Space Shuttle Cargo Integration Coupled Loads Analysis Lessons Learned

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When a system experiences a loading environment characterized by rapidly varying forces, such as a rocket launch, a transient analysis is used to analyze the response of the system. The most common transient analysis methodology is the Coupled Loads Analysis (CLA). CLAs are used by the automotive and aerospace industry to analyze cars, trucks, planes, helicopters, spacecraft, etc. The Space Shuttle program used the CLA methodology to assess the compatibility of the payload with the Orbiter and the flight environment. The Space Shuttle Verification Loads Analysis (VLA) was a standardized CLA process that started between ten and thirteen months prior to launch, and included several meetings as well as analysis by both the Space Shuttle Program and the payload developers to verify that the payloads were compatible with the flight loads environment and would not interact negatively with the vehicle. Over the course of the Space Shuttle Program, many improvements were made to the process, which reduced cycle time and improved manifest flexibility. There were several issues which were never fully addressed, but workarounds were developed to keep the process flowing. The lessons learned included automation of some processes and standardization of others, early assessments, improved documentation and better coordination with all stakeholders in the process. Lessons learned also included the limitations in the current process, and what to do to avoid the same issues in the future.

I. Introduction

Each flight of the Space Shuttle required structural analysis to verify that the Space Shuttle Vehicle and the Cargo Elements (CEs) (called payloads) in the Orbiter cargo bay could withstand the flight loads environment. For the Space Shuttle Program (SSP), this analysis was accomplished via the Integrated Vehicle analysis and the Cargo Integration (CI) analysis. The Integrated Vehicle analysis assessed the External Tank (ET), Solid Rocket Boosters (SRBs), and

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the Orbiter against a pre-determined set of limits. The CI analysis generated internal loads for the assessment of CEs and determining interface compatibility against the Orbiter capability.

Over the final ten years of the SSP, nearly 100 Coupled Loads Analyses (CLAs) were conducted as part of the CI Verification Loads Analysis (VLA) process. The VLA provided the flight environment loads that were used for the final assessments of the Orbiter capability and the CE hardware strength and life. The VLA was also the last check to verify adequate clearances between the Orbiter and CEs.

During the course of the VLA process, various issues "arose, ensued and were overcome[4]" by the dedicated engineers and analysts involved with the Space Shuttle and its payloads. This group included those at the National Aeronautics and Space Administration (NASA) Johnson Space Center (JSC), Boeing Space Exploration, United Space Alliance (USA) and Jacobs Engineering as well as all the CE/payload developers and other NASA centers and their support contractors. Some of the issues encountered had a potential to derail the process to verify the hardware for flight (technical issues) and others were significant impacts to the schedule, threatening the launch date. Along the way, VLA process efficiencies and improvements were developed and implemented. This paper addresses these issues and the lessons learned from them.

Although the Space Shuttle was unique among space vehicles due to its reusability and return capability as shown in Figure 1, it is hoped that these lessons learned will be applicable to new vehicles and programs in the future. These lessons learned cover items from the final ten years of the program, although some go back further. The lessons address issues with the VLA process itself and do not address the dynamic and static math model verifications, which were usually completed by the time the VLA started; however, the impacts of verifications not completed prior to the start of VLA are included.

Most of the Space Shuttle flights in the final ten years carried hardware to the International Space Station (ISS), which consisted of a cargo element carrying smaller payloads, rather than one large payload element; therefore, the term cargo element will be used most often throughout this paper.

[4] Quote from the character Jack Sparrow, *Pirates of the Caribbean: Dead Man's Chest*

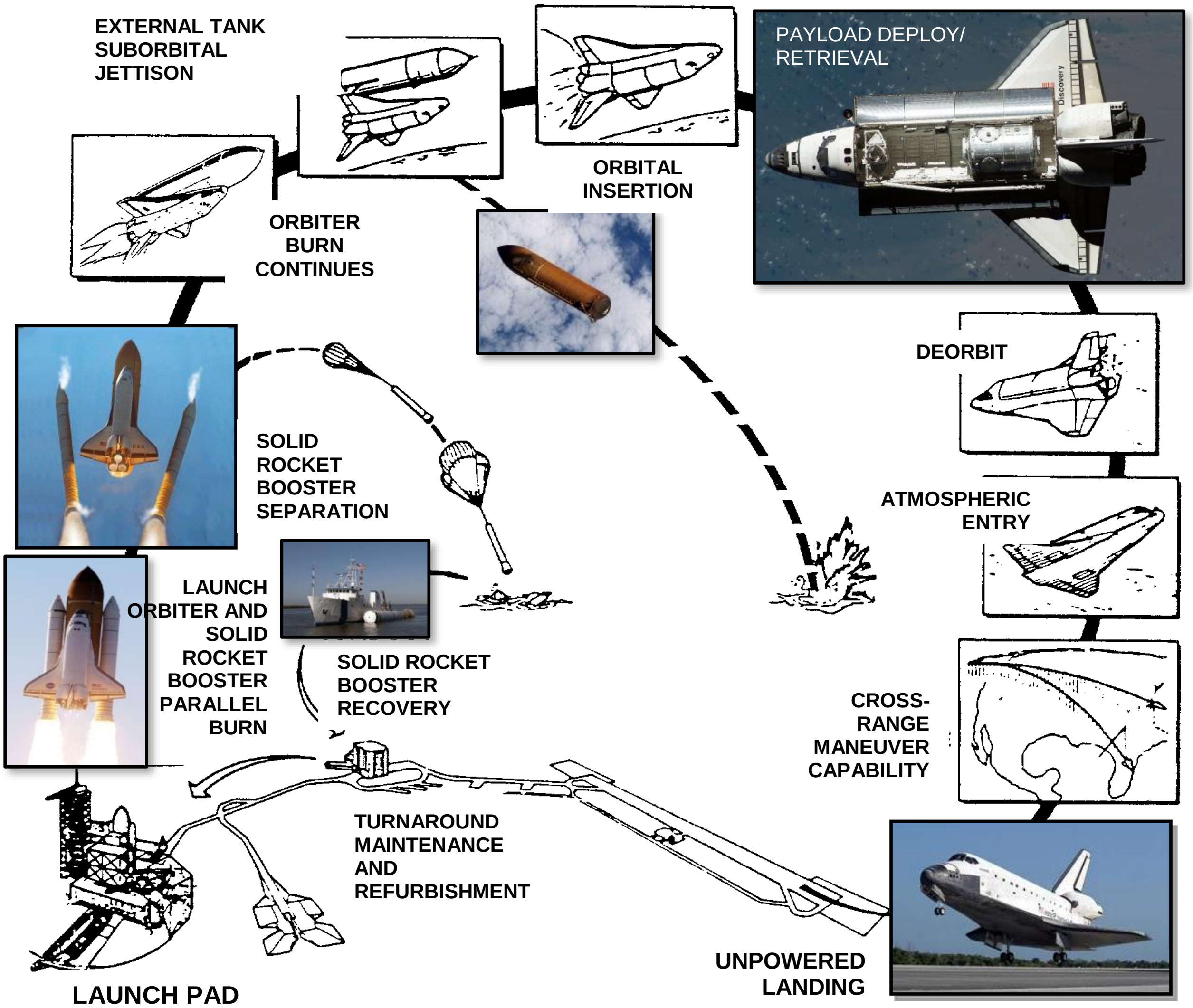

Fig. 1: Mission Cycle of the Space Shuttle

II. CLA Overview – What is it, who uses it, when and why

When a system experiences a loading environment characterized by rapidly varying forces, such as a rocket launch, a transient analysis is used to analyze the response of the system. The most common transient analysis methodology is the Coupled Loads Analysis (CLA). CLAs are used by the automotive and aerospace industry to analyze cars, trucks, planes, helicopters, spacecraft, etc. A CLA solves multiple equations of motion simultaneously to recover loads, accelerations and displacements due to forces applied over a specific flight time.

