked eye immediately following the test although NDE using a handheld ultrasonic probe identified a region of delamination just below the panel apex. The delamination region is the oval shaped marked area in the upper, left corner of Figure 4.2-28. The panel and T-seal assembly was removed from the test article. The T-seal was removed from the panel to extract the foam. Upon re-assembly of the T-seal to the panel a permanent deformation or set in the panel was observed. This is shown clearly in Figure 4.2-28.
Significant witness marks were found on the panel 6L shear fittings. The shear pins that went into the shear fittings appeared to be slightly bent with gouges on the contact surface. These observations suggested that very high loads had been imparted to the fittings. It was not possible to determine if the spar fittings had been damaged from the impact although they were dimensionally consistent with their pre-test condition. This damage gave rise to the concern that the attach hardware and spar fittings may not be able to survive multiple impact tests—a real concern since spares of anything other than the shear pins were unavailable.

The principal strains for panel 6L are shown in Figure 4.2-29. Relatively high strains were recorded not only on panel 6L, but on the adjacent fiberglass panel 7L as well. The high in-plane compression stresses achieved in panel 7L indicated that it was providing a significant boundary condition during the foam loading of panel 6L. Since stress strain curves for the fiberglass panels were not available, it was not possible to precisely determine the strength margin remaining in the fiberglass panels; however, if the minimum properties for fiberglass shown in Table 4.2-4 were considered, the structural margins appeared to be relatively low. In order to conduct a representative test on an RCC panel 6L it was vital to retain an undamaged fiberglass panel 7L. Although no damage to the panel was observed either visually or by NDE, the continued structural integrity of the panel remained in question. Given the potential for structural damage to the test article itself (panels and attach hardware), it was recommended by the test team and concurred by the NAIT and CAIB to proceed with the panel 6L RCC test using the Case 1 conditions.
Figure 4.2-26: Photograph showing skid mark and foam immediately following test FG6-1.

Figure 4.2-27: Photograph of foam wedged under T-seal after test FG6-1.
Figure 4.2-28: Photograph of fiberglass panel 6 showing permanent deformation after FG6-1.

Figure 4.2-29: Schematic of panel 6L and T-seal summarizing the principal strains achieved during test FG6-1 [16].

Legend: Strains are in με, positive values are ε1, negative values are ε2, values in 0 are for gage behind location shown.
Table 4.2-4
Material Properties for RCC and Fiberglass [16]

<table>
<thead>
<tr>
<th>PROPERTY</th>
<th>1581 Fiberglass (2) - PSI</th>
<th>RCC (3) - PSI</th>
</tr>
</thead>
<tbody>
<tr>
<td>E_{l}</td>
<td>3.00 \times 10^6</td>
<td>2.10 \times 10^6</td>
</tr>
<tr>
<td>E_{c}</td>
<td>3.40 \times 10^6</td>
<td>4.50 \times 10^6</td>
</tr>
<tr>
<td>E_{b}</td>
<td>3.12 \times 10^6</td>
<td>2.30 \times 10^6</td>
</tr>
<tr>
<td>G</td>
<td>1.05 \times 10^6</td>
<td>1.05 \times 10^6</td>
</tr>
<tr>
<td>F_{t u}</td>
<td>45000</td>
<td>29700</td>
</tr>
<tr>
<td>F_{c u}</td>
<td>55000</td>
<td>36300</td>
</tr>
<tr>
<td>F_{b u}</td>
<td>36800</td>
<td>24288</td>
</tr>
<tr>
<td>F_{s u}</td>
<td>21120</td>
<td>21120</td>
</tr>
<tr>
<td>F_{b r g} (1)</td>
<td>35500</td>
<td>35500</td>
</tr>
<tr>
<td>\mu</td>
<td>0.15</td>
<td>0.15</td>
</tr>
</tbody>
</table>

(1) Ultimate Bearing Allowable for Lug
(2) Reference (d), Minimum Properties
(3) Reference (c), "As Fab", "A" Values At Room Temperature

This chart Updates the RCC Material Properties to the Loral Vought Systems 221RP00614 "Leading Edge Structural Subsystem Mechanical Design Allowables for Material with Improved Coating System"

4.2.1.2.2 Results of LESS RCC panel 6L (RCC6)

The first impact test on an RCC panel, and the second test in the LESS test series, was conducted on June 6, 2003. A 5.63"x11.56"x21.38" piece of BX 250 foam with a mass of 1.68 pounds was launched at RCC panel 6L. The aim point was Case 1 as shown in Figures 4.2-1 through 4.2-3 (the same used in the initial fiberglass test). This aim point would yield an impact angle of 20.6 degrees with respect to the local surface normal. The foam projectile impacted the panel at 768 ft/s. The projectile struck inside the 1" diameter target circle at approximately 0.25" below center vertically and on center
horizontally. The projectile exhibited no measurable pitch or yaw. The ink lines on the foam projectile left a long black streak or skid mark across panel 6L, its T-seal and panel 7L as shown in Figure 4.2-30.

![Image](image_url)

**Figure 4.2-30: Photograph of the ink streak on panels 6L and 7L.**

Similar to fiberglass panel 6L test 1, pieces of foam were lodged between panel 6L and its T-seal although to a lesser extent. Figure 4.2-31 shows the lodged foam. The likely reason for the smaller amount of foam in this test is that the RCC panels are stiffer than fiberglass and consequently the lock side rib should deflect less. Since there is less step or in-plane offset between the panel and the T-seal there is less foam wedged in the gap between the two parts.

Step and gap measurements were taken post-test and compared with pre-test measurements. Figures 4.2-32 through 4.2-35 show the step and gap results for panel 6L.
and its T-seal on both the lock and slip sides [23]. The measurements show a maximum step and gap on the lock side of 0.030" and 0.020" respectively. These maxima were measured at a station on the lower surface within only a couple of inches of the lower flange. Likewise, on the slip side of the interface the maximum step and gap measured 0.040" and 0.030" respectively. This time the maxima were measured within 4" of the apex of the panel. Of note is the fact that the gap at a distance 8" from the apex measured 0.012". This is, as will be documented later, next to a crack identified in the T-seal.

Although the panel initially appeared undamaged, closer visual inspection revealed a crack. The crack started on the lock side rib of the panel and traversed its entire width as shown in Figure 4.2-36. The crack continued up onto the lock side channel of the panel underneath the T-seal as shown in Figure 4.2-37 and terminated on the panel facesheet about 0.75 inches forward of the T-seal as shown in Figure 4.2-38. The total length of the crack was approximately 5.5 inches. NDE indicated that the crack was through thickness. Figure 4.2-39 shows film radiographs of the rib showing crack indications on both surfaces of the rib thereby proving that it is through thickness. In addition, the L12 view captured the crack propagation event and clearly shows the nature of the crack (Figure 4.2-40).

The panel crack occurred relatively close to the RCC6R-16 strain gage on the lock side rib. Figure 4.2-41 shows the principal strains from the gage. The state of strain at the gage location is almost pure tension with a small amount of in-plane shear present. The data showed the gage failed at a time near 1.8 msec. This strain measurement provides not only a strain to failure that may be used for numerical analysis, but it also provides a time when the failure occurred which will prove valuable for code validation.
activities. Additional summary of principal strains for panel 6 and its T-seal are shown in Figure 4.2-42.

Further inspection of the test article showed that the T-seal rib was also cracked. The location of the T-seal rib crack was very close to the panel rib crack down the lower surface 6 inches from the apex (Figure 4.2-43). The crack was approximately 2.5 inches long, essentially all the way through the T-seal rib and into the T-seal flange radii. The crack actually propagated through one of the strain gages on the T-seal rib. Principal strains calculated for this gage are shown in Figure 4.2-44. Again, the state of strain is predominately tension, with some in-plane shear present. The crack occurs at approximately 2.4 msec after foam impact, as the foam passes over the upper surface of the T-seal. NDE using film radiography showed that the crack was through thickness.

Two additional damages were noted on the panel. A panel crack was found on the upper outboard flange of the panel just above the upper shear fitting. The crack was measured to be approximately 0.375 inches long and was not through thickness (surface coating crack). Figure 4.2-45 shows the crack. Subsequent review of work logs for the panel showed that this area had undergone repair in the past. Additionally, tile damage was found on upper carrier panel 6 as shown in Figure 4.2-46. The damaged area was one that had also been previously repaired. The repair was dislodged as well as some adjacent coating. The entire damaged area measured approximately 1 inch x 0.75 inches.
Figure 4.2-31: Photograph of foam wedged under T-Seal after RCC6 test.

Figure 4.2-32: Step measurements for the panel 6L/T-seal lock side interface. Results are shown as pre-test measurement/post-test measurement. The difference between these two numbers indicates the change in step.
Figure 4.2-33: Gap measurements for the panel 6L/T-seal lock side interface. Results are shown as pre-test measurement/post-test measurement. The difference between these two numbers indicates the change in gap.

Figure 4.2-34: Step measurements for the panel 6L/T-seal slip side interface. Results are shown as pre-test measurement/post-test measurement. The difference between these two numbers indicates the change in step.
Figure 4.2-35: Gap measurements for the panel 6L/T-seal slip side interface. Results are shown as pre-test measurement/post-test measurement. The difference between these two numbers indicates the change in gap.

Figure 4.2-36: Photograph of the panel 6L rib crack. The scale is in centimeters.
Figure 4.2-37: Photograph of the panel 6L rib crack showing its progress across the lock channel and onto the facesheet. The scale is in centimeters.

Figure 4.2-38: Photograph of the panel 6L rib crack showing its termination on the facesheet.
Figure 4.2-39: Radiograph images of the outboard RCC panel 6L rib, lock and front of panel. The two lines in the images show the feature is a through crack [24].

Figure 4.2-40: Images from the L12 high speed video camera showing the Panel 6L outboard rib crack.
Figure 4.2-41: Plot of principal strains for gage RCC6R-16 near RCC panel 6 rib crack. Note the gage is a rosette and the plot shows maximum and minimum strains.

Figure 4.2-42: Schematic of panel 6L and T-seal summarizing the principal strains achieved during the test [16].
Figure 4.2-43: Photograph of the RCC panel 6 lock side rib and T-Seal showing cracks in each.

Figure 4.2-44: Plot of principal strains near panel 6L T-seal crack. The TS6R-20 strain gage is a rosette and the plot shows maximum and minimum strain. Note the erratic behavior of the strain measurement between times 0.138 and 0.139 seconds as the gage fails.
Figure 4.2-45: Photo of the RCC panel 6L upper shear fitting area crack.

Figure 4.2-46: Photograph of the panel 6L upper carrier panel tile damage.
Load cells were mounted under the multi-panel test fixture in order to compare the relative magnitude of the load induced by each impact. Installing the load cells closer to the RCC panels, say between the panel hardware and spar attach fittings, would have provided a more accurate reading of the impact force but would have placed the test article out of the LESS flight configuration. It was determined that maintaining flight configuration was more important for the purposes of the test than measuring the loading precisely. The compromised load cell installation was the result.

