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FACTORS THAT AFFECT OPERATIONAL RELIABILITY OF TURBOJET ENGINES

By LEWIS CENTER STAFF

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FACTORS THAT AFFECT OPERATIONAL RELIABILITY OF TURBOJET ENGINES [1]

CHAPTER I

OBJECTIVES

By Benjamin Pinkel

This report discusses the problem of obtaining high operational reliability for turbojet engines. By high operational reliability is meant operation with low probability of flight accident. High operational reliability is easier to achieve and operating costs are reduced with engine components of increased life. The considerations involved in increasing component life are briefly discussed.

High operational reliability can be obtained by proper practices in engine design, manufacture, flight operation, and maintenance based on an accurate knowledge of the characteristics of the engine and its components. Studies of these characteristics have been made at the Lewis laboratory. This report reviews the pertinent characteristics of the engine and its components and discusses in the light of this information the kinds of action necessary to improve operational reliability.

As a starting point, the failure data on jet engines in military service were examined. These data served mainly to reveal the components that were the principal sources of difficulty, and they provided only a rough indication of their modes of failure and failure times. A study was also made of flight accident records to determine the severity of the operational difficulties that resulted from failure of these various components.

The service records revealed the failure of a large variety of components. Some of these components have no unusual stress or wear problems and with proper design should last indefinitely. Other components have finite lives because of wear, fatigue by vibration and thermal cycling, and creep. Turbine buckets, turbine disks, bearings, and combustor liners are examples of parts with finite lives. The lives of these components in many of the current military engines are considerably shorter than desired times to overhaul. Unpredictable failures of some of the components come from environmental causes such as foreign-object damage. Foreign-object damage ranges from immediate destruction of the engine to nicking of the compressor and turbine blades which can result in reduction in their lives. Shortened life of components in the hot end of the engine results from overheating or overstressing as a result of malfunctioning of the automatic control or bad handling practice by the pilot.

If the wear-out or failure times could be accurately specified, high operational reliability could be achieved even with components of short life by a proper replacement schedule. However, the normal scatter in material properties and differences in severity of the operational histories of individual engines place part of the burden of preventing failure in flight on service-inspection procedures. Hence, the scheduling of replacements and of inspections is part of the procedure for improving operational reliability.

[1] Supersedes NACA Research Memorandum E55H02 by Lewis Laboratory Staff, 1956.

These inspection and replacement procedures must be derived from a knowledge of the failure mechanisms of the components. This derivation will be discussed.

Increased reliability can be obtained at a sacrifice in performance or an increase in cost. Examples of methods of increasing reliability at a sacrifice in thrust per unit weight are:

(1) Strengthening the engine (e.g., through the use of a centrifugal compressor in place of the axial compressor and more rugged critical components)

(2) Reducing operating temperature and stress

Examples of methods of increasing reliability at the expense of greater initial cost or greater maintenance cost are, respectively:

(1) Increasing development effort and quality control

(2) Increasing frequency of inspection and replacement of parts

The finding of the best compromise between reliability, performance, and cost is a special study for each engine and each application. For example, specific thrust is given more importance relative to these other factors in the fighter than in the transport application. This type of analysis will not be attempted. However, it is hoped that the insight presented into the characteristics of the engine and its components will provide guidance for these special analyses.

Unfortunately, all the data needed for the discussion of the operational reliability problem were not available. The additional information needed will be pointed out.

In summary therefore, an attempt will be made to provide the following:

(1) An analysis of statistical data on failure of engine components in service to reveal (a) the most critical engine components, their modes of failure, and failure times; and (b) the tendency of the component to cause engine failure and flight accident

(2) A review of the theory and experimental data relating to the engine and its components that reveal (a) the causes of component failures, (b) the manner in which the components fail, (c) the factors that influence failure time and the relation between performance and failure time, (d) the phenomena that indicate incipient failure and the grace time between incipient and final failure, and (e) the effect of the component failure on the engine

(3) A discussion of the measures required to improve operational reliability, covering the following activities:

(a) Design and manufacture
(b) Inspections
(c) Replacements
(d) Maintenance practices
(e) Flight operational practices

(4) Additional information needed

The records reveal failure of a large number of miscellaneous engine components. However, this report covers only components which are pressed by turbojet performance requirements to operate at conditions where uncertainty exists regarding the design factors and the behavior of materials. Because foreign-object damage and difficulties arising from inadequate or defective control can greatly reduce component life, these topics are also included.

The NASA is indebted for statistical data and valuable discussion to personnel at Wright Air Development Center, the Air Materiel Command, Oklahoma City Air Materiel Area, the Directorate of Flight Safety, the Bureau of Aeronautics, and the aircraft engine industries.

In order to avoid proprietary difficulties, engines and aircraft are designated by a code system.

CHAPTER II

FAILURE STATISTICS

By Floyd B. Garrett and G. M. Ault

SUMMARY

Service records of turbojet engines in Air Force military service were sampled and studied. Although the records were not designed for this purpose, considerable insight was gained as to the causes and frequency of turbojet-engine failure. Data required for an improved future study are suggested.

The time to overhaul of the average turbojet engines of three models for which considerable experience has been obtained varied with engine model and application from 105 to 760 hours. Most of the engines were near the low side of this band. The most frequent and consistent cause of overhaul was foreign-object damage, from 26 to 59 percent of the engines being overhauled for this reason.

Failures in the hot section (i.e., in the combustor and turbine sections) are a frequent cause for engine removal from aircraft. Also, large percentages of the engines going through field repair and major overhaul require replacement and repair of these components.

Failures of some engine parts have caused flight accidents. In 1953, 205 accidents were due to jet-engine failure or malfunction. The responsible component was determined for 182. In decreasing order of frequency, these were fuel-control failure (68 accidents), compressor failure including foreign-object damage (54 accidents), turbine bucket failure (16 accidents), turbine disk failure (14 accidents), main bearing failure (10 accidents), and finally miscellaneous causes (20 accidents from 16 different causes). Of the 54 accidents resulting from compressor failure only one involved a centrifugal compressor, although the flying time accumulated for engines having centrifugal compressors was about the same as for engines having axial compressors.

INTRODUCTION

Service records of turbojet engines were studied in order to obtain insight into the causes of engine failure. Since the most extensive use of these engines has been by the U.S. Air Force, the records of this organization were examined in some detail.

This paper summarizes the results of this study and indicates, where possible, the time to major overhaul obtained on these engines, the engine components that most commonly failed, and the operating lives of some of the components. In addition, accident records for jet-powered aircraft for the year 1953 were reviewed to indicate component failures that have been important causes of accidents. Since the records available were not designed specifically for the study of engine reliability, comments are offered as to the records desired to facilitate future studies such as described herein. Two appendixes are included to describe in general terms the actuarial method of determining engine life and the two general laws of failure, "chance" and "wear-out."

The statistical data herein are for engines of older design that have been operated in the military services. Sufficient operating experience is not yet available for later engines to permit such a study. It is inaccurate to extrapolate all the implications of these data on older engines to the performance of engines of the latest design and to operations in other services, such as in commercial transport. The later designs have corrected many faults of the older designs; however, they also can be expected to possess faults yet to be discovered. By bringing to light all possible information on several engines for which considerable operation has been experienced, it is hoped that the later designs may avoid the faults of the older designs.

In addition, for reasons given herein, this statistical study is limited as far as possible to engines that had never been previously overhauled and, therefore, to engines of relatively short operating times. For some components, particularly the turbine disk, long-time operation can be expected to introduce failures not revealed herein. In these cases, more data are presented in the other parts of this report that discuss the individual components.

The engines and aircraft are coded. For aircraft, the letter B indicates bomber aircraft, C indicates cargo aircraft, and F indicates fighter aircraft.

SOURCES OF DATA

The Air Force records from which data were available are shown in table I. The first column lists the titles of the records, and the third column tabulates information obtained from these records for this study. Since the records were collected by the Air Force for purposes other than the type of analysis presented in this report, there are some limitations; these are listed in the fourth column. The time periods studied are indicated in the last column. The first source is the Aircraft Engine Life Expectancy Exposure Table, published monthly. This record gives the operating time since manufacture or since last overhaul of all engines installed in Air Force planes. Each engine is not listed separately; but, rather, the engines of each model are grouped into 10- or 20-hour intervals of flying time. This record also gives the number of hours flown each month and the number of engines removed for major and minor overhaul. From these data the Air Force calculates the expected time to overhaul (life expectancy) for each engine model. For the present study one limitation is that this record does not distinguish between "new" (not previously overhauled) and overhauled engines.

The second source is the Engine Technical Order Compliance and History Record, commonly called the 60B. This is essentially an engine log that stays with the engine throughout its life. The record is used to note compliance with Technical Orders, that is, to note whether recommended engine modifications have been made and to note when the engine was transferred or overhauled. Since it is not specifically required that all part changes be noted, it is not possible to follow the life of the engine components from this record; therefore, the 60B's were not used in the present study.

The Engine Removal or Loss Report (ER), which is published monthly, tabulates for each engine removed from an airplane (that is not immediately reinstalled in the same position in the same airplane) the cause for engine removal, the operating time on the engine, and whether the engine is "new" or has been previously overhauled. The cause for removal is based on the information available to the pilot and the crew chief or line officer who orders the removal. And, since the engine is only partially disassembled and inspected, limitations exist. For example, an engine may have been removed because of excessive vibration; it would be important to know whether the vibration resulted from a fatigued bearing or a failed compressor blade, but that will not be known until the engine is disassembled. Also, if a part is changed while the engine is mounted in the airplane or if the engine is removed, repaired, and immediately reinstalled in the same position in the same airplane, no ER notation would be made.

When an engine is removed because repairs are needed, it can either be repaired in the field or shipped to an overhaul depot. The Air Force has in operation an extensive program of field "minor repairs" for jet engines that permits all hot-section components (combustors, nozzle diaphragms, disks, and buckets) and turbine-shaft bearings to be replaced in the field rather than requiring the engines to be sent to major overhaul. Disassembly of the compressor rotor to replace failed parts was not permitted in the field at the time of these data. Therefore, field repair data emphasize hot-section part replacements whereas data giving reasons for major overhaul tend to emphasize difficulties within the compressor, including foreign-object damage, and within the accessory drive section of the engine. No single source of data gives the complete story. Information on repairs made in the field are available from (1) Unsatisfactory Reports (UR), and (2) a special Summary of Field Maintenance and Repair.

A UR is written at the option of field personnel when a difficulty is found with an engine or engine component. Since a UR is written at the

TABLE I.—USEFULNESS AND LIMITATIONS TO PRESENT ANALYSIS OF STATISTICAL DATA FROM SEVERAL SOURCES

Source	Frequency of publication	Information useful in present analysis	Limitations for present analysis	Time period of data used in analysis
Aircraft Engine Life Expectancy Exposure Table	Monthly	1. Operating time on all engines in service 2. Number of hours flown per month 3. Usage removals for major overhaul per 10-hr period 4. Total removals for major overhaul per 10-hr period 5. Removals for other than major overhaul per 10-hr period 6. Engine life expectancy (actuarial calculation)	Mixes new and previously overhauled engines	July to September 1954
Engine Technical Order Compliance and History Record (60B)	Continual	1. Data and operating time of: a. Technical order compliances b. Transfers c. Overhauls 2. Reasons for removal of accessories 3. Work performed at depots	Incomplete record of parts replaced	Not used
Engine Removal or Loss Report (ER)	Monthly (confidential)	1. Reason for removal 2. Maintenance required (whether major, minor overhaul, or none) 3. Operating time since new or last overhaul 4. Total time since manufacture 5. Number of overhauls since manufacture 6. Number of minor repairs since new or overhaul	1. Information based on limited disassembly 2. Engine must be exchanged to be included as a removal (an engine removed, repaired, and returned to same aircraft and position is not recorded as a removal) 3. Data incomplete because information reported by code letter	May to July 1953
Unsatisfactory Report (UR)	As needed	Description of unsatisfactory condition and time since overhaul	1. Form not mandatory 2. Life of parts not recorded	Not used
Statistical Summary of Deficiencies Reported by Unsatisfactory Reports	Monthly	Summary of unsatisfactory conditions reported by UR's	No information on engine or component life	Not used
Summary of Field Maintenance and Repair (not a standard report—especially prepared by OCAMA)	Special	Summary of components repaired or replaced	1. Life of parts not recorded 2. New engines not separated from old (or new or repaired parts)	April to June 1954
Disassembly Inspection Report (DIR)	As needed	1. Cause for overhaul 2. Other parts that needed replacement during overhaul (simultaneous replacement) 3. Total time since manufacture 4. Total time since last overhaul 5. Number of previous overhauls 6. Previous minor repairs where given	Life of parts unknown	August to October 1953
Statistical Summary of DIR's	Monthly	Current month's trend and accumulated trend for year of causes for overhaul	Gives only cause for overhaul; other necessary repairs not recorded	Not used

discretion of field personnel, there is no assurance that one will be written each time a repair or part change is made. UR's, therefore, cannot be said to give a complete picture of a difficulty. Generally, however, they do give a qualitative picture of problems that occur. Such a report gives a description of the unsatisfactory condition encountered and the operating hours on the engine since "new" or since the last overhaul. The operating time on a part that may have been previously changed several times is not given. The UR's are summarized monthly, and the summary is published as a Statistical Summary of Deficiencies Reported by Unsatisfactory Reports. This form tabulates for each engine model the number of times for the month and accumulated for the year that a particular difficulty has been reported. The summary is in very broad terms (e.g., number of "internal failures," "vibration difficulties"). This report does not indicate engine operating times.

Another source of field maintenance and repair data is a special Summary of Field Maintenance and Repair made available by OCAMA. This tabulation indicates the percentage of each engine model going through the field repair program that has a particular part replaced. It does not indicate the operating time on the engines, or whether the engines are new or have previously been overhauled or repaired.

When an engine goes through major overhaul, a Disassembly Inspection Report (DIR) is written. This describes, in the opinion of the inspector, the particular part failure or other reason (e.g., foreign-object damage) that caused the engine to require overhaul and describes in considerable detail all the part replacements made for each engine. Currently, this is the most complete of all engine records. The major limitation is that, even for an engine never previously overhauled, it is not certain that a part replaced in overhaul has not also been previously replaced in the field. Thus, the operating time on any part cannot be stated with certainty. Engines A and B overhauled at OCAMA had previous minor repairs noted on the DIR's insofar as the data were available to the inspector from the engine 60B's. Minor repair data are not noted on DIR's for C engines. The DIR's are summarized monthly, and the summary is published as a Statistical Summary of Disassembly Inspection Reports. This summary tabulates for each engine model the number of times for the month and accumulated for the year that a particular part failure, environment (e.g., foreign object), or operating condition (e.g., overtemperature) has caused an engine to require major or minor overhaul. The summary separates new engines from those previously overhauled. The engines of each model are grouped on the basis of reason for overhaul, and the average operating time since manufacture or the last overhaul is given for each of these groups.

The remaining sections of this paper describe the results obtained from a study of several of these sources for various models of three engines. For each source used, data covering a complete 3-month period were studied as indicated in table I. The same time period could not be used for all sources because the data were not available. In the case of DIR's, the time period represents the time period of overhaul. Some of the engines had been removed from service several months before overhaul. The DIR's were not available to permit grouping on the basis of time of removal from service, although it might have been preferable to do so.

The Air Force revised its methods of collecting data on part failures and replacements in February of 1955 (ref. 1). An important difference in the new system is that every part replacement or repair must be noted on a special form and sent to a central agency. With these data the true magnitude of part replacements will be readily available. In addition, when an engine undergoes field repair or inspection, a DIR will be written similar to that now written for overhaul. Records are discussed more fully in the section SERVICE RECORDS DESIRED.

COMPARISON OF JET- AND RECIPROCATING-ENGINE LIFE

The times to overhaul (engine lives) of aircraft engines used in military aircraft are given in reference 2. To provide a crude yardstick for measuring engine life, the times to overhaul of the jet engines are compared with the times to overhaul of reciprocating engines of 2000 horsepower or greater also in use in military service.

Comparisons of the lives of jet and reciprocating engines may not be entirely fair, because the engines do not perform the same function. The reciprocating engines are used by the Air Force to power bomber and cargo aircraft, whereas the jet engines power bomber and fighter aircraft. Comparisons of engine lives made herein are based on the times to overhaul determined by actuarial computations described in references 2 and 3 and appendix B herein. For reasons described in appendix B, engine lives are compared herein on the basis of the median time to overhaul rather than the mean. The median time to overhaul, which is the time when 50 percent of the engines of a sample will go to overhaul, is called "the life of the average engine" as contrasted with the mean, which is the average life.

In general, the life of the average jet engine is appreciably less than that of the average reciprocating engine. This may be seen in table II, where for the jet engines the life of the average engine ranges from 105 to 760 hours, with most models on the low side of this band. For several models of reciprocating engines, the life of the average engine ranges from 340 to 1140 hours, with many models above 650 hours.

The characteristic shape of these distribution curves is indicated by the sample cumulative frequency distributions plotted in figure 1 (see appendix B for description of plot). Besides the indicated differences in lives of the average engine, another comparison can be made by noting the percentage removed for overhaul after a particular time period. For example, at the end of 100 hours, for the jet engine, 31 percent of the C–7's, 28 percent of the B–3's, and 11 percent of the A–7's would go to overhaul; whereas, for the

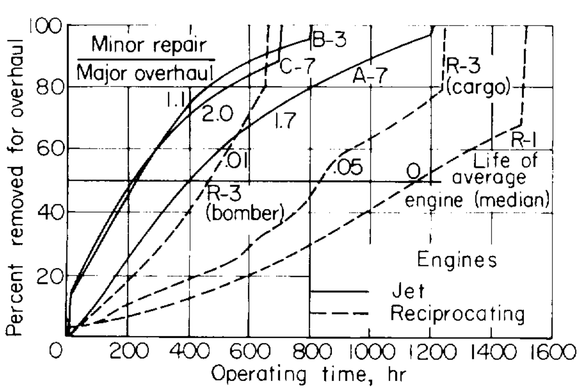


FIGURE 1.—Comparison of time to overhaul of several engines on the basis of cumulative frequency distribution.

TABLE II. COMPARISON OF MEDIAN TIME TO OVERHAUL FOR JET- AND RECIPROCATING-ENGINE AIRCRAFT IN MILITARY SERVICE

Engine code	Aircraft code	Median time to overhaul (ref. 2), hr	Number of engines in service as of February 1955 (ref. 10)
Turbojet:			
A–6	F–4	105	34
A–9	F–4	110	485
A–10, 13	F–4	110	90
B–7	B–2	120	1, 143
A–3, 4, 5	F–2	140	278
B–6, 11	F–3	150	1, 213
B–1, 2, 3, 4	F–3	160	405
A–7	F–2	220	527
B–3	B–4	240	427
B–2, 4, 5	B–3	260	275
B–10	F–3	270	1, 532
C–7	F–1	410	2, 998
C–6	F–5	440	254
B–8	B–4	[a] 610	855
B–9	B–4	760	5, 783
Reciprocating:			
R–4c	B2–1	340	433
R–4c	B2–2	350	179
R–3a	B–5	350	2, 532
R–3b	B–6	400	375
R–4b	B7–2	450	112
R–4b	C3–2	460	232
R–4e	B2–1	460	810
R–4b	B7–1	470	795
R–4e	B2–2	470	296
R–3c	C–1	520	901
R–4b	C3–1	680	252
R–4f	C3–2	690	2, 096
R–2b	B–1	740	1, 162
R–2c	B1–3	750	518
R–4a	C–1	800	604
R–4d	C–4	800	44
R–1a	B–8	820	1, 899
R–4a	C–2	850	764
R–4g	C–2	910	870
R–4b	B7–3	920	152
R–2a	C–5	940	582
R–2d	B1–2	1, 140	566

[a] From ref. 11.

reciprocating engines, only 4 percent of the R–1's and R–3's used in a cargo aircraft and 8 percent of the R–3's used in a bomber aircraft would go to overhaul.

It should be emphasized that these data indicate only the time to major overhaul. In the case of the jet engines, an extensive field minor repair program is in operation. The minor repair program for jet and reciprocating engines can be compared on the basis of the number of removals for minor repair per removal for major overhaul. The ratio is indicated in figure 1 and given for several other engines in figure 2. For the jet engines the number of removals for minor repair per removal for major overhaul varies from 0.5 to 17, whereas it varies from 0 to 0.1 for the reciprocating engines. From table II and figures 1 and 2, it is apparent that times to overhaul are much shorter for the jet engines in spite of the extensive jet-engine field minor repair program.

In addition to time to overhaul, it may be of interest to consider the number of removals of the engines from aircraft for repair or overhaul per unit of flight time. This comparison is shown in figure 3 on the basis of number of removals per 10,000 hours of flight time. The jets obviously have many more removals than

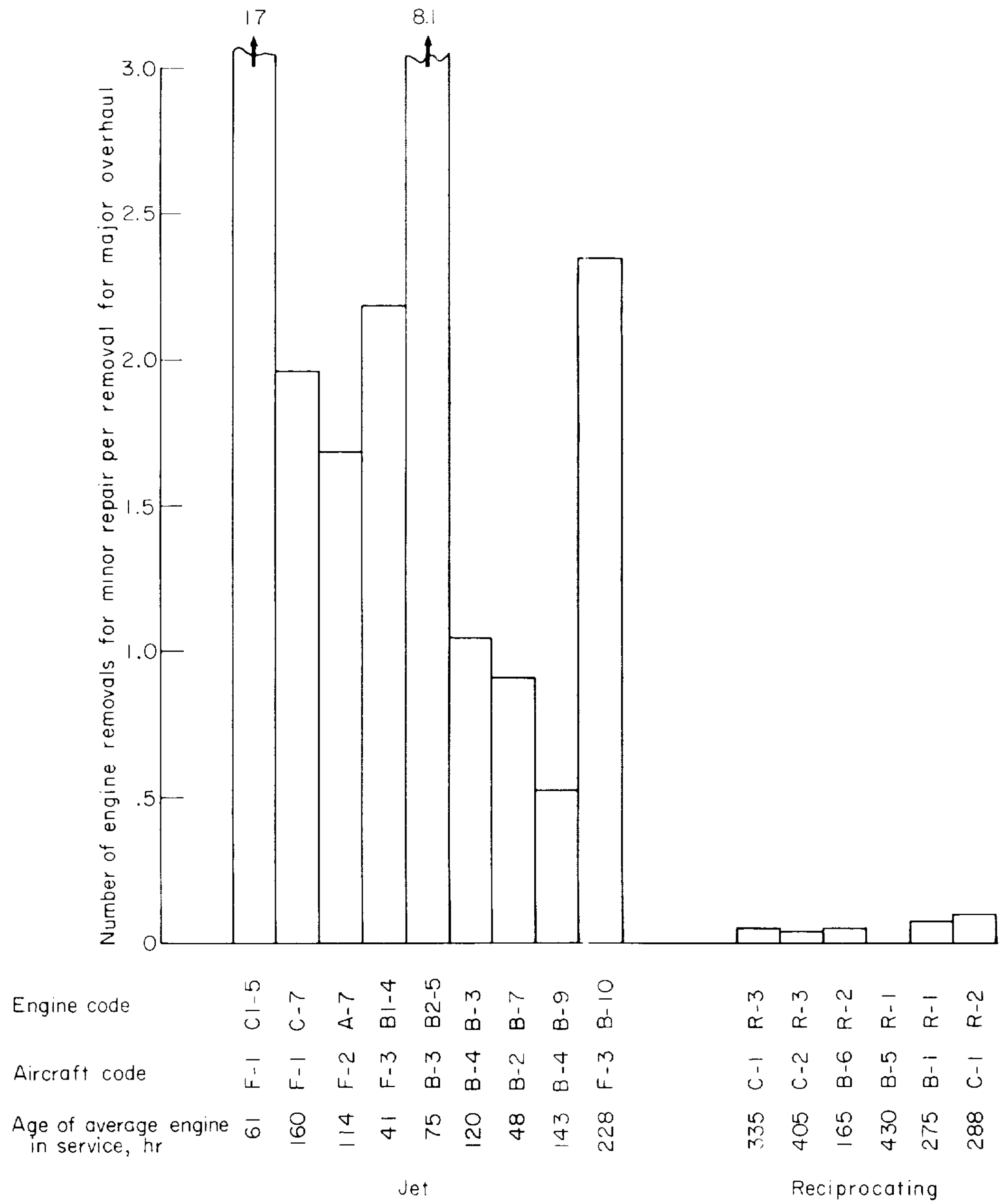


FIGURE 2. Comparison of jet and reciprocating engines on basis of number of minor repairs per major overhaul.

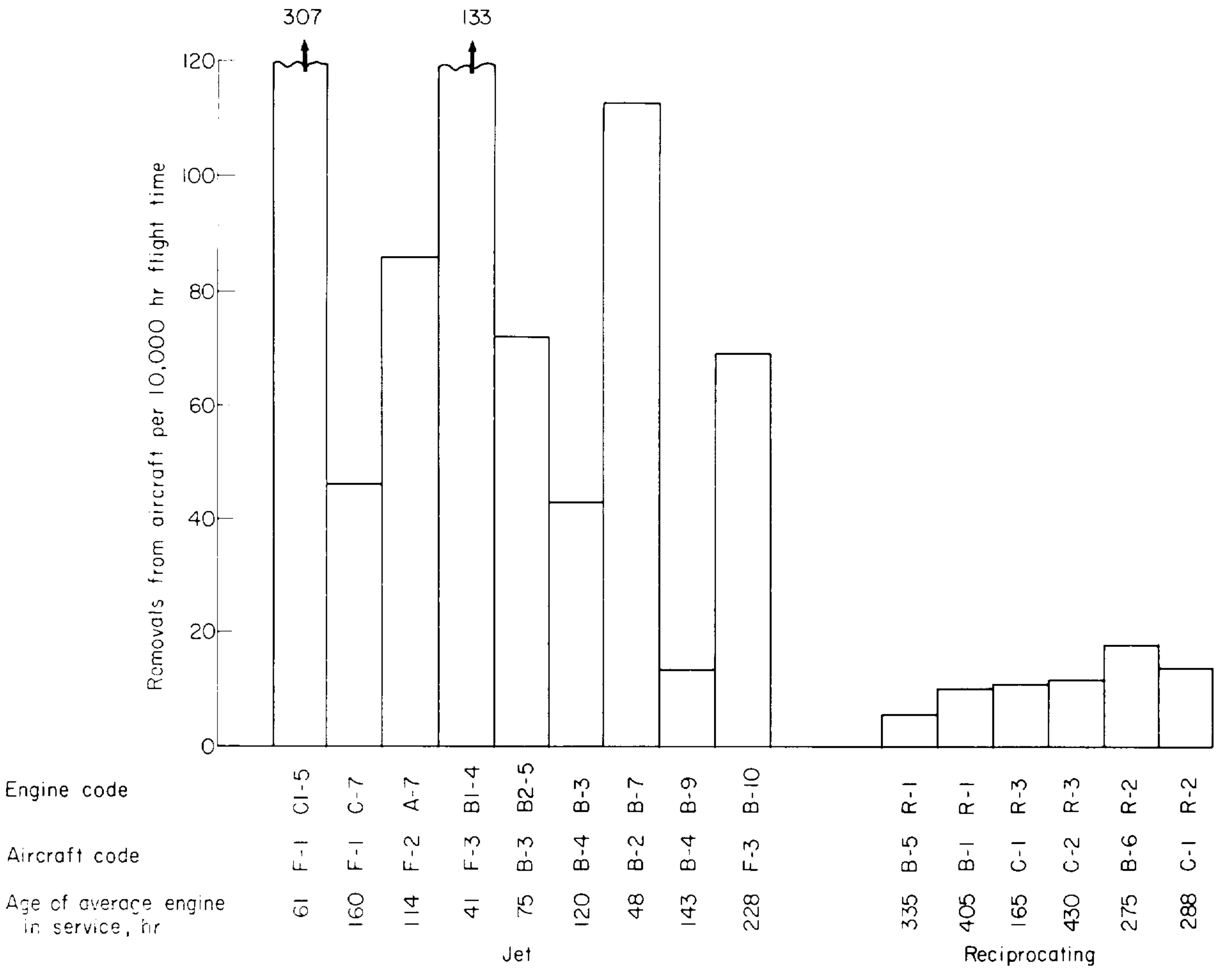


FIGURE 3. Comparison of jet and reciprocating engines on basis of frequency of engine removals for repair.

the reciprocating engines. The removal rate varies by engine model and application. The C-1 to C-5 jet engines in a training fighter had 307 removals per 10,000 hours of flight, whereas in the same application another model of the same engine, the C-7, had 46 removals.

The time to overhaul of jet engines is a function of the application and engine design, as is indicated in figure 4, where the distributions of time to overhaul of five models of the B engine are compared. The lowest median overhaul time, about 110 hours, is for the B-7 in a bomber. Another model of the same engine, the B-9, used in another bomber (B-4) has appreciably greater life (750 hrs). In fact (based on actuarial data), this engine has the longest median time to overhaul of any jet engine for which data are avail-

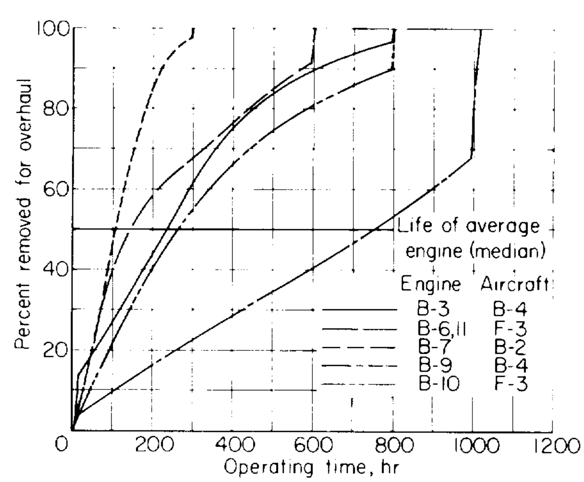


FIGURE 4. Effect of engine model and application on time to overhaul.

able. Another model of the B engine, the B-3, used in this same bomber aircraft (B-4) has an appreciably shorter median life (220 hrs). Another interesting point results from a comparison of the B-6 and B-11 with the B-10 engine. The engines are similar and power essentially the same fighter airplane, but the B-6 and B-11 engines having the lower life are equipped with afterburners and also operate more time at maximum engine speed.

ENGINE REPAIR AND OVERHAUL DATA

CAUSES OF ENGINE REMOVAL

It is of interest to consider why engines are removed from aircraft, what parts most frequently need repair in the field, and what is repaired in major overhaul. The causes for removal of engines from aircraft are summarized in figure 5 for five engine models from the Engine Removal Reports. (More accurately the figure summarizes causes of engine removal that result in installation of a different engine in the aircraft, since this is the only time an ER is written.) The stated cause reflects the information available to the pilot and crew chief or line officer who orders the removal. These men have probably looked into the inlet for compressor blade damage from foreign objects and perhaps removed the tailcone to examine the turbine section. Since the data are to be later tabulated by automatic machines, they are limited to a code system indicating the reason for engine removal. If a satisfactory code letter does not exist, the cause for removal is indicated with a code letter designating "other known reason, not specified by code."

The statistics on five engine models were studied. The high mark on each bar (maximum) represents the model having the largest percentage found for the specified removal cause, while the low mark (minimum) represents the model having the lowest percentage of this removal cause. The spread for each cause is represented in figure 5 by the height of the cross-hatched area.

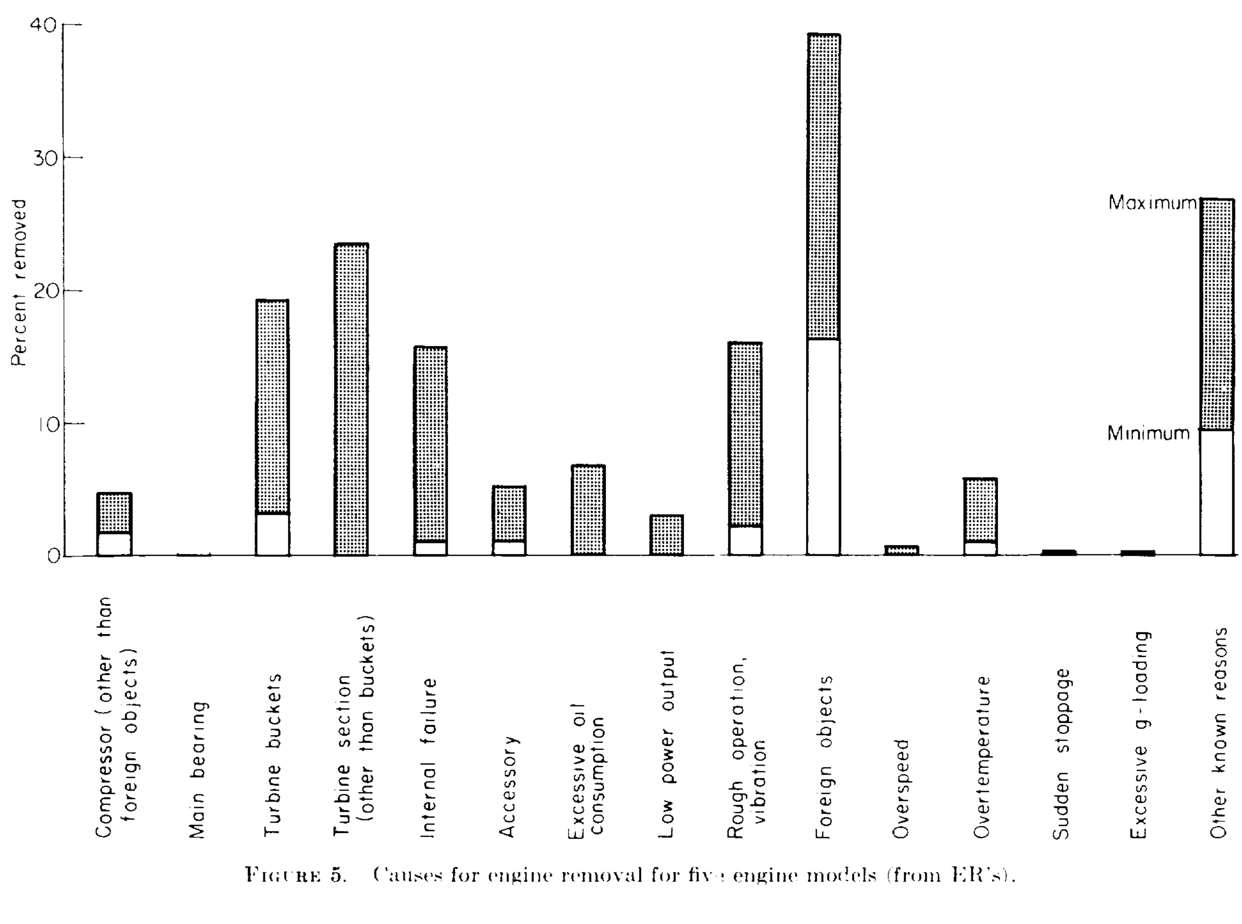


FIGURE 5. Causes for engine removal for five engine models (from ER's).

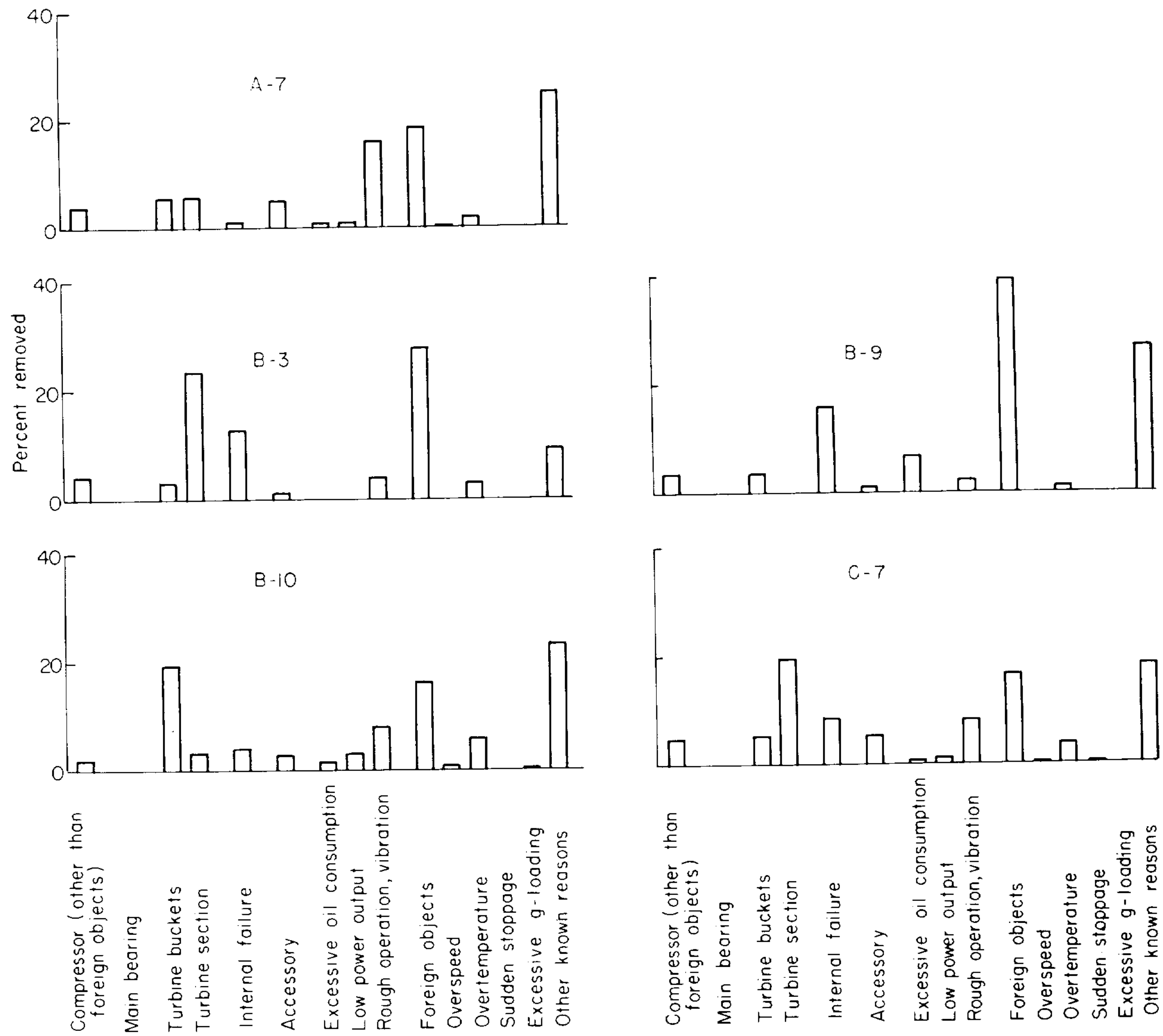


FIGURE 6.—Causes for engine removal by engine model (from ER's).

The distribution of reasons for removal among the five individual engine models is shown in figure 6. One of the largest categories for every engine model is the number removed for other known reasons not specified by code, which varied between 9 and 27 percent (fig. 5). The bearing category rarely shows any entries. If a main bearing were bad, it might be more likely noted under rough operation or excessive vibration, since the engine had not been disassembled to a point where those making out the engine removal form could see the damaged bearing. The cause of the rough operation or vibration would not be known until the engine could be disassembled in field repair or in overhaul. The importance of a particular component in causing engine removals varies from engine to engine. For example, the turbine section other than buckets caused 23 percent of the removals in one engine, but none in another engine. This is indicated by the fact that the shaded area goes down to the axis (fig. 5).

Another point is that foreign-object damage is consistently a problem. At a minimum, it caused 9 percent of the removals of one engine model and up to a maximum of 39 percent of another model. Other causes that were high were turbine buckets, internal failure, engine accessories, and rough operation or vibration.

PARTS REPLACEMENTS MADE IN THE FIELD

As mentioned earlier, when an engine is removed from an airplane for repairs, it can either be repaired in the field or sent to major overhaul. A 3-month summary of the field repair data is shown in table III and as a bar graph in figure 7, which shows the percentage of the A, B, and C engines going through field repair that had particular parts replaced. As mentioned previously, at the time these data were collected, replacement of all hot-section parts was permitted in the field. Disassembly of the compressor to replace stator vanes or rotor blades was not permitted, however. If a stator vane or rotor blade in the compressor needed replacing, the engine was sent to major overhaul. Thus, field maintenance data emphasize hot-section repairs, and major overhaul data tend to emphasize foreign-object damage and compressor repairs. The compressor could be reworked in the field, however, to "stone out" minor nicks or dents resulting from foreign objects; and, on an average, 18 percent of the B engines going through overhaul had compressors reworked because of foreign-object damage. In the later section Parts Replaced at Overhaul, those parts replaced because of foreign-object damage are isolated from other causes, but these field data apparently do not make this distinction; thus, some replacements of turbine buckets, for example, may have been because of nicks and dents from foreign objects as well as from cracking or fracture resulting from fatigue or stress-rupture. Also, the replacement of turbine wheels apparently does not necessarily indicate failure of the disk, since entire wheel assemblies are occasionally replaced in the field, even though

TABLE III.—SUMMARY OF FIELD MAINTENANCE AND REPAIR [a]

[Percentages obtained from 340 A, 1261 B, and 322 C engines that received jet-engine field maintenance ("minor repair") and were returned to service]

Engine model	Percent of engines having specified part repaired								
	Number 3 bearing	Number 4 bearing	Number 4 oil seal	Inner liners	Transition liners	Nozzle diaphragm	Turbine buckets	Turbine wheel assembly	Compressor reworked
A-5	0	0	0	73.7	7	50.5	41.4	25.3	1
A-7	14.3	19.8	1.2	44.4	0	40.4	17.9	13	29.8
A-8	0	0	0	55.5	5.5	38.8	0	0	44.4
A-10	0	0	0	53.3	0	33.3	0	0	6.7
A-6	5.6	0	5.6	33.3	5.5	50	0	7.1	0
A-13	0	0	0	71.4	0	14.3	0	0	28.6
Average A	3.3	3.3	1.1	55.3	3.0	37.9	9.9	7.6	18.4
B-3	0	0	4	56	52	60	4	8	8
B-1	14.3	0	28.6	57.1	42.8	42.8	42.8	0	0
B-2	0	12.5	0	37.5	37.5	62.5	50	25	50
B-4	2.8	4.9	9.2	33.1	33	38	28.1	28.2	9.9
B-5	0	6.3	0	6.2	37.5	46.8	43.7	15.6	9.4
B-6	2.8	8.9	7.2	22.8	55.5	23.9	7.8	39.6	10.6
B-7	4.8	7.1	0	16.6	9.5	11.9	4.7	28.6	7.1
B-8	0	4	5.1	37.3	24.2	5	23.2	4	30.3
B-9	1	6.2	6.2	17.6	26.9	4.1	12.4	10.9	40.9
B-10	2.5	8.2	8.2	51.1	3.9	16.3	34.2	8.5	23.7
Average B	2.8	5.8	6.8	33.5	32.4	31.1	25.1	16.8	19.0
C-1	0	42.8	44.3	0		14.3	14.3	14.3	0
C-3	0	100	0	0		100	0	100	0
C-4	50	0	0	0		50	0	50	0
C-6	40	20	12	44		64	68	12	0
C-7	19.9	16	7.7	35.2		46.4	29.2	1	6.3
Average C	22.0	35.8	12.8	15.8		54.9	22.3	35.5	1.3

[a] Computed by OCAMA from Form 20's RCS-2-AMC-A7. Submitted for April to June, 1954.

only some of the buckets have failed. The disks are subsequently rebladed and returned to service in a different engine.

These data from field repairs show that many hot-section parts are being replaced on engines going through minor repair. For example, 35 percent of the C engines had wheel assemblies replaced (perhaps because of either bucket or disk failures), 25 percent of the B engines had buckets replaced, 55 percent of the C engines had nozzle diaphragms replaced or repaired, and 55 percent of the A engines required replacement of the combustor inner liner. Although bearing replacements were high, it will be pointed out later that, because of the absence of an accurate criterion of bearing failures, the fact that a bearing is replaced does not necessarily mean that the bearing was bad. Also, it is clear that the life of a component varies with engine design. For example, replacement of a combustor inner liner was required in 55 percent of the A engines but in only 15 percent of the C engines. Data were not available to permit association of these replacements with operating times of the engines.

CAUSES OF ENGINE OVERHAUL

The number of Disassembly Inspection Reports available for engines being overhauled for the first time in the 3-month study period are

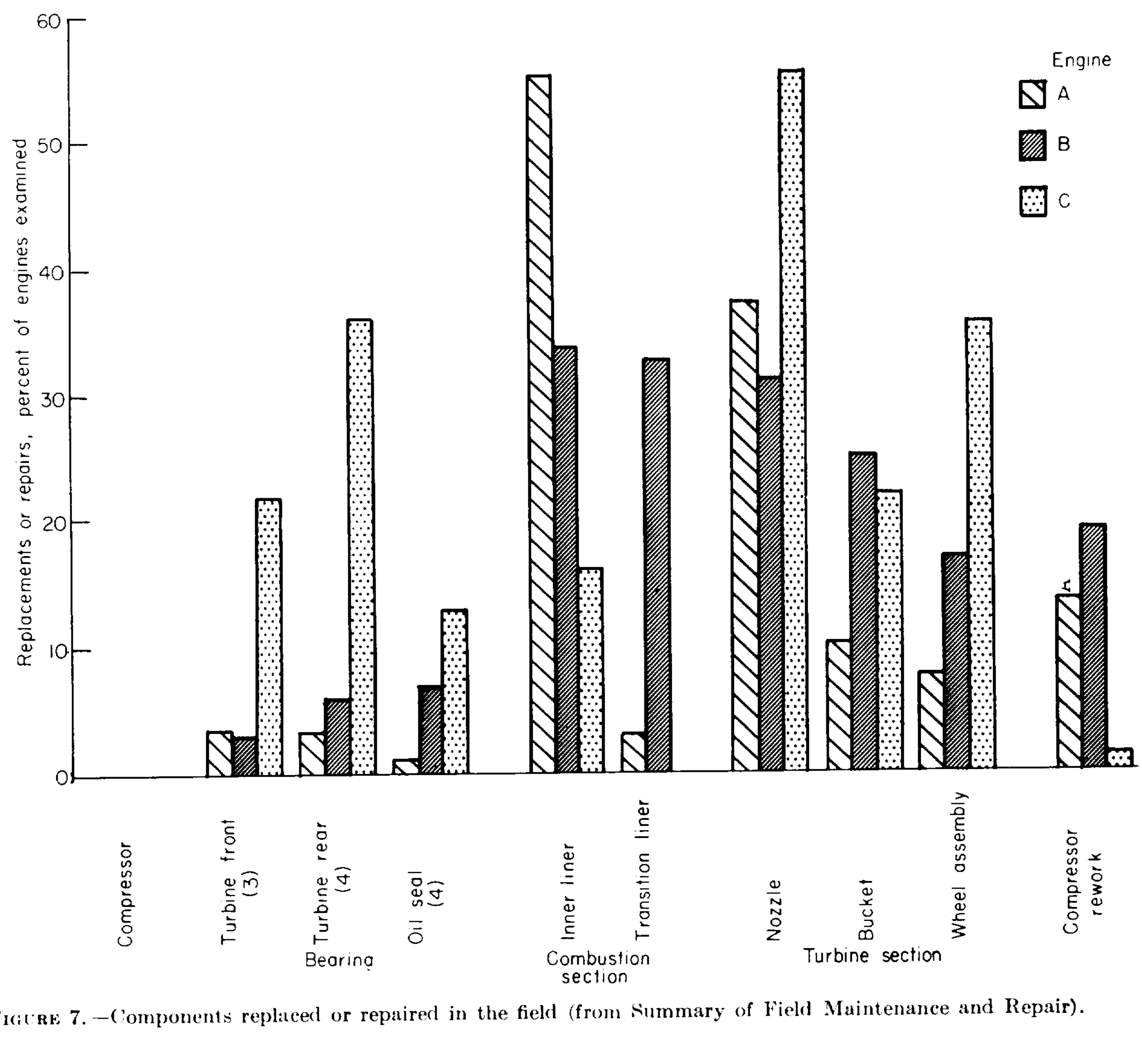


FIGURE 7.—Components replaced or repaired in the field (from Summary of Field Maintenance and Repair).

shown in table IV. The fact that the engine life varies with engine model and application (fig. 4) means that engines should not be grouped when studying causes of overhaul and part failure. Because sample sizes in some cases were so small, a decision was made to study only those engine models having more than 50 engines overhauled in the study period. In addition, only new engines (not previously overhauled) were studied to minimize the uncertainties regarding operating times on the parts. These factors limited the study to the five engine models indicated by asterisks in table IV.

This table also indicates for the five engines studied the maximum and median operating time on the engines in overhaul. The fact that the data are limited to engines having relatively short operating times must be kept in mind when drawing conclusions from the data. For example, one reason that a disk problem was not revealed by these DIR statistics is that most of the disk failure mechanisms are time- or cycle-dependent, and the engines studied have not operated sufficiently long for disk failures to be encountered. A better insight into the magnitude of disk problems can be gained from chapter VIII, TURBINE DISKS, since the authors have also reviewed some of the statistics for engines that had one or more overhauls and have thus accumulated more operating time. The B–7 engine is not reviewed herein, because it is used in an unusually severe and uncommon application.

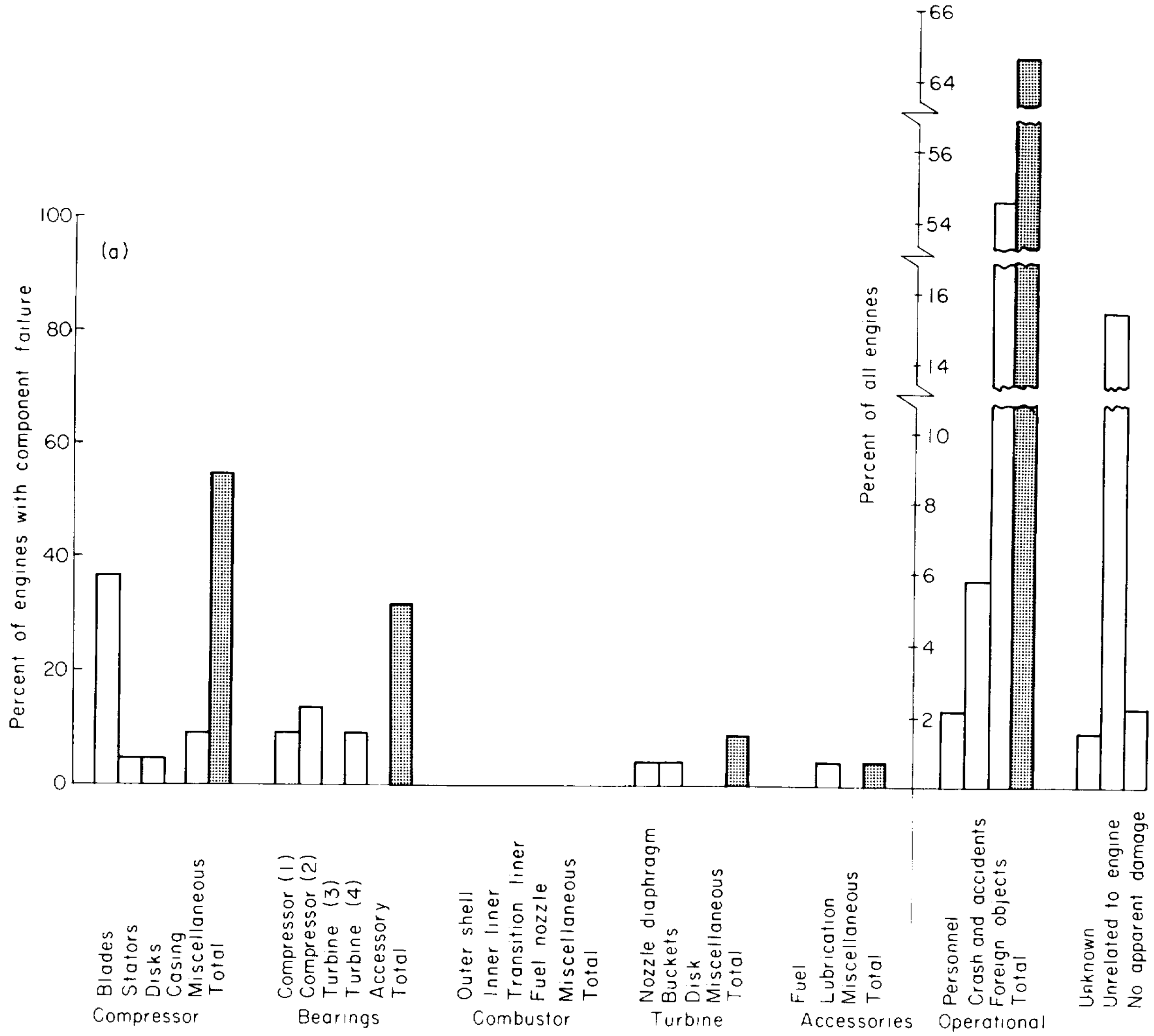
The cause for engine overhaul for five jet engines is shown in detail in figure 8. These data are from the DIR's that list, in the opinion of the inspector, the single failed part, the environment, or the other reason (e.g., personnel errors, crash, or accident) that caused the engine to come to overhaul. The data are based upon an inspection after disassembly of the engine.