CLAs can be used throughout the design of a product to develop loads for use in strength and fatigue analyses. For spacecraft, dynamic clearances between different hardware pieces may also be checked. The accurate determination of structural loads and dynamic clearances during all phases of the flight environment is crucial to the development of spacecraft structures (Reference 1). A CLA allows the hardware developer to refine the design by adjusting parameters based on the vehicle-specific load environments. The creation of accurate dynamic models, usually verified by testing, in addition to accurate forcing functions, is required for valid CLA results.

A CLA includes developing the required payload models, verifying that the models reflect the actual payload hardware (to the best ability of the analyst), acquiring the vehicle model, coupling the models together, applying forcing functions to simulate the externally applied transient environment, and solving the equations of motion. CLAs usually require on the order of weeks to complete. As efficiencies in solving the equations of motion were incorporated and computer speeds increased, there were also equal growth in the complexities and sizes of the models and so the time it took to complete the analysis did not changed much over the years.

Oftentimes, various parts of the structure are developed by different entities, such as a launch vehicle and a CE which are built by different companies (sometimes in different countries), and must be coupled together to conduct the analyses. The payload developer usually conducts CLAs for spacecraft during the design process using generic models and forces provided by the launch vehicle developer. The launch vehicle developer will then run a final verification CLA with a model provided by the payload developer(s) and the mission-specific launch vehicle model and forcing functions.

Among the inputs to the CLA are the Finite Element Model (FEM) of the hardware and Output Transformation Matrices (OTMs) provided with the model to recover the desired parameters (usually force, acceleration, and displacement).

III. Lessons Learned and Solutions to Problems Encountered

Lessons learned are presented in no particular order, although a chronological order of the VLA process is somewhat followed. This section will review those problems for which solutions were developed.

A. Developed Early Compatibility Assessment

Problem: The first several large diameter habitable (pressurized) modules being delivered to the ISS exhibited significant clearance issues with the Orbiter, which were not discovered until very close to launch. A near-interference between the Remote Manipulator System (RMS) elbow camera and the United States (US) Laboratory (US Lab) module, as shown in Figure 2, was not identified until the module was actually installed into the Orbiter and a technician noted the small clearance between the camera and the module. All other ISS pressurized modules – Multi-Purpose Logistics Module (MPLM), Columbus, Nodes 2 and 3, and the Japanese Experiment Module (JEM) Pressurized Module (PM) and Pressurized Section (PS) - were built to the same outer dimension as the US Lab but only the JEM PM was long enough to cause a similar issue with the RMS elbow camera. The other modules were shorter and did not reach as far forward as the RMS elbow camera. However, other external items such as handrails, electrical and fluid connectors, and Grapple Fixtures (GFs) on these modules created close clearances with Orbiter hardware including the top centerline latches for the cargo bay doors.

Cause: The defined payload 90-inch radius thermal and dynamic envelope was not uniform along the length of the Orbiter. The Interface Control Document (ICD) defined several items which protruded into the envelope including the RMS elbow camera, wire trays and top centerline latches for the cargo bay doors. CE developers were not sufficiently planning around the hardware that could intrude into the envelope and were exceeding the defined envelope in areas where they thought there was additional clearance. Since the habitable volume of the modules was maximized early in the design, any required external items often protruded outside the allowable 90-inch radius thermal and dynamic envelope or came in close proximity to Orbiter hardware intrusions. The module design process was such that the final as-built Computer Aided Design (CAD) model with all externally mounted items included was not provided until very close to launch.

Lessons Learned:

- Thermal/dynamic envelopes should be defined simply. The more intrusions into the space by the launch vehicle, the more complex the envelope that payload developers must follow.
- Designs of pressurized modules need to account for any additional external hardware volume required outside of the pressure shell when making tradeoffs between the outer mold line and internal volume.
- If a payload wishes to maximize their volume to take advantage of every inch of available space, they must be willing to work with the launch vehicle team throughout the design phase to preclude any late (and expensive) hardware re-designs to prevent inadvertent contact.

- Document actual clearance requirements, whether no contact or a specific clearance, and distinguish from these the values used to determine when more rigorous clearance assessments are required to verify the requirement is met.
- Predicted displacements should be verified through testing, especially if they are deemed critical.

Solution: An early compatibility assessment process was implemented to track the close clearance items as the module development progressed and locations for external items were finalized. These early compatibility assessments followed a process similar to the final VLA process. Thermal and manufacturing tolerance data and design CLA (DCLA) data were provided by the CE developers to the SSP. An initial meeting would be held to verify the data received for use in the compatibility assessment. The SSP would then conduct a loads and clearance assessment based on the DCLA data and the best available CAD model of the CE. Static clearances would be generated from the CAD model of the CE placed into the Orbiter CAD model. Dynamic clearance losses were calculated from the DCLA displacement data. A compatibility assessment review would be held with the CE developers after the SSP had completed the analysis. The SSP presented the data used including the interface load compatibility, the dynamic clearance loss at each point, and the dynamic clearance (or interference) information on all the points. All results were formally documented for later reference during the VLA.

These early compatibility assessments influenced the outer mold lines of the Node 2 and Node 3 developed by Thales-Alenia, the JEM-PM and -PS developed by the Japanese Aerospace Exploration Agency, the Columbus module developed by the European Space Agency, and the MPLM developed by Thales-Alenia for NASA.

Since the cargo bay door Top Centerline Latches were a significant intrusion into the 90-inch radius envelope, modification of the pressurized module Micro-Meteoroid/Orbital Debris (MM/OD) shielding closest to the latches was the most common issue. In many cases, the hardware modifications were as simple as removing handrails and Work-site Interfaces (WIFs) for launch and installing on-orbit. For Node 3, a connector issue required the back shell of the connector to be re-oriented to provide greater clearance to the RMS. When the JEM-PM was launched, a Velcro™ strap was used to restrain the RMS elbow camera in pan and tilt to keep it from contacting the JEM-PM or the Orbiter radiators.

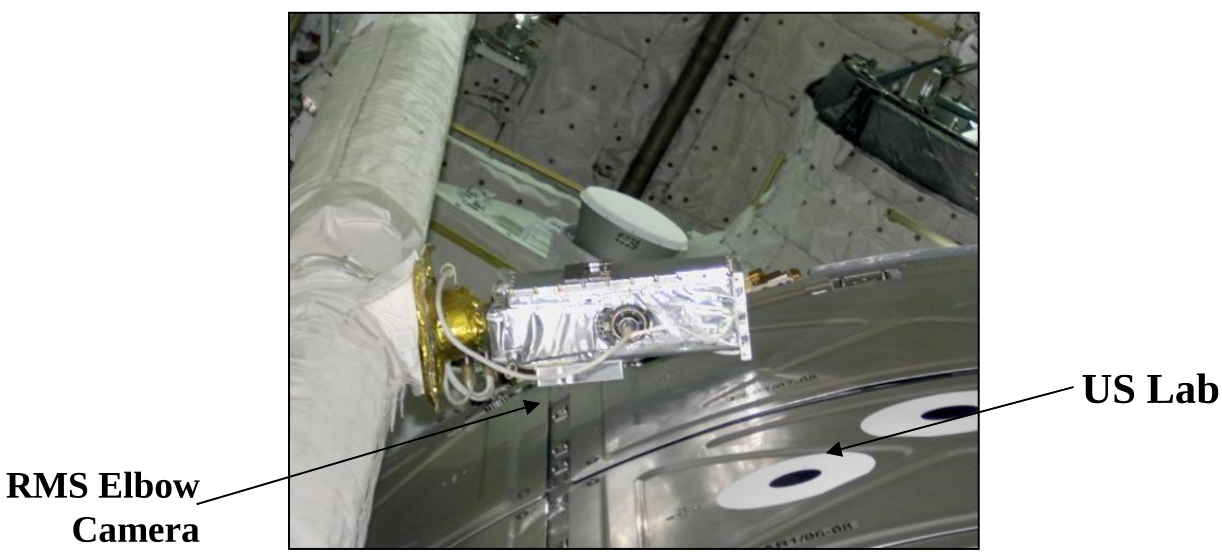

Fig. 2: Near interference between the RMS Elbow Camera and the US Lab Module

B. Automated Selection of Clearance Points

Problem: During the early compatibility analyses, each pressurized module identified a list of close clearance points. This list could easily include hundreds of items, as each handrail, connector and WIF had to be assessed along with other general points on the module. If the pressurized module MM/OD shield panels did not have captive fasteners, the four corner fasteners on each panel would have a retention lanyard installed and each of these lanyards had to be assessed for clearance. For each clearance point on the CE FEM, a corresponding point in the Orbiter FEM had to be selected to determine the relative dynamic deflections. Due the number of points, this process could be quite time-consuming to identify all the corresponding points. Sometimes points too far from the CE point were chosen and/or errors in selection were made.

