PROCESS ANALYSIS AND TOOL COMPENSATION FOR A COMPLEX COMPOSITE PANEL

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ABSTRACT

This paper presents a "building block" approach, where simple and complex process models together with focused validation testing are used to develop geometric tool compensation for a large, complex composite panel. A methodology is presented in which potential causes and mechanisms are evaluated one-by-one using simple approximate analysis. The mechanisms determined to be significant are then implemented in a full three-dimensional finite element model where all the geometric and material complexities are addressed. The predicted and recommended tool compensation is in good agreement with measurements on a test article.

INTRODUCTION

New aircraft designs have focused on the use of composite materials and large integrated components. Although large integrated components are associated with many advantages, one disadvantage is that the components must meet very stringent dimensional tolerances in order to fit together in assembly. Composite materials flow, shrink, and build up residual stress when processed, which often make the dimensions of the processed component different from that of the room temperature tooling [1-3]. Anything that makes the dimensions of the cured part different from that of the room temperature tooling will be referred to as process-induced deformation in this manuscript. For smaller components the process-induced deformation may be small enough that it can be ignored, but for larger components geometric compensation of the process tool is often necessary. There is often ample *qualitative* knowledge among the experienced composite manufacturers of how different types of composites components deform due to processing. However, *quantitative* knowledge of the accuracy needed for geometric tool compensation is seldom available. If geometric tool compensation is expected to be necessary, a number of prototype components are often made to get quantitative information on the degree of compensation required. This approach is both feasible and cost effective when components are small and have low complexity and when this activity can be included in the component development schedule. When components are large and complex the cost of prototypes can be very large and a "building block" approach, where process simulation and modeling supported by focused testing, is often more cost effective for developing appropriate tool compensation factors. An advantage of using a building block approach is that process models, based on physical laws rather than empirical observations, can be used to extrapolate experimental data from sub-scale articles to the full-scale design space. This can be very valuable for providing updates to compensation factors in response to last minute design changes or unexpected process conditions. Process analysis and simulation also forces a deeper thinking and understanding of the phenomena which is of benefit to process engineers in the long term.

The presented work was part of the "Low Cost Fairings and Doors (LCF)" project, a collaboration between Boeing Canada Technology Winnipeg (BCTW), the Composites Innovation Centre Winnipeg (CIC), and Convergent Manufacturing Technologies Inc. (CMT). The objective was to determine the appropriate tool compensation for a complex composite panel. It was decided that a combined modelling-experimental "building block" approach should be taken to minimize the number of test articles and to maximize the understanding of the main drivers for process-induced deformation of this and similar components. For proprietary reasons some technical detail has been omitted from the manuscript. However, the approach and methodology are clearly described.

The panel under investigation is a thick honeycomb structure with carbon/epoxy composites skins (Figure 1). The component is approximately 3.5 m long (fore-aft) and 2.5 m wide (inboard-outboard), with a maximum thickness of about 0.1 m. It has a distinct curvature in the y-z plane and a slight curvature in the x-z plane. The honeycomb core is chamfered down to a solid laminate at the edges. The component is processed at 177°C on a steel tool.

Figure 1. Cured and trimmed test article placed back on process tooling to measure dimensional fidelity.

ANALYSIS AND MODEL DEVELOPMENT

The goal of the project was to develop and validate engineering models that capture the main drivers for process induced deformation of a complex composite panel. By developing several models, ranging from simple to complex, their results could be checked against each other, increasing the confidence in the predictions. To develop the models three questions were considered: 1) What can potentially cause process induced deformation in the panel? 2) Which of these causes or mechanisms are significant enough that they need to be included in the models? 3) What is the best way to incorporate them into the models?

Process-induced deformations are often in the order of millimeters, a small fraction of the overall dimensions of the component. Some deformations are important and some are not depending on the magnitude and on how the component is integrated and fastened to the surrounding structure. For this investigation, the main concern was deformation normal to the tool surface (out-of-plane deformation). Minor in-plane deformations can easily be addressed by trimming the component to the appropriate dimensions after cure.

