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United Airlines Flight 232: The Impossible Landing

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Executive Summary

On July 19, 1989, after a catastrophic engine failure and loss of hydraulic flight controls, United Airlines Flight 232 crashed in an attempted emergency landing at Sioux Gateway Airport. The pilots were able to maneuver the plane to the closest runway; but, 111 of the 296 people on board were fatally injured. The underlying cause of failure was the fatigue crack propagation of a 13mm long crack on the fan disk of the tail engine that was not detected during inspections. Had proper inspection techniques been implemented and a defect tolerant approach been taken when calculated the service life of the fan disk, the accident could have been avoided.

1 Introduction

United Airlines (UAL) Flight 232 (N1819U) was en route from Denver, Colorado to Philadelphia, Pennsylvania with a planned stop in Chicago, Illinois. The aircraft was a McDonnell Douglas DC-10 with three General Electric (GE) engines, all which were compliant with Federal Aviation Administration (FAA) regulations. The principal control surfaces of the DC-10 (rudder, elevators, ailerons), as well as the flaps, brakes, and landing gear, all functioned with three independent hydraulic systems.

At 3:16 PM on July 19, 1989, the stage-1 fan disk on the No.2 engine failed, causing the fan blades to break into sharp fragments, destroying all three hydraulic systems. Without an operable hydraulic system, the pilots were only able to control the aircraft by adjusting the thrust of the two remaining engines. This led to an emergency landing of the aircraft at Sioux Gateway Airport, in Sioux City, Iowa. During the landing attempt, the right wing of the aircraft contacted the ground first, breaking the aircraft into multiple pieces and igniting a fire. Out of 296 people on board, 111 were killed and 172 were injured [1].

2 Case History of Component Material

2.1 Material Properties

The stage-1 fan disk of the GE CF6-6 engine was made of a typical aerospace grade titanium alloy (Ti-6Al-4V) (Table 2) [1; 2]. This alloy can develop three different types of anomalies during the melting portion of the manufacturing process: type-I hard alpha inclusions [3], high density inclusions [4], and segregation [5]. Specifically, type-I alpha inclusions, which were the type of melt-related anomaly that was present in the failed fan disk, formed from localized excess amounts of oxygen or nitrogen sources. These inclusions have a melting temperature significantly higher than the normal structure, making them difficult to diffuse and eliminate. Additionally, the lack of ductility in these inclusions makes them a probable site for development of voids or initiation of fatigue cracks [1].

2.2 History of Manufacturing

Processing a stage-1 fan disk generally consists of three major steps [1]: (1) material processing, where raw materials (alloy source compositions, Table 2) are combined as a heat and melted into a titanium alloy ingot, then are mechanically elongated into a billet; (2) forging, where a billet is cut into multiple forging blanks and forged into forging shapes; and (3) final machining, where a forging shape is machined into the final configuration, followed by a shot peening process that creates local compressive stress fields on fatigue-critical areas (Figure 6).

Titanium Metals Corporation (TIMET) provided the initial raw materials for the fan disk. The heat (K8283) was melted on February 23, 1971 using a double-vacuum-melting process. The ingot (from heat K8283) was reformed into a 16-inch diameter billet. Before shipping to Aluminum Company of America (ALCOA), TIMET performed a contact-ultrasonic inspection on the billet and removed the top 6.5 inches of the part, which contained subsurface flaws. The billet was then shipped to ALCOA on March 26, 1971.

At ALCOA, the billet was cut into eight forging blanks and each blank was mechanically tested for its tensile strength and notch stress rupture life. The blanks were also inspected for their alpha phase and hydrogen content. The blanks were then machined into forging shapes, which were shipped to GE in May 1971.

At GE, the specimens were machined into rectilinear forging shapes and inspected using immersion ultrasonic and macroetch inspection techniques. By December 1971, the rectilinear machined forging shape (fan disk) had been processed by shot peening, grit blasting, and metal spraying. Additionally, the fan disk was inspected through fluorescent penetrant inspection (FPI) for the final time during the manufacturing process. The qualified fan disk was then installed in a new CF6-6 engine (S/N 452-251) and shipped to McDonnell Douglas Company.

