TURBINE DISKS FOR IMPROVED RELIABILITY

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SUMMARY

The trend toward higher turbine-blade tip speeds and inlet gas temperatures makes it increasingly difficult to design reliable turbine disks that can satisfy the life and performance requirements of advanced commercial aircraft engines. Containment devices to protect vital areas such as the passenger cabin, the fuel lines, and the fuel tanks against high-energy disk fragments would impose a severe performance penalty on the engine. The approach taken in this study was to use advanced disk structural concepts to improve the cyclic lives and reliability of turbine disks. Analytical studies were conducted under NASA contracts by the General Electric Company and Pratt & Whitney Aircraft to evaluate bore-entry disks as potential replacements for the existing first-stage turbine disks in the CF6-50 and JT8D-17 engines. Results of low-cycle fatigue, burst, fracture mechanics, and fragment energy analyses are summarized for the advanced disk designs and the existing disk designs with both conventional and advanced disk materials. Other disk concepts such as composite, laminated, link, multibore, multidisk, and spline disks were also evaluated for the CF6-50 engine.

INTRODUCTION

A disk burst is one of the most catastrophic failures possible in an aircraft engine. Flight failures of disks in commercial airliners have caused fires, rupture of fuel tanks, penetration of passenger cabins, wing damage, ingestion of disk fragments by other engines, and aircraft control problems (ref. 1).

Aircraft engine companies generally endeavor to use conservative design practices and modern quality control procedures in producing turbine disks. However, failures occur because of design errors, undetected manufacturing defects, uncontrollable operating factors, errors in engine maintenance and assembly, and failure of other engine components. To attempt to design turbine disks to preclude failure from any of these causes would result in prohibitively low allowable stresses. Containment devices to protect vital areas of the aircraft against high-energy disk fragments would impose severe performance penalties on the engine.

The approach taken in this program was directed toward improving turbine disk reliability by using more advanced structural concepts to increase low-cycle fatigue life, to impede crack propagation, and to reduce fragment energies that could be generated

event of a disk failure. This paper reports the results of NASA-sponsored ana-
lytical studies by the General Electric Company and Pratt & Whitney Aircraft (refs. 2 and 3) to evaluate bore-entry disks as potential replacements for the existing first-stage turbine disks in the CF6-50 and JT8D-17 engines, respectively; these engines were selected because of their extensive use in commercial passenger aircraft. Other concepts such as composite, laminated, and multidisk designs were also studied for the operating conditions of the CF6-50 engines.

The bore-entry disks were compared with the existing disks (henceforth called the "standard disks") on the basis of cycles to crack initiation and overspeed capability for initially unflawed disks and on the basis of cycles required to propagate initial flaws to failure. Comparisons were also made of the available kinetic energies of possible burst fragments. All of these comparisons were also made for the standard disk with the material of the bore-entry disk so that improvements resulting from changes in material properties could be distinguished from those resulting from structural design changes.

DISK CONCEPTS

CF6-50 Turbine Disk Designs

The standard disk and the disk concepts considered as potential replacements are illustrated in figure 1. The standard disk (fig. 1(a)) is machined from an Inconel 718 (Inc-718) forging. Local bosses on both sides of the disk provide reinforcement around the bolt holes to increase the low-cycle fatigue life at the hole rims. Cooling air from the compressor is channeled through the shaft, cools the disk bore, is pumped up radially between the stage 1 and 2 rotors, cools the aft side of the disk between the bolt holes and rim, and then enters the blades through openings in the dovetails.

The bore-entry disk (fig. 1(b)) is a two-part disk of integral construction. The two disk halves are connected by radial webs for channeling coolant up the center of the disk from the bore to the blades. Among the advantages of the bore-entry concept are improved cooling effectiveness, reduced axial thermal gradients, and increased resistance to crack propagation in the axial direction. One of the main attractions of the bore-entry concept for the CF6 program was that it lent itself to a redundant construction where the disk would be overdesigned so that if half was failing, the undamaged disk half would be able to assume a larger portion of the load and sustain the damaged part; however, this would require a substantial increase in total disk weight. The integral bore-entry disk would be fabricated from a single-piece forging of René 95 alloy with the material between disk halves removed by electrochemical machining.

The composite disk (fig. 1(c)) uses high-strength filament or wire hoops to provide most of the load-carrying ability of the disk except at the dovetail attachments. The hoops would have to be pretensioned in order to assure an even load distribution among the filaments; this could be accomplished by filament winding, by interference fitting, or by the selection of filament and matrix materials so that the desired hoop pretension would be applied by differential thermal expansion under engine operating conditions.

In the laminated design (fig. 1(d)), a disk is constructed by bolting together a large number of sheet-metal laminates. A stepwise variation in thickness provides more laminates at the rim and bore but leaves gaps between laminates in the web region. In the link design (fig. 1(e)) a disk is constructed of pinned sheet-metal link segments. Both the laminate and link concepts are directed toward low-cost fabrication, isolation of propagating cracks, and generation of small burst fragments rather than toward improving disk life.

The multibore disk (fig. 1(f)) separates the highly stressed bore region into a number of circumferential ribs in order to prevent a crack or flaw at the bore from propagating axially. At the ends of the ribs, the tangential stresses due to centrifugal loading would be less and, therefore, the crack propagation rate should be slower than at the bore of the standard disk.

The purpose of the multidisk design (fig. 1(g)) is to obtain improved disk cooling and to provide for a redundant construction by transference of loads from a failed disk member to the undamaged ones through the bolts. The spline disk (fig. 1(h)) is essentially a two-piece design where the members are coupled through splines on their center faces. In order to counter the tendency of each disk-half to straighten out due to the lack of axial symmetry, the splines would have to be radially interlocked through pins. The mechanical coupling of the multidisk and spline designs prevents cracks in one disk member from propagating to another.

These concepts are described in more detail in reference 2.

JT8D-17 Turbine Disk Designs

The standard disk shown in figure 2(a) is machined from a Waspaloy forging. Cooling air is bled from the combustion chamber liner and discharged at high velocity through nozzles toward the front side of the disk near the rim. The cooling air is delivered to the blades through angled holes at the disk rim. These holes result in elliptical exit openings with high stress concentrations; these are the limiting low-cycle fatigue locations.

A split-bonded, bore-entry concept was selected as a possible replacement for the standard disk. As with the integral bore-entry disk (fig. 1(b)) for the CF6-50 turbine, cooling air would be introduced at the bore, would be pumped up radially through channels formed by radial webs, and would enter the blades through openings in the bases. The two halves of the bonded bore-entry disk would be fabricated from separate forgings

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of Astroloy and diffusion brazed together at the center surfaces of the radial webs. Dovetail broaching and final machining operations would be performed on the bonded disk assembly. The emphasis in the design of the bonded bore-entry disk was on improving the cyclic life without providing redundancy or increasing the disk weight.

DESIGN CONDITIONS

Design properties of the materials for the standard and bore-entry disks are presented in table I. The simplified flight cycles used for the cyclic heat transfer and stress analyses are shown in figure 3 for the CF6-50 engine and in figure 4 for the JT8D-17 engine. The flight cycle shown in figure 4 was the cycle used in the original design of the first-stage turbine disk for the JT8D-17 engine. The analytical methods are discussed in references 2 to 4.

DISCUSSION OF RESULTS

Preliminary Analyses of CF6-50 Disk Concepts

The results of preliminary analyses of the seven candidate design disk concepts are summarized in table II. Two of the designs, the laminated and link disks, proved to have excessive mechanical stresses and to be unsuitable for the CF6 operating conditions. The multibore design exhibited high transient thermal stresses in the region above the bore rims; therefore, the desired benefit of this design in retarding the propagation of rib flaws was not fully realized. Analysis of the multidisk design under various failure conditions revealed that the bolts could not contain a failed outer disk and that a crack in a center disk would reach critical length before the load could be redistributed to the undamaged members.