Based on experience with similar components, the following list of potential causes for out-of-plane process-induced deformation was developed.

1. Thermal expansion/contraction (part and tool)
2. Cure shrinkage
3. Tool-part interaction
4. Chamfered core at edges
5. Lay-up imbalance
6. Through-thickness temperature and cure gradients

The potential causes were evaluated one-by-one in an approximate manner to determine if they were significant enough to require incorporation into the process models. Significant was defined as causing an out-of-plane deformation relative to room temperature tooling of more than 0.25 mm (0.01 inches).

Approximate analysis: Thermal expansion/contraction (part and tool)

To assess the effect of tool and part expansion/contraction on dimensional fidelity of the curved panel, consider a circle sector of radius R_0 and sector angle ϕ_0 (Figure 2). The arc length s_0 of the sector is then given by

$$s_0 = R_0 \cdot \phi_0 \tag{1}$$

If the circle sector is subject to strains ε_R and ε_ϕ in the radial and circumferential directions, respectively, we have that

$$R_1 = R_0 \cdot (1 + \varepsilon_R) \tag{2}$$

$$s_1 = s_0 \cdot (1 + \varepsilon_\phi) \tag{3}$$

$$\phi_1 = \frac{s_1}{R_1} = \phi_0 \cdot \frac{(1 + \varepsilon_\phi)}{(1 + \varepsilon_R)} \tag{4}$$

R_1, s_1, and ϕ_1 are the radius, arc length and sector angle due to strains ε_R and ε_ϕ. For composite laminates we have that $\varepsilon_R \neq \varepsilon_\phi$, which gives that $\phi_1 \neq \phi_0$ and a change in angularity ϕ [4]. If two circle sectors of radius R_0 and R_1 are fitted up at the centre (Figure 2), the gap δ between the two is approximately

$$\delta \approx \left(R_1 - R_0\right) \cdot \left(1 - \cos \phi_0\right) \tag{5}$$

Thus the gap is smaller than the difference in radii for sector angles less than 180°.

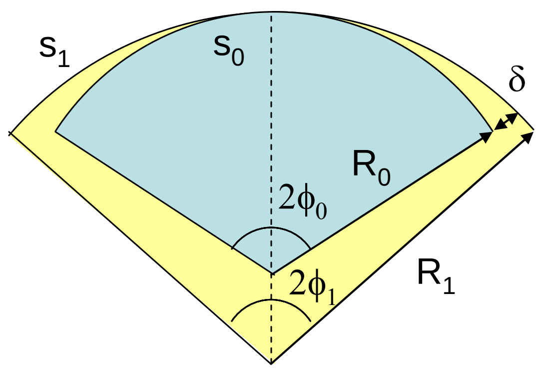

Figure 2. Geometry of circle sectors.

We can use these simple geometric arguments to assess the effect of thermal expansion/contraction on the dimensional fidelity of the part. The panel is not a perfect circle sector but half the part in the y-z plane is approximately flat and the other half approximately a circle sector (Figures 1 and 5).

When temperature of the tool is increased an amount ΔT, all dimensions will increase proportionally if the material is isotropic as in the current case (steel tool). When the tool is in its expanded state, the part cures and sets in shape. During cool-down to room temperature, the part will shrink due to thermal contraction. If the coefficient of thermal expansion (CTE) of the tool and part (in the direction of interest) are different, they will not have the same dimensions at room temperature. To estimate how the radius of the cured composite part at room temperature, R_P, compares to the radius of the room temperature tooling, R_T, consider the following sequence of events: initial tool radius, R_T, tool growth due to thermal expansion, ΔR_T, part sets in place, radial contraction of the part due to thermal contraction, ΔR_P, giving final composite radius R_P:

$$R_P = R_T + \Delta R_T - \Delta R_P \tag{6}$$

$$\Delta R_T = R_T CTE_T \Delta T \tag{7}$$

$$\Delta R_P = R_T \left(1 + CTE_T \Delta T\right) CTE_P \Delta T \tag{8}$$

$$R_P = R_T \left(1 + CTE_T \Delta T\right) \cdot \left(1 - CTE_P \Delta T\right) \approx R_T \left[1 + \left(CTE_T - CTE_P\right)\Delta T\right] \tag{9}$$