In each case the height of the shaded bar indicates the percentage of engines that were overhauled because of failure in a particular section of the engine (reading that right ordinate scale); for example, 8.8 percent of the A–7 engines were in overhaul because of failure in the compressor section. The height of bars to the left of the shaded bar indicates the relative distribution of replacements among the particular parts of the compressor; for example, 6 percent of the A–7 engines were overhauled because of compressor rotor blade failure and about 0.8 percent because of stator vane failure. The left ordinate scale indicates component failures as a percent-

TABLE IV.—NUMBER OF NEW ENGINE DIR'S AVAILABLE FOR STUDY IN 3-MONTH SAMPLE (AUGUST TO OCTOBER, 1953)

Engine model	Total number	Fighter	Bomber	Not designated	Minor overhaul	Maximum operating time on these engines in overhaul, hr	Median operating time on these engines in overhaul, hr
A–5	2	1		1			
A–6	2	2					
ᵃA–7	136	110		4	22	489	160
A–8	16	4			12		
ᵃB–3	73		68		5	634	247
B–4	33	20	2		11		
B–5	7		6		1		
B–7	88	1	82		5		
B–8	93		38		55		
ᵃB–9	72		51		21	285	55
ᵃB–10	161	99			62	301	88
C–4	1	8		1			
C–5	8	8					
C–6	40	39		1			
ᵃC–7	210	209			1	885	305

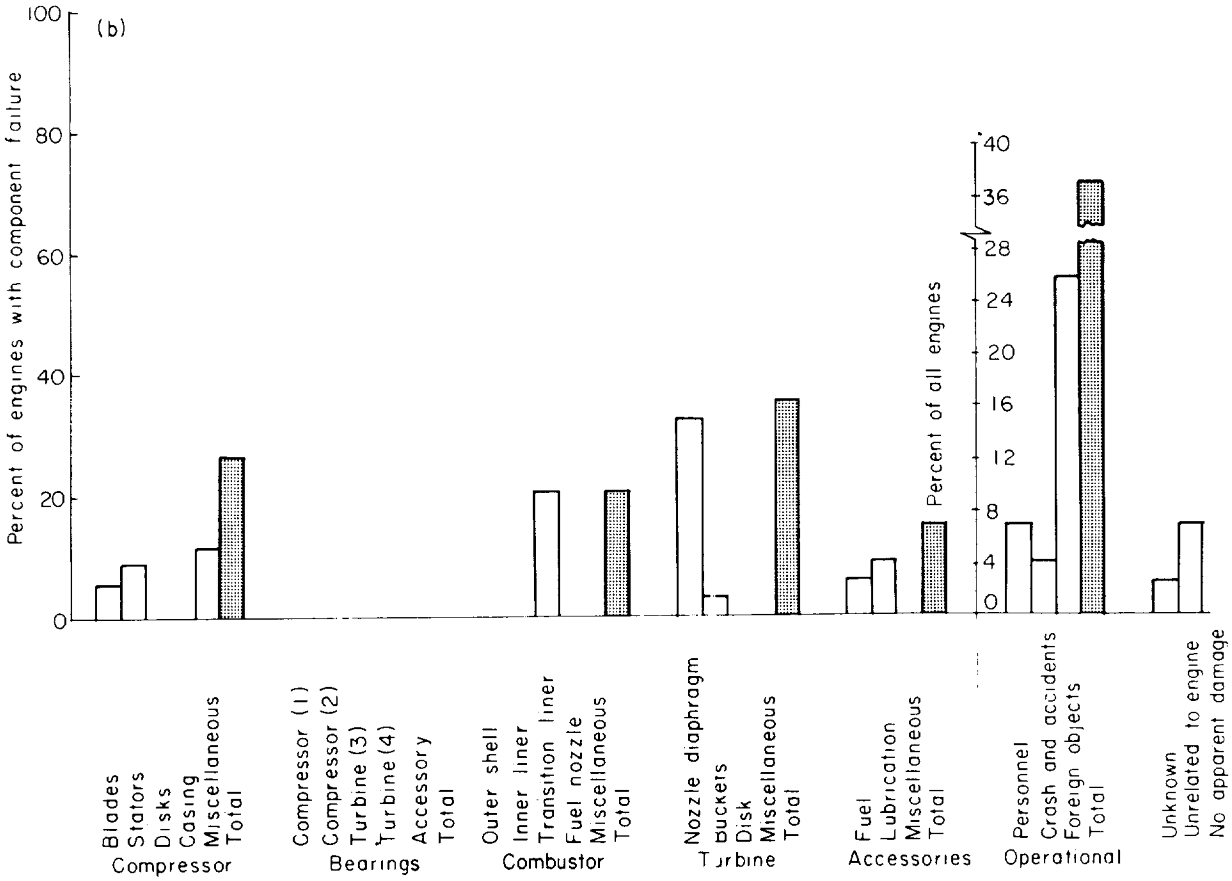
ᵃ These samples were used for statistical study.



(a) A-7 engine in aircraft F-2. Median time on engine in overhaul, 160 hours; number of engines in sample, 136.

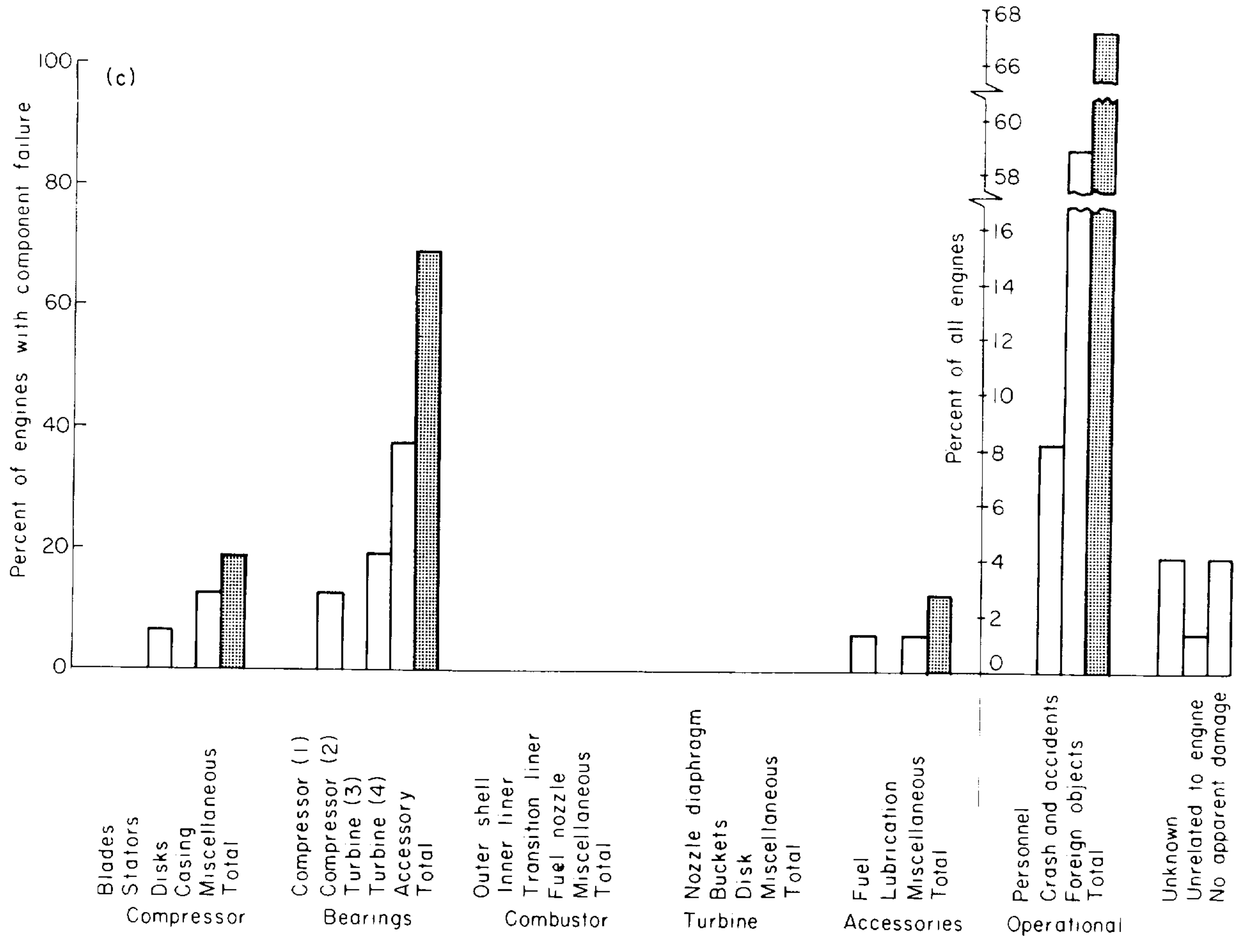
FIGURE 8.—Causes for engine overhaul (from DIR's).

age of engines for which component failures are the cause for overhaul; for example, failures in the compressor section of the A-7 engine represented 55 percent of the engine component causes of overhaul but only about 8.8 percent of all causes. If a part failed because of foreign-object damage, the cause of overhaul was noted as foreign-object damage and not charged to the particular engine part. The causes of overhaul are summarized by section of the engine (e.g., bearings, combustor assembly, etc.) for all five models in figure 9. The most frequent reason for engine overhaul was foreign-object damage (varying between 26 and 59 percent), the minimum being higher than the maximum for any other cause.



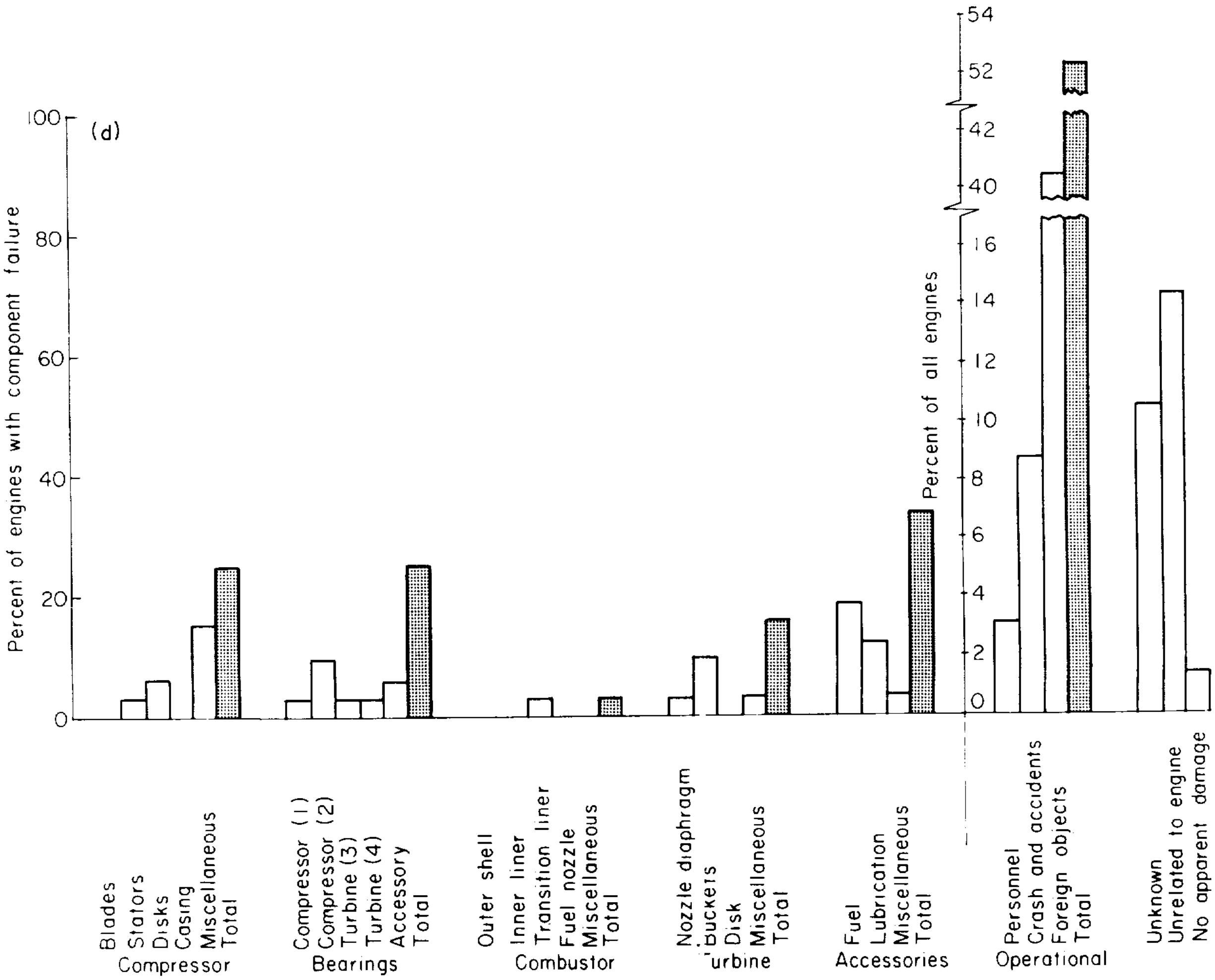
(b) B-3 engine in aircraft B-4. Median time on engine in overhaul, 247 hours; number of engines in sample, 73.

FIGURE 8.—Continued. Causes for engine overhaul (from DIR's).



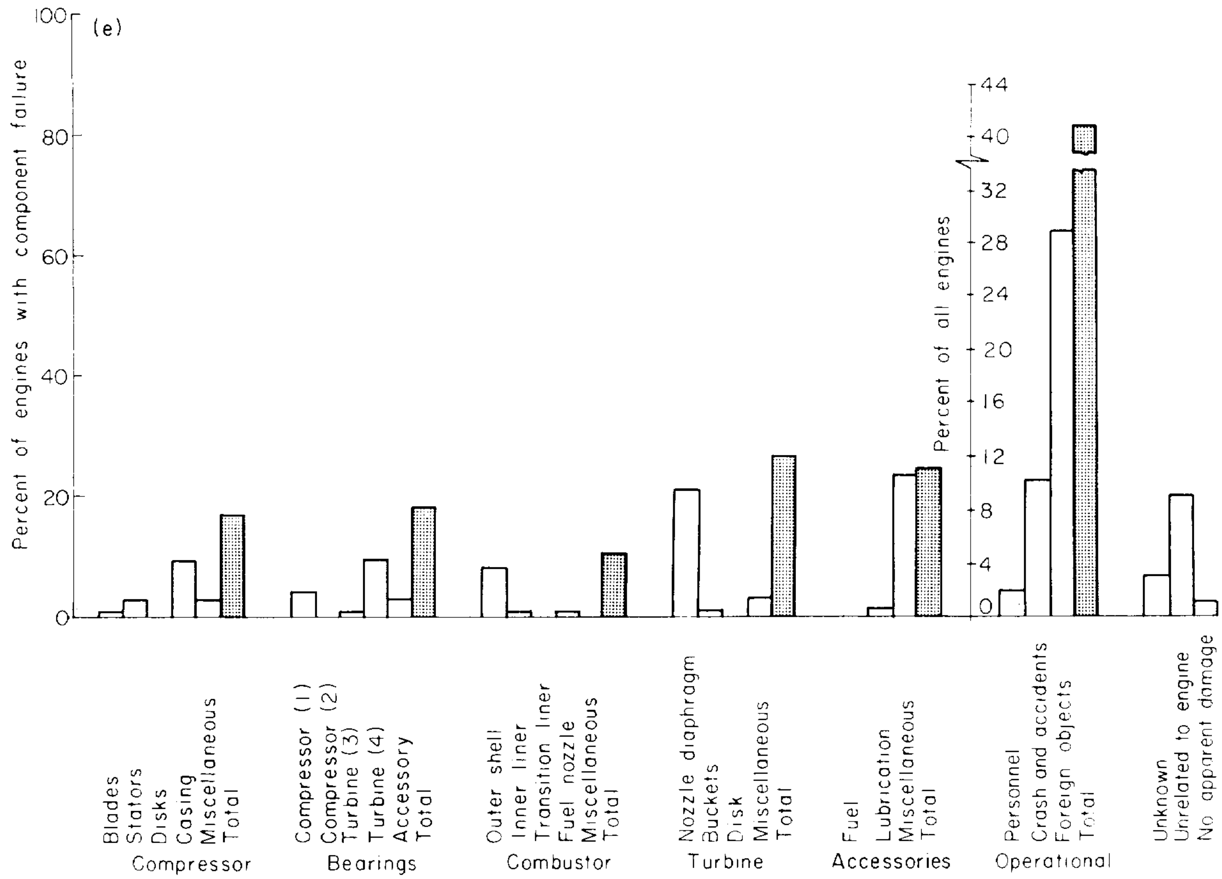
(c) B-9 engine in aircraft B-4. Median time on engine in overhaul, 55 hours; number of engines in sample, 72.

FIGURE 8.—Continued. Causes for engine overhaul (from DIR's).



(d) B-10 engine in aircraft F-3. Median time on engine in overhaul, 88 hours; number of engines in sample, 161.

FIGURE 8.—Continued. Causes for engine overhaul (from DIR's).



(e) C–7 engine in aircraft F–1. Median time on engine in overhaul, 305 hours; number of engines in sample, 210.

FIGURE 8.—Concluded. Causes for engine overhaul (from DIR's).

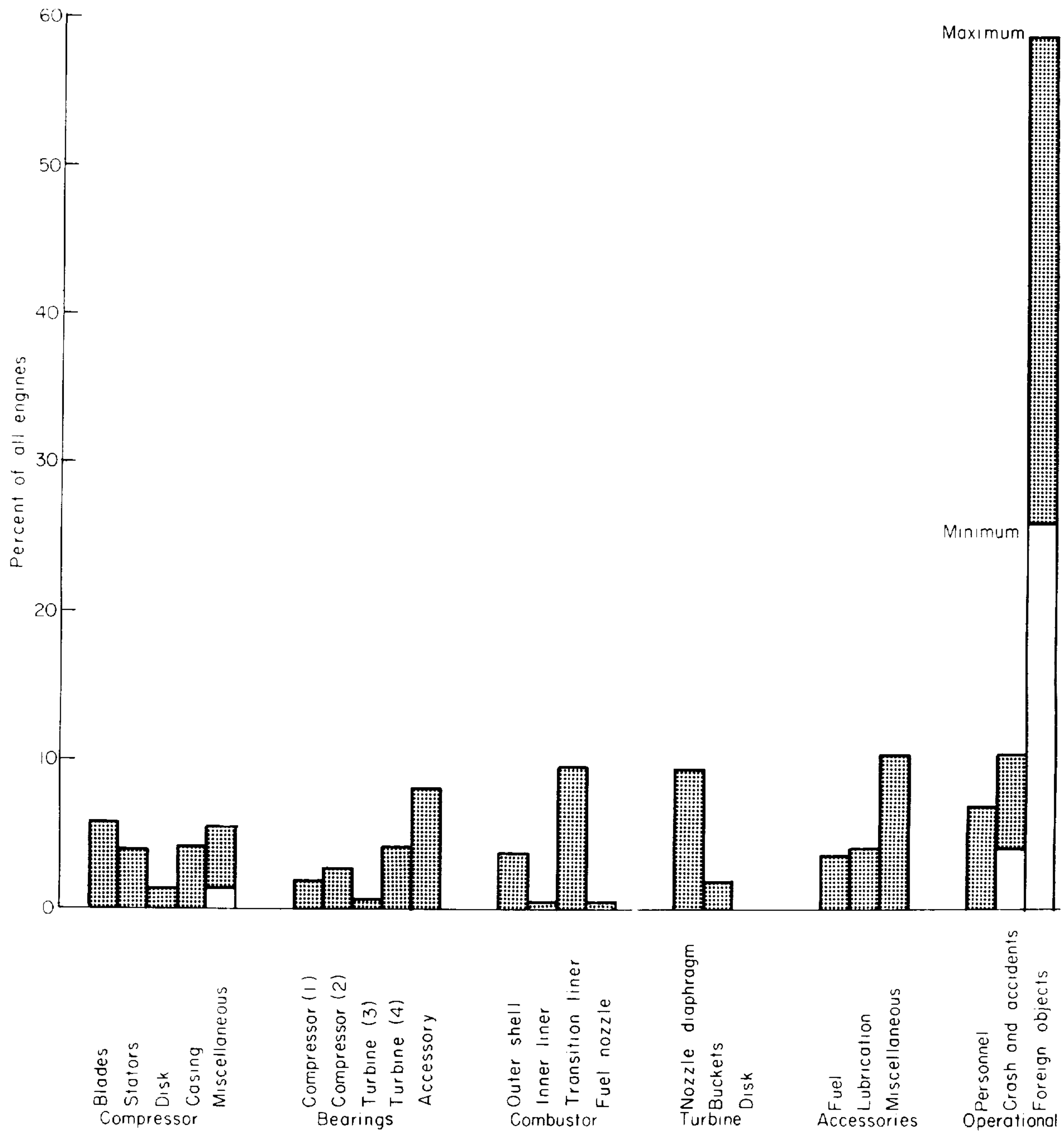


FIGURE 9.—Causes for engine overhaul for five engine models (from DIR's).

Table V indicates that, in all cases where engines were overhauled because of foreign-object damage, the damage causing overhaul was noted in the compressor. The relative importance of an engine part in causing overhaul varied with engine design or application (fig. 8 and table V); for example, the turbine section in the A engine was an infrequent reason for overhaul, but in the B–3 the turbine section is second only to foreign objects.

As the field minor repair program continues to expand, part failures in the engine hot sections, such as the combustor assembly and the turbine section, will be a reason for sending an engine to major overhaul even less often, since repairs of hot-section parts will be made in the field.

PARTS REPLACED AT OVERHAUL

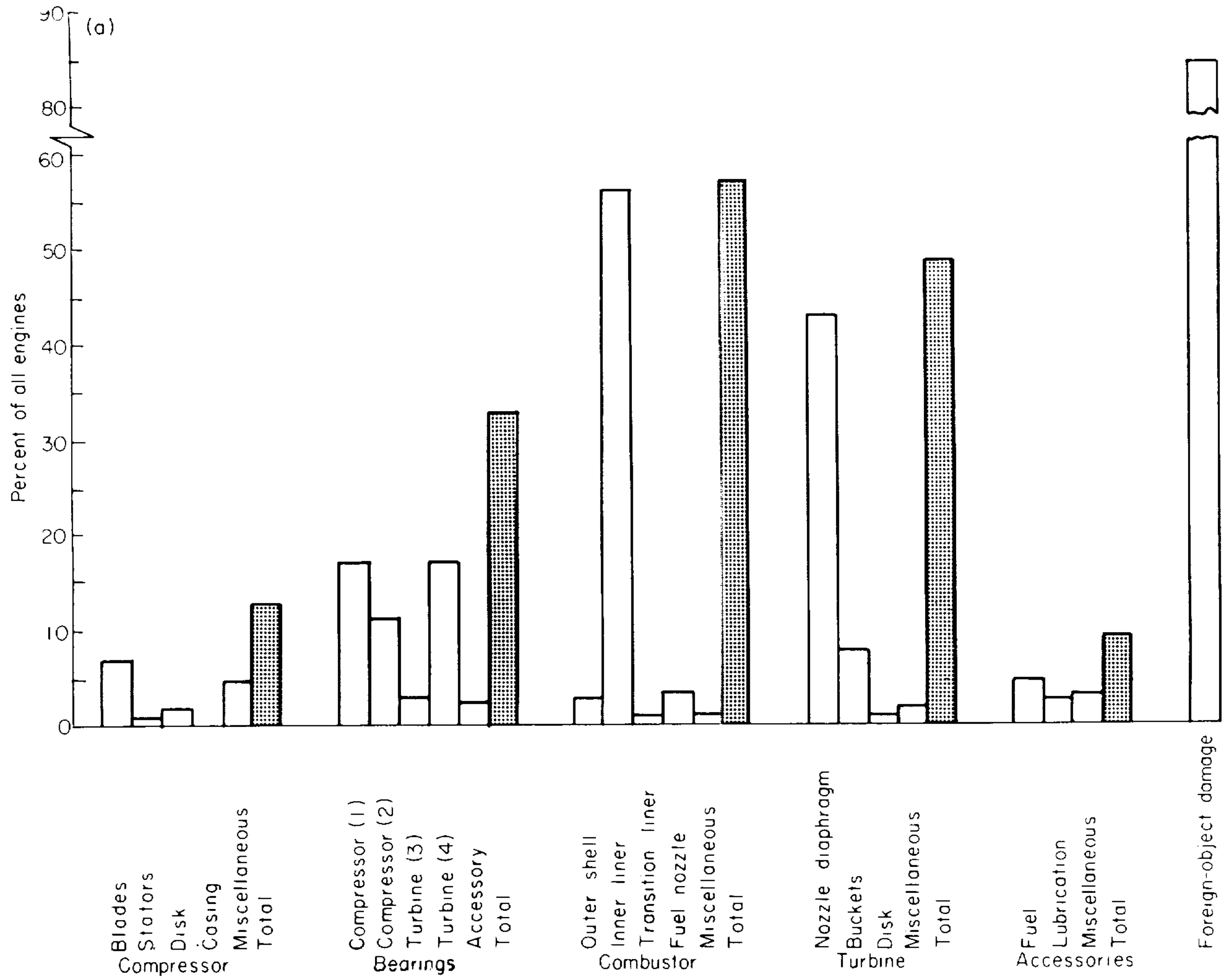
All parts replaced in overhaul on five models of jet engines undergoing their first overhaul during a 3-month period are summarized in this section, again based on data from DIR's. Figure 10 and table VI give a detailed breakdown on part replacements. The data are summarized by section of the engine (e.g., bearings, combustor assembly) in figure 11. All replacements noted by component in figures 10 and 11 are for reasons other than foreign-object damage. Very large percentages of some parts were replaced in overhaul. For example, from 25 to 60 percent of the engines had one or more of the main bearings replaced, and 87 percent of the C–7 engines required part replacements in the turbine section. Part replacements because of foreign-object damage were very high in all engines.

Table VI includes a breakdown of parts replaced or repaired because of foreign-object damage. It is apparent that both stators and rotating blades and buckets suffer damage from foreign objects. The C–7 engine, the only centrifugal-flow engine among the five studied, had a lower frequency of compressor damage than turbine damage, whereas the axial-flow engines suffered more compressor than turbine damage.

TABLE V.—REASONS FOR OVERHAUL FOR GROUP OF ENGINES OVERHAULED IN THE PERIOD AUGUST TO OCTOBER, 1953

Failures associated with—	Percent of engines overhauled				
	C–7	A–7	B–3	B–9	B–10
Compressor [a]	8	9	12	4	5
Blades	.5	5.9	2.7	0	0
Stators	1.4	.7	4.1	0	.6
Disks	0	.7	—	1.4	1.2
Casing	4.3	—	—	0	0
Miscellaneous	1.4	1.5	5.5	2.7	3.1
Main bearings	7	5	0	7	4
Accessory bearings	1	0	0	8	1
Combustor assembly [a]	5	0	10	0	1
Outer shell	3.8	0	0	0	0
Inner shell	.5	0	0	0	0
Transition liner	0	0	9.6	0	.6
Fuel nozzle	.5	0	0	0	0
Miscellaneous	0	0	0	0	0
Turbine section [a]	11	1	16	0	3
Nozzle diaphragm	9.5	.7	15.1	0	.6
Buckets	.5	.7	1.4	0	1.9
Disks	0	0	0	0	0
Inner gas baffle	0	0	0	0	0
Miscellaneous	1.4	0	0	0	.6
Accessories	11	1	7	3	7
Foreign objects	30	57	26	59	42
Compressor	29.9	55.2	26	59.2	41.6
Turbine section	0	0	0	0	0
Other causes [b]	28	27	29	19	39
Mean time to overhaul, hr	305	160	247	55	88
Number in sample	210	136	73	73	161

[a] Does not include foreign-object damage.
[b] E.g., modification, crash and accident, unknown, changed in handling.



(a) A-7 engine in aircraft F-2. Median time on engine in overhaul, 160 hours; number of engines in sample, 136.

FIGURE 10. Components replaced or repaired during engine overhaul (from DIR's).

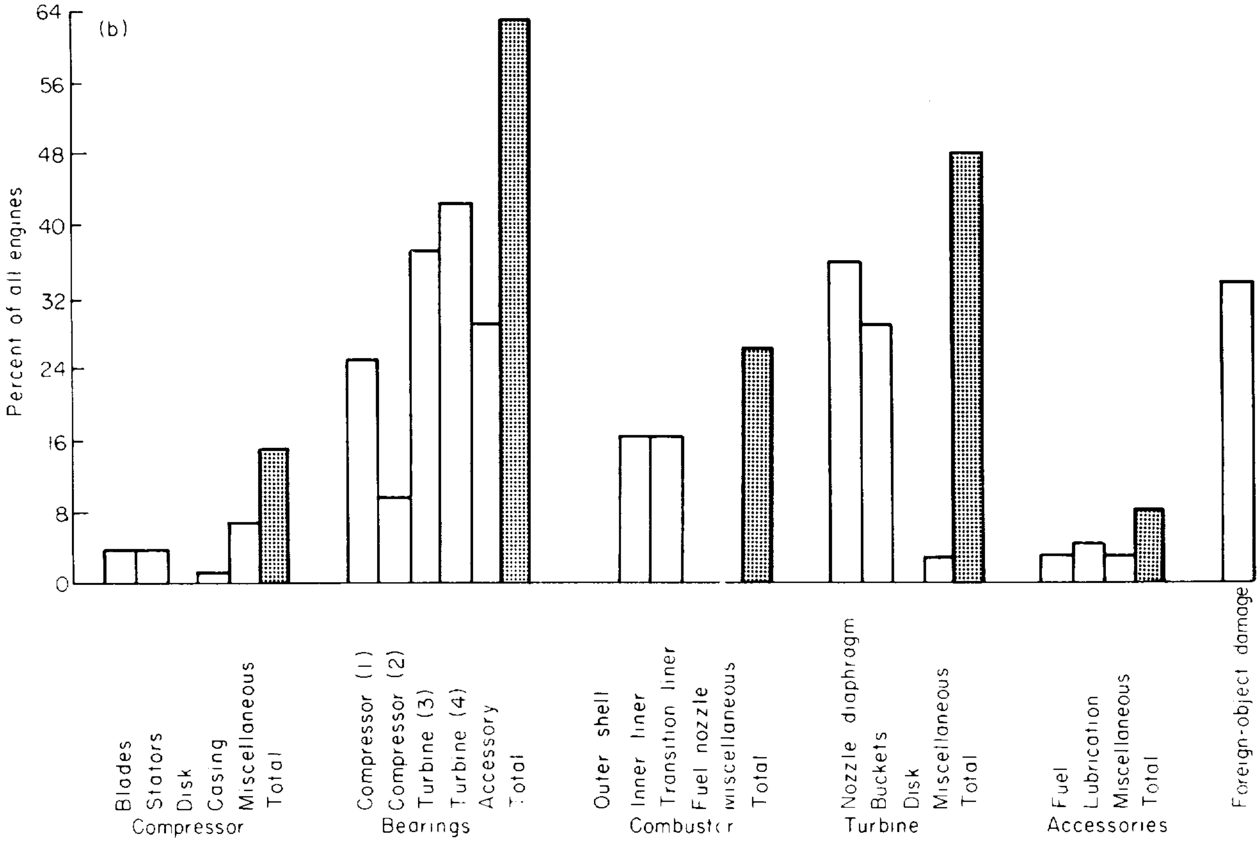
TABLE VI.—PARTS REPLACED OR REPAIRED DURING OVERHAUL (AUGUST TO OCTOBER, 1953)

Parts of engine replaced	Engines requiring specified part replacement or repair, percent of engines overhauled				
	C–7	A–7	B–3	B–9	B–10
Compressor [a]	22	12	15	5	7
Blades	.5	6.6	4	1.4	0
Stators	9	.7	4	0	.6
Disks	0	1.5	0	1.4	1.9
Casing	8.1	0	1.4	0	1.2
Miscellaneous	6.2	4.4	6.9	2.7	3.7
Main bearings	39	31	60	29	24
Accessory bearings	64	2	29	11	2
Combustor assembly [a]	44	57	26	3	25
Outer shell	5.2	2.2	0	0	.6
Inner shell	43.3	55.9	16.4	2.7	23
Transition liner	0	.7	16.4	0	1.2
Fuel nozzle	.5	2.9	0	0	0
Miscellaneous	0	.7	0	0	.6
Turbine section [a]	87	49	48	4	14
Nozzle diaphragm	85.7	42.7	35.6	1.4	8.7
Buckets	1.4	7.4	28.8	2.7	5.6
Disks	0	.7	0	0	0
Inner gas baffle	3.8	0	0	0	0
Miscellaneous	11	1.5	2.7	0	1.2
Accessories	14	9	8	10	12
Engines with foreign-object damage, percent	84	85	33	73	66
Compressor [b]	50.3	81.1	31.5	69.7	62.7
Blades	49.3	83.4	28.8	69.7	62.1
Stators	32.3	72.1	28.8	67.6	55.9
Casings	1.9	1.4	1.4	0	1.8
Miscellaneous	24.2	14.7	2.7	0	1.8
Turbine section [b]	75.4	52.4	16.4	43.4	43.5
Nozzle diaphragm	66.8	22.4	2.7	34.2	31.1
Buckets	70.6	49.6	16.4	13.4	40.4
Mean time to overhaul, hr	305	160	247	55	88
Number in sample	210	136	73	73	161

[a] Does not include foreign-object damage. [b] Foreign-object damage.

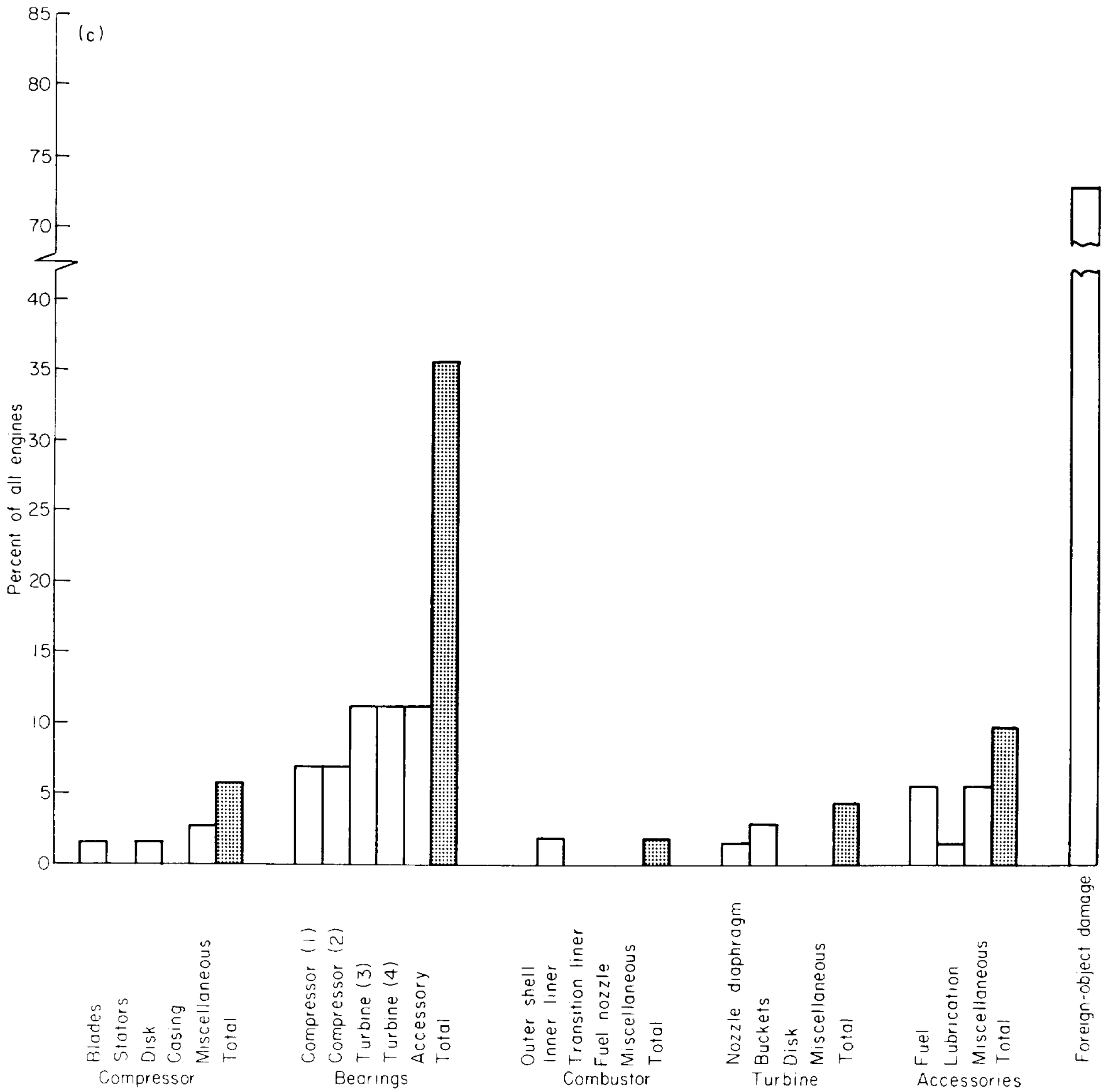
The fact that parts were replaced or repaired in overhaul does not necessarily mean, of course, that the engine would not have been operative if the part had not been replaced. For example, a turbine bucket may have been replaced because it had a cracked airfoil; if not replaced, appreciable additional operating time might be achieved in some cases before the airfoil completely fractured. Cracked parts cannot be left in, however, unless it definitely is known that the progression to fracture will be very slow and that fracture will never cause an aircraft accident. The data do indicate that the parts show at least incipient failure.

None of these engines had ever been overhauled previously; and the average engine in overhaul had short operating times since new, ranging from 55 hours for the B–9 engine to 305 hours for the C–7 engine. It might be thought that these high replacement rates are not typical for all engines in service, since the data are only from engines in overhaul. Because a majority of these engines came to overhaul because of foreign-object damage, a "chance" phenomenon (appendix A), the data do tend to represent all engines in service. From these figures and the earlier one on field replacements (fig. 7), it is seen that whenever these engines are carefully examined large numbers of parts will need replacement or repair. This is particularly true of hot-section parts and perhaps main bearings. It is not certain when a bearing is replaced that it has clear indications of damage, however. Also, many parts will need replacement because of foreign-object damage.



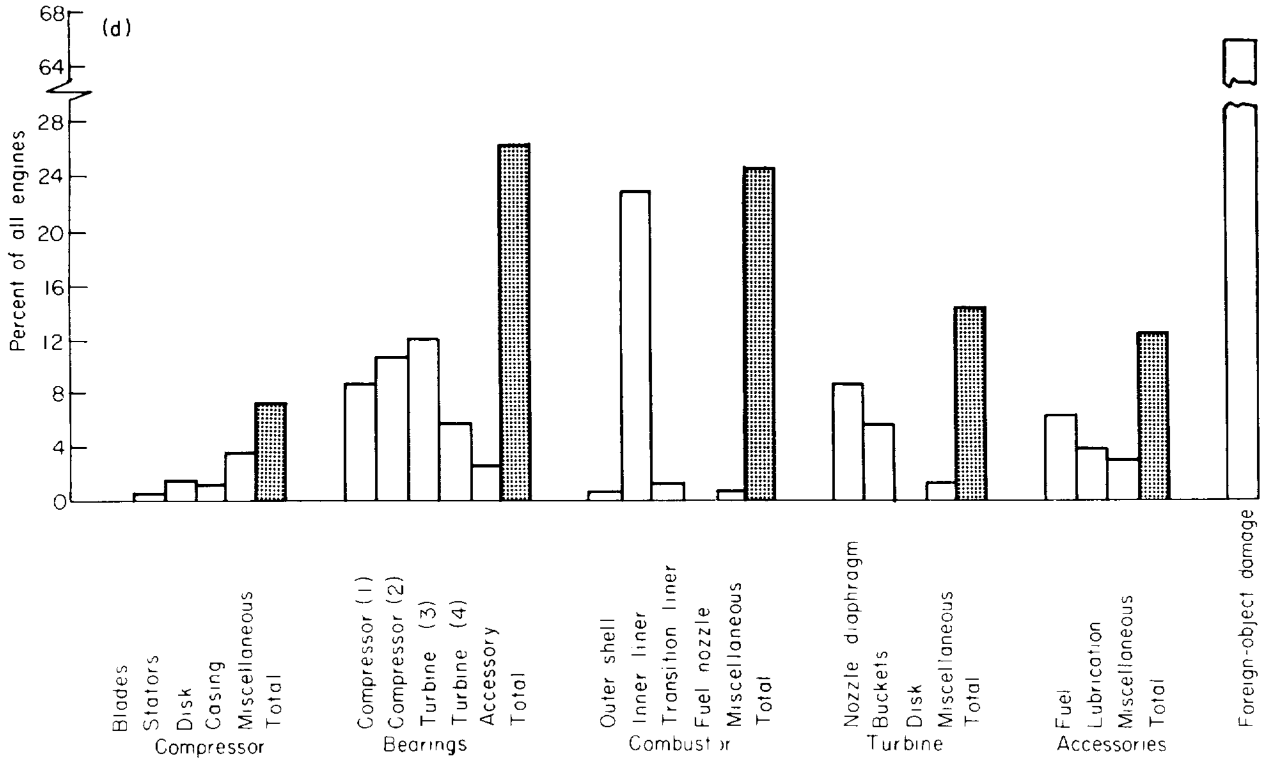
(b) B-3 engine in aircraft B-4. Median time on engine in overhaul, 247 hours; number of engines in sample, 73.

FIGURE 10. Continued. Components replaced or repaired during engine overhaul (from DIR's).



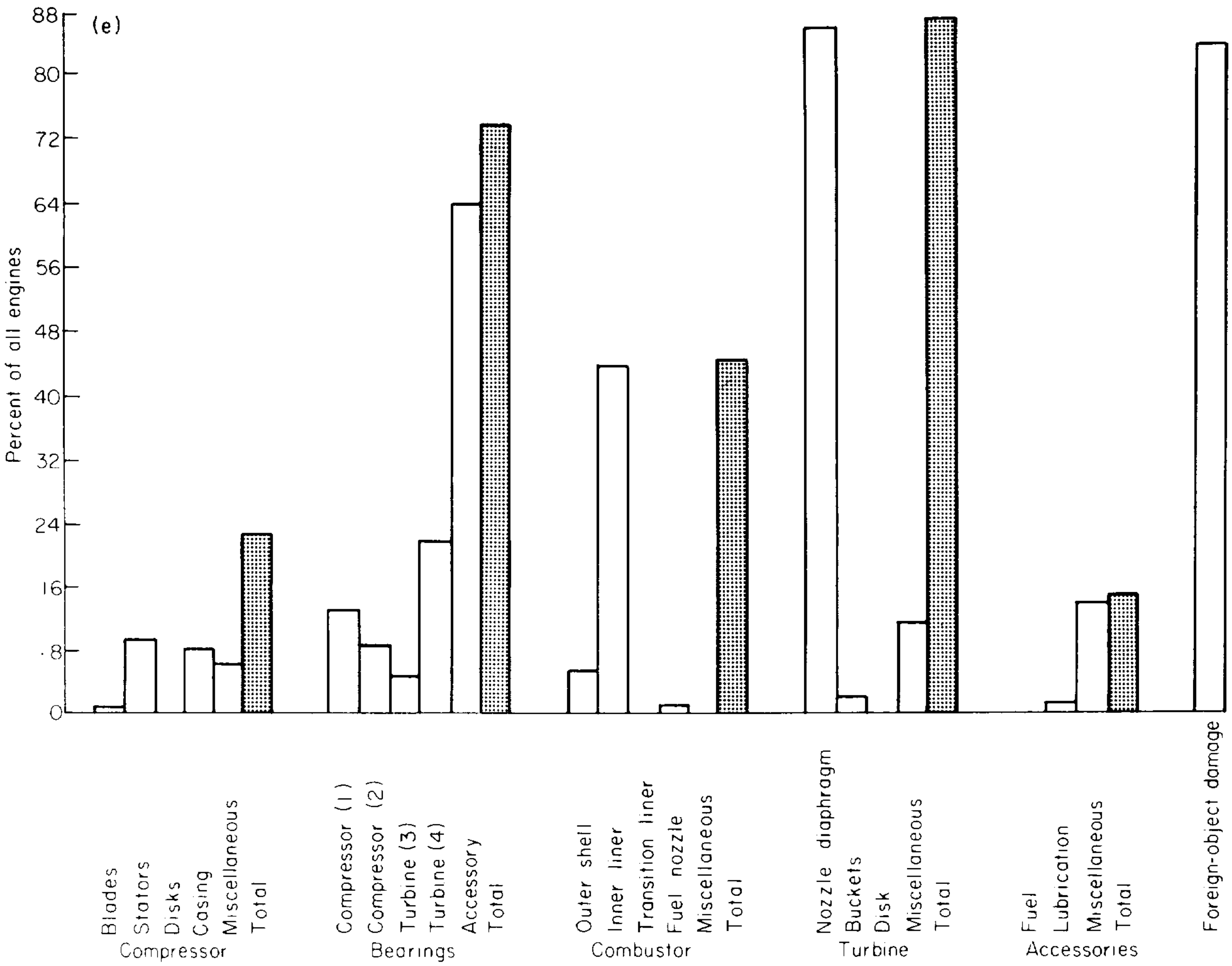
(c) B-9 engine in aircraft B-4. Median time on engine in overhaul, 55 hours; number of engines in sample, 72.

FIGURE 10. Continued. Components replaced or repaired during engine overhaul (from DIR's).



(d) B-10 engine in aircraft F-3. Median time on engine in overhaul, 88 hours; number of engines in sample, 161.

FIGURE 10.—Continued. Components replaced or repaired during engine overhaul (from DIR's).



(e) C-7 engine in aircraft F-1. Median time on engine in overhaul, 305 hours; number of engines in sample, 210.

FIGURE 10.—Concluded. Components replaced or repaired during engine overhaul (from DIR's).

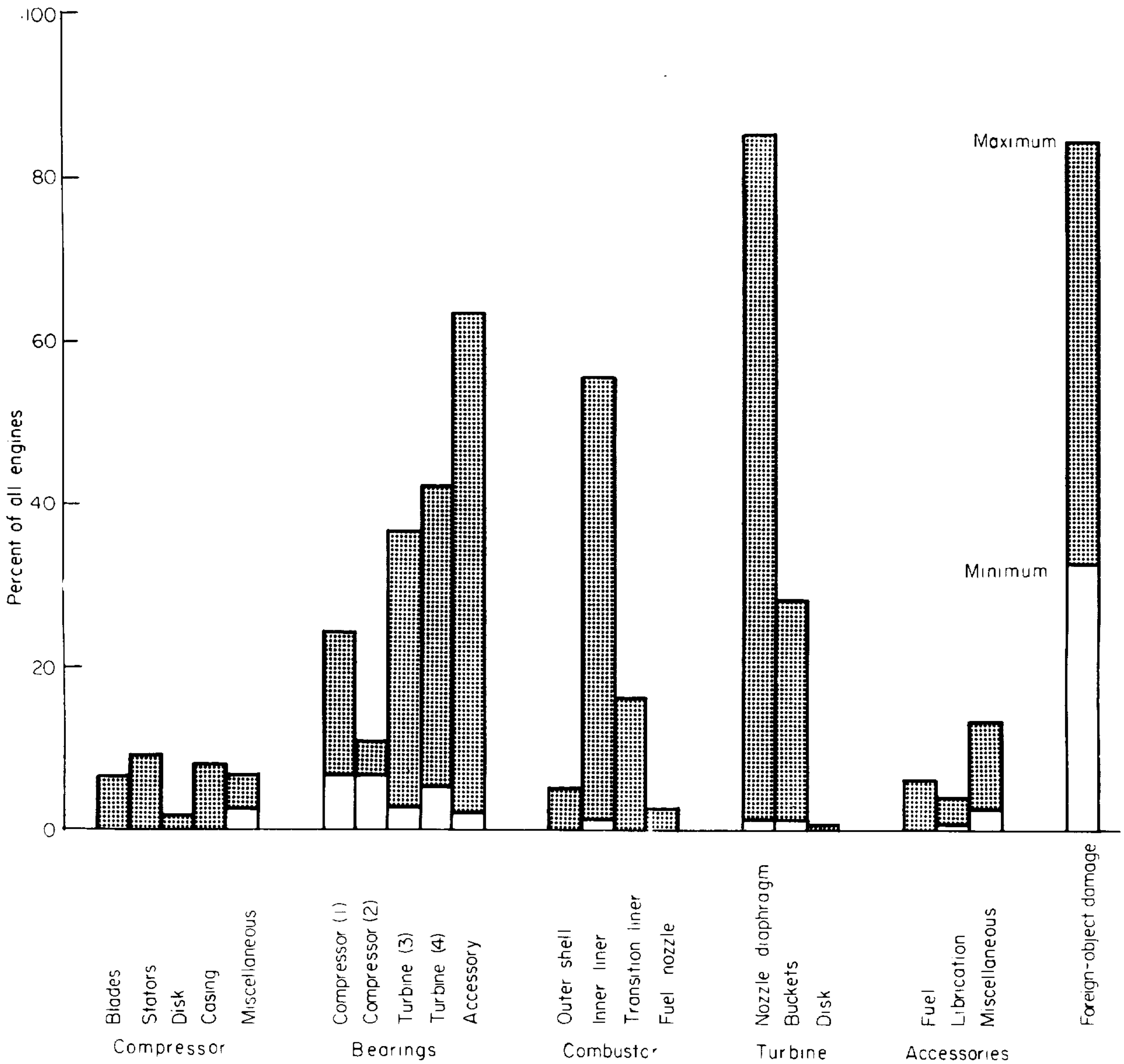


FIGURE 11.—Components replaced and repaired during major overhaul of five engine models (from DIR's).

The number of replacements of any part varies with engine design; for example, replacements of the nozzle diaphragm were made in only 8 percent of the B-10 engines but in 88 percent of the C-7 engines (fig. 10). The number of replacements also varies with the application of the engine.

TIME DEPENDENCY OF PART FAILURE

The most important data needed from service experience to assess the reliability of any engine part accurately and objectively are the rates of failure against operating time. (Rate of failure is the percentage of the parts in service that fail per unit of time.) If the failure rate against operating time is known, the seriousness of reported part failures will be known and need not be based on conjecture. For example, as stated in the section Causes of Engine Overhaul, foreign-object damage was the most frequent and consistent cause for overhaul and from 26 to 59 percent of the engines in overhaul came for that reason. This number is not of importance, however, unless it is related at least to the operating life of the engines. If the operating life were very long, say 10,000 hours, the fact that an engine came to overhaul because of foreign-object damage would not be nearly so significant as in the case of an average life of only 250 hours. The true importance would be clear if the percentage of engines requiring overhaul because of foreign-object damage per unit of flying time were known. In addition, from data of failure rate against operating time it can be determined whether the part failure follows a "chance law" (failure rate time-independent) or "wear-out law" (failure rate time-dependent). These laws are described in appendix A.

If the part failure follows a chance law, scheduled replacements will be of no help in avoiding failures. Inspections to search for incipient failures, say cracks that may lead to complete part fracture, may still be helpful in some cases, however. The failure rate can be reduced by reducing the severity of the environment (e.g., screening the engine or cleaning runways to protect against foreign objects) or by making the component better able to withstand the environment with improved materials or design.

If the part failure follows a wear-out law, then a grace period may be found during which no failures occur. Replacements can be scheduled before failures start or when the rate reaches a certain value. The failure law followed by a part should be determined from service records, because unpredictable environments might cause a part failure that was expected to follow a wear-out law to follow essentially a chance law. Also, failure rates may be higher in service than predicted by design or by test-stand operation.