Only the bore-entry, composite, and spline disks appeared suitable for the CF6-50 turbine disk applications. From the standpoint of strength-to-density ratio, the composite disk was the most promising concept. However, the composite design is furthest removed from the current state-of-the-art of fabrication and material processing technology of any of the concepts considered. Because of the considerable fabrication development that would be required, the composite disk was not further considered. The spline disk presented special problems in analysis because the load distribution among the splines is dependent on the fabrication tolerances and it is not readily apparent how the loading would be redistributed should one disk-half fail. The integral construction of the bore-entry disk gives more assurance that the loading due to a failed disk member would be more evenly redistributed on the undamaged member. The integral bore-entry concept was, therefore, selected for more detailed study to replace the CF6-50 standard disk.

Analyses of CF6-50 Standard and Bore-Entry Disks

The rim and bore average temperature responses during the flight cycle of the standard and bore-entry disks are shown in figure 5. Average effective stresses are also indicated at the start and end of takeoff, climb, cruise, and thrust reversal on descent. In both disks the maximum rim and bore temperatures occurred at the end of takeoff and climb, respectively; the maximum stresses also occurred in the bore at the end of climb.

Bore temperatures in the bore-entry disk are only slightly lower than bore temperatures in the standard disk since the bore is cooled in both cases. Rim temperatures were somewhat higher in the bore-entry disk because the coolant picks up some heat from the center faces of the disk, whereas the coolant only comes into contact with the sides of the standard disk near the rim.

Figure 6 shows the predicted cyclic lives to crack initiation in the initially unflawed standard and bore-entry disks. The limiting fatigue life of 30 000 cycles in the Inc-718 standard disk was at the aft dovetail post rabbet, where the side plate is fastened to the disk. This location was not further considered in the study because fragment generation due to failure would be limited to the dovetail post and adjacent blades. The next most critical location in the Inc-718 standard disk was at the bore with a predicted crack initiation time of 63 000 cycles. The initial FAA certified life of the first-stage turbine disk was 7800 cycles based on one-third of the minimum design life for the original design cycle, which was somewhat different from the simplified cycle used in this study; this FAA approved life is subject to increase as the result of ground tests of three fleet leader engines.

Calculated crack initiation lives for the René 95 standard and bore-entry disks were over 100 000 cycles. Since the crack initiation analyses were based on minimum guaranteed material properties, it is evident that even the standard disk is very conservatively designed provided the design conditions are not exceeded and the disks are initially unflawed.

The cyclic lives for cracks propagating from initial semielliptical surface flaws 0.635 centimeter (0.250 in.) by 0.211 centimeter (0.083 in.) to critical crack size are shown in figure 7 for the most critical locations in the three disks. Manufacturing flaws of this size should be readily detectable by modern nondestructive evaluation techniques. However, in the past, large defects in turbine disks have occasionally escaped detection through human error and have caused problems in some military engines in flight.

The most critical locations for flaws were at the dovetail slot bottom in the Inc-718 standard disk and at the bore in the René 95 standard and bore-entry disks. Although the bore-entry disk showed an improvement in the minimum crack propagation life of more than 300 percent as compared with the Inc-718 standard disk, part of this increase was due to the superior strength properties of the René 95 alloy. If the effect of different materials was eliminated by comparing the bore-flawed bore-entry and René 95 standard disks, the improvement in crack propagation life resulting solely from the structural change was 136 percent.

The crack propagation lives given in figure 7 for the Inc-718 standard disk with a dovetail slot bottom flaw and the bore-entry disk with a bore flaw are only 5 and 20 percent of the FAA certified life of the disk. However, the probability of such large flaws occurring at critical locations and passing modern inspection procedures is statistically remote. Of greater significance is that a substantial improvement in the crack propagation life is added insurance against sudden catastrophic failure due to unforeseen design, manufacturing, maintenance, or operating problems. The overspeed burst margins of the bore-entry disk were 18 and 11 percent greater than for the Inc-718 and René 95 standard disks, respectively.

The redundant construction of the bore-entry disk resulted in an increase in weight of 66 percent over the standard disk. This extra weight is equivalent to an increase of 0.29 percent in installed specific fuel consumption (SFC) for an average DC10-30 aircraft flight.

The extra disk weight could also be added to the standard disk design to reduce the centrifugal stresses due to the blade loads. However, this mechanical stress reduction would probably be offset by the increased transient thermal stresses resulting from the slower thermal response of the bulkier disk. Also, a heavier standard disk would lack the redundancy of the bore-entry disk and would generate even higher fragment energies from a burst disk.

Some possible fragment patterns resulting from manufacturing flaws are illustrated in table III. The available kinetic energies that would be generated from these failures are also indicated. The highest energy fragments are caused by failures initiating at and propagating radially from the bore, as shown by the 120° disk and blade fragment pattern for the standard disk in table III. However, the redundant construction of the integral bore-entry disk would enable the undamaged member to contain such a failed part. The only possibility of a segment separating in this way would be if the radial failure propagated through a web to the opposite disk face; however, this is highly unlikely because the total thickness for all the webs is only 20 percent of the bore circumference and, as one web started failing, its load would be transferred to adjacent webs. The most likely mode of fragment generation is a rim fragment resulting from defects or crack initiation sites at the dovetail slot bottom or bolt hole rim. Based on spin pit experience, the rim-initiated crack would result in the loss of three dovetail posts and four blades, as shown in table III. The fragment energy of the bore-entry disk rim fragment was only about 10 percent of the 120° disk segment that was assumed to be generated from a bore defect in the standard disk.

Analyses of JT8D-17 Standard and Bore-Entry Disks

The average temperature responses for the JT8D-17 turbine disks in figure 8 show consistently lower bore and rim temperatures throughout the cycle in the bore-entry disk as compared with the standard disk. The lower temperatures in the bore-entry disk were the result of its superior cooling effectiveness and the use of cooling air bled from the compressor midstage. Maximum temperatures and stresses occurred at the end of takeoff and climb, respectively.

Predicted cyclic lives for the initially unflawed standard and bore-entry disks are presented in figure 9. The FAA-certified life of the Waspaloy standard disk is 16 000 cycles based on the limiting low-cycle fatigue life at the exit of the cooling air hole. These results indicate an improvement in the cyclic crack initiation life of the Astroloy bore-entry disk of 88 percent over the Waspaloy standard disk and 67 percent over the Astroloy standard disk. The most critical location in the bore-entry disk was in the bore region at the entrance to the cooling air channel.

Defects and manufacturing flaws in the JT8D-17 turbine disks were considered for the critical locations indicated in figure 10. Subsurface flaws of 0.119 centimeter (0.047 in.) in diameter were assumed in the bore and web regions for all three disks; this diameter was selected because it is at the threshold of detectability by ultrasonic inspection. The web flaws shown in figure 10 were at the radius of maximum radial stress in the standard disks and at the radius of maximum axial stress at the bond surface in the bore-entry disks. The surface flaws at the disk rim or bore were assumed to be 0.081 centimeter (0.032 in.) in length.

The most critical location in the Waspaloy standard disk for a flaw was at the exit of the cooling air hole with a predicted crack propagation life of 2900 cycles. Substituting Astroloy properties for the Waspaloy reduced the calculated crack propagation life to 1150 cycles because of the lower ductility. However, there are indications that if the crack propagation data had included hold-time effects, the crack propagation life of the Astroloy standard disk would have been superior to that of the Waspaloy standard disk. This would also mean that the values given in figure 10 for the bore-entry disk are too low.

The calculated improvement in the minimum crack propagation life of the bore-entry disk over the Waspaloy standard disk was 124 percent. This improvement is significant in increasing the capability of the disk to survive uncontrollable factors that might result in catastrophic failure of conventionally designed disks. There was a slight reduction in the overspeed burst margin of the bore-entry disk as compared with the standard disk because the overall disk weight was kept constant and that portion of it due to the radial webs was of small structural importance.

A substantial reduction in fragment energy is shown in table III for the JT8D-17 bonded bore-entry disk even though it was not designed for redundancy. This improvement would result from the confinement of the fragmentation from a bore flaw to one disk half; the other half would probably experience failure at the rim from the increased blade loading.