CTE_T, and CTE_P are the tool and part coefficients of thermal expansion in the radial direction, and $\Delta T = T_{cure} - T_{room}$. The gap between the cured part and room temperature tooling can be calculated from eqs. (5) and (9)

$$\delta \approx R_T \left(CTE_P - CTE_T\right)\Delta T \cdot \left(1 - \cos\phi_0\right) \tag{10}$$

The coefficient of thermal expansion of a steel tool $CTE_T \approx 12*10^{-6}/°C$. A honeycomb panel obeys the geometric relationships presented above provided that through-thickness shear is prevented. With the core tapering down to a solid laminate at the edge band (Figure 1) this is approximately the case and the effective through-thickness coefficient of thermal expansion is a weighted average of the skin and the core [5]

$$CTE_P \approx \frac{c \cdot CTE_{core} + 2t_{skin} \cdot CTE_{skin}}{c + 2t_{skin}} \tag{11}$$

where c and t_{skin} are the core and skin thickness, respectively (Figure 3).

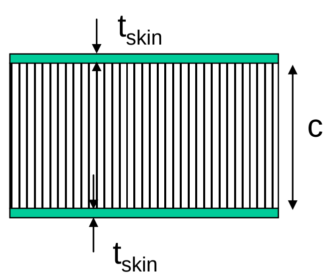

Figure 3. Through-thickness geometry of sandwich panel.

Equation (10) can be used to estimate the maximum displacement δ due to tool and part expansion/contraction. Inputting the specifics of the panel configuration into Equation (10), we have that

$$\delta \approx 3.3mm \tag{12}$$

This displacement is significantly larger than the 0.25 mm limit and the mechanism needs to be included in the detailed process models.

Approximate analysis: Cure shrinkage

Cure shrinkage of curing composites laminates can add as much as 35% to the thermal through-thickness shrinkage of a laminate [6]. In the current case the maximum effect of cure shrinkage is ≈ 35%*2t_{skin}/c. Given the small t_{skin}/c ratio for a honey comb panel, cure shrinkage was excluded from the detailed models.

Approximate analysis: Tool-part interaction

Due to the higher in-plane thermal expansion of the tool in comparison to the part, a gradient of residual stress may build up through the thickness of a laminate during heat up [2]. This stress gradient causes a net bending moment at the neutral axis, which causes laminates to warp away from the tool [7]. A semi-analytical model gives that for flat laminates, the maximum warpage δ induced by tool-part interaction can be estimated from [7]

$$\delta \approx 1.5*10^{-8}\frac{L^3}{t^2} \tag{13}$$

where δ, L, and t are all in meters. δ is the maximum gap between the tool and the cured warped part of length L and total laminate thickness t, when the part is placed back on the process tool. For a honeycomb panel $t = 2\cdot t_{skin} + c$ (Figure 3). We can use this expression to estimate if warpage due to tool-part interaction is important. The maximum inboard-outboard dimension of the panel is approximately 2.5 m (Figure 1) and the total panel thickness is 0.1 m, thus the warpage due to tool-part interaction is estimated to be

$$\delta \approx 1.5*10^{-8}\frac{L^3}{t^2} \approx 1.5*10^{-8}\frac{2.5^3}{0.1^2} \approx 2.3\cdot 10^{-5}[m] \tag{14}$$

which is small enough that it can be excluded from the analysis. Although the panel is longer in the fore-aft direction (approximately 3.5 m), the curvature will restrict deformation in that direction.

Approximate analysis: Chamfered core at edges

The solid laminate edge band outside the core area tends to pull up or “spring-in” due to the angled geometry of the chamfer. Figure 4A illustrates the geometry of the upper and lower skin before cool-down; Figure 4B illustrates the anticipated spring-in of the upper skin due to the 45° angle in the upper skin; Figure 4C illustrates the actual spring-in because the upper and lower skin are bonded together at the edge band.