The failure rate for components could be determined by introducing a known sample of new engines into service and determining the percentage of the particular part failed after the engines have operated through various time periods. Also, the failure rate of components could be determined by the actuarial method (appendix B) if the component failures were reported as related to total exposures to failure in a manner similar to that now reported by the Air Force for the engines as a whole. Since data such as these are not now available, variation of failure rates with operating time cannot be determined for engine components.

Data are available from the DIR's that give the operating times on the engines in overhaul, the part failures that caused the engine to come to overhaul, and all additional parts repaired or replaced in overhaul. An attempt was made to see whether something about time dependency (other than failure rates) could be learned from these data. It is obvious that causes for overhaul must be related in some way to the engines in service. The parts that cause the engine to come to overhaul are essentially the "bad" parts, and any study that considers only the engines in overhaul would be basing conclusions on the bad parts and neglecting the part of the sample that is still in service. It is basic that any discussion of time dependency of failure must relate those failed to the sample as a whole. This subject is discussed in more detail for the engine as a whole in appendix A. The causes for overhaul could not be related to the total engines in service, because the service information was not available.

Based on the arguments that follow, some efforts were possible to determine time dependency of failure for the parts replaced or repaired in overhaul (in addition to those causing overhaul). If a random sample of engines from service can be selected and the parts thoroughly examined for failure, some information can be obtained by determining the percentage of these examined per unit of operating time that have a particular failure and plotting these data against operating time. At least some insight may be gained as to whether service time affects part failure, and some idea of percentages of failures may be indicated. Generally, these failures found will be "incipient" failures as contrasted to failures that will make the engine inoperative. For example, if a sample of engines is chosen from service and examined for failures, the turbine buckets may show cracks, but very few would have the airfoil missing, since missing airfoils would have made the engine inoperative and such buckets would have already been repaired.

The engines examined in overhaul tend to meet the needs for a random sample, in that the majority of these engines were in overhaul because of foreign-object damage (which is a chance phenomenon) or for causes unrelated to the engine (e.g., damage in handling). The time dependency of the failure of parts replaced because of other than foreign-object damage was examined. The engines were grouped into classes of about the same number of engines, usually about 20, but in order of increasing operating time. The failure rates for the first 20, then the second 20, and so forth, were determined and plotted as histograms against operating time (fig. 12). If no failures were noted, the end of the sample interval is indicated by a short vertical line. Smooth curves are drawn through the midpoints of the tops of the bars to produce the final curves. Plots were made for all main bearings, compressor blades, combustor inner liners and transition liners, nozzle diaphragms, turbine disks, and turbine buckets.

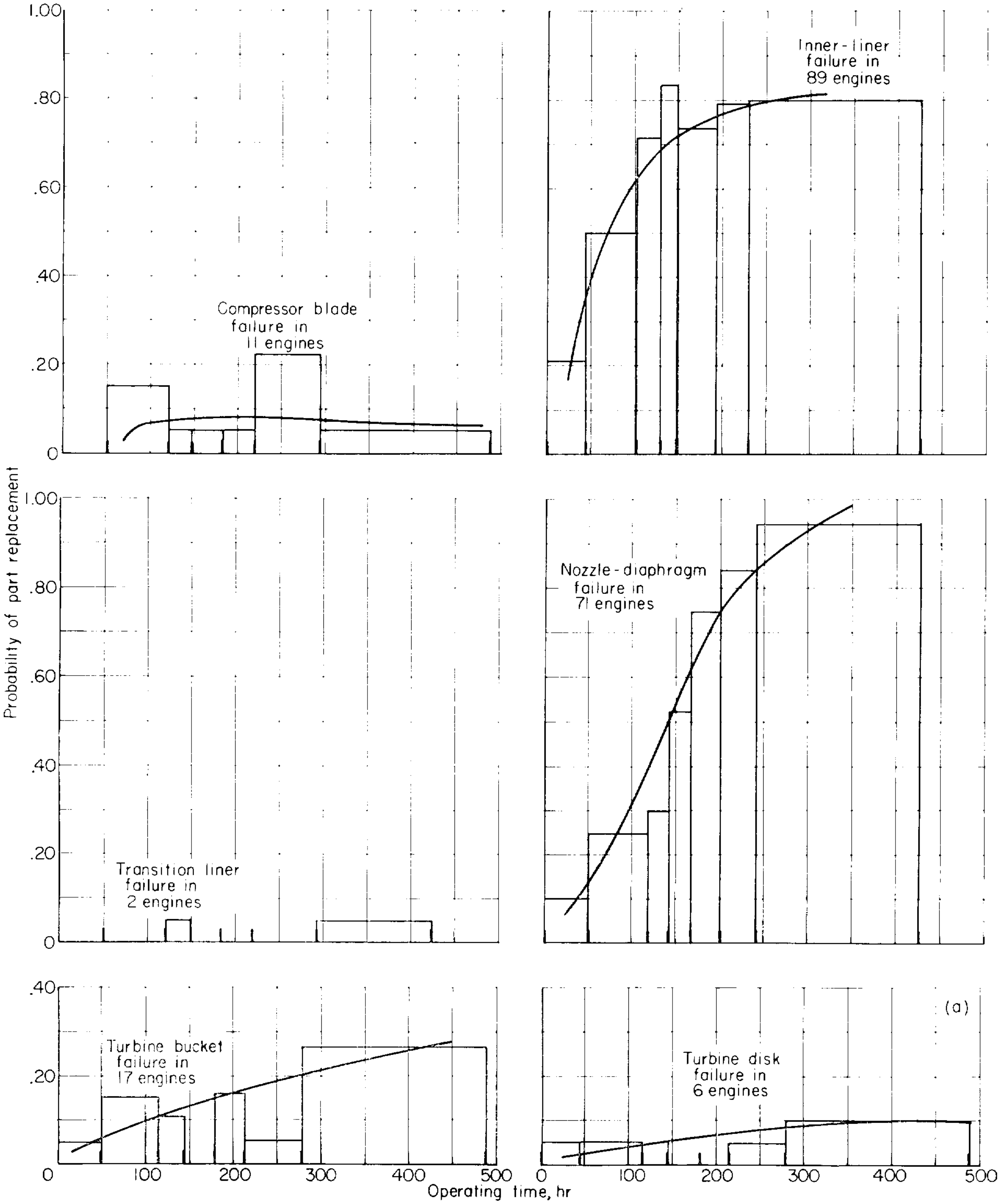
For the bearings it was quickly found that as a rule a straight horizontal line would fit the histograms quite well, suggesting that the probability of replacing a bearing in overhaul is independent of operating time on the bearing. Although these bearing data indicated that bearing *replacement* is independent of age of the bearing, bearing *failure* is not necessarily time-independent. The inspector who makes the decision to replace the bearing does not have an accurate criterion for rejecting bearings. Rejection is often based on his intuition plus the reasonable philosophy that as long as the engine is disassembled anyway, new bearings may as well be installed. Chapter IX of this report proves that many good bearings are replaced during overhaul. Since bearing replacement does not indicate bearing failure, nothing can be learned from an examination of such data, and no curves are presented.

For most other parts, however, the basis for replacement is cracking, warping, or fracture of the part, and part replacement indicates incipient failure. "Replacement probability" curves are presented for each of the other parts by engine model in figure 12. In many cases the probability of the need for part repair or replacement is quite high and increases rapidly with time. For example, figure 12(a) shows data for the A-7 engine. The curve shows that, for engines having only 50 hours of operating time, 15 percent needed nozzle diaphragm replacement or repair. The probability increased rapidly until, after 275 hours of operating time, more than 90 percent of the engines needed nozzle diaphragm repair or replacement. A similar curve is shown for inner liners. The curve starts high, and the probable necessity of inner liner replacement increases rapidly with age.

Some of these parts in service engines show a grace period. This is indicated for the turbine buckets in the B-3 engine (fig. 12(b)), where none of the engines that had run less than 138 hours needed turbine bucket replacement; then the probability started to increase. In order to achieve good reliability, all components should have a grace period so that replacements can be scheduled to avoid failures. The grace period should be very long, preferably greater than the desired time to overhaul, so that replacements do not have to be made before major overhaul. Other parts also showed a grace period. For example, no turbine disk failures were indicated out to the maximum time on these B-3 engines in overhaul, 635 hours. The sample size was very small, however.

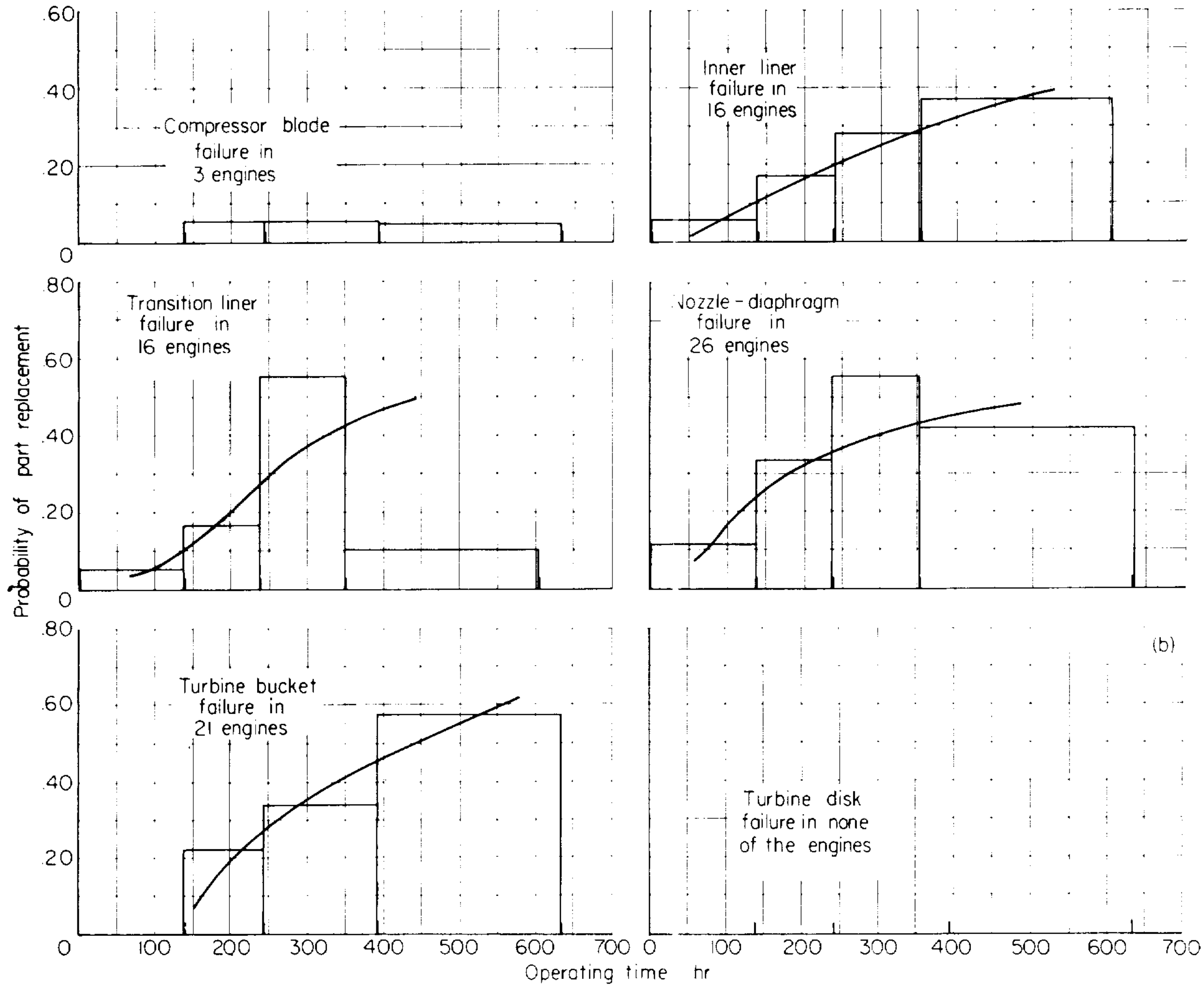
Unfortunately, many of the hot-section components exhibit failures starting near zero time. It is of interest, however, that, although high rates were found for a component in one engine, the failure rate for this component may be negligible in another engine for the operating time for which data were available.

If data plots like these are to be used, they should be based on much larger sample sizes and they must be very carefully interpreted. For example, the C-7 engine (fig. 12(e)) gives no indication of incipient bucket failure. The chapter on turbine buckets (ch. VII) points out that the nature of the failure mechanism of the buckets in this engine is such that incipient failure will not be found. The buckets progress from cracking to fracture so rapidly that the first indication of fracture is actual bucket fracture. The method of inspection must also be considered. This was discussed in connection with bearing failures. Also, no incipient failures of turbine disks were found in the B-9 engine. New inspection procedures have since been introduced that are now finding quite high percentages of incipient cracks in disks in overhaul in this engine.



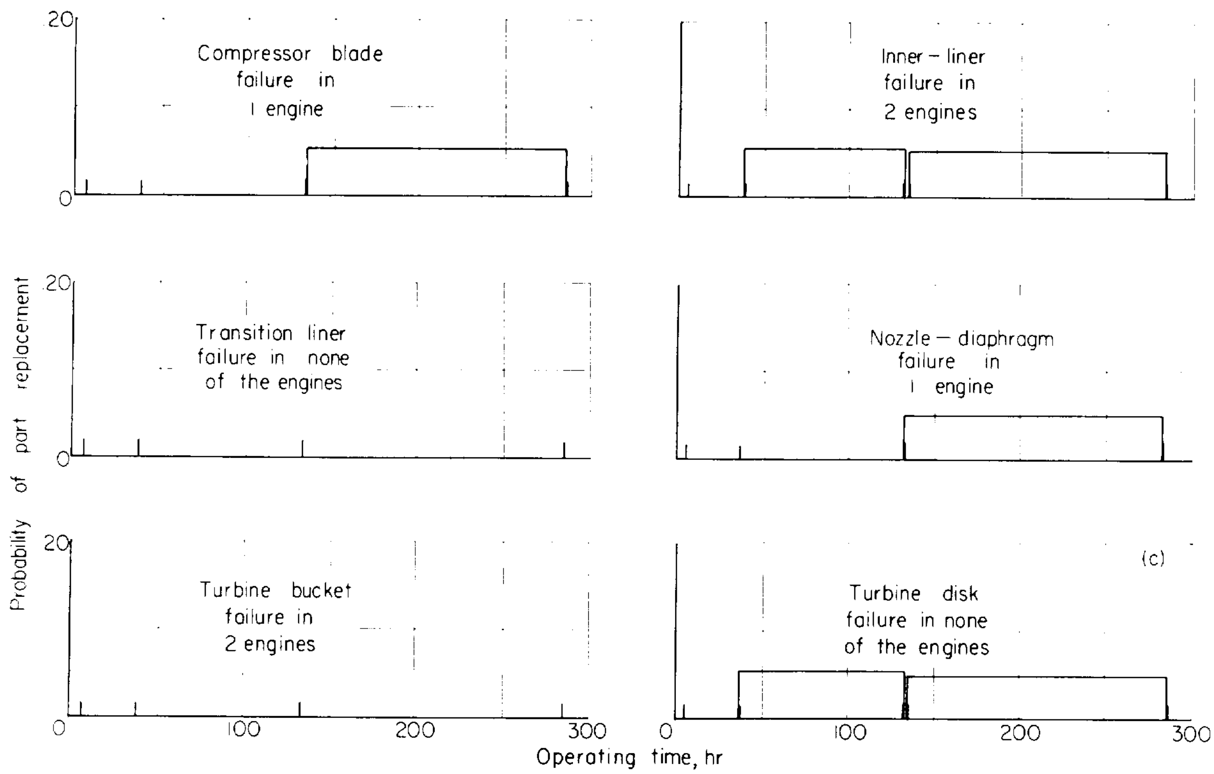
(a) Engine A–7. New engines, 136; number in interval, 19.

FIGURE 12.—Probability of engine component replacement with life for five engines (from DIR's).



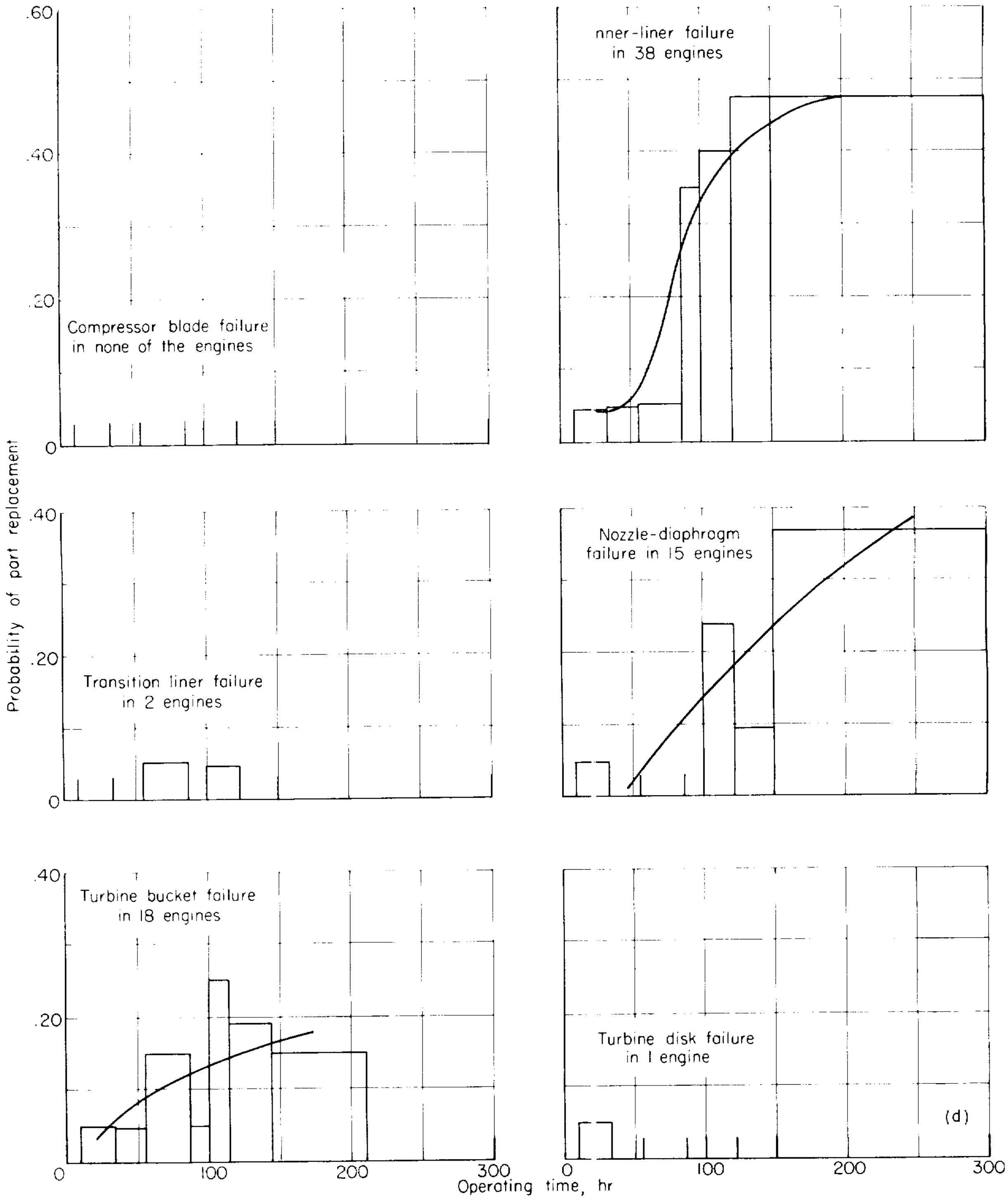
(b) Engine B-3. New engines, 73; number in interval, 18.

FIGURE 12.—Continued. Probability of engine component replacement with life for five engines (from DIR's).



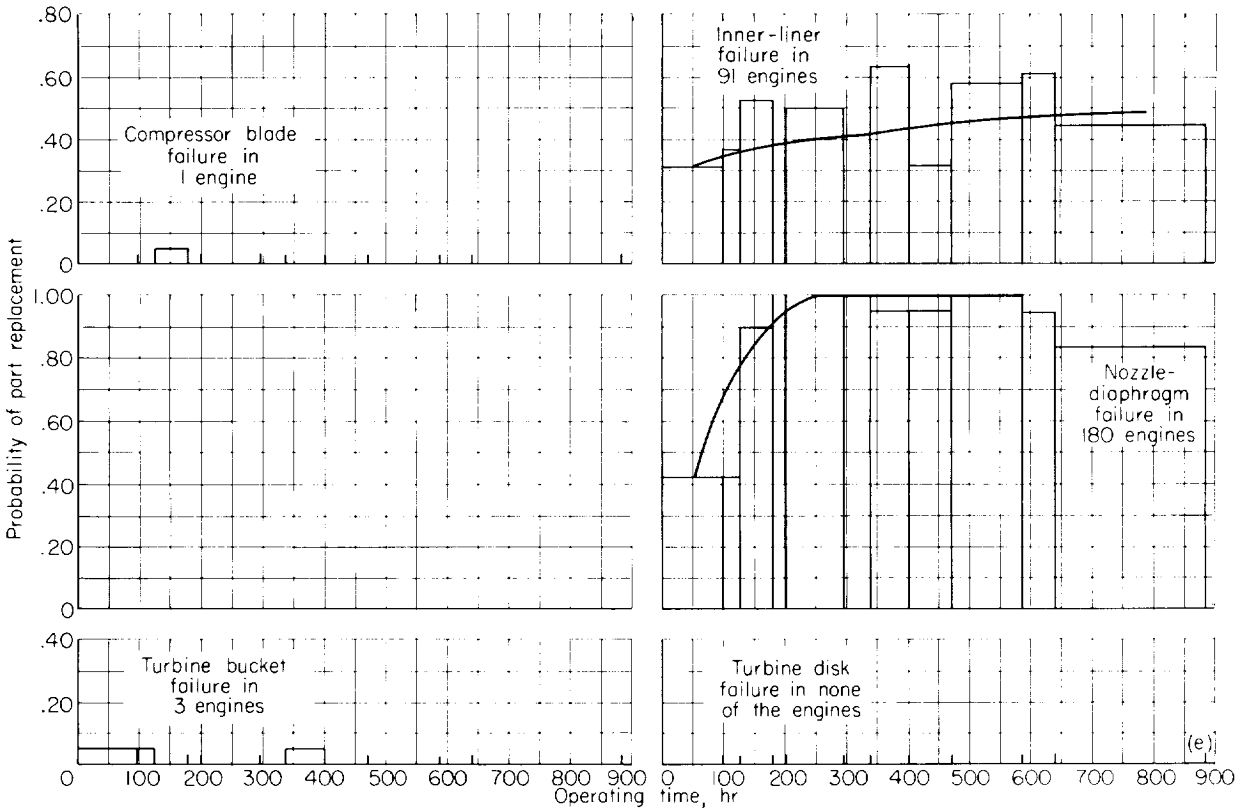
(c) Engine B–9. New engines, 72; number in interval, 18.

FIGURE 12.—Continued. Probability of engine component replacement with life for five engines (from DIR's).



(d) Engine B-10. New engines, 161; number in interval, 20.

FIGURE 12. Continued. Probability of engine component replacement with life for five engines (from DIR's).



(e) Engine C-7. New engines, 210; number in interval, 19.

FIGURE 12.—Concluded. Probability of engine component replacement with life for five engines (from DIR's).

ACCIDENT STATISTICS

A summary of the causes of engine failure or malfunction that resulted in 205 jet-powered aircraft accidents in the year 1953 is shown in figure 13 (data from ref. 4). The part failure responsible for the accident could not be determined in 11 percent of these accidents. The biggest single offender was the fuel control, which caused 33 percent of the accidents. Second was compressor failure, including that due to foreign objects; following this was turbine buckets, then turbine disks, bearings, and finally miscellaneous causes, including 20 accidents from 16 different causes. For each of the components listed in figure 13, about half the failures resulted in destruction of the airplane.

Of these 205 accidents, 173 were listed as major accidents, of which 100 resulted in total destruction of the aircraft and 73 in substantial damage to the aircraft. The rate for major accidents caused by engine failure or malfunction was 7.9 per 100,000 aircraft flying hours (ref. 5). It is of interest that the failure rate for axial-flow engines was almost three times that for centrifugal-flow engines (12.0 and 4.1, respectively, per 100,000 aircraft flying hours). Of the 54 accidents attributed to compressor failure, only one involved a centrifugal compressor, although the aircraft flying time accumulated for each engine type was the same, about 1,100,000 hours.

Of the 205 accidents listed in figure 13, 188 involved single-engine aircraft and 17 involved multiengine aircraft; however, the multiengine aircraft had much less exposure to failure (i.e., less operating time). The number of major accidents for multiengine aircraft per unit of operating time (axial-flow engines) was about the same as for all aircraft having axial-flow engines (12.2 and 12.1 per 100,000 aircraft hours, respectively). For the single-engine aircraft, 98 of the 188 accidents, or 52 percent, resulted in

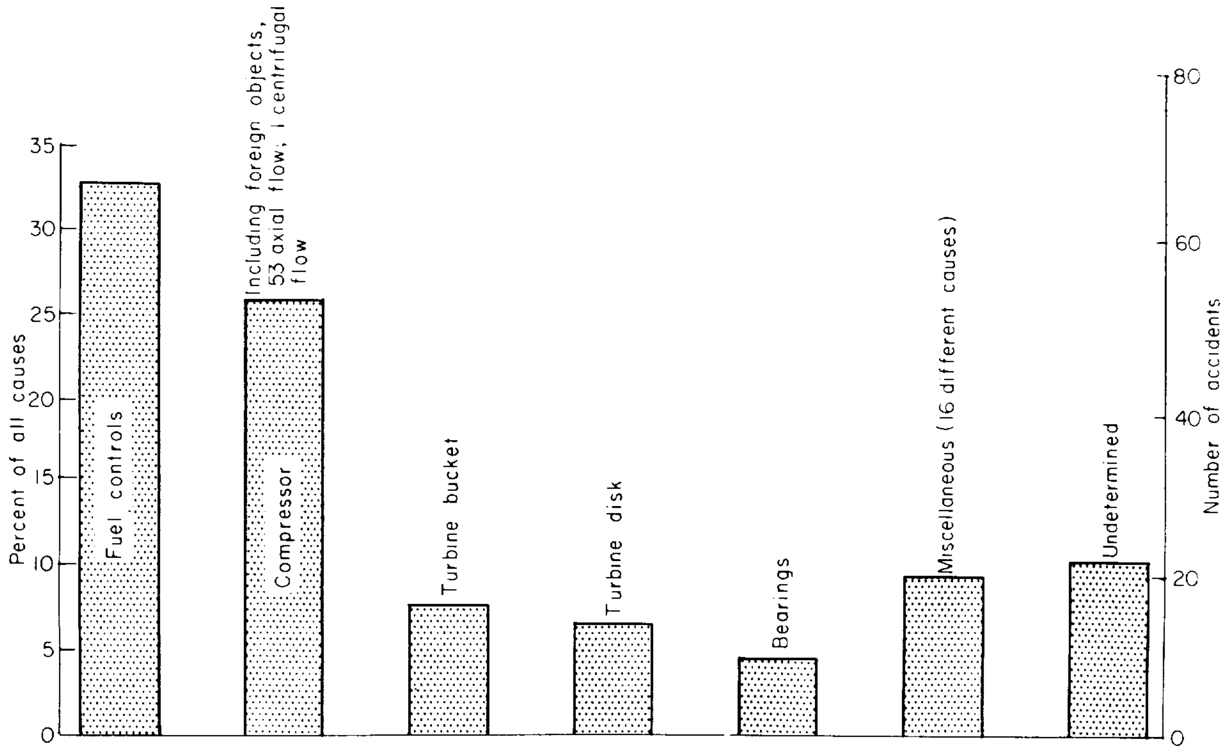


FIGURE 13. Distribution of causes of engine failure or malfunction that resulted in 205 Air Force jet aircraft accidents during 1953 (ref. 4). (Aircraft destroyed in about half the accidents resulting from each cause.)

destruction of the aircraft; whereas, for multi-engine aircraft only 2 of the 17 accidents, or 18 percent, resulted in destruction of the aircraft.

Although only limited flying time (about 105,000 hr) was available in 1953 for aircraft having pod-mounted engines, the major accident rate was about one-third of the average, or 3.8 per 100,000 hours (four major accidents). Only one of these four major accidents resulted in destruction of the aircraft, and in that case the fragments of the turbine wheel went through the fuselage, ruptured the refueling manifold, and set fire to the aircraft. In two of the four accidents, the engines that failed tore loose from the aircraft.

SERVICE RECORDS DESIRED

As pointed out earlier in the section on TIME DEPENDENCY OF PART FAILURE, the most important data needed from service records are the rates of failure of the engine components against operating time. With these data the problem areas would be accurately illuminated and the importance of the problem would be quantitatively determined. If the correct information were provided, the service data would provide a basis for the following:

(1) Improved design, both as a basis for quick fixing of urgent problems and as a basis for building up long-range design criteria so that future designs may be improved

(2) Improved operating conditions: improved personnel procedures, improved engine controls, and runway cleaning in the case of foreign-object damage

(3) Provision of safeguards such as screens for protection against foreign objects and warning devices to warn of impending part failure

(4) Scheduling of replacements and inspections to reduce the probability of flight accidents

From a review of the current failure data it is apparent that great economies would be gained from rapid inflow of quantitative failure rate statistics that indicate accurately where quick fixing is needed. Besides the savings obtained by reducing the probability of future aircraft accidents, the logistics problem and the number of spare engines might be reduced. Frequently, problems continue for a long period of time before they are recognized and studies for a solution are initiated.

The desired records involve two main features: First, an accurate operating history of engines in service is required, and second, all part failures for the engines in service must be known. These data must be reported in such a way that the two can be put together to determine a failure rate. In addition to the identification of the failed part, information is needed as to the nature of the failure and the location of the failure on the part (see item II. C. that follows).

Both the Air Force (ref. 1) and the Navy (ref. 6) have initiated new programs of record collecting as a basis for product improvement. The programs emphasize the collection of part failure data similar in part to that described in section II that follows.

Although the magnitude of the required data may seem large, that suggested in section II is not inconsistent in magnitude with that already initiated by the Air Force and Navy. In addition, the Air Force is collecting data on engine life for the actuarial method that at least approximates that suggested in section I. The magnitude of the program could be reduced by collecting data only for engines that will be used in appreciable numbers in the future, not data from retiring engine models.

The following records appear desirable:

I. ENGINE HISTORY—An engine log should be maintained that will stay with the engine.

A. Initial information recorded on this log should include:

1. Serial number of engine
2. Manufacturer and date of manufacture
3. Modifications incorporated from date of manufacture to date of installation
4. Date of placing in service
5. Model of aircraft and position in which engine is installed

B. The following history during usage should be recorded:

1. Operating time and date of modifications
2. All part replacements or repairs, including operating time, and date; where several of a particular part are used (e.g., turbine buckets), a log of each part and a method of identifying its location in the engine must be kept so that histories are not mixed
3. History of replacement parts installed (whether new or used)
4. Engine operating time when minor repairs and overhauls are performed and description of work performed
5. Operating history including:

a. Time at maximum rated conditions
b. Number of accelerations and decelerations
c. Number of starts
d. Duration and severity of each overspeed, overtemperature and associated engine speed, and hot start
e. Air base of using activity

At regular intervals (e.g., every 3 months), data for each engine by serial number should be sent to a central data agency. These data should describe the total operating time on the engine and the operating time at which overhauls and repairs were performed and summarize the history of operating conditions.

In view of the extensive minor repair program for jet engines, considerable doubt is raised as to the usefulness of data describing time to overhaul for these engines. Since the entire hot section (combustor, turbine section, turbine bearings, and tailcone) and external accessories can be replaced in the field, engines are now sent to overhaul primarily because of problems of the compressor and accessory drive section of the engine. It is understood that changes of even these parts in the field may also be permitted at a later date. Therefore, the average time at which an engine is sent to overhaul will tell only a very small and difficult to interpret part of the engine failure story. The specific data that are needed are the rates of part failure against operating time as developed by sections I and II.

II. PART REPLACEMENTS AND REPAIR—Whenever a part is repaired or replaced whether in the field or during overhaul a record should be sent to a central data agency indicating:

A. The part and whether it was replaced or repaired and reinstalled

B. The operating time on the part

C. Failure of the part (e.g., whether cracked, fractured, nicked, dented, or distorted) and location on the part where failure occurred; (the man in the field cannot be expected to interpret the cause of failure, whether fatigue, stress-rupture, etc., but with a report of the location and general characteristics of the failure, a pattern will develop enabling technical personnel to go into the field to study and interpret a particular type of failure)

D. Model and serial number of engine and aircraft and base of activity

E. Circumstances that resulted in finding the failure (e.g., flight inspection, inspection during overhaul)

F. Effect of part failure on engine operation and associated engine parts

G. Disposition of part replaced, whether condemned or repaired

Data described in sections I and II are believed to represent all the data required for an understanding of reliability of an engine model. The central agency can enter part failures by engine serial number into a card-type tabulating system. Having a record of every part failure or repair and of the operating time on every engine in service as described in section I, the central agency can determine the failure rate against operating time of every important engine component. This would simply be a matter of determining how many engines had completed a time period of flight, say from 0 to 25 hours, and what percent had a particular part failure in this time period.

Although the difficulty in compiling operating data as under I.A. 5. is appreciated, particularly where large numbers of engines are involved, the correlation of component life against operating conditions would be of great value in aiding in the definition of the importance of service operating conditions and in providing a basis for scheduling replacements. (Additional comments are made on this subject in the other chapters of this report, particularly in ch. VII.)

As mentioned previously, the main features of recording of part replacements as described in section II have already been adopted by the Navy (ref. 6) as well as the Air Force (ref. 1). In addition, DIR's will be written for all minor repairs as are now written for major overhauls. The location of the failure on the part (II. C.) is believed to deserve consideration for inclusion into the systems of references 1 and 6. Addition of section I would provide a basis for determining the important data on component failure rates.

SUMMARY OF RESULTS

Service records of turbojet engines in Air Force military service were sampled and studied. Although the records were not designed for this purpose, considerable insight was gained. Suggestions were presented as to data required for any improved future study similar to that conducted herein. Among the more important findings from the available data were the following:

1. The time to overhaul of the average turbojet engines of three models for which considerable experience has been obtained varied with engine model and application from 105 to 760 hours. Most of the engines were near the low side of this band. In addition to major overhauls, a very extensive program of field minor repairs is in use for the jet engines. The engines are given 0.5 to 17 minor repairs for every major overhaul. A minor repair can consist of replacement of all hot-section components.

2. The most frequent and consistent cause of overhaul was foreign-object damage, from 26 to 59 percent of the engines being overhauled for this reason. The relative frequency of other causes for overhaul varied appreciably with engine design and application.

3. Failures in the hot section of the engine, that is, the combustor and turbine sections, are a frequent cause for engine removal from aircraft. Also, large percentages of the engines going through field repair and major overhaul require replacement and repair of these parts. In one engine model, for example, 87 percent of the engines going through overhaul required repair or replacement of a part in the turbine section, either bucket, disks, or nozzle diaphragm. The median time to overhaul of this engine was 305 hours. In another engine model, 15 percent of the engines in overhaul having 50 hours of operation had their nozzle diaphragms repaired or replaced. After 275 hours, 90 percent of these engines required replacement or repair of the nozzle diaphragm. In other engines the "failure rate" for the same part was low.

4. The failures of some engine parts have been causes of flight accidents. In 1953, 205 accidents

were caused by engine failure or malfunction. In decreasing order of frequency, these were fuel-control failure (68 accidents), compressor failure including foreign-object damage (54 accidents), turbine bucket failure (16 accidents), turbine disk failure (14 accidents), and main bearing failure (10 accidents). Of the 54 accidents resulting from compressor failure, only one involved a centrifugal compressor, although the flying time accumulated for engines having centrifugal compressors was about the same as for engines having axial compressors.

APPENDIX A

CLASSIFICATION OF FAILURES

Failures have been classified on the basis of their governing laws into three broad categories (refs. 7 to 9). These categories are initial failure, chance failure, and wear-out failure.

Initial failure results from the fact that a component is defective at the time it is first put into operation. Such defects result, for example, from errors in manufacture or from the pre-use environment such as damage in storage, transit, or handling. Proper testing or "green-running" should prevent initially defective components from being put into service.

Chance failure results unpredictably from environmental causes. The fundamental characteristic of chance failures is that, for fixed environmental conditions, the hazard of a failure-causing condition is equally likely during equal times in the operating period; that is, the probability of failure is independent of operating time. An example of chance failure is damage resulting from foreign objects coming into the engine inlet. A foreign object is equally likely to enter the inlet at any time in the life span of the engine.

Wear-out failure results from the depletion of some material or property of the component that is essential to its proper operation. The depletion process may occur through abrasion, corrosion, or through the "using up" of life as in stress-rupture or fatigue. The probability of failure increases with age. An example of "wear-out" failure is the failure of turbine buckets by stress-rupture (see ch. VII).

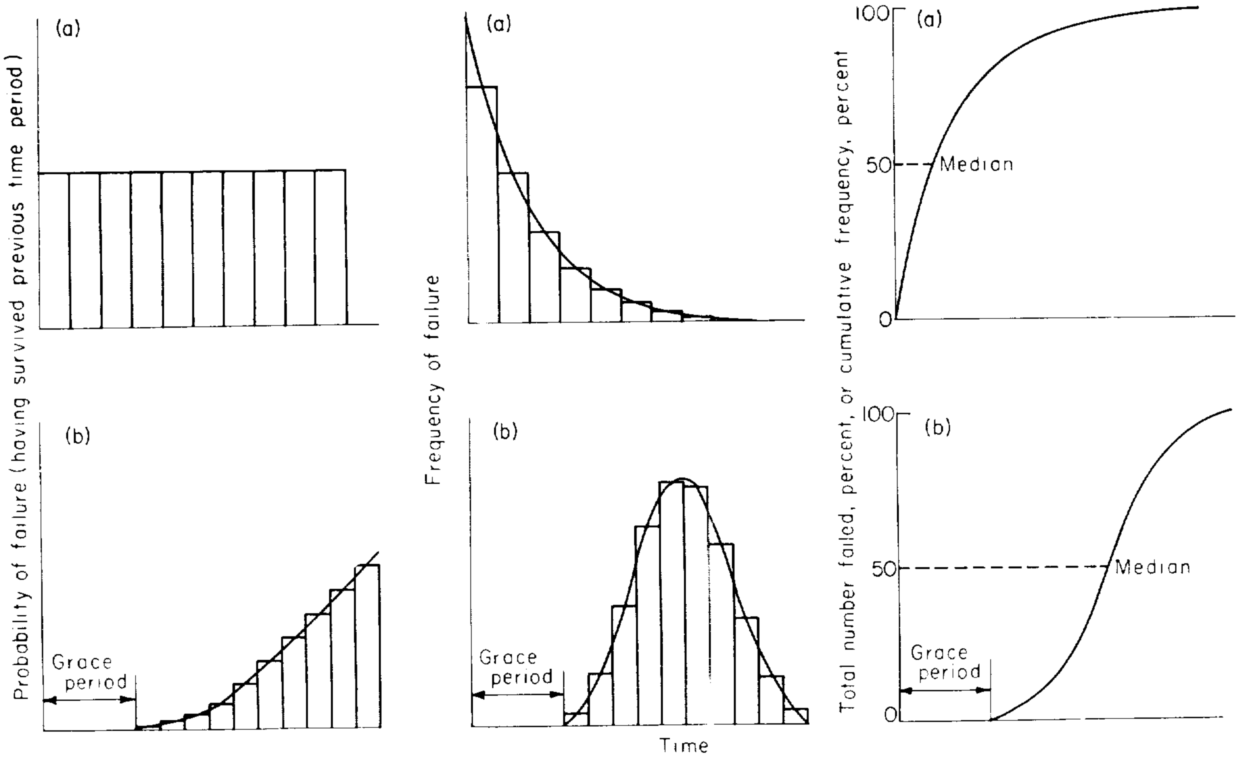
Figure 14 shows the shape of the curves for several functions for chance and wear-out failure. Both left and middle curves are instantaneous failure rates; the left curve is based on the part of population that has survived the previous time interval, whereas the middle is based on the starting population. The middle foretells at the start of operation what fraction of the starting sample will fail per time interval at some time in the future; that is, it evaluates a group performance. In contrast, the left curve deals with the individual that is still successful part way through its life span and gives its failure rate in the immediate future. The left curve is commonly called (ref. 9) the "hazard" of failure function, and the middle curve is called the "distribution" of failure function. The curves are, of course, mathematically related. If one is known, the other may be determined.

Comparing the chance and wear-out failure types, several facts are evident. With chance failure, a new component is as likely to fail as an old one, and nothing can be gained by scheduling replacements or by preventive maintenance. The failure rate can be reduced only by reducing the severity of the environment (the hazard), or by making the component better able to withstand the environment.

With wear-our failure, preventive maintenance can be extremely helpful, and replacements of old components may be scheduled to reduce failures in service. Also, only in wear-out failure is it possible to have a period of time before failure begins, a grace period.

To improve reliability, efforts must be made to reduce chance failure rates to a minimum, thereby ensuring that component failure is by wear-out. Then preventive maintenance can be used. The grace period for wear-out failures should be as long as possible, at least greater than the planned time to overhaul

The shape of the distribution of times to overhaul for a particular engine is usually the result of the additive effects of the distribution of fail-



(a) Chance.
(b) Wear-out.
FIGURE 14. –Examples of typical change and wear-out curves (ref. 8).

ure of many of its components (ref. 7). On new engines the shape of the curve may define the failure characteristics; chance distribution would probably indicate that unpredictable environment is very important, and wear-out indicates that environment is of lesser importance. For an overhauled engine, the shape of the curve of time to overhaul may be misleading. For example, an engine containing many components, each of which fails at a definite age (wear-out) but which are of mixed ages (from a previously repaired or overhauled engine), can exhibit a constant hazard of failure (chance). On the other hand, if an overhauled engine having parts of mixed ages shows essentially a wear-out failure distribution, perhaps it might be said that the important parts are being replaced in overhaul and that environmental hazards are not controlling failure.

The curves on the right side, which plot the total number of failures as a percentage of the initial population at any time of operation, are called cumulative frequency plots. All three curves are mathematically related. If one is known, the others may be determined. Use of this is made in the actuarial method (ref. 2). These cumulative frequency plots are useful for comparing engines, particularly, because the percent or numbers going to overhaul after a period of time and the median life or the life of the average engine may be read directly. The median is the 50 percent point; half the engines have a life less than this time and half have a life greater than this time. For reasons described in appendix B, the median (rather than the mean) is used in the body of this paper for comparing engines.

APPENDIX B

ACTUARIAL METHOD FOR DETERMINING ENGINE LIFE

In the study of engine reliability, it is important to know the average life of the engines and the percent of the engines that will require overhaul after various periods of time. The Air Force now uses the actuarial method to determine the overhaul distribution curves from which this information can be obtained.

Before describing the actuarial method, it is important to emphasize that the average engine life cannot be determined solely on the basis of the operating time on engines in overhaul. The primary difficulty with this practice is that the age of the engines in overhaul at any time is related to the age of engines in service. If, for example, a sample of new engines were introduced into service and the life of those engines that had gone to overhaul were examined a short time later, the average time to overhaul would be found to be very low, because the engines would not have had time to accumulate many operating hours. Obviously, the time to overhaul indicates the life of the bad engines; the good ones are still in service. If the operating time on the engines in overhaul is reexamined some time later, the average time to overhaul will be much higher and with time will probably tend to increase until all the engines of the sample have gone to overhaul. In fact, only after all the engines have gone to overhaul will the distribution of lives of engines at overhaul give the correct figures for engine life. Since the average engine in military service may fly only 20 hours per month, it would take 3 years to achieve 720 hours of operation on the average engine and perhaps twice that long for some engines of the sample.

In practice, the situation is more complicated than described, because, as engines are removed from service for overhaul, they are replaced in service by new or newly overhauled engines. Usually, this results in a distribution of operating time since new or since overhaul on engines in service, as shown by figure 15. A high percentage of the engines in service have low operating times, and thus the operating time on the average engine in overhaul will be lower than the correct distribution. Very little can be learned about overhaul times by examining the average times to overhaul of just those going to overhaul in a particular time period.

The most direct method of determining the percentage of an engine model that will go to

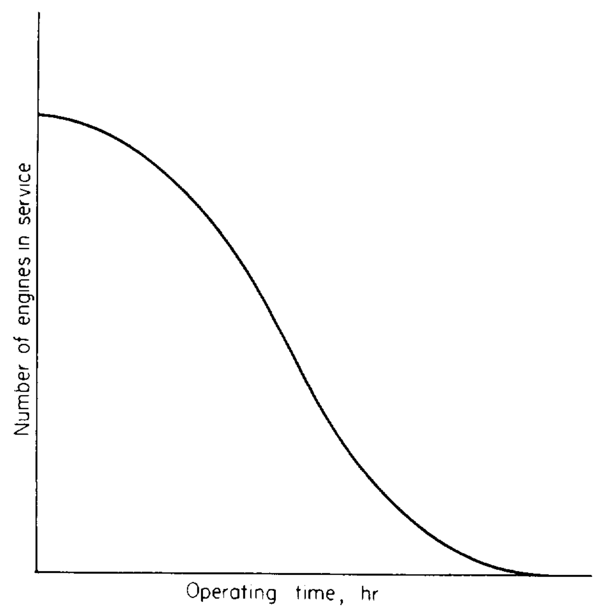


FIGURE 15.—Typical distributions of age of engines in service.

overhaul after various times of operation and the average life (information described by the middle or right curves of fig. 14) is to introduce a finite sample of engines into service and to determine the percentage of the sample that goes to overhaul after various periods of operation; for example, wait until all of the sample of engines have operated 50 hours and determine what percent went to overhaul, then wait until all had operated 100 hours and determine what percent went to overhaul, and so on. In practice, this method is almost impossible to use, because some of the sample of engines may be operated only very little and it may be a long time before all of the sample have completed even the first 100 hours.

It is usually more convenient to determine the left curve of figure 14 and from this to calculate the middle or right curve as desired. The left curve indicates the percentage of those that started a particular time interval (say from 50 to 100 hr) that fail in the time interval, whereas the middle curve describes the percent of the initial or starting sample that fail in the time interval. To determine the left curve, it is not necessary to wait until all of a sample have passed through the time interval; but rather a failure rate for any time interval can be calculated as soon as a sufficient number of engines have operated through the particular time interval to permit a statistically sound calculation to be made.

The Air Force uses the actuarial method to determine the left curve for all jet-engine models in service. The calculations are made as follows: For jet engines the operating time is divided into 10-hour intervals. Each month the Air Force determines how many hours were *flown* (number of "exposures to failure") by an engine model within each 10-hour time interval; then they determine how many engines from each time interval had to be sent to overhaul for "usage" reasons. The ratio of these two numbers is the average failure rate per hour for the particular time period. Multiplying by 10 gives the total failure rate for the 10-hour period. (A clarifying comment might be made about the importance of the term "exposure to failure." Whereas, in the case of human mortality, any living person is constantly exposed to causes of death, and this exposure cannot be turned on and off at will, engines are "exposed to failure" only when operating; thus, engines that have not operated during a study period must be excluded from the study sample.)

The same calculation is made for each 10-hour period for engine operating times as great as have been obtained. These failure rates are plotted against time, and the experimental data are then smoothed and extrapolated beyond the point where operating experience has been obtained. Figure 16 (from ref. 2) shows a typical plot of crude failure rates, the smoothing, and the extrapolation. In this particular plot the failure rate has been drawn up to 100 percent in the last time period (to 500 hr), since this is the maximum allowable operating time for the engine and overhaul is mandatory.

Basically, the preceding is the method by which the failure rates are calculated by the actuarial method. In practice, the actual calculations are

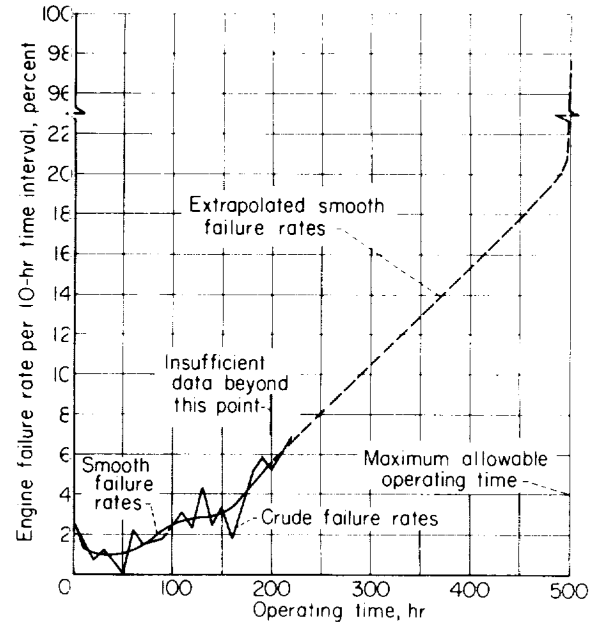


FIGURE 16.—Typical failure rate curves determined by actuarial method (ref. 2).

quite involved, because it is essential in determining the failure rate for each time period that the number of engines sent to overhaul be divided by the actual number of "exposures to failure" in the time period and not just the number of engines that happen to have, say, from 0 to 10 hours of operating time. Obtaining these data requires the use of an adaptation of the general "exposed to risk" formulas used in the calculation of mortality tables for human lives. This adaptation has been developed by the authors of reference 3.

Once the failure rate curve (the left curve) has been determined, the frequency distribution of times to overhaul (the middle curve) can be readily calculated by multiplying the number in the sample by the failure rate of the first time period to give the number failed in the first period, subtracting this number from the original sample size, and multiplying by failure rate for the second time period to give the number failed in the second period, and so forth.

The life of the average engine (the median) is recommended and is used in this paper as a basis of comparison rather than the average engine life (the mean), because the median is determined from only the early portion of the curve (e.g., fig. 16) that is based on experience and thus also does not include the number that arbitrarily go to overhaul because of maximum operating time (the final upswing of fig. 16). The median is the time when 50 percent of the engines of a sample will go to overhaul and can be read directly from a cumulative frequency plot (right curve of fig. 14); half the engines will go to overhaul in less than this time and half in more than this time.

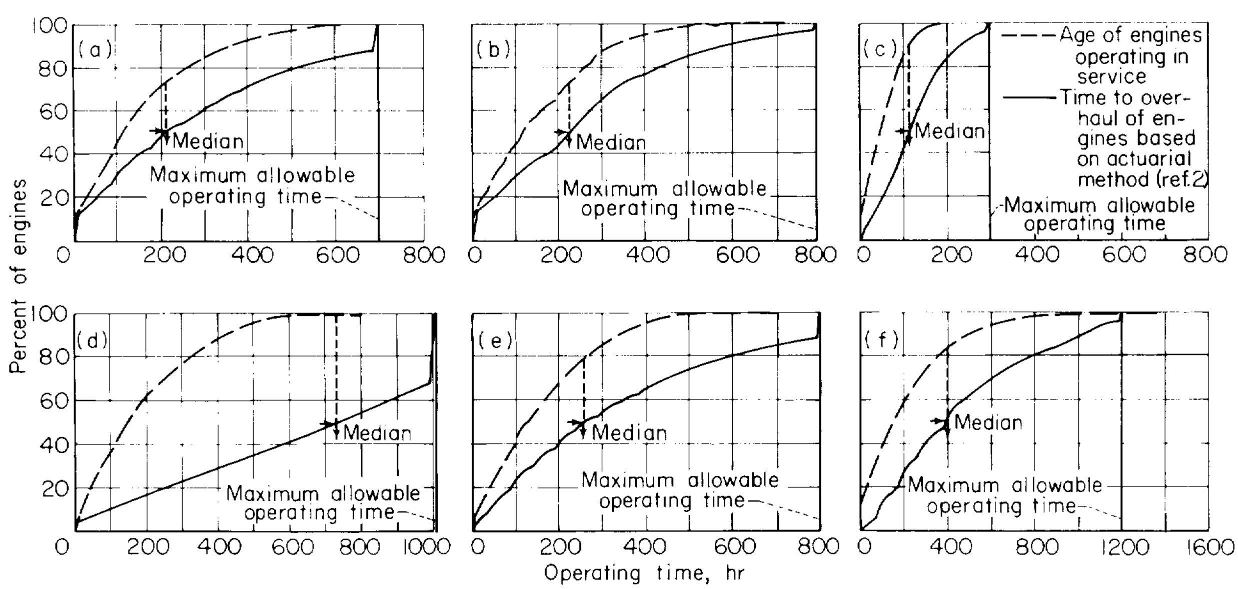
The failure rate curves determined for several engine models as of January 1955 are given in reference 2. As previously described, the curves have been extrapolated beyond the point where operating experience has been obtained. An indication of how much of the curves must have been extrapolated can be had by comparing the time to overhaul based on the actuarial calculation with the operating times on engines now in service. These distributions are compared for six engine models in figure 17. A comparison of the life of the average engine (the median) from the actuarial calculation with the number and percent of engines in service having more operating time than this is shown in table VII. These data are determined from figure 17. For example, for the A-7 engine the median time to overhaul (the 50-percent point) is about 208 hours.