CONCLUDING REMARKS

Some advanced turbine-disk structural concepts have been analytically studied as potential replacements for the existing first-stage turbine disks in the CF6-50 and JT8D-17 engines. An integral bore-entry design was selected for more detailed evaluation for the CF6-50 engine as a result of preliminary analyses of seven disk concepts including composite, laminated, and multidisk designs. The integral bore-entry turbine disk was designed to improve disk life and to prevent high-energy fragmentation by using redundant construction at the expense of an increase in disk weight.

A split-bonded, bore-entry design was selected for evaluation for the JT8D-17 engine. This bore-entry disk was designed to improve disk life without redundance or an increase in disk weight.

Cyclic thermal, stress, and fracture mechanics analyses of the bore-entry and standard disks demonstrated that substantial improvements in the cyclic lives of both initially unflawed and flawed disks could be achieved with the bore-entry disk designs. The benefits of the advanced disk designs are influenced by differences in design philosophy, disk cooling method, fabrication procedure, and engine operating characteristics.

REFERENCES

1. National Transportation Safety Board Special Study: Turbine-Engine Rotor Disc Failures, 1975.
2. Barack, W. N.; and Domas, P. A.: An Improved Turbine Disk Design to Increase Reliability of Aircraft Jet Engines. NASA CR-135033 (R75AEG, General Electric Co.), 1976.
3. Alver, A. S.; and Wong, J. K.: Improved Turbine Disk Design to Increase Reliability of Aircraft Jet Engines. NASA CR-134985 (PWA-5329), 1976.
4. Kaufman, Albert: Advanced Turbine Disk Designs to Increase Reliability of Aircraft Engines. NASA TM X-71804, 1976.

TABLE I. - DESIGN PROPERTIES OF TURBINE DISK MATERIALS

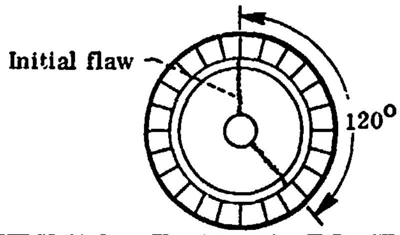
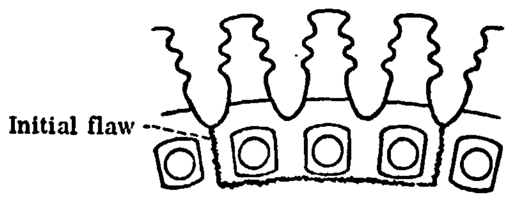
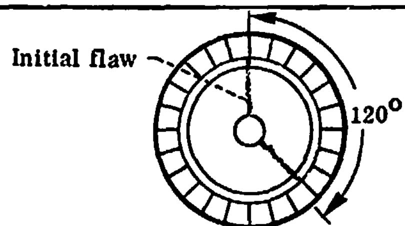
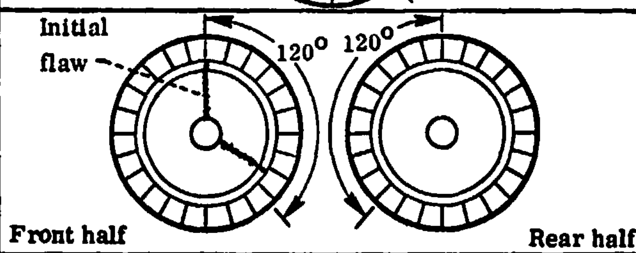
Property	CF6-50 engine		JT8D-17 engine	
	Inc-718	René 95	Waspaloy	Astroloy
Ultimate tensile strength, N/cm^2:				
At 294 K	126 000	150 000	124 000	134 000
At 811 K	110 000	145 000	110 000	117 000
Yield strength (0.2 percent offset), N/cm^2:				
At 294 K	101 000	116 000	86 000	97 000
At 811 K	92 000	110 000	76 000	87 000
Elongation at failure, percent:				
At 294 K	20	8.5	24	19
At 811 K	30	8.5	21	14.5
1000-Hour rupture strength at 867 K, N/cm^2	68 000	103 000	79 000	84 000
Stress range for crack initiation in 10 000 cycles at 811 K (minimum stress, zero), N/cm^2	81 000	93 000	85 000	[a]85 000
Critical stress intensity factor at 894 K, N/cm$^{3/2}$	93 000	88 000	>68 000	[a]>68 000

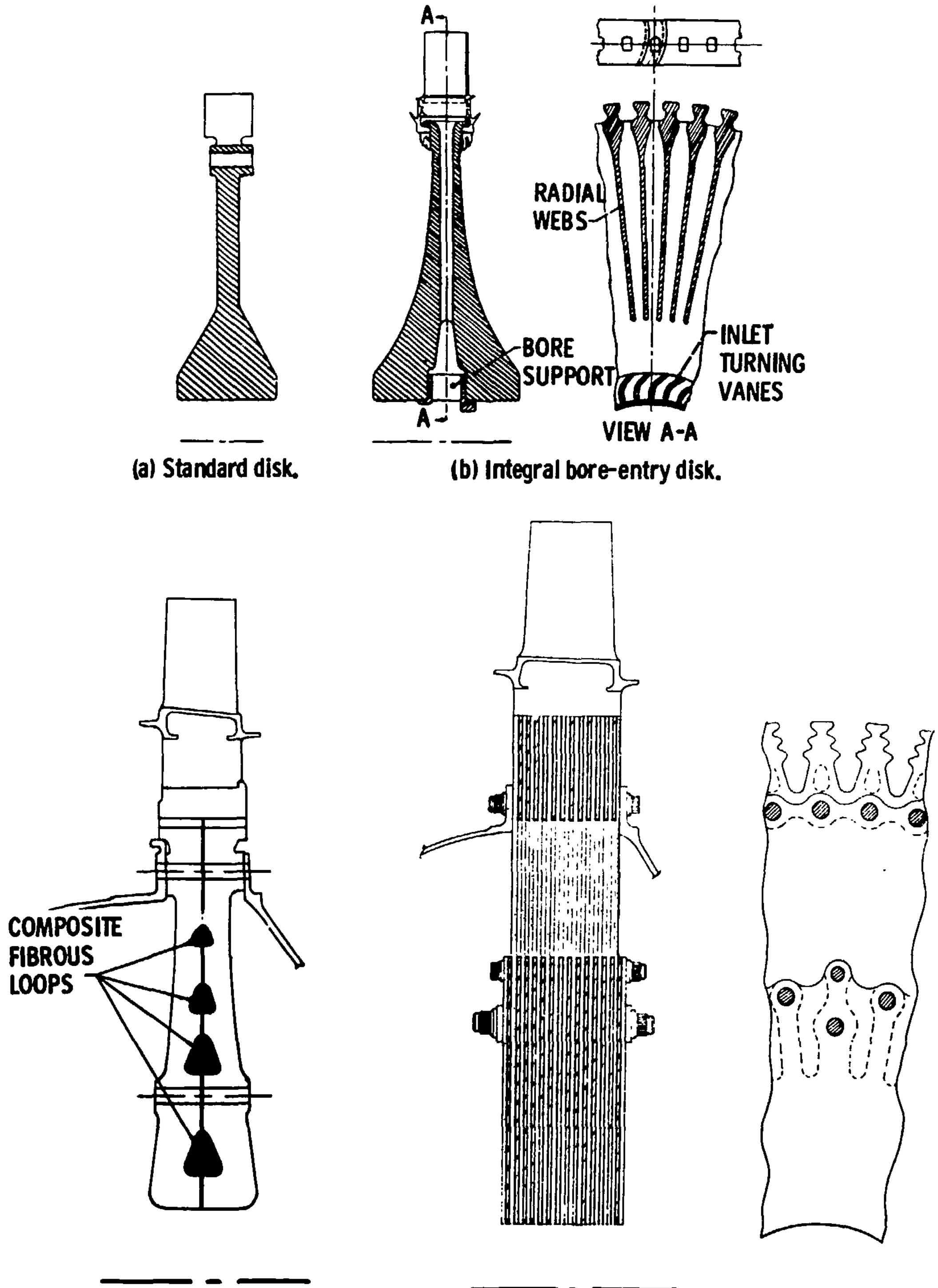
[a]Estimated.

