

Analysis Methods used on the NASA Composite Crew Module (CCM)

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The process in use by a national team from NASA research centers and industry partners for the analysis and design of a full scale composite crew module is presented. Methods, failure analyses, and design sizing approaches used by the team are identified. The HyperSizer® software coupled with FEA was used for trade studies of crew module shape configurations, material selections, sandwich panel dimensions, and composite layups and is being used for final analyses and stress report margins-of-safety. Different types of analyses were performed throughout the design maturity and their impacts to the evolving design discussed. These analyses include shell and backbone panel buckling, composite strength to damage tolerance criteria, laminate interlaminar shear, ply drop-offs, sandwich panel specific failures such as facesheet wrinkling, core shear, and core crushing, flat wise tension, core taper ramp downs, and adhesively bonded joints and bolted joints.

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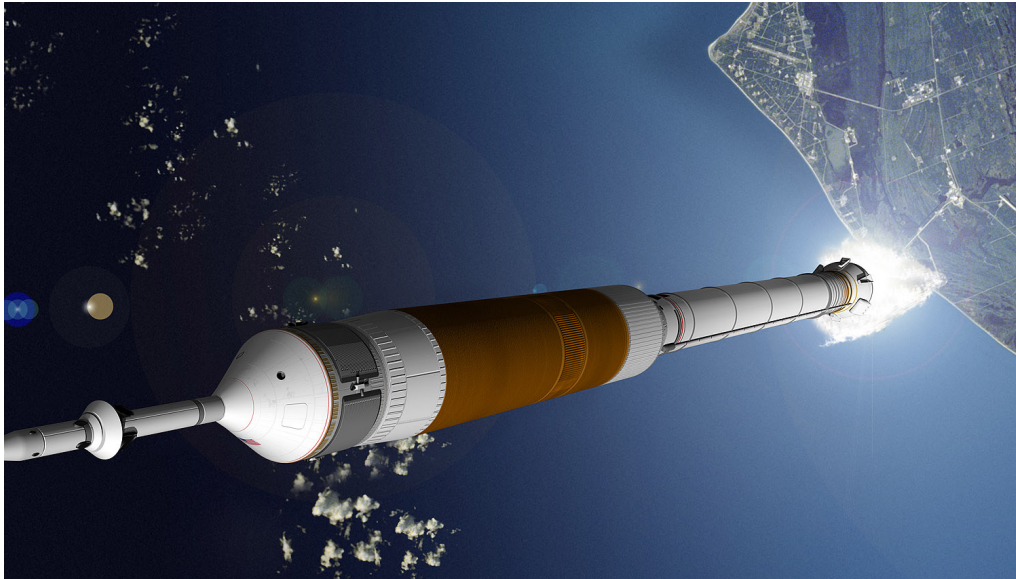


Fig.1, Ares I Crew Launch Vehicle that launches the NASA Crew Module.



Fig.2, The Crew Module (CM) interfaces with the Service Module (SM) and the Launch Abort System (LAS).

1 Introduction

Fig. 1 and Fig. 2 depict how the NASA and Lockheed Martin metallic crew module (CM) referred to as Orion will be launched and interfaced with other hardware modules. Fig. 3 illustrates the primary structural assemblies of the crew module. In a parallel effort, the NASA Engineering and Safety Center (NESC) is designing a full scale composite material crew module. The composite crew module (CCM) is being designed to the same loading environment, baselined mid year 2007, as the metallic crew module. A primary intent by NASA is to gain experience designing, analyzing, and testing flight weight composite structures for potential future space missions [1,2,3,4]. Fig. 4 introduces the CCM and the figure insert in red shows how the pressure shell fits into the existing aeroshell.

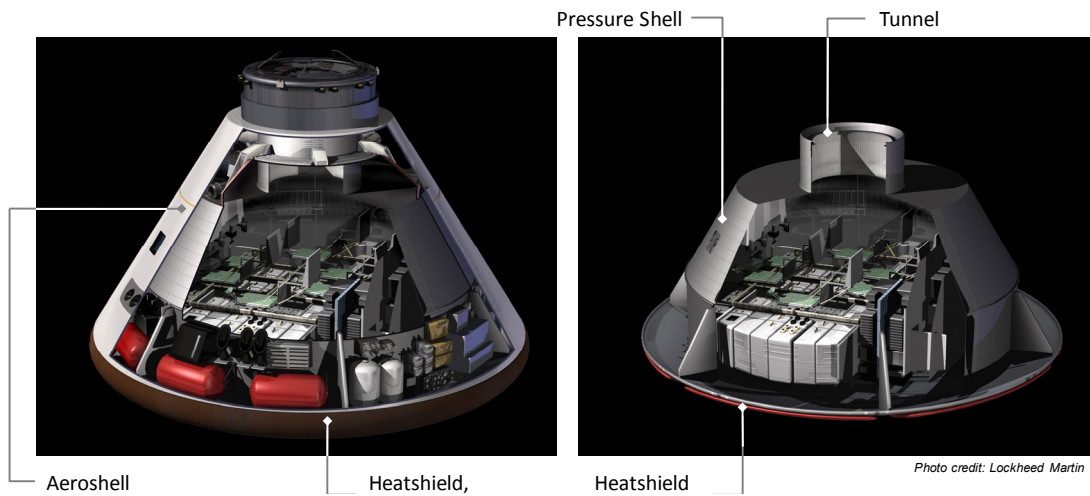


Fig. 3, Left – metallic crew module with the aeroshell and heatshield shown. Right – a cutaway view of the metallic pressure shell and heat shield carrier panel.

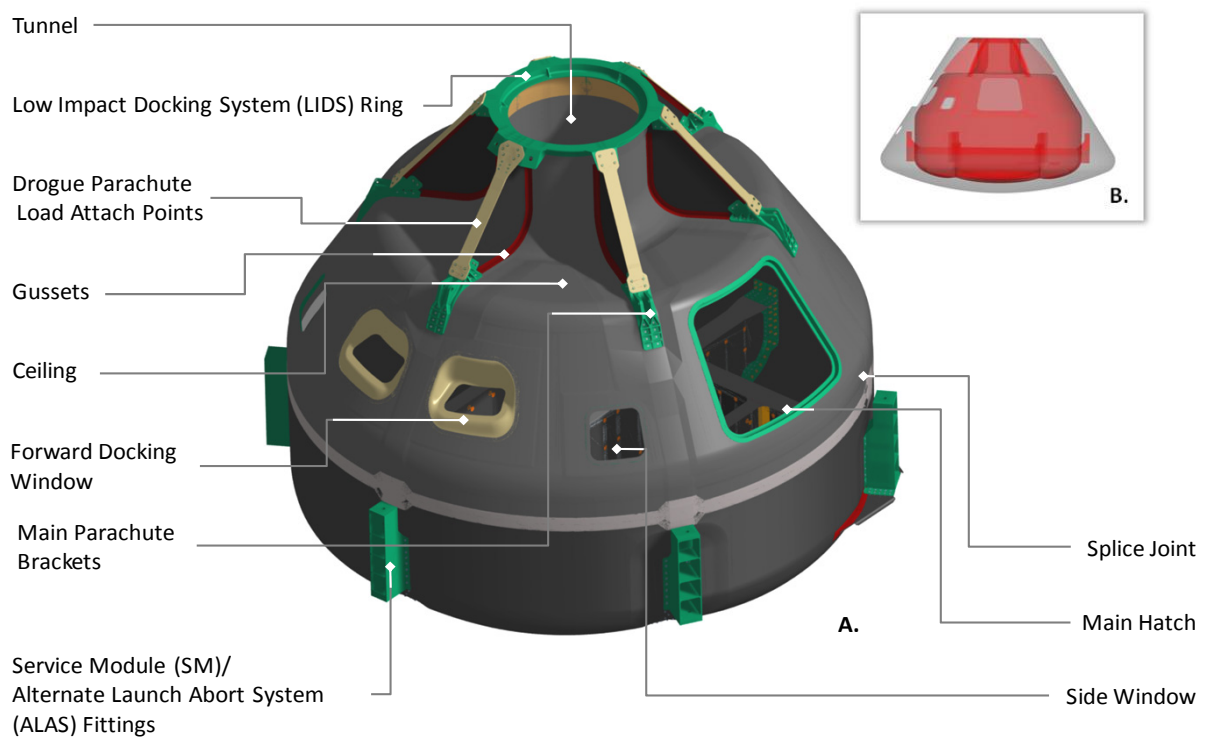


Fig. 4, The Composite Crew Module project is an alternate design of the pressure shell and maintains the essential design intent and interfaces of the baseline crew module illustrated in Fig. 3.

2 Loadings

The range of loadings is:

- Launch
- Abort
- Cabin pressure
- Trans Lunar Insertion
- Reentry (parachute forces)
- Landing (water impact landing)
- Ground support

Throughout the mission, these various loading environments present a challenge to designing robust light weight structure. In essence, the crew module is fundamentally a pressurized shell that has to be designed for two times atmospheric pressure while also withstanding large inertial body forces, concentrated parachute forces, and impact landings.

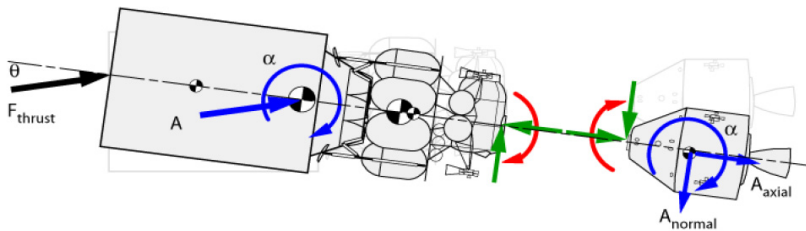


Fig. 6, Trans Lunar Insertion loading from the earth departure stage (left) causes substantial externally applied bending moment to the crew module (right) through the tunnel structure.

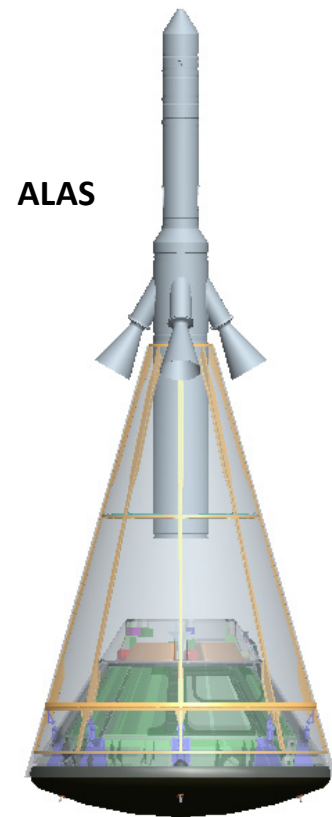


Fig. 5, Pad abort and high altitude abort thrust forces from the Launch Abort System causes high concentrated forces on the fittings and inertial forces throughout the pressure shell and particularly in the backbone.

Ground support loads were not considered. Launch loads were determined not critical. Land landing loads were considered early in the project but were later removed. The fundamental primary load case is the internal cabin pressure which is required to have a 2.0 load factor.

Other fundamental load cases are water landing and high altitude abort, Fig.5, for the lower pressure shell. For the upper pressure shell trans lunar insertion, Fig. 6, and parachute loads, Fig. 7, are critical.