2.3 History of Usage

The engine was first installed in June 1972 in the number 3 engine position in United Airlines DC-10 airplane (N1814U). In September 1988, the engine was removed and installed in UAL airplane N1819U number 1 engine position. Eight weeks later, the engine was moved to the number 2 position in the same aircraft (Figure 7). By the time of the accident, the engine had accumulated 42,436 hours and 16,899 cycles.

The stage-1 fan disk ran for 17 years and disassembly and overhaul were performed on 6 occasions. During each of these shop visits, the fan disk passed inspection. The final inspection occurred 760 cycles prior to the accident.

3 Analysis of Stresses

This section addresses the basic methodology used to estimate the forces and stresses the fan disk experienced prior to failure. In addition, the main mode of fatigue failure using both total life and defect tolerant approach are employed. Detailed calculations can be found in Appendix A.

3.1 Stresses Acting on the CF6-6 Engine Stage-1 Fan Disk

The fan disk of the CF6-6 engine is shown in Figure 1. The assembly consists of the large stage-1 fan disk and the attached fan blades (Figure 8). The analysis will focus only on the fan disk as it was the component that initially failed. The fan disk weighed 370 pounds, consisting of the rim, the bore, the web and the disk arm as labeled in Figure 2. The primary loads imposed on the fan disk were radially tensile loads. These loads were from the disk holding the fan blades against centrifugal forces during rotation of the assembly. The loads imposed by the fan blades caused radial stresses in the disk rim. These radial stresses decreased towards the bore, becoming zero at the bore since there was no material inboard of this location to resist the stress. Hoop stresses were greatest along the inside diameter of the bore, with the forward corner of the bore experiencing the maximum hoop stress (as shown in the finite element analysis (FEA), Figure 2).

3.2 Total Life Approach

In analyzing the stresses using the total life approach, we note that a take-off/landing couple is considered one cycle. Thus, in order to proceed with Basquin's equation (Eq. 1) to estimate the total life (N_f) of the stage-1 fan disk, we first had to calculate the effective stress $\sigma_{a,i}$, the mean stress $\sigma_{m,i}$ and the number of cycles to failure $N_{f,i}$ of each stage i within a cycle. Each cycle would therefore have 3 stages; take-off, landing and total (this is the stage within a cycle that goes from $\sigma_a = 0\ MPa$

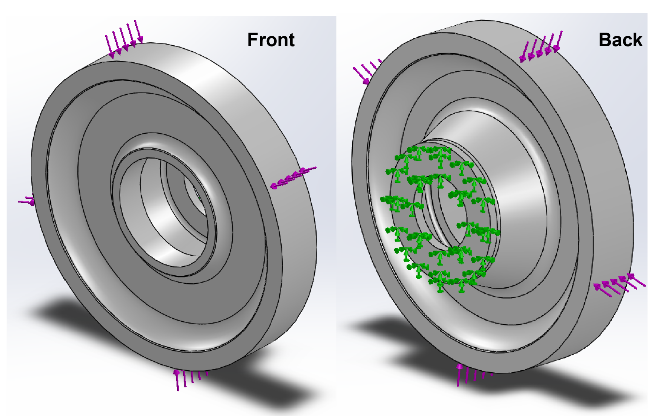

Fig. 1: Loads applied to fan disk during operation.