TABLE II. - RESULTS OF PRELIMINARY ANALYSES OF CF6-50 DISK CONCEPTS

Disk concepts	Advantages	Disadvantages
Bore entry	Redundancy, improved thermal response, longer life	Increased weight to provide redundant design
Composite	Reduced stress levels, longer cyclic life	Limited material possibilities, fabrication development required
Laminated	Redundancy, low fragment energy, low cost	Excessive weight, high stresses at bolts and bolt holes, thermal mismatches between laminates
Link	Redundancy, low fragment energy, low cost	Excessive link stresses, difficult to seal disk to prevent coolant leakage
Multibore	Ribs prevent axial flow propagation at bore	High transient thermal stresses at rib outer diameter
Multidisk	Improved thermal response, some redundancy	Increased weight, bolts would fail if outer disk failed, no load shift if inner disk failed
Spline	Redundancy, longer life	Increased weight to provide redundant design, difficult to analyze load shift with one failed disk

TABLE III. - FRAGMENT ENERGIES OF TURBINE DISK DESIGNS

Disk design	Fragment pattern	Available kinetic energy, J
CF6-50 standard disk		1 172 500
CF6-50 integral bore-entry disk		110 500
JT8D-17 standard disk		678 600
JT8D-17 bonded bore-entry disk		513 000



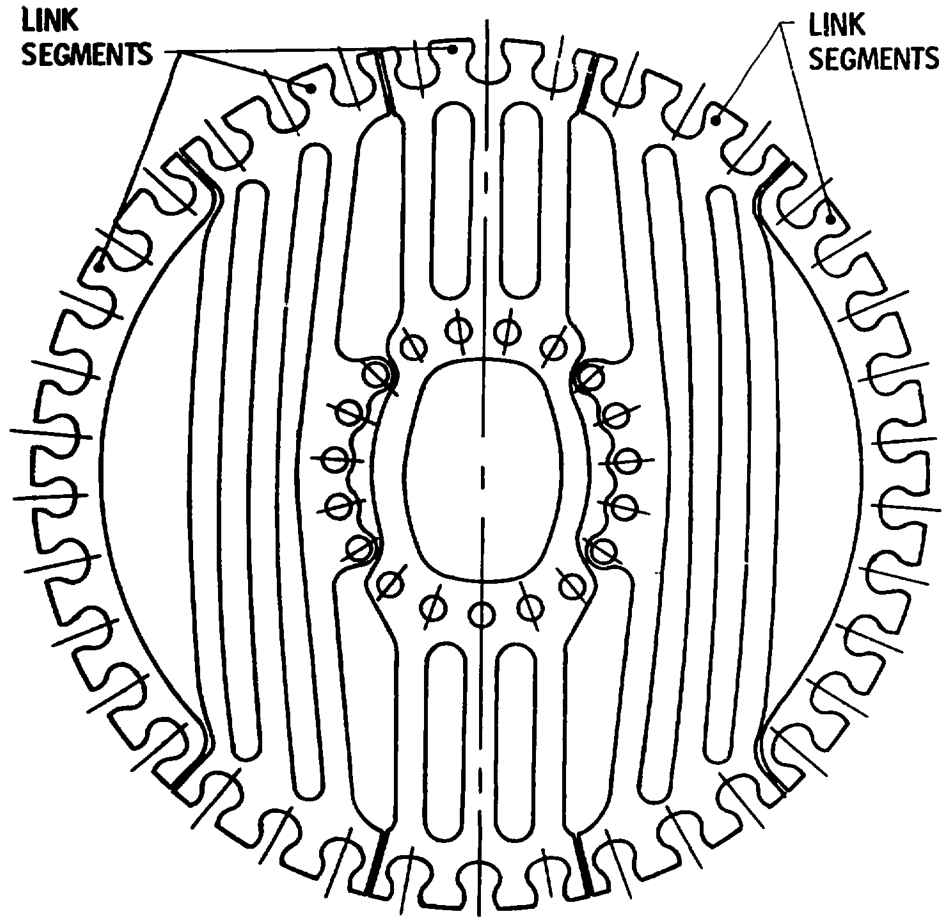
(a) Standard disk.

(b) Integral bore-entry disk.

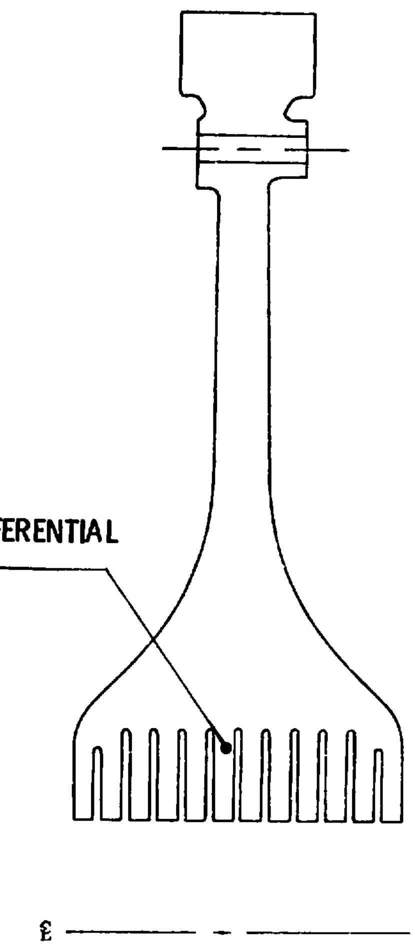
(c) Composite disk.

(d) Laminated disk.

Figure 1. - CF6-50 first-stage turbine disk designs.

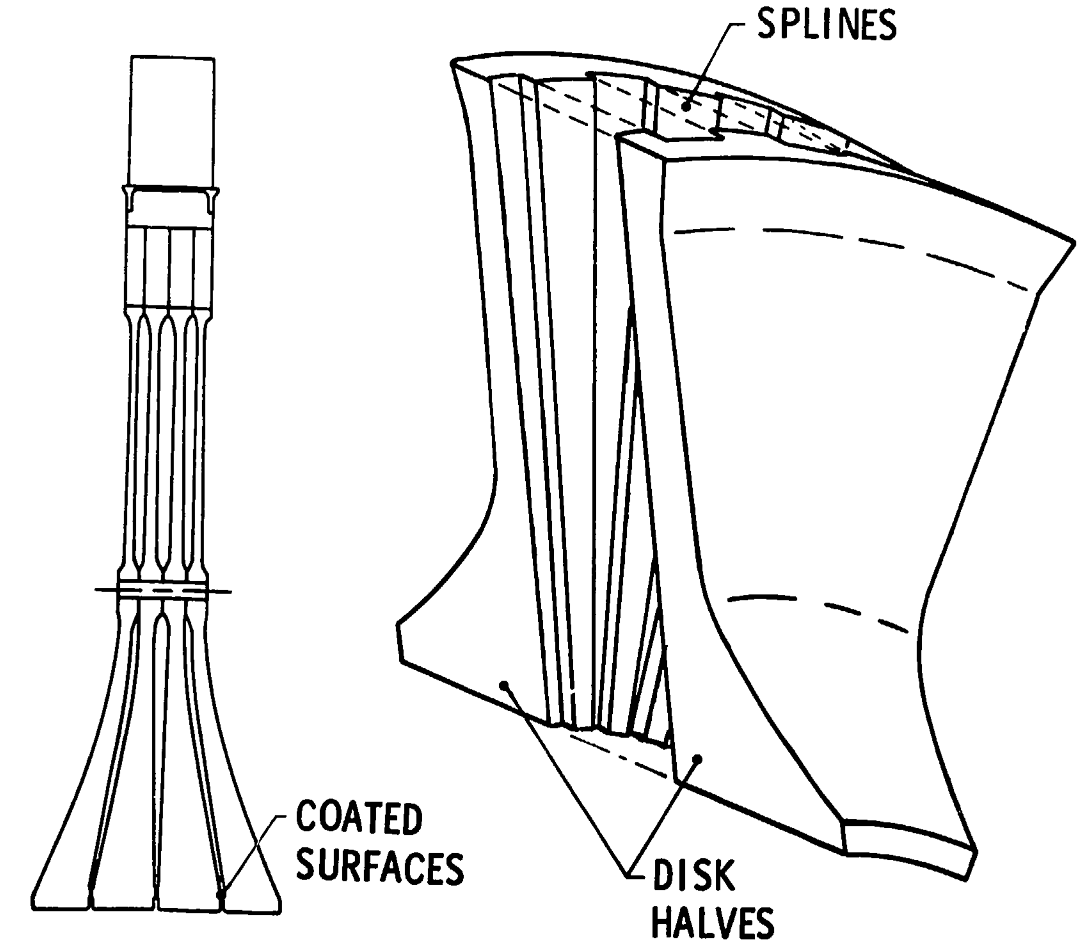


(e) Link disk (typically a disk would contain 20 to 40 layers each clocked axially relative to the next).