The loads measured during the RCC panel 6L test were very similar to the loads measured during fiberglass 6L test 1. This confirmed the hypothesis in section 3.2.1 that the low foam stiffness would make the impact loads comparable. The root sum squared (RSS) loads from both tests are shown in Figure 4.2-47. The peak load in this figure occurs at approximately 0.147 seconds (about 11 msec after impact) and the duration of the pulse is about 10 msec. The reaction of the load cells is spread out in time and less than expected in amplitude due to the inertial effects of the 3300 pound test fixture.
Figure 4.2-47: Plot of load cell data for panel 6L fiberglass test 1 and RCC panel 6L test. Loads are plotted as the root sum squared of the x, y and z measured loads at the base of the test fixture for FG6-1 and RCC6.

4.2.1.2.3 Results of LESS fiberglass panel 6L test 2 (FG6-2)

Although the results of the RCC panel 6L test showed damage to the lower surface and rib, the damage was not substantial enough to be consistent with the estimated panel hole size (10 inch diameter) incurred during the STS-107 mission. The test team decided to initiate additional tests on the previously impacted fiberglass panel 6L in order to better understand the relationships between impact location, foam orientation and failure mode. Additionally the tests were needed to evaluate and improve modeling of the fiberglass panels. These tests were conducted prior to conducting the tests on the panel 8L fiberglass or RCC so the final RCC impact test would yield damage as close as possible to that incurred during the STS-107 ascent. For the FG-2 and 3 tests,
several strain gages were added to the fiberglass panels. Three strain gages were added to fiberglass panel 8L and one gage each were added to the panel 6L outboard rib and the panel 7 inboard rib back to back with gages 23 and 10 respectively. Figure 4.2-13 shows the panel 6L ribs while section 4.2.2.1.1 details the layout of the gages added to panel 8.

The aim point for the second fiberglass test was approximately 3 inches further aft from the FG6-1 impact along a line from the panel apex to the trailing edge (parallel to the inboard T-seal interface). This aiming scenario is called “case 2” in section 4.2.1 and Figures 4.2-7 through 4.2-9 show its details. This aim point was selected to try and impart more load to the panel surface. It was anticipated that this would yield a more STS-107 like failure of the panel.

The second impact test on fiberglass panel 6L (third LESS test), was conducted on June 16, 2003. A BX 250 foam projectile of dimensions 5.63”x11.56”x23.0” and mass of 1.69 pounds was launched to an impact velocity of 774 ft/s. The projectile struck the panel within 0.6 inches of the intended target. Ink lines on the foam projectile left a long black streak or skid mark across panel 6L, its T-seal and panel 7 (Figure 4.2-48). Even more foam was wedged underneath the panel 6L T-seal than in the first fiberglass 6 test (Figure 4.2-49) and can be explained as follows. First, the deflection of the panel surface increased from 1.5 inches in FG6-1 to 2.1 inches, exposing more foam area to the T-seal. Second, the small permanent set in the panel from the initial test had increased the step between the panel and the T-seal. Both of these increases in step exposed more of the T-seal edge to the projectile and enabled the contacting foam edge to shear off and wedge into the exposed panel/T-seal gap. Immediately post-test the maximum lock side panel 6L step and gap measured 0.625 inches and 0.078 inches respectively. After
disassembly of the parts and removal of the foam, the permanent deformation of the panels allowed for a 0.370 inch step and a 0.136 inch gap. These gaps were substantial (certainly not within flight specifications) and would tend to yield a non-representative LESS response if additional test were conducted in this configuration. The test team decided to repair the T-seal by mechanically deforming the part until the fit was more flight like.

Figure 4.2-50 shows a crack on the panel 6L facesheet. The crack was approximately 12” below the apex of the panel and measured approximately one inch in length. NDE using a one-side ultrasonic probe showed no indications except in the area where surface damage was visible. Furthermore, the crack was not a through crack, but there was an indication of delamination on and about the crack. Principal strains recorded on panel 6 during the test are shown in Figure 4.2-51. Overall strains in the shell surface for the FG6-2 test were approximately 23% higher than those recorded for FG-1.

Figure 4.2-48: Photograph of the panel 6L area showing the impact skid mark incurred during test FG6-2.
Figure 4.2-49: Photograph of foam wedged under the T-seal after test FG6-2.

Figure 4.2-50: Photograph showing a close up view of the fiberglass panel 6L facesheet crack after FG6-2.
Figure 4.2-51: Schematic of panel 6L and T-seal summarizing the principal strains achieved during the test FG6-2 [16].

4.2.1.2.4 Results of LESS fiberglass panel 6L test 3 (FG6-3)

The purpose of the third fiberglass panel 6L (FG6-3) test was to further understand the impact response of the leading edge structure with the ultimate goal of producing damage on the final RCC impact test similar to that incurred during STS-107 ascent. Results of the previous two fiberglass tests showed that the face sheet strains and panel deflections both increased as the aim point was moved further aft on the panel. Furthermore, there was increased damage of the panel and more displacement of the T-seal with this change in impact point. One would expect a greater opportunity for material failure as strains and deflections increase.

For the third test on fiberglass panel 6 the impact point remained the same as for FG6-2 but the gun clocking angle was changed to allow the long leading edge of the
foam to impact the panel first instead of the corner as in previous tests. This foam orientation shown in Figures 4.2-10 and 11 was identified by pre-test analysis as generating the higher facesheet stresses than either corner or short edge impacts [22].

The third impact test on fiberglass panel 6L and the fourth LESS test was conducted on June 18, 2003. A BX 250 foam projectile of mass 1.68 pounds and dimension 5.63”x11.56”x22.5” was launched to a velocity of 782 ft/s. The foam projectile impacted the fiberglass panel within 0.25” of the intended target. Impact at this point resulted in a local impact angle of 20.6 degrees with respect to the surface normal. Ink marks on the foam projectile left a long black streak or skid mark starting on T-seal 5, continuing across panel 6, T-seal 6 and onto panel 7 as shown in Figure 4.2-52. The ink mark on T-seal 5 was attributed to the aft end of the foam projectile slapping down after initial panel contact.

Repair of the panel 6L T-seal yielded a pre-test lock side step and gap maximum measurement both of 0.150”. These measurements were outside flight specifications, but were the best that could be achieved given the condition of the parts. During the test, once again debris from the foam projectile was caught between panel 6L and its T-seal as shown in Figure 4.2-53. Figure 4.2-54 shows that upon disassembly of the T-seal and panel it is very clear a large amount of foam was lodged in the gap. The T-seal was pushed up and aft 5/8 of an inch. The T-seal was actually pushed up on top of panel 7 as shown in Figure 4.2-55. The maximum deflection of the panel facesheet was approximately 3.2 inches for the third fiberglass test—over twice the deflection measured for FG6-1. It was recognized that some portion of this increased FG6-3 deflection could
be attributed to damage from previous tests that had probably reduced the overall panel stiffness.

Multiple cracks were produced on fiberglass panel 6 during this test. The crack that formed during the FG6-2 test (approximately 12” below the apex) propagated to 1.4” total length as a result of test 3 loading. A new crack was produced during Test 3 that runs parallel to the lock side rib on the panel. The new crack is 2.2” long and intersects the previous crack as shown in Figure 4.2-56. A second new crack formed on the upper lock side of the panel approximately 4.5-inch below the apex point (Figure 4.2-57, labeled new crack 1). This through crack measured 1.78” across the face and through the lock side on the outer surface and 1.5” on the internal surface of the panel. Another new crack was formed on the lower panel surface and lock step (Figure 4.2-58, labeled new crack 2). It is 0.43” long and is not a through crack. The most substantial crack formed during this test is a through crack approximately eight inches long formed on the lower outboard rib of the panel as shown in Figure 4.2-59 (exterior view) and Figure 4.2-60 (interior view).

The instrumentation suite for this test remained the same as that for test FG6-2. Principal strains recorded on panel 6L during the test are shown in Figure 4.2-61. The facesheet strain levels were the highest recorded in the test series—approximately 79% greater than those measured for FG6-1. These levels were high enough to suggest that had the RCC6 test been conducted with the same impact location the damage would have been more severe in the lower facesheet area. Further shots at fiberglass panel 6 were not proposed because of concerns that the observed damaged would distort the results of any
additional tests. Since the fiberglass panel 6L was so substantially damaged, the test team decided to proceed to the fiberglass 8L test program.

Figure 4.2-52: Photograph of fiberglass panel 6L and its neighboring parts after test FG6-3. The black skid mark extends from the panel 5L T-seal and across to panel 7L.
Figure 4.2-53: Photograph of panels after test FG6-3 showing foam wedged under the T-seal.

Figure 4.2-54: Photograph of panel 6L and T-seal post-test. The T-seal was removed from the panel to show the large amount of foam material that was wedged between the two parts.
Figure 4.2-55: Photograph of the panel 6L T-seal post test FG6-3. Notice the slip side of the seal is riding up onto panel 7L by 0.625 inches.

Figure 4.2-56: Photograph of extended existing and new cracks on fiberglass panel 6L after test 3
Figure 4.2-57: Photograph of fiberglass panel 6L post test showing close up views of new crack 1 and its location relative to the existing crack.

Figure 4.2-58: Photograph of fiberglass panel 6L post test showing close up views of new crack 2 and its location relative to the existing crack.
Figure 4.2-59: External view photograph of the panel 6L rib crack after test FG6-3.

Figure 4.2-60: Interior view photograph of the panel 6L rib crack after test FG6-3.
Figure 4.2-61: Schematic of panel 6L and T-seal summarizing the principal strains achieved during the test FG6-3 [16].

4.2.1.3 Discussion of panel 6L impact testing

Impact testing of the fiberglass and RCC panel 6L parts yielded valuable insights for a successful test of RCC panel 8L. The details of the projectile size and velocity for each test are listed in Table 4.2-5. Foremost, testing showed that RCC panels could be failed due to high velocity impact of ET TPS foam. It also demonstrated that the achieved loads could plastically deform panel attach hardware and fittings. This was identified as a concern for the successful future impact testing due to limited spares availability. Load cell measurements showed that impact loads were comparable for a fiberglass or RCC impact test (all other variables the same); therefore, fiberglass panels of the same geometry as RCC panels yield insight into RCC response.

Strain measurements proved that one downrange panel from the target is needed for impact testing to provide a response representative of flight configuration. That is,
the next downrange panel provides a significant boundary condition and must be included (at a minimum) for future impact testing. Most importantly, these data showed that by moving the impact point downrange from that chosen in FG6-1 the resulting panel strains were increased by 23%. Additionally, changing the impact from a corner to a leading edge impact further increased the panel strains by 79%. These trends suggested that panel 8L should be tested by initially impacting the panel near the center span and with a leading edge rather than a projectile corner.

