

The Effect of Machining–Induced Micro Texture on Lightning Current Arcing between Fasteners and Composite Structure

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ABSTRACT

Drilling fastener holes in composite is much more difficult than in aluminum or other metallic materials since individual carbon fibers fracture at irregular angles resulting in numerous microscopic voids. These voids can trap excess sealant inhibiting the intimate electrical contact between the fastener and the composite structure. As the cutting tool wears there is an increase of surface chipping and an increase in the amount of uncut fibers or resin. This condition is referred to as machining–induced micro texture. Machining–induced micro texture has been shown to be associated with the presence of arcing between the fastener and the composite structure during lightning strike tests.

Lightning protection of composite structure is more complex due to the intrinsic high resistance of carbon fibers and epoxy, the multi-layer construction and the anisotropic nature of the structure. The inherent conductivity of metallic fasteners and the large number of fasteners used in aircraft construction combine to create a condition of a high probability of lightning attachment to fasteners. Intimate contact between bare metallic fastener and the hole in the composite structure has been found to be the best condition for the electrical current dissipation

Improved machining techniques could improve machining-induced micro texture and reduce the

development of microscopic voids. Fasteners developed in this study which conform to the inherent machine–induced micro texture offer another approach for reducing arcing between the fastener and the composite hole. The test results presented in this paper indicated that conforming fasteners decrease the voltage drop across the interface and reduce the dielectric effect caused by the sealant thus minimizing the possibility of arcing between the sleeve and the composite panel.

INTRODUCTION

Continuous fiber reinforced composites are seeing an expanded use in the design of aircraft components for variety of applications where the light weight, higher strength and corrosion resistance are primary concerns. Composites are typically made up of fine fibers such as carbon or glass that are oriented at certain directions and surrounded in a supportive matrix material. Although a wide variety of matrix materials are commercially available, elevated temperature cured epoxy resins are by far the most commonly used. In most component design, the plies of the composite material are arranged at a variety of angles depending on the direction of major loading. This manufacturing technique produces a stacked laminated structure which is highly anisotropic and structurally inhomogeneous. These near net-shape fabricated composite parts still need to be drilled in order to facilitate joining the components using mechanical fasteners. Machining fastener holes in composite

material can be quite problematic. While machining ductile materials such as steel or aluminum results in shearing of the material and the formation of consistent chips, machining of composites involves a more complex mechanism with fracture of the fibers under compression or bending combined with shearing of the matrix material [1-2]. Drilling fastener holes in composite does not compare to the uniformity of aluminum or steel, which can be expected to remain consistent from a drill's entry point to its exit. Individual carbon fibers fracture at irregular angles and form microscopic voids between the sleeve and the hole. As the cutting tool wears these are an increase of surface chipping and an increase in the amount of uncut fibers or resin and delamination [3]. The most common defects observed in drilled composite holes are fiber fractures, fiber pull-outs, chipping, drill entry and exit delaminations, voids and pitting [4-7]. The composite microstructure containing such defects is referred to as machining-induced micro texture.

In addition to their machining challenges, the composite structures in aircrafts are more susceptible to the lightning damage compared to metallic structures. Metallic materials such as aluminum are very conductive and able to dissipate the high currents resulting from a lightning strike. Carbon fibers are 100 times more resistive than aluminum to the flow of current. Similarly epoxy, which is often used as a matrix in conjunction with carbon fibers, is 1 million times more resistive than aluminum [8]. The composite structural sections of an aircraft often behave like anisotropic electrical conductors. Consequently, lightning protection of composite structure is more complex due to the intrinsic high resistance of carbon fibers and epoxy, the multi-layer construction and the anisotropic nature of the structure.

Some estimates indicate that on average each commercial aircraft in service is struck by lightning at least once per year [8]. An aircraft generally experiences a lightning event when flying through a heavily charged portion of a cloud. In such cases the lightning discharge typically originates at the aircraft and extends outward

from the aircraft. While the discharge is occurring the point of attachment moves from the nose of the aircraft and into the various panels that comprise the skin of the aircraft. The discharge usually leaves the aircraft structure through the empennage. Since commercial aircraft contain relatively large amounts of fuel and also include very sensitive electronic equipment, they are required to comply with a specific set of requirements related to the lightning strike protection in order to be certified for operation.

Although there have been substantial amount of works done on the effects of machining induced defects on the mechanical strength of the composite structures there is not much research done on the effects of such defects on the lightning current behavior of a composite joint, fastener arcing in particular [9-11].

Enhanced machining techniques could improve machining-induced micro texture and reduce the development of microscopic voids [12, 13]. Fasteners developed in this study which conform to the inherent machine-induced micro texture offer another approach for reducing arcing between the fastener and the composite hole. This paper presents the direct lightning test results obtained with conforming fasteners and how the machining induced hole defects affect the arcing phenomena which occurs between the fastener and the composite structure during lightning testing.

I Experimental Procedure

1.1 Material

18.5X18.5X0.25 inches panels as shown in Figure 1 were prepared from two different composite materials. The first group of panels was fabricated using woven type graphite/epoxy (Gr/Ep) laminates. The second group was manufactured using unidirectional Gr/Ep plies with stacking sequence of [(45/135/0/0/90/0/135)]. They were vacuum bag molded and then cured at elevated temperature using an autoclave.