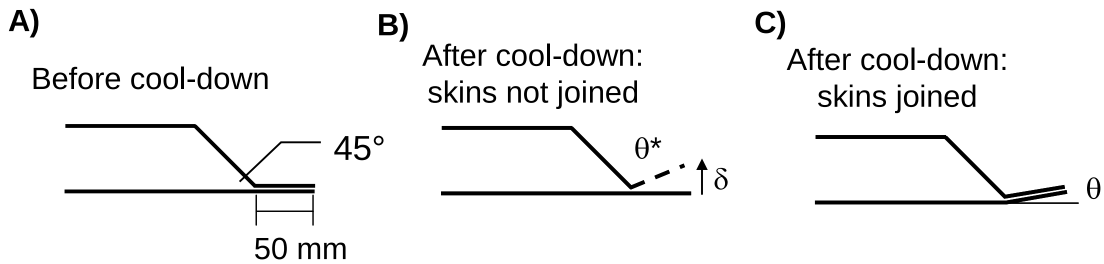

Figure 4. Schematic of "spring-in" at end of core chamfer.

The free spring-in angle θ* of the upper skin due to the 45° chamfer angle is estimated to be about 0.5° (Figure 4B) based on a typical 1° spring-in angle for a 90° carbon-epoxy laminate. The actual spring-in angle θ will be maximum one half of that, and perhaps as low as one eighth of that if we assume that beam theory applies. The maximum out-of-plane displacement δ due to this spring-in is therefore

$$\delta \approx \frac{0.5^\circ}{2} \cdot \frac{\pi}{180} \cdot 50 \approx 0.2[mm] \tag{15}$$

which is slightly less than the 0.25 mm limit. This conclusion was later verified with detailed finite element models.

Approximate analysis: Lay-up imbalance

The actual lay-up is quite complex with several different types of core densities and laminate build-ups over the panel area. This complexity may lead to slight through-thickness material asymmetry locally. The effect of this cannot easily be estimated using simple analysis and will therefore be incorporated and addressed in the detailed process models.

Approximate analysis: Cure and thermal gradients

Large cure gradients through-the-thickness can create gradients in residual stress and thereby warpage of cured laminates. Large cure gradients require large temperature gradients through the laminate that are sustained for an extended period of time. A simplified 1D analysis of the thermal gradients within part and tool during a cure cycle has been presented in the literature [8]. Using this analysis to current tool materials, tool and part thicknesses, and autoclave heat transfer characteristics, it was seen that temperature gradients are relatively small in the current case and they are not sustained for any significant length of time. Thermocouple data from a test article supported this analysis, and cure gradients were ruled out as a significant contributor for dimensional control.

Summary of causes and mechanisms controlling dimensional fidelity of the complex composite panel

Based on the assessment of potential drivers for dimensional fidelity of the panel we have the following summary:

1. Thermal expansion/contraction (part and tool)
   - Main driver
2. Cure shrinkage
   - Small effect as skins are thin compared to core
3. Tool-part interaction
   - Small effect due to length to thickness ratio of panel
4. Chamfered core at edges
   - Small effect. Will only affect deformations in the edge band a minor amount
5. Lay-up imbalance
   - Likely small in current structure but will be addressed in detailed models
6. Temperature and/or degree of cure through-thickness gradients
   - Small in current structure

There are three main advantages to going through an assessment of potential causes and mechanisms in a processing problem: 1) It focuses thinking on the fundamental cause of the problem. 2) It minimizes the time and effort required to set up detailed process models as well as eliminating unnecessary materials and boundary condition characterization. 3) It provides a basis for verification of the detailed models as well as guidance of how to focus test article manufacturing and data collection.