<table>
<thead>
<tr>
<th>Test Number</th>
<th>Projectile Mass (lb)</th>
<th>Projectile dimensions (inches)</th>
<th>Test case</th>
<th>Impact Velocity (ft/s)</th>
<th>Clocking angle (degrees)</th>
</tr>
</thead>
<tbody>
<tr>
<td>FG6-1</td>
<td>1.68</td>
<td>5.63x11.5x21.5</td>
<td>Case 1</td>
<td>784</td>
<td>0</td>
</tr>
<tr>
<td>RCC6</td>
<td>1.68</td>
<td>5.63x11.56x21.38</td>
<td>Case 1</td>
<td>768</td>
<td>0</td>
</tr>
<tr>
<td>FG6-2</td>
<td>1.69</td>
<td>5.63x11.56x23.0</td>
<td>Case 3</td>
<td>775</td>
<td>0</td>
</tr>
<tr>
<td>FG6-3</td>
<td>1.68</td>
<td>5.63x11.56x22.5</td>
<td>Case 3 (modified)</td>
<td>782</td>
<td>30</td>
</tr>
</tbody>
</table>

* Desired impact velocity was 775 ft/s for all tests.
4.2.2 LESS fiberglass and RCC panel 8L testing

Completion of the panel 6L test program enabled a better understanding of the response of leading edge panels under impact loading; however, the test team, NAIT and CAIB decided additional testing of the fiberglass panel 8L was necessary. These tests would ensure that all targeting and instrumentation issues were resolved prior to RCC test. In addition, these tests would allow for the team to better refine the impact location for the RCC test that would yield damage most representative of that incurred during STS-107 ascent.

Table 4.2-6 shows the test matrix for the panel 8L fiberglass and RCC tests. The test case, or aim point, for each test is described in Figures 4.2-62 through 4.2-65.

<table>
<thead>
<tr>
<th>Test Number</th>
<th>Projectile Volume (in³)/mass (lb)</th>
<th>Nominal Projectile dimensions (inches)</th>
<th>Test case</th>
<th>Velocity (ft/s)</th>
<th>Clocking angle (degrees)</th>
</tr>
</thead>
<tbody>
<tr>
<td>FG8-1</td>
<td>1200/1.67</td>
<td>5.5x11.5x22</td>
<td>24.1 degree, edge impact</td>
<td>775</td>
<td>30</td>
</tr>
<tr>
<td>FG8-2</td>
<td>1200/1.67</td>
<td>5.5x11.5x22</td>
<td>22.0 degree, edge impact</td>
<td>775</td>
<td>30</td>
</tr>
<tr>
<td>RCC8</td>
<td>1200/1.67</td>
<td>5.5x11.5x22</td>
<td>22.0 degree, edge impact</td>
<td>775</td>
<td>30</td>
</tr>
</tbody>
</table>
Figure 4.2-62: Drawing showing the 24.1 degree impact test case used in test FG8-1 (oblique view of the article).

Figure 4.2-63: Drawing showing the target and aim points for the 24.1 degree impact test case used in test FG8-1.
Figure 4.2-64: Drawing showing the 22.0 degree impact test case used in tests FG8-2 and RCC8 (oblique view of the article).

Figure 4.2-65: Drawing showing the target and aim points for the 22.0 degree impact test case used in tests FG8-2 and RCC8.
4.2.2.1 Instrumentation and high speed video setup

4.2.2.1.1 Sensor placement

Section 4.2.1.1.1 describes the sensor placement for the panel 6L test series. The load cell, spar fitting accelerometer and leading edge spar strain gage layout are identical for the panel 8L test series. Figures 4.2-66 through 4.2-73 show the panel strain gage, deflection gage and accelerometer layout for the panel 8L test program. The manufacturers and models for all gages are identical to those used in the panel 6L test program. During testing, however, a subset of the installed gages were actually monitored. Data was acquired on these sensors as follows:

Panels 5, 6, 7 and 6 and 7 T-seals: none
Panel 8 and T-seal 8: all accelerometers, deflection gages and strain gages
Lower carrier panel 8: none
Panel 9: all except 8, 9, 16 and 17
Lower carrier panel 9: all strain gages and accelerometers
T-seal 9: 19 and 20 only
Panel 10: 1, 6, 8, 5 (longitudinal) and 7 (longitudinal) only
Lower carrier panel 10: none
Spar fittings: tri-axial accelerometers 8, 14, 15, 11, 17 and 18
Stongback load cells: all
Leading edge spar: no strain gages
**Proposed Test Panel Strain Gage Layout:**

1. 6 x S = Rosette strain gages bonded back to back on inboard rib
2. 4 x S = Rosette strain gages bonded back to back on outboard upper and lower rib near the attach points
3. 2 x S = Uni-axial strain gages bonded on inboard rib at the lower stagnation point next to back to back rosettes
   - These uni-axial should be as close to the edge as possible
4. 1 x S = Rosette strain gages bonded on interior of outboard rib near the lower stagnation
5. 2 x S = Uni-axial strain gages bonded on outboard rib at the lower stagnation point next to lone rosette at stagnation
   - Gages near lug attach should be as close to the edge as possible and as close to the attach as possible without getting into the transition thickness area

**Figure 4.2-66:** Schematic of instrumentation locations on panel 8L.

**Proposed Test Panel Strain Gage Layout:**

1. 5 x S = Rosette strain gages bonded on interior of lower section of exposed RCC panel
2. 2 x S = Rosette strain gages bonded anterior back to back to locations #2 and #4 (added 6/20/02)
3. 2 x D = Deflection gages mounted to leading edge (for x deflections of RCC)
4. 2 x D = Deflection gages mounted to leading edge (for z deflections of RCC)
5. 1 x T = Tri-axial accelerometer
6. 2 x S = Bi-axial accelerometer

**Figure 4.2-67:** Schematic of instrumentation locations on panel 8L ribs.
Proposed Test Panel Strain Gage Layout:

4 x S = Rosette strain gages bonded on the outboard side of the rib only (not back to back)

Gages near lug attach should be as close to the edge as possible and as close to the attach as possible without getting into the transition thickness area

Note: T-seal 9 is the T-seal attached to the outboard side of Panel 8 and slips on the inboard side of Panel 9

Figure 4.2-68: Schematic of instrumentation locations on panel 8L T-seal.

Proposed Test Panel Strain Gage Layout:

5 x S = Rosette strain gages bonded to interior only on lower section of exposed RCC panel
2 x S = Rosette strain gages bonded to lower flange (back to back)

Locations (3) and (5) are not on the measurement list for Panel 8 strike

Deflections and Accels for Panel 9 have been moved to Panel 8

Figure 4.2-69: Schematic of instrumentation locations on panel 9L.
Fiberglass Panel

**Existing Test Panel Strain Gage Layout:** (Additional in red)

(3). \(2 \times S\) = Uni-axial strain gages bonded on Inbd rib at the Lower Stagnation point next to back to back rosettes

These uni-axial should be as close to the edge as possible

(5). \(2 \times S\) = Uni-axial strain gages bonded on Outbd rib at the Lower Stagnation point next to lone rosette at stagnation

These uni-axial should be as close to the edge as possible

*Channel ids 8, (9), 10, 12, 14, 16, (17) are NOT on the measurement list for Panel 8 strike.

1 \( \times S\) = Rosette strain gage bonded on inside of outbd rib just below stagnation point

2 \( \times S\) = Rosette strain gages bonded back to back on Inbd rib just below stagnation point

2 \( \times S\) = Rosette strain gages bonded back to back on Outbd rib just forward of lug attach

2 \( \times S\) = Rosette strain gages bonded back to back on Inbd rib just forward of lug attach

2 \( \times S\) = Rosette strain gages bonded back to back on Inbd rib just below upper lug attach

2 \( \times S\) = Rosette strain gages bonded back to back on Outbd rib just below upper lug attach

**Figure 4.2-70:** Schematic of instrumentation locations on panel 9L ribs.

**Proposed Test Panel Strain Gage Layout:**

4 \( \times S\) = Rosette strain gages bonded on the outbd side of the rib only (not back to back)

*Channel ids 19, 20, 21, 22 are NOT on the measurement list for Panel 8 strike.

**NOTE:** Tseal 10 is the Tseal attached to the outbd side of panel 9 and slips on the Inbd side of panel 10

**Figure 4.2-71:** Schematic of instrumentation locations on panel 9L T-seal.
Proposed Test Panel Strain Gage Layout:

*Channel ids (1), (4) are NOT on the measurement list for Panel 8 strike.
Channel 10-(2)-1 maps to 8-4-1 (closest location) for FG6-3 test. Should we read it?
2 x S = Rosette strain gages bonded to interior only on lower section of exposed RCC panel
2 x S = Rosette strain gages bonded to lower flange (back to back)

Figure 4.2-72: Schematic of instrumentation locations on panel 10L.

*Channel ids (5), (7), are NOT on the measurement list for Panel 8 strike.
Channels 10-(6)-1 and 10-(8)-1 map to gages read on FG6 Test 3. Should we read them?

Proposed Test Panel Strain Gage Layout:

2 x S = Rosette strain gages bonded on Inbd rib just below stagnation point and just forward of lug attach (lower rib)
2 x S = Rosette strain gages bonded on Outbd rib just below stagnation point and just forward of lug attach (lower rib)

These rosettes are not back to back
Dimensions for gages just forward of lugs are given on installation drawings

Figure 4.2-73: Schematic of instrumentation locations on panel 10L ribs.
4.2.2.1.2 High speed video camera configuration

For the LESS panel 8L tests a total of up to 16 high speed cameras enabled the measurement of projectile launch and target impact. Figure 4.2-74 shows the camera layout for the panel 8L tests while Table 4.2-7 documents the camera type and settings used for the tests. The camera views and their description are the same as described in section 4.2.1.1.2 for the panel 6L tests. Three new views were added. L14 and L16 were added to image the projectile as it exited the barrel looking at the leading edge and trailing edge of the projectile respectively. View L15 was added to capture an additional view of the panel 8L downrange (lock side) rib and improve the coverage of this area of the panel.
Figure 4.2-74: Camera layout for LESS panel 8L impact tests. (a) Schematic of the external camera layout for all tests. (b) Schematic of the internal camera layout for panel 8L tests.
### Table 4.2-7
Camera settings for LESS panel 6L tests