Cause: The amount of data to be handled was growing with new CAD models and the improved dynamic Orbiter FEM, which provided more choices for clearance points. The ISS Program wanted to minimize the amount of work on-orbit (e.g., Extra-Vehicular Activities (EVAs)) and therefore wanted to fly as many items in place as possible. Only those items with adequate clearance could be flown in place so each external item had to be checked and re-checked for clearances.

Lesson Learned:

- Automate as many processes as much as possible to improve accuracy and turnaround speed.

Solution: An automated clearance point selection tool was developed in conjunction with the improved Orbiter dynamic math model to determine reliable point-to-point relative deflection logic. This improved the fidelity for early compatibility assessments, requests by a CE developer for Space Shuttle Vehicle math models to be used in DCLAs, and for each VLA.

C. Reduced Cycle Time

Problem: VLA cycle time required reduction to align with the goals of the program to support increased flight rates and ISS manifest flexibility.

Cause: The SSP identified several goals in the late 1990's which included increasing the flight rate, reducing the total cycle time, reducing the manifest template, reducing the response time to add additional flights, improving manifest flexibility (i.e., swapping out payloads closer to launch or changing stowage content in larger modules), improving existing levels of safety, and decreasing the operational costs. Reduction of the VLA cycle time and moving the schedule to the right supported almost every one of these goals.

Lessons Learned:

- Invest in structural math model improvements as you go.
- Automate as many processes as much as possible to improve accuracy and turnaround speed.
- Acquire or develop tools suited to the task.

Solution: A reduction in the cycle time was accomplished over several years in a step-down process. The total cycle time was reduced by two months (33%) and the end of the schedule was shifted to the right by half a month. Figure 3 graphically illustrates the reduction steps. The cycle time improvement was achieved by the implementation of process automation by both the SSP and the CE developers.

It was determined at the beginning of the cycle time reduction task that a complete overhaul of the process was not beneficial due to the time and money it would take to develop and implement new software. Since there had been no major process or methodology breakthroughs in recent years, that kind of approach was not justified. Instead, additional automation of current processes was implemented. These automations included report generation, interface force checks against the CE Output Transformation Matrix (OTM), rigid body OTM checks, chart preparation, and latch mass coupling at the CE to Orbiter interface. Automated clearance point selection, mentioned above, was also part of the cycle time reduction. The intent, successfully achieved, was to develop an integrated loads analysis process that eliminated hand-offs, performed internal checking, replaced numerous user utility programs and created a flight unique database.

Another contribution to the cycle time reduction was an updated Orbiter math model. The model used in the VLA at the start of the process improvement was a reduced model that had last been updated in 1983. The updated model, often called the "high fidelity" or "hi-fi" model, was a physical model of the Orbiter that corrected several modeling errors and improved test-correlation in several areas to provide better load application and dynamic response performances (see Table 1). With the improved model, the processes for the selection of close clearance points and the development of mission-specific models were more easily automated.

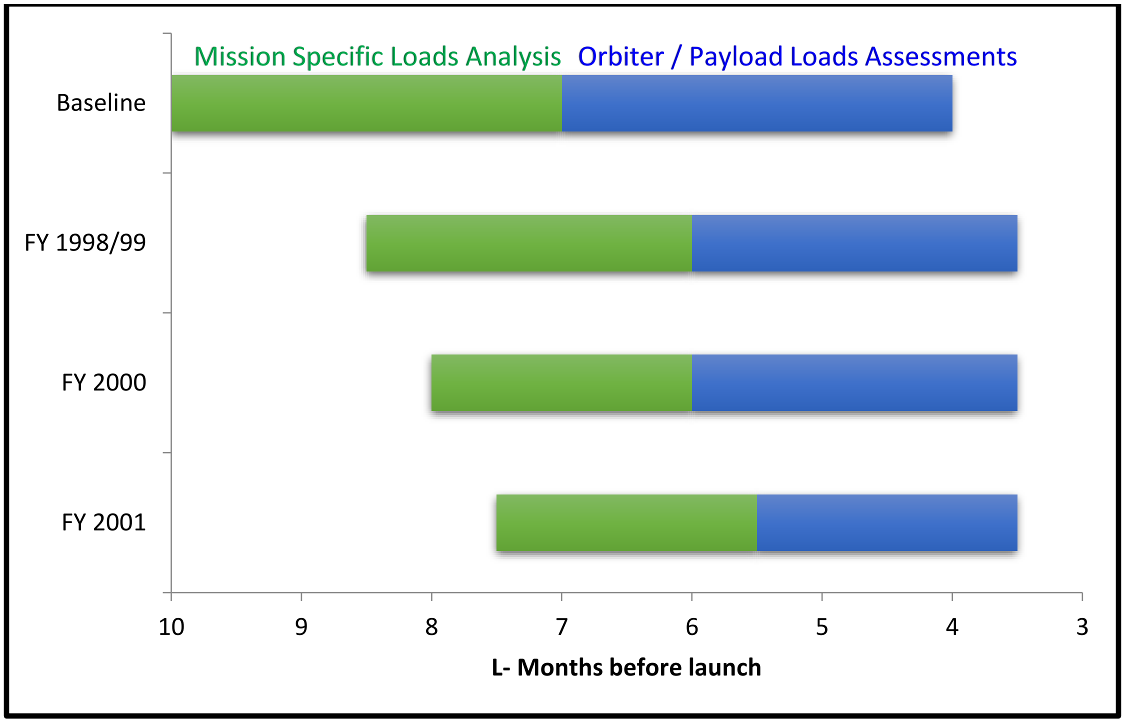

Fig. 3: Structural Verification Cycle Time Reduction

Table 1: Improvements to the Orbiter Model

Remodeling/re-meshing of mid-fuselage
Added heavy mass items under the cargo bay liner
Implemented secondary model updates throughout the vehicle
Revised mid-fuselage stub frames ties to sill longeron
Significantly increased the number of nodes available for clearance checks and FF application

D. Define and Review Contingency Cases Early

Problem: For STS-121, a late contingency case was requested by the Extra-Vehicular Activity (EVA) community after the VLA cycle had completed. For STS-127, additional contingency cases were defined by the NASA Safety Panel after the start of the VLA cycle. Any flight with the pressurized MPLM logistics module also created problems in defining the landing contingency cases as the amount of cargo to be returned was always in flux until the hatch was closed on-orbit. Non-deployable logistics carriers with multiple Orbital Replacement Units (ORUs) that were deployed by the RMS or by EVA changed configurations multiple times on-orbit.

Cause: Contingency cases were defined based on mission timeline and recommendations from the NASA Safety Panels. The mission timeline was used to determine the cargo bay configuration based on CE deployments and EVA reconfigurations. Initially, contingency cases for logistics carriers were defined at the beginning and at the end of an EVA with no interim configurations. At some point, the question of rapid safing during an EVA was discussed with the crew and with the NASA Safety Panels. It was determined that there could be a rapid-safing event during an EVA which would put the carrier in an intermediate configuration not previously considered. With several payloads and/or ORUs on the carriers, this grew to be quite a large number of contingency configurations that were being requested for inclusion in the VLA. Knowledge of the order in which items were removed and/or replaced was necessary at the start of the VLA to properly identify the configurations, as additional models would have to be requested from the CE developer.