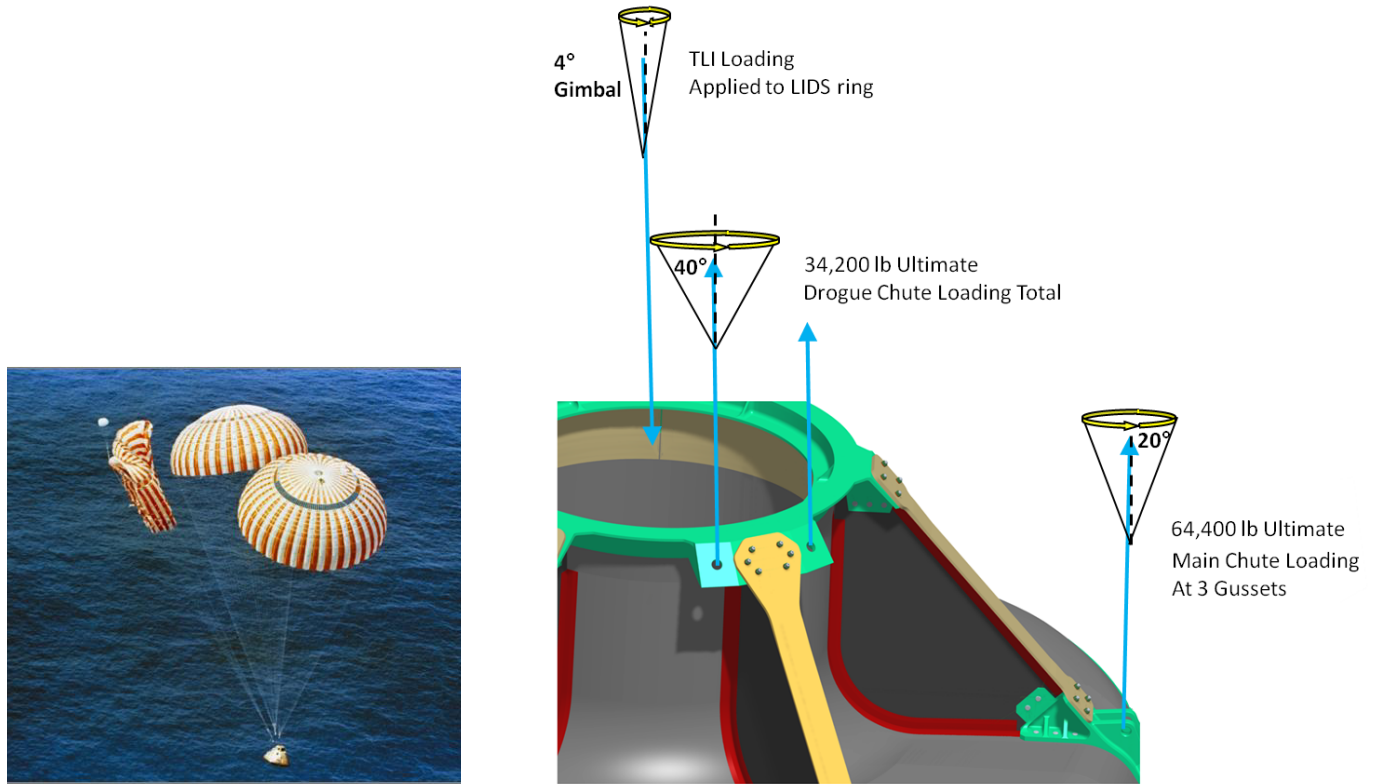


Fig. 7, Water impact loading and parachute concentrated force loadings. The drogue parachutes have a 40 degree angle of application in any direction, represented by a cone. There are three main parachutes that are limited to a 20 degree angle of application. Also portrayed is the 4 degree range of trans lunar insertion forces imparted by the earth departure stages (EDS) on the low impact docking system (LIDS) ring.

2.1 Load Factors

Table 1: Load Factors

Factor description	CxP 70135	CCM CDR
Ultimate for uniform areas	1.4	1.4
Ultimate for discontinuity areas	2.0	1.4* η
Qualification test factor	1.4	1.4
Fitting Factor	1.15	1.15
Internal Pressure	2.0	2.0

The proposed effective factor of safety (FoS) approach can be expressed as:

$$FoS_{\text{effective}} = FoS_{\text{uniform}} * \eta$$

η is an analysis specific correlation factor [6 7,8,9], which, for the CCM, a typical value may be = 1.15. The η value is intended to be based on analysis correlated failure predictions and/or

higher level building block data. However, if this supporting test data is not available, then η starts at 1.43, such that $(1.43 \cdot 1.4 = 2.0)$.

3 Trade Studies

Since the cabin pressure load case is a fundamental load, early attempts to optimize the overall shape of the pressure shell focused on this primary loading. As seen in Fig. 8., different OML shapes were attempted for the aft dome (floor) by using geometric stiffening to more efficiently support the large pressurized surface. Classical domed surfaces, Fig. 8(a), have long unsupported spans and develop large biaxial membrane and bending moments at the shoulder radius. One of the first attempts at designing a more efficient floor was a geometric pattern referred to as the plastic coke bottle Fig. 8(b) [ref. Ian Fernandez]. Though well suited for homogenous materials, the sharp edges developed detrimental interlaminar shear stresses in the layered composite material. Fig. 8(c) depicts the OML shape that evolved. This design intent is to define a shape for the OML that would support internal pressure in membrane tension while dramatically reducing pressure-induced panel bending moments.

Other trades were performed early in the design cycle to quantify the impact of differing loading scenarios. Two trade studies of notable interest are briefly described. The first was load sharing between the TPS carrier panel and the crew module backbone. The issue is the high loadings caused by water impact and spanning these pressures across a large unsupported carrier panel. See Fig. 3. Incremental load sharing studies were performed to quantify the optimal amount of load to transfer directly from the carrier panel into the backbone vs. load spanned to the perimeter and taken out in the LAS/SM fittings. It was determined that a 50% load share was optimal. A similar study was performed to quantify the weight impact of reducing the rocket thrust angle from the earth departure stage during a trans lunar insertion maneuver, see Fig. 6. These trades were performed using HyperSizer as described in the following sections.

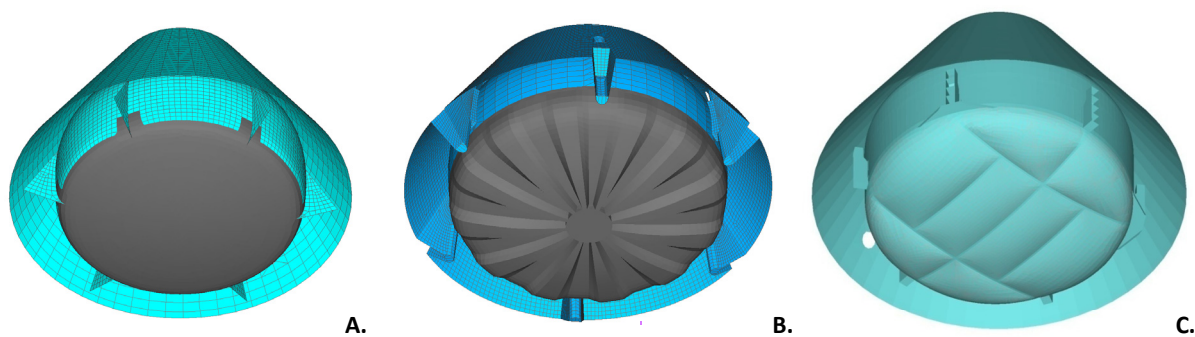


Fig.8, Different composite aft dome floor architectures shown in gray color: a) conventional OML shape, b) plastic coke bottle geometrically stiffened shape, and c) the selected lobed membrane shape which attaches to the backbone configuration. Composite materials permit the convenient fabrication of these geometrically efficient shapes.

4 Scope of Analyses

An early design decision was to baseline honeycomb sandwich panels comprised of composite facesheets and aluminum core. This panel construction forms the acreage area of the pressure shell. In areas of high out-of-plane forces, and areas of metal attachments and fittings, the constant thickness sandwich panel ramps down with a core taper to solid composite laminate. The scope of the different required analyses is identified in Fig. 9.

Acreage area analyses include laminate strength and sandwich specific failure analyses such as facesheet wrinkling and core shear strength. Detailed analyses include interlaminar shear and flat wise tension and bonded and bolted joints. Like many structures, the structural integrity of the CCM is highly dependent on the strength of the joints. The CCM has many crucial joints that are quite large. The fabrication process is to build and autoclave cure the upper and lower pressure shells separately. They are attached by a non-autoclaved splice joint that covers the perimeter of the mid section. Another primary bonded joint uses pi preforms to attach the lobed floor to the backbone. In this area of the backbone are cruciform clevis joints that connect the intersecting floor beams. The metallic LIDs ring is bonded to the composite tunnel and pi preforms are used again to bond the upper shell gussets to the tunnel. Bolted joints are used to attach the parachute fittings and the heavily loaded launch abort system/service module fittings to the solid laminates.

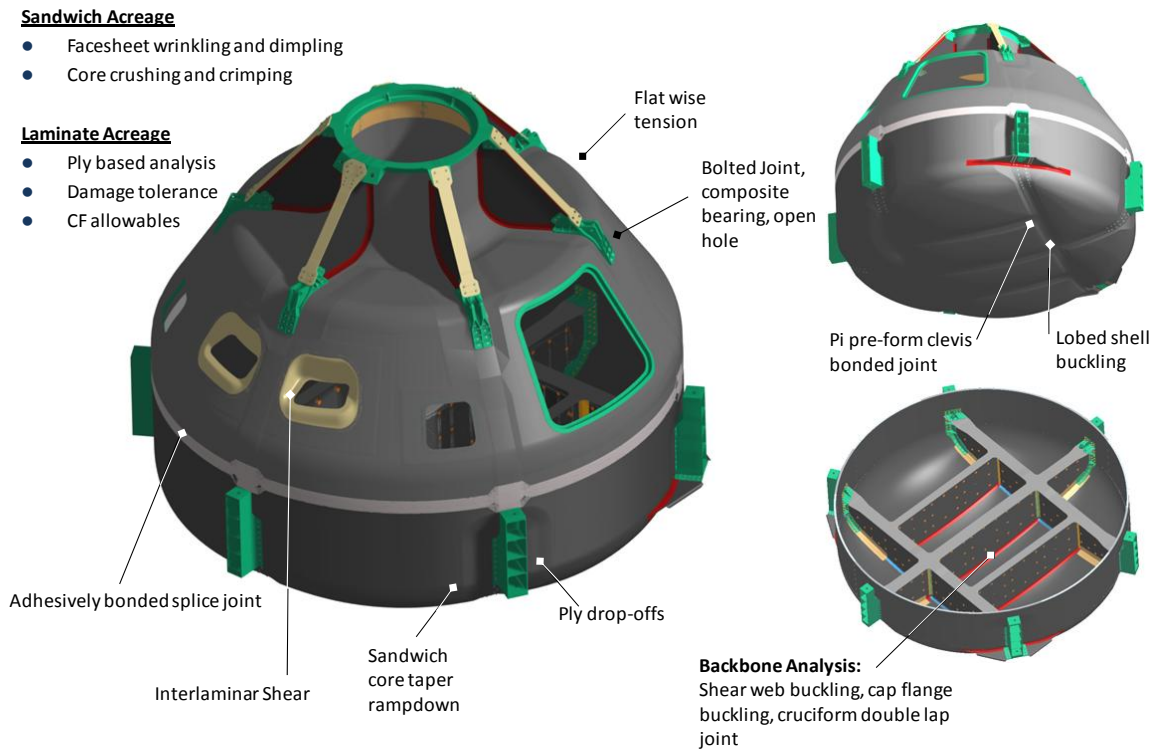


Fig.9, Scope of analyses for the CCM acreage area and details. These analyses are specific to sandwich panels, solid laminates, and bonded and bolted joints. These analyses were performed using the HyperSizer software coupled with FEA for computed internal loads. A damage tolerant design is achieved by use of appropriate composite material stress/strain allowables.