(f) Multibore disk.

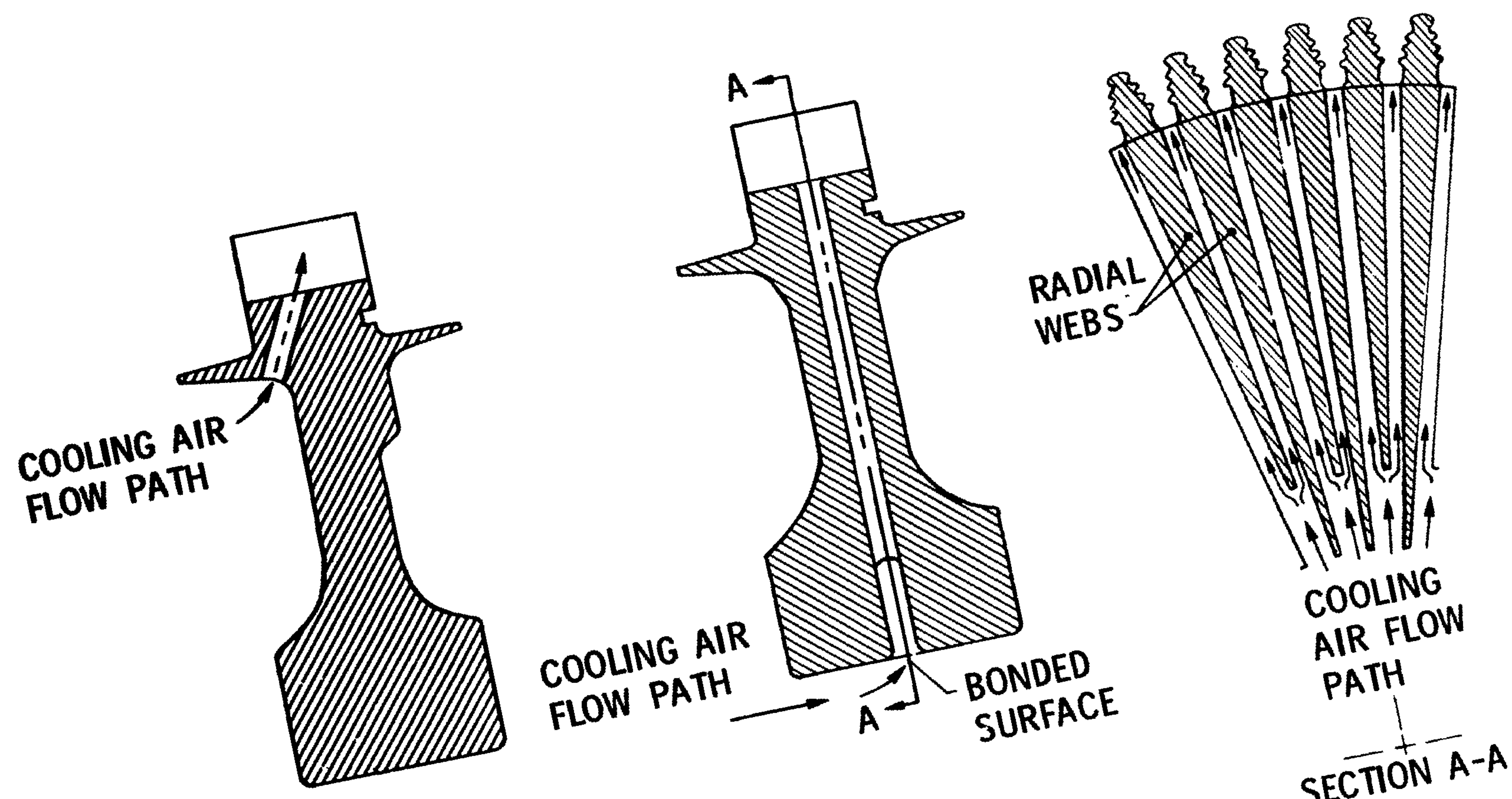
Figure 1. - Continued.



(g) Multidisk.

(h) Spline disk.

Figure 1. - Concluded.



(a) Standard disk.

) Bonded bore-entry disk.

Figure 2. - JT8D-17 first-stage turbine disk designs.

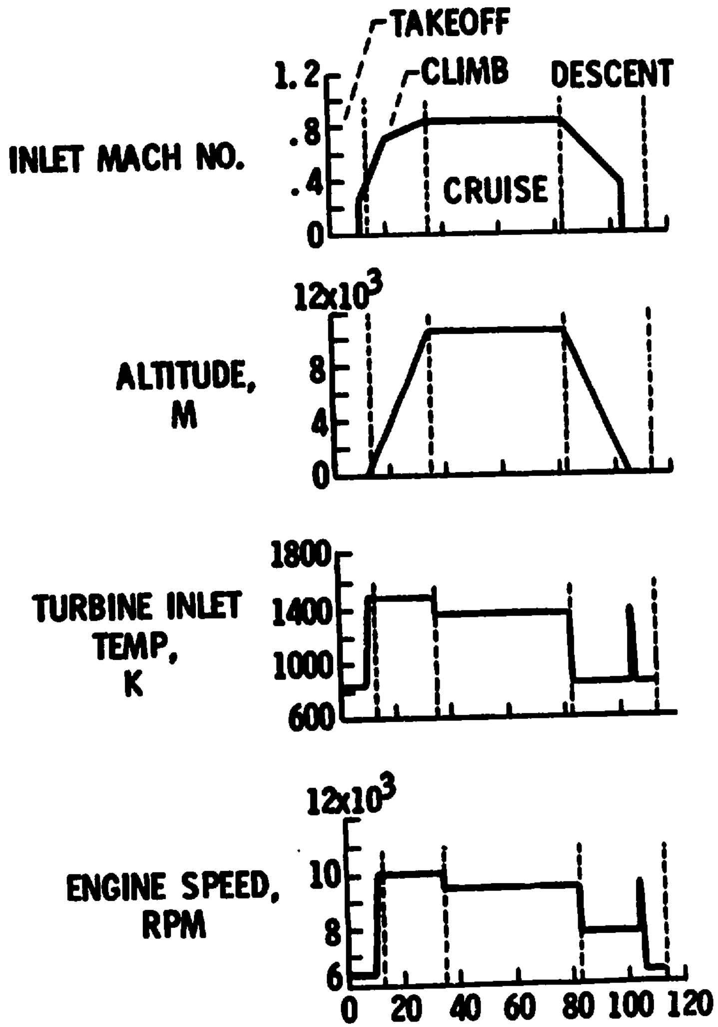


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Figure 3. - CF6-50 simplified engine ,cle.

