Safety and Airworthiness Verification on a Typical Civil Turbofan Engine*

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Abstract: Based on a typical civil aircraft powerplant unit—the PX10 turbofan aero engine, this paper elaborates the primary test demonstration and safety analysis process in terms of bird and hail ingestion, vibration, overspeed, and blade containment, etc. The entire airworthiness verification process covers the load spectrum analysis, stress calculation and physical tests under simulated and real operating conditions, and all the results show the compliance to the success criteria of each item of FAR Part 33, which implies the sufficient safety margin was achieved during the design and manufacture procedures of PX10 engine. Since there are few successful examples in indigenous aero engine certification process, the illustrations presented here could be a helpful guideline to the engine airworthiness certification procedure of domestic engine manufacturers.

Key words: Engine safety; Airworthiness Verification; Method of compliance; Overspeed; Containment

CLC number: V235.1
Document code: A
Article number: 1001–4055 (2022) 01–200415–23
DOI: 10.13675/j.cnki.tjjs.200415

Introduction
Operational safety is always the most critical issue of gas turbine engines for civil transportation applications. During the development procedure of each type of aero engine, the inherent safety character is selected as the principle consideration rather than engine technical superiority and performance, and this point is usually verified by a series of strict rig tests and safety analysis procedure. Although there are still some acceptable engine failures occurring in the commercial daily flight route, the verification work is rigorous enough to

* Receive date: 2020–06–04; Acceptance date: 2020–08–12.

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strengthen the confidence of both the public and aircraft operators.

The safety requirements of aero engine issued by the relevant authorities are now known as the airworthiness regulations, which are the direct outcome of all the lessons learned from engine failures in human aviation history. There are two authoritative issued engine airworthiness specifications popularized in the world, i.e., the FAA Part 33\(^1\) and the EASA CS–E\(^2\). In spite of some minor difference presenting between the two standards, they are harmonizing with each other all the time to achieve an ultimate goal of engine safety\(^3\). Based on the requirements of airworthiness, the aero engine must show its compliance to the applicable rules by running under different critical conditions, including the icing, rain, overspeed, and vibration. Only the ones that pass all the proof tests successfully can obtain the type certificate and be approved to enter into service.

Over the past two decades, many efforts have been dedicated to probe into the aero engine failure mechanism and malfunction protection. The fruitful research results reveal most of root causes of engine failure and have played an important role in the continuous airworthiness period. In view of different directions of research, some meritorious studies are summarized herein.

1.1 Engine cooling

The actual cooling effect is a vital consideration for aero engine installation and hot section protection. In early years, Campbell\(^4\) provided an empirical correlation to evaluate the effects of each of engine parameter on the cooling requirements. Pinkel\(^5\) treated the engine as a heat exchanger and correlated the general heat exchanger parameters to the engine operating conditions. Based upon the NACA cooling correlation analysis, Wardley et al.\(^6\) demonstrated a flight test procedure to develop a more effective air-cooled engine installation configuration, and their proposal was well verified by a Piper PA–41P aircraft with Lycoming TIO–540 engine installation. As a further study, Ward and Miley\(^7\) presented a similar modification to the traditional NACA cooling method to show compliance to the engine cooling requirements. Since the gas turbine engine has played an important role in the transportation aviation history, people’s more attention is paid to the cooling technique of engine hot section. Based on the adaptive response surface method, Wang et al.\(^8\) developed an optimization technique to investigate the engine cooling problem. Compared with the plotted objective function, this technique provided reasonable prediction to improve the thermal effectiveness of aircraft de-icing strategy. To investigate the effusive cooling effect on gas turbine hot section components, Cerri et al.\(^9\) presented a numerical model to evaluate the geometric and thermodynamic features. Finally, they concluded two cooling solutions for the first stage turbine blades. Recently, Yang et al.\(^10\) performed a parametric study to assess the general applicability of inlet air cooling systems. The discussion on off-design point performance presented in the results may bring about new inspiration for the selective mode of engine cooling.

1.2 Rotor integrity

To maintain the engine structural integrity is the primary requirement in rotating system design, and lots of studies are focused on this research scope. Oakley and Newell\(^11\) described an elastic predictive approach to forecast the residual fatigue life on aircraft engine blades after foreign object damage. Verified by the Ti6Al4V blade impact experiments, the prediction presented satisfactory accuracy. Bhaumik et al.\(^12\) carried out a systematic analysis to the crack failure on the blade trailing edge of a turbine rotor blisk, and Dhandt and Kohl\(^13\) examined the effects of different inner to outer radius ratios and stress level ratios on the dynamic failure of engine rotating disks. By using different residual compressive stress profiles, Chan et al.\(^14\) attempted to counteract the undesirable effects of contact stresses and to mitigate the fretting fatigue in the blade disk attachment region. They combined the contact stress and residual stresses into a probabilistic crack growth model, and predicted the risk of disk failure under simulated loading conditions. Aiming at the service life prediction of turbine rotor, Millwater and Osborn\(^15\) developed a methodology to determine the sensitivities of probability of fracture on turbine disk design. This study was directly derived from the engine industrial demand. In addition to the above achievements, some other valuable conclusions on the fatigue lift prediction and damage tolerance evaluation of rotor disks can be also found in the
open literatures [18-21].

1.3 Foreign object damage

Objects such as sand, birds, small stones and other debris may be sometimes ingested into the engine inlet which damage the fan and compressor rotor, and this is treated as a FOD event. A famous case showing the huge risk caused by the engine FOD is the A320 Hudson river forced landing event, 2009 [22]. In early 1990s, the FAA published a detailed survey to the engine bird ingestion events [23], and issued an operational procedure to avoid the occurrence of severe engine decay. Howard [24] presented a computational tool to predict the effects of bird ingestion, and demonstrated its accuracy by actual engine testing. Teichman and Tadros [25] used both the analytical and experimental program to investigate the FOD resistance capability of fan blade on a small turbofan engine, and Mao et al. [26] calculated the nonlinear transient response of bird–striking fan system by employing the explicit 3–D finite element code LS–DYNA. Except for the bird threat, some other studies presented the effects of rain and hail ingestion on engine performance and operational safety [27-29], and even the water ingestion [30-31].

1.4 Icing condition

Safety concerns over engine icing have risen for a long time since 1980s. Willcocks [32] provided a review of engine icing conditions, and examined various methods of icing prevention. This study presents the earlier icing attention from industrial engine design project. Through the wind tunnel tests, Venkataramani et al. [33] investigated the ice accretion on the booster stage airfoils, and Rowley et al. [34] studied the icing evolvement on the anti–icing nacelle inlet of a tri–engine helicopter. The two works illustrate typical approaches on engine icing exploration adopted by the engine manufacture. As the effect of icing crystal on engine performance becomes more attractive, some researchers present their different computational models to predict the particular icing accretion phenomena on the engine core module [35-36]. Recently, Yang and Zhou [37] presented the normal airworthiness certification process of engine icing based on a special turbofan model. Through introducing the critical point analysis and icing test, this study provided an acceptable guidance to show compliance with the pertinent regulation.

1.5 Blade containment

As a specific item of aero engine certification requirement, an engine must demonstrate its capability to contain a critical rotor blade which is released at full operating speed. Based on the industrial experience, Horsley [38] described the fan blade–off test of RB211–535E4 aero engine which employed an innovative lightweight aluminum/Kevlar casing structure. The engine model now powers the Boeing 757 aircraft for daily operation. Naik [39] and Stahlecker et al. [40] compared the containment effects of three different high strength woven fabrics cases with Kevlar, Zylon 500D and Zylon1500D, and also developed a constitutive model to predict the ballistic impact with the help of commercial software LS–DYNA. In fact, the utilization of composite rotor case has become the development trend for next generation engine. Sarkar and Atluri [41] employed a finite element analysis to investigate the containment aspects of rotor blade fragments. They inspected the collision impact caused by both a single fragment and several blades, and verified the numerical results with the experimental data obtained from the rotor spin rig test. Aside from the above mentioned, more studies focusing on the engine blade containment capability by test approach are continued [42-44].