TABLE VII.—PERCENT OF ENGINES IN SERVICE THAT HAVE MORE OPERATING TIME THAN PREDICTED LIFE FOR AVERAGE ENGINE (FIG. 17)

Engine model	Time to major overhaul for median engine (actuarial method), hr	Engines in service that have more operating time than predicted life for average engine (column 2)	
		Number [a]	Percent
A-7	208	142	27
B-3	231	118	27.5
B-9	743	58	1
B-10	258	322	21
C-7	402	495	16.5

[a] Calculated from total number in service as of Feb. 1955 (ref. 10 and table II).

The curve of age of engines in service (just above the actuarial curve) indicates that about 27 percent (100—73) of the engines in service have had longer than 208 hours of operation. From the data for number in service from table II, about 142 of the engines in service in February 1955 had operating times longer than this. Thus, the actuarial curve out to the median time to overhaul is based on a reasonable amount of experience. Table VII shows that the actuarial curves



(a) Engine A-7. (b) Engine B-3. (c) Engine B-7.
(d) Engine B-9. (e) Engine B-10. (f) Engine C-7.

FIGURE 17. Comparison of operating times on engines in service with time to overhaul as determined by actuarial method.

generally are based on appreciable operating experience out to the life of the average engine (the median) and should be quite accurate in this range. The major exception is the B–9 engine having the longest operating time to overhaul of all jet engines. In this case, of the 5783 engines in service, only 1 percent or 58 of the engines in February 1955 had operated beyond the median time. Therefore, the median might be subject to appreciable error in this case.

The actuarial curves predict that many of the engines now in service should go to overhaul, because they have reached their maximum allowable operating time; yet most of the distribution curves of figure 17 show that very few engines now in service are approaching such times. There are several possible reasons for this:

(1) The average number of hours of operation per month for these military engines is small. A spot check indicated 17 and 20 hours per month for two fighter engines and 24 and 31 hours per month for two bomber engines. If an engine were operated 20 hours per month, it would require $3\frac{1}{3}$ years to accumulate 800 hours of operation, the maximum allowable operating time for the B–10 engine.

(2) When a new engine is introduced into service, the maximum allowable operating time may be set as low as 50 hours and moved upward as experience is obtained. Thus, the low operating times on any group of engines may indicate that the allowable operating time may have only recently been moved upward, and no engines have had time to accumulate a large number of operating hours.

(3) The actuarial method considers only those engines that go to overhaul because of "usage reasons." Ten to thirty percent of some models sent to overhaul were found to have been removed from service because of "nonusage" reasons (e.g., for modification, because the engine was to be transferred overseas, or because of accident or combat damage).

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CHAPTER III

FOREIGN-OBJECT DAMAGE

By Lewis A. Rodert

SUMMARY

A review of U.S. Air Force service records shows that foreign-object damage in jet engines is a major maintenance and safety problem. Few objects are identified; however, most damage is caused by air-base surface debris, parts of failed inlet components, and objects left by personnel. Centrifugal engines are less vulnerable to damage than axial types. Aircraft designs providing maximum engine-inlet height have minimum engine damage. Screens for axial-flow engines have not been effective in preventing damage necessitating premature overhauls but probably have prevented accidents. Personnel training, improved air-base debris removal, avoidance of mass taxi operations, improved engine screens, and engines of rugged construction are suggested as remedial measures.

INTRODUCTION

The ingestion of foreign objects into gas-turbine engines has been the cause of many minor repairs, premature engine overhauls, and major aircraft accidents. An effort has been made by the military services and the aviation industry to reduce foreign-object damage (refs. 1 to 3), but opinions differ on how the reduction may best be accomplished (refs. 4 and 5).

Records of service experience with gas-turbine engines and other available sources have been reviewed for information on the following questions:

(1) What is the effect of foreign-object damage on reliability and safety?

(2) What are the origins and modes of ingestion of objects which damage engines?

(3) What efforts have been made to reduce foreign-object damage?

(4) What steps should be recommended for further damage reductions in the future?

This report attempts to answer these questions.

ANALYSIS OF PROBLEM

MAINTENANCE AND SAFETY

When damage caused by the ingestion of foreign objects is slight, the engine is repaired at the maintenance base of the operating squadron. In a 3-month period of 1954 the required reworking of compressors due to foreign-object damage accounted for about 18 percent of the minor repairs on axial-flow engines performed at the squadron bases. Maintenance work necessitated by foreign-object damage on centrifugal engines was less than 2 percent of all minor repairs during the same period.

Air Force Technical Orders specify nick and dent tolerances for gas-turbine engines. When damage caused by the impact of an object exceeds that allowed in the technical orders, the engine must then be sent to an overhaul base for complete disassembly and repair.

Air Force Disassembly Inspection Reports (DIR's) prepared by the gas-turbine engine overhaul bases during the period extending through August, September, and October, 1953, have been reviewed in detail.[1] The sample includes reports for 222 type A engines, 666 type B engines, and 726 type C engines. Engines A and B are representative of axial-flow engines, and engines C are representative of centrifugal engines. Disassembly Inspection Report Summaries for 1953 and the first 6 months of 1954 were studied for a further indication of the trends in the problem.

The inspection of a gas-turbine engine during overhaul may reveal indications of several mal-

[1] Engine model and dash numbers are given in code to conceal the identity of proprietary products.

functions. Each malfunction or unsatisfactory condition found is described in the inspection report. A primary reason for the removal of the engine from service, based on information obtained during the inspection, is indicated in the report. Malfunctions, therefore, are reported either as the primary reasons for removal from service or as simultaneous failures also found during inspection.

The data on the damage by foreign objects were compiled as (1) total damage composed of both primary and simultaneous damage, and (2) primary damage only.

In the NACA study, foreign-object damage included damage caused by objects originating outside the aircraft, objects probably left in the inlet duct by personnel, objects generated by failure within the inlet duct including screen components, and objects of unknown origin. Damages caused by objects generated by failure behind the inlet screen or engine face were considered internal failures. Foreign-object damage was reported only when a nick or dent was observed that was clearly caused by the passage of an object through the engine. If an unexplainable major internal breakup of the engine had occurred, nicks or dents on the rotating parts were considered to have been caused by debris from the unknown failure and not by foreign objects.

The engines were divided according to primary cause for removal from service under the following headings:

(1) Foreign objects
(2) Other factors

The criteria for category (1) have been noted previously. Category (2) includes engines damaged as a result of internal failures, malfunctions caused by personnel error, damages caused by crashes or accidents, engines overhauled for unknown causes (primary failure not identified), engines removed from service without apparent damage, and malfunctions resulting from accessory or control-system failures. The compilations were made for all engines of each model irrespective of engine dash number and previous overhaul experience.

The inspection of extensively damaged engines frequently fails to lead to the identification of the initial failure cause. The difficulty of the inspection is increased if the engine has been subjected to the impact of a crash. For these reasons, not all engines damaged by foreign objects have been so listed in the compiled data. The results, therefore, are in error, the problem being of greater magnitude than indicated by the overhaul records.

Information contained in the DIR's on damage and overhauls caused in engines A, B, and C by foreign objects is summarized in table I. Most of the reported damage to engines A and B was in the compressor. Most of the foreign-object damage to engine C was in the form of nicks and dents on turbine-section components.

The Air Force DIR Summaries provide additional information on the frequency of premature overhaul caused by foreign-object damage. Thirty two percent of all jet-engine overhauls in 1953 were prematurely caused by foreign-object damage. The percentage of premature overhauls

TABLE I.—SUMMARY OF FOREIGN-OBJECT DAMAGE AND JET-ENGINE OVERHAUL CAUSES

Foreign-object damage	Engine A		Engine B		Engine C		All	
	Number	Percent	Number	Percent	Number	Percent	Number	Percent
Damaged	167	75	366	[illegible]5	492	68	1,025	64
Not Damaged	55	25	300	[illegible]5	234	32	589	36
Total	222	100	666	100	726	100	1,614	100
Overhaul causes								
Foreign objects	119	53	251	[illegible]8	110	15	480	30
Other factors	103	47	415	62	616	85	1,134	70
Total	222	100	666	100	726	100	1,614	100

due to foreign objects in engines B increased from 38 percent in 1953 to 42 percent in the first 6 months of 1954.

The hazard to flying safety created by the ingestion of foreign objects into gas-turbine engines cannot be evaluated directly from engine-overhaul statistics. Foreign objects may have caused major jet-engine aircraft accidents, but confusion of the debris usually concealed the evidence required to prove the fact. Studies by the Air Force Directorate of Flight Safety Research have led to the statement that "axial flow compressor failure is the largest single factor contributing to the jet engine accident rate. Foreign object damage and metal fatigue in compressor rotor parts are principle [*sic*] factors in these failures" (ref. 1). Therefore, it is important that foreign-object damage be reduced in order that jet-engine maintenance problems be minimized and flight safety improved.

OBJECTS IDENTIFIED

The objects that damaged the engines are listed in table II, insofar as identifications were made or inferred in the DIR's. The identifications are based on the following:

(1) The object was found in the damaged engine or otherwise positively identified.

(2) The nature of the damage indicated the kind of object responsible.

(3) A missing part from the inlet components could have caused the damage.

The significant observation to be made from the information presented is that comparatively few of the objects that damage engines are identified (table II). Large objects may cause major breakups of engine interiors or accidents and then become lost in the debris. Small objects may pass through the engine and become lost. Pebbles and other frangible objects may become broken into very small pieces and make identification improbable or of dubious value. Much of the damage is therefore attributed in the official reports to objects of unknown origin or identity.

Rivets, screws, special fastenings, and screen segments that fail and/or become loose in the engine inlet are a known source of damage. Spare parts and metallic debris left in the inlet by manufacturing, maintenance, and operating personnel are also known to have contributed to the problem.

The DIR's indicate that most of the damage in engines sent to overhaul bases is of moderate severity in the form of nicks and dents, the depths of which exceed limits specified by Air Force Technical Orders. The sizes of the nicks indicate that most of the damage is done by small objects. Small objects generated by the failure of inlet-

TABLE II.—IDENTIFIED OBJECTS CAUSING PREMATURE OVERHAUL OF ENGINES

Objects	Number	Percent of engines overhauled	Objects	Number	Percent of engines overhauled
Engine A			Engine B		
Screen segments [a]	23	19.3	Metal pieces	22	8.8
Rocks and pebbles	5	4.2	Rocks and pebbles	17	6.8
Battle debris	1	.8	Screws and bolts	12	4.8
Unknown	90	75.7	Failed parts	10	4.0
			Safety wire	6	2.4
Total	119	100.0	Tools	5	2.0
			Cloth	2	.8
Engine C			Battle debris	1	.4
			Bird	1	.4
Tool	1	0.9	Animal	1	.4
Unknown	109	99.1	Unknown	174	69.2
Total	110	100.0	Total	251	100.0

[a] All on A-7 engines.

duct components are sometimes identified (table II).

Large objects do not pass through the engines and are therefore found unless a major accident results and the evidence is lost. Large objects such as tools, failed parts, and spare parts left in the engine inlet have been identified, although comparatively infrequently (table II).

Objects left in engine inlets, including tools, parts, and scrap, during final assembly and preparation for initial flight tests by aircraft manufacturers and overhaul agencies may constitute the major cause of damage when engines are initially operated. Debris from the air-base surface, including pebbles, concrete, and metallic objects, is believed to cause most of the damage after the engines have been placed in service by the operating agencies.

MODES OF ENTRY

References 2 and 5 indicate possible ways in which foreign objects enter engines. The mode of entry of objects generated or left in the engine inlet is self-evident. Airport debris may be blown into engine inlets by the blast of other jets or may be thrown in by landing-gear wheels. Other investigations of the modes of entry (refs. 4, 6, and 7) conclude that the engine air-inlet stream, unaided by outside influences, will not cause the ingestion of objects from the ground surface if the airflow is uniform and undisturbed by vortex formations.

The ingestion of foreign objects from the ground by vortices formed between the engine inlet and the ground was investigated at the Lewis laboratory (ref. 8). The presence of vortices is evidenced by dust and water whirls and occasionally by visible cores (fig. 1). The visible core of a vortex formed at an engine inlet is composed of condensed water droplets. The condensation of these droplets results from a static temperature equal to or less than the dewpoint temperatures at the vortex-core pressure. The reduced temperature in the vortex core is indicative of a low-pressure region.

The NACA study of ingestion by vortices showed that "pebbles, typical of objects that damage jet engines, were projected into the air by the vortices and were drawn into the engine by the high-velocity inlet-air stream." Vortex formation

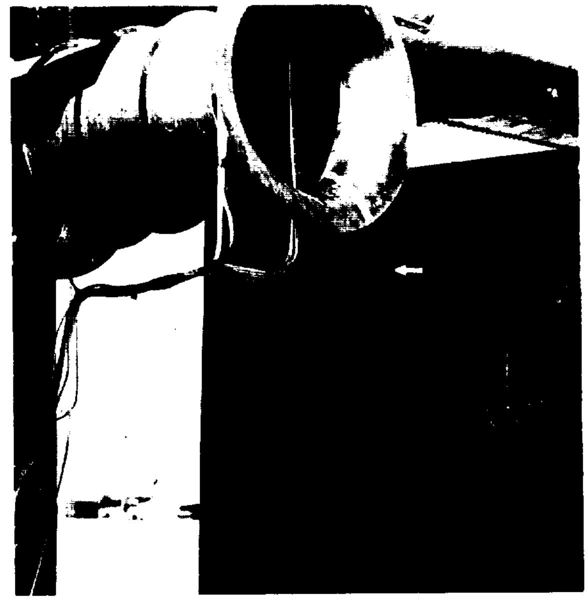


FIGURE 1.—Vortex formation between air inlet of gas-turbine engine and adjacent plane surface.

depended on engine speed, engine height, and surface wind (ref. 8). The possibility of ingesting airport surface debris is enhanced by

(1) Increased engine speed

(2) Increased engine size

(3) Reduced engine-inlet height above the ground surface

(4) Reduced wind or taxi speed

Pebbles on smooth surfaces are less likely to be projected upward into the inlet by a vortex than when they are lodged in a crack. When exposed on a smooth surface, the pebbles were swept aside by the circular motion outside the vortex core but were not projected upward. Pebbles lodged in cracks and thus constrained from lateral motion were projected into the air when a vortex core passed over the crack.

Thus, from available information, the nature and modes of entry of foreign objects of major importance are as follows:

(1) Inlet components released by failure and drawn into the engine

(2) Objects left in the inlet by personnel and drawn into the engine

(3) Air-base debris thrown into the engine inlet by the blast of other jets or aircraft landing-gear wheels

(4) Air-base debris ingested by engine-inlet vortices

DAMAGE PREVENTION

The damage caused by the ingestion of foreign objects into gas-turbine engines may be reduced by the following:

(1) Ruggedly constructed engines

(2) Aircraft design, particularly increased engine air-inlet height above the ground

(3) Air-base construction and operation

(4) Engine-inlet screens and inertial separation devices

RUGGED ENGINE CONSTRUCTION

The rugged nature of the centirfugal engine C is indicated by the data in table I. Foreign-object damage occurred in 68 percent of the centrifugal engines and in an average of 60 percent of the two axial-flow engines. However, foreign-object damage was the cause of premature overhaul in only 15 percent of the centrifugal engines, as contrasted with about 42 percent in the axial-flow engines.

New axial-flow engines presently coming into service are expected to be less vulnerable, but service data are not yet available on these engines. Speculation on reduced vulnerability is based on the benefits to be derived from design features of the new engines, several of which are

(1) Loosely held compressor blades that will reduce impact damage

(2) Increased safety factors in blade design that will extend the nick tolerance of blades

(3) Shrouded stator blades that will tend to arrest the propagation of damage through an axial-flow compressor

These factors tend to reduce the hazard caused by the ingestion of foreign objects. However, whenever a blade is nicked, repair will still be necessary before flight is possible.

To obtain high airflow per unit frontal area and high pressure ratio per stage, compressor design trends of the future are expected to include longer blades (high tip to hub ratio) and higher rotating speeds. Both trends will increase blade stress and hence reduce blade factors of safety and will require improved materials and construction methods, if future engines are to be less vulnerable to foreign-object damage.

AIRCRAFT DESIGN

The types of aircraft in which the engines were installed were indicated in the reports of most engine inspections. These data made possible an analysis of aircraft design effects on the frequency of jet-engine damage. The following factors were examined:

(1) Landing-gear wheel locations in relation to engine-air-inlet location

(2) Height of engine air inlet above ground surface in ground operating condition

No consistent changes in damage rates were found for wide variations in wheel location, indicating that objects tossed up by wheels are responsible for little damage.

The relation of engine damage to the distance from the engine air inlet to the ground surface or engine height is indicated in figure 2. The results of this analysis show that the percentage of engines damaged in the various aircraft increased as the height of the engine air inlet decreased. Variations in engine height probably affect the ingestion of objects picked up from the air-base surface.

The study of foreign-object ingestion by engine-inlet vortices (ref. 8) indicates that the maximum height at which a vortex will form between the inlet and ground depends on the operating power of the engine. Increasing the power resulted in the formation of vortices at greater heights. The use of higher powered engines mounted in twin pods is believed to provide more favorable conditions for vortex formation and may possibly result in a higher damage rate at a given height than indicated in figure 2.

The details of the air-inlet duct design can also affect the foreign-object problem. Design ingenuity and safety margins determine inlet com-

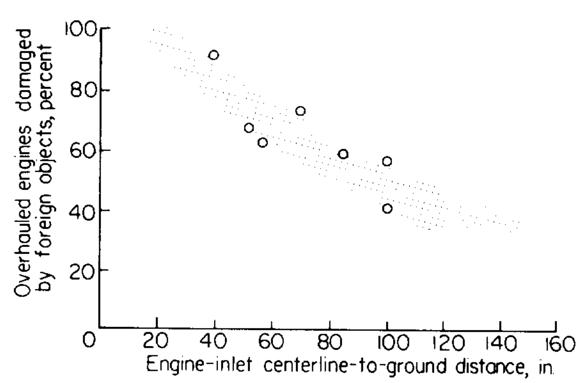


FIGURE 2.—Variations in damage to engines with engine-inlet height plotted for several representative Air Force aircraft.

ponent failure frequency rates. The configuration of the duct determines the ease and efficiency with which inspections may be made for objects left in the inlet by personnel and also affects the efficiency of the engine-inlet screen installation.

OPERATING TECHNIQUES AND AIR-BASE CONSTRUCTION

The "Name and Location of the Last Using Activity" of the engines for which inspection reports were analyzed indicated the local base or region where each was operated. Operating environmental factors believed to influence foreign-object damage include the effectiveness of debris removal, mass taxiing and takeoff maneuvers, operating personnel efficiency, and air-base surface material. Foreign-object damage was compiled for engines installed in various aircraft for the various air bases designated as the "Last Using Activity." Foreign bases and bases from which fewer than ten engines had been received and inspected were not analyzed in the study. The results of the analysis of engines B damaged by foreign objects are given in table III. The data were analyzed for each aircraft to avoid complex interrelations with effects of engine height given in figure 2. Foreign-object damage to engines varies over wide ranges with variations in base of operations. As noted for bomber aircraft B–2 and B–4, the engine damage rate at some bases may be twice as great as at others. The variation of fighter-engine damage with bases is significant but not as large as for the bombers.

The study indicates that the overall damage caused by foreign objects might be reduced by universally applying the operating techniques and air-base construction methods followed at the bases having the lowest rates. The following actions are suggested:

(1) Develop surface-cleaning devices capable of covering the large areas of an air base in a short time and removing all debris, including objects lodged in cracks

(2) Avoid mass taxi and takeoff maneuvers

(3) Train personnel to inspect for and remove objects in inlets and on ground under inlets in run-up areas

(4) Eliminate airport debris by the development of improved materials and construction techniques

Investigations of runway surfaces should include a search for materials that do not generate debris when subjected to repeated freezing cycles, heating cycles from jet exhaust, fuel spillage, and other forms of destructive exposure.

SCREENS

The use of screens as protection against damage to gas-turbine engines by foreign objects is a controversial issue. In support of screens is the claim that the ingestion of large objects that can cause complete and sudden engine failures followed by aircraft accidents may be prevented by the use of screens. In 1952 the removal of fixed screens from most Air Force aircraft due to icing difficulties was accompanied by a sharp rise in major accidents. When retractable screens were later installed, flying safety improved, indicating that the use of screens had prevented the ingestion of objects that can cause accidents (ref. 2).

TABLE III.—AIR BASES FROM WHICH DAMAGED ENGINES B WERE RECEIVED

Base or airport [a]	Total sent to overhaul	Foreign-object damage, percent		
		Bomber B–4	Bomber B–2	Fighter F–3
a	11	91		
b	28			79
c	27	78		
d	23			74
e	24	71		
f	15	67		
g	22			64
h	10		60	
i	19			53
j	30	50		
k	10		50	
l	95	47		
m	14	36		
n	22			
o	38		24	

[a] Denoted by code letter.

Nevertheless, strong objections to the use of screens have been raised (ref. 4). The arguments against their use are

(1) Air-pressure loss across screens reduces engine thrust and moves the condition of engine operation closer to the stall region of the compressor.

(2) Small objects pass between screen elements, and large high-velocity objects break through and cause engine damage.

(3) Retractable screens as presently designed and operated dump the collected debris into the engine when they are retracted during flight.

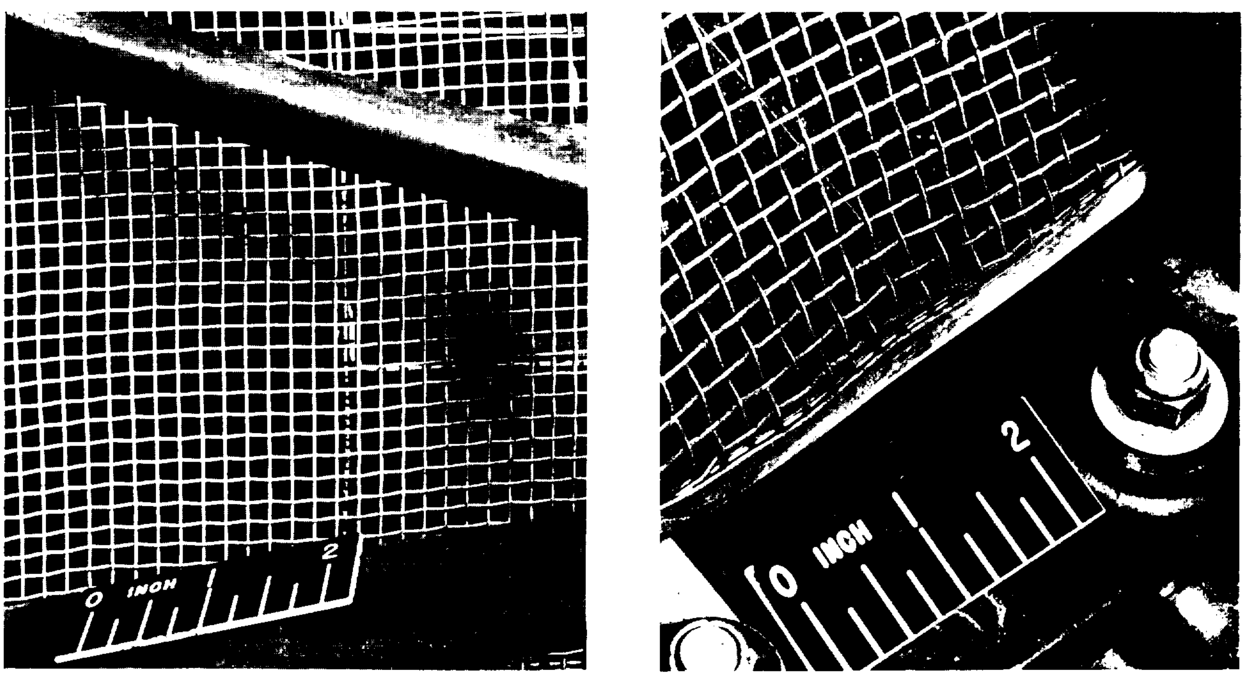
(4) Warped screens become inefficient, and those that fail in fatigue provide an additional source of engine damage.

(5) Screens are vulnerable to icing and add weight and complication.

Screening effectiveness.—Sections of screens used on centrifugal engines are shown installed in figure 3. Fixed screens used on early axial-flow engines are shown in figure 4, and a retractable screen is shown in figure 5. These screens are representative of equipment developed and used over the past several years. Other screen equipment is currently under development by the aviation industry.

Centrifugal engines in the present study were fitted with screens shown in figure 3. The number of engines fitted with each screen, the mesh dimensions, and the percentage of engines damaged by foreign objects are given in table IV. Early models of the centrifugal engines were fitted with the fine screen having 0.132-inch-square openings. More recent engines were fitted with the coarse screen having 0.216-inch-square openings.

The two screen types were inspected after service on engines in NACA research projects. The fine mesh screens were in excellent mechanical condition after extensive usage; the coarse mesh screens were deteriorated. Broken wires, enlarged mesh dimensions, and large edge clearance between screen and engine frame resulted in openings as large as 0.50 inch in the coarse mesh screen (fig. 3 (b)) through which objects might pass. Thirty-six percent of the engines with the fine mesh screen were damaged, while



(a) (b)

(a) Openings 0.132 inch square.

(b) Openings 0.216 inch square.

FIGURE 3. Fixed air-inlet screens on centrifugal engines.

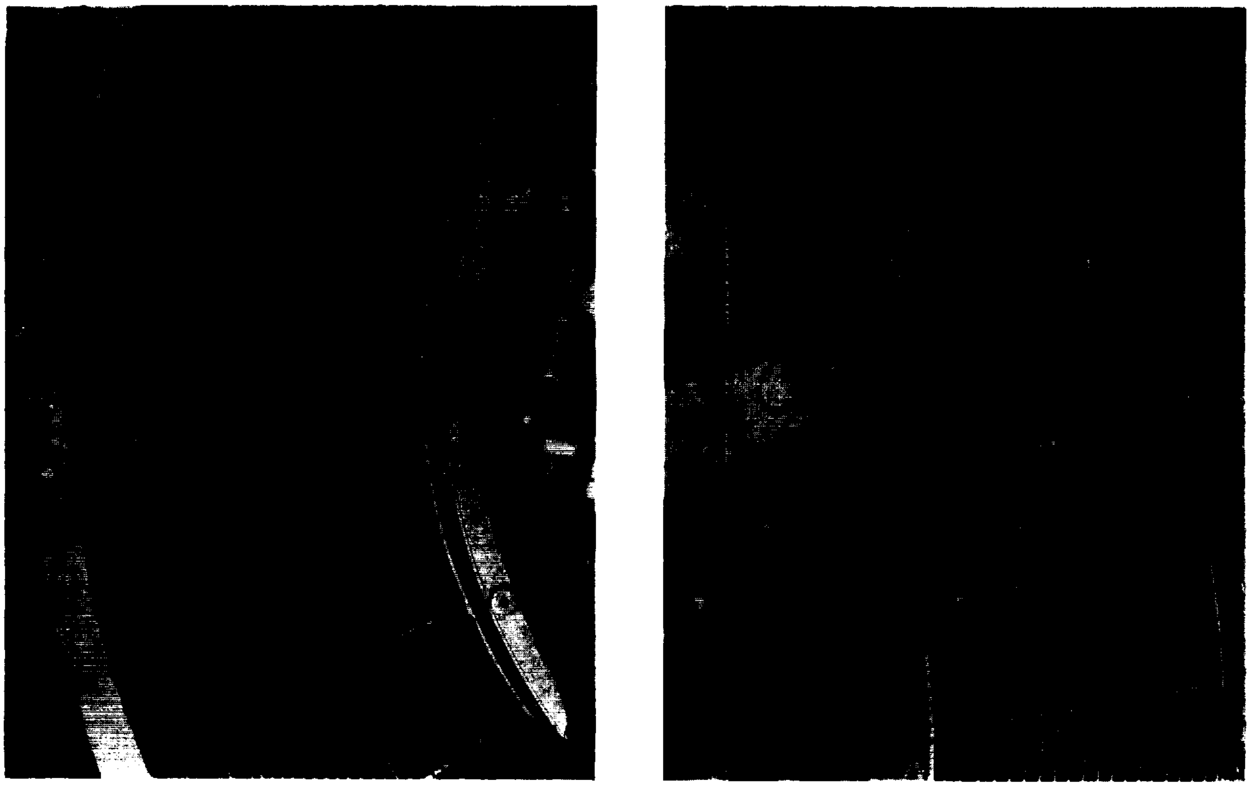


FIGURE 4. Fixed inlet screens on early axial-flow engines.

84 percent of the engines with the coarse mesh were damaged, which indicated the need for small screen openings and the importance of construction details that ensure reliability in screen design.

TABLE IV.—FOREIGN-OBJECT DAMAGE TO ENGINES C WITH VARIATIONS IN SCREEN INSTALLATION

Number of engines	Engine	Screen installation	Percent of engines damaged
239	C-1 to 4	Fixed, 0.034-inch wire diameter spaced 0.166 inch on center giving 0.132-inch-square openings (fig. 3(a))	36
487	C-5 to 7	Fixed, 0.034-inch wire diameter spaced 0.250 inch on centers giving 0.216-inch-square openings (fig. 3(b))	84

Insufficient data hampered the study of the effectiveness of screens for axial-flow engines. The DIR's show that about 40 percent of all axial-flow-engine overhauls are prematurely caused by foreign objects. Thus, the general implication of the results of the present study and of the data in the Air Force DIR Summaries is that screens are ineffective. While present axial-flow-engine screens may prevent accidents, the maintenance problem due to the ingestion of foreign objects still exists. These observations are confirmed by the results of Air Force studies of the need for screen improvements (ref. 2).

Performance penalties.—The inlet screen acts as a restriction in the engine inlet and causes a reduction in thrust due to reductions in air weight flow and pressure ratio across the engine. The static-thrust reduction caused by inlet-screen pressure losses has been computed for engine B operating at a sea-level static pressure of 2,116 pounds per square foot, a nozzle-outlet temperature of 1,740° R, and a nozzle pressure ratio of 1.91. For small

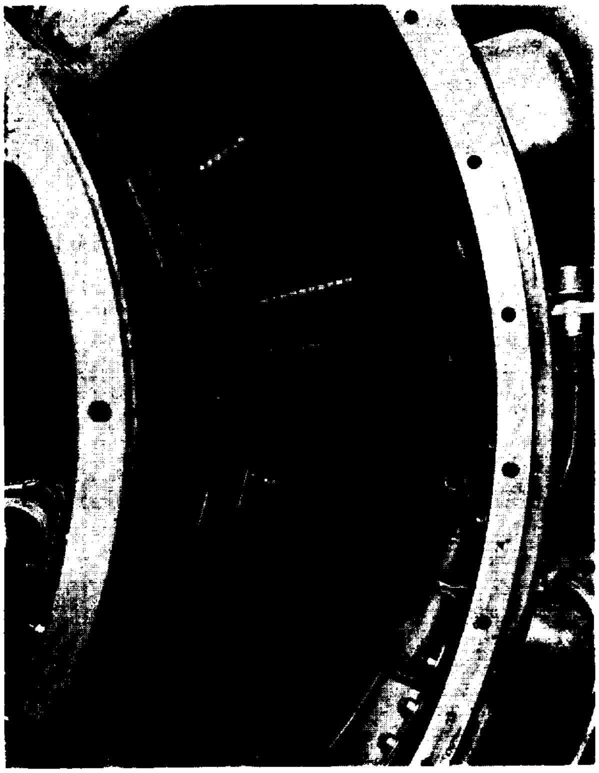


FIGURE 5.—Retractable inlet screens on recent axial-flow engine.

inlet pressure losses ΔP_1, the thrust loss ΔF_j for engine B is given by the equation

$$\frac{\Delta F_j}{F_j}=1.85\frac{\Delta P_1}{P_1} \quad (1)$$

in which F_j is the normal thrust and P_1 is the inlet total pressure. The relation of equation (1) is shown graphically in figure 6.

The pressure loss across an engine-inlet screen composed of streamlined section elements may be computed from the equation

$$\Delta P=C_D\frac{c}{s}q \quad (2)$$

in which

- P pressure, lb/sq ft
- C_D section drag coefficient
- c section chord, ft
- s section spacing, ft
- q dynamic pressure of flow through screen, lb/sq ft

The design of screens involves several conflicting requirements, as may be seen from equation (2). High impact strength requires large screen section chord and thickness. Thickness is important when objects strike between the sections. High resistance against vibration stresses also requires large thicknesses. High screening effectiveness, however, requires small screen section spacing; and low thrust loss requires large spacing and small chord, location of the screen in a region of minimum velocity, and a minimum drag coefficient.

The drag coefficient depends on several additional factors, including chord-thickness ratio, chord-spacing ratio, Reynolds number range, and aerodynamic smoothness. Turbulence or flow distortion may cause the streamlines to deviate from the angle of minimum drag with the screen and thereby increase the pressure loss. Therefore, the pressure loss across the screen also depends on the design of the air-inlet duct.

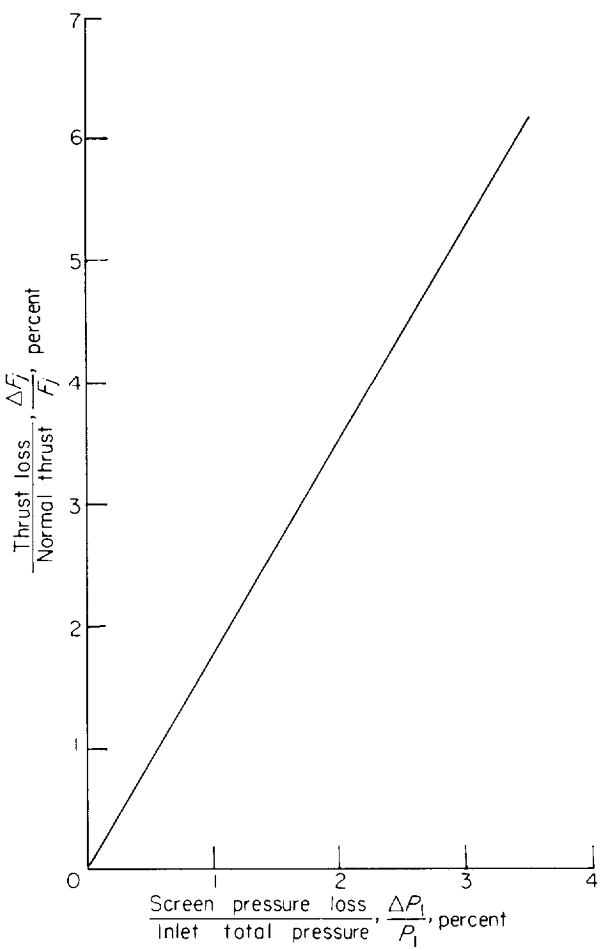


FIGURE 6.—Thrust loss caused by inlet-air screen.

The effect of engine-inlet screens on the performance of an F-86D airplane was investigated by the U.S. Air Force (ref. 9). The report states that "a static thrust calibration showed that extending the inlet screens caused a blocking effect on the engine air-intake ducts which resulted in a loss of thrust under sea-level standard day conditions, of 3.3% for the 'Military Power' thrust selector position (afterburner off) . . ." The drag coefficient for the F-86D aircraft screen computed from the thrust loss and the screen dimensions (substituted in eqs. (1) and (2)) is about 0.145. An experimental engine-inlet screen (ref. 10) gave pressure losses corresponding to a drag coefficient of about 0.06. This screen was of the fixed type and was mounted in a laboratory duct setup. The F-86D airplane screen drag is greater than that of the experimental screen, probably because of flow distortions and turbulence in the airplane air inlet and greater aerodynamic roughness of the service equipment. A review of available knowledge on pressure losses across screens and grids (ref. 11) shows the need for additional drag data on screens of streamlined sections.

Efforts to improve screens should be aimed at the following design objectives:

(1) Aerodynamically smooth screens and inlet ducts

(2) Undistorted inlet airstreams

(3) Low-velocity screen location

(4) Small section spacing

(5) Retention of objects caught

(6) Structural ruggedness of screen

The achievement of these goals will overcome most of the objections to screens listed previously. Efforts to improve engine protection should also include consideration of inertial separating devices. Inertial separation of objects in combination with the use of screens is also of interest among the possible ways of solving the problem.

Ground run-up screens.—The military services have attempted to reduce foreign-object damage by the use of screens attached only during some engine ground operations (ref. 12). The following ground-screen problems were revealed in collecting data for this report and in interviewing personnel:

(1) Ground screens impose high pressure loss and thereby prevent rated power operation of engines. All engines must be operated at rated speed in order to make fuel-control adjustments.

(2) Screen mesh openings are not small enough to stop many objects.

(3) Ground screens that are large enough to reduce pressure losses and that have small enough mesh to stop all objects are bulky and cause a hazard to personnel who apply or remove them from operating engines.

Perhaps future improvements in other methods of protection will eliminate the need for ground run-up screens. In the interim, however, ground screens are important in protecting engines against foreign-object damage, and their continued development is needed.

ADDITIONAL INFORMATION NEEDED FOR IMPROVEMENT OF OPERATIONAL RELIABILITY

Reduction of foreign-object damage to gas-turbine engines may be effected by further engine development, airplane design, improved operating techniques, and special protective devices such as screens.

FOR IMMEDIATE APPLICATION

Some measures may be applied from existing knowledge with little or no equipment development, while others will require additional information and development. The following measures are suggested for immediate application:

(1) Improve and uniformly apply training and supervision of manufacturing, maintenance, and operating personnel in engine-damage avoidance. Emphasis should be placed on preflight inspection of engine inlets and ground areas under engine inlets.

(2) Improve debris removal from ground operating areas, particularly from paving cracks, and eliminate sources of debris.

(3) Avoid mass taxi and takeoff maneuvers and other operations that cause debris to be thrown into engine inlets.

(4) Employ available ground and engine-inlet screens as much as possible.

(5) Inspect engines carefully after each operation for screen damage and nicks and dents in the entering stages of the compressors. Inspection of all blades would be ideal, but in absence of a practical method by which this may be accomplished in current engines, a significant percentage of damage can be detected from the inspection of entering stages and screens. Entering stages are the most highly stressed, have the lowest factors of safety, and are most vulnerable to foreign-object damage.

FOR FUTURE APPLICATION

Measures to be recommended and on which additional information is needed are:

(1) Improved air-base cleaning equipment

(2) Redesigned programs for the elimination of unreliable air-inlet components and fastenings that can release objects and damage engines

(3) Improved engine-inlet screens

Information on screen drag should be provided for variations in screen geometry and Reynolds number to enable designers to achieve the most favorable compromise between preserving maximum thrust and protecting against damage. Investigations should also be made of the effects of screen location on inlet air pressure recovery, uniformity of flow to the engine face, and ice prevention. The results should enable designers to select the most favorable screen location in consideration of all the problems and the types of aircraft involved. Also, installations that take advantage of inertial separation should be studied.

Since no single material contains the optimum characteristics required for compressor and turbine blading, it is possible that the ruggedness of such blading can be improved by combining several materials into a composite structure. Thus, the various components may individually contribute to static strength, fatigue strength, abrasion resistance, internal damping, and so forth. Research should be undertaken to determine the optimum combination of materials and the structure into which they should be assembled in order to provide the maximum resistance to denting and the minimum reduction in life when denting occurs.

CONCLUSIONS

The ingestion of foreign objects into jet engines is a threat to flying safety and necessitates maintenance on many engines which otherwise would have remained in service. Few objects that damage engines are identified. Air-base surface debris blown in by other jets or ingested by vortices provides a major part of the problem, but failed inlet components and objects left in inlets by personnel also contribute. Centrifugal engines are less vulnerable than axial types, although both have high damage rates. Increasing engine-inlet height reduces the foreign-object damage rate; however, increased height is not a panacea. Significant variations in foreign-object damage exist in engines operated at different bases; thus, the operating environment and techniques are also important factors in determining damage rates. Screens have been ineffective in preventing damage necessitating maintenance; however, the record indicates that screens have improved flying safety somewhat.

Steps to reduce foreign-object damage should include personnel training, better debris cleanup, improved screens and air inlets, and engines of increased resistance to foreign-object impact. Research on the criteria of screen design, screen installation, and rugged engine construction is suggested. Until the danger of foreign-object damage is completely eliminated, it is essential that a scheduled inspection be set up for damage to the compressor and turbine blades.

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CHAPTER IV

COMPRESSOR BLADES

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SUMMARY

Military failure statistics indicate that foreign-object damage, vibration fatigue, and stress-corrosion cracking are among the more important failure causes placing compressor blades high on the list of engine components responsible for engine overhaul and flight accidents. These statistics are reviewed, background information on the failure mechanisms is described, and possible remedial measures are discussed. Design, operation, and inspection practices that would have a beneficial effect on service reliability, as well as future research that would provide a firmer basis for achievement of reliability, are also discussed.

INTRODUCTION

One of the most vulnerable components in a gas-turbine unit is the axial-flow compressor. Foreign objects entering the engine inlet cause the largest portion of compressor damage, and vibration failures of the rotor blading also account for many compressor losses.

In early engines using centrifugal compressors, the few problems encountered were quickly solved, and the centrifugal compressor gained a reputation of considerable endurance and reliability (ref. 1). The large diameters associated with centrifugal compressors and their limited pressure ratio eliminated their application in later units. They were replaced almost exclusively by the more efficient and higher pressure ratio axial-flow compressor.

Even if centrifugal compressors are used in early commercial transports because of added reliability, there will ultimately be a trend toward use of axial compressors because of their inherent performance advantages. This chapter, which is concerned mainly with axial-flow units, points out sources of compressor failure and suggests methods of improving compressor reliability. Failure statistics will also be discussed.

FAILURE STATISTICS

Study of failure statistics of military units over the first 6 months of 1954 reveals that 26 to 59 percent (depending on engine model and application) of premature engine overhauls were due to foreign-object damage in the compressor (ch. II). The ingestion of foreign objects may cause a direct impact failure but more frequently will produce nicks, gouges, and dents in both rotor and stator blades. The immediate effect in some cases is impaired efficiency, but the more serious effects are points of stress concentration, which may with time result in fatigue failure.

Statistics on compressor failures other than foreign-object damage are relatively ineffective in determining the true severity of the compressor reliability problem. For example, available statistics show that 6.5 percent of one engine type were removed prematurely from aircraft primarily because of compressor difficulties. In another engine type, 8.8 percent of the engine removals were attributed to the compressor. However, neither the Engine Removal Reports (ER's) nor the Disassembly Inspection Reports (DIR's) used for statistical failure data include any of the fatigue failures where blades actually separated from their attachments. Since a broken compressor blade is generally carried through succeeding compressor stages and starts a chain reaction of blade failures, a single blade failure may virtually destroy the entire engine, and frequently the aircraft. Consequently, these results are not listed in the DIR's, since the engine is not repairable and must be scrapped. A compressor failure in which a single blade failed in the first stage is shown in figure 1. The damage

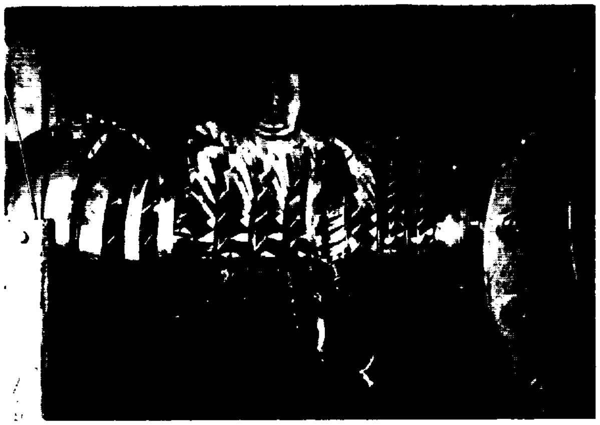


FIGURE 1.—Typical axial-flow-compressor failure resulting from fatigue of one first-stage blade.

accumulated until it burst through the casing at the seventh stage. The damage resulting from a fatigue failure in a blade root is shown in figure 2. Six other blades in the same stage had extensive fatigue cracks in the roots and fractured upon impact. The remains of a more disastrous compressor failure are shown in figure 3. Part of the compressor went through the roof of the building in which the unit was tested. The row of blades visible in the picture is that of the last rotor stage. The blades of the second-last rotor disk were stripped off. These failures occurred in test stands at the NACA laboratories.

FIGURE 2.—Damage resulting from fatigue of blade root in second rotor stage.

A limited amount of compressor statistics can be obtained from accident reports. In many cases, however, the damage is so extensive that the original cause cannot be traced and thus is listed as unknown. Accident reports for 1953, which include 205 accidents caused by jet-engine failure or malfunction, show the compressor to be responsible for 26.8 percent of the accidents. In almost half of these cases the airplane was also destroyed.

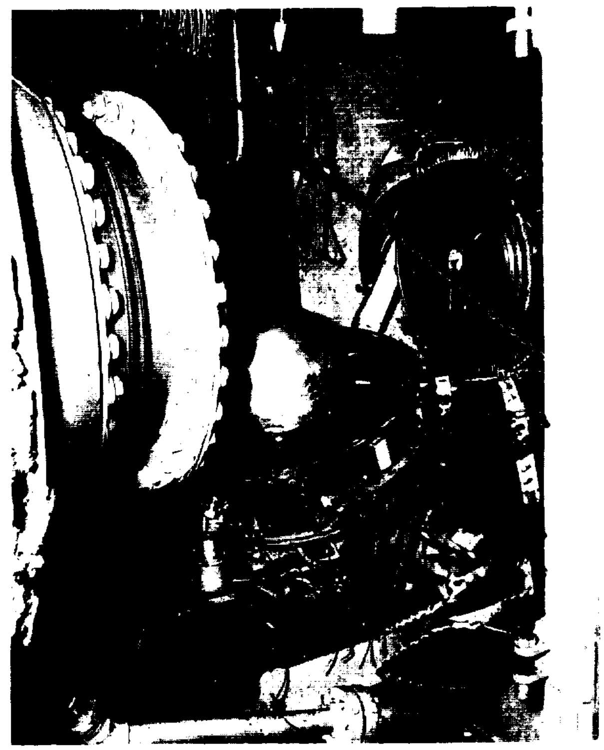


FIGURE 3.—Axial-flow-compressor failure.

Another shortcoming of the DIR's, ER's, and the accident reports utilized in this study is that the reports analyzed cover only a limited number of Air Force engines in active use. Some of these use centrifugal compressors and therefore do not reveal the axial-flow compressor problem. Other engines omitted from the statistical study, including Navy engines and newer engines with limited service experience, have had considerable difficulty with the compressors. Engines of high compressor pressure ratio are especially subject to durability problems. Many engines not covered in the study have been grounded at various times as a result of excessive fatigue failure in the compressor.

Because the high percentage of engines damaged by foreign objects was manifested mainly in the compressor, and because fatigue failures in compressors were so disastrous, it is apparent that the weakest single item in the jet engine is the rotor blading of the axial-flow compressor. The centrifugal compressor is far more reliable than the axial-flow unit. Of the 55 failed compressors mentioned in the 1953 accident records, only one was a centrifugal compressor. DIR's and ER's show that centrifugal compressors have almost as many failures as axial-flow units, but these failures are only the repairable type.

DISCUSSION OF FAILURE MECHANISMS

Very few compressor failures have occurred as a result of direct stress imposed by centrifugal or steady aerodynamic loading. The centrifugal stress varies with the stage (fig. 4). Even the highest centrifugal stress of 36,000 pounds per square inch (psi) in the first stage at the rated speed of the engine is only a fraction of the strength available from the compressor blade material. The bending stresses due to the steady-state gas forces at rated speed are only 15 to 25 percent of the centrifugal stresses. Obviously, other factors are primarily responsible for the failures.

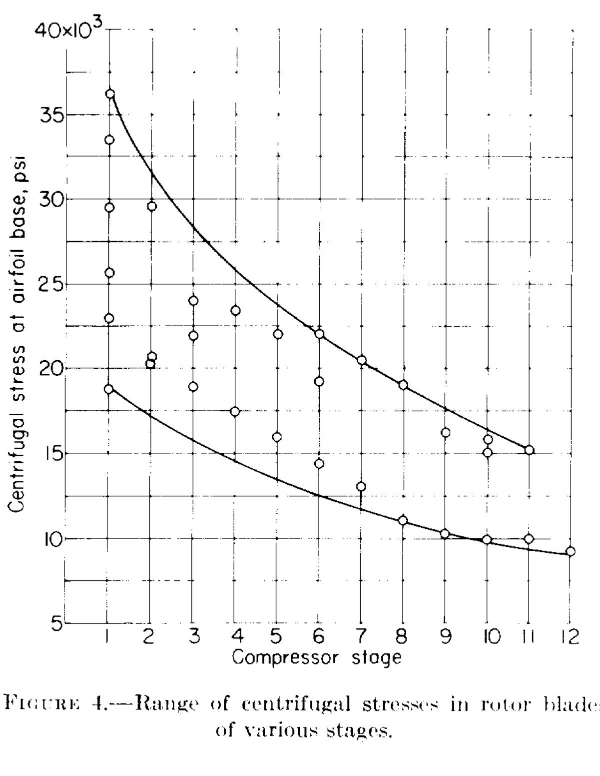


FIGURE 4.—Range of centrifugal stresses in rotor blades of various stages.

PRINCIPAL TYPES OF FAILURES

The principal causes of compressor failures, in order of frequency of occurrence, are: (1) foreign-object damage, (2) vibration fatigue, (3) stress-corrosion cracking, and (4) blade-surface erosion.

Foreign-object damage.—The axial-flow-compressor rotor blades are the first moving objects encountered by foreign matter entering a jet engine. Because every other row of blades in a compressor is stationary and immediately obstructs any motion imparted to foreign particles by the intervening rotor blades, damage is caused to a large percentage of the total number of blades in a multistage compressor. Also, the cantilever-type attachment and the thin leading and trailing edges of the airfoils make the axial-flow units extremely susceptible to foreign-object damage (fig. 5).

With the conventional blade materials used, the direct impact of a foreign object seldom causes the blade to break, unless the object is unusually large. The damage consists mostly of small gouges and nicks in the airfoil (fig. 6). These nicks in themselves are not disastrous, but they multiply the vibratory stresses normally present and thus induce or greatly accelerate

FIGURE 5.—Damage resulting from foreign object entering inlet.

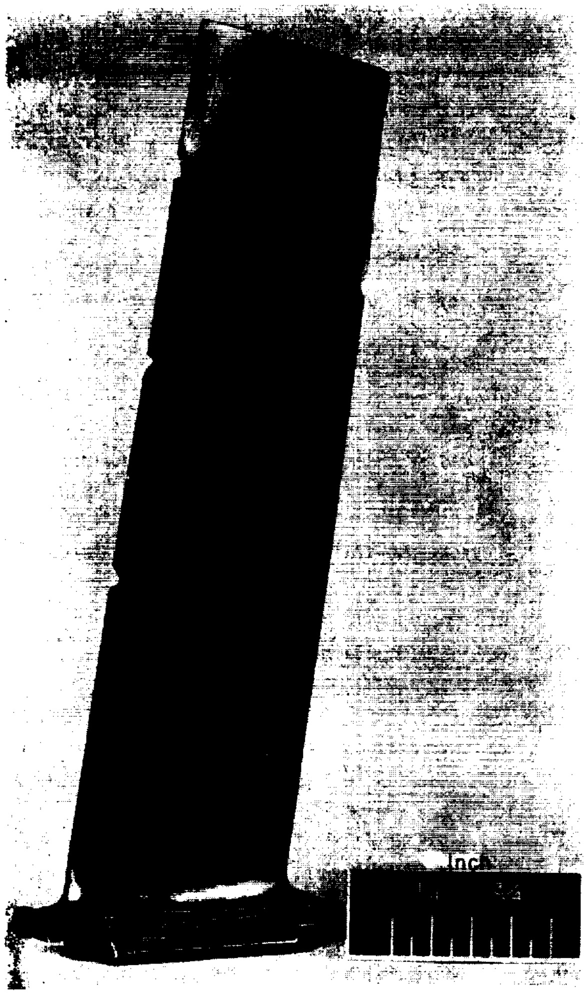


FIGURE 6.—Gouges in compressor blade caused by foreign object.

fatigue failures. Therefore, the nicks near the base of the airfoil or at the sharp leading or trailing edge are of greater concern than damage near the tips of the airfoils. Engine manufacturers have set up arbitrary limits on the number, size, and location of such nicks (ref. 2).