to $\sigma_a = 315\ MPa$ and back to 0). According to the FEA performed by the National Aeronautics and Space Administration (NASA), the stage-1 fan disk of an aircraft engine typically running at a disk speed of 6000 revolutions per minute (rpm) experiences critical circumferential stresses that range between 40 to 50 ksi [6]. Using a median value between 40 and 50 ksi, we used an effective stress of $\sigma_a = 45\ ksi \approx 315\ MPa$. However, during take-off and landing, the engine deviates from its cruising behavior and accelerates to a maximum disk speed of $rpm_{max} = 8500$ rpm. Using this assumption and a linear interpolation method, a $\sigma_{a,max} = 446\ MPa$ was calculated for the take-off and landing stages. We then calculated the proportion $\frac{N_i}{N_{f,i}}$ of each stage within a cycle to get a total number of cycles to failure of $N_f = 42,382$ cycles. According to the National Transportation Safety Board Office of Aviation Safety (NTSB), GE estimated a defect-free service life of $54,000$ cycles during certification [1], a value that is 22% higher than our prediction. A factor of safety (FOS) of 3 was used to establish a safety limit, reducing the service time to $18,000$ cycles. Our calculations showed that applying a FOS of 3 would have ended the service time of the engine at $\sim 14,000$ cycles, which is $2,899$ cycles before the No.2 engine total cycles to failure of $16,899$ cycles.

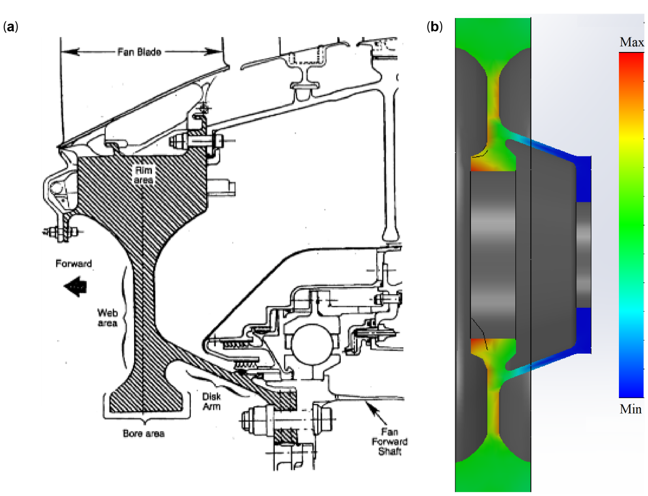

Fig. 2: (a) Cross section of fan disk and surrounding components [1]. (b) Effective stress distribution in the fan disk during operation.

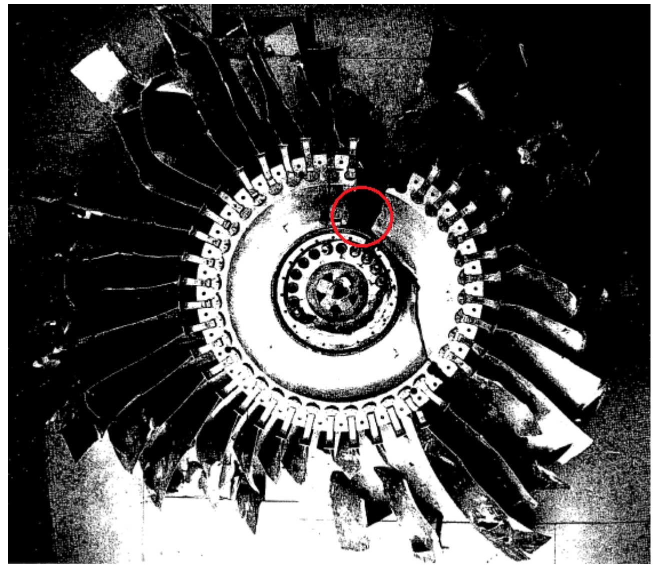

Fig. 3: Reconstruction of the broken fan disk. The red circle shows the initial separation. [1].