The literatures discussed above have covered most of aspects on the aero engine safety requirements and demonstration. From the airworthiness point of view, each study only emphasizes a given aspect of consideration for aero engine design rather than the comprehensive safety evaluation. Based on a special aero engine model, the present work aims to illustrate the primary rig tests and safety analysis during the type certification procedure, and elaborate the methods of compliance to each applicable regulation item. It will be a helpful guide line for the domestic industrial community to understand the airworthiness rules, and pass the series of final demonstration tests successfully.

2 Research methodology

The FAR Part 33 is generally accepted as the most authoritative standard for civil aero engine airworthiness certification all around the world. This regulation is is-
sued by the Federal Aviation Administration of US government, and revised continuously to promote the safety level of aero engine certification. Many countries adopt this rule as the reference to formulate their own engine approval specification. The Part 33 regulation requires each type of engine to exhibit its failure tolerance ability under different cruel conditions, such as the bird ingestion, vibration, icing endurance, and fan blade-off situations. This verification process usually leads to a severe trial to the product manufacturers. On the basis of given engine model, the current study will also select the FAR Part 33 as certification basis to reveal the main test demonstrations during the airworthiness approval procedure.

A typical two-spool turbofan engine named PX10 was selected for the airworthiness verification, as shown in Fig. 1. By employing a gear box, the engine fan assembly rotates counterclockwise while the high pressure (HP) and low pressure (LP) spools rotate clockwise when viewed aft looking forward. Each spool is mounted on two bearings, a ball bearing to react the thrust and radial loads, and a roller bearing to support radial loads and tolerate the differential axial thermal growth between the supporting structures and rotating components. An annular splitter, located behind the fan serves to separate the fan air into bypass and core flow. The core air flows through the four-stage axial LP compressor, single-stage centrifugal HP compressor and then enters into the reverse-flow annular combustor. The engine hot section consists of a single-stage HP turbine and three-stage LP turbine, and all the accessories such as the fuel pump, hydromechanical fuel control, and oil pumps are supported and driven by the engine main gearbox. As a modern engine model, the PX10 is equipped with a digital electronic control unit and relevant required sensors, which can facilitate the engine performance management while achieving a high level of fuel efficiency.

The PX10 engine has a maximum takeoff thrust of 24kN at the static, sea level conditions up to ambient temperatures of 25℃. Moreover, an automatic performance reserve (APR) rating will be provided to allow an additional thrust augmentation for one engine inoperative operation. The engine steady state limits of \( n_1 \) and \( n_2 \) are 21000r/min and 31485r/min, respectively. To improve the performance and durability of PX10 model, the typical advanced techniques applied are: (1) optimization of tangential on board inducer (TOBI), which provides increased cooling to the HP turbine; (2) single crystal blades for the HP and first-stage LP turbines; (3) advanced aerodynamic and mechanical fan design.

3 Verification process and results

3.1 Engine cooling

Item 33.21 of the FAR Part 33 requires that engine design and construction must provide the necessary cooling effect under conditions where the airplane is expected to operate. To meet this rule, the worst condition combined with temperatures and stresses for the engine hot section need to be declared in the first place. Fig. 2 shows the identification of PX10 engine station, and the engine operating limits for the mechanical design point (MDP) and three other conditions are listed in Table 1. The MDP status of PX10 is the sea-level maximum-power takeoff condition on a break point ambient temperature of 25℃, with the engine turbine inlet temperatures and spool speeds set to their respective rating limits.

![Fig. 2 Engine station marks of PX10](image)

All the data presented in Table 1 denotes that the MDP status is the most critical to the turbine compo-
ments among all the operating conditions. The difference between the turbine gas path temperature and cooling air sources ($T_1$ and $T_{2a}$) is the largest, which implies that the driving potential between the source and sink coolant temperatures for the turbine is higher at MDP condition, resulting in higher turbine rotor temperatures at the rim, and much greater temperature differences and the thermal stresses. Therefore, the MDP operation is selected for both the compressor and turbine transient thermal analysis and the subsequent low cycle fatigue (LCF) calculations.

To confirm the ground operation with the hottest turbine disk cavity gas temperatures and demonstrate proper function of the turbine cooling system throughout the flight envelope, an internal cooling integrity flight test was employed. The standard gas temperature thermocouples and necessary pressure instrumentations were installed at the HP and LP turbine interface in the flight tested engine to verify that the crucial cooling flows at the interface do not drop appreciably at different altitudes. The configuration of flight test bed and associated engine installation is displayed in Fig. 3.

Table 2 summarizes the HP turbine disk aft cavity gas temperatures measured under engine maximum continuous power and at two radial locations for the flight test points. These temperatures decline with the increment of altitude, and commensurate with the reduced $T_{45}$ and $T_{50}$. Note that the outer radius gas temperature for ground operation at the height of 0 ft is hotter than the intended value. This discrepancy is attributed to an axial mismatch of 0.244 cm between the disk and static structure at the disk rim flow discourager. It is apparent from the data in Table 2 that ground operation defines the hottest HP turbine disk aft cavity gas temperatures, and that no detrimental changes in cooling performance between the engine ground and altitude operation. This also matches the engine actual operational characteristics.

The engine internal cooling integrity is also substantiated by the measured first-stage LP turbine disk forward cavity gas temperatures for the flight test points as well as the scaled temperatures for the maximum allowable altitude $T_{45}$ of 991.1°C, as shown in Table 3. The engine ran under maximum continuous power, and both the measured and scaled worst condition disk cavity gas temperatures decline with the increasing altitude. It can be found that the ground operation defines the highest first-stage LP turbine disk cavity temperatures, and that moderate gas temperatures are maintained between the ground, altitude and custom bleed operation.

The transient and steady state thermal analysis

<table>
<thead>
<tr>
<th>Condition</th>
<th>MDP</th>
<th>SLS/ISA</th>
<th>Maximum compressor temperature</th>
<th>12.2 km cruise</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ambient temperature°C</td>
<td>25.6</td>
<td>15.0</td>
<td>51.7</td>
<td>-56.7</td>
</tr>
<tr>
<td>Mach number</td>
<td>0</td>
<td>0</td>
<td>0.6</td>
<td>0.8</td>
</tr>
<tr>
<td>$F_2/N$</td>
<td>24000</td>
<td>24000</td>
<td>9088.9</td>
<td>5200</td>
</tr>
<tr>
<td>$n_1$, HP/(r/min)</td>
<td>31900</td>
<td>31348</td>
<td>31900</td>
<td>30158</td>
</tr>
<tr>
<td>$n_1$, LP/(r/min)</td>
<td>21500</td>
<td>20856</td>
<td>19546</td>
<td>20824</td>
</tr>
<tr>
<td>$T_{2a}$/C</td>
<td>67.8</td>
<td>56.1</td>
<td>110.0</td>
<td>12.2</td>
</tr>
<tr>
<td>Maximum $T_{2a,ef}$/C</td>
<td>271.7</td>
<td>252.8</td>
<td>278.3</td>
<td>213.3</td>
</tr>
<tr>
<td>$T_3$/C</td>
<td>490.0</td>
<td>465.0</td>
<td>495.6</td>
<td>415.6</td>
</tr>
<tr>
<td>$T_{45}$/C</td>
<td>1237.8</td>
<td>1190.6</td>
<td>1226.1</td>
<td>1136.1</td>
</tr>
<tr>
<td>$T_{50}$/C</td>
<td>1021.7</td>
<td>980.6</td>
<td>1011.1</td>
<td>935.0</td>
</tr>
<tr>
<td>$p_r$/kPa</td>
<td>1916.7430</td>
<td>1916.7430</td>
<td>1572.0050</td>
<td>668.7915</td>
</tr>
</tbody>
</table>

Fig. 3 Engine flight test bed for altitude operation

Table 1 PX10 engine operating conditions
were performed for the turbine and compressor rotors, and the FAA service cycle shown in Fig. 4 was used for all the transient thermal analysis. Thermal models of the HP spool and LP turbine were calibrated against temperature plug data taken at the steady state maximum power conditions. The convective gas temperature boundary conditions and their transient response behavior were obtained from the full engine thermal survey data. In order to obtain metal temperatures for the HP and LP turbine rotors, steel pins of a known hardness were embedded into the parts at locations of interest. After a 60 min steady state engine dwell at maximum power, the hardness of each pin was measured again. Through metallurgical correlations, the change in pin hardness was related to the operating metal temperature experienced at the particular location. Thus, the steady state temperature distribution was obtained at the takeoff power condition. Under the ideal test conditions, the temperature sensitive plug uncertainty is around ±2°C.