<table>
<thead>
<tr>
<th>Camera Location</th>
<th>Camera Type</th>
<th>Lens Focal Length (mm)</th>
<th>Image Format (WxH)</th>
<th>Frame Rate (fps)</th>
<th>Scene Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>L1</td>
<td>Phantom 7</td>
<td>50</td>
<td>512x512</td>
<td>8000</td>
<td>Close-up of impact zone from ahead</td>
</tr>
<tr>
<td>L2</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile prior to impact</td>
</tr>
<tr>
<td>L3</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile after impact</td>
</tr>
<tr>
<td>L4</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile and debris</td>
</tr>
<tr>
<td>L5</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Projectile and debris</td>
</tr>
<tr>
<td>L6</td>
<td>Phantom 5</td>
<td>50</td>
<td>1024x512</td>
<td>2000</td>
<td>Overhead close-up of impact zone</td>
</tr>
<tr>
<td>L7</td>
<td>Phantom 4</td>
<td>35, 60</td>
<td>512x256, 256x512</td>
<td>2101, 2, 1931, 3</td>
<td>Close-up of panel 8 alignment pin</td>
</tr>
<tr>
<td>L8</td>
<td>Phantom 5</td>
<td>14</td>
<td>512x1024</td>
<td>1901</td>
<td>Panel 8 interior surface</td>
</tr>
<tr>
<td>L9</td>
<td>MotionCorder</td>
<td>25</td>
<td>256x240</td>
<td>500</td>
<td>Gap between panel 8 and carrier panel</td>
</tr>
<tr>
<td>L10</td>
<td>MotionCorder</td>
<td>6</td>
<td>256x240</td>
<td>500</td>
<td>Gap between panel 9 and carrier panel</td>
</tr>
<tr>
<td>L11</td>
<td>Phantom 5</td>
<td>15</td>
<td>512x1024</td>
<td>1901</td>
<td>Panel 8 interior surface</td>
</tr>
<tr>
<td>L12</td>
<td>Phantom 5</td>
<td>18, 10, 24</td>
<td>800x600, 256x512, 512x1024</td>
<td>4808, 1901, 1901, 1901</td>
<td>Panel 8 outboard rib</td>
</tr>
<tr>
<td>L13</td>
<td>Phantom 7</td>
<td>16, 10, 35-105</td>
<td>512x256, 512x384</td>
<td>2101, 1901, 1000</td>
<td>Close-up of impact zone from below</td>
</tr>
<tr>
<td>L14*</td>
<td>Phantom 4</td>
<td>10</td>
<td>512x256</td>
<td>2101</td>
<td>Down barrel of gun</td>
</tr>
<tr>
<td>L15*</td>
<td>Phantom 5</td>
<td>60</td>
<td>512x1024</td>
<td>1901</td>
<td>Panel 8 outboard rib</td>
</tr>
<tr>
<td>L16*</td>
<td>Phantom 4</td>
<td>25</td>
<td>512x512</td>
<td>1078</td>
<td>Downrange from gun barrel</td>
</tr>
</tbody>
</table>

* Only used for test RCC8

1 Setting for fiberglass panel 8L test 1 only
2 Setting for fiberglass panel 8L test 2 only
3 Setting for RCC panel 8L test only

### 4.2.2.2 Results of LESS panel 8L impact testing

#### 4.2.2.2.1 Results of LESS fiberglass panel 8L test 1 (FG8-1)

Although the plan was to test panel 9L when the LESS test article was designed and constructed, forensic evidence during the Investigation suggested that panel 8L was
the most likely impact location during STS-107 ascent. Since panel 8L is located at the
transition from the intermediate wing spar to the wing spar the panel has a unique
gеometry that skews the T-seal planes on each side. Also, panel 8L has a 7-ply doubler
that reinforces the aft-half of the lower surface facesheet. This doubler is designed to
react increased aerodynamic loads that are not present on panel 6L. For the initial shot on
a fiberglass panel 8L, an impact location similar to the final fiberglass panel 6L location
was chosen because facesheet strains recorded during the FG6-3 test were the highest
recorded to date. Additionally the incidence angle was adjusted to reflect a debris
trajectory representative to a panel 8L location.

The fifth LESS impact test was an impact test of fiberglass panel 8L (FG8-1)
conducted on June 27, 2003. The foam projectile, 5.6”x11.6”x22.6” in dimension and
1.69 pounds in mass, impacted the fiberglass panel at 770 ft/s within 1.0 inch of the
intended target. Ink lines on the foam projectile left a long black streak or skid mark
across panel 8L and its T-seal as shown in Figure 4.2-75. There was no notable change
of step and gap after the foam was removed from the panel/T-seal gap.

Figure 4.2-75: Photograph of the ink streak on fiberglass panel 8L after test FG8-1.
Principal strains recorded on panel 8L during the test are shown in Figure 4.2-76. The peak facesheet strain recorded for FG8-1 was approximately 37% higher than that recorded for FG6-3. These strains were the highest seen in any test. For this test condition, the facesheet stress on an RCC panel 8L would be approximately 18 ksi—a value likely high enough to cause failure. The load cells mounted to the base of the test fixture also recorded their highest RSS load to date. The load cell data is shown in Figure 4.2-77. The peak load for FG8-1 was approximately twice the load seen in FG6-1.

Inspection of fiberglass panel 8L after the test showed that the skid marks were not a constant darkness across the panel face—the ink became lighter as the trajectory path approached the outboard T-seal. This indicated that the foam was not as fully compressed as desired on the downrange panel face since loading was decreasing the further the projectile moved along the trajectory path. In other words, the foam was rebounding off of the panel before it reached the lock side. Measuring the length of the foam path across the panel surface and calculating the time for the foam to transverse this path, it is likely that the foam had begun to rebound and unload at 2.0 msecs, thus before it reached the outboard T-seal. Since the maximum facesheet strains had always been achieved on the downrange gages it was expected damage would occur on the downrange panel face or T-seal. This theory was supported by analysis that indicated the stress wave in front of the foam always maximized at the outboard edge where the deflected shape had the smallest radius of curvature. Since the forensic data indicated lower outboard panel 8L or T-seal 8 damage it was advantageous to stress the T-seal as well as the panel. Therefore assuming the ink marks reflected the magnitude of the pressure loading and since there was a desire to insure that the foam loading was maximized on outboard side
of the panel, the impact point for the second fiberglass 8 shot was translated approximately 6 inches along the original FG8-1 trajectory toward the center of the panel for the second fiberglass panel 8L test.

Figure 4.2-76: Schematic of panel 8L and T-seal summarizing the principal strains achieved during the test FG8-1 [16].

Figure 4.2-77: Plot of load cell data for test FG8-1.
4.2.2.2 Results of LESS fiberglass panel 8L test 2 (FG8-2)

A second impact test of fiberglass panel 8L, the sixth LESS impact test overall, was conducted on June 30, 2003. As stated previously the impact point for the second fiberglass 8L shot was translated approximately 6 inches along the original FG8-1 trajectory toward the center of the panel as shown in Figure 4.2-78. The foam projectile was of dimension 5.63”x11.56”x22.63” and its mass was 1.69 pounds. It was launched to an impact velocity of 778 ft/s and struck the panel within 0.7 inches of the intended target. Ink lines on the foam projectile left a long black streak across panel 8L and its T-seal as shown in Figure 4.2-79. Pre-test the maximum lock side step and gap were 0.02” and 0.04” respectively. The panel’s slip side maximum step and gap were -0.028” and 0.088” respectively prior to the test. The test increased the lock side step and gap to 0.27” and 0.813” respectively while the slip side step and gap opened to -0.287” and 0.193”. The T-seal was pushed approximately 9/16” onto the forward side of fiberglass panel 9L (Figures 4.2-80 and 4.2-81). Figure 4.2-82 shows the large amount of foam stuck in the panel/T-seal gap. Disassembly of the panel showed three external cracks on fiberglass panel 8L and one on the T-seal. Figure 4.2-83 shows a 0.75” long non-through crack located approximately 14.5” below the apex. It was formed on the upper lock side. Another crack was formed about 21” below the apex on the lower lock side. Figure 4.2-84 shows the non-through crack measures 0.85” across the face and the through the lock side. The outboard rib displayed a through crack 5.9” in length. Figures 4.2-85 and 86 shows the crack runs from the lower flange 10.2” towards the apex. NDE ultrasonic inspection of the panel showed three areas of delamination and one area on the T-seal. Figure 4.2-87 shows lock side delamination measuring 0.4”x3.75” and 0.2”x3.15”. An
area of delamination was found on the internal lock side rib measuring 1.15”x10.45” running from the lower flange toward the apex (Figure 4.2-88). The T-seal damage is shown in Figures 4.2-89 and 4.2-90. The delamination is on the slip side of the T-seal and measures 14.5” long and runs from the bottom of the T-seal toward the apex. Figure 4.2-91 shows a crack on T-seal 8L measuring approximately 3.44” starting at the bottom and running toward the apex. As noted previously the T-seal was pushed approximately 9/16” onto the forward side of fiberglass panel 9L.

Principal strains recorded on panel 8L during the test are shown in Figure 4.2-92. Peak strains on the facesheet were lower in this test than in the first fiberglass 8L test, although strains on the outboard rib and T-seal were substantially higher. The test damage on fiberglass panel 8L made the value of additional shots questionable so the decision was made to move on to the RCC panel 8L test article.

Figure 4.2-78: Drawing showing the initial impact point for fiberglass panel 8L test 2 (FG8-2). The impact point was translated approximately 6 inches down from the FG8-1 trajectory.
Figure 4.2-79: Photograph of fiberglass panel 8L and its neighboring parts after FG8-2.

Figure 4.2-80: Photograph of panel and T-seal after test FG8-2 showing foam wedged under T-seal.
Figure 4.2-81: Photograph of panel and T-seal after test FG8-2 showing T-seal damage. The T-seal was pushed approximately 9/16" onto the forward side of fiberglass panel 9L.

Figure 4.2-82: Photograph of the panel 8L T-seal showing the substantial amount of foam wedged in the panel/T-seal gap.
Figure 4.2-83: Photograph of a crack formed as a result of the FG8-2 test. The crack measures 0.75" across the face and through the lock side.

Figure 4.2-84: Photograph of a second crack formed as a result of the FG8-2 test. The crack measures 0.85" across the face and through the lock side.
Figure 4.2-85: Photograph of the external surface of a lock side rib crack formed as a result of the FG8-2 test. The through crack is 10.2” on the outer surface and 5.9” on the inner surface.

Figure 4.2-86: Photograph of an internal lock side rib crack formed as a result of the FG8-2 test. The through crack is 10.2” on the outer surface and 5.9” on the inner surface.
Figure 4.2-87: Photograph of a delamination on the fiberglass panel 8L lock side rib formed as a result of the FG8-2 test. The damage measures 0.4”x3.75”.

Figure 4.2-88: Photograph of a delamination on the internal fiberglass panel 8L lock side rib formed as a result of the FG8-2 test. The damage measures 1.15”x10.45” and runs from the lower flange toward the apex.
Figure 4.2-89: Photograph of the panel 8L T-seal delamination. The damage is located on the slip side, measures 14.5" long and runs from the bottom of the T-seal toward the apex.

Figure 4.2-90: Photograph of the panel 8L T-seal delamination. The damage is located on the slip side, measures 14.5" long and runs from the bottom of the T-seal toward the apex.
Figure 4.2-91: Photograph of a crack formed on the panel 8L T-seal as a result of the FG8-2 test. The crack starts at the bottom of the T-seal and runs toward the apex 3.44".