5 Design Engineering

The analysis methods identified in Fig. 9 are performed with the HyperSizer® software [5]. This tool has resulted in significant design-cycle time reduction from software integration and analysis automation that has resulted in the ability to analyze a large number of design configurations of the CCM. The benefit of tool integration on the CCM design and analysis process is presented, along with risk reduction from the use of final analysis methods earlier in the design process.

5.1 Software for Composite Design and Analysis

This project, like any other aerospace design, is making use of composite material's strength and weight efficiency, and flexibility of fabrication. To gain the most benefit with composites, engineers perform many trade studies to explore the design space and find an optimum set of panel concepts, dimensions, and layup stacking arrangements, referred to as sizing optimization. The need in their set of analysis tools is to evaluate many design alternatives very rapidly and with enough analysis fidelity to discern true differences in performance in competing vehicle configurations and design features.

To accomplish this level of composite specific analysis automation, NASA is using a two part combination of software tools. The first more widely known software is FEA. The FEA packages used on this project are NEi Nastran, NX/Nastran, MSC/Nastran, and Abaqus. The other type of software, HyperSizer, is used to perform most of the composite analysis and sizing optimization. First developed by NASA in the late 1980's, HyperSizer has been commercially developed and sold by Collier Research Corporation since 1995.

5.2 FEA

FEA packages are used primarily in aerospace for computing internal loads (load paths) of a vehicle using a relatively coarse meshed FEM referred to as the global model or master model. Literally hundreds of external loads are applied to the model to represent the complete scenario of loading events that occur from ground handling, taxi, flight, and landing. For space applications this includes launch, reentry, and abort. Considering all of the local design features to all load cases is a tremendous effort, even with the use of software automation. Composite materials further complicate the engineering design and analysis effort.

First cut composite material analyses were based on smeared laminate properties. Later, as the design matures, analyses based on actual and more accurate ply-by-ply approaches were used. Shell elements are common for acreage area analyses and many localized design features. Bar elements were used to represent structural beams. For structure that requires more in depth calculation of the multiaxial stress state, such as including peel and interlaminar stresses of a bonded joint, both solid element FEMs and HyperSizer advanced joint analyses [10,11,12] were used.

Though FEA codes have incorporated composite specific analyses, such as implementation of classical lamination theory (CLT) to the more sophisticated progressive failure Virtual Crack Closure Technique (VCCT) methods, FEA is limited to computing stress/strains of the individual plies and does not in itself predict the strength of a part, that is its failure load. FEA codes need to be supplied the failure theory and corresponding material stress and/or strain allowables that are calibrated to the failure theory. In order to capture localized stress/strain gradients such as those around bonded and bolted joints requires very fine meshes, usually incorporating solid elements. Though this level of effort is performed by researchers to investigate the impact of different kinds of details, this level of modeling is not typically performed on production projects due to high element count, modeling challenge of mesh transitions, difficulty of modeling the off axis 45 plies, and applying proper boundary conditions. Another important consideration to composite analyses is the writing of the final stress report that is performed along with full scale testing to achieve airworthiness certification. Any software automation process chosen must include detail documentation of controlling load cases, failure modes and locations, and all resulting analysis margins-of-safety.

Large aerospace companies like Boeing, Lockheed Martin, and Northrop Grumman perform these additional composite failure analyses and margin-of-safety stress reporting using a collection of analytical specialty legacy codes. Developed over decades, these non-FEA codes have been correlated to proprietary test data (validated) and have been used over many years successfully. Smaller aerospace companies, startup companies, and companies transitioning from exclusively performing metallic design to including composite design work do not have these specialty analysis programs available for their use, nor the necessary proprietary data. In lieu of their own tools and data, these companies turn to commercial software tools like HyperSizer that include the required detail analyses, material database, and open literature test data used as starting point for establishing their own fabrication process specific allowables.

5.3 *HyperSizer*

HyperSizer incorporates almost all composite analyses required for aerospace structures in a comprehensive user interface that couple very tightly the individual analyses and their corresponding margins-of-safety stress reporting. Starting with importing FEA computed internal element unit forces from the global finite element model of the vehicle's panels and beams, HyperSizer solves for hundreds of different failure modes very rapidly using material allowables and its failure criteria that are specifically correlated to test results. Its rapid analyses allows full vehicle models to be analyzed to hundreds of load cases while also including stress/strain gradients from local detail effects.

Metallic stress analysts have been successful in selecting worst case; FEA computed internal loads by filtering out and inspecting element maximum and minimum loads. Attempts to intuitively pair down hundreds of load cases to just a dozen controlling cases are not plausible with composite materials. Post processing and filtering down loads based on element individual max/min force components does not capture the combination of biaxial or multiaxial loadings that are actually critical for composites.

With composites there are many design variables to consider such as ply orientations, layup stacking sequence, and the possibility of hybrid laminates of tape and fabric, and/or different material systems such as Gr/Ep and metal foils (fiber metal laminates) that provide more opportunity to tailor the material to the application. In aerospace, this equates to the number one goal of reducing weight. To gain the most weight savings, a manual, intuitive trial and error process is not likely going to find the best design especially when strength, stability, and damage criteria need to be simultaneously considered. Inherent in any discussion of composite analysis is then the companion topic of sizing optimization of the composite structure and the process of updating the global FEM to reflect a change in load path.

The HyperSizer process fundamentally can be very briefly described in two steps. Using the CCM acreage honeycomb sandwich composite panel as the example:

Step 1) Provide the list of candidate laminates for the sandwich skins, and candidate honeycomb core materials and thicknesses.

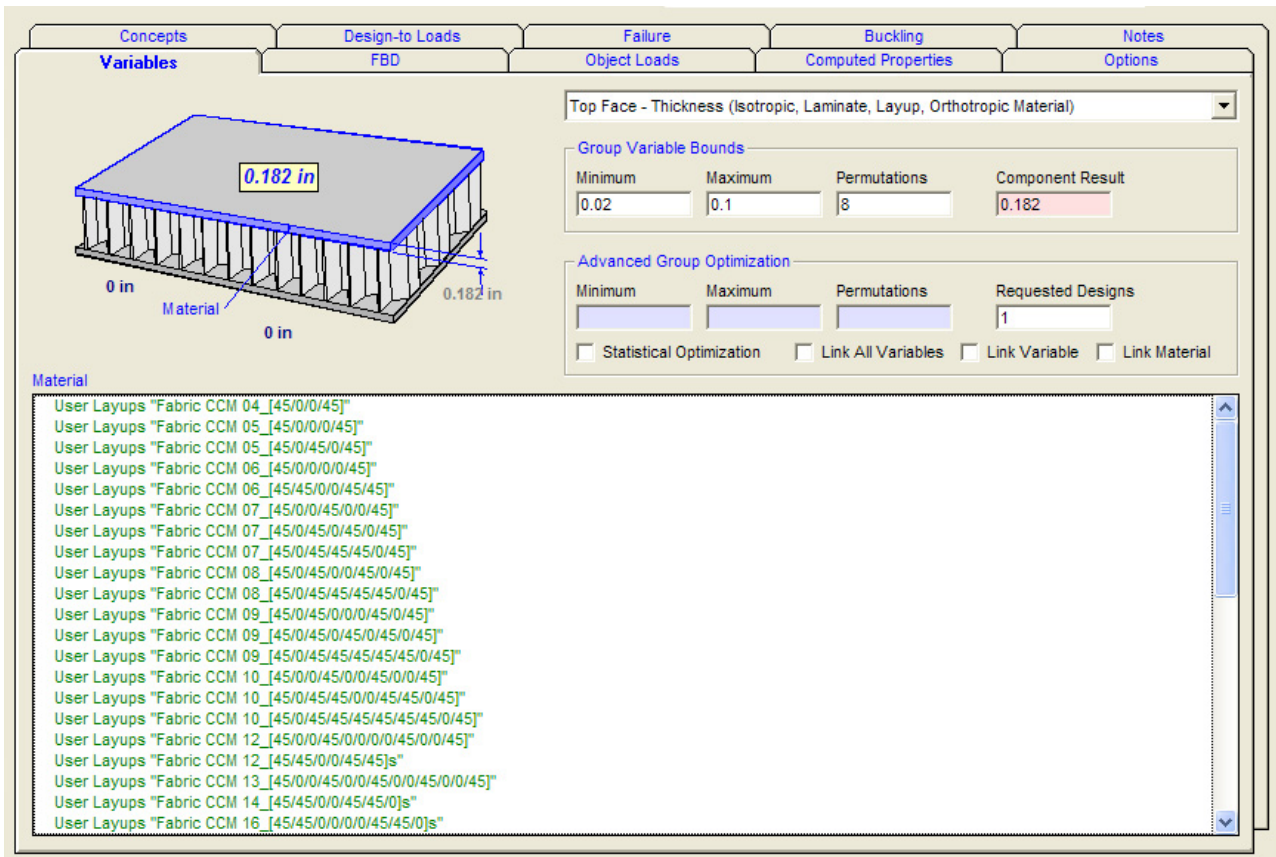


Fig. 10, Composite laminates selected by the user to be candidates for optimization. The facesheet can size to any of the layups.

Step 2) Select which failure analyses to include for margins-of-safety calculations, and that have to be satisfied during sizing optimization.

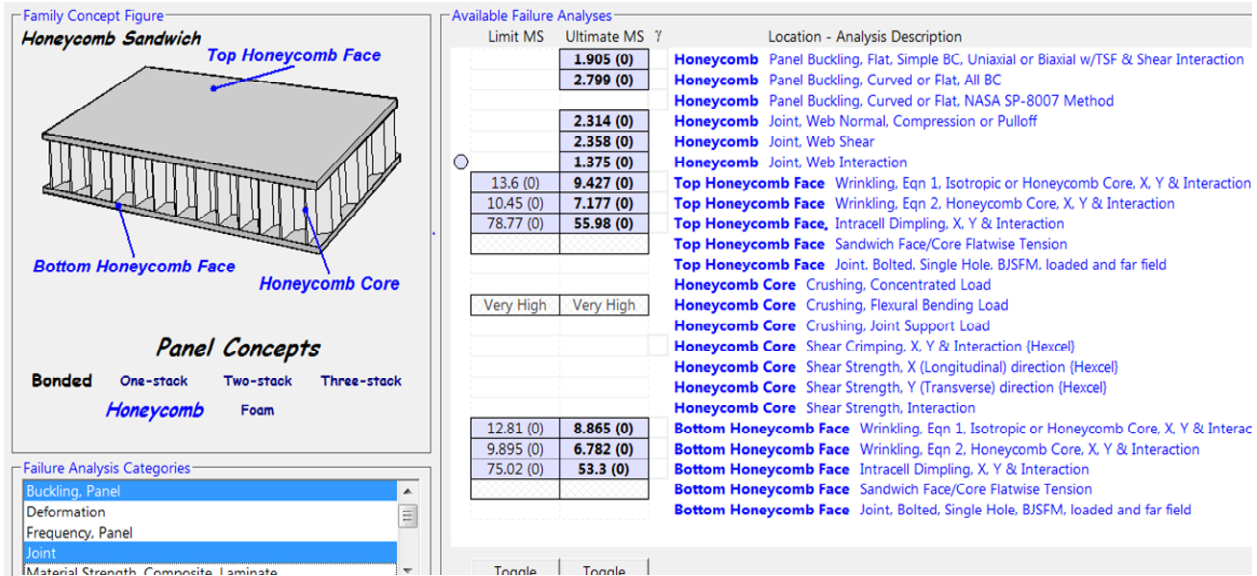


Fig.11, Failure analyses specific to sandwich panels. In this example, the lowest margin-of-safety is due to the clevis joint of the web in the bonded cruciform. This backbone web is in compression due water landing loads.