Fig. 5 and 6 show the HP compressor and turbine temperature plug results for the tested engine. Imposed on these plots are the disk temperature predictions for the operating condition at which the test was run. The numerical simulation was carried out by using the commercial software CFX with calibrated SST near wall function. Note that the thermal model agrees quite well with the temperature plug data. The only significant disparity occurs at the HP turbine disk forward rim. The thermal prediction produces a hotter rim and a somewhat larger bore thermal stress than the engine test data, which implies that the more conservative analytical result was therefore used in life predictions.
All the temperature data obtained from the thermal analysis model and engine test are within the endurable capabilities of engine materials. In other words, the cooling effect of engine secondary air system is fully realized so that each component of hot section works well during the normal engine operation.

The PX10 engine external cooling system which is shown in Fig. 7 provides the necessary cooling air to maintain zone I and II component temperatures within allowable operating limits. The zone I is comprised of all the cavities between the engine bypass duct and fan case. The main components in zone I are electronic engine control (EEC) unit, starter and associated accessories. The ventilation of air in this cavity is from the leakage across assembly gaps in the nacelle and flanges.

Zone II is comprised of the engine core section and its associated harnesses, valves, lubrication plumbing, and the pressure bleed tubing. An air supply to zone II is from the bypass air through six thumbnail scoops located strategically on the core fairing.

The external engine cooling test was conducted by using the flight test bed as shown in Fig. 3. During the test procedure, the flight test bed provided sufficient altitude environment for data obtaining to demonstrate compliance with the requirements of item 33.21. Peak component temperatures are presented in Table 4, along with the projected 51.7°C hot day values and allowable limit temperatures. This table shows that the hot day component temperatures are within acceptable limitations.

### 3.2 Low cycle fatigue analysis

A cycle shown in Fig. 4 consisting of 30 s acceleration from zero speed to ground idle (GI), 5 min at GI, 6s acceleration from GI to sea level takeoff (SLTO) condition, continuous operation at take off until steady-state is achieved, 15s deceleration to GI, 5min at GI, 60s deceleration to shut down, and 10min of soakback was selected for the transient analysis for low cycle fatigue (LCF) of the PX10 engine. For the LP and HP rotors LCF analyses have been performed at the MDP condition. Cycle analyses of the PX10 engine through the flight envelope reveal that the chosen MDP drives the engine closer to the limits than any other operating
point. This MDP, additionally, provides the largest temperature differences at sea level between $T_{L4s}$ and $T_{L2}$; affecting thermal stresses in the LP compressor; between $T_{H4}$ and $T_{H4s}$ affecting thermal stresses in the HP compressor; between $T_{A1}$ and $T_{A2}$ affecting thermal stresses in the HP turbine, and finally, between $T_{4}$ and $T_{24s}$, affecting thermal stresses in the LP turbine.

For the LCF analysis of turbine components, both the steady-state and transient thermal analyses were performed under the MDP status. Fig. 8 displays the results of steady-state thermal analysis for the HP turbine, and the transient temperature results are plotted along with the stresses for each potential critical location in Fig. 9. Since the stress analysis forms the basis of component life prediction, the similar procedures are also applied to the LP turbine, engine shafts and compressor critical parts.

To obtain accurate boundary conditions for each LP compressor rotor and to get bore and web stresses, an entire LP compressor rotor was modeled for the SLTO 25°C day condition, as shown in Fig. 10. This model includes all the disks, shrouds, and seals in 2 dimensions and blades in 3 dimensions, and the units used for load and interface expression are lb. and inch, respectively. The thermal and mechanical boundary conditions were applied to the finite element model to conduct stress analyses for stresses in the bore, web, and other axisymmetric regions.

It was determined that maximum bore and web stresses occurred during the transient takeoff condition.

### Table 4 Engine external cooling test matrix

<table>
<thead>
<tr>
<th>Component</th>
<th>Altitude/km</th>
<th>Mach number</th>
<th>Power setting</th>
<th>40.6°C day measured takeoff temperature/°C</th>
<th>Projected hot day takeoff temperature/°C</th>
<th>Allowable maximum temperature/°C</th>
</tr>
</thead>
<tbody>
<tr>
<td>EEC box</td>
<td>0</td>
<td>0</td>
<td>Idle, max takeoff</td>
<td>70.6</td>
<td>79.4</td>
<td>82.2</td>
</tr>
<tr>
<td>Fuel control unit</td>
<td>0</td>
<td>0.20</td>
<td>Takeoff, APR</td>
<td>101.1</td>
<td>109.4</td>
<td>287.8</td>
</tr>
<tr>
<td>Fuel nozzle</td>
<td>1.52</td>
<td>0.40</td>
<td>Takeoff, climb</td>
<td>121.7</td>
<td>130.6</td>
<td>176.7</td>
</tr>
<tr>
<td>Angle gear box housing</td>
<td>3.05</td>
<td>0.40</td>
<td>Climb, max power, cruise</td>
<td>128.9</td>
<td>137.8</td>
<td>176.7</td>
</tr>
<tr>
<td>Ignition excitor</td>
<td>4.57</td>
<td>0.43</td>
<td>Climb, cruise, idle, windmill</td>
<td>76.7</td>
<td>85.0</td>
<td>121.1</td>
</tr>
<tr>
<td>Rear mount bracket</td>
<td>6.10</td>
<td>0.53</td>
<td>Climb, cruise, idle, windmill</td>
<td>262.8</td>
<td>271.7</td>
<td>371.1</td>
</tr>
<tr>
<td>ITT harness</td>
<td>9.14</td>
<td>0.60</td>
<td>Climb, cruise</td>
<td>126.9</td>
<td>135.8</td>
<td>176.7</td>
</tr>
<tr>
<td>$n_1$ monopole</td>
<td>11.00</td>
<td>0.52, 0.70</td>
<td>Climb, cruise</td>
<td>130.7</td>
<td>140.1</td>
<td>190.6</td>
</tr>
<tr>
<td>Air oil separator</td>
<td>13.72</td>
<td>0.60, 0.70, 0.78</td>
<td>Climb, cruise</td>
<td>144.4</td>
<td>153.3</td>
<td>232.2</td>
</tr>
</tbody>
</table>

**Fig. 8** HP turbine steady-state temperatures at MDP (K)
whereas the minimum stresses occurred during the transient thermal soakback condition. Fig. 11 shows the LS-DYNA predicted LP compressor rotor transient effective stresses at 36s after takeoff.

Table 5 presents the predicted LCF lives for each engine critical parts. Through combining the previous calculated stress distributions with the crack propagation properties of base material, the cycle life based on minimum ($-3\sigma$) material properties of each part can be deduced reasonably. For the engine APR power rating application, a scaled factor originated from the previous service experience was added to predict the lives of turbine components. The start-stop cyclic tests and engine service history have also proved that all the calculated life values of critical parts are conservative enough so that no crack is found within the stated life cycles of each life-limited component. Actually, the lives of couplings and shafts are unlimited.

### 3.3 Rotor overspeed

The requirement of item 33.27 requests that the turbine, compressor and fan rotors must withstand the overspeed tests for five minutes. Following the test, each rotor should maintain its structure integrity and may not be cracked.