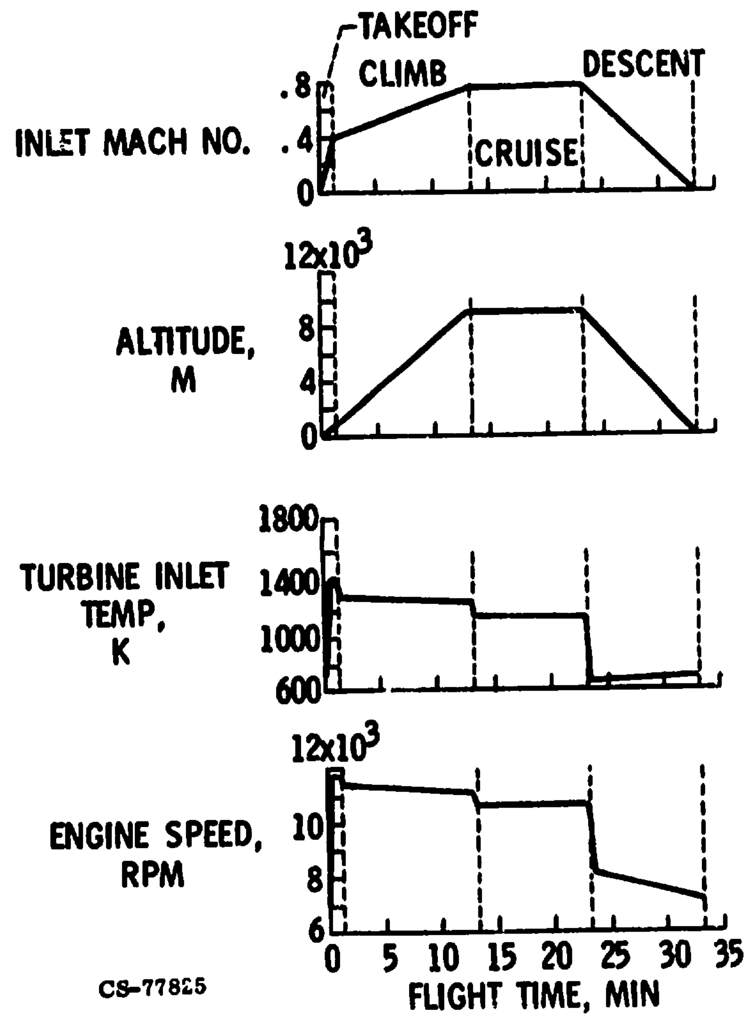

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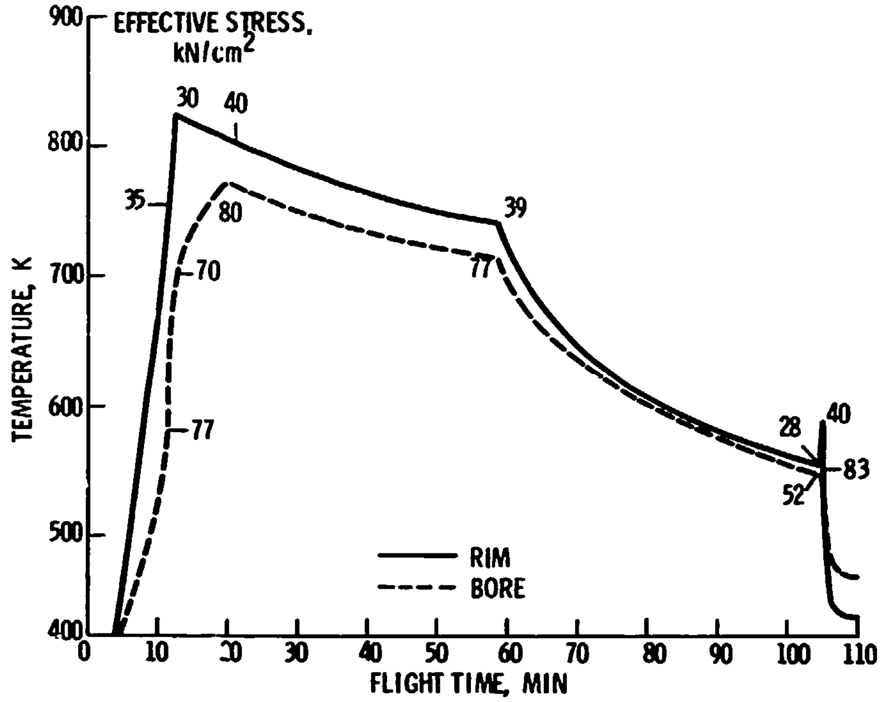
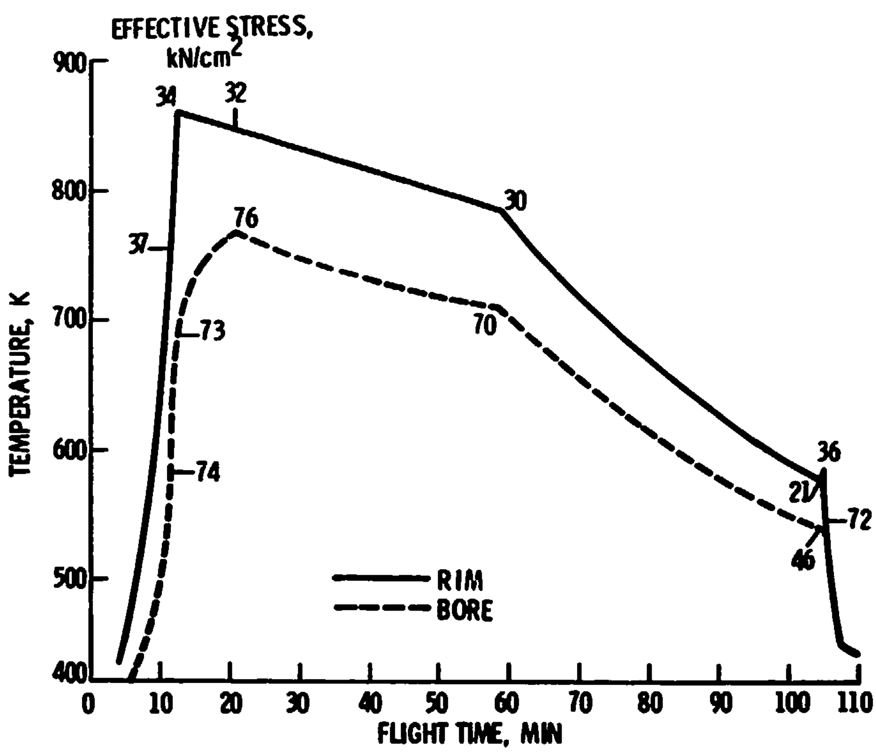
Figure 4. - JT8D-17 simplified engine cycle.

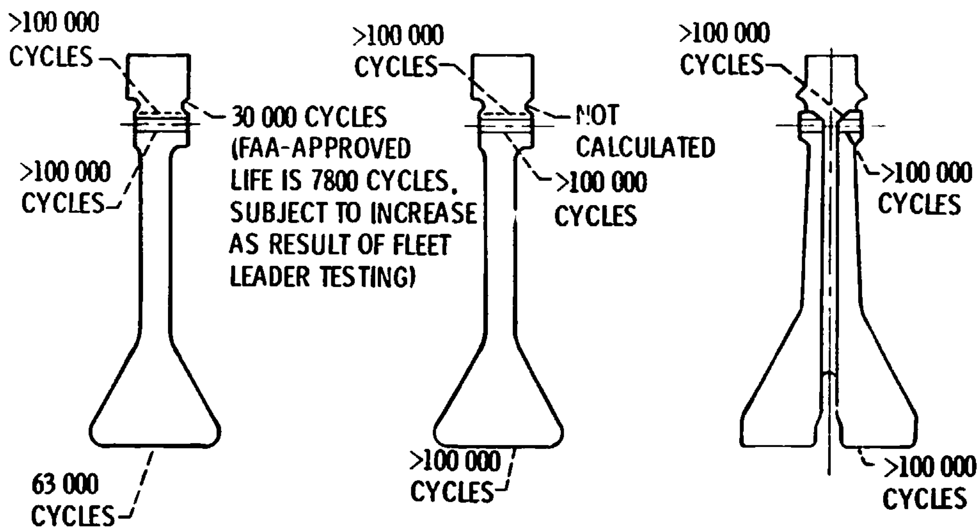


(a) INC-718 standard disk.