In some cases, additional contingency return configurations were recommended based on the failure tolerance of hardware that was removed or installed during the flight. Failure tolerance is defined by the number of failures required before the hardware no longer functions as intended or becomes a safety issue. Unfortunately, these cases were usually identified after the start of the VLA. This continued to be an issue throughout the program due to the relative timing of the VLA and NASA Safety Panel review schedules. Due to the number of configurations that could be assessed within the template, the VLA typically only looked at one failure at a time in determining the contingency cases. On rare occasions, the NASA Safety Panel developed rationale that required a case that was two failures deep.

Lessons Learned:

- Coordinate contingency cases with *all* possible stakeholders and at several different times during the process, making them aware of the impacts to other groups.
- Develop alternate verification methods (and define their limitations) for last minute changes when adequate time is not available for a full analysis.

Solution: For missions with logistics carriers, an initial determination of contingency cases would be made at the start of the process at a meeting called the Cargo Integration Review (CIR) and formalized at the VLA kickoff. These cases would then be presented at either a splinter SSP Integrated Product Team (IPT) meeting or the regular IPT meeting, depending on the meeting schedule. A splinter meeting was eventually added to the CIR just for discussing the contingency cases since a large number of stakeholders were regularly available at that event. Both the CIR and the IPT meetings enabled a wider audience review of the cases and included EVA, Robotics, and the NASA Mission Operations Directorate (MOD). NASA Safety Panel representatives were also usually present at one of the meetings. These meetings opened the discussion for which additional cases may be required or which cases may not be needed due to extremely low likelihood of occurring. All contingency cases had to be defined no later than six weeks prior to math model delivery for the VLA since they often required development of unique CE models to represent the contingency configurations.

It was also important to stress to the larger SSP community how many cases could be accommodated within the given template as every possible permutation of carriers and failures could not be reasonably accommodated within the process.

An agreement between the ISS program and the SSP was established to include up to three MPLM models for return from orbit to cover the potential manifest variations. These models would be spaced based either on the generic allowable mass tolerance of +/- 200 pounds for large CEs or based on a “layered loading” approach to repacking the module, depending on the transfer time identified for the flight. However, in some cases the amount of return cargo identified just before the flight was very different from the modeled manifest (which was based on data 6 months prior to launch) and the execution of a full additional VLA cycle was not possible prior to the flight. This necessitated the use of alternate analysis methods to assess the return configuration. In most cases, this involved having the MPLM integrator run a CLA with an updated model since the landing Orbiter models were usually available to them due to sensitivity studies performed for the manifested racks. The NASA Structures Working Group would then review the CLA data to define any additional Uncertainty Factors (UFs) to be applied to VLA results to conservatively cover the model differences. CLA results generated outside the SSP were not used directly due to the limited verification of the tools used by the contractors in comparison to the benchmarked and configuration controlled VLA tools. This did lead to some conservative UFs; however, it was usually not an issue for the Orbiter interfaces, the MPLM or the racks inside. The results with the UFs were included in the final VLA acceptance review around L-2.5 months.

After all the years of flying logistics to the ISS, only once did the cargo bay return in a configuration other than the planned returned configuration. STS-131 returned without a piece of hardware on an external carrier due to an Extra Vehicular Activity (EVA) taking longer than expected which prevented that task from being completed. That configuration was one of the VLA contingency cases and allowed a quick decision to not return the hardware and still be in a safe configuration for landing. All configurations included in the VLA were documented in the flight rules by the NASA MOD to allow quick decisions when an issue arose during the flight.

E. Define a Pathfinder Approach

Problem: Even with a shift to the right for model delivery, some CE developers were not delivering a correlated dynamic model due to other delays in their schedule. Often, significant changes affecting model dynamics were occurring to the hardware just after model delivery. Due to an on-orbit failure of the ISS Trailing Umbilical System – Reel Assembly (TUS-RA), this ORU was a late addition to STS-121 replacing another ORU that was manifested on the logistics carrier and had already gone through the VLA process. The VLA had to be re-started with updated models for the logistics carrier. On at least three of the last few Space Shuttle flights, the MPLM module mass was either decreased or increased just after the final VLA models were developed and delivered.

Cause: Schedule or technical issues during testing can delay development of the correlated model. In the case of ISS, late manifest definition or even on-orbit failures requiring late manifesting of an ORU, would lead to re-running the VLA. The ISS manifest definition schedule for pressurized module racks and content did not match up with the VLA schedule for model delivery. The ISS schedule had final manifest definition at L-6 months whereas the VLA models were due at L-7.5 months. Even after the manifest definition, models would not be available for six to eight weeks.

Lesson Learned:

- Pre-determined parameter limits are a great way to reduce assessment time.
- If pre-determined parameter limits are not an option, plan for multiple CLAs with the first being the baseline to compare future CLAs against, using an uncertainty factor on the first set of loads to provide margin against later load increases.
- Identify ways to address schedule risks due to manifest changes.
- Match up schedules between different groups as best as possible and identify critical paths.

Solution: In July 1998, the SSP Cargo Integration office hosted a Technical Interchange Meeting (TIM) to discuss alternate loads analysis approaches in an effort to reduce cycle time without sacrificing accuracy of results. Although many methods and suggestions were reviewed at the TIM, there was no community consensus on a best option.

The SSP Orbiter and Systems Engineering and Integration offices used a “load indicator” based method for their vehicle assessments. For a system that is analyzed repeatedly, it is efficient to define a set of redlines that can be used to compare analysis results against. Redlines are those values above which the analysis results should not exceed and are usually based on approximately a zero margin of safety. Consideration was given to apply this approach to the CE developers’ CLA results but defining redlines for payloads that only fly once would be time-consuming and an inefficient use of the CEs resources. Therefore, an alternate method was evaluated.

An approach was developed based on the idea that multiple cycles of the VLA could be run using the first cycle as a baseline. The second and subsequent cycles could be completed in half the time compared to the first cycle since analysis runs would already be set up and the results would be compared to the first cycle which would include a small uncertainty factor sufficient to cover most changes. With the implementation of the uncertainty factor, the subsequent cycle results were mostly or completely enveloped by the first cycle and the amount of work required to assess the subsequent cycles was reduced. This process is similar to the load indicator process of checking loads against a set of redlines. The responsibility for clearing the CE remained with the CE developers as they reviewed the comparison data and addressed any areas of concern.

In the first implementation of this approach, an uncertainty factor of fifteen percent (15%) was applied only to the dynamic portion of the VLA results. This was a complicated application and hard to remove if the CE developers encountered an issue with the loads. It was later changed to ten percent (10%) on all transient loads and five percent (5%) on quasi-static loads and was referred to as the Manifest Uncertainty Factor (MUF).

The first cycle of the VLA was run using the standard template of 4 months from model delivery to Verification Acceptance Review (VAR) where results were shared and reviewed. The CE developers had 2 months to assess the hardware after the SSP program delivered the CLA data. A subsequent cycle was started at L-4.5 months, three months after the start of the first cycle and one month after data delivery. The second cycle was only two months long, with the data provided to the CE developers at about the time of the first cycle Verification Acceptance Review (VAR), which limited the amount of overlap between the two cycles for the CE developers. The second cycle data delivery included a limited set of comparison tables with the first cycle. The CE developers could then focus their efforts on those items, which demonstrated an exceedance over the first cycle, rather than assessing every item as they only had one month in the second cycle to assess their hardware.

After its development, this multiple cycle approach was used by SSP on almost all of the flights. Oftentimes the CEs changed significantly enough between cycles that the comparisons could not be done easily unless the OTM was in the same order as the original delivery. Most of the CE developers eventually automated their assessments of the VLA data such that they just processed the subsequent cycle data with their software rather than reviewing the SSP provided comparisons. Since the SSP provided comparisons were not in widespread use among the CE developers, they were eventually dropped and only provided when requested. The first cycle was still valuable, however, in working through unexpected loads and setting up the software to read the data delivered.