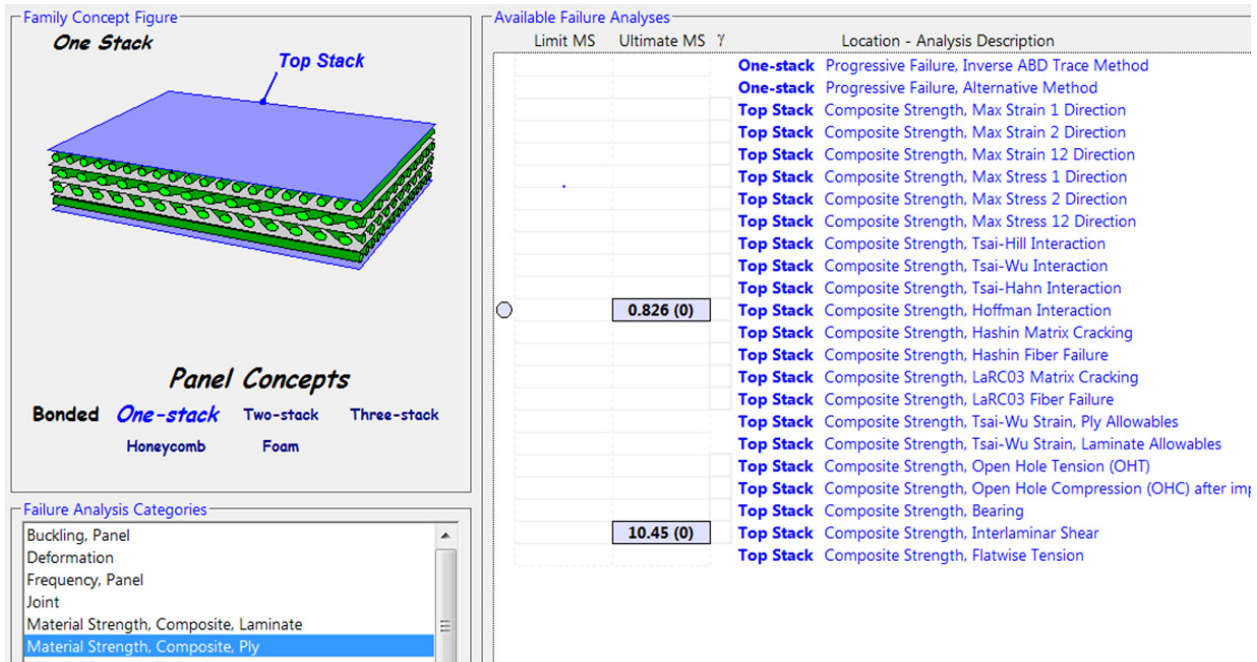


Fig.12, Solid laminate failure modes. Primarily its Hoffman composite strength criteria or out-of-plane interlaminar shear.

6 Master Model

A coarse meshed, loads model can accurately compute internal running design-to forces if the full definition of panel membrane, bending, and membrane-bending stiffness matrices are defined, including off diagonal terms, and entered into the element properties. Once the loads are properly computed, then hundreds of different analyses such as panel buckling, crippling, beam-column, bonded and bolted joint, composite strength to damage initiation and damage tolerance criteria, etc. can be performed for the entire CCM. A primary foundational capability is to accurately analyze any panel concept without the need to discretely mesh with finite elements the shape of the stiffeners or their spacing. This permits tremendous flexibility and rapid turnaround of trades with different panel concepts all from the same coarsely meshed FEM.

There is no limit to the number of FEM elements, grids, or load cases, permitting the ability to rapidly handle large FEMs. A linear relationship between run times and model size is apparent, not exponential which can become detrimental when going from demonstration to full production FEMs.

Such an approach can analyze and optimize all structural components of space structures to thousands of load cases. Statistical post processing of the FEA computed element forces provide appropriate design-to loads. These loads are used for failure analyses and are further resolved for specializing in composite analyses and optimization. A progressive Global-Local-Detail process of computing stresses and strains allows hundreds of different failure analyses to be included. Interlaminar shear and peel stress variation is computed in the adhesive for linear and five different non-linear material methods.

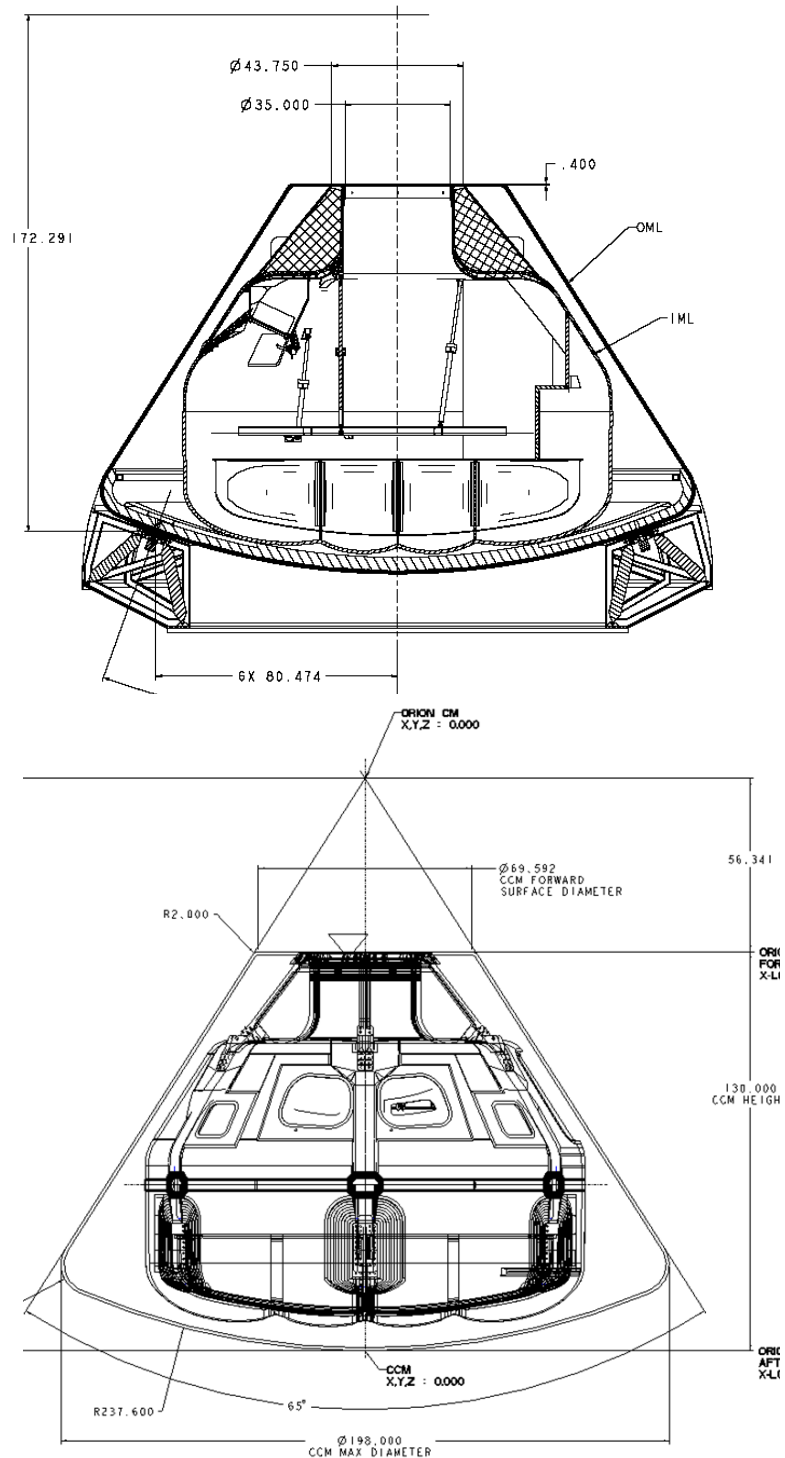


Fig. 13, CCM preferred sandwich closeout design. This design usually

7 Preliminary Sizing

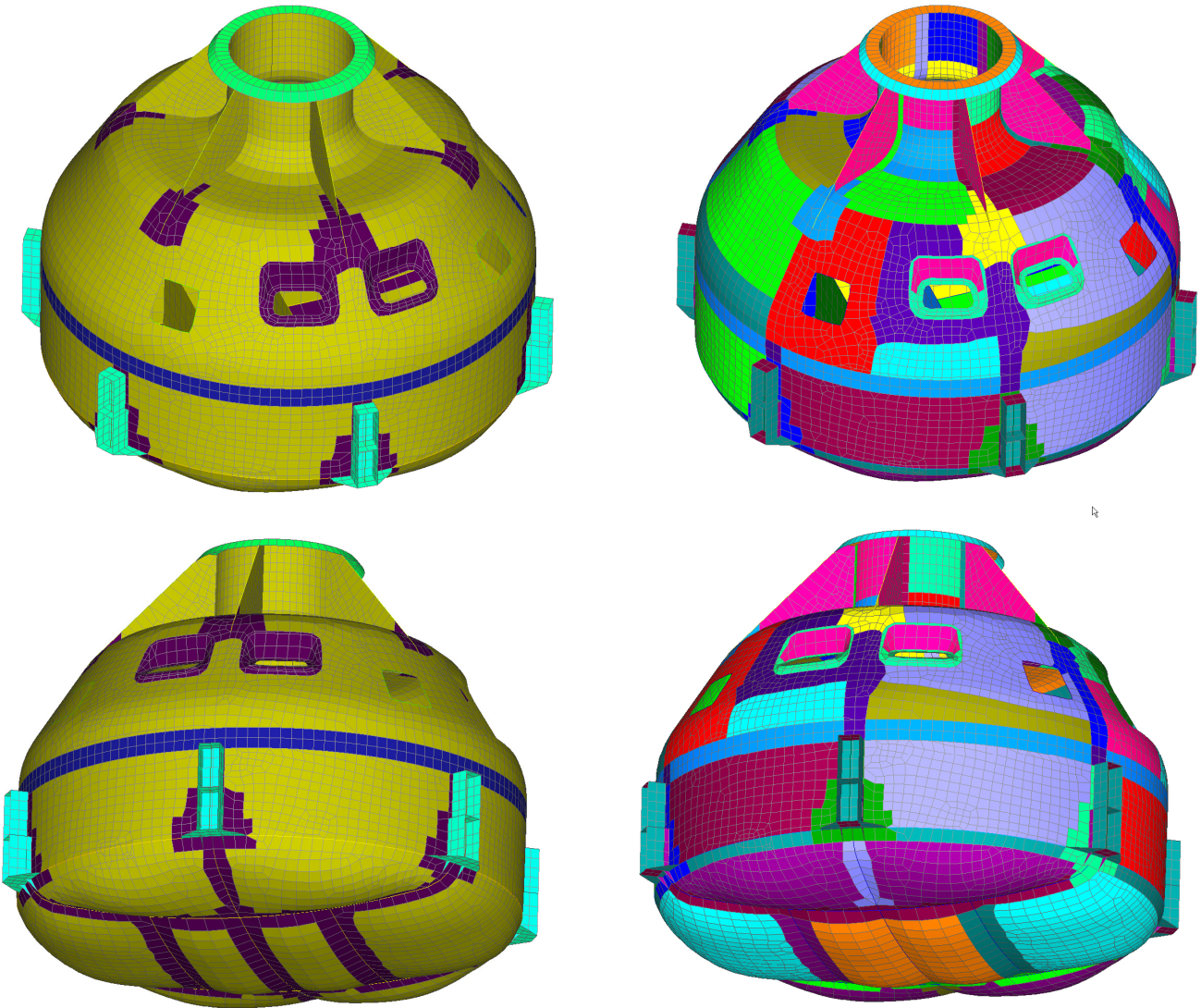


Fig. 14, Left images are the panel concept and material. The olive green represents areas of sandwich, and the purple areas of solid laminate. Right images are the composite sizing optimization areas of constant layup and sandwich core transitions

- HyperSizer imports FEA computed internal loads from the ‘Master’ FEM and performs detailed sandwich and laminate failure analyses for each structural component.
- A component is an area where the layup and core density is constant. On the FEM, a component is identified by grouping elements together as indicated with the same color. Elements of the same color share the same FEM property PCOMP layup data.
- A component’s “design-to” load is the highest element load. That is, each element’s load must obtain a +MS for the component layup and core density.
- As the loads FEM is modified during design maturation, HyperSizer automatically controls the iterative convergence between its sizing optimization and NASTRAN FEA solutions. Additionally, HyperSizer provides interactive modification of a FEM’s element property assignments for redefining layup drop-offs.