Figure 4.2-92: Schematic of panel 8L and T-seal summarizing the principal strains achieved during the test FG8-2 [16].
4.2.2.2.3 Results of LESS RCC panel 8L (RCC8)

The fiberglass testing and RCC analysis indicated that by moving the previous RCC panel 6 impact point (FG6-3) further aft and towards the center of the panel, the overall facesheet deflections would increase, making the step between the panel and the outboard T-seal larger. The larger step would allow more of the foam to impact the side of the T-seal and thereby increase the load going into the T-seal.

The RCC panel 8L impact test, the seventh and final LESS impact test preformed for the accident investigation, was conducted on July 7, 2003. The BX 250 foam projectile, of dimension 5.63"x11.5"x22.56" and mass 762.9 g, was launched to a velocity of 777 ft/sec. The foam projectile pitch and yaw angles were very nearly zero and the projectile leading edge struck within an inch of the intended target. The impact produced a large hole in panel 8L, as shown in the before and after images of Figure 4.2-93. The hole was roughly rectangular and measured approximately 16" by 16.75" inches as shown in Figure 4.2-94.

The foam projectile broke up into multiple pieces after impact. Recovered pieces of the projectile are shown in Figure 4.2-95. The largest of these pieces measured 5.63"x11.50"x10.75". Most of the largest foam debris was recovered outside the target; however, a great deal of the smaller foam debris was recovered from inside the test article.

Pieces of the broken RCC panel 8L facesheet were recovered only from inside the test article. The five largest pieces were able to be pieced together using the pattern of photogrammetry targets located on the inside of the panel (as shown in Figure 4.2-96). Dimensions of these pieces were: 8.1"x13.1", 7.4"x11.5", 3.9"x10.9", 4.5"x3.9" and
5.75" x 2.6". RCC panel 8L also exhibited cracks running on both the lock and slip sides from the hole to the apex. Figures 4.2-97 and 4.2-98 show the lock side cracks. Consulting the figure, a 3.25" long through thickness crack starts at the hole and runs toward the apex. A second, through crack extends from the top of the hole to the lock side (C to B) and measures 3.25" long. The crack then turned up the panel over the apex and on to the upper panel surface (B to A). This through crack measured 3.25" long. Figure 4.2-99 shows NDE radiographic images that support the conclusion that the cracks were indeed through the thickness. Similarly, the slip side of the panel exhibited through cracks. Figure 4.2-100 shows a 1.5" long crack that runs from the hole toward the slip side rib. The crack turns up 9.4" over the apex to the upper surface. NDE radiography (shown in Figure 4.2-101) show that the cracks were through the thickness. Slip side step and gap measurements showed a -0.080" max delta step and 0.090" max delta gap. The lock side showed a more significant max delta step and gap. The max delta step and gap were 0.128" and 0.133" respectively for the lock side.

Upon removal of the panel/T-seal assembly from the test article, a crack was found on the lower outboard T-seal lug as shown in Figure 4.2-102. Upon removal of the T-seal from the panel, a 9/16" x 7/8" fragment of RCC broke off of the lug as shown in Figure 4.2-103.

Principal strains measured on panel 8L and its T-seal during the test are shown in Figure 4.2-104.
Figure 4.2-93: Pre and post test RCC8 photographs of panel 8L

Figure 4.2-94: Photograph of the hole created in panel 8L during test RCC8
Figure 4.2-95: Photograph of recovered projectile fragments after test RCC8

Figure 4.2-96: Photograph of the largest recovered pieces of panel 8L
Figure 4.2-97: Photograph of lock side cracks on panel 8L

Figure 4.2-98: Photograph of lock side cracks on panel 8L viewed with respect to the large hole
Figure 4.2-99: Radiographic image of the lock side crack showing it is a through thickness crack.

Figure 4.2-100: Photograph of slip side cracks on panel 8L
Figure 4.2-101: Radiographic image of the slip side crack showing it is a through thickness crack.

Figure 4.2-102: Photograph of the T-seal lug crack prior to disassembly of RCC8
Figure 4.2-103: Photograph of the T-seal lug failure

Figure 4.2-104: Schematic of panel 8L and T-seal summarizing the principal strains achieved during the test RCC8 [16].
4.2.2.3 Discussion of RCC panel 8L impact testing

The fiberglass panel 8L impact tests supported the notion developed during fiberglass panel 6L testing that moving the initial impact point further to the aft and middle of the panel would yield higher strains in the RCC panel and potentially greater achieved damage. Greater deflections were expected since the projectile would load a larger, less stiff area of the panel than in previous experiments. The test results show that although strains and thus deflections in the panel 8 facesheet were higher, the tension stress in the rib near the panel 6L crack location were significantly lower. Figure 4.2-105 is a plot of the maximum principle strains in the outboard rib near the apex. The foam impacts the panel at t=0.1363 s in both tests. Looking at the high speed video images failure of panels 6L and 8L occur at approximately 1.6 and 2 ms respectively after foam impact. The difference in failure times can be attributed to the difference in impact point and trajectory of the projectile across the panel surface. It is notable that both panels fail when the foam projectile just reaches the T-seal. This is likely due to the fact that foam is fully compressed at this time and begins to rebound off of the panel. There may also be some contribution in strain due to stress wave moving through the complicated panel geometry.

The foam impact on RCC panel 8L created a hole some 16.75” x 17” in dimension. Forensic evidence supported by thermal analyses suggest that a hole 10” in diameter would be sufficient to generate the temperature rises observed in the MADS data and the overall structural damage noted in the panel 8/9 area. The damage produced during the test is a bit larger. Another significant observation is that all of the RCC fragments were found inside the panel/LESS cavity. Immediately after the accident the
Figure 4.2-105: Plot of RCC panel 6L and 8L test maximum principle strains. The strains were measured in rosettes on the outboard rib adjacent to the apex in both tests [16].

Air Force Space Command reviewed the Space Surveillance Network records looking for any anomalies during the STS-107 mission. They found that on flight day 2 a small object was in orbit with Columbia. This object’s orbit decayed by flight day 5. An extensive test program was conducted to match the object’s ballistic coefficient and radar cross-section with Orbiter materials. It was determined that only a fragment of an RCC panel approximately 140 in$^2$ or greater would match the observations made of the flight day 2 object. The fragments produced during testing are consistent in size with the flight day 2 object suggesting further that the RCC breech was the physical cause of the accident.
The kinetic energies for the panel 6L and 8L RCC tests were comparable (15,483 versus 15,852 ft-lbs respectively); therefore, if the damage were based strictly on kinetic energy one would expect very similar results for the two tests. This is clearly not the case. Clocking the projectile from a corner to an edge impact increased the loading imparted to the panel. In addition, the geometry of the panel itself allows the stiffness to vary from one location to another. In the apex region the panel is stiffer due to its arched geometry. Likewise in the lower and upper panel surfaces near the flanges the RCC material is thicker by design (these areas are called doublers). This stiffens the panel locally as well. Impacts in between the apex and doublers will tend to yield higher deflections than the stiffened regions just as was observed in the RCC panel 8L test. It is clear from this discussion that the threshold conditions required to fail a panel will vary with impact location on a panel and even panel to panel. Given the expense of full scale testing, the Orbiter program has chosen to evaluate panel response using analytical techniques.
5.0 Conclusions and Future Work

5.1 Conclusions

The results of the impact test program were invaluable to the CAIB. The board had already written their report, but delayed submittal of the findings to Congress until the outcome of the RCC panel 8L test was known. Although many factors contributed to the loss of Columbia and her crew (as identified by the CAIB), determining the physical cause of the accident was of critical importance. Impact testing on the MLGD TPS tiles showed that the tile could not be critically damaged given reasonable STS-107 ascent conditions. On the other hand, the RCC panel 6L test showed that RCC panels could indeed be damaged by foam debris during ascent. This fact was sufficient to enable the CAIB to identify the debris impact as the “likely” physical cause of the accident. After the conduct of the RCC panel 8L test the board changed the wording to the following:

“The physical cause of the loss of Columbia and its crew was a breech in the thermal protection system on the leading edge of the left wing. The breech was initiated by a piece of insulating foam that separated from the left bipod ramp of the External Tank and struck the wing in the vicinity of the lower half of Reinforced Carbon-Carbon panel 8 at 81.9 seconds after launch. During re-entry, this breech in the Thermal Protection System allowed superheated air to penetrate the leading edge insulation and progressively melt the aluminum structure of the left wing, resulting in a weakening of the structure until increasing aerodynamic forces caused loss of control, failure of the wing, and breakup of the Orbiter.”

The test methods used at SwRI enabled accurate measurement of all critical impact conditions. In particular, the use of high speed video cameras provided insight
into the deformation of the foam debris and targets. Finally, of note is the fact that RCC panel response varies widely with impact location. The results of the RCC panel 6L and 8L tests show that only a slight change in impactor orientation meant the difference between critical and non-critical TPS damage.

5.2 Future Work

5.2.1 CAIB Recommendations

After a thorough review of the accident and NASA’s processes, the CAIB made a number of observations and specific recommendations that the Agency must implement before returning the Shuttle to flight. A total of 29 recommendations were made—14 of which are required to be completed prior to return to flight (RTF) [25]. Seven of these recommendations require the completion of additional impact testing to implement the changes suggested by the CAIB. Recommendations R3.3-2 and R3.8-2 call for a comprehensive impact test and analysis program to evaluate RCC and TPS impact resistance. These recommendations are as follows:

*R3.3-2 Initiate a program designed to increase the Orbiter’s ability to sustain minor debris damage by measures such as improved impact-resistant Reinforced Carbon-Carbon and acreage tiles. This program should determine the actual impact resistance of current materials and the effect of likely debris strikes.*

*R3.8-2 Develop, validate and maintain physics-based computer models to evaluate TPS damage from debris impacts. These tools should provide realistic and timely estimates of any impact damage from possible debris from any source*
that may ultimately impact the Orbiter. Establish impact damage thresholds that trigger responsive corrective action, such as on-orbit inspection and repair, when indicated.

Given these recommendations, the Orbiter Debris Impact Assessment Team (ODIAT) was formed to fully investigate RCC and TPS tile performance for ascent debris impacts. Impact test plans were developed to evaluate the effects of ascent debris of various types including: ET TPS foam, SRB ablator, cryogenic surface accumulated ice and frost, and even metallic pad debris. In addition to the test capability developed at the SwRI, NASA activated new impact test facilities at White Sands Test Facility (WSTF) and Kennedy Space Center (KSC). The tile impact test plan was developed to be empirical in nature. Foam impact testing is predominately conducted at SwRI while the ice and ablator impact tests were conducted at WSTF and KSC respectively. Furthermore, all full scale RCC panel impact tests were conducted at SwRI while a series of RCC coupon impact tests were conducted at the Glenn Research Center. The two new facilities (WSTF and KSC) were developed in order to meet RTF critical milestones in a timely fashion. The lessons learned at each facility were passed on to each other so that all testing groups benefited from the collective knowledge of the team.