Foreign-object damage is a serious problem because of the large numbers of blades involved and the expense of disassembly, blade replacement, and compressor rebalancing. The number of rotor blades per stage varies from 17 to over 130 (fig. 7), the total per compressor averaging about 800 rotor blades. Occasionally, as many as 75 percent require replacement. Figure 8 shows the typical result of a foreign object passing through a compressor. The number of blades hit in the first eight stages was relatively constant.

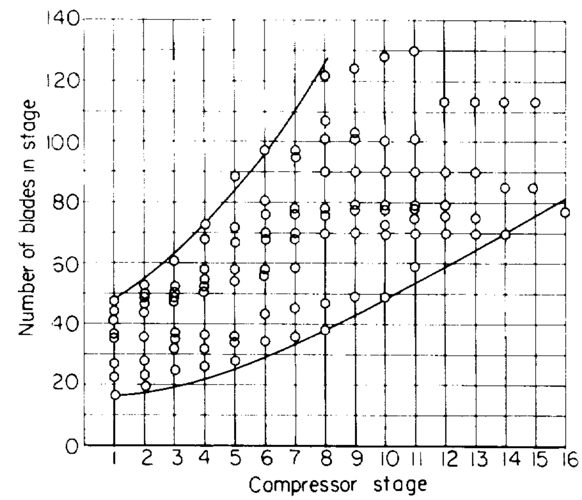


FIGURE 7.—Range of number of rotor blades in each stage of conventional jet-engine compressor.

Because the number of blades per stage increases toward the rear of the compressor, the first stage had the highest percentage of damage. The peaks in the curves indicate that the foreign object lingered slightly in the tenth stage. The values do not include stator blades, which are not nearly as critical because of the absence of centrifugal loads.

Vibration fatigue.—Fatigue usually occurs in the rotor blades, but is occasionally experienced in the stator blades or rotor disks at the blade recesses because of vibratory loads from the blades. The blade vibrations are generally induced by pulsations in the airflow. In most cases of blade failure, the fractures occur in the airfoil section near the base and show progressive damage typical of pure fatigue (fig. 9(b)). The origin of

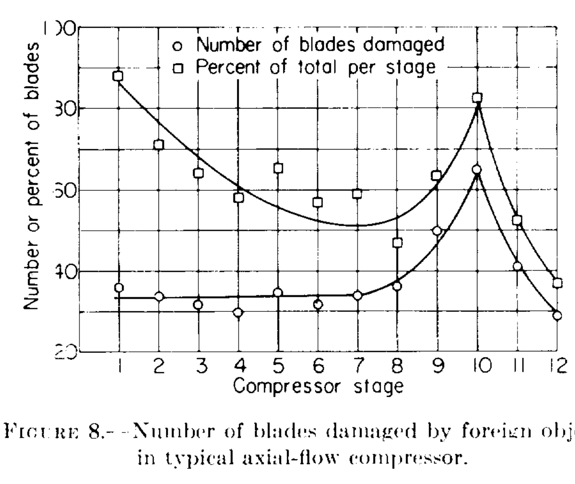
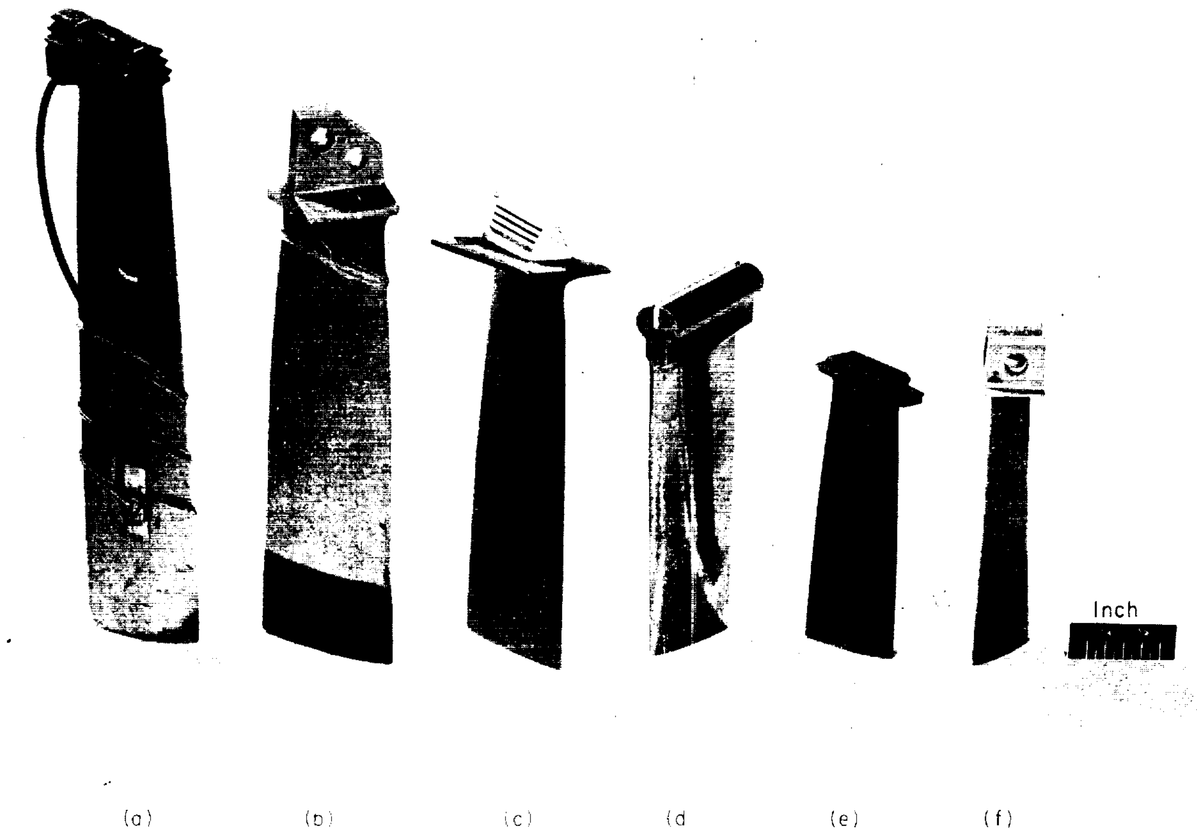


FIGURE 8.—Number of blades damaged by foreign object in typical axial-flow compressor.

(a) Failed blade.
(b) Cross section AA from (a).
(c) Cross section CC from (a).
(d) Foreign object damage.

FIGURE 9.—Initiation points of fatigue in typical compressor blade.



(a) Fir tree. (b) Riveted. (c) Fir tree. (d) Ball root. (e) Dovetail. (f) Loose pin.

FIGURE 10.—Types of blade root fastenings.

the failure is usually on the convex surface at the maximum camber point (point O), where the bending stress is maximum during vibration. Fatigue failures may also originate in the trailing edge of the airfoil (point P). In other cases, the failure is in the fastening (fig. 9(c)). Frequently fatigue failures originate at random locations on the airfoil where stresses are not ordinarily high but are magnified by stress concentrations arising from toolmarks or nicks made by foreign objects (fig. 9(d)). Six types of fastenings that have been used for compressors are shown in figure 10. In the fir-tree types (figs. 10(a) and (c)), high stress concentrations occur in the fillet between serrations, paticularly if the radii are small. The bending stresses due to vibration are superimposed on the steady centrifugal and gas-bending stresses. All stresses are magnified by the stress concentration caused by the abrupt changes in cross section

Vibratory stresses of $\pm 30{,}000$ to $\pm 40{,}000$ psi have commonly been measured in conventional jet engines (refs. 3 and 4), while vibratory stresses as high as $\pm 80{,}000$ psi have been observed in experimental units. Figure 11 is a modified fatigue diagram, a plot of vibratory stress against number of cycles to produce failure for a typical blade material. The horizontal portion of the curve establishes the endurance limit, the vibratory stress that can be endured indefinitely without causing failure. This endurance limit is lower than that normally presented for this material, but the curve has been modified to include the effect of the steady stresses imposed on a rotor blade at the speed at which blade vibrations are most prevalent. The high vibratory stresses generally occur at 50 to 70 percent of rated speed. The time required to produce failure can be readily determined from the blade natural fre-

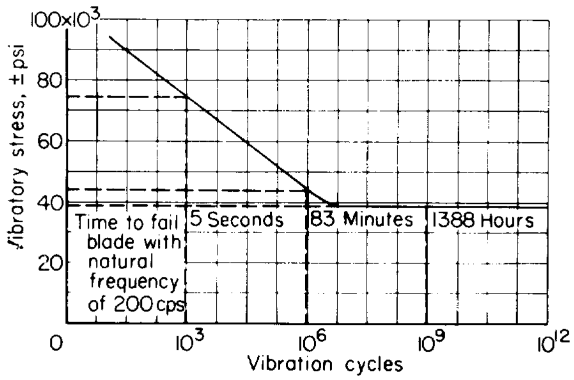
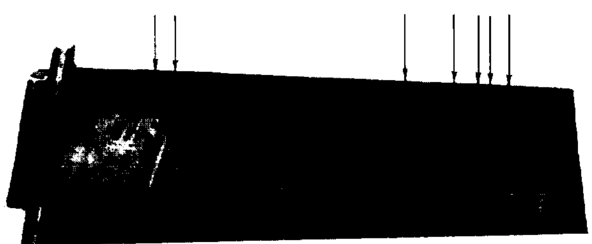


FIGURE 11.—Modified fatigue diagram for typical compressor-blade material.

quency and its vibratory stress. The causes of fatigue are discussed in more detail in the section NATURE OF FATIGUE PROBLEM.

Stress-corrosion cracking.—A difficulty called stress-corrosion cracking has recently been encountered with the 12-percent-chromium iron alloy currently being used for nearly all compressor blades (refs. 5 to 7). Although the actual mechanism for this process has not been completely established, the cracks (fig. 12) are known to appear under the following circumstances.



(a)

(b)

(a) Location of cracks.
(b) Photomicrograph of crack.

FIGURE 12.—Typical blade cracked by stress corrosion.

As a result of the strengthening heat treatment and quenching at 1750° F, residual stresses due to fabrication technique are frozen into the material according to the theory outlined in references 5 and 6. The residual stresses tend to assist moisture in the air and particularly in salty atmospheres in penetrating and reacting with the grain boundary material (fig. 12(b)). Another theory suggests that the problem is simply one of hydrogen embrittlement and is independent of residual or operating stresses (ref. 7). In either case, because of extreme thinness, the trailing edges of the blades are most vulnerable to cracking.

During fabrication, these blades are given several heat treatments. The first, which occurs after the blade has been formed, is intended to produce an optimum microstructure and relieve residual stresses due to forging. The blade is then finished and the edges polished. A second heat treatment is applied to relieve residual stresses introduced during the finishing operation. The properties of the blade, and in particular its resistance to stress corrosion, depend on the temperatures used in these heat treatments. The practice of one manufacturer in applying the first heat treatment is to solution-treat the blade at 1750° F and blast-air-cool it. The tempering treatment following the quenching is at 1000° F for approximately 2 hours. This results in high Rockwell hardness values of C–32 to –38, which give high tensile and endurance strength (fig. 13); but such blades have had stress-corrosion problems. Another manufacturer also uses a solution-treatment temperature of 1750° F, but tempers at a higher temperature of 1140° F for 2 hours during the first heat treatment with a resultant softening to a Rockwell hardness level of C–20 to –26. The final heat treatment in both cases is a stress-relief anneal at 950° F during which the hardness is not changed. Although the softness of blades made by the second method results in lower tensile strength and endurance limit (fig. 13), improved resistance to stress-corrosion cracking, as well as other improved properties such as higher internal-damping capacity and higher impact resistance (ref. 6), accompany the lower hardness. In a study by this manufacturer of 109 compressor failures due to stress corrosion, only one compressor contained

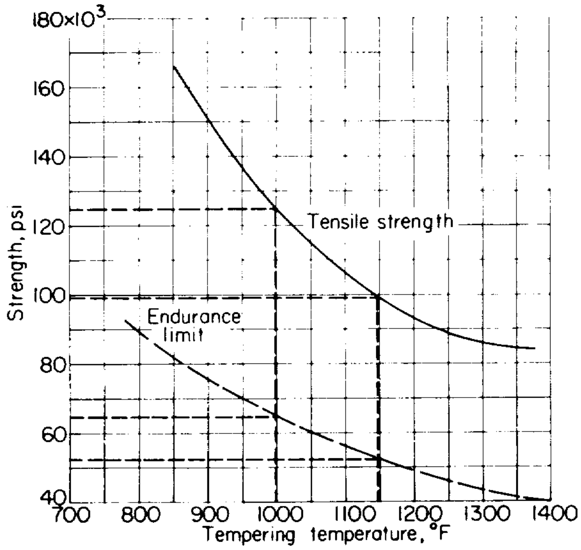


FIGURE 13.—Effect of tempering temperature on tensile strength and endurance limit of 403 stainless steel.

soft blades, while 101 contained only hard blades. The other seven compressors had both hard and soft blades.

The stress-corrosion cracks themselves are not very serious, but in the presence of vibration, the crack is propagated until ultimately the blade fails from fatigue.

Blade-surface erosion.—Erosion may be due to dust, sand, water, or similar particles. British conversion from aluminum to steel blading was due to the common occurrence of erosion of aluminum blades. Erosion effects are gradual and readily detectable during inspection, so that any blades in an aircraft suffering from erosion would probably be withdrawn from service before disintegration occurred.

NATURE OF FATIGUE PROBLEM

Fatigue rsults from alternating stresses produced by blade vibrations. Listed in order of importance, factors that may induce these vibrations are: (1) rotating stall, (2) aeroelastic coupling, (3) obstructions, and (4) transmission through mountings.

Rotating stall.—Rotating stall occurs at speeds considerably below rated speed because of aerodynamic complications arising from operating the compressor at speeds other than design speed. The axial-flow compressor is designed to operate under a given set of conditions, and the design point is usually the rated speed of the engine. Thus, all the angles of the various blades are set in their proper relative orientation for this operating condition. When the engine operates at any condition other than its design point, the blades in each stage are improperly arranged relative to the blades of adjacent stages.

The effect of part-speed operation of a multistage engine is shown in figure 14. For proper passage of the air during this part-speed condition, the hub contour for optimum stage matching is that shown by the dotted line, whereas the machine, which was designed for full-speed operation, actually has the solid-line contour. The passage in the rear part of the compressor is not large enough to handle the air induced by the compressor. Thus, the rear stages tend to choke and restrict the airflow through the compressor. Because of the reduced axial velocity of the air, the front stages operate at a higher angle of attack, thereby being brought close to the stall conditions.

Figure 15 shows in detail what happens in one of the stages when it approaches stall. The angle α is the angle between the chord of the airfoil and the air velocity. As the angle is increased the lift increases, but beyond a certain critical angle the airflow separates from the airfoil and the lift decreases. This is blade stalling.

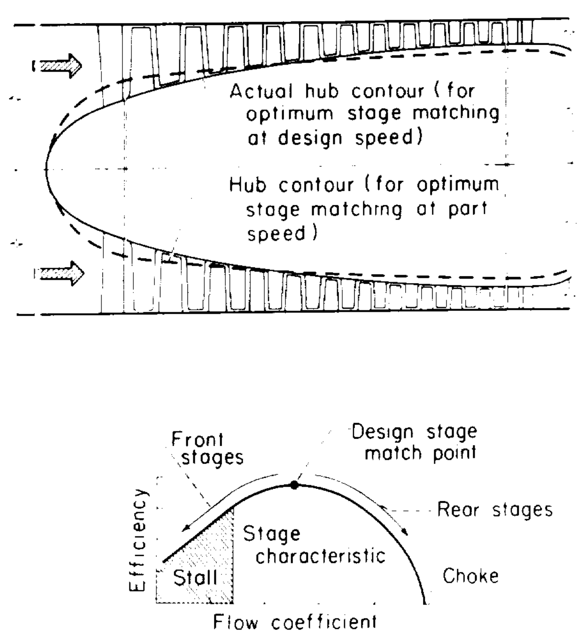
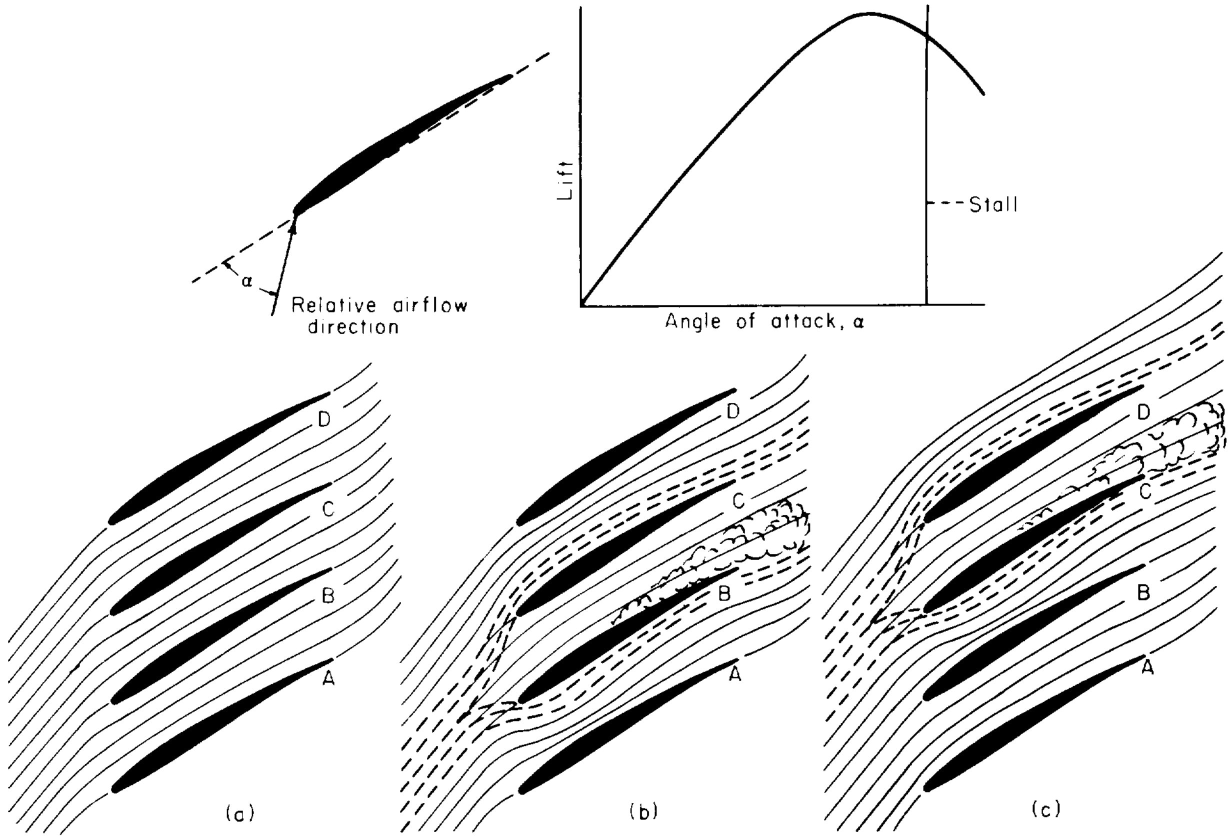


FIGURE 14.—Desired compressor profile for full-speed and part-speed operation and effect on efficiency.



(a) Unstalled rotor row. (b) Blade B stalled. (c) Blade C stalled.

FIGURE 15. Sketches illustrating progression of rotating stall.

Smooth airflow associated with operation below the stall point is shown in figure 15(a). Each of the many blades of a stage approaches closer to stall, but there is always one blade (e.g., B) which will tend to stall before the others because of manufacturing differences. When this blade becomes stalled the adjacent passage can no longer pass its normal quantity of air, and the air normally using the passage BC must be diverted to adjacent passages, as shown by the dotted lines (fig. 15(b)). The additional air forced into passage CD approaches blade C at a higher than normal angle of attack and stalls it. The air diverted into passage AB strikes blade B at a lower angle of attack and unstalls it. Thus, the stall region has moved from BC to CD (fig. 15(c)). Similarly, the stall of blade C diverts air to the next passage and blade D stalls and so forth. In this way, the low-flow-velocity region is caused to move successively from one passage to the next. This is called rotating stall.

By means of hot-wire anemometers, rotating-stall patterns have been measured and examples are shown in figure 16. As many as eight equally spaced low-velocity zones have been detected simultaneously in a compressor annulus (ref. 8),

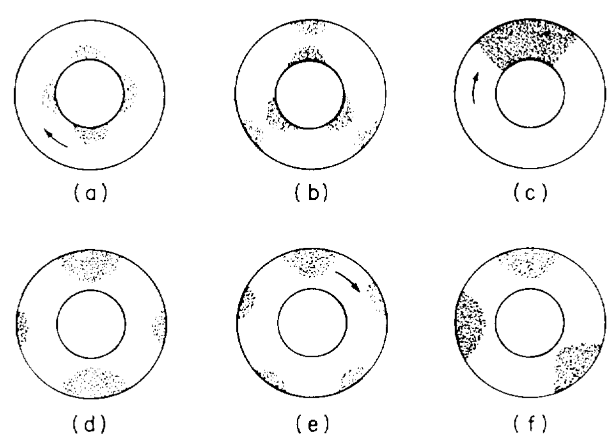


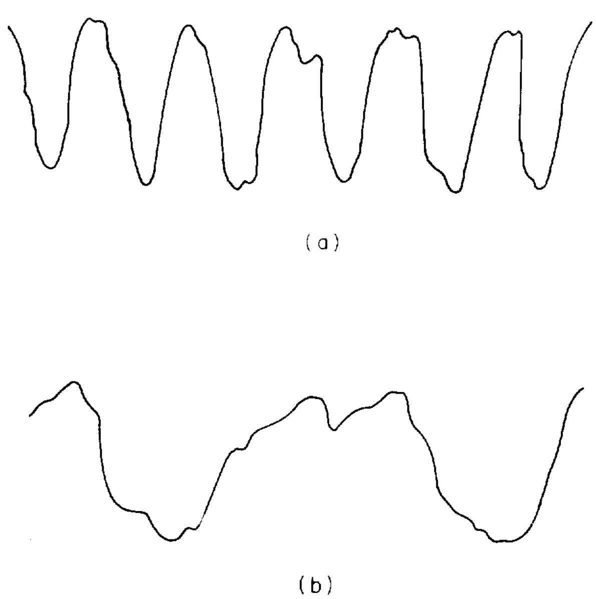
FIGURE 16.—Some rotating-stall patterns detected by hot-wire anemometers.

but generally, the number of stalled zones is from three to six (refs. 4, 9, and 10). The zones in a given pattern may be of different intensity and sometimes are unequally spaced.

Typical signals from hot-wire anemometers (ref. 11) used to detect low-velocity zones are shown in figure 17. Because the signal is not of a sinusoidal-wave form, it contains many harmonics which can induce blade vibrations. Many potential critical speeds are possible in any axial-flow compressor because of the multitude of natural frequencies of the different stage blades in an axial-flow compressor, the many stall patterns which have been observed, and the harmonics of each pattern.

Figure 18 shows a hypothetical critical-speed diagram. The intersection of any of the natural frequency lines with the exciting frequency lines for both the various stall patterns and their harmonics constitutes a possible condition of blade vibration. The magnitude of the vibratory stress depends on intensity of the harmonic causing the vibration, the total damping in the system, and the proximity of the exciting frequency to the blade natural frequency.

Rotating-stall patterns have induced very severe blade vibrations. The catastrophic compressor failure shown in figure 2 was caused by rotating stall. Fortunately, rotating stall of a strong periodic nature occurs only between 45 and 70 percent of rated speed. The jet engine is seldom subjected to this speed range for long periods of operation, as the range is traversed only upon acceleration and deceleration. Nevertheless, rotating stall constitutes the primary source of fatigue failure.



(a) Signal for three stall zones.
(b) Signal for one stall zone.

FIGURE 17.—Typical hot-wire-anemometer signal for rotating stall.

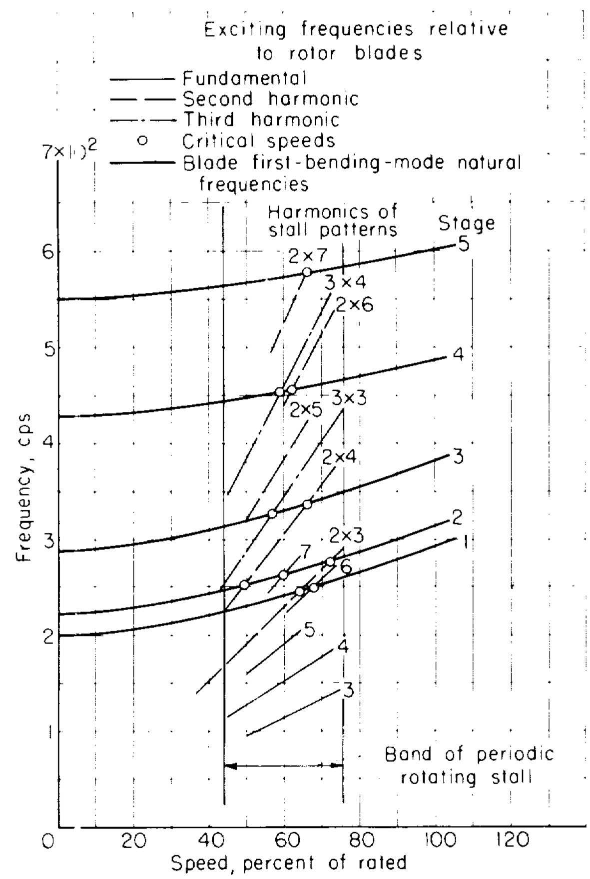


FIGURE 18.—Hypothetical critical-speed diagram for axial-flow compressor.

The number of stalled zones constituting a stall pattern depends on many factors. A hypothetical compressor performance map with the speed ranges through which specific numbers of stalled zones occur simultaneously is sketched in figure 19. Line AA is a typical steady-state operating line. This line shifts upward if the jet-nozzle area is decreased and downward when the area is increased, thus varying the speed range over which a particular rotating-stall pattern is present. Line BB is a typical path followed when the engine is accelerated. This line is also vari-

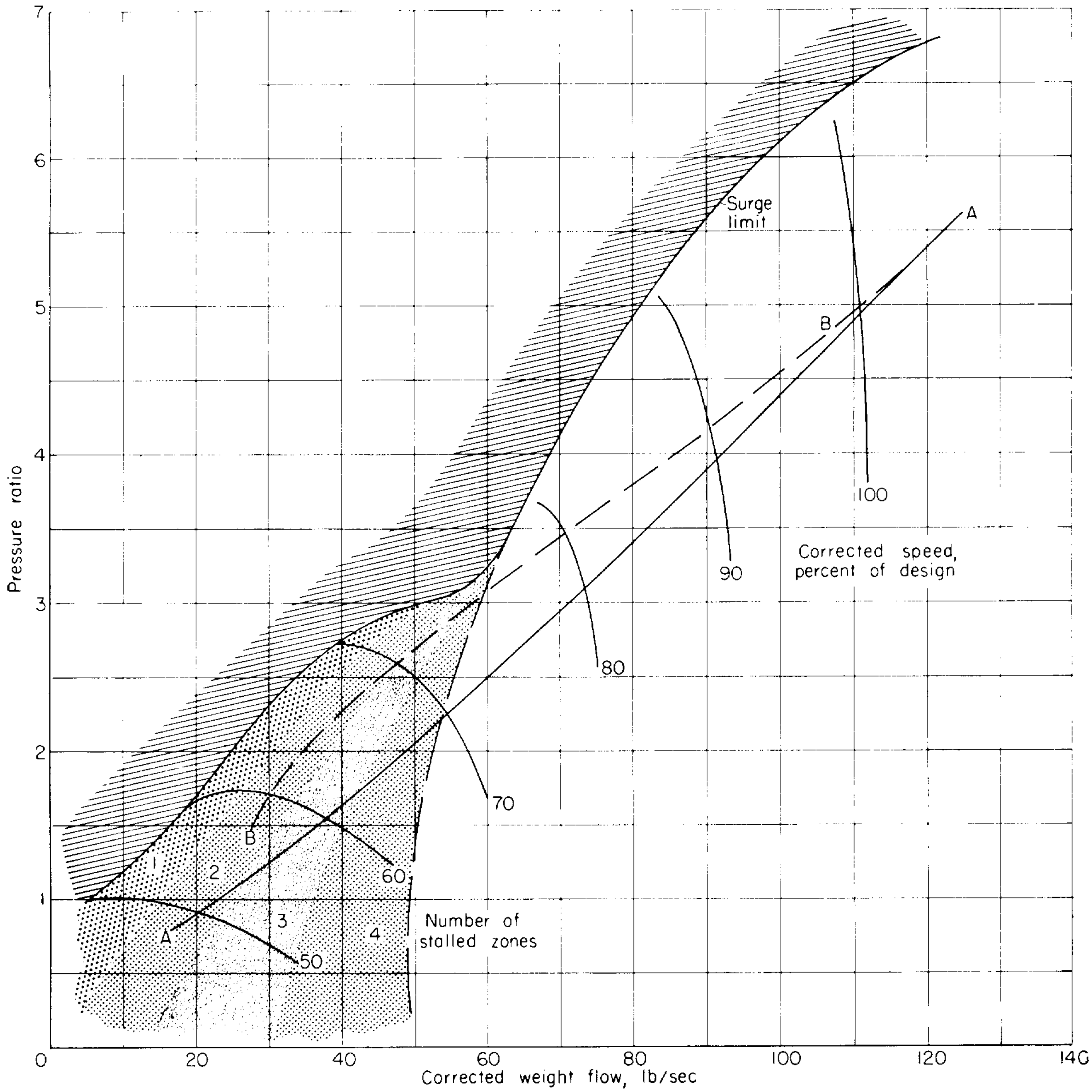


FIGURE 19. Hypothetical performance map for compressor showing regions of stall.

able, depending on the rate of acceleration. Deceleration will produce still another family of curves below the steady-state line AA. The speeds marked on the graph are corrected to standard atmospheric conditions. Inlet temperatures lower than standard would shift all the corrected-speed lines on the map to the right; higher inlet temperatures would shift them to the left. Thus, on one day a severe vibration may be observed in an engine; the next day the same vibration cannot be reproduced in the same engine because of a change in ambient temperature. One engine of a certain model may vibrate because of rotating stall, while another of the same model may never vibrate because it has a slight different tailcone or nozzle-diaphragm area. One engine may vibrate because of resonance between the stall and blade frequency at a certain speed. In another engine, rotating stall may cease, or a different stall pattern, and consequently a different stall frequency, may occur at this speed. Thus resonance is not produced, even though the blade frequencies may be the same. Therefore, many factors are involved in the development of a vibration condition.

The stall-pattern regions are never as sharply defined as in figure 19, because there is considerable overlap between these regions. During operation at any given speed near the boundaries, the stall pattern may change instantaneously from one configuration to another and back again and, under certain conditions, even to a third or fourth configuration. Figure 20 is a continuous anemometer record at a constant engine speed. Initially, a single-stall zone was present. Next, a four-stall pattern took place with two weak and two strong zones, followed by a five-zone pattern with only one strong zone. Finally, three zones of equal velocity variation were established in the annulus. The light vertical lines represent time increments of 0.01 second, showing how rapidly these changes can take place.

Aeroelastic coupling.—The interaction of aerodynamic forces and the elastic behavior of the blading can result in an instability which produces blade vibrations. Flutter is an instability commonly occurring in airplane wings (ref. 12) and has been observed in stationary cascades of untapered, untwisted airfoils simulating compressor blades (ref. 13). In compressors, the

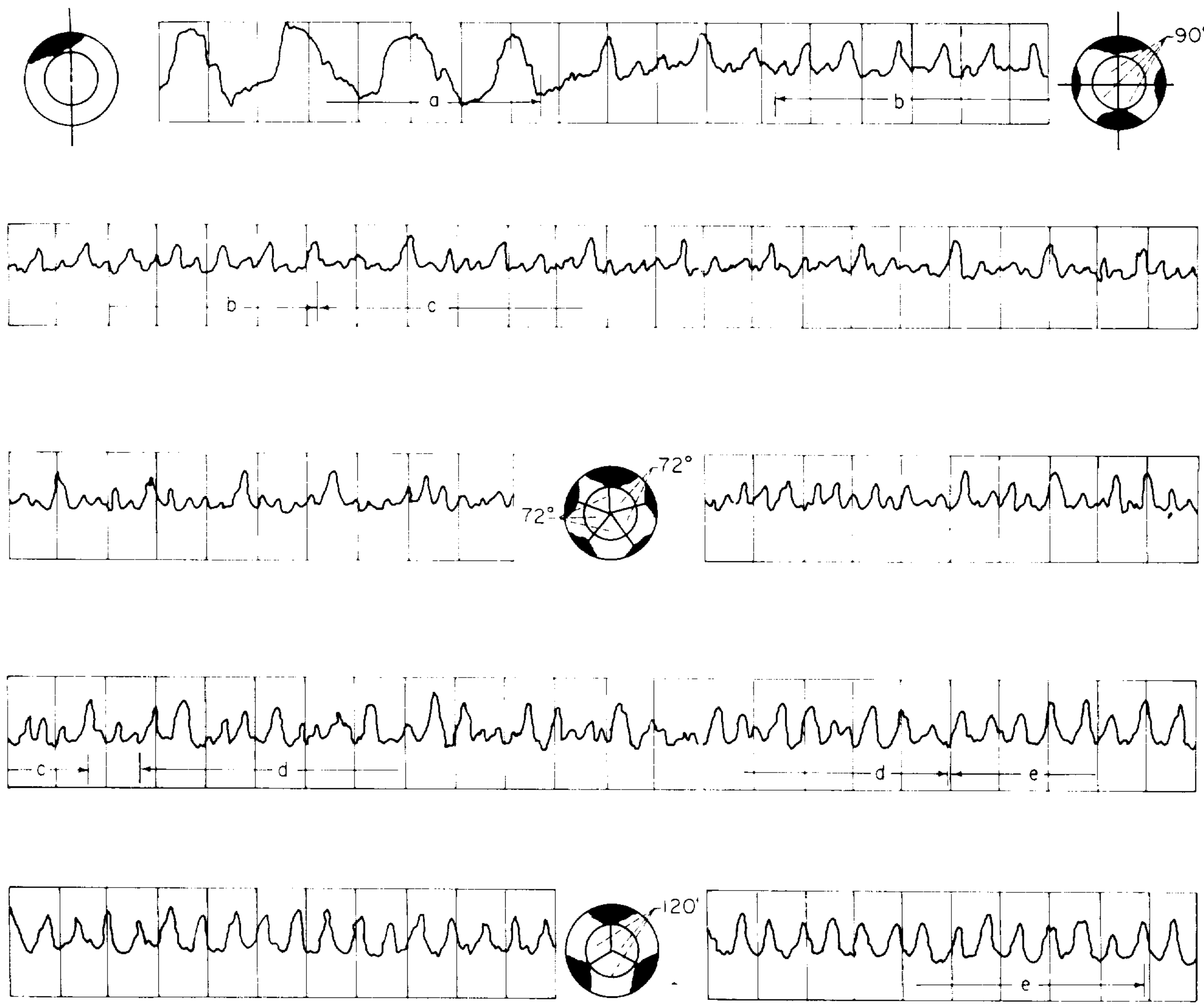


FIGURE 20.—Tracing of oscillogram of continuous anemometer signal at constant engine speed. (Durations of various stall patterns a, b, c, d, e indicated by arrows.)

problem may be aggravated by high aerodynamic loading, proximity to stall, pressure rise across the stage, and certain cascade effects (refs. 14 and 15). Such excitation occurring in the vicinity of stall is known as stall flutter, which is entirely different from rotating stall. Rotating stall is a periodic nonuniformity in the airflow that only excites blade vibrations when the impulse frequency approaches the blade natural frequency. Stall flutter is self-induced vibrations of the blade in a uniform flow field due to aerodynamic instabilities arising from high angles of attack. Although considerable research has been directed toward stall flutter in compressors, no clear-cut experimental evidence of its occurrence in operating units has been observed. The British, however, are strong advocates of the existence of stall flutter in compressors based on their many stationary cascade tests.

Obstructions.—Vibration excitations occasionally are produced by bearing supports, stationary blades, bleed ports, and measuring probes (ref. 15). Wakes shed from these obstacles strike the rotor blades as they revolve and constitute a periodic exciting force. Strain-gage investigations usually detect some vibrations due to the obstructions, but the amplitudes are seldom large enough to cause difficulty.

Transmission through mountings.—Transmission of periodic forces through the blade mounting from accessory-drive and propeller-reduction gearing, pulsations from the various pumps, and cycling from the combustion process are all possible causes of fatigue. However, these forms of excitation never have been encountered to a serious degree and are included only for completeness. Unless jet engines become drastically different from present units, the problem is not expected to be of serious concern.

FACTORS AFFECTING STRESS AMPLITUDE

The stress amplitude developed depends on the magnitude of the exciting force, on the relation of the exciting frequency to the natural frequency of the system, and on the damping tending to suppress the vibration. The first factor is self-evident; the greater the exciting force, the higher the vibratory stress amplitude. When harmonics are involved, the magnitude of the exciting force is dependent on the magnitude of the harmonic energy content present in the initial impulse. For example, an engine may have both a single- and five-stall pattern at nearly the same speed. Generally, the greater the number of stalls in the annulus, the weaker the wake impulses. However, in most cases, the higher the harmonic, the lower the energy content of that harmonic in the original impulse. The harmonic content generally decreases more rapidly for the higher harmonics than does the energy in the stalled zones, as the number of stall zones in the annulus increases. In other words, the vibration amplitude caused by the fundamental of the five simultaneous stall zones will be much higher than the vibration amplitude from the fifth harmonic (having approximately the same exciting frequency) of a strong single-zone pattern.

The second factor affecting stress amplitude is best illustrated by figure 21. One of the abscissa scales of this figure is the exciting frequency; the other is engine speed. The stress amplitude is maximum when the exciting frequency is equal to the natural frequency of the blade. The illustrated resonance curve is typical in shape for a conventional compressor blade. Operation at only 100 rpm off the resonance point will cut the vibration amplitude to half the resonant value.

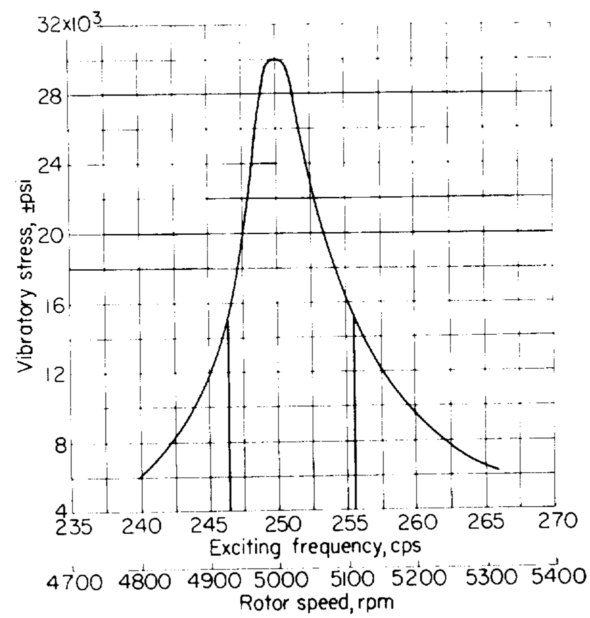


FIGURE 21.—Typical resonance curve.

In some cases of rotating stall, true resonance may not occur because the stall condition may vanish before the stall frequency equals the blade natural frequency.

The third factor affecting stress amplitude is damping, which actually controls the peak value of the resonance curve (fig. 21). The damping of a vibratory blade can be separated into three parts: (1) material damping (sometimes called hysteresis or internal damping), (2) root or mounting damping, and (3) aerodynamic damping. Material damping is the internal friction developed in the blade material when it is stressed. Each material has its specific value (logarithmic damping decrement, table I) and there appears to be no consistent correlation with other common mechanical or physical properties of the material. Damping is affected by stress, heat treatment, temperature, and frequency of vibration (refs. 16 to 19).

Root damping is the result of external friction arising from the rubbing between the mating rotor and blade root surfaces or surfaces especially provided for this purpose and is dependent on root geometry and centrifugal load of the blade (fig. 22). The root damping can be increased and extended over a considerably higher centrifugal load or speed range by providing lubrication of the root (ref. 20).

Whenever a blade vibrates in a moving airstream, part of the vibration energy of the blade is imparted to the airstream. The amount of energy dissipated in this way is a function of air velocity, air density, density of the blade material, and blade geometry. Aerodynamic damping can be computed by the procedure used in reference 15 or 21. The blade amplitude then depends on the amount of vibration energy absorbed by the air, the blade material, and friction in the mounting instead of being spent in shaking the blade.

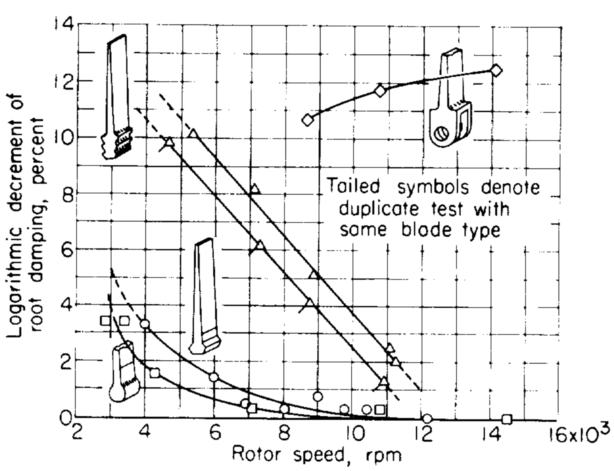


FIGURE 22.—Damping of various blade root designs as affected by rotational speed.

MITIGATION OF THE FAILURE PROBLEM

IMPROVED COMPRESSOR DESIGN

The failure problem can be greatly minimized if the designer anticipates vibrations when proportioning the blade and its attachment. Experience has shown only first-bending-mode vibrations to be of major concern.

Design of the blade and attachment.—The designer should provide generous fillets at the base of the airfoil, keep the stress concentrations in the root as low as possible, and avoid overhanging any parts of the airfoil from the base. The blade roots should be designed to allow easy inspection of the critical points and easy replacement in case of foreign-object damage before fatigue produces ultimate failure. In the design stage, special roots and devices can be incorporated to increase the total damping in the system. By introducing external-friction damping, the stress amplitude may be diminished even if the exciting force persists. Blades loosely fitted in their mounts have beneficial effects in minimizing the amplitude built up in a vibrating blade (ref. 21). At the part-speed condition (where rotating stall occurs) there is some benefit derived, but at rated speed the high centrifugal forces increase the friction until no root motion takes place in the root and no vibration energy is dissipated by this means. Introducing a lubricant such as a dry film of molybdenum disulfide MoS_2 reduces the friction to the point where root motion again takes place even under the high centrifugal load. Figure 23 shows the effect of MoS_2 on root damping of a loose blade for various centrifugal loads on a ball-type root mount.

One British engine utilizes a very loose pinned type of blade attachment like the one shown at the upper right-hand side of figure 22 or in figure 10(f). This particular type of blade takes advantage of the fact that serious vibrations in the first bending mode of a fixed cantilever blade

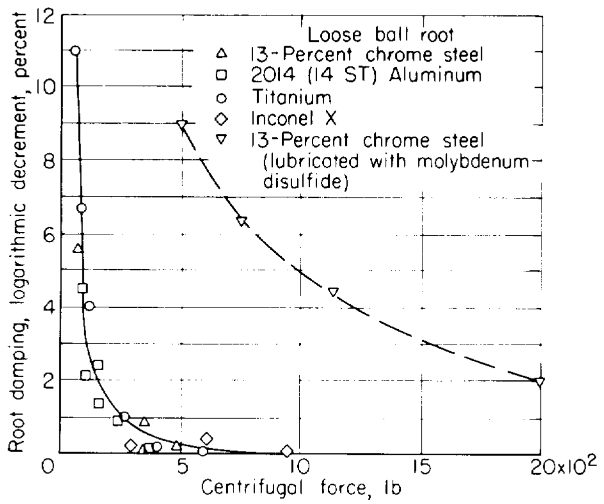


FIGURE 23.—Effect of solid film lubricant molybdenum disulfide MoS_2 on damping of loose blade.

cannot be excited if the pinned arrangement is properly designed (ref. 22). In higher-vibration modes, root stresses are also reduced because the blade is not mechanically restrained from bending. In addition, damping friction is provided by the side faces of the root tongues on the disk and blade. This damping is not reduced by centrifugal force as the speed increases. Another special root that provides supplementary damping is shown in figure 24. In this vibration damper, developed at the Lewis laboratory, pins are retained in radial holes in the rim of the rotor disk. Centrifugal force presses the pins against the blade appendage. Oscillations of the blade cause movement of the pins and produce friction between the mating surfaces of the blade, pins, and disk. In the model investigated, the special root device reduced the stress amplitude to one-tenth the magnitude of the same assembly with the pins removed.

The device shown in figure 25 has been suggested at the British National Gas Turbine Establishment as a means of adding internal damping by design and fabrication. The blade is split to induce supplementary friction, the two parts being held together by high-damping elastic cements. Vibration causes the mating surfaces to move relative to each other, thereby inducing friction and limiting the stress amplitude. This device has never been evaluated in a full-scale compressor.

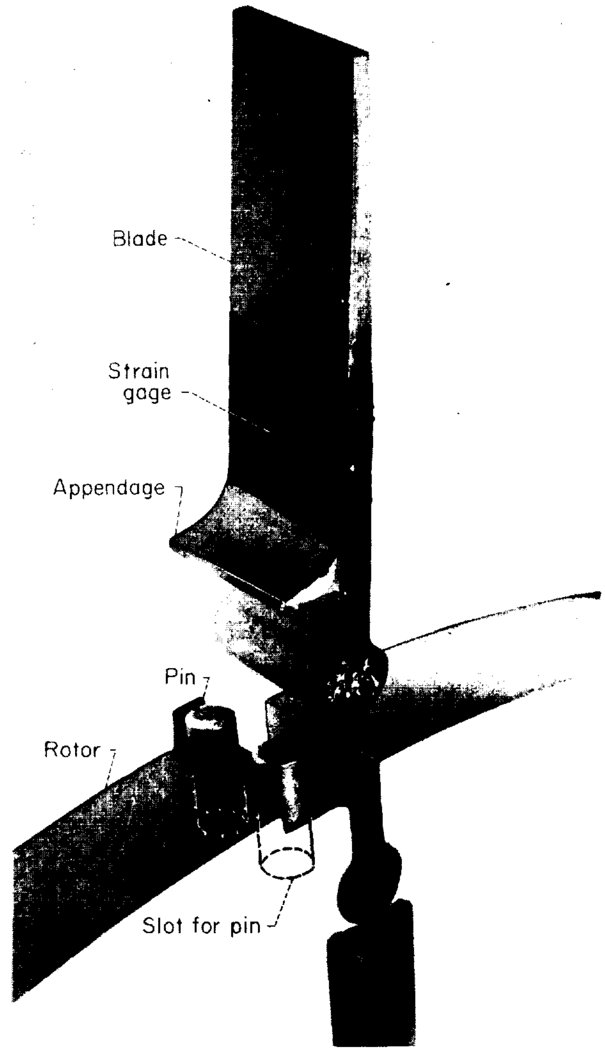
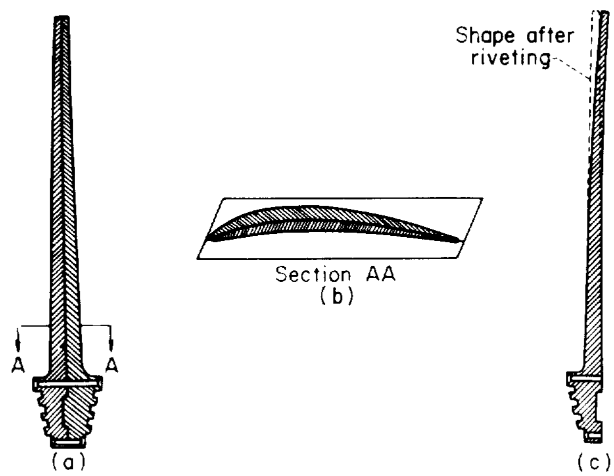


FIGURE 24.—Special device providing additional external damping.

Excitation by obstructions.—In addition to minimizing amplitude by adding damping and improving the resistance of the blades to fatigue, the designer should reduce the exciting forces likely to cause vibration. Obstructions to smooth airflow should be minimized. When such obstructions are necessary, such as in the case of front bearing supports, a vibration analysis should be made to ensure that the frequency of excitations will not coincide with the natural frequency of the blade at likely speeds of operation. Streamlining, unequal spacing, and placing the obstructions as far ahead of the blading as possible in a relatively low velocity region will help minimize the excitation. The effect of



(a) Assembled blade.

(b) Airfoil cross section.

(c) Prestrained blade half.

FIGURE 25.—Means of increasing internal damping of compressor blade.

artificially introduced wakes from obstructions on the vibration amplitude of compressor blades is described in reference 23.

Excitation due to rotating stall.—Through extensive investigation of rotating stall, the principles involved and the exciting forces produced are beginning to be understood (ref. 24). Means of eliminating the effects of rotating stall are becoming evident (ref. 25). For example, blocking off some of the incoming air at part speed by baffles, or orifice plates so that the intake will not exceed the air-handling capacity of later stages has proved successful in some engines. The vibration amplitudes with and without such baffles are shown in figure 26. Without baffles, a strong three-stall pattern was present which produced stresses as great as ±26,000 psi. With the hub-type baffle a turbulence was created around the hub, but the stress was reduced to ±5000 psi. The baffle, ramp, or similar blocking device must be retracted at full speed of the compressor. Reference 26 reports the improvement in vibration conditions and corresponding loss in performance of a whole series of baffles over the entire speed range. In this investigation, a baffle blocking only 5 percent of the inlet area broke up the periodic nature of rotating stall, and the maximum stress was reduced from ±60,000 psi to ±20,000 psi. The loss in compressor efficiency resulting from the 5-percent baffle was less than 2 percent, even at rated speed.

The use of variable-angle inlet guide vanes has also been considered as a means of reducing rotating-stall characteristics. In one case studied at the Lewis laboratory, it was found that such guide vanes changed the stall pattern from a tip (figs. 16(d) to (f)) to a hub pattern (fig. 16(a)). This latter condition reduced the exciting energy absorbed by the rotor blades (cantilever mounted, attached at the hub). The stress level with the standard angle setting was ±40,000 psi. With the optimum angle setting for this vibration condition, the vibratory stress in the rotor blades was reduced to ±5000 psi. The variable inlet guide vanes are not as effective as other means of alleviating rotating stall because they readily control only the angle of attack to the first stage. Changes in the other stages are relatively small, however. If the stator blades are of cantilever construction attached to the outside casing, the change of location of the stall zones from tip to hub, as observed in the particular engine investigated, may induce severe stator blade vibrations.

Designing both several front stages of variable stator blades and variable inlet guide vanes should prove far more effective than varying the guide vanes alone. Several advance-design engines use variable stators, which, however, greatly increase the complexity of the engine.

A very effective and less complicated remedy for rotating stall is the addition of interstage bleed at part-rated speed. The bleed diverts

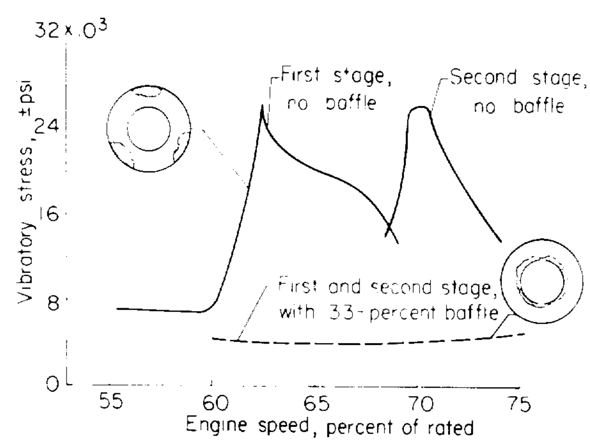


FIGURE 26.—Effect of inlet baffles on vibratory stresses caused by rotating stall.

some of the incoming air from the intermediate and later stages which are unable to pass it on. Bleed provides the added flow area desired at the rear of the compressor (fig. 14). Data illustrating the direct effect of bleed on rotating stall and vibration amplitude have not yet been obtained.