To determine the most critical overspeed requirement for the LP compressor rotor, the most critical flight operating point had to be defined for each engine and overspeed category. Through a detailed analysis to the engine potential failure, it was found that the critical single component failure maximum overspeed for the LP compressor rotors is 24176 r/min, which occurs at 1.22 km and 15.6°C ambient condition. For a dual failure to occur, as required by the item 33.27(f), the fan front shaft failure would be combined with the overspeed solenoid system failing to detect the overspeed. The latter failure is considered to be a dormant failure of the hydro-mechanical system. For the PX10 engine, however, this failure going undetected during a preflight or during normal flight operations has been precluded. The EEC software has been modified to perform a three-step test to completely verify the $n_1$ overspeed shutoff function. This function is checked for failure during preflights, autostarts and continuously monitored during normal operations; therefore, it cannot fail without being undetected. In order to ascertain the overspeed margin of each LP compressor stage, a validated finite-element model was created by using ANSYS approach. The model was analyzed with room temperature average material properties, and included the conservative PX10 rotating seal loads at each rotor pilot. The results of plastic burst analysis show the minimum burst margin of 1.15 occurs at the 4th stage LP compressor under the overspeed condition, which means that the LP compressor rotor has 130% overspeed capability at least when comparing to the normal operational rotating speed.

For the HP rotor overspeed capability verification,
the compressor impeller and turbine disk were tested respectively. These tests were conducted at laboratory ambient conditions in an evacuated whirlpit, and the rotating speeds selected were 120% of the maximum permissible speed at the maximum operating temperature. The necessary corrections for the difference in minimum ultimate tensile strength at the maximum operating temperature and room temperature were also performed during the tests.

Fig. 12 displays the HP impeller and turbine rotor fixtures in the tests. Per the approved test procedure, simulated blades were installed on the turbine disk. These blades produce a lower blade pull load on the disk rim when compared to the production blades and therefore, an additional speed correction factor was applied to the test speed to account for this deviation.

Prior to the overspeed test, the finite element plastic analyses have been carried out on the HP compressor and turbine rotor to ascertain the burst margins of each component. To obtain the conservative result, the maximum engine operating temperatures, minimum material characteristics and adverse tolerance of disk and blades were selected as the worst case conditions in the analyses. Plastic analyses have been shown to be consistent with the measured residual growth data taken during production overspeed testing of the HP compressor impeller (42500±500 r/min), and the HP turbine disk (36500±200 r/min). As part of the assessment of plastic analysis validity, results of specimen tests obtained during the forging substantiation process have been used to define the pedigree of material that has been overspeed tested. Plastic analysis results have been consistent with all the measured data. Analysis results, in the form of maximum disk strain versus rotating speed, were evaluated against a minimum material burst characteristic to define the HP compressor and turbine disk burst speeds. The results indicate minimum burst speeds relative to 32378 r/min (maximum permissible speed) of 126% for the HP impeller and 127% for the turbine disk.

Conclusions deduced from the PX10 engine safety analysis present that the maximum $n_2$ overspeed due to any single point failure will not exceed 110% of its normal speed, and the rotating speed under multipoint malfunction is even less than the above value. Thus, the compliance will be demonstrated by conducting a room temperature whirlpit test to 120% of the maximum permissible speed. The average tangential stress in the disk depends only on speed and is dependent of temperature.
Therefore, to maintain an equivalent burst margin at room temperature, the speed increased to account for the increase in minimum ultimate strength at room temperature, over the minimum ultimate strength of rotor disk material at the average disk temperature and maximum operating temperature condition.

The critical engine operating points, which produce the maximum operating temperature condition, were used to define the rotor overspeed requirements. For the HP impeller, the highest average disk temperatures occur at SLTO breakpoint condition. An average impeller temperature of 351.7°C was used with a maximum permissible rotor speed of 32378 r/min (101.5% speed), resulting in a required test speed of 45113 r/min. For the HP turbine disk, the average temperature is 544.4°C, and an additional speed correction factor was applied to the test speed to account for the simulated blades installation. Finally, the resulting required test speed is 40542 r/min.

Before the test, each rotor was measured at the
bore and rim locations as the basis to determine the final dimensional growth. The rotors were successfully tested for five minutes. At the completion of each spin test, a visual, fluorescent penetrant, and dimensional inspection of the test rotor was conducted. Table 6 illustrates the dimensional variation of HP compressor impeller and turbine disk. Posttest inspection showed that the parts were within the approved dimensional limits for the overspeed condition, and no evidence of incipient failure, or cracking of any rotor was present.

As for the LP hot section, the 3rd stage turbine rotor was chosen to be overspeed tested for five minutes due to its minimum material burst margin of 2.39. The actual tested speed was 27306 r/min, which was 125% higher than the maximum permissible value during the engine normal operation. Posttest inspection indicated that no evidence of latent failure was found, and the disk diametral growths of bore and rim were 25μm and 2.5 μm, respectively.

### Table 6  Bore and rim measurements before and after test

<table>
<thead>
<tr>
<th>Part</th>
<th>Test speed (r/min)</th>
<th>Disk diameter growth/μm</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Actual minimum</td>
<td>Actual maximum</td>
</tr>
<tr>
<td>HP compressor impeller</td>
<td>45200</td>
<td>45500</td>
</tr>
<tr>
<td>HP turbine disk</td>
<td>40550</td>
<td>40800</td>
</tr>
<tr>
<td></td>
<td>bore mid (rim)</td>
<td>612</td>
</tr>
<tr>
<td></td>
<td>(rim trailing edge)</td>
<td>160</td>
</tr>
<tr>
<td></td>
<td>724</td>
<td>500</td>
</tr>
</tbody>
</table>

### 3.4 Bird ingestion

The PX10 engine has a wide chord heavy fan blade design. This damperless wide chord design feature creates a more severe condition for a fan blade-off event, but results in a more robust design for large bird ingestion events. Per the requirement of item 33.76, a 1.81kg bird was ingested by the PX10 engine to demonstrate its durability under unexpected conditions. The primary concern for a 1.81kg bird ingestion event centers on preventing an ingestion related fan blade-off condition from occurring. An ingestion related fan blade-off event would create a large rotating unbalance of the fan rotor. The other classical 1.81kg bird ingestion issues are shock torque loading and compressor ingestion. The damage resulting from these issues is minimal due to the robust design features built into and proven in the current production PX10 engine fleet.

The bird ingestion design methodology includes the engine as a whole and the fan blade design which is considered critical. Before the final test demonstration, six rig tests and three engine tests were completely successful. For the wide chord damperless fan blade of PX10, the most critical region with respect to large bird ingestion is the root section. During the fan blade design process, both the structural intensity and aerodynamic requirements were highly emphasized. A design practice from the previous experience proves that to make the outermost retention area of fan blade much strong can limit the failure mode in bird ingestion to the airfoil and not a full blade-off event. This principle was adopted for PX10 fan blade design.

The mechanism of bird ingestion with relative velocity triangle expression between the bird and blade is shown in Fig. 13. The normal velocity component creates shear and moment loading in the blade, with the root section being the most critically loaded with respect to blade loss. The PX10 engine fan rotor has a lower blade count which leads to a larger passage between blades, and a correspondingly larger amount of bird mass engaged per blade in the momentum exchange. This is offset to some extent, by slower fan rotational speed on the PX10 engine. For a 1.81kg bird event, the airfoil root shear capacity is a more critical feature than its bending capacity in a Ti–6Al–4V fan blade that has a material elongation of 10 % minimum. The blade centrifugal loading will hold the ingestion loads in equilibrium even with a pinned or fully plastic root condition. However, the root must withstand the ingestion shear loads to be in equilibrium. Based on the real material characteristics, a nonlinear finite element stress analysis was also performed on the PX10 engine fan blade and verified the peak plastic bending strains to be less than the material minimum elongation. Results show the maximum strain condition occurs 0.73 ms into the ingestion event and peaks in the airfoil root section at the 1/3 chordal location. Fig. 14 displays the strain distribution with the maximum tensile plastic strain equal to 2.7%, and the maximum compressive strain at 1.8 %. This is well below the 10% minimum material ductility for Ti–6Al–4V material.