3.3 Defect Tolerant Approach

Using the defect tolerant approach, we first found the values of the initial ($a_i = 4.6\ mm$) and final crack length ($a_f = 14\ mm$) along the radial direction from the analysis done by the NTSB [1]. We note that the a_i was discovered after the crash occurred, since the methods used during the last inspection did not detected the flaw. According to Ritchie et al. cracks of size $> 5\ mm$ are categorized as "large cracks" and therefore fit a two parameter Paris regime (Eq. 2) to predict the crack growth [7]. Using separation of variables and integration of the Paris regime equation, we estimated the number of cycles needed to grow the initial crack to a critical size that resulted in premature failure. We calculated $N_{f,\ calculated} = 789$ cycles needed to achieve failure, a value within $\sim 4\%$ of the actual number of cycles to failure ($N_{f,\ actual} = 760$ cycles).

4 Determination of Root Cause

When the fan disk failed, it blew out of the containment ring, making it almost impossible to understand why the failure had occurred. Fortunately, three months later, the broken fan disk was recovered from a corn field and the determination of the root cause became plausible.

The inspection process started with a thorough visual examination. During this visual inspection, 2 main fracture areas were identified (Figure 3).

The first fracture area progressed circumferentially through the rim of the disk. The second fracture area progressed in the radial direction through the bore, web, disk, arm, and rim, showing signs of overstress. These markings of overstress were linked to a preexisting radial/axial fatigue crack in the bore of the fan disk. [1].

The preexisting crack was further examined by metallography, fractography and chemical analysis. This analysis revealed that the crack initiation occurred in a cavity on the surface of the disk bore. This cavity was created by a nitrogen-stabilized hard alpha inclusion that went undetected during the ultrasonic inspection performed by GE. The cavity went undetected because during the time of the inspection, the fatigue origin was nearly filled with hard alpha material, making it undetectable by ultrasonic inspection.

Scanning electron microscopy (SEM) showed that the stabilized alpha inclusion had microcracks oriented parallel to the cavity surface and that microporosity was present around the core of the inclusion. Additionally, areas with brittle fracture features and no fatigue striations were found intermixed with ductile-appearing bands and fatigue striations [1]. The fatigue striations were found near the end of the stabilized-alpha area of the inclusion. The fatigue striations spacing was measured and was found to vary over the area. The closer spaced striations were referred to as minor striations and the wider spaced striations were referred to as major striations. The total number of these major striations was correlated with the total number of takeoff/landing cycles (15,503 cycles) [1] and showed evidence of the growth of the fatigue crack since the early life of the fan disk.

During the life of the stage-1 fan disk, six detailed inspections were performed by UAL using FPI. NTSB suggested that the type-I alpha inclusion (together with the cavity and microcracks produced by the shot peening process) introduced an initial crack with the application of stress during the disks initial exposure to a full thrust condition (confirmed by correlation of number of major striations to cycles and a sister fan disk manufactured at the same time). Unfortunately, none of the six FPI inspections indicated the surface crack.

The last inspection was performed in February

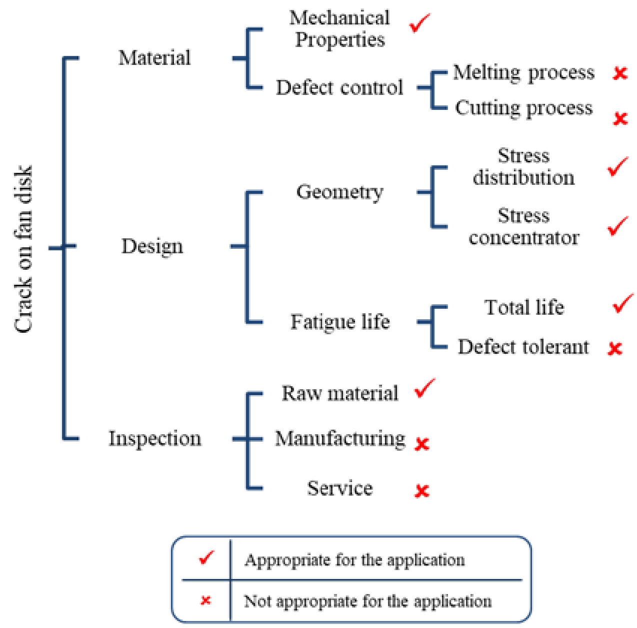

Fig. 4: Fault tree analysis of the fatigue crack in the fan disk.