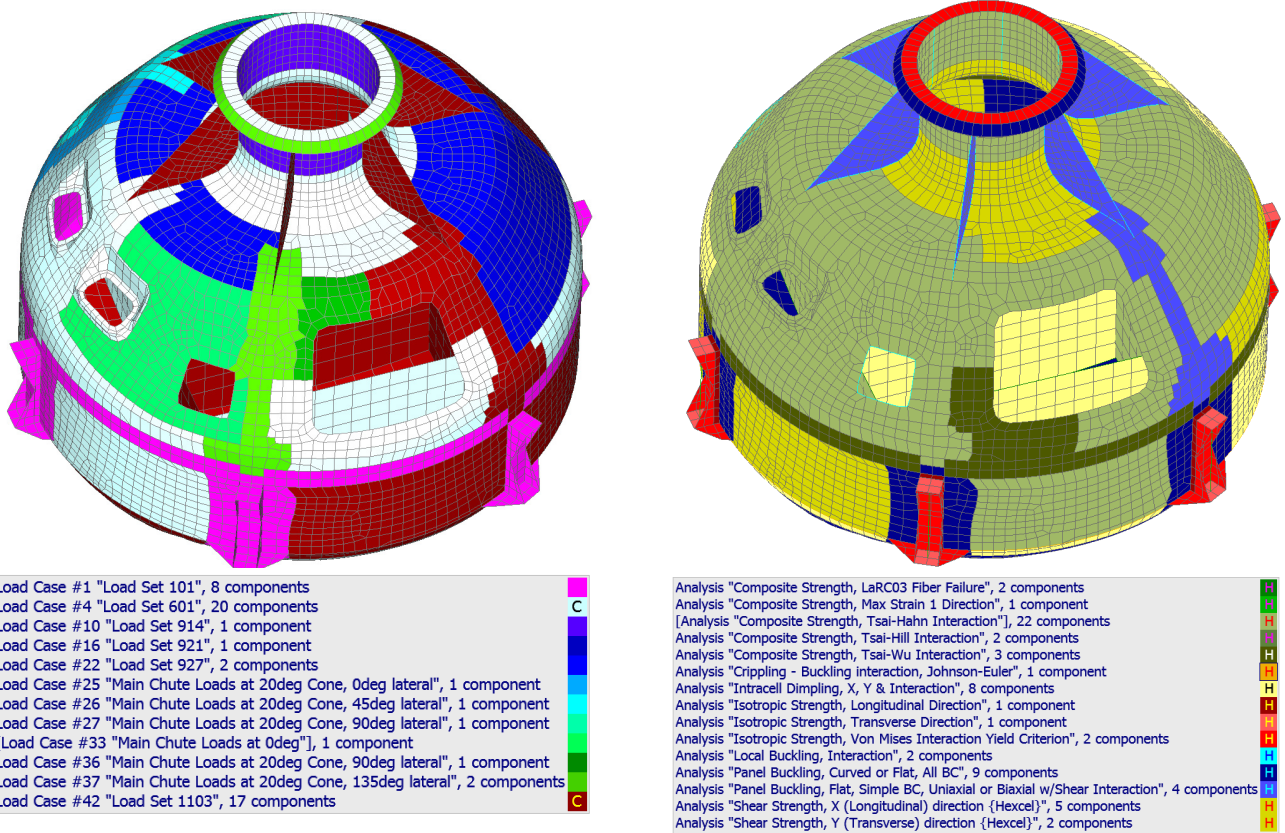
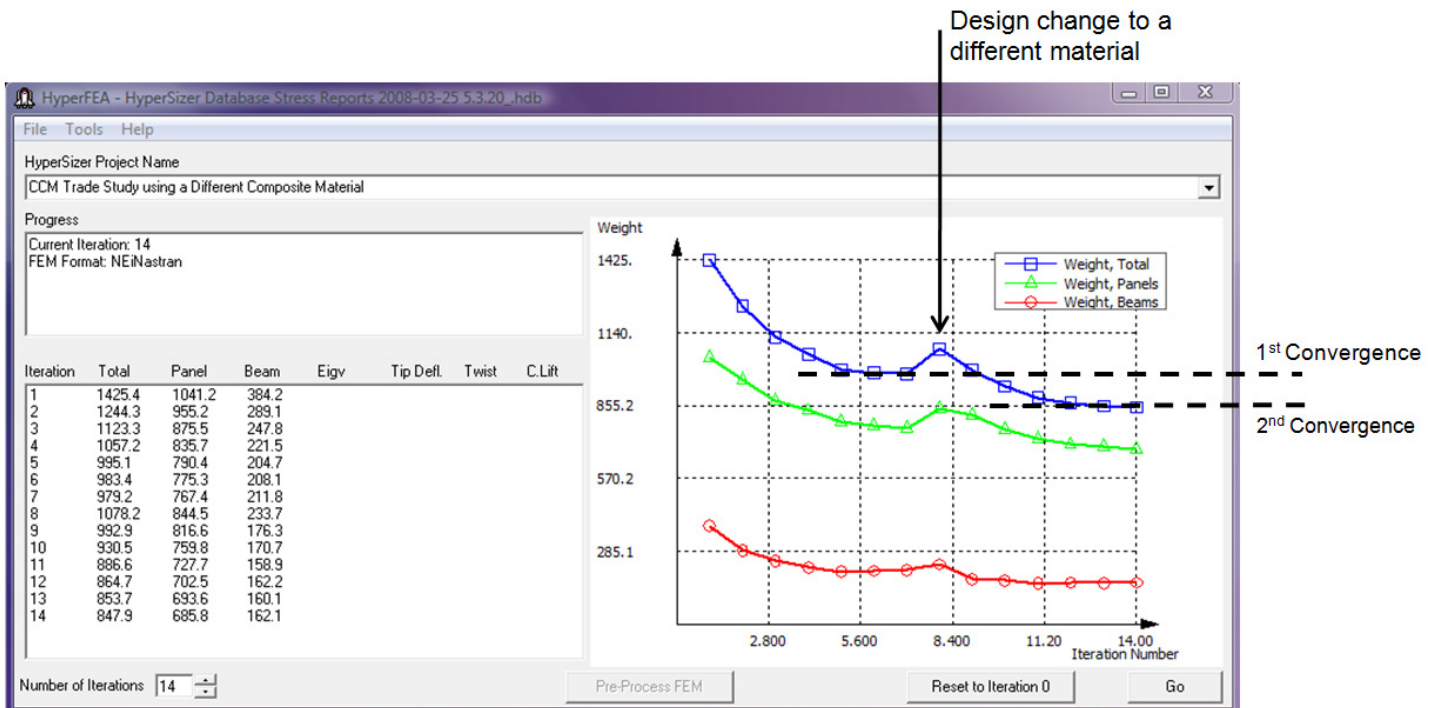


Fig. 15. Left - controlling load case: right- areas of controlling failure analyses.



8 Final Sizing and Analysis

The analytical model shown in Fig. 17 identifies the sandwich panels in green color, and the solid laminates in pink color. The dark lines define the ply drop offs.

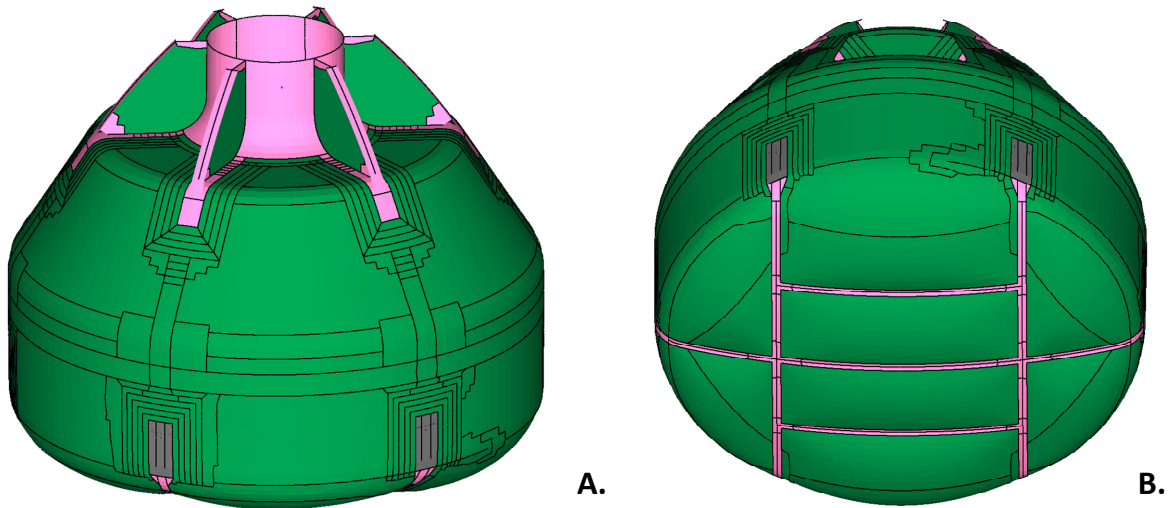


Fig.17, Areas of sandwich panels (green) and solid laminates (pink).

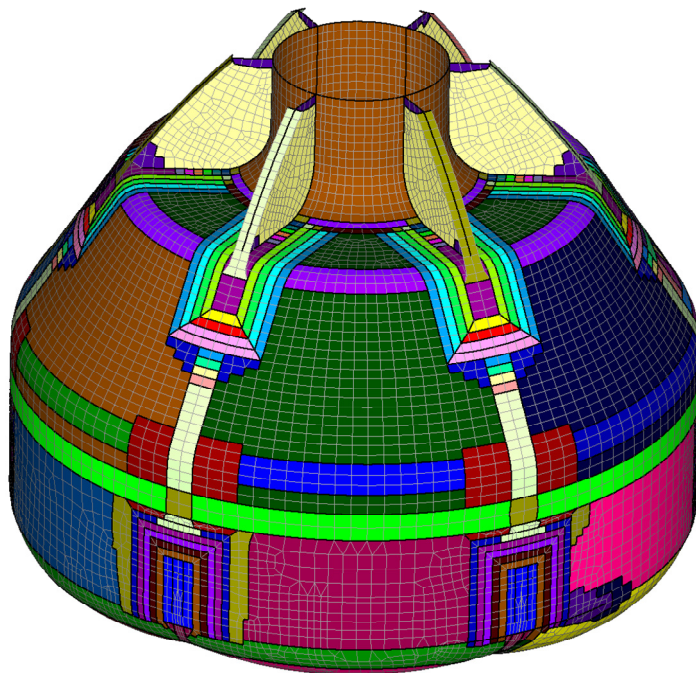


Fig.18, Each color is either a unique solid laminate layup, or sandwich design.