The primary criterion invoked to limit the damage
of bird ingestion event is to preclude a fan blade-off condition from developing. This is dominated by the fan design. Protection against main shaft failure due to the transient torque developed during an ingestion event has also been carried over to the PX10 engine. This consists of a shear section in the shaft coupling system between the fan and the LP spool. The shear section is sized the weak link in the shafting system for a fan rotor sudden stoppage or fast deceleration rate. While the shear section exists to protect the main shafting system, the transient torque during a 1.81 kg bird ingestion event is too low to trigger the shear section release. Development, field, and certification ingestion events have not resulted in any shear section or main shaft separations.

Although the above analysis seems to fully exhibit the bird ingestion tolerance, the engine test is still the premier method to show compliance to rule Part 33.76. Compared to the bird ingestion test, the damage resulting from a fan blade-off condition is more severe than the bird ingestion event. Moreover, the secondary damage produced by the rotor unbalance due to fan blade-off is much worse than the total engine damage caused by a 1.81 kg bird ingestion event. Therefore, the bird ingestion rig test was substituted by the fan blade-off test. In fact, this is an alternate approach which is often selected by the engine manufacturer to demonstrate the engine operational safety under the bird ingestion occurrence. If the engine could tolerate all the damage caused by fan blade-off event, it can be deduced that the design philosophy of engine fan module will also meet the airworthiness requirement of 1.81 kg bird ingestion.

3.5 Inclement weather

Item 33.78 of FAR Part 33 requires that each engine should have sufficient safety margin to run in the inclement weather. The PX10 fan section is designed for safe operation under the rain and hailstorm conditions, based on the experience gained in the water and hailstorm ingestion tests. Components in the PX10 engine pertinent to hail and water ingestion are: the spinner, fan blades, bypass duct splitter, downstream vanes, compressor blades and the combustor. The arrangement optimization of these components is a key factor for the engine rain ingestion operation. The special test equipment required for water ingestion testing included a special water cart designed to provide water flow rates up to 265 L/min, as shown in Fig. 15. Water flow nozzles were mounted in front of the engine inlet. Before the formal test, the nozzles were pre-calibrated to provide the water flow rates required for each condition. Adjustment of water flow rates was accomplished through the use of three adjustable solenoid valves on the facility water cart. Different combinations of the solenoid valves allowed multiple flow rates to be pre-calibrated without shutting down the engine between test points.

Prior to the test, all the installed instrumentation was calibrated. The PX10 engine was operated with a typical flight inlet condition firstly to determine the steady state baseline performance. This performance data provided the engine corrected air flow rate versus cor-
Fig. 15 Water ingestion test setup

rected fan speed \(n_t\), which was then used to obtain the minimum required water-to-air ratio for each operating condition.

By using the engine air flow curve, ambient temperature, and ambient pressure, the required water flow rates were calculated for the minimum flight idle, takeoff and 11190 r/min \(n_t\) (minimum flight idle rating at 300 knts) engine power settings. To determine the water cart flow rates in the test, the engine inlet temperature (IT), barometric pressure (PBAR) and the actual physical \(n_t\) speed were recorded at each condition. The PBAR and IT were used to calculate the physical airflow. IT was taken from three thermocouples mounted on the water nozzle manifold upstream of the engine inlet. Secondly, the corrected \(n_t\) speed was calculated for each power setting using \(n_t\) and IT. The corrected air flow was then determined from the corrected airflow curve established during the preset performance calibration. Thirdly, the water flow rates were determined for each power setting based on the required water-to-air ratios. Another 0.25% to the flow rates was added conservatively to account for variations in ambient temperature and flow rig variability during the testing. In addition, a constant water density of 1kg/L was used in the calculations.

The formal test demonstration of engine water ingestion consisted of two steps. The steady-state water ingestion with 4% water-to-air ratio by weight was completed by ingesting water for three minutes at flight idle and takeoff power settings. To inspect the transient reaction, the water was abruptly ejected into the engine for three minutes at the takeoff power, and three minutes later the engine was decelerated to flight idle rating with one-second throttle movement, and then the engine was accelerated to takeoff power again in three minutes. During the test, the engine flameout-relight logic remained enabled, but one igniter was disconnected at the ignition exciter for single-channel dispatch ability. The other igniter was instrumented for the spark detection. Both the engine feedbacks of steady and transient water ingestion were recorded in the engine manual so as to provide a flight instruction to the pilots.

The water flow requirements versus the actual test values are shown in Table 7. As can be seen, the use of preset values to provide the desired water flow rates resulted in slightly higher test flow rates than required. During both the transient and steady-steady water ingestion test, no significant change in engine speeds or thrust was observed. Moreover, no spark “on” indication occurred on the test record while water was being ingested, indicating that the digital electronic engine control unit did not sense a blowout of the combustor. Following completion of the water ingestion test, visual in-

<table>
<thead>
<tr>
<th>Flight condition</th>
<th>IT/°C</th>
<th>PBAR/MPa</th>
<th>Physical speed (n_t)/ (r/min)</th>
<th>Physical airflow/(kg/s)</th>
<th>Water-to-air ratio required%/</th>
<th>Minimum water flow required/ (L/min)</th>
<th>Actual water flow rate/(L/min)</th>
<th>Actual water-to-air ratio%/</th>
<th>Targets</th>
</tr>
</thead>
<tbody>
<tr>
<td>Takeoff</td>
<td>11.1</td>
<td>0.097</td>
<td>20580</td>
<td>78.69</td>
<td>4.00</td>
<td>199.89</td>
<td>207.42</td>
<td>4.51</td>
<td>Steady state</td>
</tr>
<tr>
<td>Flight idle</td>
<td>9.4</td>
<td>0.097</td>
<td>6821</td>
<td>23.8</td>
<td>4.00</td>
<td>60.98</td>
<td>205.9</td>
<td>14.52</td>
<td>Transient</td>
</tr>
<tr>
<td>Flight idle</td>
<td>10.6</td>
<td>0.097</td>
<td>6828</td>
<td>23.76</td>
<td>4.00</td>
<td>60.52</td>
<td>205.9</td>
<td>14.55</td>
<td>Transient</td>
</tr>
<tr>
<td>Takeoff</td>
<td>11.1</td>
<td>0.097</td>
<td>20613</td>
<td>78.42</td>
<td>4.00</td>
<td>200.45</td>
<td>206.28</td>
<td>4.47</td>
<td>Transient</td>
</tr>
<tr>
<td>Takeoff</td>
<td>12.2</td>
<td>0.097</td>
<td>20646</td>
<td>78.07</td>
<td>4.00</td>
<td>200.42</td>
<td>206.28</td>
<td>4.47</td>
<td>Steady state</td>
</tr>
<tr>
<td>Flight idle</td>
<td>6.7</td>
<td>0.097</td>
<td>6788</td>
<td>23.92</td>
<td>4.00</td>
<td>61.28</td>
<td>61.32</td>
<td>4.25</td>
<td>Steady state</td>
</tr>
<tr>
<td>6.1km Flight idle</td>
<td>8.3</td>
<td>0.097</td>
<td>11195</td>
<td>38.75</td>
<td>6.70</td>
<td>155.75</td>
<td>206.66</td>
<td>8.90</td>
<td>Steady state</td>
</tr>
<tr>
<td>6.1km Flight idle</td>
<td>7.2</td>
<td>0.097</td>
<td>11170</td>
<td>38.78</td>
<td>6.70</td>
<td>156.13</td>
<td>178.65</td>
<td>7.78</td>
<td>Steady state</td>
</tr>
</tbody>
</table>
spections of the fan inlet housing, core stator, bypass stator, front frame, spinner and fan rotor assemblies were conducted. No damage or distress of the hardware was observed as the result of water ingestion. Furthermore, the borescope inspection to the LP compressor revealed that no abradable material on the rotor tips was noticed.

According to the rule requirement, five 5.1 cm and five 2.54 cm diameter hailstones with specific gravity of 0.8~0.9 were employed to demonstrate the tolerance capability of PX10 engine. The hailstones were formed in Teflon molds, and stored at -17.8°C in a refrigerator after removal from the molds, as shown in Fig. 16. Sabots were weighed and selected for each hail gun barrel for the test. Before the test, hailstone diameters were measured to the accurate size. The hailstone guns were located in front of the engine test stand for targeting and calibration. A paper target with engine centerline and targets position marked was posted on a wooden board, which was bolted to a metal support. The hailstone guns setup is presented in Fig. 17.