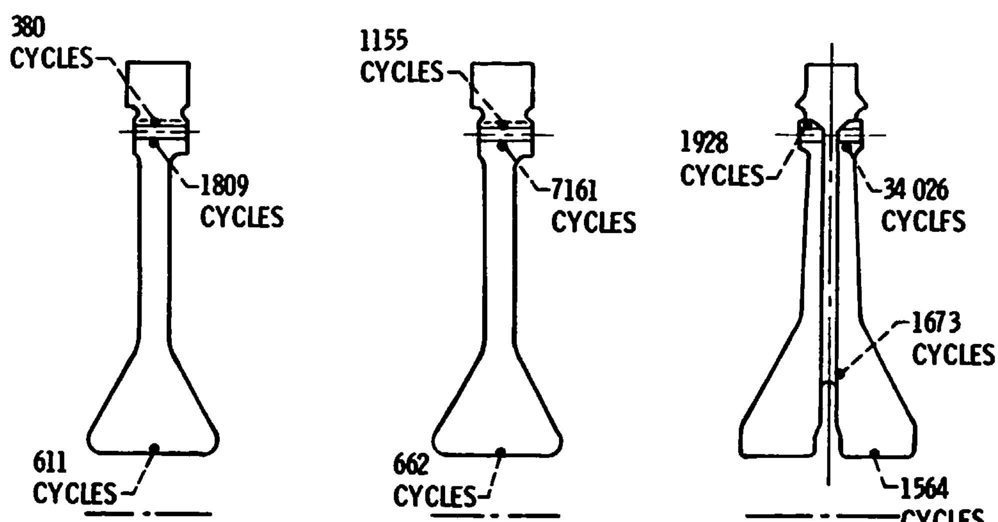
(b) Bore-entry disk.

Figure 5. - CF6-50 turbine disk temperature response.



(a) Standard disk (INC-718). (b) Standard disk (Rene' 95). (c) Integral bore-entry disk (Rene' 95).

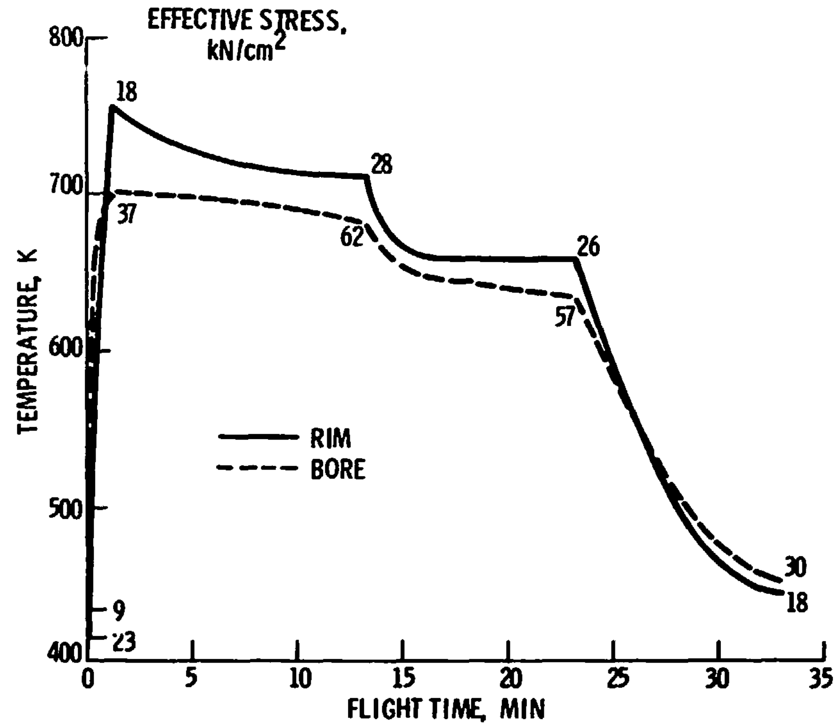
Figure 6. - Crack initiation lives of CF6-50 first-stage turbine disk designs.



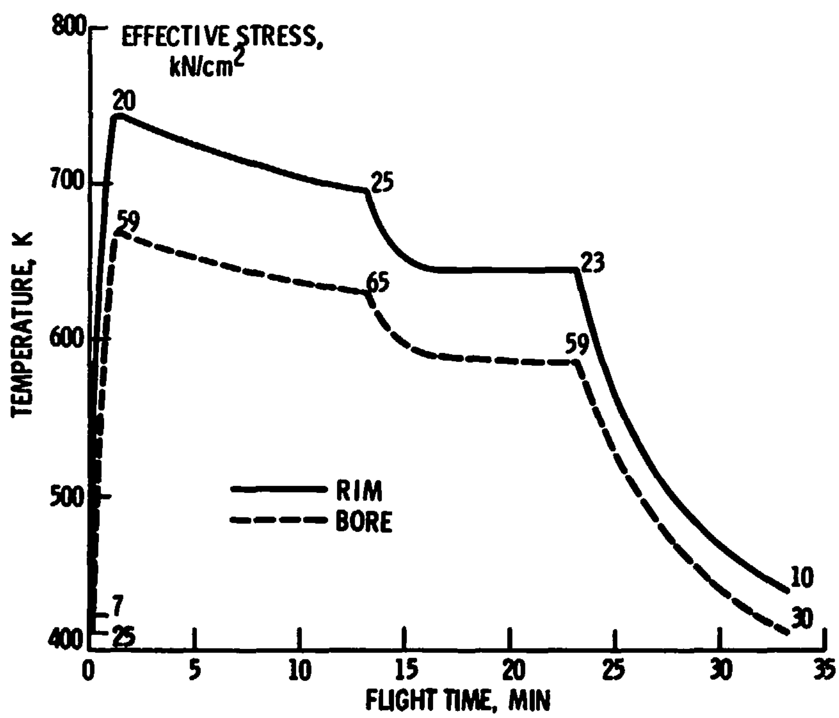
	(a)	(b)	(c)
BURST SPEED, PERCENT OF MAXIMUM TAKEOFF SPEED:	126	134	149
WEIGHT CHANGE, PERCENT OF STANDARD DISK WEIGHT:	---	0	+66

(a) Standard disk (INC-718). (b) Standard disk (Rene' 95). (c) Integral bore-entry disk (Rene' 95).

Figure 7. - Crack propagation lives of CF6-50 first-stage turbine disk designs.

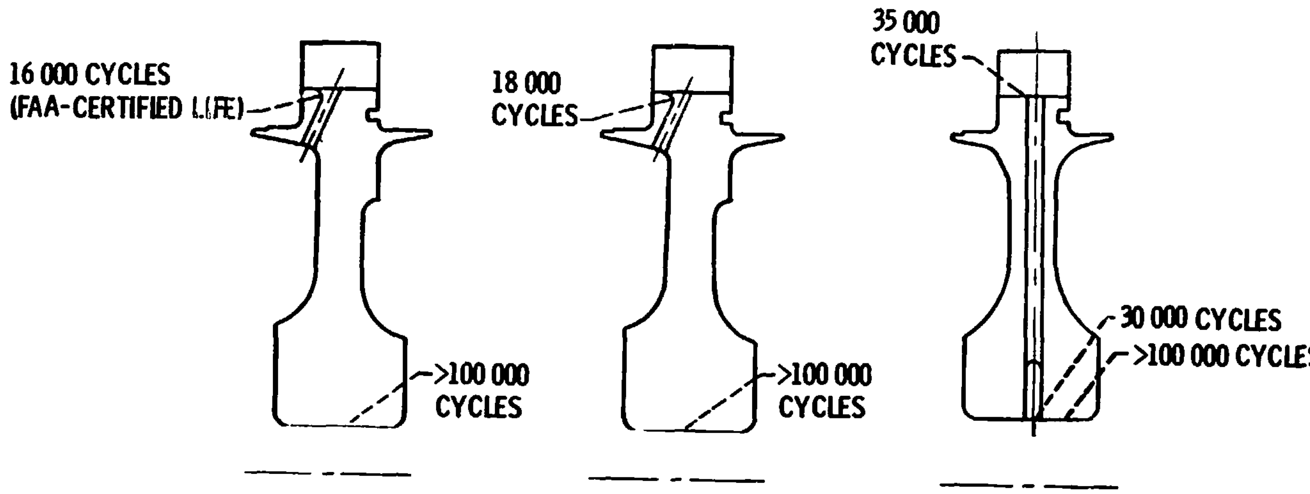


(a) Waspaloy standard disk.