F. Measure Hardware and Conduct Walk-downs

Problem: Early versions of CE CAD models did not include all externally mounted hardware. Multi-Layer Insulation (MLI) blankets and handhold brackets and straps were often not included in the CAD models. Connectors would either be omitted from the CAD model or be located in a position different from the actual hardware. This led to the late identification of close clearances, interferences or snag hazards just prior to or at CE installation into the canister used to transport the hardware to the launch pad. (Canister installation was usually scheduled for one month before flight.) The issue of better CAD model representation had been discussed numerous times with various hardware developers at program boards, but the issue remained unresolved. For STS-120, late identification of unrestrained fastener lanyards at numerous locations on the outer mold line required significant effort to document and certify Node 2 for flight by showing positive clearances to Orbiter hardware. On STS-129, late identification of a blanket not properly modeled required significant effort to approve for flight, including the development of an additional strap to restrain the blanket during flight to prevent snagging. Figure 4 shows a comparison between the Node 1 module as assessed for launch (in green) and the final flown configuration (in red). Additional hardware added to the outer mold line close to the 90-inch radius thermal and dynamic envelope is shown in red.

Thermal and dynamic envelope exceedances were not the only concern when it came to the CE CAD model. The RMS group verified clearances during deployment to adjacent cargo bay structures. Inaccurate models could cause an unexpected collision during payload deployment between the payload and Orbiter structure.

Cause: Early in the design process, not all external hardware may have been known as additional items may have been added as EVA training progressed and the need for additional EVA aid items identified. For items covered in blankets, the blankets and handhold straps or loops are difficult to model. Blankets can be modeled in either a fully billowed condition or some sort of general outline covering the hardware. Although contact between the Orbiter and a blanket may not have been recognized by CE developers as an issue, the documented hard requirement was no contact with the Orbiter. Depending on Orbiter hardware in the vicinity of the blanket, it could cause a snag hazard, even if the payload is not deployed. In general, CE developers appeared to not be properly concerned about a true representation of all items on the outer mold line.

Lessons Learned:

- CAD models should always be verified against the flight hardware.
- All soft goods, especially blankets, should be properly modeled and represented in CAD models to avoid clearance issues and snag hazards.
- Use any and all measurement tools available to verify hardware dimensions on both sides of any interface.
- Blankets should fit as snugly as possible against hardware to alleviate any issues with blankets shifting due to gravity, acceleration and billowing.

Solution: Since the CE hardware was documented in the payload-specific ICD, an ICD walk-down was conducted prior to hardware installation into the canister for transportation to the launch pad. This walk-down confirmed that the CAD model representations in the documentation reflected actual hardware. The walk-down often found blankets, loops, wires, etc. not properly reflected in the ICD. The hardware missing in the CAD model was measured for assessment using either a ruler from a known surface or a laser scanner to develop a point cloud model that could be surface-meshed and placed in the Orbiter model for a more detailed clearance assessment. Although the laser scanner is a great tool for blankets, loops and uneven surfaces, it can take on the order of days to collect and process the data. However, with plenty of notice on items needing measurements, the laser scanner was an invaluable tool in confirming CAD models and verifying clearances between CE and Orbiter hardware.

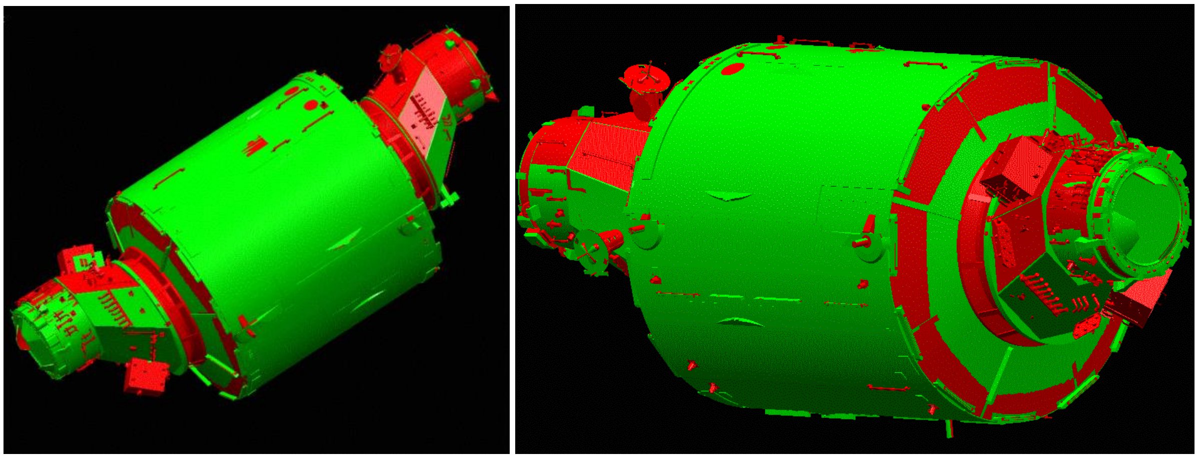

Fig. 4: Comparison of Node 1 CAD Model (in green) and As-flown Configuration (in red) (Two Views)

G. Standardize Mathematical Model Deliveries

Problem: VLA math models were sometimes delivered without all the required information and without proper documentation. Incomplete deliveries often resulted in a delay to the VLA while the data was obtained. Checks performed on received models identified errors in the models that had to be corrected. Models would include coordinate systems other than the Orbiter coordinate system (Orbiter coordinate system was required due to in-house software limitations). All models were required to be delivered in English system units. Orbiter interface latch mass would be included and be incorrect (latch mass depended on the type of latch used to restrain the payload for flight). Documentation, when received, was not in a standard format with information in a different order for each CE developer which made it difficult to process in an efficient manner.

Cause: Sometimes all the required data was not available at the time of submittal or the CE developer was unaware of what needed to be submitted. Component coordinate systems were easier to leave in the component coordinate system rather than translating into Orbiter coordinates. CE developers would not review the Pre-Verification Loads Review meeting charts to verify the proper latch mass. Although the content of the math model delivery was defined in program documentation, there was no specific format or data order for the math model documentation. In general, there was a lack of specific format requirements and data content.

Lessons Learned:

- Be specific about items to be delivered with any math model.
- Any software used for model processing and CLAs should be as flexible as possible for model formats and coordinate systems.
- If a specific unit system is required (input or output), clearly document and communicate that system to all model developers.
- Conversion of OTM items between systems of units should be avoided by creating the OTM only after the model units have been converted.
- Document as much as possible to make it easy for the CE developer to deliver everything needed the first time, correctly and in a standard format.
- Develop checklists for both the CE developer and in-house to verify deliveries include all necessary data.

Solution: A macros-based Microsoft Excel™ interactive form, called the Payload Model Submittal Form (PMSF) was developed to be used by the CE developer and delivered for every version of the CE model that was submitted. Two different versions were developed – one for across-the-bay payloads and one for Orbiter sidewall-mounted payloads.

The form asked the user for all the required model information including points of contact, model description, frequencies, attach points, manufacturing and thermal tolerances, OTMs and OTM equations, latch mass, etc. The program prompted users when an input was invalid or outside the typical range for that value. When all the information had been entered, and only when it was all entered, the form would write out a file that would be sent in with the model. This file made it much easier to manually input all the data as it was in a standard format. With additional time and resources, it could have been further automated to create the necessary file without having to manually copy the data. The form was easy to use but did encounter issues with European users when it came to the decimal versus comma notation for numbers and the software defaults.

Although the PMSF helped organize the data, CE developers were still not delivering all the required data. A checklist was developed by the SSP to help verify that all the data was delivered. That checklist was also made available to CE developers so that they could verify their delivery had all the required information. The PMSF tool and the checklist went a long way in standardizing input requirements for the CE developers' structural math model deliveries and getting all data in on time and error free.

IV. Forcing Function Lessons Learned

Forcing function (FF) updates due to program requirement changes impacted CEs that were far along in their design and analysis when implemented. New loading environment requirements imposed on hardware in development, already designed or ready to launch, caused significant resource expenditure in evaluating new loads, performing additional DCLAs, structural testing, structural modifications, additional stress analysis and on occasion, margin of safety waivers. Forcing Functions were typically developed by the Space Shuttle contractor, benchmarked and formally approved by NASA-JSC for use in DCLAs and VLAs.