During the progression of design and analysis maturity, three major classifications of analyses can be defined.

- Analysis for sizing optimization
 - Architectural trade studies
 - Optimum honeycomb sandwich design
 - Optimum composite layups
- Analysis for failure margins-of-safety for acreage areas
 - Panel buckling
 - Composite strength failure and damage tolerance
 - Sandwich specific: facesheet wrinkling and core shear
- Analysis for fabrication/manufacturing features
 - Those planned early: Cutouts, sandwich ramp downs and laminate ply drops
 - Those unplanned that become known later: Fabric ply overlap regions, fiber angle alignment

The following sections outline the various types of failure analyses performed on the CCM and the general approach adopted for those analyses. As start, the approaches follow the NASA requirements and technical documents, such as [13,14].

9 Composite Material Strength

The primary approach used to analyze the laminates was ply based. That is the stresses/strains of each ply are computed and compared to a ply allowable value. This is contrast to laminate based approaches which define allowable loads on the laminate basis as a function of the layup, usually in terms of the percentage of 45 plies. Though the CCM analysis approach was ply based, the effect of stress/strain allowables as a function of the %45 plies was also included. Fig. 19 portrays these curves.

Early on, the Hoffman failure criteria was adopted as the standard failure theory to use by the entire CCM analysis team.

Hoffman failure theory

The Hoffman criterion predicts failure using the same five terms as Tsai-Wu and Tsai-Hahn, and only the last term is different:

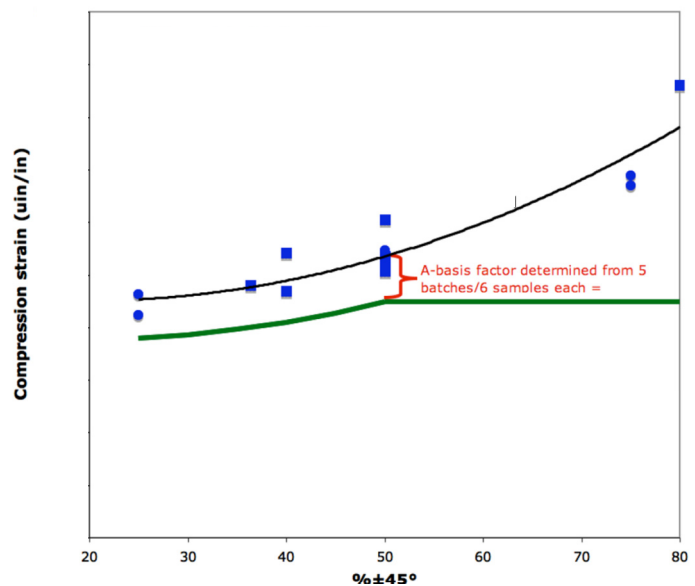


Fig.19, Caption to be determined

$$F_{12} = -\frac{1}{2X_t X_c}$$

Equation (144.2) is implemented in HyperSizer as Hoffman criterion:

$$\left(\frac{1}{X_t} - \frac{1}{X_c}\right)\sigma_{11} + \left(\frac{1}{Y_t} - \frac{1}{Y_c}\right)\sigma_{22} + \frac{\sigma_{11}^2}{X_t X_c} + \frac{\sigma_{22}^2}{Y_t Y_c} + \frac{\tau_{12}^2}{S^2} - \frac{\sigma_{11}\sigma_{22}}{X_t X_c} \geq 1.0$$

This interaction equation is rewritten as a margin of safety and performed for each ply in the laminate. The lowest MS for any given ply is reported.

$$F_1 = \left(\frac{1}{X_t} - \frac{1}{X_c}\right); \quad F_2 = \left(\frac{1}{Y_t} - \frac{1}{Y_c}\right); \quad F_{11} = \frac{1}{X_t X_c}; \quad F_{22} = \frac{1}{Y_t Y_c}; \quad F_{66} = \frac{1}{S^2}; \quad F_{12} = -\frac{F_{11}}{2}$$

$$MS = \frac{2}{F_1\sigma_{11} + F_2\sigma_{22} + \sqrt{(F_1\sigma_{11} + F_2\sigma_{22})^2 + 4(F_{11}\sigma_{11}^2 + F_{22}\sigma_{22}^2 + F_{66}\tau_{12}^2 + 2F_{12}\sigma_{11}\sigma_{22})}} - 1.0$$

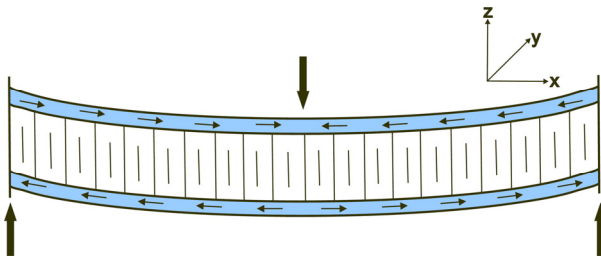
[15]

Damage Tolerance

Designing the CCM to be damage tolerant was an overriding objective. The primary approach to achieving damage tolerance was to establish fundamental stress/strain allowables that were based on test data of damage coupons. First a survey of existing data was performed to establish preliminary values, and then as more test data became available, the allowables were updated. Correction factors were established to account for elevated temperatures and wet conditions.

10 Sandwich Specific Analyses

By separating the facesheets of a sandwich panel with a lightweight core material, either foam or honeycomb, relatively specific high bending stiffness, D_{ij} , is developed. Therefore for a given panel unit weight, sandwich panels offer high compressive buckling stability and resistance to bending moments. Unlike stiffened panels, the high bending stiffnesses are in both longitudinal X, and transverse Y, directions.



By separating the facesheets, the moment of inertia is increased in the panel, and bending moments are carried more efficiently as stresses in the thin facesheets.

Because bending moments are supported by relatively thin facesheets, unique failure modes are common to sandwich panels, and are related to a buckling instability. Wrinkling, a mode shape which spans across many cells, and dimpling, which occurs in the distance of a cell, exhibit little or no post-load carrying capability and are typically catastrophic.

Likewise, failures in the lightweight core are common and also usually catastrophic. These include shearing, crimping, and crushing of the core. All of the unique sandwich specific failure modes are analyzed with HyperSizer failure methods. The HyperSizer failure methods are documented in a series of files installed with the software and are used with sample calculations that insert automatically into the generated stress reports. The equations are general and include composite materials, biaxial membrane and bending moment loadings with in-plane and out-of-plane shear and their interactions for failure.

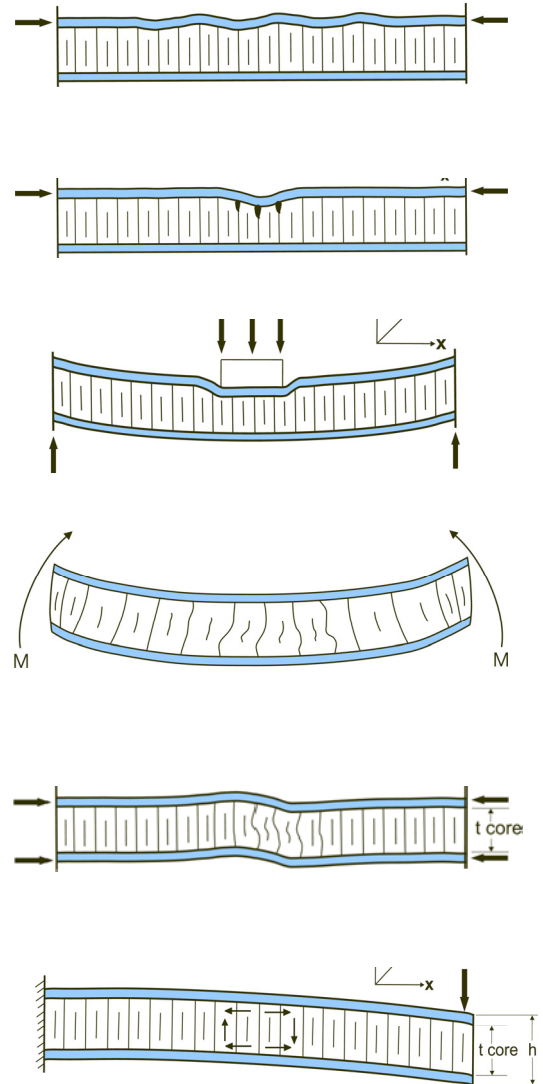


Fig.20, Different honeycomb sandwich panel structural failures: a) wrinkling, b) dimpling, c) core crushing, d) core crushing from moment, e) shear crimping, and f) shear strength

10.1 Thick cores versus thin cores

Thick cores can be fairly accurately analyzed with a simplified approach which is based on calculating peak stress by $Q_x / (t_{\text{core}} + \frac{1}{2} (t_{\text{face1}} + t_{\text{face2}}))$. For a 1" thick core with .1" thick facesheets, the difference between the simplified approach and FEA is less than 5%. However, for thick core sandwich panels, the simplified approach is about 10% over-conservative as compared with FEA. HyperSizer has a much more accurate solution to computing core shear stresses, referred to as the interlaminar shear approach. [Ref, Bednarczyk and Aboudi, NASA CR]. This method is very accurate for both thick and thin core sandwich panels.

For example, for an applied $Q_x = (100 \text{ lb/in})$, and sandwich dimensions of $t_{\text{core}} = t_{\text{face1}} = t_{\text{face2}} = (1.18" / 3) = 0.3933"$, and with facesheet material properties of $E=10.0 \text{ Msi}$, $\nu=.33$, and the Aluminum core (Hexcel 1/8-5052) with a shear modulus, G_l , longitudinal = .045 Msi:

Method	Peak Core Shear Stress (psi)
HyperSizer Simplified Approach	127.1
HyperSizer Interlaminar Shear Approach	117.4
FEA of a finely meshed verification model	117.7

10.2 Flat wise tension

The flatwise tension analysis is implemented for the case of pure bending moment applied to a curved sandwich panel. This effect can become significant for designs where the ratio of panel radius of curvature to panel thickness becomes small. The flatwise tension stress is a function of the radius of each facesheet. Also note that the radius of the inner facesheet is smaller than that of the outer facesheet, and this effect is calculated separately for each face. For any curved composite in pure bending moment, the radial stress, σ_{rr} and any radial location can be expressed [16] as:

$$\sigma_{rr} = \frac{M}{Hr}$$

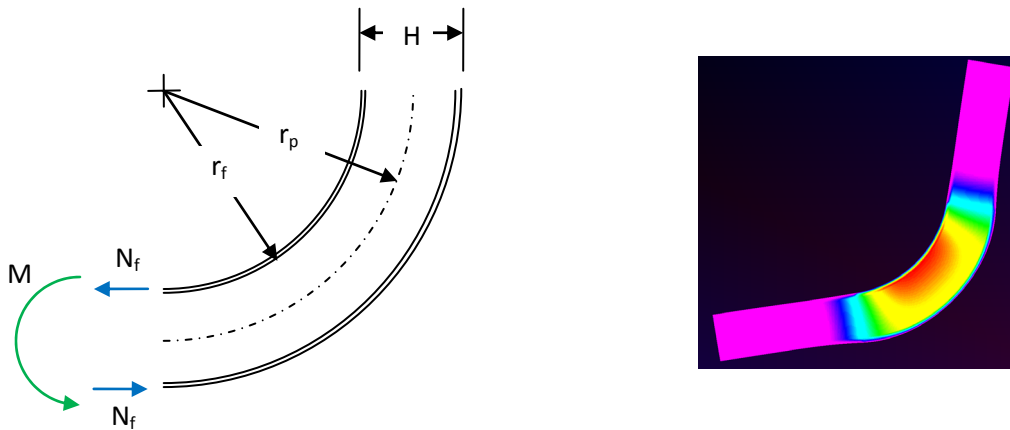


Fig.21, Left - flat wise tension fundamentals depicted, Right – FEA solution showing the effect.