COMPARISON OF ANAYSIS AND TEST RESULTS

Based on the conclusions from the approximate analysis several thermo-elastic finite element based process models were created using the COMPRO CCA – ABAQUS platform that included the mechanisms identified as important. Both two-dimensional (2D) and three-dimensional (3D) models were created for internal model verification and evaluation of required local geometrical detail in the models. The results presented are from a developed 3D finite element model that included tool and part thermal expansion/contraction as well as details of ply lay-up and core geometry. The chamfered core was not included in this particular model in order to reduce model size and computational cost. The analytical finding of the small effect of the core chamfer on out-of-plane deformation was validated with detailed 2D finite element models of the chamfer region (not presented here).

A few test articles were fabricated to develop an appropriate manufacturing process for the panel. The process induced deformations of these test articles were measured by placing the cured and trimmed part back on the process tool and measuring the gap between tool and part around the perimeter using a feeler gauge. This measurement technique was selected to simplify the measurements and the analysis of the data. These simple measurements do not give a complete picture of the process induced deformation but give an indication of the deformation around the perimeter of the part where it interfaces with surrounding structure.

Figure 5 shows the predicted out-of-plane deformation of the panel from a detailed finite element process model. There is very little deformation of the "flat" inboard section as expected, but the curved section "springs-in" in accordance with presented simple geometry arguments and analysis. To better quantify the deformation, the deformation along ten approximately parallel station lines are shown in Figure 6. Due to the almost constant cross-sectional geometry in the fore-aft direction of the test article there is little difference between predicted deformations at different station lines. The model predictions presented in Figures 5 and 6 are for a model with "thick" face sheets through-out the laminate. The actual panel has face sheet thickness that varies in a complex manner over the panel. However, as in-plane and through thickness thermal expansions virtually unaffected by these changes, the model showed that there is virtually no difference in the predictions between thick, thin or actual laminate thicknesses.

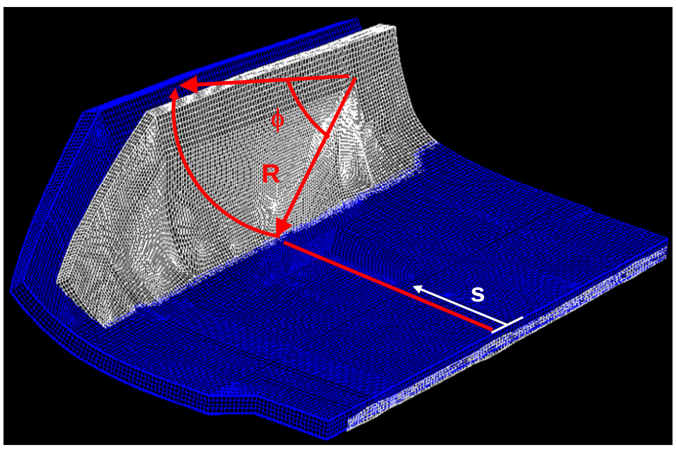

Figure 5. Undeformed (blue) and deformed (white) finite element mesh of the panel. Out-of-plane deformations magnified 10 times.

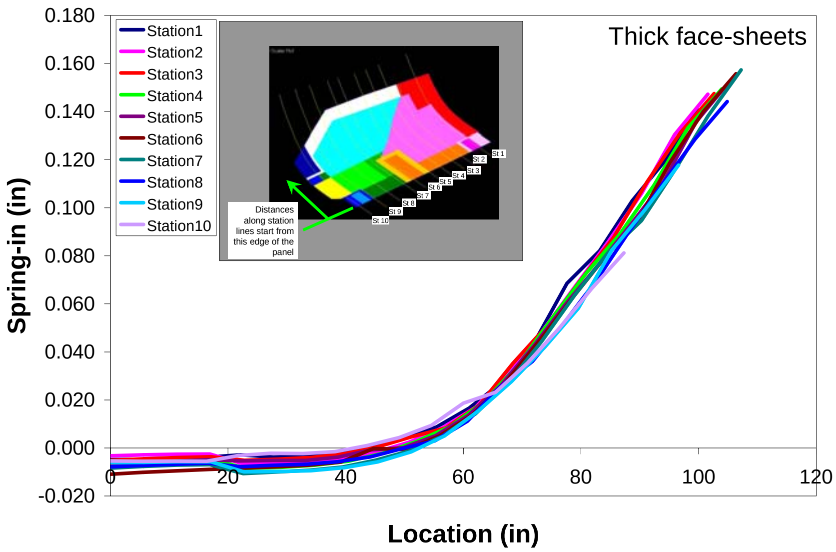

Figure 6. Predicted out-of-plane deformations of the panel along 10 station lines in the fore-aft direction.