Forcing functions can be developed in one of several ways – Monte Carlo analysis, modified two-Sigma Worst-on-Worst or a pure three-sigma approach. The Monte Carlo method is useful in studying systems with a large number of coupled degrees of freedom and significant uncertainty in inputs. However, it is computationally intensive with all the parameters that act on a space vehicle during ascent. CE developers usually are limited in the size of cases they can run in a DCLA. The smaller the number of FF, the quicker a CLA can be completed and designs finalized. A Monte Carlo approach to CLAs for payload developers was deemed impractical.

A two-sigma Worst-on-Worst approach combines various parameters at a two-sigma level to obtain a finite number of forcing functions from which a smaller subset can be down-selected. This was used for early developments of Space Shuttle Cargo Integration FF but always led to the use of supplemental forces to cover flight data.

A. Developed Landing Forcing Function (FF) Database

Problem: Landing forcing functions initially required a Rockwell International (now The Boeing Company) Internal Letter from Cargo Integration to the Orbiter Project requesting a set of forcing functions for a particular payload complement. As Orbiter had their own tasks to accomplish, the FF request fulfillment could be delayed leading to slips in the Cargo Integration schedules.

Cause: In the mid-1980's, the flight rate was increasing as were the number of payloads in development, causing significant traffic in requesting and receiving landing FFs for DCLAs and VLAs. The turnaround became a bottleneck to the schedule.

Lessons Learned:

- Spend resources up front to build necessary environment databases such that reliance on other group/function deliveries is not required and bottlenecks are mitigated.
- If a database is developed, do not lose the capability to regenerate any of the data in the database.
- In the absence of a database, automate the generation of environments to allow different group/functions to utilize the software as needed.

Solution: A database of landing FF was developed in 1986 that allowed Cargo Integration to operate independently in selecting the necessary cases based on weight and center of gravity (CG) of the integrated system. Remarkably, that same database was used through the end of the program with only two instances of additional cases required from the Orbiter project.

For STS-114, a contingency return case using a higher sink rate was requested by the SSP Office but there was no FF for that mass property available. Fortunately, just prior to that, the software that was used to generate the original data base had been located, transitioned to a new computer, and recompiled to generate some lower sink rate (5.0 and 6.0 fps) FFs for the ISS Program for sensitivity cases. It was only because that software had already been updated for a previous request that the STS-114 flight was supported per the SSP Manager's request; however, it was never intended to generate FF on a mission-specific basis. It was used only one more time on the last Space Shuttle flight due to an MPLM return case.

B. Increased Loads Due to Supplemental Forces

Problem: From 1986-1988, a formal Design Certification Review (called DCR-2) was held for the Shuttle elements as an outcome of the Challenger vehicle accident. New Space Shuttle Vehicle structural math models and forcing functions were released to the payload community in March 1988. The new FF contained supplemental forces (tuned sine waves) required to provide an adequate envelope of flight data. At a NASA-JSC meeting in April 1988, the Hubble Space Telescope (HST) program manager announced that the HST "could not fly" due to the large number of issues encountered with the new FF.

Cause: In the development of the new FFs, several benchmark payloads and their flight data were enveloped to ensure limit loading was achieved. In the effort to reach the limit loading, the tuned sine wave supplemental forces (aka "pseudo-forces") were added to the down selected FFs. The supplemental forces required to provide adequate coverage of flight data drove payload component responses to unacceptably high levels on some flight manifests.

Lessons Learned:

- Effective loads relief efforts are only successful with close coordination among all stakeholders. In the case of the SSP, stakeholders included the payload developer, Systems Engineering, Cargo Integration (CI), the United Space Alliance, the NASA Flight Operations and Integration office, and the NASA SSP CI Structure Working Group (SWG).
- Communicate often with the payload development organizations regarding potential updates to launch vehicle math models and forcing functions.
- Each cargo element must evaluate and show positive margins for the latest program approved environments. Do not allow "grandfathering" of previous math models and forcing functions when they are not truly representative of the current environment.
- When multiple flights are encountering loading issues, it is time to re-examine the inputs used to generate the loads. Understand the effects of all the parameters in a forcing function and determine which have the most significant impact on payload loads.

Solution: A multi-month effort by the HST team was undertaken to perform additional component level testing and re-perform stress analyses. Multiple DCLAs were performed as the NASA SSP Office, SWG and The Boeing Company (then Rockwell International) collaborated on modifying the FF at distinct frequencies using logic that eliminated cargo bay accelerometer data that was remote from the HST cargo bay attachments. A final set of acceptable modified FFs was approved through the payload requirements waiver process. The HST was eventually successfully launched and continues to operate thirty years later.

C. Eliminated Supplemental Forces

Problem: When the improved dynamic Orbiter FEM was developed, it made sense to also update the forcing functions. With the new FEM, there were more locations to apply forces than the previous model, which would better represent the environment and it was desired to remove the supplemental forces which could drive cargo element responses. During re-development of the FF for cargo integration use, it was found that only a few parameters influenced the cargo responses (Reference 2): Solid Rocket Booster (SRB) Ignition Overpressure (IOP), Space Shuttle Main Engine side loads, and IOP timing.

Cause: For the CI high fidelity Orbiter model, the enhanced fidelity in the math model and the increased number of IOP force application points resulted in cargo element responses that were dominated by IOP forces. The updated Orbiter modeling corrected the errors for distribution of loads and the relative influence of the forcing function changed the response characteristics. Only a few liftoff parameters were found to be important to the overall cargo bay dispersions from nominal.

Lesson Learned:

- Spend time understanding the effects of all the parameters in a forcing function and determine which have the most significant impact on payload loads.

Solution: The two-sigma approach used in all previous FF developments for CI was abandoned in favor of a new limited parameter three-sigma approach. Further investigation into the details of the FF revealed that there may be better ways to simulate the environment. For the high-fidelity Orbiter model, the parameter influences on cargo responses was extensively examined. Forcing functions were developed with cooperation across the SSP based on a three-sigma SRB IOP response. Innovative filtering and scaling techniques were implemented to eliminate the previous narrow band sine wave supplemental forces necessary to envelope flight data at particular frequencies. One two-sigma dispersed case was included to improve flight data coverage in the lower frequency range. This gave a more realistic model of the launch environment than the supplemental forces necessary in the previous set of forcing functions used to envelope flight data.

D. Forcing Function Database Limitations

Problem: Changes in the landing Orbiter system mass and CG, which fall outside of the defined forcing function mass and CG limits, such as those due to altering the aft ballast, could result in the necessity of re-running landing cases.

Cause: For landing, a set of FF was developed that was based on the system weight and CG of the Orbiter plus any CEs. A database was established based on a 5,000 pound/four-inch mass and CG tolerance range for each FF, over the allowable range of weight and CG for the Orbiter system. The SSP used two sink rates for payloads – one at 9.6 feet-per-second (fps) for nominal returns at lighter weights and one at 7.2 fps for aborts, contingency returns and heavier return weights, each with a 20-knot crosswind.

The forcing function database blocks, once developed, were never updated. There were more forcing functions for the lower sink rate since heavier Orbiters were not required to return at the higher sink rate (which could blow-out a tire or collapse the landing gear). Whenever a VLA was run, the appropriate FF were selected and used. However, a small change to integrated weight and CG could require a different forcing function if it moved outside of the "box" since there was no overlap between cases. This limited the change in mass properties before a VLA would have to be re-run.

Sensitivity studies were conducted to determine the impacts of using the wrong FF for a particular weight and CG range. It was found that the load changes were significant, even if the system was at the edge of the box.

Lessons Learned:

- Forcing functions should be based on the mission-specific mass properties rather than on a standard set of weight and CG categories ("bins" or tolerance ranges) to allow greater flexibility.
- If a database of set cases must be used, the impact of going outside the box should be well understood to determine the allowable tolerance of the system before re-analysis is required.