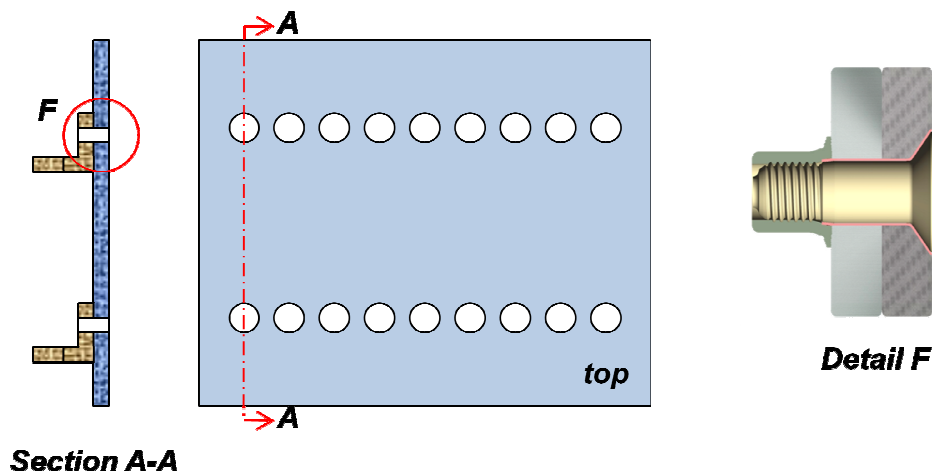


Figure 1: Composite test plates and stringer assembly

1.2 Drilling and Assembly

18 holes per panel (two rows of nine holes) were drilled using PCD (polycrystalline Diamond) coated twist drills and countersink drills. PCD coated drills were chosen since they been demonstrated to yield better hole quality and wear less compared to the conventional carbide or HSS tools [12]. The top plate, the stringer and the aluminum back-up plate were clamped tightly and drilled simultaneously.

All the drilling was conducted without coolant using a vertical mill center at a constant spindle speed of 2500

rpm and a feed rate of 3 in/min. Chips and debris created during drilling were continually removed using a vacuum system. Drills were inspected visually for any noticeable wear or chipping prior to drilling each panel. After drilling, the fastener holes were cleaned to remove any residual debris. Finally each hole was inspected and measured using a bore gage. Hole dimensions were matched to minimum interference requirements specified by the fastener manufacturer. The target hole dimension was 0.2695"- 0.2700" in diameter. Table I lists the typical hole inspection data obtained from these drilling trials.

Table I: Dimensional inspection data of drilled composite test plates

Panel 1 Unidirectional Composite				Panel 2 Woven Composite			
Fastener No.	Counter Sink Ø (in.)	Top Panel Hole Ø (in.)	Stringer Hole Ø (in.)	Fastener No.	Counter Sink Ø (in.)	Top Panel Hole Ø (in.)	Stringer Hole Ø (in.)
1	0.448	0.2673	0.2686	1	0.4530	0.26960	0.26965
2	0.449	0.2673	0.2686	2	0.4530	0.26965	0.26965
3	0.447	0.2673	0.2686	3	0.4530	0.26955	0.26965
4	0.448	0.2672	0.2686	4	0.4520	0.26960	0.26965
5	0.447	0.2673	0.2687	5	0.4540	0.26960	0.26965
6	0.448	0.2672	0.2686	6	0.4520	0.26960	0.26965
7	0.447	0.2673	0.2686	7	0.4530	0.26960	0.26965
8	0.448	0.2673	0.2686	8	0.4530	0.26960	0.26965
9	0.449	0.2672	0.2686	9	0.4530	0.26960	0.26960
10	0.449	0.2673	0.2686	10	0.4535	0.26960	0.26965
11	0.449	0.2672	0.2686	11	0.4530	0.26955	0.26955
12	0.448	0.2673	0.2685	12	0.4540	0.26965	0.26965
13	0.448	0.2673	0.2686	13	0.4535	0.26955	0.26955
14	0.448	0.2672	0.2685	14	0.4530	0.26960	0.26960
15	0.449	0.2672	0.2686	15	0.4530	0.26960	0.26965
16	0.446	0.2672	0.2686	16	0.4550	0.26955	0.26955
17	0.448	0.2673	0.2686	17	0.4530	0.26955	0.26965
18	0.449	0.2672	0.2686	18	0.4530	0.26955	0.26960

All the test panels were assembled prior to paint. An inter-fay sealant was applied between top plate and spar. Fasteners were wet assembled by applying sealant to the sleeve and pin countersink. The combined stack-up thickness was 0.452" which represents a minimum grip installation condition. The fastener system used for spar/panel assembly were

radial expansion sleeved fasteners where the pins were made of Ti-6Al-4V, collars were Ti-3.5Al-2V and the sleeves were A286 stainless steel. Figure 2 shows the general view of the cross sections and the installation sequence of radial expansion sleeved fasteners.