A third CAIB recommendation involves eliminating damaging ET TPS debris shedding.
R3.2-1 Initiate an aggressive program to eliminate all External Tank Thermal Protection System debris shedding at the source with particular emphasis on the region where the bipod struts attach to the External Tank.

While impact testing does not aid in redesigning the application of this TPS, the results of the activities identified above enables engineers to determine which TPS debris is considered critically damaging. Critical damage is defined as that damage which would result in the loss of the Orbiter vehicle and/or its crew. Elimination of all ET TPS debris shedding is a costly goal both in time and money. A more achievable goal is to eliminate as much foam as possible with the objective of eliminating all ascent debris sources that can cause critical damage to the Orbiter. In particular, as the ET team made efforts to reduce the size and mass of possible debris, these objects and associated transport variables (calculated impact velocity and angle) were compared to the set of damaging debris as determined through impact testing and analysis. Iteration between the teams determined the allowable debris that could be shed from the ET.

The impact test team has also supported the implementation of four other CAIB recommendations in a less direct capacity. These recommendations require real impact damage to test NDE methods and RCC and TPS tile repair techniques.

R3.3-1 Develop and implement a comprehensive inspection plan to determine the structural integrity of all RCC system components. This inspection plan should take advantage of advanced non-destructive inspection technology.
R3.4-3 Provide a capability to obtain and downlink high-resolution images of the underside of the Orbiter wing leading edge and forward section of both wings’ TPS.

R6.4-1 For missions to the International Space Station (ISS), develop a practicable capability to inspect and effect emergency repairs to the widest possible range of damage to the TPS, including both tile and RCC, taking advantage of the additional capabilities available when near to or docked at the ISS. For non-ISS missions, develop a comprehensive autonomous (independent of ISS) inspection and repair capability to cover the widest possible range of damage scenarios. Accomplish an on-orbit TPS inspection, using appropriate assets and capabilities, early in all missions. The ultimate objective should be a fully autonomous capability for all missions to address the possibility that an ISS mission fails to achieve the correct orbit, fails to dock successfully, or is damaged during or after undocking.

R3.3-3 To the extent possible, increase the Orbiter’s ability to successfully re-enter Earth’s atmosphere with minor leading edge structural subsystem damage.

As damage was produced during the test programs, test articles were made available to the non-destructive evaluation (NDE), inspection and repair teams. The use of NDE techniques on the articles allowed the team to evaluate methods both for ground operations and on-orbit assessment. Likewise, the boom sensor team made use of the test
articles to ensure the sensors were selected with sufficient resolution to discern damage while on-orbit. Furthermore, several of the RCC panels were cut apart post-test. In addition to material property testing, several of these samples are scheduled for arc jet testing to evaluate which damage modes are considered critical during vehicle re-entry. Finally, some of these samples will be repaired using on-orbit RCC repair techniques and then exposed to arc jet testing. These evaluations will ensure that the proposed repairs will withstand the harsh re-entry environment.

5.2.2 RTF Impact Testing

During the STS-107 investigation, the projectile for all of the impact tests was a relatively large piece of foam with dimensions of roughly 21" x 11.5" x 5.5" (see Figure 5.2-1). This projectile was representative in material type and size to the external tank bipod ramp—the ascent debris source that caused critical Orbiter damage.

Figure 5.2-1: Projectile made from external tank BX-250 foam (21" x 11.5" x 5.5").
During the on-going RTF test program, numerous projectile materials, representative of likely ascent debris sources, are being launched at both shuttle wing leading edge RCC panels and TPS tiles. A wide range of sizes, angles of impact and velocities are being tested so as to bound, through testing, every possible impact scenario that might occur on launch. Damage thresholds are determined by changing these variables and empirically based performance equations are being developed from these data. Likewise, these data are used to improve our confidence in computational and analytical methods being developed.

Figure 5.2-2: Various projectile materials, shapes, and sizes.
Figure 5.2-3: LCGG aimed at Shuttle wing leading edge panels.

Figure 5.2-4: Medium compressed gas gun.
The debris materials include foam material from the ET, ablator materials from the solid rocket boosters (SRBs), ice which forms at various locations on the ET and SRBs, and metal which could potentially fall off areas of the launch pad structure. Projectiles were fabricated of these materials in various sizes and masses. Representative examples of these projectiles are shown in Figure 5.2-2. To represent the ET foam debris, projectiles of rectangular cross-section (between 1” to 4” wide by between 2” to 8” tall) were selected. The lengths of the projectiles were adjusted to vary the mass of the simulated debris. The ablator also has a rectangular cross section, but the expected debris sizes are much smaller (typically 0.5” wide by 1” tall by 1.5” long). Various sizes and densities of ice are being produced and tested. Both low density (~37 pound/ft3) ice (as would occur when moisture from the air condenses on a surface near or in contact with a cryogenic tank) and normal density (~58 pound/ft3) ice (clear ice) with both rectangular and cylindrical cross sections are being studied. Only one size of metal projectile is being tested, a standard copper-coated BB. This projectile was selected to represent scale and rust that could be shed from the Shuttle launch pad structure.
Due to the large variation in projectile materials, size, and mass, various launcher systems are being utilized. The SwRI large compressed gas gun (LCGG as shown in Figure 5.2-3) was used for all investigation tests, mainly because it was the only gun system suitable for launching the large foam projectiles [26]. This gun system is also being used extensively during the RTF work, but with much smaller rectangular foam projectiles. The two main sizes of projectiles being launched with the LCGG system are foam projectiles with cross sections of 2” x 6” and 2” x 4” (as opposed to the 5.5” x 11.5” cross section used during the investigation). The main benefit of launching the low-strength foam with this gun system is that with the large volume pressure tank (500 gallons) and the long (40-foot) barrel length, the accelerations required to achieve the desired launch velocity are relatively low.

Two other systems are being used for the RTF testing, a medium compressed gas gun (MCGG as shown in Figure 5.2-4) and a small compressed gas gun (SCGG as shown in Figure 5.2-5). The MCGG (150 gallon tank) is used to launch foam projectiles with various cross sections. The barrel length is 20 feet so the velocity that can be achieved with this system is limited. The ice and ablator projectiles are launched using the SGG (3 gallon tank). The ablator is launched in a barrel with the same cross section as the projectile (the same way the foam is launched). The ice, however, must be saboted and fired from a 2” diameter bore barrel. The sabot is abruptly stopped at the muzzle, allowing the projectile to fly towards the target.

All three of the compressed gas gun systems operate essentially the same way. The projectile is loaded into the barrel and the barrel is then bolted to the gun breech. The high-pressure reservoir is then charged to a predetermined pressure (with nitrogen or
helium gas) based on the desired velocity. Nitrogen gas is used to achieve velocities equal to and below 1000 ft/s, while helium is used to launch projectiles to velocities in excess of 1000 ft/s up to nearly 2000 ft/s. Then, a fast-acting valve is activated which releases the high-pressure gas to the base of the projectile, accelerating it out of the barrel. All three guns are also mounted on modified car lifts to provide vertical adjustment of the shot line height. Air bearings at the base of the lift provide yaw adjustment of the gun relative to the target.

For the metal shots, the BBs are fired from a standard pump-up BB gun and shoulder fired at the target (see Figure 5.2-6). The velocity of this gun is surprisingly repeatable and can be adjusted based on the number of times the gun is pumped.

Figure 5.2-6: Gunner aiming BB gun prior to shooting an RCC panel.
5.2.3 Thermal Protection System Tile Testing

One of the most aggressive RTF test requirements is the conduct of the TPS tile testing program. A total of over 764 impact tests were scoped to empirically determine tile impact performance: 398 foam, 198 ice, 120 ablator and 48 metal projectiles. Although NASA facilities were utilized to test ice and ablator materials against TPS tile targets, WSTF and KSC respectively, SwRI’s role in this task was limited to impacting the TPS tiles with ET foam—approximately 52% of the total tests. As stated previously, many geometries of foam are being launched over a range of velocities at TPS tile arrays (see Figure 5.2-7) set at various angles. Figure 5.2-8 illustrates the variations.

Figure 5.2-7: Photograph of a TPS tile array.
Figure 5.2-8: Schematic of foam cross-sections, target angles, and projectile velocities for the foam on tile tests at SwRI.

The MCGG was used for the majority of these tests, although the LCGG was used for some of the higher velocity shots. To align the gun on the target, the system is adjusted such that the muzzle is nominally 36" from the impact point. A bore-mounted laser is used to site the desired impact point during the alignment process. The majority of the tests were performed indoors. When the foam projectiles impact and damage the tile targets, a great deal of airborne dust is generated. Precautions are taken to protect personnel from potentially hazardous debris.

5.2.3.1 Tile test results

Figure 5.2-9 shows the spectrum of damage modes demonstrated through the test program for foam impactors. Of particular note is the unexpected complete tile loss
failure mode (see Figure 5.2-10). Testing has shown that this damage mode is only encountered when tiles are impacted without gap fillers installed between the neighboring tiles. Ceramic fabric sheets called gap fillers are placed in between TPS tiles on the Orbiter where the gap between the tiles does not meet specification. The gap fillers prohibit hot gases from impinging on the structure. It appears that these gap fillers also provide an unintentional benefit for impact resistance since they tend to share the load between the neighboring tiles around an impact size, thus eliminating the tile out failure mode (for the reasonable range of impact conditions tested). The presence or absence of these gap fillers also has a significant influence on the level of observed tile damage. In particular the damage is smaller and more erratic where the gap filler is present. Gap fillers were removed for all testing starting near the beginning of the test program so that the results would yield conservative empirical performance equations. This was an important step since we cannot ensure, nor do we have any way of knowing exactly where gap fillers are installed on the vehicle (unless a local visual inspection is performed). Under review is the possibility of adding gap fillers to locations on the vehicle where likely impact conditions are conducive to tile loss.

The particular impact location on a tile has a significant influence on the level of observed tile damage (i.e. "edge" versus "center" impact). The test program uses bounding cases in an attempt to capture this influence through damage scatter. The tests to date have demonstrated more damage data scatter than anticipated since array tile damages are not as "clean" as previously tested single tile damage. It has also been observed that tile failure (fracture) occurs at the interface between the tile's densified layer and the undensified substrate. TPS tiles consist of aligned bundles of silica fibers.
Figure 5.2-9: Spectrum of damage on TPS tile for foam impacts.

Figure 5.2-10: Photograph of the complete tile loss failure mode.

Most of the substrate is very porous, yielding the exceptional thermal properties of the material; however near the bond surface of the tile, the material is densified through a chemical process to improve the tile’s structural capability. This, coupled with the strain
isolation pad, ensures that the tile remains bonded to the vehicle throughout its mission profile.