1988. At the time of this inspection, the crack was approximately 13 mm (0.476 inch) [1] in the axial direction, which went undetected due to naked eye limitation. However, the inspection left a discolored area that was evident during post-failure examination. During the 760 takeoff/landing cycles after the last inspection [1], the crack grew to a critical length and introduced a catastrophic fracture of the fan disk from the bore to the rim.

In summary, the root cause of the airplane disaster was a combination of the lack of detection of a cavity during the manufacturing process and the non-identification of an existing crack during the last inspection of the fan disk in service.

5 Failure Prevention

Numerous methods of failure prevention could have avoided the crash of UAL flight 232 (Figure 4), beginning with the original manufacturing of the fan disk. As noted in the previous section, the hard-alpha inclusion was formed during the melting process and the associated cavity was formed during the shot peening process. A macroetch inspection after the shot peening process may have detected this surface defect; however, at the time, this inspection was only performed prior to the shot peening process. The manufacturers did not perform this inspection because they feared the process would remove surface material, changing the overall mass and performance of the components. Additionally, to promote the melting or dissolution of the hard alpha inclusions during the ingot manufacturing, the temperature of the molten pool could have been increased in the furnace or the time the material was in the liquid state could have been increased (from double- to triple-vacuum-melting process). Although these changes do not guarantee the complete dissolution of the inclusions, they would have increased the amount of opportunities for dissolution of the flaw. Additionally, after ingot manufacturing, a smaller diameter billet (13- or 14-inch diameter) would have increased the likelihood of a crack being detected below the surface [1].

The manufacturer assumed a defect-free approach and calculated a predicted service life of 54,000 cycles (18,000 cycles with a safety factor of 3) [1]. Since the hard alpha inclusion introduced a cavity early in the operation of the fan disk, a defect tolerant approach should have been taken during the design phase. Using a defect-tolerant approach, we calculated the service life of 789 cycles from the last inspection.

Proper inspection of the fan disk in service would have also prevented failure of the device. Post analysis concluded that approximately a half inch crack existed on the surface of the bore during the time of the last inspection. This crack was sufficiently open, therefore the FPI fluid was able to enter into the crack, making it detectable [1]. If the inspector adequately prepared the part for inspection and rotated the part during inspection, the crack should have been detected. Additionally, nondestructive inspection operators at the time typically worked alone with very little supervision. Had regulation required another person to oversee the inspection process, it is more likely the crack would have been detected.

Further consideration should have been taken during the original design and certification of the aircraft to protect the critical hydraulic systems. If a backup system had been developed to preserve adequate flight control in the event of a catastrophic in-flight event, such as in UAL flight 232, the outcome could have been very different. Furthermore, if the engine engineers provided data about the distribution patterns and fragment energy levels of the engine components during a failure, further design precautions could have been taken when designing for the protection of the hydraulic systems. Another prevention method could have been the usage of valves to stop the fluid in the hydraulic system if a pipe is broken.

Various other steps could have been taken to mitigate the number of casualties in the accident. At the time of the accident, infants were considered in-lap occupants. During the preparations for emergency landings, parents were instructed to hold their children on the floor while they assumed their protective brace position. Implementation of a better emergency procedure for children could have reduced the number of children harmed. Additionally, the emergency management post-crash could have been improved. Due to a mechanical issue, one of the water supply vehicles was unable to pump any water, allowing the fire to intensify and spread to the inferior of the plane. This issue could have been eliminated if there was a requirement for periodic fire-service full-capacity testing.

6 Conclusions

In conclusion, the lack of proper inspection of the fan disk on United Airlines Flight 232 led to the death of 111 passengers. Since the fatigue crack on the fan disk went undetected, the prolonged fatigue stresses eventually caused the part to fracture, destroying the engine and severing the hydraulic lines. The underlying cause of this failure can be attributed to 3 factors: the microscopic defects that were not identified during initial manufacturing, the lack of attention to detail during inspections, and the absence of a contingency plan for the loss of hydraulic controls in the system design. Many safety and quality control questions emerged from this disaster, including flight control systems failure, jet engine construction processes and testing, pilot training, and flight procedures.