10.3 Sandwich Panel Edge Closeouts, Core Taper Rampdown

There are many locations on the CCM where the sandwich panel acreege thickness of a constant 1" is ramped down to a solid laminate. This is particularly true in areas of joints. These areas are design details that must be analyzed. Early in the design, joints illustrated in figure 22, were considered. The analysis issue related to these joints is out-of-plane shear stress peaks caused by the concentrated force of the supporting web. Though the contact footprint would develop an effective width to bear against the core, the stress intensity of the peak remained an issue. These joints have the advantage of easier fabrication, and though densified core would be used in the area of the bearing contact surface, the design decision was to not use these joint concepts but instead to use a rampdown joint concept of figure 23.

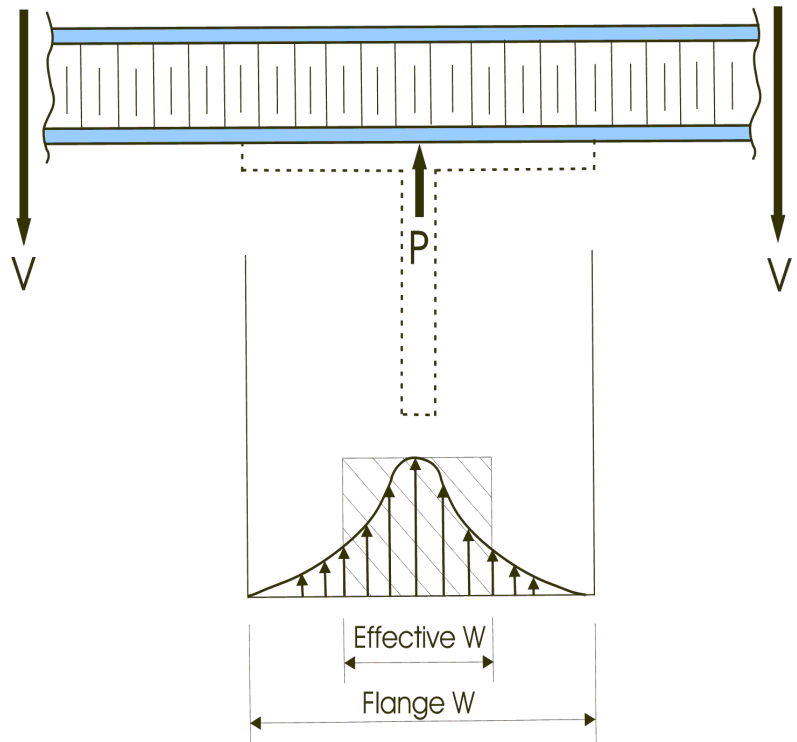


Fig 22. Compressive stress on sandwich panel from support bearing load P . The CCM design did not use these types of load bearing joints, and instead went with a rampdown closeout design shown below.

Detailed analyses show the load transfer mechanism in laterally loaded tapered sandwich panels is the tapered facesheet contributes significantly to the overall shear load transfer. This will cause a relief of the core shear stresses, and the effect may be significant even for relatively small taper angles.[17]

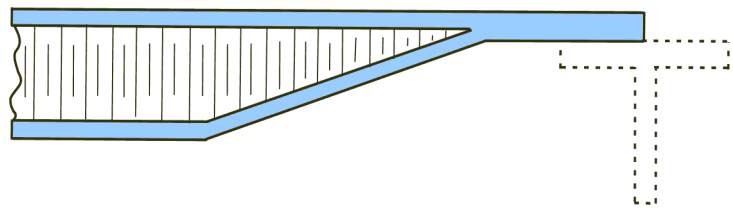


Fig. 23, CCM preferred sandwich closeout design. This design usually had core tapers approximately 7 degrees. High out-of-plane shear forces, noted as FEA Q_x and Q_y cause shear stress failures in the core. The facesheet ramp angle will carry some of this out of plane load, reducing the amount of stress in the core.

11 Ply Drop-off Transitions

Ply drop off transitions cause interlaminar shear stress failures. Layup optimization needs to consider how to best transition from thin to thicker laminates. Rules of thumb are used in industry such as a 20-to-1 slope as a drop off rate limit. Other rules such as requiring the first ply and last ply of the laminate to be 45's were used. A practical way to follow these rules and cover the entire surface of the CCM was to have the fabric layups built from a basic minimum gage layup of [45/0/0/45]. These 4 plies exist in every possible layup. Here are some examples.

45/0/0/45
45/0/45/0/45
45/0/0/0/45
45/45/0/0/45/45
45/0/45/45/0/45
45/0/0/0/0/45
45/0/0/45/0/0/45
45/45/0/45/0/45/45

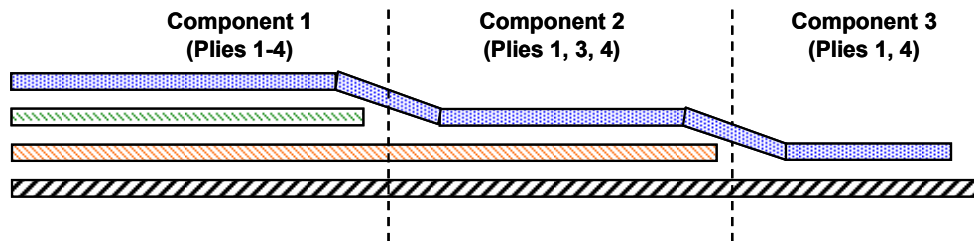


Fig.24, Possible failure initiation occurs from interlaminar shear stresses at areas of ply drop off transitions. Layup optimization needs to consider how to best transition from thin to thicker laminates.

12 Bonded Joints

Bonded joints are used extensively on the CCM. Primary locations are the out-of-autoclave, splice joint that bonds the upper pressure shell to the lower pressure shell. The splice joint is a conventional joint. However, also used extensively on the CCM are non-conventional Pi Preform joints used to bond the backbone to the lobed dome, and to bond the gusset plate to the pressure shell tunnel and ceiling.

A well planned building block test program was executed with test data providing the pull-off and shear allowables in terms of (lb/in). The allowables are defined as a function of a characteristic stress at the reentrant corner.

12.1 Pi Preform Bonded Joints

A Pi Preform joint is Tee shaped. It bonds the orthogonal oriented web to the skin. The web slides into a clevis and the flat surface of the preform is bonded to the skin. To obtain higher allowables, overwrap pairs of plies are used.

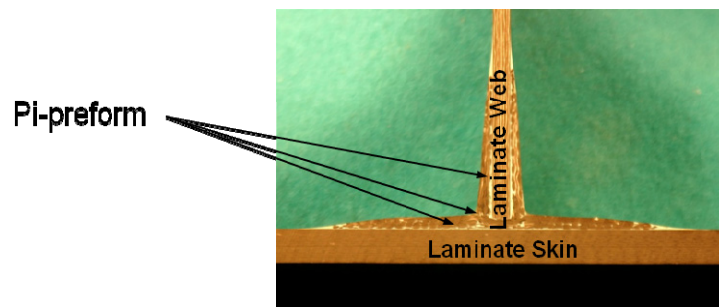


Fig.25, The Pi Preform joint as fabricated.

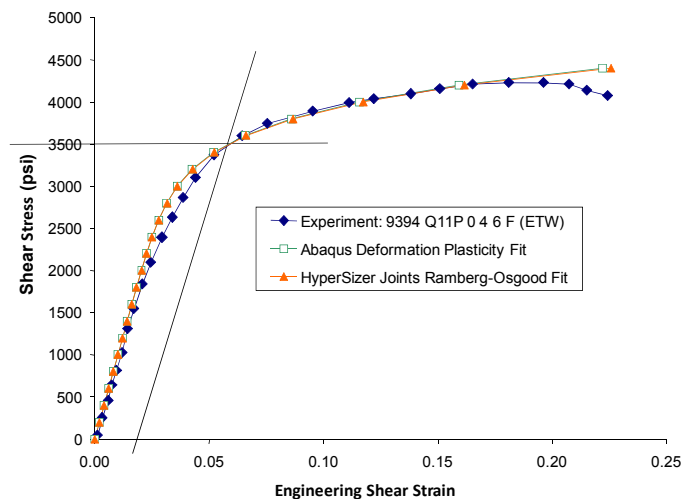


Fig.27, Nonlinear bonded joint material properties

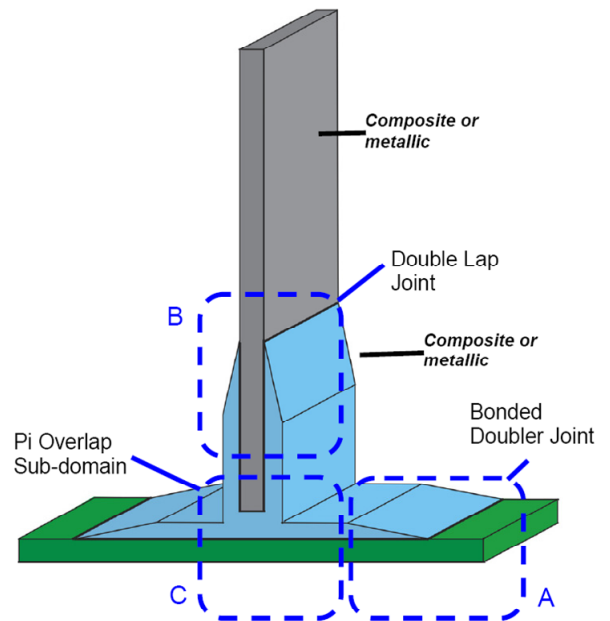


Fig. 26, HyperSizer solves the Pi joint in separate analysis domains.

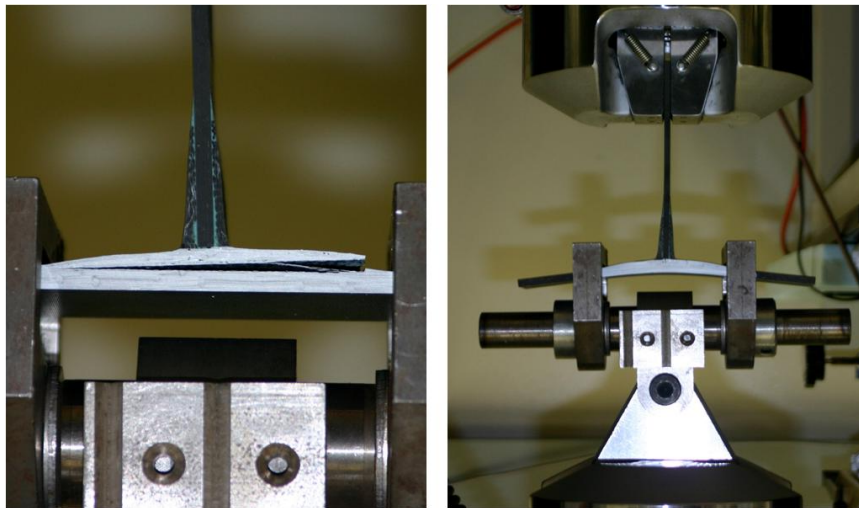


Fig.28, Pi Preform solid laminate joint testing. Failure occurs at the reentrant corner of flange and skin.