MATERIAL SELECTION

In selecting a material for compressor rotor blades, the tensile strength, fatigue strength, impact strength, density, and damping properties must all be considered. The majority of the rotating compressor blades are made from 403 stainless steel, heat-treated to give the desired tensile and fatigue strength without too great a sacrifice of impact strength and stress-corrosion resistance. The reason for the preferred use of 403 stainless steel can be deduced by comparing the properties of all other feasible materials with those of the 403 stainless steel (table I).

Tensile strength and density.—The centrifugal load is the primary load acting on a rotor blade. The centrifugal stresses in the blade are directly proportional to the material densities for a given blade design made from different materials. The comparative strength of various materials is given by the ratio of the blade-material tensile strength to either the centrifugal stress or the density. This ratio, compared with the same ratio for 403 stainless steel, along with the individual tensile strengths and densities, is listed in table I. The magnesium, aluminum, and titanium alloys appear to be superior to the 403 stainless steel. However, this comparison is based on room-temperature properties, and many of the newer, high-pressure-ratio, high-flight-speed jet engines will have compressor outlet temperatures near or above 700° F. Therefore, the different materials are also compared on the basis of 700° F properties. Only the titanium alloy now appears superior to 403 stainless steel. Obviously, existing plastics are not suitable for operation in the latter stages of high-flight-speed compressors. Materials cannot be evaluated on the basis of tensile strength alone.

Fatigue strength.—The most common cause of compressor failure is fatigue due to vibration. Therefore, an accurate material evaluation must strongly consider fatigue strength. The fatigue strength (endurance limit) or the vibratory stress required to produce failure in one hundred million vibration cycles is also listed in table I. Since the blade operates under the simultaneous action of both centrifugal and vibratory stresses, the effect of the centrifugal stress in reducing the allowable vibratory stress must also be considered (ref. 27). Except for unusual cases, the endurance limit varies directly with static tensile strength; hence, materials of good tensile strength are expected to have good fatigue strength. Materials of high tensile strength and high strength-to-density ratio are ideal for compressor blades, since these two characteristics will improve the allowable margin for vibratory stress. Type 403 stainless steel is good from both these standpoints. Fatigue strength of rotor blades is affected by compressor speed, which increases the mean stress, as shown in figure 27. The modified fatigue curve as corrected for the mean stress was presented earlier (fig. 11).

Material damping.—The damping in a system determines the amplitude, and consequently the vibratory stress, for a given exciting force. A material may have very high fatigue strength, but if the damping is very low, the material may have a much shorter life than one with moderate fatigue strength and high damping for the same exciting force. Material damping is the most reliable of all the sources available. Root damping may vanish because of high centrifugal force or malfunctions of an external damper. Aerodynamic damping may vanish because of

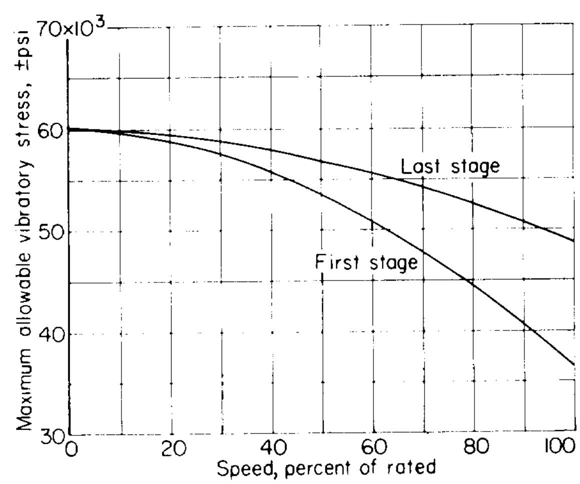


FIGURE 27.—Fatigue strength or allowable vibratory stress as affected by rotor speed.

TABLE I. PROPERTIES OF VARIOUS MATERIALS USED IN COMPRESSOR ROTOR BLADES

Material	Tensile strength at room temperature, S_t, psi	Density, ρ, lb/cu in.	Strength-weight ratio, $\frac{S_t}{\rho}$	Strength (compared with 403 stainless steel), $\frac{S_t/\rho}{S_{t,403}/\rho_{403}}$	Tensile strength at 700° F, $S_{t,700°}$, psi	Strength-weight ratio, $\frac{S_{t,700°}}{\rho}$	Strength (compared with 403 stainless steel at 700° F), $\left(\frac{S_t/\rho}{S_{t,403}/\rho_{403}}\right)_{700°}$	Fatigue strength at 10^8 cycles, S_F, ±psi	Logarithmic damping decrement, δ, percent	Elastic modulus, E, psi/10^6	Frequency factor/10^3, $\sqrt{E/\rho}$
403 Stainless steel	155, 000	0. 280	553, 600	1. 00	90, 000	321, 400	1. 00	75, 000	2. 8	28. 0	10. 0
304 Stainless steel	90, 800	. 286	317, 500	. 57	69, 500	243, 000	. 76	40, 000	. 2	29. 0	10. 1
TP-1 (Metallic powder)	113, 000	. 287	393, 700	. 71	85, 000	296, 200	. 92	25, 000	4. 0	26. 0	9. 5
SAE 4340 (Low alloy steel)	130, 000	. 284	457, 700	. 83	90, 000	316, 900	. 99	62, 200	. 3	28. 6	10. 0
AZ80-X (Forged magnesium)	46, 000	. 065	707, 700	1. 28	6, 000	92, 300	. 29	18, 000	. 6	6. 5	10. 0
24ST86 (Wrought aluminum)	74, 850	. 100	748, 500	1. 35	10, 200	102, 000	. 32	25, 500	. 5	11. 2	10. 6
75ST (Wrought aluminum)	82, 300	. 101	814, 900	1. 47	12, 000	118, 800	. 37	19, 000	. 5	10. 3	10. 1
RC-130-A (Forged titanium)	140, 300	. 166	845, 200	1. 53	96, 300	580, 100	1. 80	60, 000	. 6	16. 7	10. 0
CTL-91-LD (Phenolic resin)[a]	23, 300	. 062	375, 800	. 68	------	0	0	15, 000	12. 0	3. 03	7. 0
PDL-7-669 (Polyester resin)[a]	36, 000	. 071	507, 000	. 92	1, 000	14, 100	. 05	14, 000	12. 5	2. 48	5. 9
DC 2104 (Silicone resin)[a]	21, 200	. 064	331, 300	. 60	4, 400	68, 800	. 21	10, 000	13. 0	2. 56	6. 3

[a] Fiberglas-cloth laminate bonded with plastic resins.

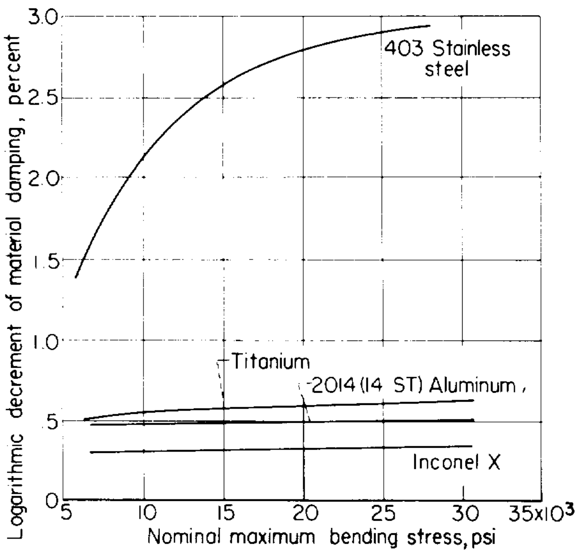


FIGURE 28.—Effect of vibratory stress on internal damping of various materials.

unfavorable coupling between the aerodynamic and elastic forces. Material damping, on the other hand, is always present for limiting amplitude buildup. Some materials display increased internal damping as the vibratory stress increases (fig. 28). Thus, the damping is highest when the need for it is greatest.

The damping values vary widely depending on the type of material (table I). The damping is given as logarithmic decrements in percent or the rate of decay of the vibration amplitude in successive cycles after the exciting force has been removed. Type 403 stainless steel has the highest value (2.8 percent) among all the conventional metals. A theory on the cause of exceptionally high values found for 403 stainless steel is discussed in reference 20. The principal reason for the wide usage of type 403 stainless steel is its high damping value. The internal damping of steel depends largely on its composition. Note, for example, that type 304 stainless steel has only 0.2 percent damping. The internal damping of 403 stainless steel depends also on heat treatment (ref. 6). An incorrect temper of the material may result in a logarithmic decrement of only a fraction of the value indicated in the table. Therefore, it is very important not only to use the proper steel, but to apply the proper heat treatment.

Aluminum and titanium show logarithmic decrements approximately one-fifth those of 403 stainless steel in its optimum condition. This is one reason why the early use of aluminum in compressor blade materials was discontinued, and why titanium must be fully evaluated before it can be adopted.

The metal TP–1, which consists of iron powder infiltrated with copper, has a logarithmic decrement of 4 percent, which is high for compressor blade materials. However, the reliability of the material for rotating blades has not been established, and at the present time, its use is limited to stator blades.

The plastics CTL–91–LD, PDL–7–669, and DC–2104 all show outstanding internal-damping characteristics. Preliminary tests indicate values of about 12 percent (ref. 28). Blades of such materials would, therefore, be very difficult to excite to high vibration amplitudes. Structurally, the plastics have sufficient strength, as verified by the 100-hour endurance run at rated speed in a full-scale engine (ref. 28). Initial tests, however, indicate that the present plastic laminates have only one-fourth the impact resistance of type 403 stainless-steel blades and therefore may become a problem from foreign-object damage. Also, because of their low modulus of elasticity, they can be broken during normal handling of compressor rotors by mechanics. Preliminary tests of the plastic blades show a resistance to erosion comparable or superior to that of steel blades.

Comparative evaluation of materials.—Up to this time, no reliable system has been devised which enables the designer to select the best material for a given application. For a given blade size and shape and exciting force, the vibratory stress is roughly inversely proportional to the material damping, if the natural frequency remains the same for the materials compared. Fortunately for ease of evaluation, the frequency changes very little for the different metals (table I). The natural frequency is proportional to the square root of the elastic modulus divided by the density. Under this restriction of constant natural frequency, the best material is the one in which the product of fatigue strength and internal damping is highest. However, in rating blades of two materials having unequal natural frequency, the relative merit depends on the relative proximity of the point of operation to resonance. Thus a blade having better strength and higher damping may be poorer in a particular application than another blade, if its natural frequency is closer to the exciting frequency. Because the blade frequency deviates appreciably when plastics are used, no direct comparison is possible. For one engine, plastics may be better than any of the metals; for another they may be worse, depending on the relation between the natural and exciting frequencies.

Accurate comparison of materials is further complicated by variations in aerodynamic damping for different material densities. In addition to factors already discussed, resistance to normal corrosion, erosion, and impact damage must be considered when evaluating materials. Notch sensitivity, also an important factor in material selection, influences the root design and the rate of propagation of fatigue cracks originating at points of foreign-object damage or stress-corrosion cracks.

INSPECTION PROCEDURE TO AVOID FAILURE

Inspection during manufacture.—Inspection performed at the present time during the manufacture of compressor blading pertains mostly to dimensional accuracy and material soundness. An improvement in engine reliability can probably be attained by the addition of the following simple inspection techniques:

(1) A natural-frequency check of each blade produced is very informative and is relatively easy and inexpensive. It would reveal any defects in the important regions by a lower-than-normal frequency and would indicate unfavorable dimensional inaccuracies, such as heavy tip sections or undercut root sections, which would also lower the frequency.

(2) Internal-damping measurements using the same equipment as for frequency checks would be simple. After noting the frequency, the exciting force could be abruptly stopped and the die-away curve measured, from which material damping could be computed. This damping measurement would check the material composition and heat treatment.

(3) Nondestructive tests have been greatly improved in recent years. The latest techniques should replace the older methods. For example, the improved zyglo process, known as post-emulsification, detects defects not indicated by previous methods.

(4) Composition and hardness checks are undoubtedly made by some manufacturers, but extended use would be helpful. Frequently the material supplier is relied upon to deliver the correct material, but mistakes can be made by both the supplier and the manufacturer. For an item as critical as a compressor rotor blade, composition checks should be made of each batch of material just before the cutting begins. To ensure against stress-corrosion cracking, the hardness of each blade should be checked upon completion. Only one defective rotor blade is required to wreck an engine and possibly destroy an aircraft.

Preflight inspection.—At the present time, engines are subject to standard acceptance tests prior to release for service. Since vibration is one of the principal causes of disastrous compressor blade failures, and since a definite number of vibration cycles is required to produce failure, the acceptance tests may not serve adequately to ensure successful service operation.

The rotating-stall characteristics vary for different engine models, and variations even exist among engines of the same model. The "green" run every engine must undergo prior to release for service should be designed to include operating conditions in which rotating stall and other suspected vibration inducers are most prevalent. Thus, the effectiveness of this test in detecting specific engines which might be subject to vibration problems is increased. Changes can even be made to aggravate vibration conditions, such as very rapid acceleration or abnormally small exit nozzle areas, to duplicate unusual conditions which may be reached in service. Performing a vibration survey on each engine is desirable, but this procedure is prohibitive with present instrumentation. To give a reasonable degree of assurance, at least five or six different engines of a given model should be instrumented for vibration and flow-fluctuation measurements under a variety of operating conditions, including idle, acceleration, part-speed range (50 to 70 percent of rated), and maximum power under both sea-level and altitude conditions.

An intensive effort should be directed toward developing equipment to detect blade vibrations without altering the engine. For example, a vibration detector under development at the Lewis laboratory entailed no alteration to the engine other than the addition of small permanent magnets to the tip of one or more blades. Large induction coils were wound around existing engine parts. If necessary, the magnet-tipped blades can be replaced by conventional blades after testing, although they have the same vibration characteristics and their strength is not impaired by the magnet. If the technique can be developed, the blades with magnets may be used in service for periodic checks or for a continuous record of the presence of vibrations. These precautions should be taken until a means of eliminating destructive vibration amplitude is devised or until experience shows that it is no longer necessary. Those engines indicating high vibration during tests should not be allowed to appear in service until adequate changes are made to lower the vibrations to a safe level.

On the initial installation of an engine into a new airframe, several engines should be surveyed for vibration characteristics. Because of the effect of distortion on rotating stall (ref. 29), when inlet ducting changes are made, the engine should again be checked. In fact, it would be advisable to conduct the first runs of a new engine with the engine connected to ducting similar to that intended to be used with the engine in service.

Inspection in service and overhaul.—Airport cleanliness and screening will reduce the effects of erosion and foreign-object damage. However, there is a minimum size of particle that can be economically removed from runways or screened from the inlet. Because of this limitation, engines and particularly the compressors should be carefully inspected on a regularly scheduled basis.

The engine and the engine nacelle should be designed to permit ready field inspection of a segment of the rotor blades for foreign-object damage. The early stages are most susceptible to foreign-object damage. The effect of nicks is greatest in these stages, which operate at the highest centrifugal stress and are most susceptible to vibration. Therefore, a particular effort should be made to provide easy inspection of the early stages as a means of detecting foreign-object damage. In applications where removal of the casing for a complete inspection is impractical, the principal purpose may still be served by facilitating inspection of the early stages. Sighting through the inlet may be adequate in some cases.

Inspection for stress-corrosion cracking should also be scheduled periodically. For this defect, however, the evidence is less obvious and special techniques, such as zyglo inspection, will be necessary. When the problem becomes less prevalent through the use of better material or heat treatment, this inspection should no longer be needed.

The practice of blade scrambling during overhaul should receive considerable study. Research should be undertaken to determine whether blades gradually deteriorate in fatigue resistance with operating time. A compressor may have many hard blades susceptible to stress corrosion. If the blades are removed and run through an inspection line with other blades, the hard blades may become distributed among a number of compressors which are then all potential sources of engine failure. The current practice is to mark blades for replacement in their original position, because the slight discrepancies in length of scrambled blades make necessary a grinding operation to cut all blades to the same external diameter. This affects the balance, increases tip clearance, and results in a loss of aerodynamic performance.

The maintenance of good records for each rotor blade for several engines of each new model would be helpful in establishing the normal life of blades in a particular application. The removal of all blades after a definite time period in service, regardless of the presence of damage, may possibly forestall some engine failures. This time period may be increased as improvements in materials and reduction in exciting forces are effected.

Checks for indications of blade deterioration may be devised by suitable research. For example, the natural-frequency changes or blade elongations are signs of impending failure. The largest change in frequency takes place in the very last stages of fracture, when failure is imminent. Elongation of blades is evidence of creep resulting from high tensile stresses, and if excessive, indicates that failure is near. Creep may become more of a problem than it is now when higher temperatures from higher flight speeds and pressure ratios become more common. Periodic frequency and blade-length inspections may serve as a guide in retiring blades at overhaul.

FUTURE RESEARCH

A number of aspects of the compressor reliability problem have not been discussed because there is no quantitative information at the present time on which to base decisions. For example, it is apparent that the structure and material of the compressor outer casing have a decided effect on the extent of damage if a rotor blade fails. A rugged casing would protect the remainder of the aircraft but the internal damage would be heavy; a very light casing would permit a failed blade to escape, thereby minimizing internal failure, but making possible damage to other vital points of the aircraft. Experience indicates that conventional aluminum-alloy casings usually retain plastic, aluminum, or bronze blade failures, but not steel blades (figs. 1 and 2). The optimum degree of failure retainment should be established by future research. The greatest protection may possibly be obtained by using a dual casing. The inner casing would be made of a thin material which could readily be pierced by the blade fragment. An air space or other retaining medium would prevent the broken blade from starting a build-up of failure within the compressor. The outer casing would be very strong to prevent penetration of the broken fragments, thereby protecting other engines, aircraft, and passengers.

Another suggestion warranting future research is the investigation of purposeful irregularities in the blade rows as a means of suppressing rotating stall. Rotating stall depends on successive stalling of adjacent blades for its propagation. If, for example, the angle of attack of several blades in each stage is made much lower than that of the other blades in the stage, the propagation may be disrupted by keeping one or more blades from stalling even with the increase in angle of attack due to the additional airflow from adjacent passages.

Research should also be conducted to determine whether there is a certain operating life after which it is economically advantageous to replace certain rotor blades. Front-stage blades, in particular, are subjected to higher static and vibratory stresses and, therefore, are more likely to fail than latter-stage blades. Useful operating life is largely a function of the particular engine model and type of service imposed on the unit.

Shrouding and lacing wires are used to prolong blade life on some jet-engine gas turbines and in the steam-turbine industry. Analytical and experimental stress analyses should be carried out to evaluate the merits of minimizing vibrations in rotor blades in this way. Shrouding of stator blades greatly decreases susceptibility to stator vibrations. In the stator blades, the shrouding does not impose any additional steady stresses, as it does with rotor blades.

Considerable effort is warranted in the development of new materials for compressor blades. The present plastic laminated blades are very promising but are limited in operating temperature, and even at low temperatures have very low moduli of elasticity. Improvements are being rapidly developed, however, and future plastic materials may overcome the temperature limitation. Better metallic materials should also be developed to completely eliminate the stress-corrosion problem without sacrifice of tensile and fatigue strength and internal damping; increased internal damping is always desired. Investigation of combinations of engineering materials may produce a blade that is resistant to damage by small-sized foreign particles and is less susceptible to fatigue when nicked by a foreign object.

Development of devices to detect the existence of compressor blade vibrations would greatly aid in improving compressor reliability. A small light-weight device placed on the engine that would warn the pilot of operating conditions which induce blade vibrations would be helpful. Also, instrumentation is needed, regardless of size, that could be placed near an engine during tests to detect vibrations without any alterations or wiring in the engine. Such a mechanism would enable the manufacturer to check every engine and evaluate ducting prior to release. It would also make possible the study of vibrations to devise methods of prevention or suppression.

Additional research on various means of retaining blades in the rotor and methods for providing added damping in the blade-root system may result in a configuration that would eliminate vibration-fatigue problems of rotor blades. The loose-pin or hinged blade requires additional refinement to attain the optimum proportions and arrangement of the various parts, but is already a great improvement over more conventional root forms.

CONCLUDING REMARKS

The use of axial-flow compressors in engines gives better performance characteristics, even though axial-flow compressors are less resistant to failure than centrifugal compressors. Currently, the trend is toward the use of axial-flow compressors. Therefore, the present study has been confined to the axial-flow compressor. Difficulties with compressor blades arise mainly from the following sources:

(1) Breakage by foreign objects

(2) Accelerated fatigue originating at a nick caused by a foreign object

(3) Accelerated fatigue originating at a stress-corrosion crack

(4) Fatigue caused by vibration, usually caused by rotating stall.

Failure of one compressor blade will, in most cases, cause a chain reaction of blade failures and result in immediate failure of the engine. Every effort must, therefore, be made to avoid compressor blade failure.

Blade breakage by foreign objects can be reduced by the use of engine-inlet screens. However, a lower limit to screen mesh size is imposed by considerations of the inlet-air-pressure drop and the associated reduction in performance. Small objects can pass through the screen and, while they may not cause immediate blade failure, they may result in nicks in the blade that may serve as nuclei for fatigue. Hence, it is essential to provide means for inspecting compressor blades, particularly those of the forward stages, for signs of damage. Engine modifications should be made so that inspections can be made conveniently and quickly at frequent intervals. In addition to inspection for foreign-object damage, the blades should be checked at less frequent intervals for evidence of stress-corrosion cracks, or other signs of deterioration. The presence of stress-corrosion cracks would indicate a need for a change in blade heat treatment, material, or fabrication technique.

Rotating stall is one of the principal causes of severe vibrations leading to fatigue failures, even in the absence of foreign-object damage. This phenomenon occurs mainly during acceleration of the engine through the speed range from 50 to 70 percent maximum speed. The engine manufacturer should determine the maximum vibratory stresses obtained during rotating stall for each engine. This test should preferably be made with inlet-flow distortion and reduced exhaust-nozzle area to simulate in steady operation the conditions expected during acceleration. Corrections to the engine design such as inlet baffles, interstage bleed, and variable-angle stator blades should be made if the stall condition is severe. The vibration problem in general can be considerably alleviated by using special root configurations and damping devices and investigating different blade materials. Plastic laminates show promise because of their very high internal damping, but at present they lack high-temperature (above 500° F) strength and rigidity.

Compressor reliability of existing units, as well as of more advanced design engines, can be improved by additional inspection procedures during manufacture, in service operation, before flight, and during overhaul. Provisions for convenient inspection and blade replacement should be considered in designs of new engines.

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CHAPTER V

COMBUSTOR ASSEMBLY

By Patrick T. Chiarito

SUMMARY

Service records for turbojet engines were studied to learn the types and causes of combustor failures and to obtain suggestions for improving engine reliability. The statistical data derived from the records show that the inner liner consistently accounted for the largest number of combustor failures in all engines studied, and several replacements of liners were usually made between major overhauls. However, liner failures rarely were the cause of an engine overhaul or an engine failure in flight.

Liners usually fail because of thermal stresses induced by large temperature gradients and propagated by thermal cycling associated with starting, accelerating, and stopping the engine. Failures are accelerated by such factors as improper fuel flow, carbon deposits, and severe transients. Two serious flight accidents that occurred in 1953 were attributed to igniter and fuel-nozzle malfunctioning.

Some comments are made regarding choice of materials, design and fabrication considerations, and operational practices for improving reliability. Additional information, for example, a better understanding of the complex failure mechanism, is needed in order to provide design criteria for preventing failures.

INTRODUCTION

The combustor of a turbojet engine consists primarily of sheet metal, which is required to satisfy a variety of specifications. For example: With respect to fabrication, the metal must have ductility for initial forming and must be joinable such as by welding; and, while in service, it must resist the corrosive action of hot gases, cracking due to thermal shock and fatigue, and distortion. Furthermore, small thicknesses are used to save engine weight.

Typical engine installations of combustors are shown in figure 1. The nonuniform impingement of hot gases on the liner walls causes hot spots and often abrupt temperature gradients. Temperature gradients are difficult to predict and exhibit a random behavior. Their effect is usually evident in severe warping and cracking of liners and eventual breakout of sheet-metal fragments. The transition liners and cross-ignition tubes are likewise subject to cyclic thermal stresses that can cause cracking and breakout of fragments. Carbon deposition in the combustor can cause malfunctioning of fuel nozzles and igniters.

Service records were studied in order to become acquainted with the difficulties with production combustors as background for the discussion of methods for improving engine reliability. The purpose of this paper is to present (1) the failure statistics that were derived from the service records, (2) a discussion of the types and causes of the failures, and (3) some suggestions for handling the combustor for improving engine reliability. Additional information that is needed to improve reliability is also mentioned.

FAILURE STATISTICS

SOURCES OF DATA

Approximately 1500 Disassembly Inspection Reports (DIR's) for the 3-month period from August through October of 1953 and a summary of jet aircraft accidents caused by engine failure or malfunction during 1953 (ref. 1) were studied for the three engine types to determine: (1) the way combustor components fail, (2) the mean life of a component as indicated by its

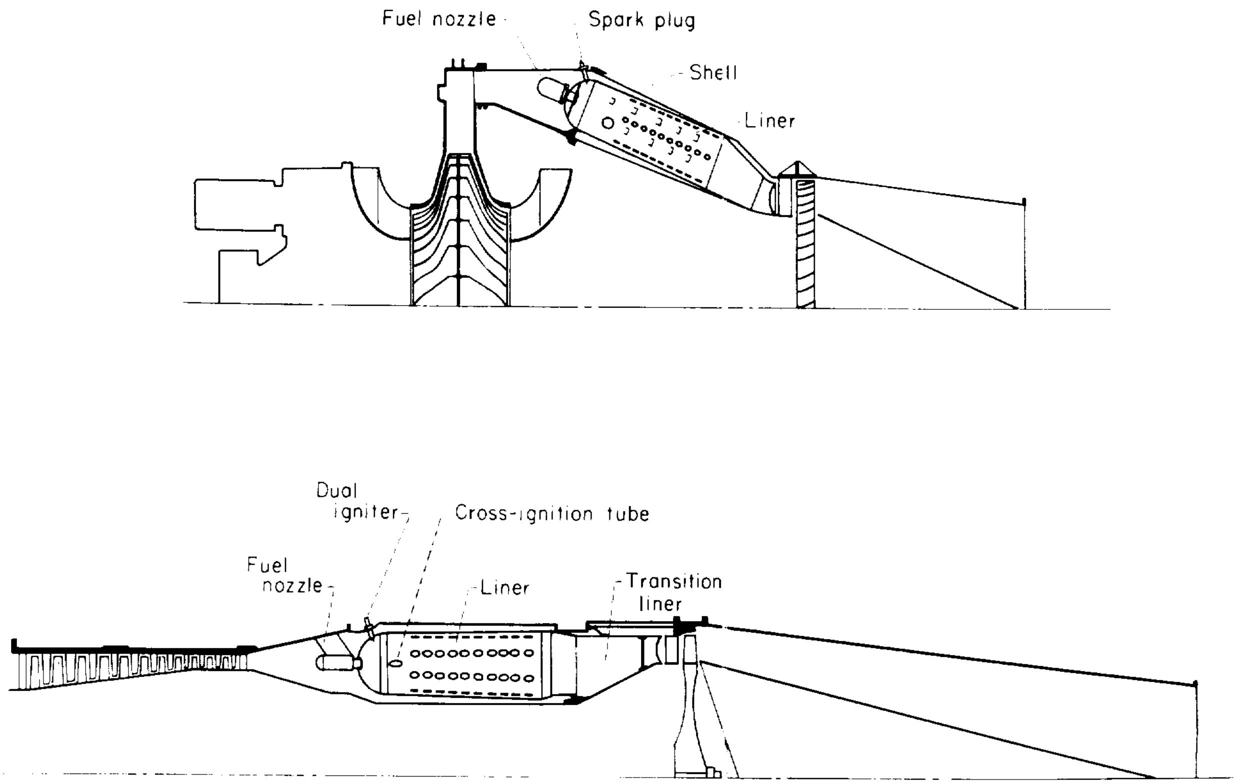


FIGURE 1.—Components of combustors in typical engine installations.

mean replacement time, and (3) the effect of a component failure on the engine and flight safety.

STATISTICS

Combustor failure statistics taken from the DIR's are broken down in table I. The data for old (previously overhauled) and new (not previously overhauled) engines are given in parts (a) and (b), respectively. The values for both are combined in part (c). It is apparent that, although a larger percentage of cracked liners are found during inspection in overhauls than any other combustor component failure, liner failure is rarely a primary cause of engine overhaul.

Of all engines analyzed, the average percentage that had some failure in the combustor was 69 percent for engine A, 27 percent for engines B, and 40 percent for engines C (table I). The sheet-metal liner consistently accounted for the most failures. In many of the engines all the liners were cracked, and almost always the liners of the ignition chambers were cracked. However, it is important to note (see column on primary causes in table I) that failure of the combustor liner is given as the reason for sending only one engine to overhaul. The primary cause for overhauling 33 engines C (5 percent) was sheet-metal failure in the outer shell, and overhauls of 11 engines B (2 percent) were due to failures in the transition liner. It is interesting to note that the largest percentage of liner failures occurred in engine A, which had the largest percentage of fuel-nozzle failures.

The mean replacement time for liners computed from the DIR data for new engines was 173 hours for A, 185 hours for B, and 365 hours for C. For engines C, the mean replacement time given is probably higher than actual time, because the DIR data for this engine do not include liner changes made in the field. In fact, the mean life (measured to the first sign of a crack) of liners for all engines would usually be less than the mean replacement time, because field inspection of liners is usually made incidental to maintenance of other engine parts. The

TABLE I.—STATISTICS OF ENGINES WITH FAILURES IN COMBUSTOR ASSEMBLY

[Data obtained from DIR's, Aug. through Oct., 1953]

Engine	A-7			B-3			B-4			B-7			B-8			B-9			B-10			C-1			C-2			C-4			C-7		
	Number	Percent total number	Number primary causes	Number	Percent total number	Number primary causes	Number	Percent total number	Number primary causes	Number	Percent total number	Number primary causes	Number	Percent total number	Number primary causes	Number	Percent total number	Number primary causes	Number	Percent total number	Number primary causes	Number	Percent total number	Number primary causes	Number	Percent total number	Number primary causes	Number	Percent total number	Number primary causes	Number	Percent total number	Number primary causes
(a) Old engines (previously overhauled)																																	
Total number engines analyzed	48			0			98			0			0			0			0			110			41			90			218		
Number with combustor failure	32	67					57	58														34	31		15	37		30	33		96	44	
Number [a] with failure in:																																	
Shell	12	25					8	8														13	12	6	6	15	4	4	4	1	29	13	15
Liner	30	63					47	48														34	31		14	34		30	33		89	41	
Transition liner	3	6					33	34	2																								
Fuel nozzle	11	23					4	4														0			0			0			0		
Cross tube	0						0															0			0			0			0		
Igniter	0						0															0			0			0			0		
(b) New engines (not previously overhauled)																																	
Total number engines analyzed	136			73			33			88			92			72			160			0			0			0			210		
Number with combustor failure	95	70		24	33		20	61		6	7		15	16		1	1		44	28											95	45	
Number [a] with failure in:																																	
Shell	11	8		2	3		3	9		0			3	3		0			1	0.6											12	6	7
Liner	92	68		16	22		9	27		4	5		6	7		1	1		41	26											94	45	1
Transition liner	3	2		16	22	7	10	30		2	2	1	1	1		0			3	2	1												
Fuel nozzle	21	15		0			0			0			0			0			0												1	0.5	1
Cross tube	0			1	1		0			0			9	10		0			0												0		
Igniter	3	2		0			0			0			0			0			0												0		
(c) All engines (old plus new)																																	
Total number engines analyzed	184			73			131			88			92			72			160			110			41			90			428		
Number with combustor failure	127	69		24	33		77	59		6	7		15	16		1	1		44	28		34	31		15	37		30	33		191	45	
Number [a] with failure in:																																	
Shell	23	12		2	3		11	8		0			3	3		0			1	0.6		13	12	6	6	15	4	4	4	1	41	10	22
Liner	122	66		16	22		56	43		4	5		6	7		1	1		41	26		34	31		14	34		30	33		183	43	1
Transition liner	6	3		16	22	7	43	33	2	2	2	1	1	1		0			3	2	1												
Fuel nozzle	32	17		0			4	3		0			0			0			0			0			0			0			1	0.2	1
Cross tube	0			1	1		0			0			9	10		0			0			0			0			0			0		
Igniter	3	2		0			0			0			0			0			0			0			0			0			0		

[a] Sum of component failures exceeds total number of engines with combustor failures because simultaneous failures occurred.

replacement of liners during scheduled inspections is adequate, however, because the cracks progress relatively slowly. Before February 1, 1955, recording of field repairs on the DIR was optional. At that time, a technical order was issued that requires a record of all field inspections, even if no replacements are made.

Other failures (e.g., cross-ignition tubes and igniters) are also included in table I.

EFFECT OF COMPONENT FAILURE ON ENGINE OPERATION

Warped liners cause abnormal temperature distributions and may result in burning of nozzle vanes and turbine rotor blades. Cracks in liners are probably tolerable until they cause fragments to break out. Sheet-metal fragments usually scratch or nick vanes and rotor blades, producing stress raisers which shorten the fatigue and stress-rupture life of the blades. At times, more damage is done. The DIR's on the 1500 engines studied listed only 5 liners and 12 transition liners in which failures had progressed to the point where fragments had broken out. The size of the largest fragment broken out of the inner liner and the resultant damage are as follows (the fragment from a transition liner that caused the most damage and the recorded damage are also listed):

(1) Inner liner: In an A–7 engine with a 3- by 5-inch sheet-metal piece missing from an ignition liner, all 95 turbine buckets were dented at the leading edges, and 26 of the 72 vanes were dented at the trailing edges.

(2) Transition liner: In a B–3 engine with 1½ inches missing from the aft retainer strip, all 96 turbine buckets were bent approximately ½ inch on the leading-edge tips, and 58 of the 64 vanes were nicked, bent, and torn along the trailing edges.

No record was found, however, of buckets that suffered immediate fracture due to the impact of sheet-metal fragments.

Four of the 205 flight accidents due to turbojet-engine failure or malfunction recorded by the U.S. Air Force during 1953 (ref. 1) were attributed to combustor-component failures. The results of the analysis of these four accidents are summarized in table II.

CAUSES OF FAILURES

MECHANISM OF SHEET-METAL FAILURE

During normal engine operation, the sheet-metal liner walls are heated by the impingement of hot gases. Because of nonuniform burning of fuel and mixing with secondary air, the liner is locally heated. The deformations of the material resulting from the local expansion may cause buckling and also local plastic flow in the hottest spots because of thermal stresses induced by the resistance of the cooler surrounding metal (ref. 2). Repeated heating and cooling of the liner associated with starting, accelerating, and stopping the engine will cause rupture by fatigue, especially at stress raisers such as louvers.

TABLE II.—FLIGHT ACCIDENTS CAUSED BY COMBUSTOR FAILURE OR MALFUNCTION

[Ref. 1]

Aircraft	Engine	Accident type	Brief	Findings
F–3	B–10	Minor	Excessive exhaust gas temperature	Two liners failed causing damage to nozzle diaphragm; hole burned in aft fuselage
F–3	B–10	Major	Flameout in flight. Aircraft destroyed after airstart attempts unsuccessful	Igniter leads fouled causing only one can to fire
F–3	B–10	Major	Engine explosion during flight	Probably spray of fuel nozzle 7 distorted and caused excessive heat in liner
F–1	C–7	Minor	Lost power on takeoff roll	Possible temporary carbon deposit on fuel nozzle spray tips

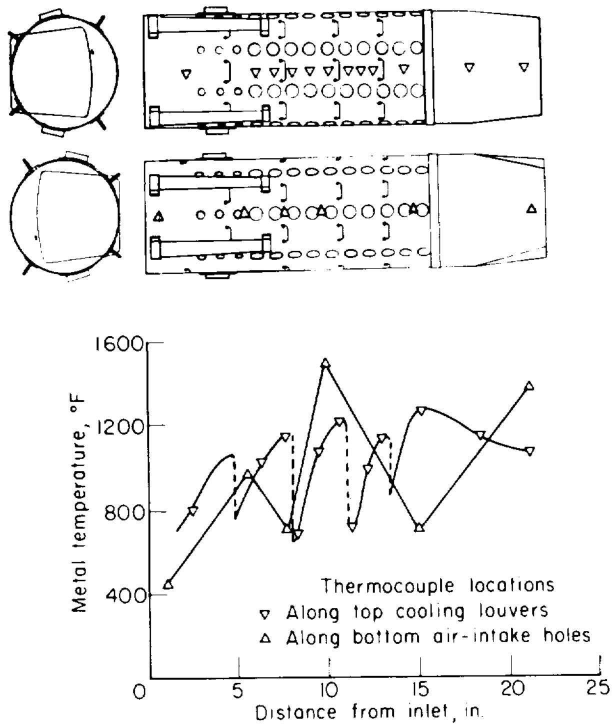


FIGURE 2.—Longitudinal distribution of metal temperature on top and bottom of liner (ref. 3).

Longitudinal distributions of metal temperature measured along the top row of cooling louvers and the bottom (or side nearest the engine shaft) row of air-intake holes in a liner are shown in figure 2 (from ref. 3). The abrupt discontinuities at the louvers and the wide variations between rather close air-intake holes may be seen. The hottest belt was near the middle of the combustor liner. The distribution of metal temperature measured around the liner near the hottest belt is shown in figure 3. The bottom is hottest, reaching a maximum of about 1600° F. A thermal gradient of as much as 700° F per inch was measured at 8000 rpm (or approximately 70 percent rated speed).

Because liners are loaded essentially only to the extent of self support, they are subjected mainly to thermal stresses. Typical liner failures are shown in figure 4. In the upper liner, the cracks progressed far enough from the louvers to the air-intake holes to break out a fragment near the hottest belt. A burned hole, buckles, and cracks can be seen in the lower liner. The corners of the transition liner (including pieces of the aft retainer strip) were also missing.

FACTORS THAT ACCELERATE COMBUSTOR FAILURES

Carbon deposits near fuel nozzles and igniters distort fuel spray patterns and may cause localized hot spots in the sheet metal. Carbon that accumulates elsewhere along the liner generally interferes with the gas flow and may cause overtemperature and distorted temperature distributions at the combustor exit (turbine inlet). Carbon deposits found around the igniters and fuel nozzles in two engine types are shown in figure 5. The deposits on the igniter of the combustor on the left will distort the top side of the fuel pattern. In the other combustor, the carbon (which grew on one electrode) obstructed more than half of the combustor passage immediately aft of the nozzle and obviously disrupted the fuel spray.

Carbon is almost always deposited on the inner walls of liners, on the fuel nozzles, and on the igniters of engines burning JP fuels. The tendency to produce carbon deposits depends upon the fuel analysis, combustor design, and engine operating conditions. Reference 4 shows that the tendency for a fuel to produce carbon is a function of its hydrogen-carbon ratio and

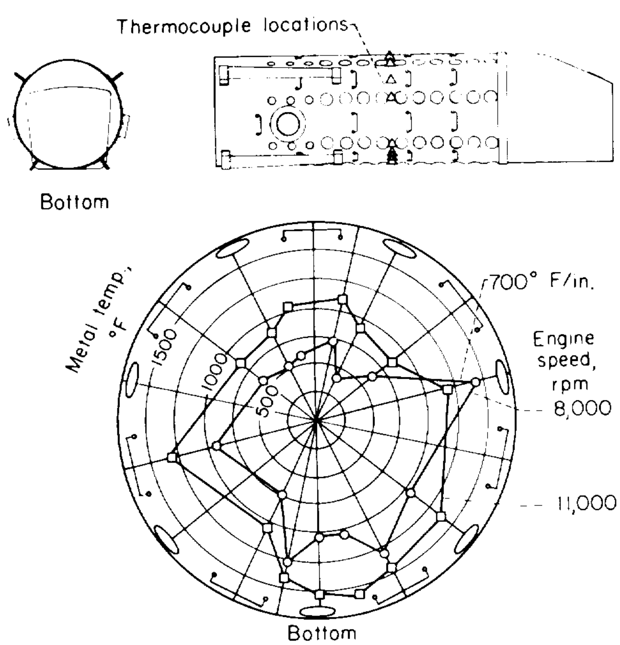


FIGURE 3.—Circumferential distribution of metal temperature near hottest belt of liner (ref. 3).

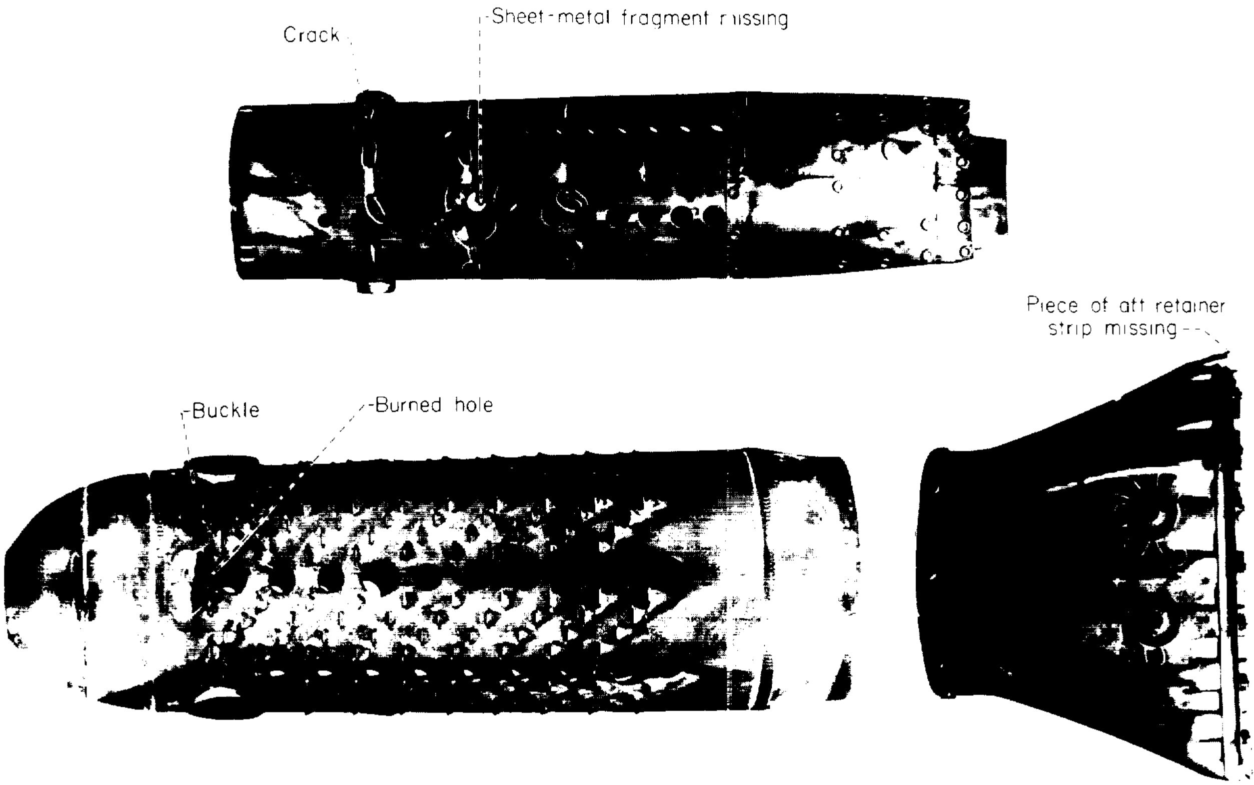


FIGURE 4.—Typical failures in turbojet liners.

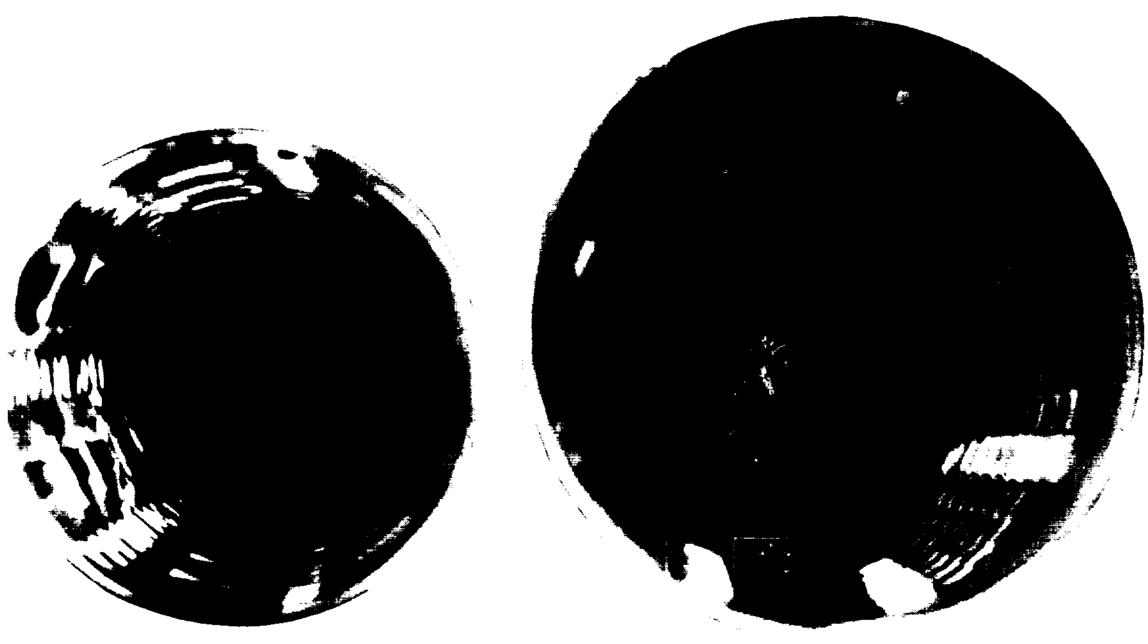


FIGURE 5.—Carbon deposits on igniters of two engine types.

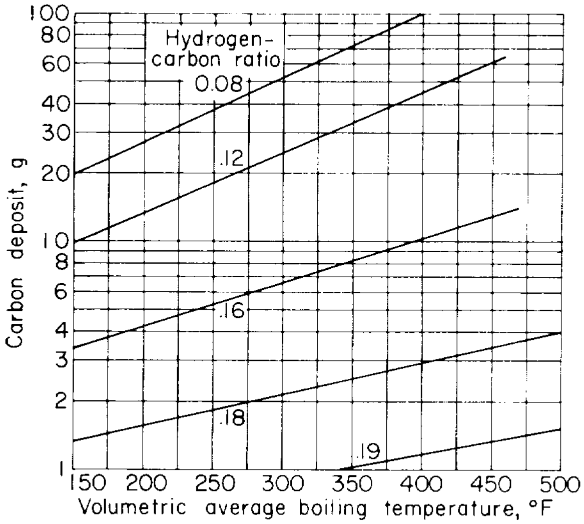


FIGURE 6.—Tendency of a fuel to produce carbon deposits as function of hydrogen-carbon ratio and volumetric average boiling temperature.

volumetric average boiling temperature, as illustrated in figure 6. Reference 4 also describes a simple laboratory smoke test for determining the carbon-producing tendency of fuels. Because the selection of fuels is based upon other combustion properties, as well as availability and cost, some carbon deposition cannot be avoided.

Clogging of fuel-nozzle passages and screens and fuel-orifice wear affect the fuel-flow rate and pattern and also cause liner damage. Fuel-flow rate beyond the permissible limits was noted primarily in engine A–7, which had the highest percentage of fuel-nozzle malfunction (17 percent) and probably accounted for the highest percentage of liner failures (66 percent, see table I(c)). Although no data have been found, it is expected that heat soak-back may account for deposits within the nozzle caused by thermal decomposition of fuel constituents.

Faulty igniters cause unsuccessful starts and probably produce thermal shock and large temperature gradients in the sheet metal. Severe transients, such as accidental hot starts and rapid acceleration or deceleration, also hasten sheet-metal failures in the transition liner.

Small cracks caused in manufacture by punching air-intake holes accelerate fatigue failure in liners. Thermal cycling causes the initial cracks to grow and eventually to extend to louvers.

EFFECT OF MATERIAL PROPERTIES

The tendency for failure of the liners by thermal cycling is affected by the properties of the material of the liner, such as thermal conductivity, coefficient of thermal expansion, strength at high temperatures, and the response of the material to repeated cycles of mechanical deformation and plastic flow.

The use of materials with increased thermal conductivity will relieve localized hot spots by dissipating the heat. Flame impingement tests on small cylinders of several materials showed the expected decrease in thermal gradients with increasing conductivity (ref. 2). Figure 7 shows gradients of 1000° F per inch measured for the commonly used Inconel. This gradient was reduced to 710° F for Inconel-clad nickel and to 370° F for Inconel-clad copper.

The number of flame impingement cycles to failure is shown in figure 8 for the materials included in figure 7. A wide spread of values is indicated, and no consistent relation was found between number of cycles and severity of temperature distribution (fig. 8). The benefits of clad metals that approach optimum properties

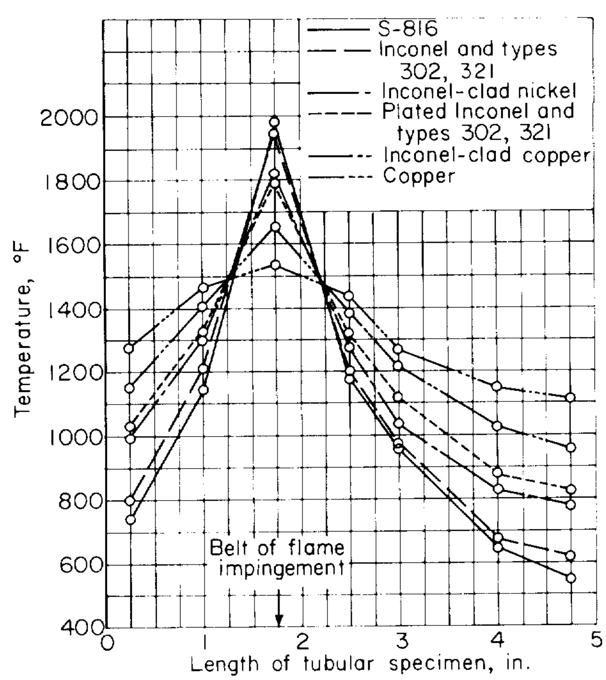


FIGURE 7.—Temperature gradients measured during flame impingement tests (ref. 2).

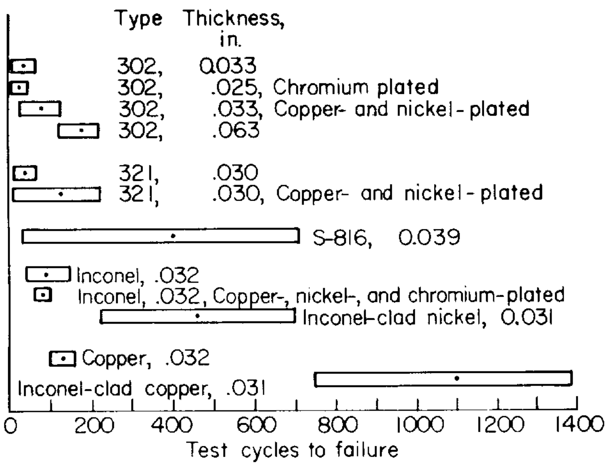


FIGURE 8.—Fracture due to repeated flame impingement. (Dot indicates average number of cycles for specimens of each material.)

necessary to satisfy many of the requirements of sheet metals in high-temperature service is, however, brought out in figure 8. Unfortunately, the use of cladding is hampered by fabrication difficulties.

A decrease in coefficient of thermal expansion will be beneficial in reducing the amount of differential expansion caused by thermal gradients.

The mechanism of failure of components by thermal cycling is complex, and much additional research is required on this subject. From this research should come ideas for the selection of improved materials and for improvements in design.

Current liners are usually made of Inconel which has high corrosion resistance. Liners are being placed in production made of mild steel coated with aluminum for corrosion resistance. Ceramic coatings are also being developed for protecting the surfaces of liners made of the lower strategic alloy materials. Materials selected for high thermal conductivity may not in themselves have adequate corrosion resistance, but satisfactory protective coatings could possibly be developed.