Based on several detailed process models and simple analytical analysis, a tool compensation scheme, which is essentially the inverse of the predicted part deformation shown in Figure 6, was developed. No compensation was required from the inboard edge to approximately 1.5 m (58") into the panel where the curvature starts. In the curved region the recommendation was to compensate the tool linearly from no compensation at the start of curvature to 3.8 mm (0.15”) compensation at the outboard edge. Note that this recommendation is in approximate agreement with the simplified “spring-in” analysis presented earlier (3.3 mm).

After the predictions and tool compensation recommendations were made, measurements were taken on a test article. A comparison of predicted and measured deformation for the full scale test panel is shown in Figure 7. The agreement was deemed satisfactory given the variability in the data and approximate measurement technique used (feeler gauge). Higher fidelity contour data is required if confidence in model predictions is to be further increased.

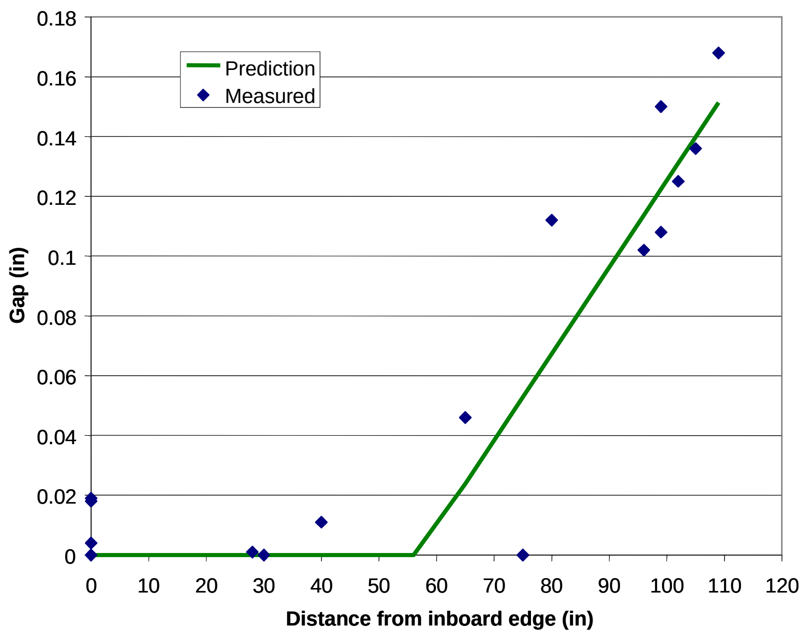

Figure 7. Comparison of predicted and measured out-of-plane deformation for the full scale test article.

DISCUSSION

The building-block approach, using a combination of simple and complex models together with focused validation testing, showed that the main driver for out-of-plane process-induced deformation of a complex composite panel is the difference in through thickness CTE of the honeycomb core and the tool CTE, see Eq. (12) and Figure 6. If a core with a CTE greater than that of the steel tool ≈ $12{*}10^{-6}$/°C as in the current case, the part will "spring-in" and create a gap between the room temperature tool and the cured part. Contrary to common belief, a higher CTE tool material (e.g. aluminum) would reduce "spring-in" for the panel and result in a part with better dimensional fidelity. A potential design choice would be to match the through thickness CTE of the core to that of the tool, which would eliminate the need for geometric tool compensation. The value of the presented simplified analysis is to develop fundamental understanding of the main drivers of the problem and how the process-induced deformation is affected by changes in part, tool, and process parameters. The simplified analysis also serves to guide and verify the development of more complex process models. With the insight gained from this project the team has a much better understanding of how to perform approximate process analysis of honeycomb panels as well as how to set up simple yet high fidelity finite element models of this class of problems.

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