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Appendix A:
Basquin's Equation

$$\sigma_a = (\sigma' - \sigma_m)(N_f)^b \tag{1}$$

Paris Regime

$$\frac{\mathrm{d}a}{\mathrm{d}N} = C(\Delta K)^n (K_{max})^m = C(f\Delta\sigma\sqrt{\pi a})^n (K_{max})^m \tag{2}$$

Total Life Approach

We recall from Lewicki et al. that at an engine speed of 6000 rpm the hoop stress at the bore of the Fan Disk is about 315 MPa [8] . During Take-off and Landing the engine experiences a maximum stress due to the higher engine speed of 8500 rpm, which by linear interpolation becomes $\sigma_{max} = 446\ MPa$. The different stages of the flight are Take-Off and Landing as well as the total cycle from σ_{min} to σ_{max} and back. The number of cycles to failure $N_{f,\ i}$ for each stage is calculated using Basquin's equation, where $\sigma' = 1500\ MPa$ and $b = -0.2$ are obtained from Janeč̃ek et al. [6]

Table 1: Total Life approach Stress Calculations

Stage	σ_a (MPa)	σ_m (MPa)	$\frac{N_i}{N_f}$
Take-Off	446	315	1.37×10^{-7}
Landing	446	315	1.37×10^{-7}
Total Cycle	157.5	157.5	2.22×10^{-5}

$$\begin{aligned} N_f &= \frac{1}{2 \times (1.37 \times 10^{-7}) + 2.22 \times 10^{-5}} \\ &= 42,382 \ \text{ Cycles} \end{aligned}$$

Defect Tolerant Approach

In the following section calculations made to estimate the Defect tolerant life of the fan disk are shown. Recalling from Lewicki et al, the initial crack length during the last inspection was $a_i = 4.6$mm and the final crack length was $a_f = 14$mm [8]. Categorizing these lengths as 'Large' cracks and using the modified Paris Equation from Ritchie et al. the number of cycles required to achieve failure N_f is calculated [7]. By using separation of variables and integration over the crack size and Number of cycles we get:

$$\int_{N}^{N_f} \mathrm{d}N = \int_{a_i}^{a_f} \frac{1}{C(f\Delta\sigma\sqrt{\pi a})^n (K_{max})^m} \mathrm{d}a$$

$$N_f = \frac{2}{(n-2)Cf^n\Delta\sigma^n\pi^{\frac{n}{2}}K_{max}^m}\left(\frac{1}{a_i^{\frac{n-2}{2}}} - \frac{1}{a_f^{\frac{n-2}{2}}}\right)$$

where $f = 1.12$, $\Delta\sigma = 315\ MPa$, $C = 5.2 \times 10^{-12}$ and $n = 2.5$, $m = 0.67$ and $K_{max} = 75\ MPa\sqrt{m}$.

$$N_f = \frac{2}{(n-2)(5.2 \times 10^{-12})(1.12)^n (315)^n \pi^{\frac{n}{2}} (75)^{0.67}} \left(\frac{1}{(4.6)^{\frac{n-2}{2}}} - \frac{1}{(14)^{\frac{n-2}{2}}} \right) \approx 789 \text{ Cycles}$$