V. Lessons Learned but Not Implemented

These are lessons that were never implemented due to time and budget constraints. Many of these are more generic than those in the previous sections. Although solutions were never implemented, the issues were worked around. These issues still exist today for programs and the lessons learned still have relevance.

A. Software Tools

Problem: Software developed to support the SSP VLA, including model processing and running the CLA, was written using custom in-house code rather than using commercial software. Over the years, issues arose as computer platforms changed and programming languages advanced. Due to the age of the software at the end of the program, it was becoming difficult to find a system on which the compiled code could run. If recompiling was required, it was going to require extensive software updates as compilers in the original language version were no longer available. As computers and computer languages changed, keeping the software running increased in difficulty.

Cause: The automated VLA software used the Computer Software Management and Information Center (COSMIC) open source NASA Structural Analysis (NASTRAN) software. While using the open source NASTRAN software allowed greater freedom in adapting the code originally, it also required someone very familiar with the code to be available at all times to address any issues or errors found as there was no "help desk." As computer systems aged, the available platforms available to run the code without re-compiling became severely limited. This limitation also led to the risk that the computer would crash and not be recoverable.

A backup set of software based on the commercially available MacNeil-Schwendler Corporation (MSC) NASTRAN was verified and validated but did not include all the automated tasks programmed into the baseline software.

Lessons Learned:

- In-house software requires resources throughout the life of the code to upgrade and test for compatibility as computer systems and operating systems are upgraded.
- Tools more heavily based on commercial codes are easier to upgrade and port to other systems.
- Investment in the automation of repetitive tasks is well worth the time and money for when analyses must be repeated.
- Rigorous benchmarks/run matrices must be defined and executed to ensure the correct results when software and computer platforms change.
- Cooperation and constant coordination among management, analysts, analysis tool providers, operating system providers, and computing platform providers during software or computing platform changes are imperative.
- Any tool issues must be diligently tracked and resolved quickly.

B. Mass Properties Management

Problem: For across bay CEs and payloads, the SSP had a mass property tolerance of two hundred pounds and one-inch root-sum-squared (RSS) on CG between the VLA math models and the actual hardware. Orbiter sidewall-mounted payloads were limited to fifty pounds and one-inch RSS. Final weights for CEs were sometimes found to be outside of the tolerance requiring additional work or even additional VLA cycles. On STS-97, the large Integrated Truss Segment (ITS)-Port 6 (P6) had a mass discrepancy of three hundred fifty pounds and a three-inch CG shift. On STS-110, the ITS Starboard-0 (S0) had a mass discrepancy of two hundred eighty pounds. Both flights required additional VLAs to address the discrepancies. Several flights later in the program required additional VLA work due to late changes in the manifest of the MPLM resulting in either a heavier module or a lighter module than was provided for the VLA.

Cause: For the ITS-P6, the hardware was not weighed during interim build stages and therefore it was very difficult for the CE analysts to determine where the discrepancies occurred. For the ITS-S0, a joint agreement at the Joint Mission Integration Control Board between the ISS Program and the SSP had the ISS Program tracking their mass properties on the large trusses more diligently than they did for ITS-P6. In exchange, the SSP was not levying a MUF on the integrated truss segment VLAs. Since the STS-110 VLA did not include the MUF and the mass properties were violated, a new VLA was required to address both the model change and an unanticipated increase in Orbiter propellant.

The MPLM was a logistics module whose mass properties were constantly changing until hatch closure. Models were built on estimates of launch and return hardware. Sometimes that hardware was not actually manifested, and sometimes more hardware was manifested than was originally planned. Since the ISS Program goal was to be as flexible as possible allowing changes up until hatch closure, it was almost impossible to identify the final configuration prior to VLA model delivery. Hatch closure for the MPLM was typically only 6-8 weeks prior to flight. Return flexibility often required real time analysis during a flight to address return configurations that exceeded the VLA tolerances.

Lessons Learned:

- Integrated hardware should be weighed at various stages of assembly and piece parts weighed when available integrated hardware cannot be measured.
- Mass property verification should be scheduled at various milestones during design and assembly, and the models updated as required to match the actual hardware as much as possible.
- Early analysis to expand the tolerance on hardware that will be flown multiple times should be investigated to minimize re-work and increase flexibility (and therefore reducing the amount of real time mission work).

- Low margin, light weight designs require more sophisticated and costly analysis efforts than robust structures when it comes to manifest flexibility.
- Conduct final verification close to launch to incorporate as many changes as possible and to reduce exposure to additional changes.

Solutions Investigated: Early in the Shuttle program, Spacehab (SH) took the initiative to do some up-front work to increase the mass and CG tolerance on the SH Logistics Double Module (DM). The Logistics DM was a pressurized module built from two single SH modules. Stowage in the module was in racks, in lockers on the aft and forward bulkhead, or in bags strapped down to the floor. Working with the SWG, they increased their tolerance from two hundred pounds, one-inch RSS in CG to six hundred pounds and three inches RSS on CG. They accomplished this through a parametric sensitivity study in which the stowage options were varied, and loads compared. All racks were still limited to the fifty pound/one-inch RSS tolerance. The initial design of the module was very robust which helped tremendously in the assessment.

In July of 1998, a memo was published by NASA-JSC regarding the SH DM Flight Experience (Reference 4). With the expanded tolerance, it was believed that some of mission specific VLAs could be avoided. According to the memo, this avoidance of VLAs was never achieved because the outlined criteria, such as mass and CG being within tolerance of a previous flight or the cargo bay manifest being identical, were violated. One of the SH structural analysts noted, "The logistics missions created complex problems, associated mostly with the desire for unlimited manifest freedom. We were forced to react to these changes, sometimes several iterations during the same mission." The same comment could be applied to the ISS logistics module, the MPLM.

A frequent flyer study was conducted in 2000 through 2002 for two different Shuttle manifests that were expected to fly multiple times - the SH Logistics DM and Integrated Cargo Carrier (ICC), and MPLM and ICC. The goal was to increase the mass tolerance to fifteen hundred pounds for the pressurized modules. Of the three CEs in the study, the SH DM alone was able to define a process to react to changes in the co-manifested carrier (Reference 5). New loads for design would be established and only those hardware items sensitive to the co-manifested CE would be verified. The MPLM variations (16 rack locations with at multiple variations of racks) proved to be too numerous to provide any useful data for defining a new mass tolerance. The ORUs on the ICC were too varied in their mass, CG and frequency to also be able to define anything useful. The final results for both of the manifests were inconclusive; however, the assessment of the interfaces between the manifests and the Orbiter showed no concerns for the Orbiter side of the interface. Unfortunately, after the study was completed, the DM was deleted from future Shuttle manifests.

C. VLA Scope

Problem: The SSP was often asked to run either multiple VLA cycles or multiple models in the same cycle. This changed the final verification analysis from the mission-specific analysis, as originally intended by the SSP, to a process to perform mission planning "what-if" and "highly desired" cases covering a range of differing scenarios. After the VLA cycle time reduction was implemented, more analysis cases were being added to the process, which increased the time to run the analyses. These additional cases added up to CLA cases numbering in the dozens with multiple launch configurations. These changes impacted other groups that used the payload developer models.

Cause: Variations were often identified late due to delayed manifest definition. Potential impacts to other areas of flight and vehicle analysis (such as flight design and Systems Engineering and Integration's Flight Margin Assessment) were not as obvious as those to potentially affecting the VLA process. Late manifest changes often impacted these other disciplines as well. However, these other groups were generally not set up to handle large, last minute changes outside of the cargo integration mass properties tolerance of two hundred pounds and one-inch RSS on CG. These other areas would often then become the critical path in the schedule.

Lessons Learned:
- Understand the limitation of all areas of the vehicle performance affected by mass changes and have them clearly documented, including impacts when violated.
- Verification cycles are to be used for final verification of a manifest, not for conducting variational analysis.
- Define early in the program a "reasonable" limit on the contingency and alternate configurations to be run in the final verification, as some may be needed.