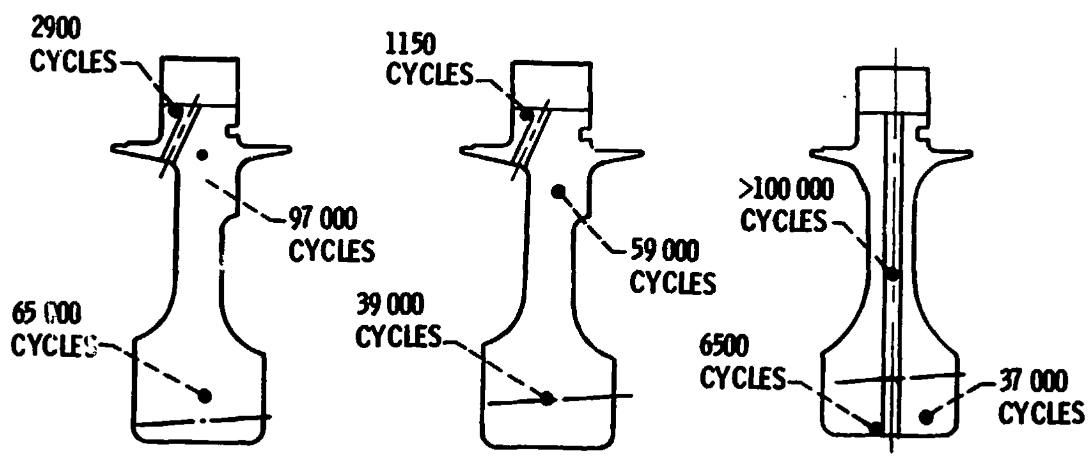
(b) Bore-entry disk.

Figure 8. - JT8D-17 turbine disk temperature response.



(a) Standard disk (Waspaloy). (b) Standard disk (Astroloy). (c) Bonded bore-entry disk (Astroloy).

Figure 9. - Crack initiation lives of JT8D-17 first-stage turbine disk designs.



	(a) Standard disk (Waspaloy).	(b) Standard disk (Astroloy).	(c) Bonded bore-entry disk (Astroloy).
BURST SPEED, PERCENT OF MAXIMUM TAKEOFF SPEED:	136	136	133
WEIGHT CHANGE, PERCENT OF STANDARD DISK WEIGHT:	---	-4	0

Figure 10. - Crack propagation lives of JT8D-17 first-stage turbine disk designs.

DISCUSSION

J.H. Gerstle, Boeing

With respect to that bonded bore entry disk, it is a fortunate thing from an airframer's point of view that turbine disk failure fragments tend to always stay in the plane of rotation. If you remember Denis McCarthy's figure, he showed a dispersion of about ± 3 degrees. We looked into it and saw something very similar: very narrow dispersion angles. I wonder if you would care to comment on if you had such a split disk design, whether the pieces would tend to fly out of the plane, which would gain us a little bit in reduced fragment energy, but from a configuration point of view the problem would be worse.

A. Kaufman, NASA-Lewis

Well, I really cannot comment much of that; that is kind of speculative. This isn't really answering your question but it's somewhat similar to it. One of the reasons that there is a disagreement between General Electric and PWA over whether you should bond disks or make them integrally is that GE is more worried about the bonded concept. I don't say this is right or wrong; but, they feel that a bonded disk is very likely to have an unbonded area which would propagate as you pile on cycles and create an unsymmetrical stress distribution. This would put some bending component on the disk, whereas an integral disk is more likely to be loaded uniformly and is more likely to do what your analysis predicts it will; it's more dependable. Pratt & Whitney seems to believe that this isn't a concern. I guess they have run some tests on this bonded design, but I don't think the tests are extensive enough to really set this concern to rest.

D.T. Poland, Lockheed-Calif.

One of the considerations in redundant structure is to have a safe-load life after you've experienced a failure so that safe operation will continue until we discover the failure. This means having a sufficient safe life to carry you between inspection periods so you can find cracks and failures on a scheduled inspection of the airframe and engines. I was wondering what considerations you had taken into account in this redundant design philosophy to allow for discovery of a failure in one of the redundant parts.

A. Kaufman, NASA-Lewis

This involved a little personal disagreement I have with the redundant design that GE did. I think it's a little over-redundant. The way they designed it was that in case of a failure, the undamaged disk could contain the failed part, and thus complete the operational life for which the disk was initially designed. I'm not sure that's not a dangerous concept because that could mean that the pilot would not be aware of the failure of part of the disk and could carry that failed part along. It seems to me that you want to have some redundancy but not too much. It would seem to me that the ideal redundant construction should be one in which you could contain the failed part long enough to get the airplane down to the ground, but by rubbing or some other means the pilot would have some warning that something is wrong. If you overdesign it so it has a long life, the operator could be containing a failed part for thousands of cycles between overhauls, and then it could let go unexpectedly. I hope I've answered your question.

Unknown Speaker

I think it's worthwhile to point out that these pilots are very sensitive to any situation that happens. If you lose a piece of disk I do not think there is much doubt that he will not know it.

A. Kaufman, NASA-Lewis

In that redundant design if you initiate a crack in the bore, you're not going to lose a piece of disk. What you say is true if you lose a piece near the rim. But, I'm concerned that you may initiate a radial crack which may go up an inch or so and then be contained on the failed part. I don't think they have really wrung through that analysis. I'd be afraid that the pilot might not be aware of it, and the thing could come apart somewhere between overhauls. So, I think, a small amount of redundancy is what you want to aim for, and you don't want.that crack to ride along in the aircraft too long.

G.L. Gunstone, CAA-UK

It's quite a thought, Mr. Chairman, that in the whole airplane, the disks and the shafts in the engines are the only parts which are single-element items, failure of which is potentially catastrophic. On the engine side, we are 20 or 30 years behind the aircraft people in that particular respect. I think that getting redundancy or fail-safe systems is about the only solution I see as valid unless we adopt the other types of approaches we've talked about (that is, making the aircraft withstand the debris). If we're going actually to prevent the disks from failing in their own right, this is about the only way I see of it being possible.

J.C. Wallin, BAC

I just want to follow up briefly on the comments of the last two speakers. Following airframe practice, it is not going to be any good having redundancy in engines unless you have the inspection methods to go with it, so that you can detect the cracks before they become catastrophic. The way we use redundancy in the airframe structure is to have routine inspections which will pick up the flaws so you can do something about them before they get out of hand.

A. Kaufman, NASA-Lewis

I think we're talking of a redundant construction that when you get that plane down you're going to have a fairly good-size crack, maybe an inch long, maybe longer. There's always a possibility the failed part will be hidden but I would think (that aside) the inspector should be able to detect it readily.

J.T. Dixon, P&W-Florida

What the man from GE said, it's hard for me to believe, first off, that you can contain that portion with the estimated big eccentric load that you have there. But, if you do and if you can, I don't think you're going to worry about the pilot knowing. You're going to have such large deflection that those blades are going to contact the vanes, unless you're going to increase engine length, and you add extra weight for extra engine length. So, I don't think you have to worry about excessive redundancy. A failure will be picked up pretty quick.

A. Kaufman, NASA-Lewis

It may not be detected readily but I can see what you're saying: if that flaw propagates really extensively, this will redistribute the stresses and throw an eccentric load on the disk. But suppose that the flaw grows to just an inch from the bore, I'm not sure you're going to get that much of an eccentric load that it is going to be felt.