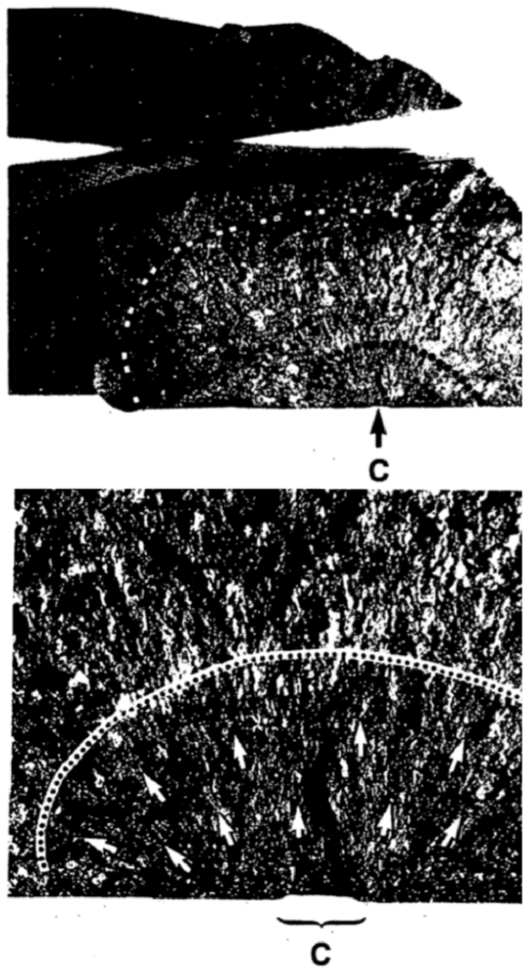

Fig. 5: Crack initiation site on fan disc [1]

Appendix B:

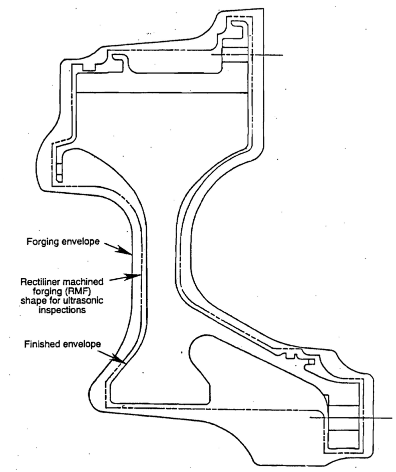

Fig. 6: Stage 1 fan disc cutaway view [1]

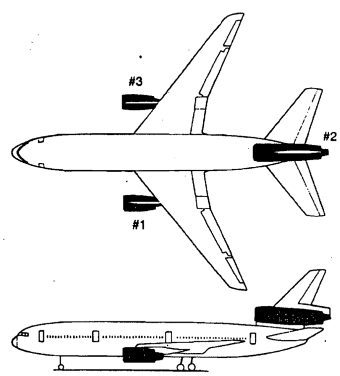

Fig. 7: DC-10 airplane top/side view with engine arrangement[1]

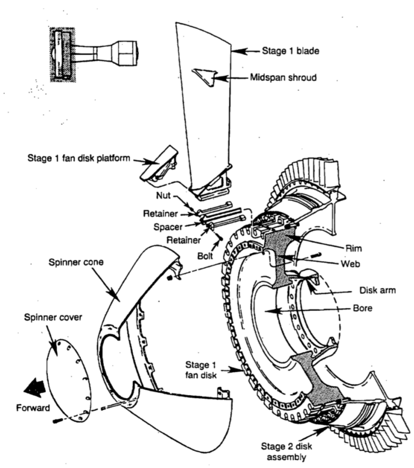

Fig. 8: Fan rotor assembly [1]

Table 2: Material Properties for Ti-6Al-4V [2]

Properties	Values
Composition	Aluminum (6) + Iron (¡=0.25) + Oxygen (¡=0.25) +Titanium (90) + Vanadium (4) (in Wt.%)
Modulus	113.8 GPa
Poissons Ratio	0.342
Tensile Yield	880 MPa
Tensile Ultimate	950 MPa
Elongation at Break	14%
Compressive Yield	970 MPa
Fracture Toughness	75 $MPa\sqrt{m}$
Fatigue Strength	240 MPa @ $k_t = 3.3, N_f = 10^7$
Fatigue Strength	510 MPa @ $N_f = 10^7$