METHODS OF IMPROVING OPERATIONAL RELIABILITY

MATERIALS

The life of liners can, of course, be increased by the use of improved materials. The importance of such properties as high thermal conductivity, low coefficient of thermal expansion, and other material properties, and the need for additional research have already been discussed.

DESIGN AND FABRICATION CONSIDERATIONS

Some sheet-metal problems may be solved by the use of heavier gages. Besides increasing the strength and stiffness, increased thickness provides more heat-flow area and reduces thermal gradients. This practice is limited, however, by the weight penalty imposed. Some sheet-metal failures, for example those found in outer combustion chambers, are easily fixed by patching (fig. 9) and do not constitute a serious problem.

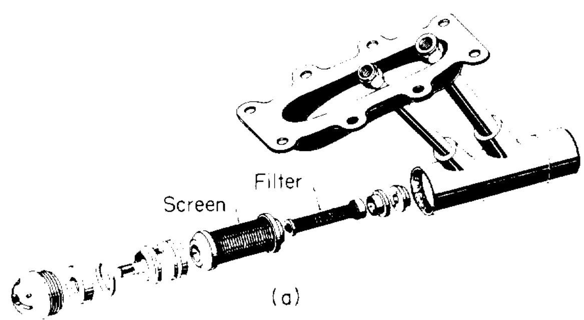
Because thermal gradients will probably always occur in sheet metal, a design principle that should be explored is the relieving of thermal stresses rather than resisting them. This may be approached by using a segmented structure for the liner in which expansions in any one segment do not impose stresses on neighboring segments.

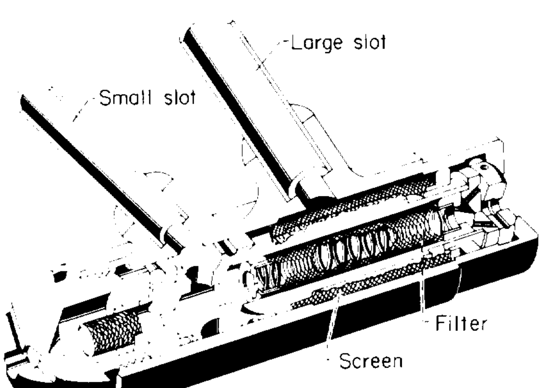
As pointed out previously, faulty fuel flow is probably the biggest source of combustor troubles. Besides the fuel nozzle itself, any of the several parts of the fuel control and supply sys-

FIGURE 9.—Failure and patch in outer combustion chamber.

tem may contribute to this deficiency. With regard to the nozzle, clogged screens or filters (fig. 10) will interfere with the flow. Such was the case with another engine type, resulting in frequent collapse of the screens and severe interference with fuel flow. Removing these screens from the hot section of the nozzle adequately solved the problem; the screens in the flow dividers proved to be ample. In some fuel nozzles the small slot provides fuel continuously, and the large slot supplements this supply during periods of increased demand. If the outer annular passage in the nozzle were used for the small flow, some cooling of the inner passage would probably result and the harmful effects of heat soak-back might be reduced.

Previous research (ref. 5) shows that the use of a shielded fuel nozzle to prevent carbon deposition also improves its performance, especially at altitude.





(a) Parts of fuel nozzle.

(b) Cutaway view of fuel nozzle.

FIGURE 10.—Typical fuel nozzle showing location of screen and filter.

The igniter for a turbojet engine must produce a spark that will ignite the fuel-air mixture that flows past the electrodes. If a start is not made within a few seconds, the combustor must be cleared of fuel before a new start is attempted in order to prevent overtemperature. Flameouts during flight impose still more severe requirements upon the igniter for a restart. If the igniter is coated with carbon (see fig. 5), the combustible mixture may be deflected away from the electrodes or the electrodes may be shorted. Surface-discharge igniters in which carbon deposits do not cause electrical interference are being developed (ref. 6). In fact, carbon deposits may help their performance.

Because igniters are not needed during normal flight, they cause unnecessary interference with fuel spray and provide a projection upon which carbon may grow. Retractable igniters (ref. 7) would probably overcome this objection.

On the basis of current statistics, it can be anticipated that several liner replacements will be made between engine overhauls. It is therefore essential that the design provide for accessibility of the combustor for ease of field maintenance. Disassembly of the combustor to inspect and replace damaged parts as necessary would help to extend the major overhaul interval for the engine. If this disassembly is made without removal of the engine from the aircraft, the "down-time" for the aircraft would be kept to a minimum.

Because overtemperature of the liner may result from several causes, such as (1) excessive fuel flow, (2) distorted fuel spray pattern, and (3) distortion of the flame by carbon deposits, an indicator of excessive liner temperature would be very useful in avoiding burning of sheet metal and incipient cracking.

Cracks in the liner caused by thermal cycling will progress slowly to breakout of a sheet-metal fragment that will damage the turbine rotor blades. Because there is usually sufficient time for detecting these cracks during a field inspection, it seems adequate to base a liner replacement on an examination of its condition during a scheduled inspection.

Mechanical finishing of the edges of punched air-intake holes retarded cracking in liners (ref. 8). The holes were reamed, sanded, and vapor-blasted to remove small cracks and worked metal caused by the manufacturing method. The benefits are given in the following table:

Time in accelerated life runs, hr	Average number of cracks in seven as-fabricated liners	Average number of cracks in seven mechanically finished liners
8	8	2
16	20	9

Although mechanical finishing helped, cracking consistently started at the stress-relieving holes even though they were also reamed and vapor-blasted.

OPERATIONAL PRACTICES

With regard to reliability, it would be preferable to replace cracked liners with new ones. However, cost requires consideration of the repair of liners. For example, one Navy facility claims that a saving of $350,000 was realized during 1953 by patching damaged liners (ref. 9).

The mean replacement time of combustor components is considerably less than desired times to major overhaul. Scheduled field inspections and replacements as necessary will therefore help to extend the overhaul interval.

In view of the flight safety and economic considerations, the engine manufacturer and air-line operator will have to work out a policy that might include the following:

(1) Scheduling inspections of combustor components depending on operating conditions

(2) Replacing liners regularly with approved used or new liners

(3) Procedure for logging time on each liner used

(4) Definition of extent of damage that requires repair or replacement

(5) Definition of repairable damage

Since rapid engine failure can result from a badly distorted gas temperature distribution, a device for warning of such occurrence would permit the pilot to take action that might prevent a flight accident.

CONCLUDING REMARKS

The statistics derived from the service records that were studied indicate that the inner liner consistently accounted for the largest percentage of combustor component failures in all three engine types. Even under normal operation, the life of a liner was relatively short and several field replacements were made between major overhauls. The failures generally started as buckles that caused cracks. Fortunately, cracks progressed relatively slowly to breakout of sheet-metal fragments, and it seems sufficient to base a liner replacement on an examination of its condition during a scheduled field inspection. Although there was evidence that sheet-metal fragments from liners damaged turbine rotor blades, there was no evidence of pieces even as large as 3 by 5 inches causing immediate fracture of a blade. Sheet-metal failures were, however, the primary causes of overhaul for 45 engines.

Failures in sheet metal are caused by large temperature gradients and thermal cycling. The induced thermal stresses may cause buckling and local plastic flow. Repeated cycles of temperature change associated with starting, accelerating, decelerating, and stopping cause repeated working of the material and cracks. Cracks are tolerable until they permit breakout of fragments. The danger of engine destruction in flight by a sheet-metal fragment passing through the turbine is greater for the multistage than for the single-stage turbine engine, but comparatively little experience has been gained with the multistage turbine engine. The effect of the hole left by the fragment is usually of small importance. Severely warped liners cause abnormal temperature distributions and may result in burned nozzle vanes and buckets. Probably the most important contributory cause of sheet-metal failures is improper fuel flow. These failures are accelerated by factors such as carbon deposits, severe transients, and stress raisers caused by fabrication methods.

Two major and two minor Air Force flight accidents during 1953 were attributed to combustor-component failures. Because the mean replacement time of combustor components is considerably less than desired times to major overhaul, scheduled field inspections are necessary to extend engine life and improve reliability.

ADDITIONAL INFORMATION NEEDED

In order to increase combustor-component life and engine reliability, additional information is needed, including the following:

(1) A better understanding of the complex mechanism of sheet-metal failure by thermal cycling would provide a basis for improved design and selection of materials. Laboratory tests on simple plate elements could be extended to simulate production combustors in their operating environment. As an interim measure, the number of cycles of transient engine operation that will cause buckling, cracking, and eventual breakout of fragments would be helpful.

(2) Detailed combustor gas and metal temperatures are needed to guide the laboratory tests mentioned in (1) and as a design criterion. The performance of the fuel system should be correlated with measured temperatures and component life.

(3) Methods of reducing the effects of localized heating should be explored. For example, segmented construction of the liner should minimize the interaction of adjoining elements and thereby extend service life.

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CHAPTER VI

NOZZLE DIAPHRAGMS

By Francis J. Clauss

SUMMARY

Repeated thermal stresses are the most important factor causing damage to nozzle diaphragms. These stresses arise from nonuniform temperature distributions and from constraints which prevent the nozzle diaphragm from expanding and contracting freely during heating and cooling. Repetition of these stresses occurs during starting, acceleration, and stopping and eventually causes cracks along the edges of the nozzle vanes, at the trailing edge of the vane slot in the inner ring, and in the weld joining the inner ring to the mounting flange. These cracks usually do not progress to complete failure of the unit.

Although cracks in nozzle diaphragms are frequent and add to the cost of engine maintenance, they rarely cause engine failure. Damage to nozzle diaphragms is not a problem with respect to reliability at the present time. This may be due to frequent repairs and replacements in the field and during major overhauls for other types of damage.

Cracks, because of their rate of propagation and effect on engine operation, can be safely handled by scheduled inspection. This involves costly repairs and replacements.

Cracking of nozzle diaphragms can be reduced in the following ways:

(1) Incorporate provisions to allow the parts to expand and contract freely.

(2) Distribute the temperature more evenly throughout the nozzle diaphragm and the attaching parts of the engine.

(3) Use materials with better resistance to repeated thermal stressing.

(4) Eliminate vane slots in the inner ring or reduce their tendency to promote cracks.

Rapid deterioration of the nozzle diaphragm occurs when, through malfunction of the combustor, severe overtemperature occurs. Overtemperature may cause sections of nozzle vanes to be burned off. Nicks and dents in the exposed surfaces are frequently caused by foreign objects.

INTRODUCTION

Damage to nozzle diaphragms is found in the majority of jet engines received at overhaul depots. Such damage must be repaired before the engines are returned to service, and frequently entire nozzle diaphragms must be replaced. This chapter reviews damage to nozzle diaphragms and its effects on the reliability of turbojet engines.

The schematic sketch of a turbojet engine in figure 1 shows the approximate operating temperatures of the various parts. The nozzle diaphragm is located between the combustion chambers and the turbine wheel. Combustion of the air-fuel mixture takes place within the combustion chambers, and the hot gases expand with increasing velocity through the nozzle diaphragm. Here nozzle vanes direct the gas stream against the blades of the rotating turbine wheel which, in turn, drives the compressor.

The nozzle diaphragm is a stationary element of the engine. It consists essentially of a number of airfoils, or vanes, held between two rings, as shown in figure 2. The vanes are subjected to no mechanical stresses other than those which arise from the force of the gas stream. This force can cause the vanes to deform unless the material has an adequate yield strength and creep resistance at high temperatures. Lack of strength or excessive temperatures produce light bows at the trailing edges of the vanes.

The temperature of nozzle vanes approaches the gas temperature. Metal temperatures as high as 1900° F have been measured (ref. 1), although normally the nozzle vanes operate at tempara-

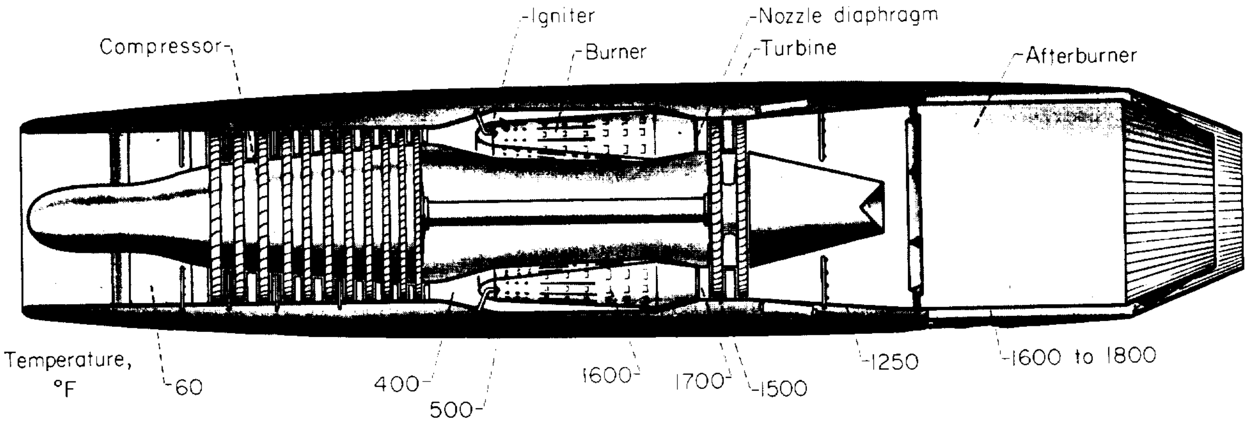


FIGURE 1.—Turbojet engine with afterburner.

tures several hundred degrees less than this (ref. 2). The inner and outer rings, which hold the nozzle vanes, operate at lower temperatures. In some engines these rings fasten rigidly to other parts of the engine, and the nozzle vanes, in turn, are fastened rigidly to the rings. The thermal expansion and contraction of the vanes then produce stresses in the inner and outer rings which cause them to warp or crack.

Each time the engine is started, the nozzle vanes are heated rapidly to their operating temperatures. Lesser temperature changes occur whenever the engine changes speed. During a normal shutdown, the vanes are again subjected to rapid cooling. If, for any reason, flameout (or combustion blowout) occurs during flight, the vanes are cooled rapidly by the inrush of cold air. Rapid heating then follows on reignition of

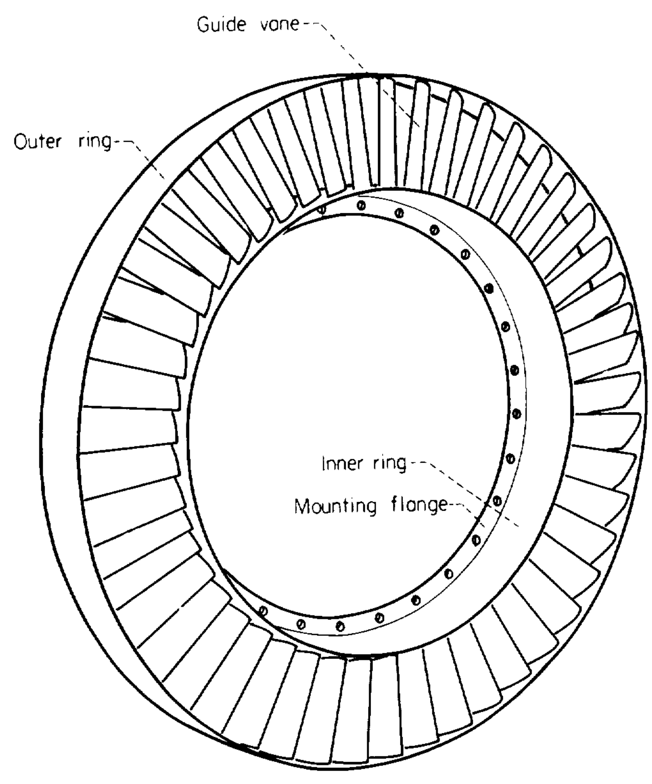
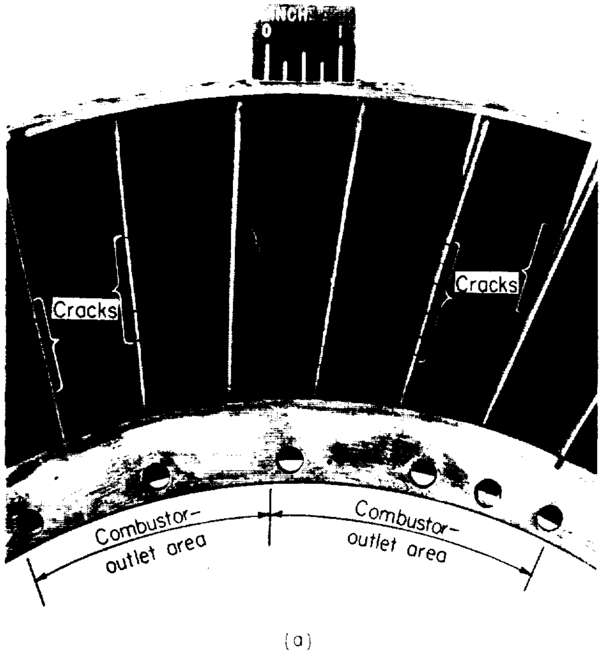
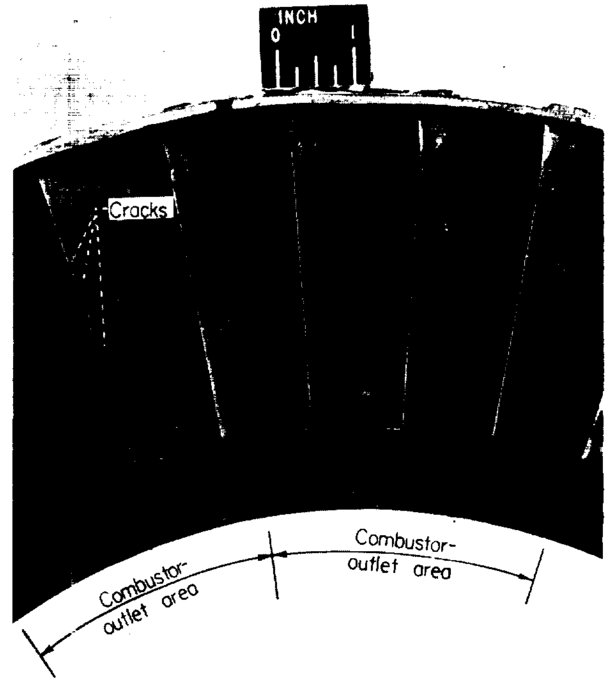


FIGURE 2.—Nozzle diaphragm.



(a) Upstream edges.

FIGURE 3.—Nozzle diaphragm showing cracks on edges of vanes (ref. 9).



(b) Downstream edges.

FIGURE 3.—Concluded. Nozzle diaphragm showing cracks on edges of vanes (ref. 9).

the flame. All these conditions expose the nozzle vanes to repeated cycles of thermal stressing which can cause them to warp, crack, and ultimately fracture. Figure 3 shows the cracks produced at the edges of nozzle vanes by this repeated thermal stressing. The vanes in the middle of the combustor outlets, where the temperature is hottest, show more cracking than the other vanes. Figure 4 shows more detail of the cracks developed by thermal stresses after a number of engine cycles.

The hot combustion gases to which the nozzle diaphragm is exposed are corrosive. Besides being oxidizing, the gases sometimes contain harmful elements from the fuel, such as lead and vanadium. These elements attack the vanes and penetrate into the material along the grain boundaries. The cracks so developed act as stress raisers and promote failure from other causes. In practice, the corrosion problem is avoided by using materials of adequate oxidation resistance and specifying that jet fuels do not contain large amounts of harmful elements. Surface coatings might be used for protection, but they present other problems.

Solid particles in the gas stream exert a scrubbing action against the nozzle vanes. Erosion of the vane surfaces can occur unless proper materials are used.

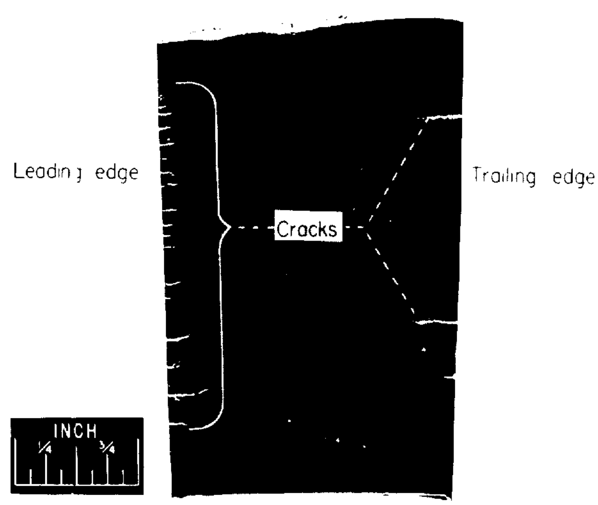


FIGURE 4.—Typical cracks in nozzle-diaphragm vane after 319 cycles (ref. 9).

Particles impinging against nozzle vanes can cause them to fail by impact. These particles can be foreign objects ingested through the intake system, parts that shake loose during operation, or fragments of compressor blades or combustor liners that break off. The damage to the nozzle diaphragm may be limited to nicks or dents that do not seriously interfere with engine operation. Occasionally, however, large objects chip off sections of the vanes and these, in turn, can cause further damage to the turbine blades and to the tailpipe and afterburner. This problem is discussed in chapter III.

In short, the design and materials used in nozzle diaphragms should provide adequate resistance to

(1) Aerodynamic loads
(2) Thermal stresses
(3) Corrosion
(4) Erosion
(5) Impact

The purposes of this paper are to review the following:

(1) The importance of nozzle-diaphragm damage to the reliability of turbojet engines

(2) The types of damage that occur in nozzle diaphragms

(3) The mechanisms of damage to nozzle diaphragms and the factors of engine operation that cause damage

(4) The ways in which nozzle diaphragms might be made more reliable

(5) The areas in which additional information is needed

To accomplish these purposes, operating records on turbojet engines and literature pertinent to the problems have been examined. This information has been analyzed in the light of research on turbojet engines.

FAILURE STATISTICS

IMPORTANCE OF DAMAGE TO NOZZLE DIAPHRAGMS

The importance of damage to nozzle diaphragms can be examined from two points of view:

(1) Its effect on engine operation, such as reduced thrust or a flight accident

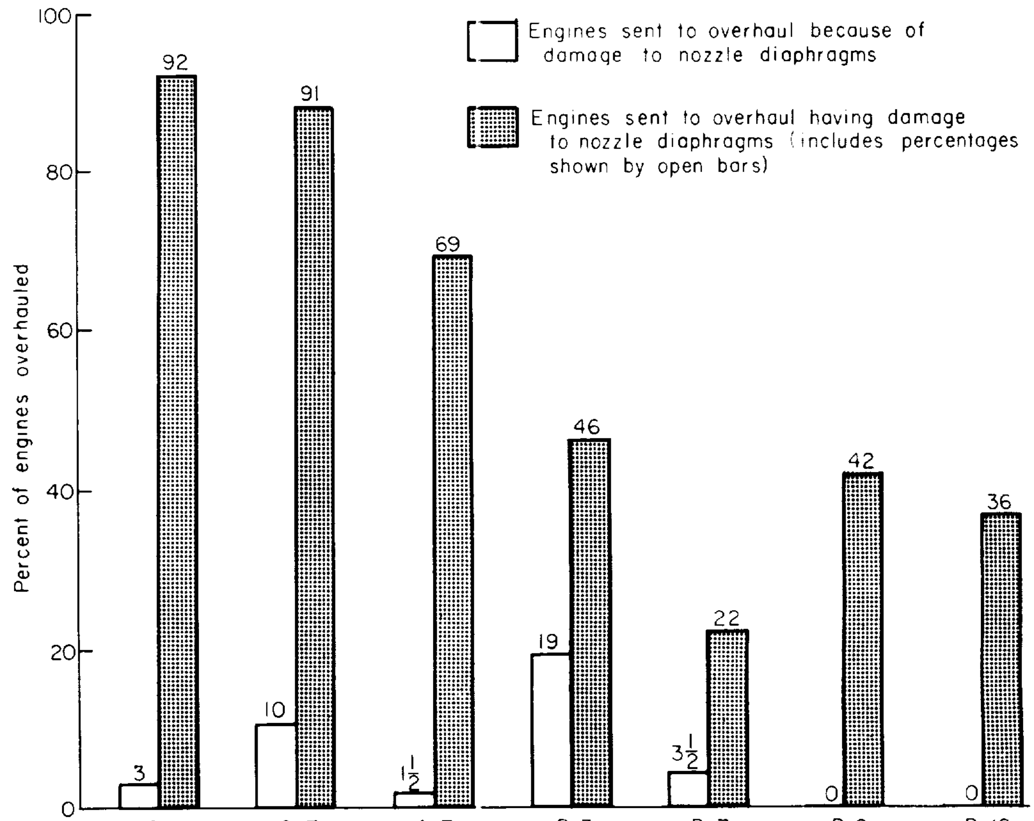
(2) Its effect on engine maintenance

Although nozzle diaphragms require frequent repair or replacement, failure of nozzle diaphragms is not an important factor causing flight accidents in turbojets at the present time. This fact is shown by a study of 205 U.S. Air Force jet accidents during 1953 in which failure or malfunction of the engine or its accessories was the primary cause of the accident (ref. 3). In two cases, overtemperature had damaged the hot section of the engine, including the nozzle diaphragm, and caused a major accident. In another case, foreign objects had damaged the nozzle diaphragm and the plane threw three turbine buckets, causing a minor accident. In these accidents, the nozzle diaphragm was only one of the items affected rather than the cause of the accident.

None of the records studied reported any loss of thrust from minor damage to nozzle diaphragms. Distortion of the vanes would change the flow characteristics and, if severe enough, require changes in the exhaust-nozzle area to maintain operating conditions. Excessive bowing of the vanes could affect engine performance (ref. 4).

Ordinarily, damaged nozzle diaphragms are repaired or replaced in the field. When base facilities are inadequate, turbojet engines may be sent to overhaul depots for this work. The open bars in figure 5 show the percentages of several types of engine that were sent to overhaul depots because of damaged nozzle diaphragms. For example, of the C–6 engines received at the overhaul depots, 3 percent were sent there because of damage to the nozzle diaphragm. This figure is low for most other engines, but several engines, such as the C–7 and B–3, show much higher percentages of overhauls because of damaged nozzle diaphragms.

Despite the repairs made in the field, a large number of nozzle diaphragms are found damaged during inspection at the overhaul depots in engines sent there for other reasons. This is shown by the cross-hatched bars in figure 5. In the C–6 engines, for example, 92 percent of all the engines sent to overhaul had damaged nozzle diaphragms. (This value includes the 3 percent sent to overhaul because of damaged nozzle diaphragms.)



Engine	C-6	C-7	A-7	B-3	B-7	B-9	B-10
Installation	Fighter	Trainer	Fighter	Bomber	Bomber	Bomber	Fighter
Engines overhauled	37	208	134	69	86	71	161
Maximum operating time, hr	651	885	425	634	165	285	301
Median operating time, hr	350	301	136	242	60	48	88

FIGURE 5.—Damage to nozzle diaphragms reported at overhaul depots. (Data obtained from DIR's, Aug., Sept., Oct., 1953.)

Damage to the nozzle diaphragms does not pose a problem of engine reliability because the damage progresses so slowly that it can be repaired before a failure occurs. The necessary repairs or replacements may be made in the field or during overhauls for other types of damage. As these other types of damage are reduced or eliminated, damage to nozzle diaphragms may become a more important problem to engine reliability.

OPERATING RECORDS

The manner and frequency of damage to nozzle diaphragms can be learned from a study of operating records on turbojet engines. Aircraft engine Disassembly Inspection Reports (DIR's) were made available to the NACA to furnish this data. These reports were prepared by the overhaul depots of the Air Research and Development Command, U.S. Air Force, and each report contains the findings on an engine at the time of overhaul. The limitations of the DIR's in furnishing reliability data are discussed in chapter II.

TYPES AND FREQUENCY OF DAMAGE TO NOZZLE DIAPHRAGMS

The sketch in figure 6 shows the types of damage most common to nozzle diaphragms. The frequency with which they occur during operation is shown in figure 7 for the various engines studied.

The most common type of damage was cracking of the nozzle vanes. Thus, for engine C-6, fully 92 percent of the engines overhauled (or every engine in which any damage to the nozzle

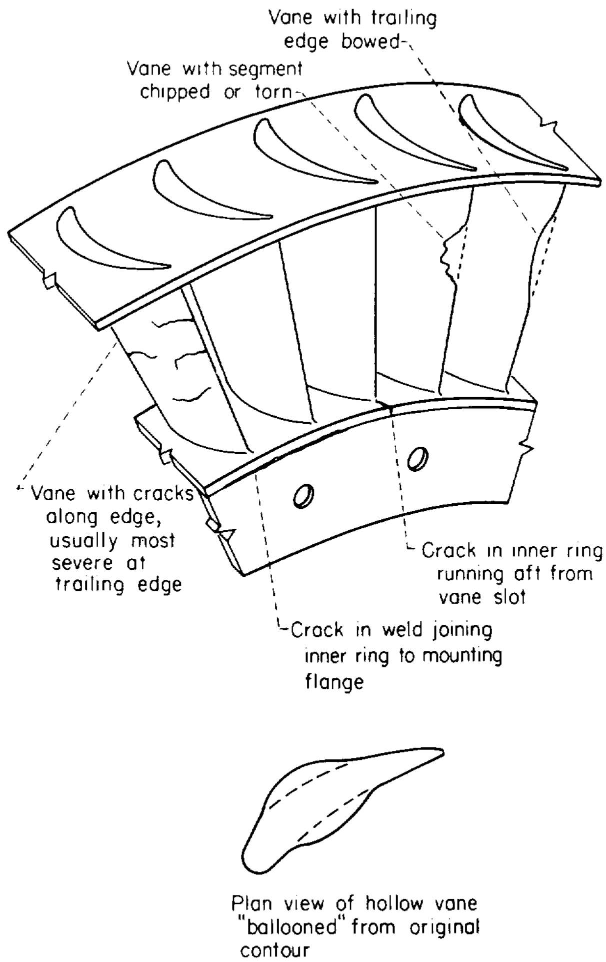


FIGURE 6.—Types of damage most common to nozzle diaphragms.

diaphragm was found) had one or more cracked vanes. The cracks started most often from the trailing edges of the vanes. Two other common types of cracks were circumferential cracks in the weld joining the inner ring to the mounting flange and axial cracks in the inner ring running aft from the vane slots. In several engines, the weld joining the inner ring to the mounting flange was cracked the full 360°. Cracks in the inner ring running aft of the vane slots frequently progressed into the mounting flange. The presence of any of these types of crack apparently did not interfere with the operation of the engine. The damage to nozzle diaphragms in engines overhauled for other reasons was just as severe as that found when damage to nozzle diaphragms was the primary cause for overhaul.

The other type of damage found most frequently in nozzle diaphragms was nicks and dents in the vanes. This type of damage was present, for example, in 57 percent of the C–6 engines overhauled.

Types of damage observed less frequently include the following:

(1) Metal deposits on surfaces; surfaces "sandblasted"

(2) Nozzle vanes bent, torn, chipped, warped, distorted, "ballooned," or burnt to varying degrees of severity

(3) Inner and/or outer ring cracked or warped

(4) Aft edge of inner ring, mounting flange, or outer ring worn or grooved

A further breakdown of damage statistics for the C–7 engine is presented in figure 8, which breaks down the data shown in figure 7 into 50-hour intervals for the first 700 hours of operation. Beyond this time, there were not enough engines for a valid analysis.

The frequency of overhaul is shown in figure 8(a). The frequency was highest during the 50-hour interval from 150 to 200 hours, and about 12 percent of the engines were overhauled during this period. About one-third of the engines had been overhauled at the end of 200 hours of operation.

One-half of the engines removed during the first 50 hours of operation had some damage to the nozzle diaphragms (fig. 8(b)). This was true despite the fact that none of these engines were overhauled primarily because of this damage. The number of engines removed with some damage to the nozzle diaphragms increased rapidly with the time of operation, until practically all those removed after 200 hours had some damage to the nozzle diaphragms. A few engines overhauled after 200 hours' operation had no nozzle-diaphragm damage recorded, probably because of field replacements that were not reported in the DIR's. The conclusions that may be drawn from this figure are that practically all nozzle diaphragms will be damaged to some extent after 200 hours of operation, but that this damage will generally not seriously interfere with continued operation of the engine.

As shown in figure 8(c), the edges of the nozzle vanes were cracked in 25 percent of the engines removed during the first 50 hours of oper-

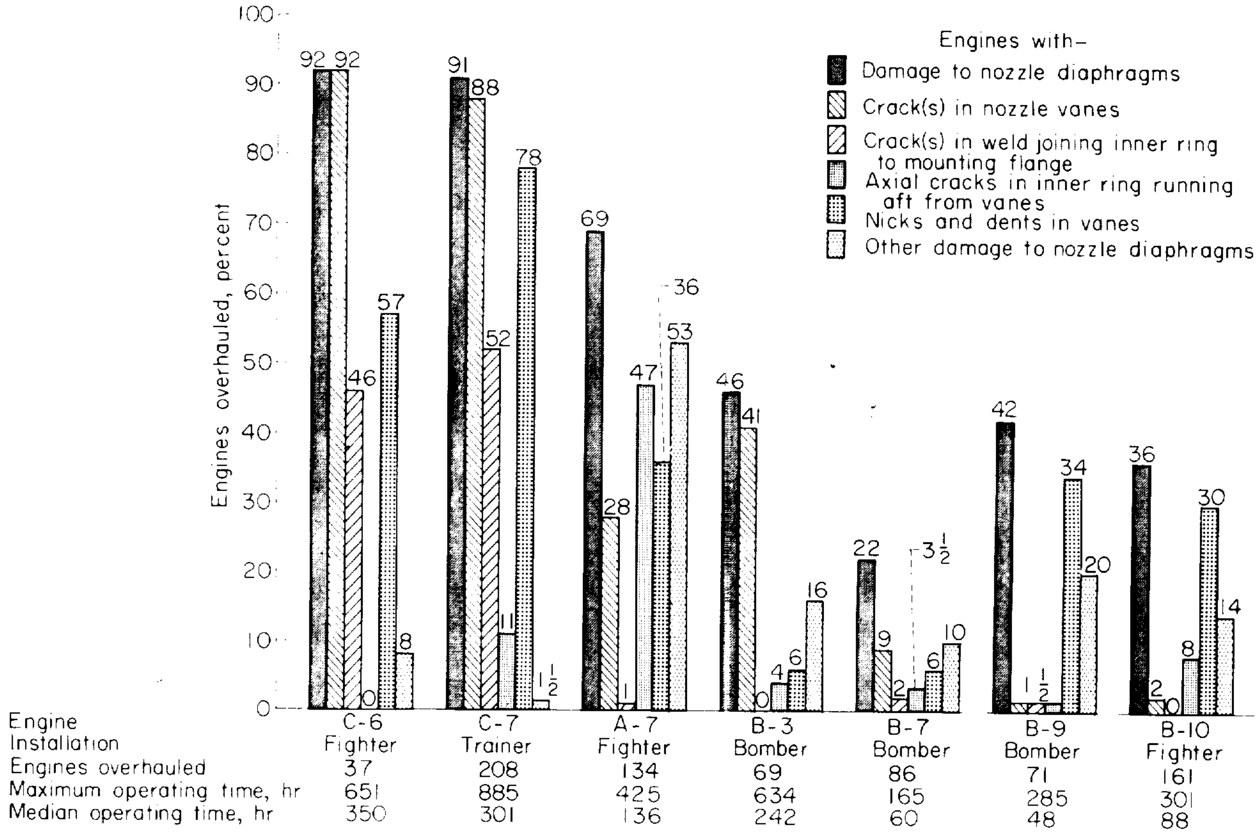


FIGURE 7.—Frequency of occurrence of various types of damage to nozzle diaphragms. (Data obtained from DIR's, Aug., Sept., Oct., 1953.)

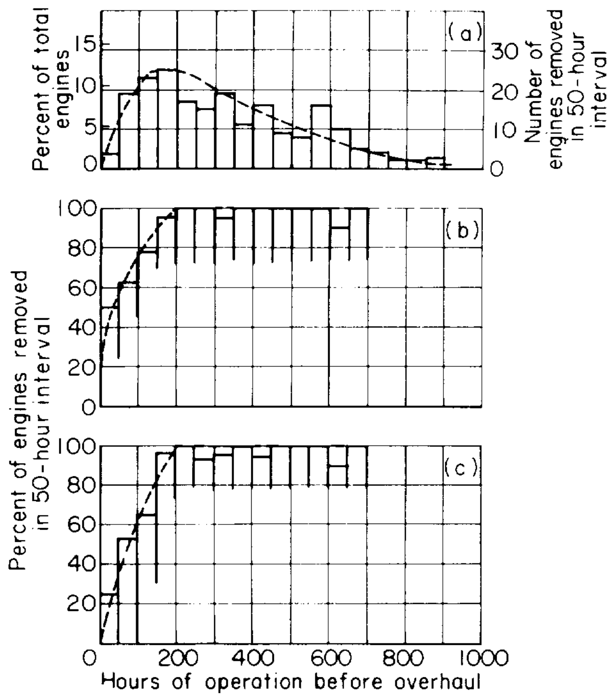
ation. After 200 hours, vanes in practically all the engines were cracked. Cracking was the most frequent type of damage to nozzle diaphragms, being reported in 88 percent of all engines of this type overhauled. The curve roughly parallels that shown in the previous figure for all types of damage.

Nicks and dents were present in the nozzle vanes of 50 percent of the engines removed during the first 50 hours of operation, as shown in figure 8(d). The rate leveled off at about 85 percent of the engines overhauled after 200 hours of operation. Field repairs probably account for this value's not reaching 100 percent for long operating times.

The weld joining the inner ring to the mounting flange did not crack until the second 50-hour interval of operation (fig. 8(e)). The frequency of weld cracking increased with time, although at a lower rate than that of nozzle-vane cracking (fig. 8(c)), and leveled off at about 90 percent after 500 hours. Again, field replacements not reported on the DIR's are probably responsible for the absence of this type of damage in a few engines overhauled after long times of operation; if these replacements were reported, the frequency of this type of damage might reach 100 percent. As for the effect of this type of damage, the records show that the weld in several engines was cracked the full 360° without causing an accident or being the primary cause of overhauling the engine.

Cracks running aft of the vane slots developed in the inner ring at a still slower rate. No cracks of this type appeared until the third 50-hour interval of operation, as shown in figure 8(f). The frequency increased continuously to a value of 30 percent after 700 hours of operation.

Any changes in design or materials to reduce the rate of one type of cracking must not, of course, unduly accelerate one of the other types of cracking.



(a) Frequency of overhaul.

(b) Frequency of damage to nozzle diaphragms.

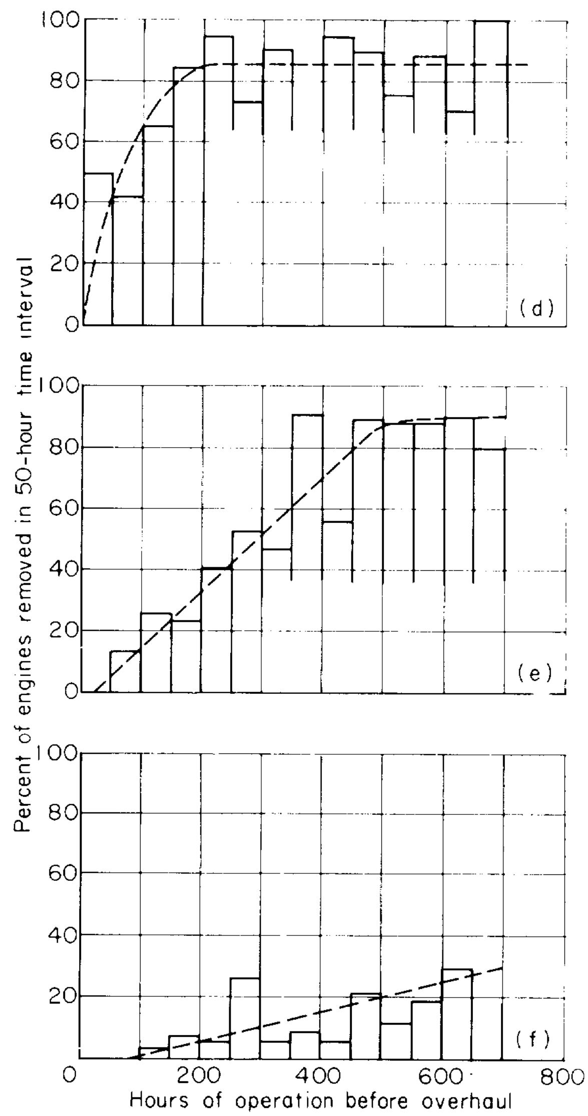
(c) Frequency of cracks in nozzle vanes.

FIGURE 8.—Overhaul data for 208 C–7 engines. (Data obtained from DIR's Aug., Sept., Oct., 1953.)

FACTORS OF OPERATION CAUSING DAMAGE TO NOZZLE DIAPHRAGMS

TEMPERATURE CONDITIONS

An understanding of the cracking of nozzle diaphragms must begin with a knowledge of the temperature conditions in the engine. In a study of temperatures of nozzle vanes in a J47 engine (ref. 2), the temperatures of nozzle vanes varied with their position relative to the combustor outlet. Vanes directly aft of the highest temperature zone of the combustor are naturally hotter than those behind the division points between combustors. Gas temperature across the outlet of the transition liner of one combustor varied as shown in figure 9. This figure shows the temperature distribution during operation at an engine speed of 7950 rpm and a tailpipe gas temperature of 1260° F (rated conditions for the J47 engine). The lines of constant temperature are approximated by the dashed lines in the upper part of the figure. The center of the 1900° F isothermal is displaced radially outward about 20 percent from the center of the transition liner, apparently because of the deflection of the hot gases by the inclined surface of the transition liner just ahead of the point at which temperatures were measured. The zone of highest temperature is displaced to the right side of the



(d) Frequency of nicks and dents in nozzle vanes.

(e) Frequency of cracks in weld joining inner ring to mounting flange.

(f) Frequency of cracks running aft from vane slots in inner ring.

FIGURE 8.—Concluded. Overhaul data for 208 C–7 engines. (Data obtained from DIR's, Aug., Sept., Oct., 1953.)

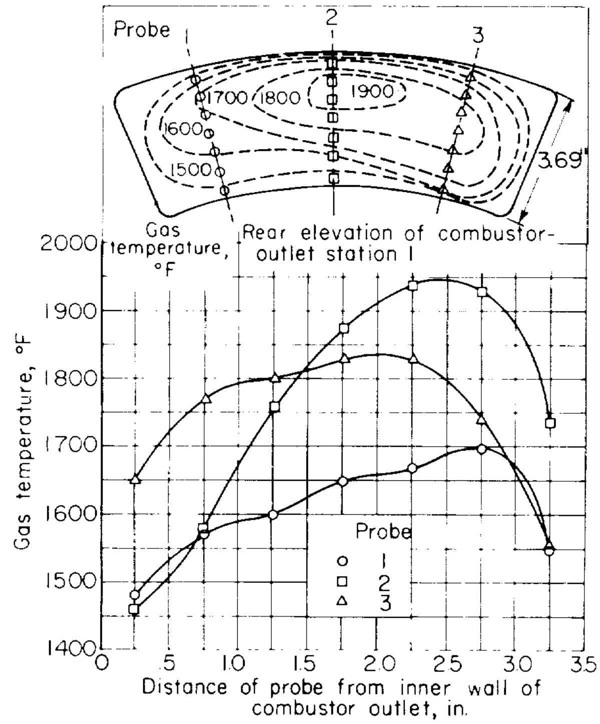


FIGURE 9.—Gas temperature across combustor of J47 engine (ref. 2).

combustor outlet. The highest gas temperature measured was 1940° F, and the lowest was 1460° F.

Temperatures are given in figure 10 for 10 adjacent nozzle vanes covering this combustor outlet under the same engine conditions. Temperatures were measured on the concave sides of the vanes, one-half of the distance between the inner and outer rings and four-tenths of the distance from the leading to the trailing edges (i.e., at midspan and 40 percent chord). The pattern shown by this figure is similar to that shown in figure 9 for the gas temperature. Nozzle vanes behind the division points between combustors are about 400° F cooler than vanes directly aft of the highest temperature zone of the combustor. Temperatures ranged from 1570° F on vane 7 to 1130° F on vane 2.

The temperature variation in a single nozzle vane is shown in figure 11. Temperatures were measured at the points marked at the midspan of the vanes (under the same engine conditions as the data of figs. 9 and 10). The important feature shown in this figure is the large difference in temperature between the edges and the body of the vane. This difference was ascribed to the passage of small amounts of cooling air through the hollow vanes used in the J47 engine. The discrepancy between the midchord temperature shown in this figure and the 40-percent-chord temperature shown in figure 10 for blade

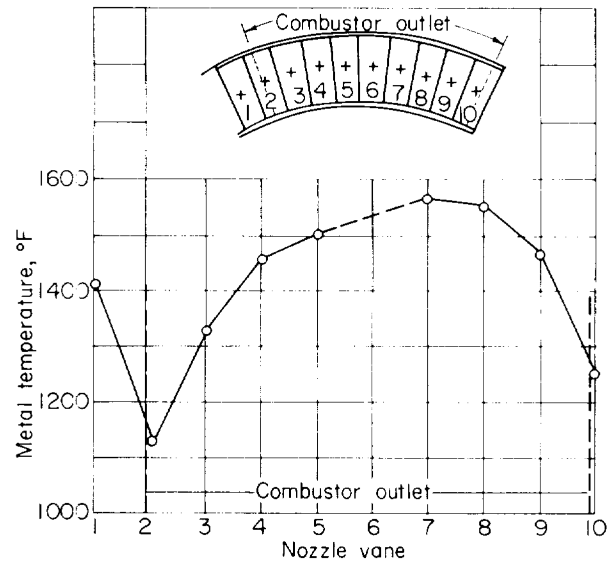


FIGURE 10.—Metal temperatures of nozzle vanes across combustor of J47 engine (ref. 2).

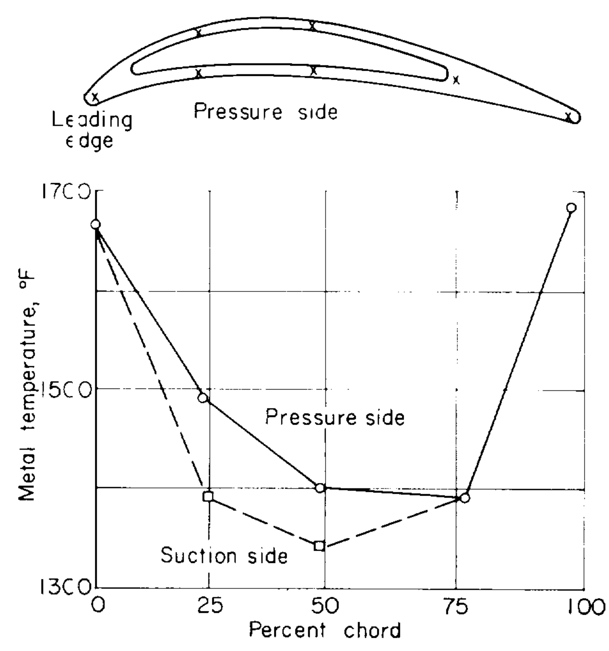


FIGURE 11.—Temperature variation across chord of nozzle vane in J47 engine (ref. 2).

6 is due to experimental changes made between the two tests.

The variation of vane temperature with time during rapid acceleration from idle to full power (3000 to 7950 rpm) is shown in figure 12(a). Temperatures were measured halfway between the inner and outer rings at the positions marked in the figure. Note that the trailing edge (position 4) responded most rapidly to changes in gas temperature and reached a peak of almost 1800° F during the acceleration period. Note also the large temperature drop from the trailing edge to the ¾-chord point (position 3) only 0.650 inch from position 4. The maximum drop is about 750° F at about 12 seconds after the start of acceleration, which was the time when the trailing edge reached its peak temperature.

Nozzle-vane temperatures during a normal and a hot start are shown in figures 12(b) and (c). Again, there is a large temperature drop from the edges to the body of the vane.

THERMAL STRESSES

When a body is heated or cooled nonuniformly, as nozzle diaphragms are, the various sections tend to expand or contract by different amounts. In order to enable such a body to remain continuous, a system of thermal strains and associated stresses is set up within the body. These thermal stresses are thought to be the most important cause of cracks in the nozzle diaphragms of jet engines. A general discussion of thermal stresses is given in reference 5.

The thermal stresses in a body depend upon

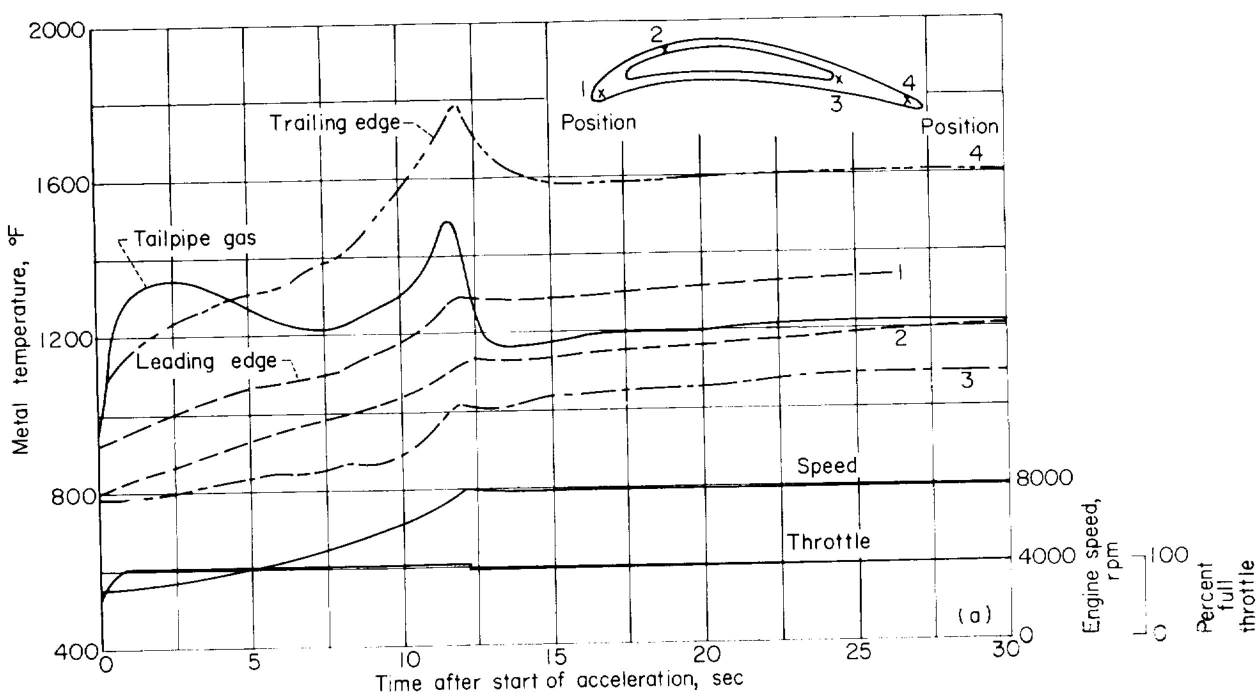
(1) The temperature distribution within the body

(2) The properties of the material

(3) The degree of constraint imposed upon the free expansion and contraction of the body

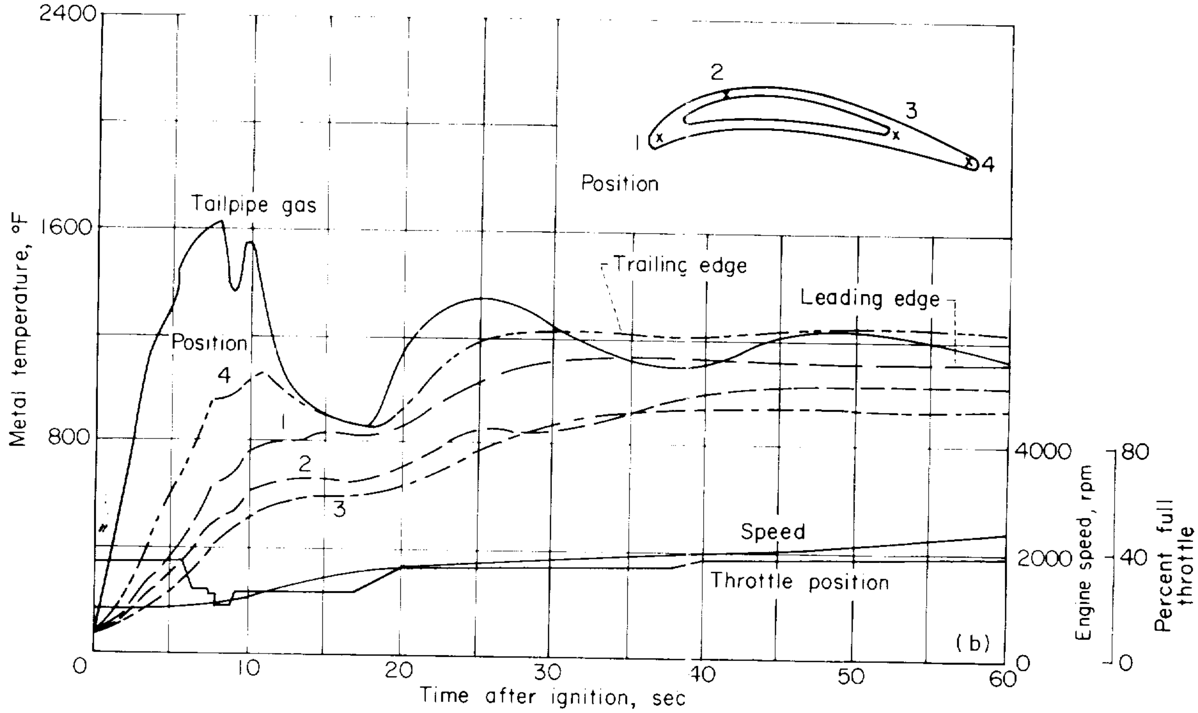
The forces constraining a body from free expansion and contraction may be imposed externally, as when a bar is heated and cooled with its ends held fixed. Or the constraining forces may be imposed internally by adjacent sections of the body, as when the temperature is changed nonuniformly. In this case, the degree of constraint depends on the size and shape of the body.