Fig. 16 Prepared 5.1 cm hailstones for PX10 engine hailstone ingestion test

Fig. 17 PX10 engine hailstone ingestion test setup

A sequencer was set up to activate the high-speed cameras and hailstone gun pressure accumulator release valves. The aiming of the gun was adjusted until the projectile impacted the locations. Prior the formal test, several calibration shots were taken from each gun for verification of the hailstone velocity versus gun accumulator air pressure. The hailstone speeds were measured using a laser trip drive to measure the time in milliseconds for the hailstones to travel a set distance of 45.72 cm.

During the hailstone ingestion test, the PX10 engine was installed in the test stand. The size of two 5.1 cm and two 2.54 cm hailstones were verified, and two preselected sabots were loaded with one 2.54 cm and one 5.1 cm hailstone each and weighed, prior to being loaded in the two pneumatic guns. As soon as the guns were loaded, the PX10 engine was started and accelerated to ground idle power, and stabilized for five minutes. The engine was then accelerated to the maximum cruise power of 19334 N thrust condition, stabilized for another five minutes. After that, two 5.1 cm and two 2.54 cm diameter hailstones were fired to the fan. The engine was run “hands off” for an additional 5 minutes after the test, decelerated to the ground idle and finally shut down. Similarly, two additional 5.1 cm and 2.54 cm diameter hailstones were ingested into the engine, and finally one 2.54 cm and 5.1 cm diameter hailstones were thirdly fired to the engine. Posttest inspection of the PX10 engine showed that no damage to fan blades and inlet hardware can be found. A comparative summary of the performance at maximum cruise thrust setting taken from the pre- and posttest data recording is presented in Table 8. The data with a measurement accuracy of ±0.6% shows no sustained power loss during the hail ingestion test.

3.6 Icing

Icing is one of the most serious hazards for the engine safety operation. It can result in certain engine operating anomalies such as surge, stall, rollback or the flameout. Item 33.68 of the Part 33 requires the safety exhibition for the engine induction system icing, and rule 33.77 demands an ice–slab ingestion test demonstration.

For the PX10 engine, components pertinent to icing are the spinner, fan blades, bypass duct splitter, and the downstream vanes. Engine operating parameters pertinent to icing are the ambient temperature, ambient pressure, rotation speed, fan blade tangential velocity, and the fan discharge temperature. Refer to a given oper-
ating condition in the icing envelope, ambient temperature and pressure are constant, while the rotation speed, blade tangential velocity, and fan discharge temperature are variable. Usually, a higher fan discharge temperature will result in the less downstream ice accretion. The higher rotation speed and tangential velocity could lead to smaller amounts of ice accretion on the rotating surfaces, and then smaller ice particles shed from these surfaces. As a result, the critical engine power setting which is prone to icing is the flight idle, when both fan speed and temperature rise across the fan are low. To achieve the satisfied anti-icing effect, a special icing flight idle schedule with a higher fan rotation speed and temperature rise was developed. This schedule is programmed in the engine electronic control unit (EEC), and is activated simultaneously with the nacelle inlet anti-icing system upon the pilot command in operation.

To fully investigate the engine anti-icing effect, the critical point analysis (CPA) was performed within the regulatory icing envelope. At last, seventeen critical points representing in-flight conditions and one ground idle icing point were studied, as shown in Fig. 18. At each in-flight icing point, flight Mach numbers of 0.3, 0.4 and 0.5 were analyzed. This speed range is chosen because air speed lower than Mach 0.3 is not a typical flight speed of any PX10 powered aircraft, and flight speed higher than Mach 0.5 would produce a high enough inlet ram temperature to prevent icing on any engine parts.

Under the well-connected icing flight and ground idle ratings, the fan section operating parameters in the CPA are analyzed. The critical parameters in icing analysis involve the ambient temperature \( T_{\text{amb}} \), ambient pressure \( p_{\text{amb}} \), altitude, Mach number, fan rotation speed \( n_1 \), blade hub tangential velocity \( v_{\text{hub}} \), and the fan discharge temperature \( T_{\text{f}} \). For all the in-flight speeds and ambient conditions studied, \( v_{\text{hub}} \) increases with the augmented Mach number, and which is helpful to restrain the spinner ice accretion and shedding characteristics. Moreover, high values of \( T_{\text{f}} \) imply that the ice accretion on the components downstream of PX10 is less severe. There are significant deviations in fan discharge temperatures at icing points 6, 7, 8 and 15. Further analysis shows that icing condition does not exist for fan downstream components at points 7, 8 and 15, because the values of \( T_{\text{f}} \) in these points are above 4.5°C. The point 6 is not a potentially dangerous icing condition since the aircraft can not sustain level flight of Mach number 0.5 at the 1.98 km altitude with flight idle power. If the engine power increases, the \( T_{\text{f}} \) can easily exceed 4.5°C, which precludes icing on the fan downstream components. More details with regard to the critical point filtration can be found in the study of Yang and Zhou[32].

Based on the above analysis, five critical points that indicate the most rigorous conditions were chosen to

\[
\begin{array}{ccc}
\text{Parameter} & \text{Performance based on strip-chart recordings} & \text{Performance based on performance recordings} \\
\text{Inlet temperature/°C} & 8.3 & 8.9 & 15.0 & 15.0 \\
\text{Fn, thrust/N} & 19334 & 19334 & 20289 & 20347 \\
\text{Tf/°C} & 855.6 & 857.8 & 881.7 & 883.3 \\
n_u/(r/min) & 18750 & 18750 & 19200 & 19200 \\
n_h/(r/min) & 29750 & 29750 & 29480 & 29527 \\
\end{array}
\]
In order to demonstrate the engine operational safety under ice slab ingestion, an ice slab with 73.9 cm length by 5.54 cm width by 0.51 cm thickness was employed as the intrusion object. The size of ice slab was selected on the basis of PX10 engine inlet area, and the test stand was shown in Fig. 19. Prior to the ice slab ingestion test, the engine performance calibration was performed. Several check runs were conducted before the formal ingestion test to ensure the ice release fixture was functional and the integrity of ice slab was maintained until ingested into the engine. With the engine installed in test rig, the ice ingestion test was carried out. The engine was firstly started and stabilized at the idle setting for one minute, and accelerated to the maximum cruise power setting of 19467 N, at prevailing ambient conditions and stabilized for one minute. Various engine parameters, such as the inlet temperature and pressure, engine speeds, HP and LP turbine inlet temperature, thrust, fuel flow rate, compressor exit pressure, fuel pressure and the engine vibration level were recorded.

After the introduction of ice slab, the engine continued to operate “hands-off” mode for five minutes, and there was no change in any monitored engine parameters. At