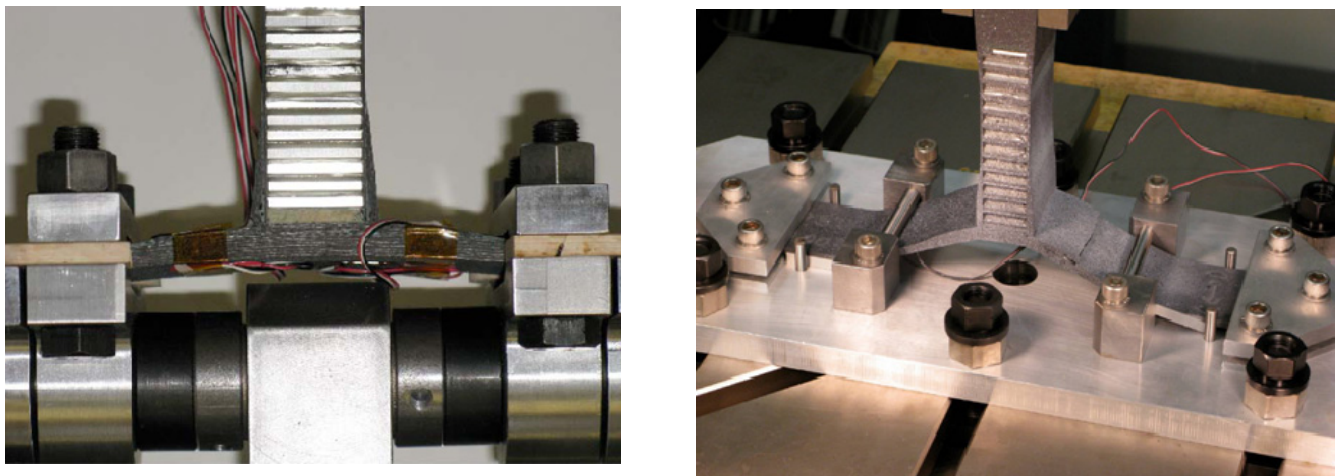


Fig.29, Pi Preform sandwich to solid laminate joint testing. Left – flat Tee shape joint. Right – 20 degree angle of the lobed dome surface.

12.2 Upper Shell Gussets

Figure 30 depicts the process of establishing the bonded joint pull-off and shear loads by using the master FEM.

The concentrated forces caused by drogue and main parachute attachments could cause failure in the Pi preform bonded joint. This is the joint that bonds the gusset plate to the pressure shell tunnel. To determine the applied shear and pull-off tension loading to the joint from in-service conditions, a new capability was developed to identify the joint grids (green grids in bottom left image). These are the grids that are in common with the intersecting shell elements. At these grids, FEA computed corner forces are generated and imported into HyperSizer. HyperSizer then transforms these forces into the ever changing joint coordinate system, depicted in the middle image. Once the element by element pull-off normal loads and shears are quantified, then they can be quickly compared to the test data allowables.

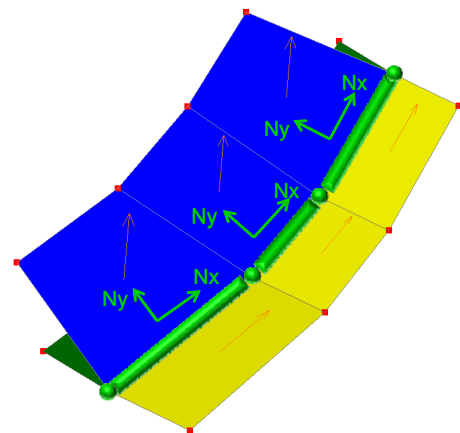
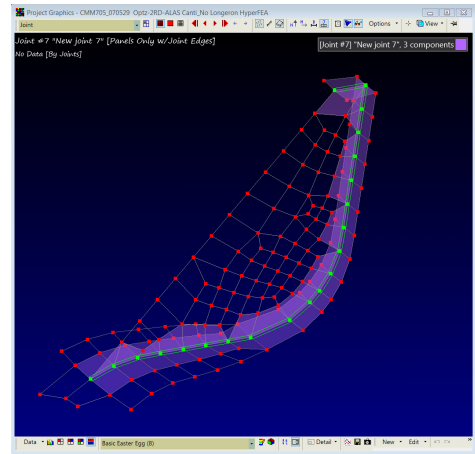
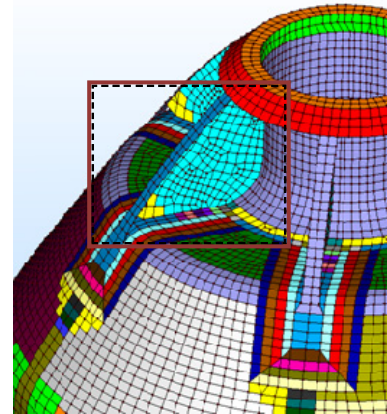


Fig. 30, Left top image is the displacement caused by a drogue parachute attachment. Right top is displacement from a main parachute load.

13 Bolted Joints

In general, bolted joints are used when appropriate, such as in areas of very high concentrated load introduction. There are two primary areas of bolted joints. One area is the six ALAS/SM attachment fittings and the other area is the six parachute fittings. In both of these areas, both the master FEM, and finely meshed models were used to compute the internal load distributions on the fasteners. The fasteners were modeled using either CBUSH, or RBE2 elements. Loads from the elements were used to establish the bearing forces. Other techniques were used to establish the bearing force on each fastener by using Free Body Diagrams (FBD) to establish the total load from one part to the other part. In this way, the count and spacing of the fasteners could be made as variables in the sizing process.

Once the bearing load on fasteners was determined, either of two approaches was used. The first, most simple approach, was to compare this bearing value to a composite material bearing allowable established from test data. The other approach was to use the BJSFM bolt program as integrated into HyperSizer. This approach considered the angle of the bolt loading, biaxial loads and shear, and the effect of biaxial far field by-pass loading and computed failure at the characteristic distance using traditional ply based failure theories.

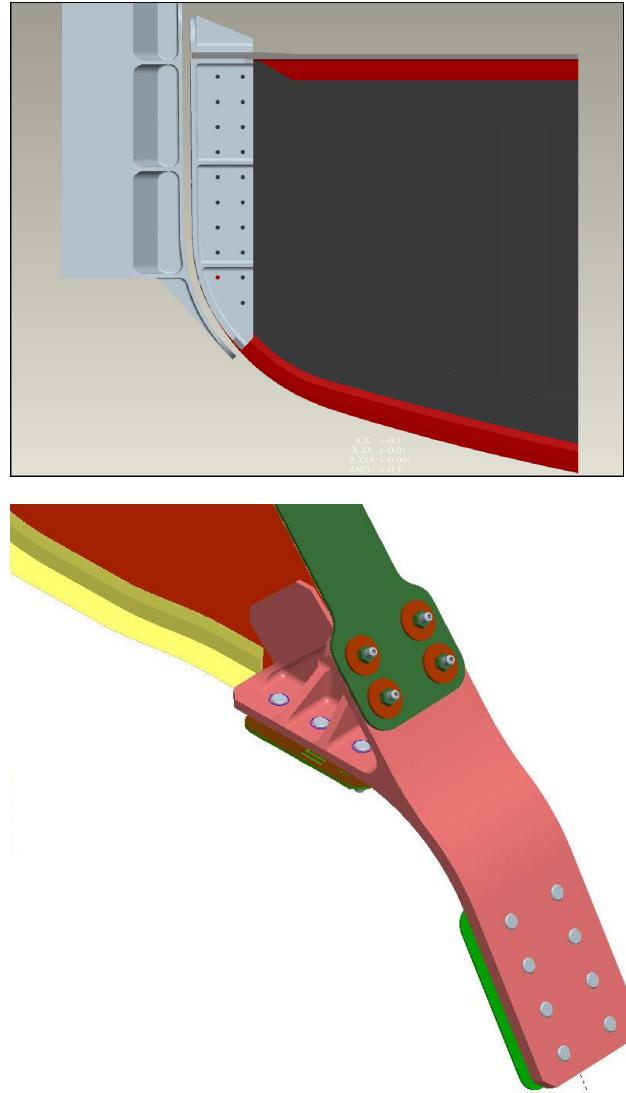


Fig. 31, Two primary bolted joint locations on the CCM. Top- the Alternate Launch Abort System/Service Module (ALAS/SM) fitting. Bottom - the parachute fitting.

14 Fabrication

The figures below are photographs taken early spring 2008. The CCM full-scale fabrication has begun.

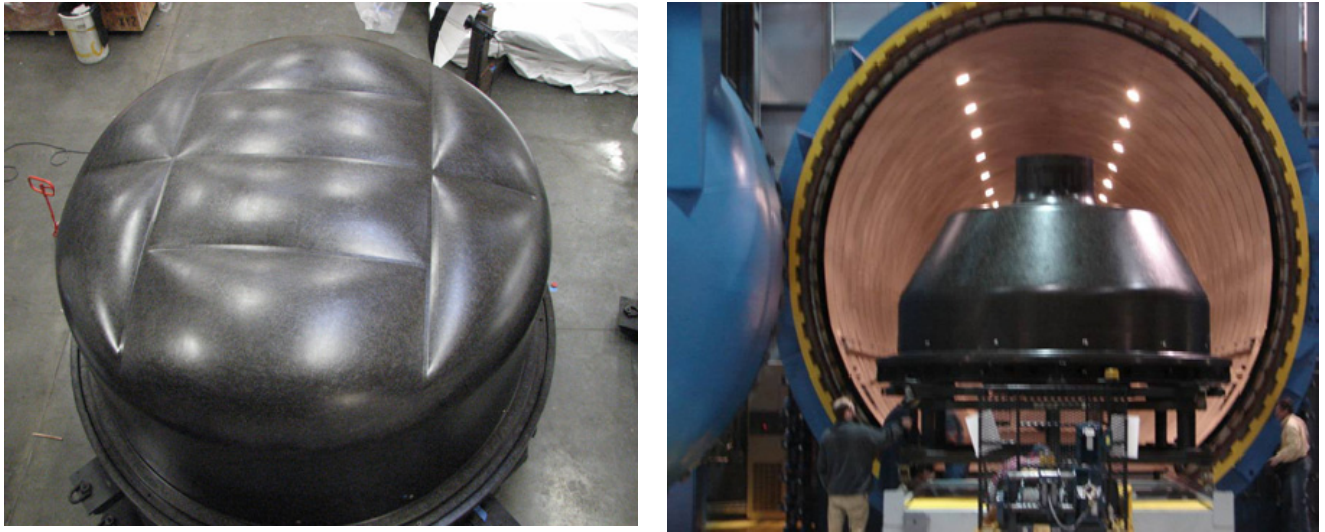


Fig.32, Left- tooling of the lobed aft dome, Right – upper pressure shell in the autoclave.

15 Conclusions

A summary of the structural analyses and composite material analyses performed for design of the full scale NASA Composite Crew Module (CCM) are described. These failure analyses are provided in the HyperSizer software for automated preliminary design sizing and final margin-of-safety stress reporting. Having the same high fidelity analyses available during preliminary design, as used in final design, is very valuable in producing hardware concepts that have less weight growth and required strength and stability during final design. By including these analyses early in the design cycle, weight growth is minimal, and weight savings can be obtained by finding appropriate alternate designs.

16 Acknowledgments

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