Another observation is that tile is easily damaged by debris for expected impact velocities and angles; however, the criticality of the damage must be assessed through thermal modeling of the re-entry heating process. That is, just because damage is demonstrated during the test program does not mean that the TPS would be compromised during re-entry heating. Criticality of tile damage will vary with site location since the heating rates are different all over the vehicle. Similarly the tile thickness varies widely to maintain thermal margins and aerodynamic mold line—another contributing factor to the need for site specific critical damage definitions. An assessment of critical depth of tile penetration and extent of crater are underway.

Typically tiles located on the underside of the vehicle are of the high reflectance surface insulation (HRSI) type (black) while those on the upper surfaces of the vehicle are typically low reflectance surface insulation (LRSI, white) tiles. Testing to date shows that the HRSI tiles show significantly greater resistance to foam impact damage than the LSRI tiles. This is contributed to the difference in the glass surface coating applied to the tiles. The HRSI tile glass coating is denser than that of the LSRI tile.

In addition to the characterization of acreage tile, some testing was performed on tiles in special locations, such as around the MLGDs and tiles attached to access panels (also called carrier panels) near the leading edge RCC panels. Based on limited characterization testing with 12” x 6” x 2” (0.20 lb) BX-265 foam at 45° impact angles, the carrier panel special configuration tests showed minimal damage from foam impacts at worst case conditions. The tiles were tested to total kinetic energy levels of 1600 to
3200 ft-lb at a 13° local impact angle. That angle is considered worst case in the RCC panel 6 locations. Testing of tiles that surround the MLGDs showed significant damage and a potential flow path into the gear cavity at extreme, worst-on-worst foam impact conditions. Total kinetic energy levels of 1600 to 3800 ft-lb at a 13.5° impact angle resulted in embedded foam and permanent deformation of the door thermal barrier and cracking of multiple tiles at the densified layer interface. These tests were repeated on the MLGD special configuration test article for more realistic foam impact conditions. The results showed minimal damage and no potential flow path at total kinetic energy levels of 710 to 770 ft-lb at a 13° impact angle and total kinetic energy levels of 940 to 1900 ft-lb at a 7.5° impact angle. These results suggested to the Shuttle program that a redundant thermal barrier should be added to the MLGDs to establish thermal margin; however, since for reasonable ascent impact conditions no damage was exhibited this design change will be implemented after first return to flight.

5.2.4 Wing Leading Edge RCC Panel Impact Testing

In an effort to better understand the impact resistance of the leading edge subsystem, an analysis and impact test program was developed. Since RCC panel resources are so limited and costly (~$3.5M per panel), an empirical based approach was not practical. Instead it was decided to evaluate RCC panel impact resistance through finite element analysis (FEA) supplemented with coupon level impact testing and very limited full scale validation testing. Called a building block approach, material coupons were extracted from panel 8L which was tested during the STS-107 investigation. Static and high strain rate tests were performed to confirm that the material properties fell within A-basis allowables established during the Orbiter development.
Next, a series of LS-DYNA FEAs were conducted for 6" square RCC flat panels. These anticipated results were compared with flat plate testing conducted at the Glenn Research Center. At this time, the results of these tests have not been thoroughly compared with FEA results; however, the qualitative indication is that the experimental results (deflections in particular) match quite well with analyses. Finally, a test program on full scale RCC panels will be conducted at the end of the 2004 calendar year. The parameters are to be determined; however, it is anticipated that foam, ice and ablator impactors will be launched at each of three RCC panel 9Ls dedicated to the validation program.

In the interim, several dozen impact tests have been conducted on full scale RCC panels for other purposes. First, a series of three impact tests were conducted on a 9L RCC panel to demonstrate the threshold energy required to fail the panel. Testing demonstrated that the panel exhibits damage if it is struck with foam at a kinetic energy between 1470 to 3384 foot-pounds. Figure 5.2-11 shows the damage produced by the third impact test conducted on the apex region of the panel. A 0.2 pound projectile, less than 10% of the mass of the projectile launched during the STS-107 investigation, traveling at 1167 ft/s struck the panel at a local impact angle of 45 degrees.

Similar testing on an RCC panel 16R demonstrated that this threshold (for the apex region of the panel) lies somewhere between 1553 to 1784 foot-pounds for its somewhat different geometry. Figure 18 shows the damage induced by the 0.06 pound projectile launched to an impact velocity of 1384 ft/sec. NDE determined that the damage was a through crack of the panel some 7" in length (Figure 5.2-12).
Figure 5.2-11: Photograph of Panel 9L (OV-103), test 3 impact damage.

Figure 5.2-12: Panel 16R (OV-105), test 2 impact damage.
Another objective for full scale panel impact testing is to demonstrate the impact penetration sensor system (IPSS). The IPSS is a network of accelerometers and wireless data acquisition systems that will be installed on the rear side of the leading edge spar on the Orbiter fleet. During a mission, the accelerometers will measure g-levels experienced at high data rate (20,000 Hz). These data will be stored on-board the Orbiter and, once on-orbit, engineers will be able to download these data selectively to determine where any leading edge strikes may have occurred during ascent. Likewise, the system will have a limited life to evaluate impact incurred on-orbit due to meteoroids and orbital debris. Full scale panel testing with these sensors in place allows the IPSS team to develop structural response models that will enable interpretation of the signals the team will measure during a mission. Early identification during a mission of an ascent impact will improve the likelihood of repair success by focusing optical and laser scanning sensors on areas of concern.

Both the LCGG and the SCGG systems are being used to for the RCC panel testing. SwRI is responsible for testing all of the previous listed materials at the RCC panels (ET foam, ablator, ice, and metal). These tests are currently being performed out doors at the SwRI ballistics range.

Up to seven high-speed cameras are used for these tests. Both Phantom V7 and V5 cameras are used. In addition to the velocity camera and the orthogonal-view camera mentioned in the previous section, there is also a camera positioned to look vertically down on the impact point. There are also 2 cameras looking at the inside surface of the RCC panel to observe the target response (see Figure 5.2-13). Additionally, there are 2 photogrammetry cameras looking at the impact point on the outside of the target. These
last 2 cameras are positioned exactly 90-degrees apart from one another to allow the NASA-JSC photogrammetry group to make precise measurements of deflection.

![Image](image.jpg)

**Figure 5.2-13:** RCC panel spar test article with cameras in place to measure target response.

### 5.2.5 Remaining RTF work

Given the enormous amount of data generated in such a short period of time, synthesis of the results and decisions regarding vehicle modifications are woefully needed. These efforts will continue until the end of calendar year 2004 in support of the first return to flight. It is anticipated, however, that the test programs conducted to date are first steps in a continuing effort to understand TPS impact resistance. These materials will likely be used for next generation crew transport vehicles and most certainly the data will be used to help design new impact hardened TPS.
Due to the efforts of the impact testing and analysis teams across the Agency (including the supporting contractor team), the Shuttle program will have a well founded understanding of how the thermal protection system will respond during an impact event. This work will greatly improve our understanding of mission risk and enable the selective reduction in risk through safe mission operations and engineering design and implementation.
6.0 Bibliography


7.0 Appendix A: Effect of Orbital Debris Shape on Lightweight Spacecraft Shielding

7.1 Introduction

Meteoroids and orbital debris (M/OD) pose a significant hazard to low earth orbiting spacecraft. Researchers have developed engineering models that describe this environment and allow for the evaluation of impact risk. Furthermore, selective hypervelocity impact (HVI) testing allows for the evaluation of spacecraft material response to this threat. Lightweight spacecraft shielding are developed to protect critical elements from penetration. This approach enables mission and program managers to manage their risk with respect to this environment.

A shortfall of this approach, as with any engineering task, are the assumptions made. In particular, the engineering model does not include a description of M/OD shape. Risk assessments performed using this tool assume that the debris are spherical although we clearly know they are not. Since STS-50, NASA scientists and engineers have collected impact data from returned Shuttle Orbiter surfaces [1]. These data include description of impact damage and impactor composition. Using semi-empirical damage equations, one may obtain an estimated impactor size from this information. Window impact damage yields valuable insight as to the shape of these M/OD for particle sizes on the order of 5 to 400 microns. For the purposes of this discussion, impactor shape will be described in terms of a length to diameter (L/D) ratio. A sphere has an L/D equal to 1. A disk has an L/D less than one and a rod has an L/D in excess of 1. Figure 7.1-1 shows a histogram of estimated M/OD impactor shapes calculated from flight data.

A statistical analysis of the distribution shows that the particles have a mean L/D of 0.87 and a standard deviation of 0.45. The data also show that the M/OD have L/D as
small as 0.2 and as large as 6.6. This means that we are most likely to encounter particles that resemble disks or rods rather than spheres. Figure 7.1-2 shows radiograph images of several equal mass impactors striking a bumper [2]. Each impactor has a different shape and yields strikingly different debris clouds. It is the distribution of projectile/bumper mass and their kinematics which will fail a given pressure wall. It has been demonstrated that disks and rods can be more damaging to spacecraft structures than equal mass spheres [3]. These preliminaries provide the basis of motivation for the following work.

![Histogram of estimated M/OD length to diameter ratios for STS-50 to 103.](image)

**Figure 7.1-1:** Histogram of estimated M/OD length to diameter ratios for STS-50 to 103.

### 7.2 Hypervelocity impact testing with non-spherical projectiles

A test series was conducted to begin to evaluate the shape effect of M/OD impactors on lightweight spacecraft shielding. The objectives of the program were as follows: to observe the effect of projectile L/D (shape) on a typical ISS shield and to
Figure 7.1-2: Radiographs of debris clouds produced by the impact of four different shapes of equal mass [2].

evaluate the experimental process for examining debris cloud composition and kinematics. The ultimate goal is to utilize these data in developing a generic debris cloud model.

A total of 10 tests were conducted on a shield similar to the International Space Station (ISS) Common Module cylinder section. All tests involved using a two stage light-gas gun to launch circular cylindrical projectiles to velocities near 7 km/s. The projectiles were of equal mass (450 mg +/- 10 mg) and had L/D ratios of 0.25, 0.5, 1, or 1.5. Flash x-ray systems were used to capture the projectile orientation in flight. In addition, a Cordin ultra-high speed camera was utilized to capture images of the impact event itself. These diagnostic systems enabled precise measurement of the debris cloud expansion and projectile orientation.
7.2.1 Results

Table 7.2-1 shows all of the pretest details while Table 7.2-2 documents the results of the testing effort. Although all targets were impacted at 0 degrees of obliquity with respect to the shot line, the projectiles themselves were launched with a random tilt. This tilt (or total angle) is the angle measured between the normal vector of the front face of the projectile to the flight line. None of the projectiles impacted such that the total angle was 0 degrees (that is, with their front face normal to the bumper). Instead, the total angles ranged between 17 and 69 degrees. Figure 7.2-1 shows the effect of projectile total angle on rearwall damage. For an L/D of 0.25, we see that a difference in total angle of approximately 30 degrees can mean the difference in rearwall hole formation. Crater size on the rearwalls provides an indication of the relative size of the fragments in the debris cloud that impinged on the plate.