---

Table 9 ~Selective points of icing test

<table>
<thead>
<tr>
<th>Case</th>
<th>A</th>
<th>B</th>
<th>C</th>
<th>D</th>
<th>E</th>
</tr>
</thead>
<tbody>
<tr>
<td>Reference point</td>
<td>10</td>
<td>15</td>
<td>11</td>
<td>12</td>
<td>18</td>
</tr>
<tr>
<td>Altitude/km</td>
<td>4.69</td>
<td>2.44</td>
<td>6.22</td>
<td>6.71</td>
<td>0.27</td>
</tr>
<tr>
<td>$T_{\text{in,}}/^\circ\text{C}$</td>
<td>-10.0</td>
<td>-10.0</td>
<td>-20.0</td>
<td>-23.3</td>
<td>-1.7</td>
</tr>
<tr>
<td>$P_{\text{in}}$/kPa</td>
<td>56.22</td>
<td>75.10</td>
<td>45.75</td>
<td>42.79</td>
<td>97.84</td>
</tr>
<tr>
<td>Engine flow/(kg/s)</td>
<td>24.8</td>
<td>31.1</td>
<td>21.7</td>
<td>20.8</td>
<td>17.3</td>
</tr>
<tr>
<td>Engine inlet duct velocity/(m/s)</td>
<td>81.4</td>
<td>76.2</td>
<td>84.4</td>
<td>85.3</td>
<td>33.5</td>
</tr>
<tr>
<td>Mean droplet diameter/μm</td>
<td>20</td>
<td>20</td>
<td>20</td>
<td>20</td>
<td>40</td>
</tr>
<tr>
<td>$L_{\text{wc}}$/MI/(g/m$^3$)</td>
<td>2.20</td>
<td>2.20</td>
<td>1.70</td>
<td>1.50</td>
<td>---</td>
</tr>
<tr>
<td>$W_{\text{wc}}$/MU/(kg/h)</td>
<td>22.0</td>
<td>20.0</td>
<td>18.0</td>
<td>16.4</td>
<td>---</td>
</tr>
<tr>
<td>$L_{\text{wc}}$/MC/(l/m$^3$)</td>
<td>0.44</td>
<td>0.44</td>
<td>0.22</td>
<td>0.19</td>
<td>0.60</td>
</tr>
<tr>
<td>$W_{\text{wc}}$/MC/(kg/h)</td>
<td>4.41</td>
<td>4.03</td>
<td>2.33</td>
<td>2.02</td>
<td>1.69</td>
</tr>
<tr>
<td>Exposure time – MC (min;sec)</td>
<td>4.08</td>
<td>4.08</td>
<td>4.13</td>
<td>4.14</td>
<td>30.00</td>
</tr>
<tr>
<td>Exposure time – MI/s</td>
<td>37</td>
<td>37</td>
<td>38</td>
<td>38</td>
<td>---</td>
</tr>
<tr>
<td>Number of cycles</td>
<td>3</td>
<td>3</td>
<td>3</td>
<td>3</td>
<td>---</td>
</tr>
<tr>
<td>Distance from spray nozzle to engine inlet/cm</td>
<td>365.76</td>
<td>365.76</td>
<td>365.76</td>
<td>365.76</td>
<td>530.86</td>
</tr>
</tbody>
</table>

Notes: $L_{\text{wc}}$/MI (Liquid water content—maximum intermittent); $W_{\text{wc}}$/MI (Water catch—maximum intermittent); $L_{\text{wc}}$/MC (Liquid water content—maximum continuous); $W_{\text{wc}}$/MC (Water catch—maximum continuous).
last, the engine was decelerated to idle setting and shut down. All the recorded data indicated that the engine operated satisfactorily during and after the ice slab ingestion test.

The post activities included inspection of the fan and inlet hardware, and the posttest performance calibration of PX10 engine. Posttest inspection of the engine which was shown in Fig. 20 revealed that it was quite hard to find any ice slab impact damage on fan blades, vanes, and the inlet hardware. Meanwhile, no indication of any fracture or nick on the fan blades was found.

3.7 Vibration

Per the requirements of items 33.63 and 33.83, the engine must operate through its declared flight envelope without inducing excessive stress in any engine part due to vibration. Usually, the vibration analysis and rig test are selected as the compliance methods to these rules.

For the vibration calibration of HP rotor, the rig test is considered as the premier choice. The PX10 engine core rig consists of the HP spool, supporting bearings and combustion system. A variable area exhaust nozzle combined with a pressurized impeller inlet was used to better match actual engine cycle conditions. During the test, a series of short speed sweeps were imposed in transient acceleration mode through each operating point of interest, and the exhaust nozzle area was adjusted for each of these sweeps to obtain the closest match to full engine performance. To obtain the stress levels of interested locations, a total of 12 strain gages were installed on the HP turbine rotor, as shown in Fig. 21. The strain gages were positioned in certain areas of peak vibratory stress, as determined previously through the finite element simulation.

The HP turbine blade Campbell diagram with the measured peak-to-peak strains are shown in Fig. 22. 5E ~ 30E present 5 ~ 30 times of basic resonance frequency. It can be seen that the maximum peak-to-peak vibratory strain level measured was 583×10^-4, which occurs at location D. When compared with the allowable stress levels of the single-crystal blade alloys, the measured maximum vibratory stress can be acceptable. After finishing a similar rig test performed to investigate the vibration features of HP impeller, the maximum peak-to-peak vibratory strain level located on the blade was found to be 1.284×10^-3. Because the HP impeller is made of thermal-resistant titanium alloy, this measured vibration level is also within the allowable limitation.

Vibration testing of fan components for the PX10 engine was conducted as well. During assembly of the ground test engine, strain gages and their leads were installed on the fan blades, disk and vanes. The strain gage instrumentation leads from the fan rotor assembly

![Fig. 20 Fan rotor condition after ice slab ingestion](image)

![Fig. 21 PX10 HP turbine blade strain gage locations](image)
were passed to an electrical slip-ring mounted in the fan spinner of the engine. The leads from the slip-ring, were mounted through a three-strut support ring in the plane of the spinner, and then connected to the data acquisition system. When performing the fan vibration surveys, a special inlet configuration consisted of a standard bellmouth, a distortion screen holder and rotator, and the forward slip ring assembly was used, as shown in Fig. 23. It can aid to measure both the clean and distortion effects on the fan blade vibration. Inlet pressure rakes were installed for calibrating distortion screens but were not present during the vibration test.

The clean inlet response for the PX10 fan blade, disk and stator vane was determined by measuring dynamic strain response during two slow accelerations and decelerations of the engine between idle and 105% of maximum physical fan speed with a standard bellmouth inlet. 12 strain gages on the fan blade and two on the fan disk were employed to obtain the strain levels. It was found that the highest strain level on the fan blade during clean inlet testing was 5.24×10⁻⁷ at the bottom of pressure side. The peak response occurred at the fan rotating speed \( n_{\text{max}} \) of 7582 r/min, and the minimum safety margin which was defined as the ratio of allowable alternating stress to the alternating stress was 9.1 at this peak strain location. During the rig test, strain gages were used to monitor the non-synchronous activity associated with flutter. It was found that at no time was the non-synchronous response observed.

The vibratory characteristics of PX10 fan rotor under inlet airflow distortion conditions were determined by measuring the dynamic strain response for the first four harmonics. The inlet distortion was imposed with screens installed in front of the fan with appropriate number of segments to produce the desired distortion patterns. The screens were designed to produce circumferential profiles with the distortion intensity controlled by variation in screen mesh grid. Equipped with the distortion screens, the engine operated from idle to maximum obtainable rotation speed, and the strain gages were also used for the data acquisition. Results showed that the measured strains on fan disk were well below 2.0×10⁻⁴ at any case, which presented an alternating stress of less than 10.34 MPa and were considered as insignificant. The maximum strain level under different distortion grades was 3.685×10⁻⁴ at \( n_{\text{max}} \) of 5580 r/min, corresponding to a minimum safety margin of 1.29.

Except for the above demonstration tests, a comprehensive flutter survey was undertaken in the altitude flight test. During the flight test, the light probes were mounted in the fan inlet housing positioned to view the fan blade leading edge and trailing edge tip passing. A total of six locations were specified, four to view the leading edge at different circumferential locations, and two to view the trailing edge. The PX10 engine was run along normal fan operating lines at extreme corners of the flight envelope. During the tests, engine operation was conducted to 100% fan physical speed, which resulted in corrected fan speeds up to 105% depending on the test conditions. When each desired altitude and Mach number condition achieved, the vibration test was conducted in two slow acceleration–deceleration cycles from idle to maximum physical speed within 90 seconds. Results of several surveys were organized with the use of waterfall plots that show vibration amplitude versus fre-
frequency and fan rotor speed, with excitation order lines displayed. Finally, at no time during flight test was non-synchronous response clearly observed. The results of flight test have demonstrated that the PX10 fan rotor is free from flutter over the entire operating range of the engine.

3.8 Blade containment

To emphasize the containment requirements in engine design, items 33.19 and 33.94 claim that the engine should demonstrate its containment capability when encountering an unexpected blade failure from the fan, compressor or the turbine module. According to the rules, the most critical compressor, turbine or fan blade at its maximum permissible rotation speed must be contained by the casings while the engine should operate safely for at least 15 seconds or self-shutdown. Moreover, the blade must be released at the outermost retention groove or at least 80% blade height. Due to the complicated instrumentation and expensive cost price, the engine fan blade off test is always a reluctant mission for the manufacturer, and scheduled for the last demonstration in the certification process.