- For each payload or manifest, define a plan on how to address additional variations prior to the final verification cycle.
- A database of all results obtained is useful to study trending and sensitivity, based on different manifests.

D. Data Transmittal

Problem: Sharing output data from the CLA with other groups became difficult. The size of files to be shared was larger than what electronic mail allowed at the time (10 megabytes).

Cause: At the beginning of the SSP, dynamic math models were not very large. Models that did not fit on floppy disks were transmitted via computer tapes. As computers got faster and software improved, dynamic models became more detailed and the file sizes to be shared increased significantly. If the CE developer requested time histories from the VLA, file sizes were huge (for the time, on the order of gigabytes). Large files need to be shared but often distribution must be limited due to sensitivity of the data or proprietary issues. Since file transfer protocol sites are usually anonymous, files placed there were not considered secure.

Lessons Learned:

- A system for sharing large files must be established early for secure and efficient sharing between all participants, whether domestic or international.
- The organization performing the CLA should always follow-up with CLA results recipients to verify data was received to mitigate any evaluation delays.

E. Formal Documentation

Problem: During the Return to Flight investigations after the STS-107 Columbia accident, supporting documentation for various analyses and waiver rationale could not be found.

Cause: Simple problems may easily be solved and therefore not formally documented, but when that solution is needed months (or even years) later, it is not there. No one may remember how they arrived at the simple conclusion of a problem and it will have to be re-analyzed. When trying to research an issue after the fact, there is no substitute for formal documentation of the process to solve the problem and the results.

Lessons Learned:

- Document all analyses, no matter how small – a short memo may be all that is needed.
- Documented analyses allow for peer review, which can catch errors, omissions or question any underlying assumptions used.
- For any and all violations of requirements, rationale should include a reference to the analysis done in support of acceptance. This is very helpful when similar issues arise that need to be assessed.
- Data specified in the requirements should be traceable to a documented analysis.

F. Clear Communication between Programs

Problem: Operational vehicles require flexibility in rapidly responding to on-orbit failures. As a result, the SSP and ISS went through a process of how to respond to Launch-on-Need ORUs. Initially, the first ORUs to be launched based on need in this new paradigm had not been through all the proper structural verification reviews prior to inclusion in a VLA. The Launch-on-Need carriers imparted different loads during launch and landing than the cargo element they initially launched on. When data was provided from the VLA, the assessment time for the ORU was longer than the allowed time in the template. Multiple meetings were required to review all the structural assessment data, as it was completed in phases, starting with clearing launch loads first, and then return loads on the failed ORU.

Cause: The first ORU of this new paradigm was manifested much earlier than anticipated and all the data for its assessment was not available. The VLA was conducted with a reduced cycle time, limiting the time available to assess the loads. There was reluctance by the hardware owner to share data in a timely fashion rather than trying to work jointly with the SSP. The Joint Structures Team, whose purpose served to work this type of joint issue was under-utilized. Data briefed to one set of program boards was not the same as presented at other meetings. Lower level reviews and approvals by advisory boards were taken as acceptance without a full review by management.

Lessons Learned:

- Clear roles and responsibilities between the payload provider and the launch authority must be established when a Launch-on-Need ORU is manifested. This is applicable to any late manifest changes.
- Issues and late manifest items must be worked jointly between programs which includes complete transparency and sharing of information in a timely fashion.
- Readiness of the Launch-on-Need hardware or late manifest items needs to be clearly communicated to determine the open work and a supportable schedule.
- A project manager or project lead as the single focal point to both programs is needed to verify data is shared equally and to keep track of what approvals have or have not been given.

G. Instrumentation

Problem: Structural instrumentation in the cargo bay for the Orbiter vehicle was very limited. On occasion, one particular accelerometer would have to be removed due to interference with a payload attach location and then reinstalled prior to the next flight. Since this accelerometer was the only one in that particular axis, flight data was not obtained for post-flight assessment in that axis. Instrumentation for CE responses was not readily available.

Cause: An accelerometer manifested at the keel in Bay 6 of the Orbiter cargo bay had an interference with the keel bridge when it was installed. On those flights, the gage would have to be removed for the flight and then re-installed post-flight after the bridge was removed. The cargo bay of three Orbiters (OV-099, OV-102 and OV-103) had less than a dozen uniaxial accelerometers for post-flight analysis of the payload environment. After Challenger (OV-099) was lost in 1986, only two remaining Orbiters contained accelerometers which could be used for post-flight payload analysis. Columbia (OV-102) was lost in 2003 which left only one vehicle with cargo bay instrumentation.

Providing CE instrumentation was always problematic. Wired instrumentation could only be used on non-deployable CEs, and the Space Shuttle on-board data system wasn't designed to capture additional instrumentation data. Limited stand-alone instrumentation was developed by NASA and used on many flights. However, there were issues with the unwieldiness of the hardware, battery life, activation and data retrieval. CEs had to dedicate mass and volume on their hardware to hard mount (four bolts) the hardware and it had to be accessible late in the flow. Reprogramming was through a small port and was not wireless. If the hardware was not oriented properly, the gravity switch would not trigger, and data would not be recorded. Often, false triggers were recorded and on occasion, the hardware would not trigger at all. Since all the units were stand-alone, there was no time synchronization between the units.

Lessons Learned:

- Verify that instrumentation locations are not in conflict with other hardware required to support the flight.
- Manifest adequate instrumentation so that in the event that one gage is unavailable, all the data for that axis is not missing.
- Develop and manifest instrumentation which is easily replaced and repaired, with downloadable data and wireless programming.
- Develop autonomous instrumentation that can be relocated to allow a greater range of data from flights, depending on the manifest.
- Create a database of flight data which allows for searching depending on the manifest and other pertinent information.
- Adequate flight data must be obtained to create a flight experience database, which can be used to indicate whether a particular flight was within the flight experience.
- Flight data is useful to assist in the development or evaluation of new launch vehicle math models and forcing functions.
- Flight data is invaluable for modifying forcing functions or relieving loads on a case by case basis (with an adequate database of flight experience).
- CE mounted instrumentation is useful for determining CE responses across the interface with the launch vehicle.
- Data should be reviewed quickly after a flight to determine if any anomalies occurred. The same data can also be used as forensic evidence in the event of known anomalies.
- It is easier to show that CLA analysis envelopes the flight data, than it is to show accuracy of the CLA.

VI. Conclusion

The Space Shuttle Program had been performing CLAs to support payload and vehicle flight load environment verification assessments since STS-1 in April of 1981. Over the thirty plus years of the program, computers and modeling had improved significantly. Some of those changes caused problems and others improved the process. The final ten years of the program saw more international cooperation and a desire for "faster, better, cheaper" assessments. The final verification assessments of the mission specific Space Shuttle manifest saw cycle-time reductions, refinement of assessments to assist domestic and international partners in utilizing all available space in the cargo bay, an increase in the contingency return variations, documentation and methodology improvements and achieved significant manifest flexibility.

Although the unique nature of the reusability of the Space Shuttle will not be seen again for several years, if ever, these lessons learned can be applied to all systems, aerospace and non-aerospace, that utilize CLAs in their verification process.

Dedication

This paper is dedicated in memoriam to Mr. Christopher Rodgers/Boeing Space Exploration, Huntington Beach, CA. Mr. Rodgers contributions during the Space Shuttle Program are immeasurable as a colleague, mentor, and a learned practitioner.

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This set of lessons learned could not have been assembled without the help of those that have been mentors and colleagues to the authors over the years. It is their extensive knowledge of past issues, how things used to be done and the improvements implemented (or not) over the years, which the authors did not wish to be lost at the end of the SSP. These individuals include but are not limited to: Mr. William Walls/Boeing Space Exploration, Houston, TX; Mr. Nelson Fox/Hamilton Sundstrand Corporation Houston, TX; Mr. Bobby Brown/NASA-JSC (retired); and Mr. Paul Romine/NASA-JSC (retired). Additional help and guidance were provided by Mr. Raymond Nieder/NASA-JSC (retired).

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