An explanation for the difference in the results is that more projectile mass is shocked to a higher pressure in case (a) compared with case (b). Figure 7.2-2 is a schematic suggesting how the projectile interacts with the bumper and the resulting debris cloud composition. When the projectile strikes the bumper, a shock wave is transmitted through the projectile and the bumper. As the shock interacts with free surfaces in either material, the intensity of the shock is attenuated. This means material closer to the leading edge of the projectile will experience the highest pressures. Material close to free surfaces (in particular the rear) will experience large tensile forces due to wave rarefaction. It is likely that this material will spall from the bulk of the projectile if the tensile strength exceeds the yield. For sufficiently high impact velocities (thus pressures) the projectile and bumper material can not only fragment, but also change state
Table 7.2-1
Projectile parameters for impact testing

<table>
<thead>
<tr>
<th>Test Number</th>
<th>Impact Velocity (km/s)</th>
<th>Projectile Shape</th>
<th>Projectile Length (mm)</th>
<th>Projectile Diameter (mm)</th>
<th>L/D</th>
<th>Projectile Mass (g)</th>
<th>Average total angle (degrees)</th>
</tr>
</thead>
<tbody>
<tr>
<td>3</td>
<td>6.8</td>
<td>Right circular cyl</td>
<td>9</td>
<td>6</td>
<td>1.5</td>
<td>0.67838</td>
<td>20</td>
</tr>
<tr>
<td>4</td>
<td>6.9</td>
<td>Sphere</td>
<td>6.75</td>
<td>6.75</td>
<td>1.0</td>
<td>0.44866</td>
<td>-</td>
</tr>
<tr>
<td>7</td>
<td>6.7</td>
<td>Right circular cyl</td>
<td>7.77</td>
<td>5.18</td>
<td>1.5</td>
<td>0.44050</td>
<td>41</td>
</tr>
<tr>
<td>8</td>
<td>6.9</td>
<td>Right circular cyl</td>
<td>3.81</td>
<td>7.52</td>
<td>0.5</td>
<td>0.45284</td>
<td>45</td>
</tr>
<tr>
<td>10</td>
<td>6.9</td>
<td>Right circular cyl</td>
<td>3.83</td>
<td>7.49</td>
<td>0.5</td>
<td>0.45025</td>
<td>48</td>
</tr>
<tr>
<td>11</td>
<td>6.9</td>
<td>Right circular cyl</td>
<td>7.84</td>
<td>5.18</td>
<td>1.5</td>
<td>0.44399</td>
<td>40</td>
</tr>
<tr>
<td>12</td>
<td>6.8</td>
<td>Right circular cyl</td>
<td>3.84</td>
<td>7.5</td>
<td>0.5</td>
<td>0.45409</td>
<td>17</td>
</tr>
<tr>
<td>13</td>
<td>6.8</td>
<td>Right circular cyl</td>
<td>3.8</td>
<td>7.5</td>
<td>0.5</td>
<td>0.45506</td>
<td>69</td>
</tr>
<tr>
<td>14</td>
<td>6.9</td>
<td>Right circular cyl</td>
<td>2.43</td>
<td>9.49</td>
<td>0.25</td>
<td>0.45708</td>
<td>57</td>
</tr>
<tr>
<td>15</td>
<td>6.9</td>
<td>Right circular cyl</td>
<td>2.42</td>
<td>9.5</td>
<td>0.25</td>
<td>0.46019</td>
<td>54</td>
</tr>
</tbody>
</table>

Table 7.2-2
Quantitative results of impact testing

<table>
<thead>
<tr>
<th>Test Number</th>
<th>Total hole area (cm²)</th>
<th>Total spall area (cm²)</th>
<th>Normalized Spall Damage</th>
<th>Damage Summary</th>
</tr>
</thead>
<tbody>
<tr>
<td>3</td>
<td>5.54</td>
<td>21.40</td>
<td>2.67</td>
<td>Rear wall failed, 1 large hole and two small; 1 large spall</td>
</tr>
<tr>
<td>4</td>
<td>0.00</td>
<td>8.01</td>
<td>1.00</td>
<td>Rear wall failed, no perf.; 1 large spall</td>
</tr>
<tr>
<td>7</td>
<td>2.33</td>
<td>11.00</td>
<td>1.37</td>
<td>Rear wall failed, 2 large holes in rear wall, 2 pinholes; 1 large spall</td>
</tr>
<tr>
<td>8</td>
<td>0.14</td>
<td>3.68</td>
<td>0.46</td>
<td>Rear wall failed, pinhole perf.; 3 large spalled areas</td>
</tr>
<tr>
<td>10</td>
<td>0.00</td>
<td>4.70</td>
<td>0.59</td>
<td>Rear wall failed, 1 spallation of rear wall, no perf.</td>
</tr>
<tr>
<td>11</td>
<td>1.98</td>
<td>10.28</td>
<td>1.28</td>
<td>Rear wall failed, 1 large hole, 1 large spallation</td>
</tr>
<tr>
<td>12</td>
<td>0.00</td>
<td>0.30</td>
<td>0.04</td>
<td>Rear wall failed, 1 spallation of rear wall, no perf.</td>
</tr>
<tr>
<td>13</td>
<td>1.10</td>
<td>11.90</td>
<td>1.49</td>
<td>Rear wall failed, 2 large holes and 3 small ones; 1 large spall</td>
</tr>
<tr>
<td>14</td>
<td>4.10</td>
<td>8.60</td>
<td>1.07</td>
<td>Rear wall failed, 1 large hole and 1 small; 1 large spall and 1 small</td>
</tr>
<tr>
<td>15</td>
<td>0.00</td>
<td>1.83</td>
<td>0.23</td>
<td>Rear wall failed, no perf.; 1 large spall and 1 small</td>
</tr>
</tbody>
</table>

upon shock release. A fraction of the projectile and bumper mass is expected to be liquid or vapor for this reason.

Total angles that maximize the projectile surface that strikes the bumper will tend to yield debris clouds with more finely disrupted projectile mass. Furthermore, the projectile mass will tend to have a greater radial velocity component. For example, in case (a) at least half of the projectile mass contacted the bumper surface. This yields the finely disrupted projectile mass that cratered the lower left side of the rear wall. The projectile mass that did not contact the bumper surface was fragmented during release of
shocks propagating from the leading projectile edge. These shocks were of lower peak pressure due to interactions with free surfaces as the shock reached the back of the projectile. Since the shock amplitude is lower, the fragment sizes tend to be larger. In case (b) the projectile (of disk shape) struck the bumper nearly edge on. In this case the diameter is 4 times the thickness of the projectile. This means that as the shock propagates across the diameter of the projectile it is being attenuated rapidly. One would expect up to 4 reflections from the free surface in the time it takes the shock to propagate to the rear free surface. Physically this means that most of the projectile saw pressures near or below the yield and so there was a large remnant fragment (or several large fragments with small radial velocity) in the cloud. The large fragment was of sufficient size to perforate the rearwall.

It is necessary to establish some measure of damage to the rearwalls in order to compare the results yielded by the various projectiles. In each case, the debris cloud cratered the rearwall. Furthermore, the rearwalls showed evidence of rear surface spallation. This is caused by tensile rarefaction waves that exceed the rearwall material yield strength. These waves are generated as the cloud impinges on the wall. Measurement of this damage indicates the momenta that were imparted to the plate. Normalizing the spall areas of rearwalls damaged by non-spherical projectiles to those of a rearwall damaged by an equal mass sphere, indicates the relative damage caused by the projectile’s shape. Values greater than one indicate that the debris cloud mass was more damaging to the rearwall than for the spherical case. Figure 7.2-3 shows this damage parameter plotted versus total angle. Note that even the L/D equal to 0.25 case is more damaging than the sphere for a total angle greater than approximately 55 degrees. It is
also observed that each projectile begins to produce spallation damage after a critical
total angle is exceeded. For these conditions and an L/D equal to 0.25, that angle is near
50 degrees and for an L/D of 0.5 the critical angle is near 20 degrees. The other
projectiles always produced spall.

Likewise by examining the total area of the hole produced in the rearwall (Figure
7.2-4), it is noted that total angles of approximately 55 and 50 yield a hole in the rearwall
for L/D equal to 0.25 and 0.5 respectively. Although the data are limited at this time, it is
clear that the damage produced by a given projectile is strongly dependant on the
impactor orientation.

![Figure 7.2-1: Rearwall damage for targets impacted by L/D=0.25. (a) Projectile impacted at a 54 degree total angle. (b) Projectile impacted at an 85 degree total angle.](image)
Figure 7.2-2: Schematic of suggested composition and state of projectile and bumper material after release from shocked state.

Figure 7.2-3: Plot of normalized spallation damage vs. total angle.
7.2.2 Debris cloud kinematics

Analysis of the Cordin shadowgraph filmstrips allows for the measurement of debris cloud expansion as it travels toward the rearwall. The rate of expansion is important since higher radial velocities suggest that the areal momentum load will be lower and thus less damage. Figure 7.2-5 shows plots of the normalized axial and radial velocities (with respect to the impact velocity). The data suggest binomial curve fits may be applied as follows:

\[
\frac{V_{\text{axial}}}{V_0} = 0.06x^2 + 0.26x + 0.69 \ (R^2 = 0.76) \quad [1]
\]

\[
\frac{V_{\text{ radial}}}{V_0} = -0.13x^2 + 0.45x + 0.3 \ (R^2 = 0.68) \quad [2],
\]

where \(x = \frac{L}{D} (V \sin(\text{total angle}))^{0.5}\).
Figure 7.2-5: Plot of normalized debris cloud axial and radial velocities.

7.3 Conclusions

The limited testing to date has to some extent demonstrated the effect of impactor shape on lightweight spacecraft shielding performance. It is clear that rearwall damage has a strong dependence on not only the length to diameter ratio of the impactor but also its orientation. For equal mass projectiles, it was shown that projectiles with an L/D even as small as 0.25 can be more damaging than a sphere for the same impact conditions. Shock wave interactions with free surfaces determine the debris cloud material state and thus rearwall damage. Two measures of rearwall damage were introduced as a technique for comparing damage generated by various projectiles. Similarly, it was shown that measurement of shadowgraph imagery is an effective means for evaluating debris cloud kinematics.
7.4 Future work

Testing with non-spherical impactors is on going. It is planned to continue testing with right, circular cylinders, but begin to vary several parameters. By changing the ratio of the bumper thickness to the effective projectile length, projectile mass, and target obliquity, more will be learned about the composition and lethality of the resulting debris clouds. Furthermore, a closer evaluation of damaged rearwalls should yield data on fragment size and population and even projectile/bumper material state. These data are essential in the development and validation of a debris cloud model.

7.5 References