Prior to the blade off test, a margin of safety analysis was performed to elevate the blade containment capability for each rotor stage by calculating the blade energy at maximum rotation speed and casing tolerance at its maximum temperatures. The blade was assumed to separate in the outermost retention for inserted blades, and the most probable fragment size for the impeller was determined by the fundamental blade vibration mode. The engine containment capability was then calculated by employing the empirically derived blade containment curves. These curves make use of test data which relates the projectile’s kinetic energy to the material ultimate strength, elongation, and the thickness requirements to contain the projectile. The containment capacities of fan, compressor and turbine modules are detailed presented in Table 10.

Table 10 demonstrates all rotor blades have positive margin to the test line used to determine the containment capability of the casing structure. The minimum margin of 1.42 was found in the HP compressor stage when the impeller was partially released. As previous noted, the impeller blade fragment is assumed to be equivalent to the most probable blade size corresponding to the fundamental blade vibration mode. Based on the engine safety analysis, this unexpected event is considered as extremely impossible failure during the engine route operation. Although the fan module showed a reasonable margin to contain the released fan blade, the fan rotor was still chosen for the engine blade off test demonstration due to the highest unbalance load generated and massive secondary damage to the engine.

The fan blade containment test was conducted on the PX10 engine with final type design configuration. Prior to the test, the laboratory simulating aircraft forward and aft mount beams were used to interface between the bookend test stand structure and the forward and aft engine mount attachment locations. The mount beams are designed to simulate the pylon stiffness. A witness shield was placed around the containment plane to define the trajectory of any uncontained fragments or any released hardware. The shield is constructed of 0.127 cm aluminum sheet and supported off of the book-

<table>
<thead>
<tr>
<th>Stages</th>
<th>Case material</th>
<th>Temperature/℃</th>
<th>Containment capacity/(N·m)</th>
<th>Fragment energy/(N·m)</th>
<th>Safety margin</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fan</td>
<td>Al6061-T6, Kevlar</td>
<td>95.0</td>
<td>39163.5</td>
<td>15668.4</td>
<td>2.49</td>
</tr>
<tr>
<td>LPC1</td>
<td>Ti-6Al-4V</td>
<td>162.2</td>
<td>3853.2</td>
<td>1847.8</td>
<td>2.09</td>
</tr>
<tr>
<td>LPC2</td>
<td>Ti-6Al-4V</td>
<td>203.3</td>
<td>1195.5</td>
<td>677.6</td>
<td>1.76</td>
</tr>
<tr>
<td>LPC3</td>
<td>Ti-6Al-4V</td>
<td>246.1</td>
<td>1562.3</td>
<td>529.3</td>
<td>2.95</td>
</tr>
<tr>
<td>LPC4</td>
<td>Ti-6Al-4V</td>
<td>287.8</td>
<td>2918.2</td>
<td>448.2</td>
<td>6.51</td>
</tr>
<tr>
<td>HPC</td>
<td>In718, Ti-6Al-4V</td>
<td>371.1</td>
<td>1325.5</td>
<td>930.3</td>
<td>1.42</td>
</tr>
<tr>
<td>HPT</td>
<td>Rene 41, In718</td>
<td>990.6</td>
<td>4068.0</td>
<td>2135.5</td>
<td>1.90</td>
</tr>
<tr>
<td>LPT1</td>
<td>M-250, In718</td>
<td>898.9</td>
<td>9982.1</td>
<td>1879.4</td>
<td>5.31</td>
</tr>
<tr>
<td>LPT2</td>
<td>M-250, In718</td>
<td>815.6</td>
<td>16434.4</td>
<td>3121.6</td>
<td>5.26</td>
</tr>
<tr>
<td>LPT3</td>
<td>In718, In625</td>
<td>704.4</td>
<td>5862.4</td>
<td>2957.5</td>
<td>1.98</td>
</tr>
</tbody>
</table>
end test stand and with ground supports. For witness purpose, the test stand side plate was painted to monitor any fragments which might be released at this location. In addition, a grid system was employed as an aid in determining the fragment trajectories, and a 1.5m by 1.5m grid pattern will be taped on the pad which is surrounding the engine. The engine has a simulated aircraft inlet for the fan blade off test. The test inlet simulates a typical aircraft installation inlet weight, stiffness, and the overhung moment at the inlet flange. Following a conservative estimation, the engine was configured with thrust reverser to achieve more severe engine loads. In the meanwhile, the engine nacelle was modified with wall thickness changes to simulate the thrust reverser stiffness. Weights and bulkheads were also used to obtain the correct weight and overhung moment from the engine attachment flange. Fig. 24 displays the PX10 engine blade off test installation, and more details can be referenced in a similar study of Yang and Zhou.

![Fig. 24 PX10 fan blade off test installation](image1)

The blade off test was conducted at the ambient temperature and pressure that produced the maximum permissible $n_1$ speed. In order to release the fan blade on required time, an explosive device was inserted into the hole drilled along the fan blade dovetail center plane to fail the blade at its outmost retention feature. A trigger signal was coupled through a rotating transformer to a control circuit on the fan shaft. The power of detonator trigger was provided by batteries carried in the shaft.

When finishing the calibration procedure for all the recorders, the blade off test started. The engine was firstly accelerated to the idle rating and stabilized for 5 minutes, and then performed a 1min acceleration to the desired $n_1$ speed. After the indication of fan blade release, no throttle movement was made in $15\text{s}$ unless the engine was commanded automatically to shut down by the EEC unit. Immediately, $15\text{s}$ after the fan blade release event, the fuel supply was shut off through the master control panel and let the engine freely run down. Through the test, several leading parameters including the $n_1$, $n_2$, power lever angle, engine fuel flow and compressor discharge pressure were monitored in real-time, and photographs were taken as necessary to document the test setup, results, any special test equipment, and the hardware condition. Video tape and the high-speed photography were also used as a visual supplemental aid, but not to be considered a necessary part of compliance demonstration. Finally, the engine ran “hands off” for a period of $18\text{s}$ after the fan blade separation, at which time the engine was shut down. At the moment of fan blade release, the high-speed photography showed that the inlet spacer of simulated customer inlet flange separated after the initial revolution, and the engine transferred to the manual mode operation. Fig. 25 displays the PX10 fan rotor condition after the blade off test. Except for the fan blade missing, the released blade fractured and a portion of the blade was found forward of the engine. It can be also observed that the rest of fan intact with blade tips curled. In addition, no leakage of fluids from the accessory drains was observed, and the oil tank mounts were still intact. This could be considered as a critical proof to exhibit the structural integrity of PX10 engine.

![Fig. 25 Missing of failed fan blade](image2)

After the test, the fan blade was blocked by the containment housing successfully. Although many secondary damages occurred at different engine locations
during the fan blade failure, such as the curvic damage of LP shaft, the event did not cause the engine to catch fire, separate from its mounts, or lose the capability of being shut down.

4 Conclusion

Based on a typical engine model, the safety evaluation and airworthiness verification procedure are elaborated in this paper. All the analysis and test results in terms of rotor overspeed, foreign object damage, vibration and blade containment and so on meet the stringent standards of FAR Part 33 successfully. Although there are many alternatives to demonstrate the method of compliance to aero engine airworthiness regulation, the approaches presented here are highly recommended in the final engine safety verification. Due to the potential huge consumption of time and cost, it is not necessary for the applicant to seek for other alternative manners since the above illustration could be fully accepted by the FAA and CAAC. The current work maybe further replenished as the research progresses, and it could be used as a guide for the engine manufacturers to pass through the formal engine airworthiness certification procedure.

Acknowledgement: With heartfelt appreciation, the author would like to express his thankfulness to Mr. Zhou Yanpei for his encouragement in the past 10 years, and also to the support of Engine Certification Center of CAAC and Commercial Aircraft Corporation of China, Ltd. (COMAC).

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(编译：刘萝威)