The temperature distribution in the body may be unchanging with time (steady-state), as when different parts of the body are held at constant temperatures. Alternatively, the temperature dis-



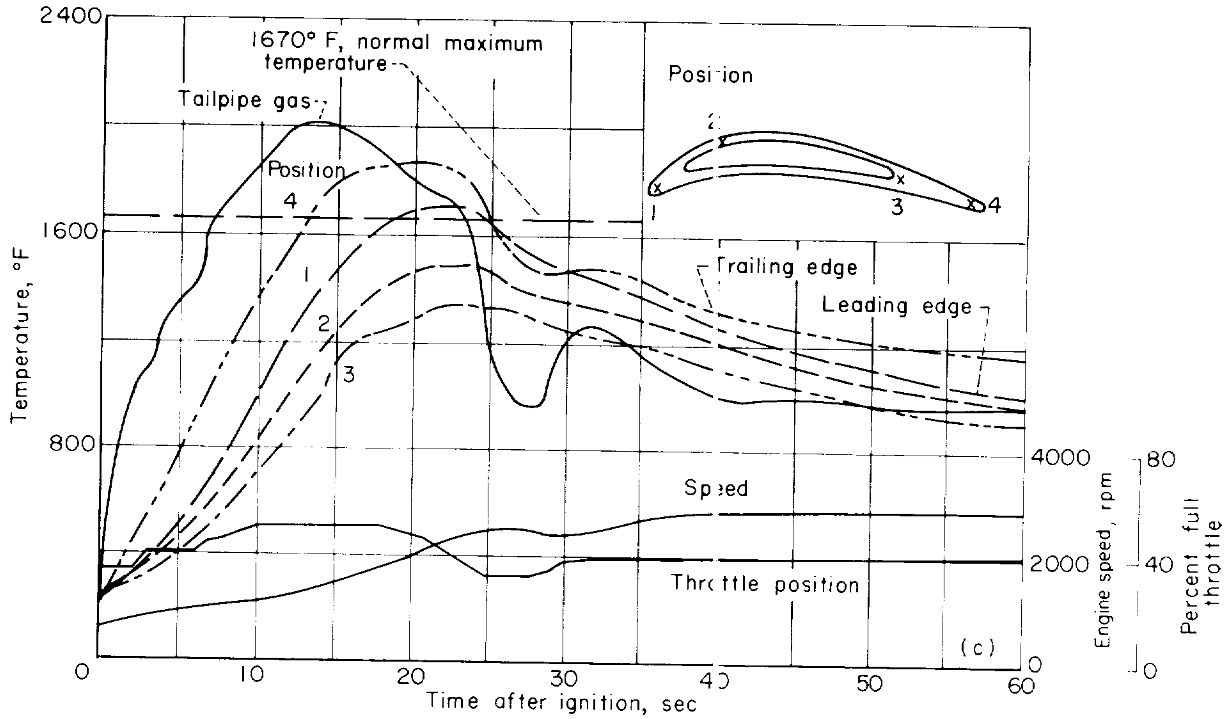
(a) Acceleration from idle (3,000 rpm) to rated speed (7,950 rpm).

FIGURE 12.—Transient temperatures (ref. 2).



(b) Normal start.

FIGURE 12.—Continued. Transient temperatures (ref. 2).



(c) Hot start.

FIGURE 12.—Concluded. Transient temperatures (ref. 2).

tribution may be transient, or changing with time. So long as the temperature distributions are equal, the thermal stresses are equal for both steady-state and transient distributions. However, the transient temperatures produced momentarily by sudden heating or cooling ("thermal shock") generally produce more severe temperature distributions from the standpoint of thermal stresses than do steady-state distributions. Thus, much higher thermal stresses may be produced during the thermal shock of a body between two temperatures than would exist at a steady-state condition. A second factor that may increase the damaging effect of thermal shock is the embrittlement of many engineering materials by the rapid application of stress. These materials may not be able to withstand a thermal-shock stress that could be readily absorbed if applied slowly so that the materials retained their ductility.

When failure occurs after a repeated number of cycles of thermal stressing, the process of failure is known as "thermal fatigue." This process is more complex than when failure occurs on the first cycle. It includes gradual changes in the material, such as those which occur during ordinary mechanical fatigue at constant temperature, as well as those which occur during exposure at high temperatures. Creep or stress-relaxation and microstructural changes may be important. Mechanical properties and the stress-strain relations may be different at the start of each cycle. The repeated straining in opposite directions during each half of the cycle is said eventually to exhaust the ductility of the material, and the body cracks.

Cracks caused by thermal fatigue have a brittle appearance with little or no apparent plastic yielding about the point of fracture. In this respect, they are similar to the fractures caused by ordinary mechanical fatigue.

Because of the similarity of the cracks in nozzle diaphragms to those caused by thermal fatigue, and in the absence of other conditions that might cause fatigue damage, thermal stresses are felt to be the most important factor causing cracks in nozzle vanes and rings during engine operation. High thermal stresses can also cause warping or distortion of the rings. The data shown in figures 10, 11, and 12 show temperature conditions that cause high thermal stresses.

The vanes near the center of the combustor are hotter than those at the edges, as shown in figure 10. Experience shows that the inner and outer rings operate at temperatures below those of the vanes. The difference in the thermal expansion of the vanes, as well as the difference between the vanes and the rings, causes stresses and distortions in the rings and vanes when both rings are attached to the vanes. There is also a difference in the expansion of the inner ring and the mounting flange to which it is welded and which, in turn, fastens to the engine.

Under steady-state conditions, the leading and trailing edges are hotter than the body of a vane, as illustrated in figure 11. Thermal stresses and plastic flow can occur along the leading and trailing edges. On cooling, the stresses are reversed, and plastic flow may occur in the reverse direction. Repeated starts and stops may ultimately cause the leading and trailing edges to crack.

The temperature gradients between the edges of the vanes and the body are more severe during starting (figs. 12(b) and (c)), accelerating (fig. 12(a)), and stopping than during steady-state conditions (fig. 11). Thus, thermal stresses are greater during these times. Hot starts can cause damage not only by the greater thermal stresses that accompany the higher rates of temperature change, but also by changes in material structure at the higher temperatures.

FOREIGN OBJECTS

Foreign objects are next in importance to thermal stresses in damaging nozzle diaphragms. As these objects pass through the engine, they nick and dent the exposed surfaces in their paths. Nozzle vanes are frequent victims, and the surfaces of the inner and outer rings are sometimes nicked and dented as well. Gravel or stones that have been pulverized in the compressor may give a sandblasted appearance to the nozzle diaphragm. Large objects may cause deep gouges or may chip or tear sections of the vanes, but the damage is generally not so severe as to limit the operation of the engine. Chapter III discusses their effect on engine reliability.

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OVERTEMPERATURE

A number of DIR's report instances where nozzle vanes have been "ballooned," partially melted, or otherwise burnt to varying degrees of severity because of gas temperatures that exceeded those normally permitted. Overtemperature may be caused by a hot start, personnel error, instrument malfunction, or compressor stall. Instrument malfunction and its effects on engine reliability are discussed in chapter X.

IMPROVING RELIABILITY

The factors that damage nozzle diaphragms can be grouped into those that may be considered part of the normal conditions in which an engine must operate and those that may be considered accidental. The accidental factors include foreign objects and overtemperature. The most direct approach in dealing with these factors would be to eliminate them entirely, as by installing protective screens and control instruments and ensuring their proper performance. These methods are discussed separately in other parts of this report. Until these changes have been made, the designer must allow for the accidental factors by designing sufficiently rigid structures or by using materials that are more resistant to impact and to overtemperature. This must be balanced with the greater cost, more difficult fabrication, and higher strategic-element content such designs and materials may have.

The predominant factor among those associated with normal conditions in the engine is thermal fatigue. This has been shown to cause cracks along the edges of the vanes, cracks along the weld joining the inner ring to the mounting flange, and cracks in the inner ring running aft from the vane slots. The solution to this problem is to develop designs and materials that will

(1) Incorporate provisions to allow the parts to expand and contract freely

(2) Distribute the temperature more evenly throughout the nozzle diaphragm and the attaching parts of the engine

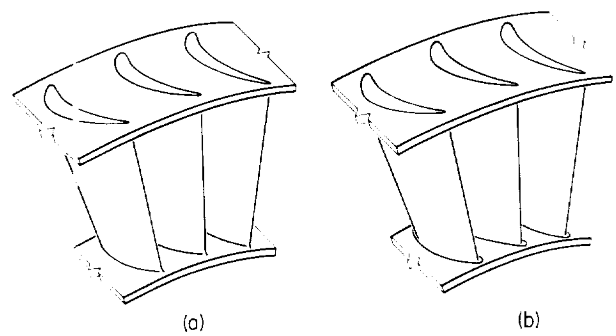
(3) Have better resistance to repeated thermal stressing

(4) Eliminate vane slots in the inner ring or reduce their effect in promoting cracks.

Until thermal cracking is eliminated, scheduled inspection of nozzle diaphragms is required to maintain operational reliability.

RELIEF OF CONSTRAINT

In the early nozzle diaphragms, the vanes were welded at both ends onto thick rings, as shown in figure 13(a). Since the outer ring does not heat as much as the rest of the unit, it constrains the expansion of the vanes during heating. This constraint causes compressive strains



(a) Vanes welded to rings at both ends.

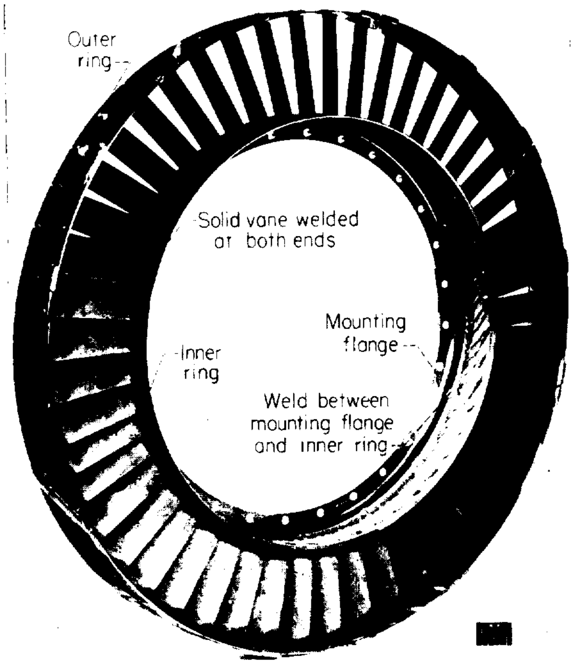
(b) Vanes welded at one end only.

FIGURE 13.—Sections of nozzle diaphragm (ref. 5).

in the vanes when hot, and, if these strains are plastic, residual tensile stresses are introduced when the vanes are cooled. Successive repetition produces thermal-fatigue cracks.

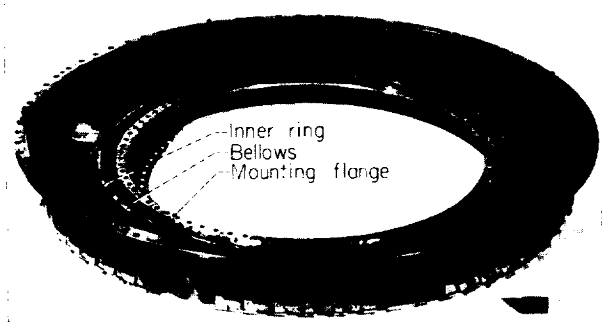
A later design that eliminates the end constraint of the nozzle vanes is shown in figure 13(b). In this design, the vanes are retained by welding to the outer ring and are floated in slots cut into the inner ring. The vanes can expand and contract freely along their length, with the inner ring serving only to position them. Other alternatives to allow free floating are possible.

The advantages of the floating design are seen by comparing operating records for different engines. Figure 14(a) shows the design in an early engine. Note that the vanes are welded at both ends to the inner and outer rings and that the mounting flange is welded to the inner ring. Figure 14(b) shows the free-floating design of a later engine. The vanes are alternately welded to either the inner or outer rings, except for a few vanes that are welded at both ends. A bellows between the inner ring and the mounting flange reduces constraints further. The records



(a) From early engine.

FIGURE 14.—Nozzle diaphragm.



(b) From later engine.

FIGURE 14.—Concluded. Nozzle diaphragm.

show that vane cracking was more frequent in the early design than in the later free-floating design. Part of this difference may have been due to differences in operating conditions and to the use of cooled hollow vanes in the later engine instead of the solid vanes used in the first engine. Cracks in the inner ring running aft from the vane slots were more frequent with the floating design, however. These cracks often extended into the mounting flange. The records did not show whether these cracks were located at the slots where the vanes were welded or were free.

Note that the free-floating design eliminates only the end constraint on the vanes; it does not eliminate those stresses in the vanes that arise from unequal temperatures in the different sections of the vanes. During heating or cooling, the surfaces of the vanes change temperature more rapidly than the interior. Also, the thin edges change temperature more rapidly than the more massive midchord sections. These differences in temperature can produce stresses well above the elastic limit of the material, so that plastic flow must occur (ref. 6). Again, repetition eventually causes cracks, warpage, and failure.

One way that might be investigated for reducing the constraint of adjacent sections of vanes on each other would be to segment the blade, as shown in figure 15. In this way, the edges of the vanes could slide past the center section during expansion and contraction and relieve the thermal stresses that would otherwise be produced. A segmented blade would not have the

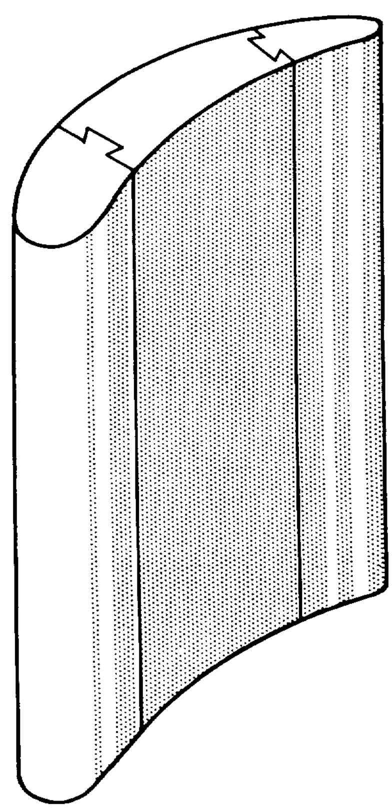


FIGURE 15.—Example of vane segmentation.

strength of an integral one, and care would be needed to avoid introducing other problems from aerodynamic loading. Thus, a segmented vane might lose the benefit of the camber in obtaining lateral stiffness and might bow more easily. The use of woven screen material covered with a thin skin might also reduce the thermal stresses set up by the constraint of adjacent sections.

Hollow nozzle vanes might have better thermal-shock resistance than solid vanes (ref. 5), perhaps because of a lower thermal inertia (or resistance to temperature changes) and a more even distribution of temperature. Hollow vanes would also have less weight and use less strategic material than solid vanes.

Segmentation of the outer ring has also been used to reduce thermal stresses there. Using a bellows arrangement between the inner ring and the mounting flange, as has been used on one engine, should provide some relief between the expanding and contracting diaphragm and the rigid engine casing.

OBTAINING MORE EVEN TEMPERATURE DISTRIBUTIONS

Means should be studied for obtaining more even temperature distributions in order to reduce thermal stresses, particularly during the transient conditions where the greatest differences in temperature are obtained within the body. Two general approaches appear possible:

(1) Reduce the rate of heating or cooling to allow more time for the temperature distribution to even out or approach steady-state conditions

(2) Increase the rate of heat transfer within the body to allow the temperature distribution to even out or approach steady-state conditions more rapidly

Coatings with high reflectivity suppress radiant heating so that coated parts heat more slowly than uncoated ones. This allows more time for the heat to be conducted from the surfaces to the interior of the part before the surface layer reaches its highest temperature. Transient temperature gradients are thus reduced, and this reduction, in turn, lowers the thermal stresses during radiant heating. If coatings can be maintained with high reflectivity, their use on nozzle vanes will merit further study.

Surface coatings can also insulate against the transfer of heat from the gases to the metal. Insulation may be an important function of a surface coating when the imposed thermal shock is of short duration (ref. 5).

Vane segmentation provides gaps that partially insulate the segments from each other. As a result, temperature distributions within each segment are more uniform.

Increasing the thermal conductivity of the material causes heat to be conducted more rapidly from the surface to the interior of the body. This would also reduce the temperature gradients and thermal stresses. Copper, for example, has a thermal conductivity about 15 times as great as high-alloy steels, so that cores of copper might be used to reduce temperature differences in nozzle vanes.

MATERIAL SELECTION

Using materials with greater resistance to thermal fatigue is an obvious means of improving the reliability of nozzle diaphragms. Many tests have been made to measure this property of a material.

One of the first studies on the effects of thermal shock on gas-turbine materials was published in 1938 in the German literature (ref. 7) and is described in an English survey by Bentele and Lowthian (ref. 6). Wedge-shaped specimens of nine different alloys were used to simulate the shape of turbine blades with sharp edges. The test cycle consisted of heating the specimens in an air-gas flame for 1 minute, followed by cooling in still air for 3 minutes. Tests were conducted with two flame-temperature ranges, 650° to 700° C and 850° to 900° C. Specimens failed after a number of cycles either by severe distortion of the edge or by the formation of cracks. The tests demonstrated that

(1) Excessive distortion, as well as cracking, is an important criterion of serviceability.

(2) The number of cycles to failure falls off very rapidly as the flame temperature is increased.

(3) The relative merit of different metals can change with the test conditions; one metal may be better than another for one set of test conditions and be inferior under other conditions.

(4) There is no clear relation between the resistance to thermal shock and the material properties such as tensile or creep-rupture strength.

In connection with items (3) and (4), Manson has derived indices that relate the thermal-shock resistance of brittle materials to their material properties (ref. 5). This index predicts that one material may be superior to a second material under one condition of quenching, and the order may be reversed under slightly different conditions. Experimental results for brittle materials have confirmed this relation. For ductile materials, the interactions of the metallurgical variables are more complex, and no index for thermal fatigue has as yet been derived. Tests of ductile materials, such as those cited, also show a change in the relative order of merit for different test conditions. Therefore, in tests for rating materials for a specific use, the conditions must closely approach those of the intended application. Otherwise, the results may be misleading.

Whitman, Hall, and Yaker conducted tests to determine the resistance of six cast high-temperature alloys to cracking caused by thermal shock (ref. 8). The alloys studied, listed in the order of decreasing resistance, were S–816, S–590, HS–21, 422–19, X–40, and Stellite 6.

Specimens of these alloys were cast in the form of wedges. These wedges were heated to a uniform temperature and then quenched by a stream of water across the narrow edge. When the specimens were cool, they were removed from the quenching apparatus and inspected. Failure was defined as the presence of a crack that extended across the entire width of the quenched edge.

No correlation was found between the thermal properties (coefficients of expansion, thermal conductivities, and specific heats) and the resistance of the materials to thermal cracking. The actual variations in the thermal properties of the six alloys were small, so that one of the results of this study was to show that materials with similar thermal properties can have widely different resistances to thermal cracking. The authors noted a similarity in the trends of notch impact strength and resistance to thermal cracking, which indicates a possible relation between these properties.

Variations of thermal-shock tests in which wedge- or triangular-shaped specimens are heated or cooled, or both, along one edge have been used by many investigators. At one industrial laboratory, for example, the test involved repeated heating of the edge of a triangular-shaped sample in a burner flame followed by sudden cooling of the edge by a blast of compressed air. Failure was considered to be reached when a crack had traveled completely across the $\frac{1}{32}$-inch edge of the specimen. The order of merit of the alloys studied (unpublished data) was as follows:

(1) Cast HS–21
(2) Wrought S–816
(3) Wrought L–605
(4) Wrought S–590
(5) Cast X–40
(6) Cast S–816
(7) Wrought V–36
(8) Wrought M–252
(9) Cast HE–1049
(10) Wrought Waspaloy
(11) Cast GMR–235
(12) Cast Guy alloy

Cast HS–21 withstood more than 10 times as many cycles as cast Guy alloy.

In these tests, cast HS–21 and X–40 were both superior to cast S–816, whereas they were both inferior in the tests of reference 8. This again emphasizes that thermal-shock tests must closely approximate the conditions of application if they are to be used to rate materials. As yet, results on laboratory thermal-shock tests have not been successfully correlated with service conditions. While material A may be superior to material B in a laboratory test, the second material may be better in actual service.

Wrought S–816 was superior to cast S–816 in the last-mentioned tests. Whether the wrought condition is always better than the cast condition of an alloy is not known. Cobalt-base alloys were superior to nickel-base alloys in both the cast and wrought conditions in the data cited, but again this may not be true in general.

SLOTS

The concentration of stresses at the bases of notches is well known to designers. Designers constantly try to eliminate notches or to provide generous rounding to lessen their effect. Axial cracks in the inner rings of nozzle diaphragms start at the trailing edges of the vanes, where the slots that are cut in the ring to receive the vanes concentrate the stresses from the thermal expansion and contraction of the system. The greater rounding of the slots at the leading edges reduces stress concentration there, and, hence, cracks do not occur there. Flaring the bottom of the vanes to increase the rounding would reduce vane cracking at the trailing edge, while designs that eliminate the slots would eliminate the stress concentration problem as well.

OPERATING PRACTICES

Frequent inspection and repair seem necessary to avoid increases in nozzle-diaphragm damage beyond safe limits; however, how often maintenance will be needed is difficult to know from the present data. The number of starts and stops, as well as the severity of operation, are probably more important than the total time of operation. DIR's, which give only the total time of operation, indicate that edge cracking will be present in practically all cases where the nozzle diaphragms have operated 200 hours or more. Other types of cracking occur at slower rates.

Fortunately, cracks in nozzle diaphragms are not important causes of engine failures or aircraft accidents. Their rate of propagation is apparently slow, and their presence is not damaging. Present inspection and maintenance practices appear adequate from this standpoint.

ADDITIONAL INFORMATION NEEDED FOR IMPROVEMENT OF OPERATIONAL RELIABILITY

From the standpoint of cost, damage to nozzle diaphragms is important. Because of this damage, jet engines require frequent repairs and replacements. Additional information is needed to extend the time between overhauls.

Much of the information needed is concerned with mechanical designs to reduce or eliminate thermal stresses. Free-floating vanes, ring segmentation, and bellows attachments can reduce the constraint between different parts. Vane segmentation, hollow vanes, composite vanes with cores of high thermal conductivity, and vanes made of flexible materials are possibilities for reducing the constraint between different sections of the same part. Factors that concentrate thermal stresses should be eliminated or their effect reduced.

Information on surface coatings that maintain high reflectivity and insulate against heat conduction is needed.

Fundamental research should be conducted on thermal fatigue. Tests that will establish the relative merits of different materials under service conditions should be developed, as should also materials with better resistance to thermal fatigue, impact, and overtemperature. Design parameters that relate the number of cycles of thermal fatigue before failure occurs with the geometry, heat flow, and material properties should be established. Production specifications should include conducting such tests as are found satisfactory on the completed components.

CONCLUDING REMARKS

Repeated thermal stresses are the most important factor causing damage to nozzle diaphragms. These stresses arise from nonuniform temperature distributions and from constraints which prevent the nozzle diaphragm from expanding and contracting freely during heating and cooling. Repetition of these stresses during starting, accelerating, and stopping eventually causes cracks along the edges of nozzle vanes, at the trailing edge of the vane slot in the inner ring, and in the weld joining the inner ring to the mounting flange. These cracks usually do not progress to complete failure of the unit.

Although cracks in nozzle diaphragms are frequent and add to the cost of engine maintenance, they rarely cause engine failure. Damage to nozzle diaphragms is not a problem with respect to reliability. Cracks in the nozzle vanes can be safely handled by inspection and repair at frequent intervals.

Rapid deterioration of the nozzle diaphragms occurs when, through malfunction of the combustor, severe overtemperature occurs. Overtemperature may cause sections of nozzle vanes to be burned off. Nicks and dents in the exposed surfaces are frequently caused by foreign objects.

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CHAPTER VII

TURBINE BUCKETS

By G. M. Ault

SUMMARY

Air Force service records indicated that for several models of the jet engine turbine-bucket replacements are very frequently necessary because of bucket cracking or fracture. Bucket fracture occasionally occurs in flight. In engines with single-stage turbines, the fragments from fractured buckets are often ejected through the discharge nozzle with no further damage and little loss in thrust. There is a risk associated with permitting bucket fracture in flight, however, as attested by the fact that in 1953, of 205 Air Force flight accidents attributed to jet-engine malfunction, 16 were traced to turbine-bucket failure. In the case of multistage turbines, the probability of catastrophe from failure of a turbine bucket is much greater. A failed bucket in an early stage may destroy the buckets in successive stages and stop or destroy the engine.

Turbine buckets are subject to the combination of centrifugal stress, vibratory stress, high and rapidly changing temperature, and a corrosive atmosphere. Turbine buckets can fracture by stress-rupture or fatigue, or a combination thereof. Fracture can be accelerated by damage from overtemperature or overstress, or damage or cracks resulting from thermal fatigue, or perhaps from corrosion, or by nicks caused by solid objects in the gas stream.

Suggestions are made for reducing the likelihood of flight failures by certain considerations in design, operation, and inspection.

INTRODUCTION

Turbine buckets are subjected to a more severe and complex combination of stress and temperature than any other jet-engine component. The buckets in current engines are subjected to very high centrifugal stresses at temperatures of the order of 1500° F. Also, they are subjected to a hot corrosive atmosphere and to gas impulses that may cause the buckets to vibrate. Rapid heating and cooling induces thermal stresses in them. Hence, turbine bucket failures are very common and occur in a variety of ways.

Buckets can fail either in the airfoil or in the root-fastening region. Since root failures are not common, and root stresses and temperatures are such that difficulty can be relieved by design using current engineering methods, the emphasis in this paper is on airfoil failures.

The various causes of bucket airfoil failures are reviewed, manifestations of failure are described, and, in some cases, methods of eliminating the failure are indicated. A discussion, based on considerations of bucket-failure causes and mechanisms, is presented of the measures required in design, manufacture, inspection, and overhaul to avoid flight accidents.

MILITARY STATISTICS OF BUCKET FAILURES

MAGNITUDE OF BUCKET-FAILURE PROBLEM

The importance of bucket failures in limiting engine reliability is defined in two ways; first, by the severity of damage to the engine and the airplane caused by a bucket fracture, and second, by the frequency with which buckets must be replaced. The severity of damage caused by bucket fracture is described later.

U.S. Air Force records were studied to determine the frequency of bucket failures, the life of buckets in service engines, and the causes of failure. Significant data, particularly in regard to life of hot-section components, are difficult to obtain. Turbine buckets are frequently replaced in the field, and recording of every bucket replacement had not been required in any field record until late 1954, and then for only one engine model. Data from these records are not yet available. From examination of the Disassembly Inspection Reports (DIR's) the operating time on a bucket cannot be determined with certainty, even on an engine being overhauled for the first time; all or some of the buckets may have been replaced in the field. Thus, an accurate history of bucket life and bucket replacements for an engine cannot be obtained.

An approximation of the number of bucket failures taking place can be obtained from a study of Engine Removal Reports (ERR's) and the DIR's. The ERR describes the reasons for removal of an engine from an airplane when the engine is replaced by another. No removal is recorded, however, if a bucket is replaced without removing the engine from the airplane, or if the engine is removed from the airplane, the bucket replaced, and the engine immediately reinstalled in the same position in the same airplane. The ERR data tend to minimize the number of bucket replacements since, of all hot-section components (excepting perhaps the tailcone), buckets are most easily replaced.

A review was made of the ERR's and the DIR's covering a 3-month period for jet engines having their first overhaul ("new" engines). (Because of difficulty in obtaining data, the 3-month period was not the same for the ERR's and for the DIR's.) Results from this review are given in table I for six engine models having more than 50 major overhauls during the study period for DIR's.

A significant figure, abstracted from the ERR's, shows the relative importance of the turbine bucket in causing engines to be removed from an airplane for repair or overhaul. From 2 to 19 percent of all engine removals are for the specific purpose of replacing turbine buckets (table I, column 4). If the figure is given on the basis of percentage of engines removed because of engine or accessory failure (omitting engines removed because of foreign-object damage, unstated reasons, etc.), turbine-bucket failure is responsible for from 3 to 44 percent of the removals (column 5). The engines for which these removal rates were found had average operating times from 50 to 155 hours (column 3).

Although the table includes the percentage of engines overhauled because of turbine-bucket failure (column 8), the values are not particularly meaningful because the buckets are often replaced in the field. Examination of the DIR's that gave bucket failures as a cause for overhaul indicated that most of the replacements could have been made in the field. Sending a particular engine to overhaul was frequently arbitrary; for example, it was felt that a complete examination of the engine was warranted.

A more useful figure to consider is the percentage of engines examined during overhaul that required replacement of turbine buckets because inspection showed cracking or fracture (from causes other than foreign-object damage). This value was between 2.6 and 36 percent (column 9) for the various engine models and applications.

TABLE I.—FREQUENCY OF ENGINE REMOVAL FROM AIRCRAFT, ENGINE OVERHAUL, AND BUCKET REPLACEMENT DURING OVERHAUL BECAUSE OF BUCKET FAILURE

[Covers only engines in for their first overhaul.]

Engine code	Airplane code	Operating time on average engine in service (since new or last overhaul), hr	Engine removals because of bucket, percent of all removals	Engine removals because of bucket, percent of engine removals because of engine failure	Number of engines overhauled	Operating time on average engine at overhaul, hr	Engines overhauled because of bucket, percent of all overhauls	Engines having buckets replaced during overhaul, percent of total
A-7	F-2	115	5.5	14	136	160	0.7	7.4
B-3	B-4	125	3	6.5	73	247	1.4	28.8
B-7	B-2	50	2.2	3.5	87	61	0	27.6
B-9	B-4	140	3.3	10.3	76	55	0	2.7
B-10	F-3	130	19	44	161	88	1.9	5.6
C-7	F-1	155	5.2	9.9	210	305	.5	1.4

The average operating time to overhaul for the several engine models examined ranged from 55 to 305 hours.

Figure 8 of chapter II shows that many buckets are replaced in the field maintenance and repair program. The percentage again varies with engine model and application.

It is apparent that (1) turbine-bucket damage or failure is an important cause of engine removal; and (2) that whenever engines are examined in field maintenance or in overhaul, large percentages of the engines require bucket replacement. The operating time for the average engine in service or being overhauled is very low, much less than any desired overhaul time. Since bucket failures, other than those from foreign-object damage, are time- or cycle-dependent (as is shown later), the frequency of failures can be expected to increase as average operating time on the engines increases.

MODES OF BUCKET FAILURE

The DIR's were studied to learn the mechanisms of bucket failure. Some of the information obtained from them is summarized in table II. (The ERR's and the field maintenance data that were available did not give this information.) The DIR's indicate the location of cracks or fracture on the bucket airfoil, and these data have been tabulated. The probable cause of failure could best be determined by a careful examination of the buckets, but since they were not available, the probable cause has been deduced from the crack locations and from experience with these engines in NACA test-stand studies. In addition to the six engine models being overhauled for the first time for which failure data are given in table I, data are included in table II for one engine model (code C-1) that had one or more previous overhauls.

For the A and B engines, the prime reasons for bucket replacement during overhaul were leading-edge cracking, probably caused by thermal fatigue, and tip cracking, probably caused by mechanical fatigue. In only two cases were the buckets actually fractured (a piece missing). One of these fractures was undoubtedly the result of trailing-edge tip fatigue; the cause of the other is unknown.

TABLE II. MODES OF BUCKET FAILURE INDICATED BY DIR'S

[Three-month period.]

Engine code	Reason for replacement (number of engines with each failure type) (a)	Probable cause (b)
A-7	7, Tip cracks	Mech fatigue
	1, Contour crack	Not known
B-3	8, Tip cracks	Mech fatigue
	8, Heat check crack, LE	Thermal fatigue
	5, Cracks, LE	Not known
	1, TE tip broken off	Mech fatigue
B-7	12, Tip cracks	Mech fatigue
	12, Heat check cracks, LE	Thermal fatigue
B-9	1, Tip crack	Mech fatigue
	1, TE crack	Mech fatigue
B-10	1, TE tip gone	Mech fatigue
	2, LE tip gone	Mech fatigue
	1, 1⅛″ of tip broken off	Not known
	2, Buckets missing	Not known
C-7	2, Tips scuffed	Excessive creep
	1, 1″ of tip broken off	Stress rupture or SR plus mech fatigue
	1, Cracks and buckles	Not known
C-1[c]	39, 1″ To 2″ of tips broken off	SR or SR plus mech fatigue
	16, Excess stretch	Creep
	1, Crack 1″ from tip	SR or SR plus mech fatigue

[a] Leading edge, LE; trailing edge, TE.
[b] Mechanical, mech; stress-rupture, SR.
[c] 109 engines previously overhauled.

In the C-7 engine, the causes of bucket failure were somewhat different: tip scuffing, probably the result of excessive creep or elongation; and fracture about an inch from the tip, probably the result of stress-rupture or stress-rupture plus mechanical fatigue. An earlier model of this engine, the C-1, using an alloy having somewhat lower stress-rupture strength, had a much higher failure rate. Of 109 engines (C-1) in overhaul, 56 required bucket replacement; bucket failure was the reason for sending 39 of these engines to overhaul. In these 39, the buckets had fractured along the chord, probably because of stress-rupture or a combination of stress-rupture and mechanical fatigue; 16 others had excessive stretch, a result of excessive creep; and in only one engine was a bucket replaced because a crack was found. There is an important difference between these bucket failures and those of the A and B engines. In the A and B engines cracks caused by tip mechanical fatigue and leading-edge thermal fatigue progress slowly enough that they could probably be found by inspection, and bucket fracture in flight could be reduced to a minimum. In the older models of the C engine, the fractures along the chord resulting from stress-rupture or stress-rupture plus mechanical fatigue propagate from cracks so rapidly that current inspection techniques probably cannot prevent blade fracture in flight. As is shown later, excessive creep may be used as a replacement criterion in some instances.

In subsequent sections of this paper, these various failure mechanisms and related inspection methods are described in more detail.

TIME DEPENDENCY OF BUCKET FAILURES IN SERVICE

In chapter II, attempts are made to determine whether bucket failure rate increases with engine operating time. It is found that in the engines where failure rates were sufficiently high to provide an adequate sample from the limited data available, failures were more likely with increased operating time. The B–3 engine exhibited an ideal type of failure curve (fig. 12(b) of ch. II); there was a definite grace period before the first bucket failure. Flight failures of buckets could be eliminated if failure curves had a known grace period (all buckets could be replaced at the end of the grace period). Longer grace periods are desired, of course. If "wear-out" failure curves do not exhibit a grace period, it is imperative that the engine be redesigned or the operating environment changed to provide one. If failure rates are independent of time ("chance"-type failures), changes must be made to eliminate the failure or reduce the failure rate to acceptable values. With certain types of failure mechanism incipient fracture can be found at regular inspection periods, thus avoiding bucket fracture in flight, and it may be possible to safely continue operation while a correction for the failure is being found.

EFFECT OF BUCKET FAILURE ON ENGINE

SINGLE-STAGE TURBINES

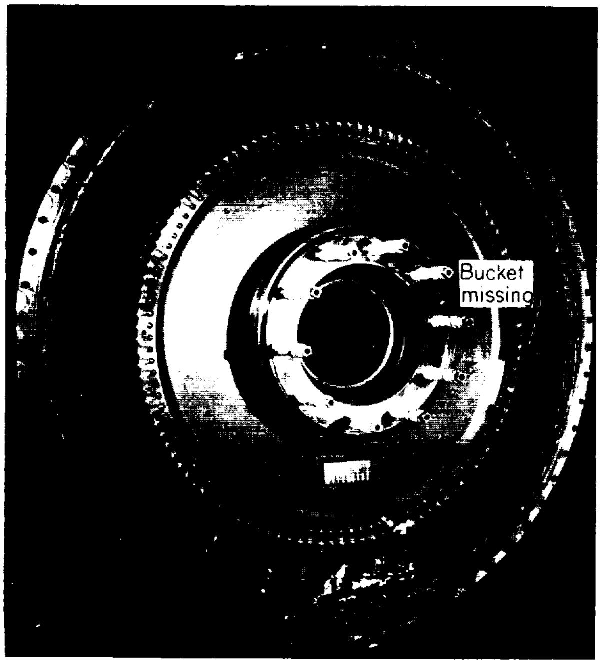
Failure of a bucket in a single-stage turbine is not usually serious. Many times a pilot is unaware that a bucket has failed until it is found missing on post-flight inspection. The fragment of the failed bucket goes out the tailpipe. In some cases, however, fragments of failed buckets have been known to catch between the tips of the rotating buckets and the turbine shroud. This can cause appreciable damage, and may wreck an engine. An example in which serious damage occurred is shown in figure 1. Here the fragment of the failed bucket (no. 54) has broken off two additional buckets and severely damaged the tips of all others on the wheel.

FIGURE 1.—Damage to single-stage turbine resulting from bucket failure (number 54).

The importance of failure in single-stage turbines is indicated by the fact that of all (205) jet aircraft accidents caused by engine failure or malfunction in the year 1953, turbine buckets caused 8 percent (fig. 13 of ch. II).

MULTISTAGE TURBINES

Damage resulting from failure of turbine buckets will likely be much more severe in multistage turbines than it has been for single-stage turbines. In multistage turbines, a fragment from a first-stage bucket failure must pass through subsequent stages where it can cause appreciable damage. A failed three-stage shrouded turbine is shown in figure 2. Failure of a second-stage



(a)

(a) Second stage of turbine; all bucket shrouds gone.

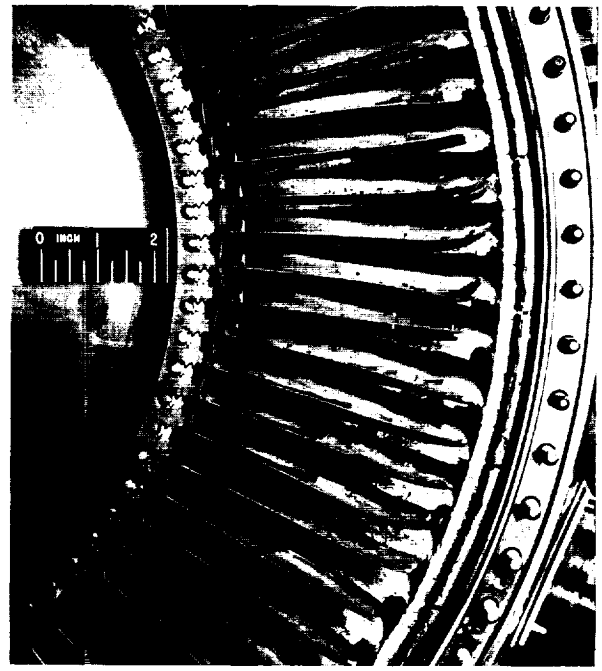
FIGURE 2.—Damage to three-stage turbine resulting from failure of second-stage bucket.

bucket broke the tips and shrouds of all the second- and third-stage buckets. The probability is very high that an engine will no longer produce thrust if a bucket fails in any stage except the last; therefore, the reliability must be much

(b)

(b) Closeup of fractured second-stage bucket.

FIGURE 2.—Continued. Damage to three-stage turbine resulting from failure of second-stage bucket.



(c)

(c) Third stage of turbine; all bucket shrouds gone.

FIGURE 2.—Concluded. Damage to three-stage turbine resulting from failure of second-stage bucket.

greater for buckets of multistage turbines than has been required for single-stage turbines.

MODES OF FAILURE

Distributions of temperature and centrifugal stress along the airfoil of turbine buckets of two typical jet engines when operated at full engine power are shown in figure 3. Temperatures were measured with thermocouples embedded in the buckets. The stress values for the two turbines are representative of high and low stress levels in current production engines.

In addition to the centrifugal load, the gas forces impose a bending load on the bucket airfoil. The designer partly compensates for the gas load by tilting the airfoil slightly downstream so that centrifugal force will induce opposite bending. The bending force can be completely cancelled at only one plane along the airfoil span. Further, since the gas bending load reduces with altitude and the engine operates at essentially constant speed, the gas bending load can be cancelled for only one altitude. If the bucket is tilted to compensate for the sea-level

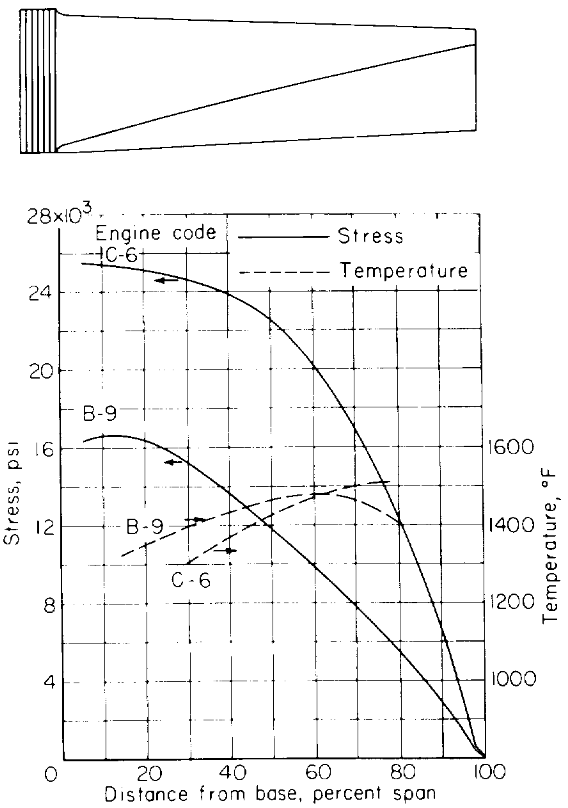


FIGURE 3.—Temperature and centrifugal stress along airfoil of turbine buckets of two typical jet engines.

gas load, it will be overtilted for the gas load at altitude.

STRESS-RUPTURE

In the range of temperatures in which turbine buckets operate, there is, for each bucket material, a finite time before fracture for each combination of stress and temperature. This time is called the stress-rupture life. Stress-rupture curves for a representative alloy, S–816, are shown in figure 4.

If the distributions of centrifugal stress and temperature of a bucket are known (as shown in fig. 3) bucket life at each point along the bucket length can be predicted from interpolations of stress-to-rupture curves like those of figure 4. If predicted life is plotted against bucket span, the curves for current production engines usually have the form shown for two engines in figure 5. The minimum in each curve identifies the critical section or "critical zone" in the bucket—the point where the bucket is most likely to fail by stress-

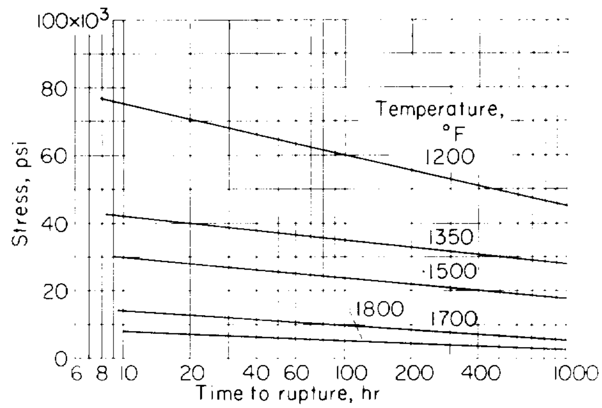


FIGURE 4.—Stress-rupture curves for representative bucket alloy, S–816. (Data from ref. 13.)

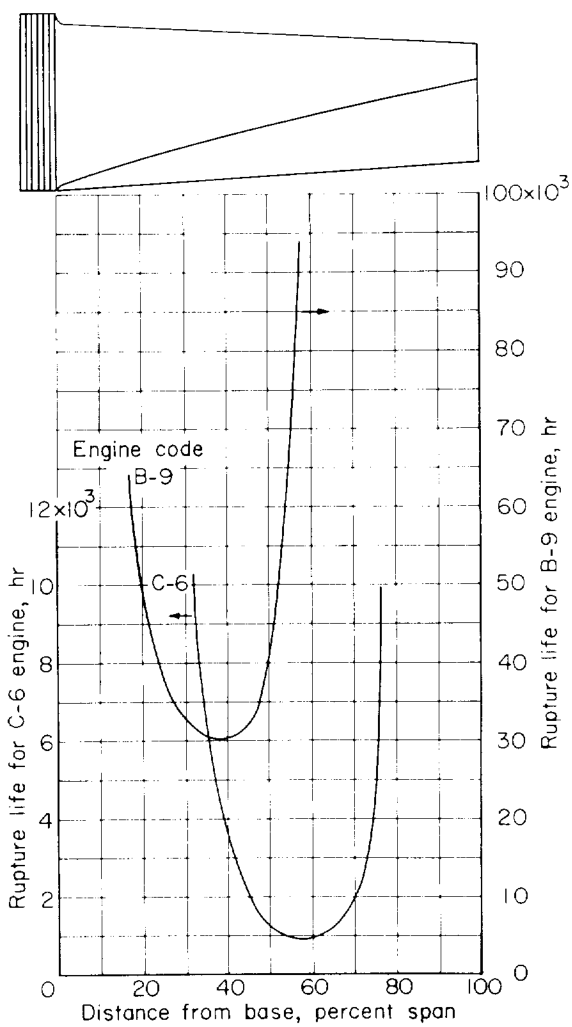


FIGURE 5.—Stress-rupture life along airfoil for two typical turbine buckets.

rupture. The wide difference in minimums for two production turbines (900 and 30,000 hr) is of interest. Large safety factors (as is indicated for engine B) favor reliability.

The curves of figure 5 apply to the stress and temperature conditions of full engine power. Since stress-rupture life is very sensitive to stress and temperature, operation at less than full engine power (e.g., at cruise) or higher than full engine power (overspeed or overtemperature) will increase or decrease the time to fracture by stress-rupture over that indicated by the minimums of figure 5.

For example, the data for S-816 alloy (fig. 4) indicate that at constant temperature a 10-percent change in stress (approximately a 5-percent change in engine speed) changes life by a factor of about 3. A 10-percent change in temperature (from 1500° F) changes life by a factor of about 100. Although the effect differs in amount from alloy to alloy, all bucket alloys are similar in that changes in stress-rupture life are large for small changes in temperature or stress.

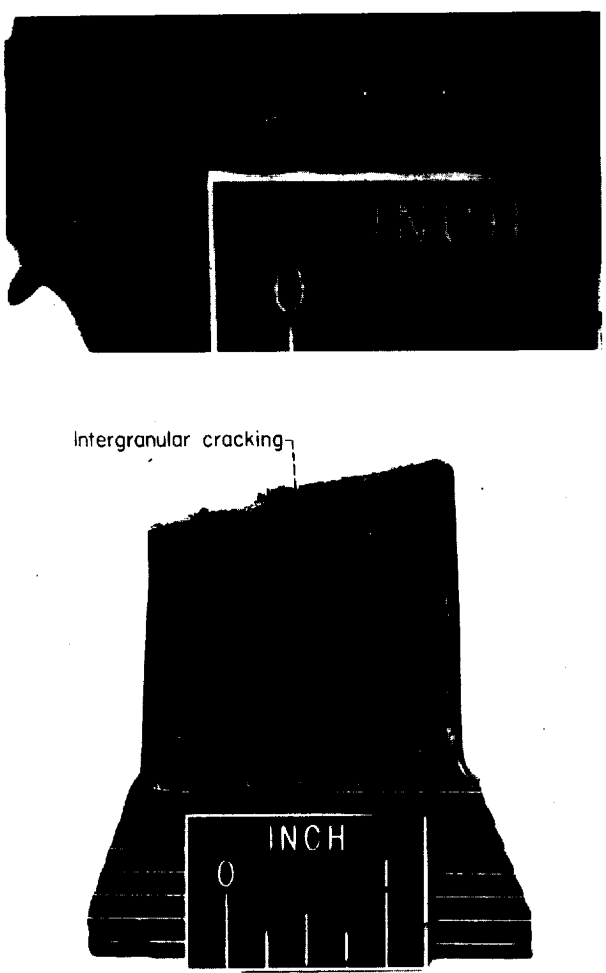


FIGURE 6.—Typical stress-rupture fracture.

In general, the life predicted from stress-rupture properties is the longest that the bucket could be expected to run at full engine power, since the only stress considered is that induced by centrifugal force. Other environmental factors, such as vibratory stress, corrosion, thermal stress, and impact, will reduce the life below this value.

Incipient stress-rupture failure appears as irregular intergranular cracking in a narrow zone of the airfoil span corresponding to the minimums of the curves (the "critical zone") illustrated in figure 5. Usually the cracks are not confined to the leading or trailing edge, but occur at random across the chord. Buckets that have completely fractured (fig. 6) have many cracks on the airfoil surface adjacent to the fracture. Since the fracture is intergranular, the fracture surface will generally be rough.

Detection of stress-rupture cracks during engine inspections cannot be counted upon as a method of avoiding bucket fracture, because complete fracture generally follows cracking very shortly.

BUCKET ELONGATION OR CREEP

At any condition of stress and temperature shown on the stress-rupture curves of figure 4, the test specimen elongates with time in a manner described by the ideal curve of figure 7. When load is applied, the specimen elongates elastically then plastically at a decreasing rate (first-stage creep) until the rate becomes approximately constant (second stage). Finally,

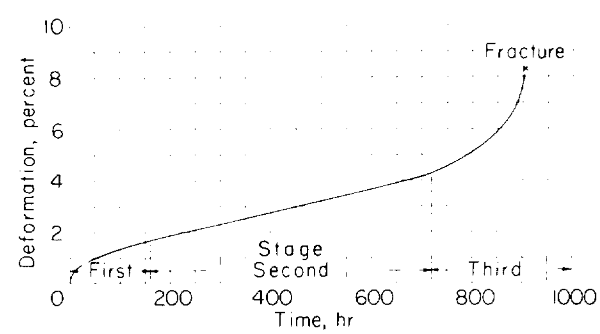


FIGURE 7.—Ideal creep curve (stress and temperature constant).

the rate begins to increase (third stage) until fracture occurs.

Unit elongation, or strain, is not uniform along a bucket length because of the nonuniform stress and temperature conditions (fig. 3). A typical distribution of strain along the length of one particular bucket after 9 hours and after 28 hours of operation is shown in figure 8. The zone of maximum creep corresponds to the zone of minimum stress-rupture life of figure 4.

Although some zones of the bucket may suffer very large local strains, total bucket elongation may be small. Figure 9 shows plots of the strain in the ½-inch zone of maximum creep and in the total bucket elongation against time.

The designer is interested in knowing the allowable centrifugal stress and temperature because they affect both the time to rupture and the time in which elongation of the bucket will deplete the operating clearance and cause the bucket tip to rub the shroud band. Typical design curves giving time and stress to cause various amounts of strain or fracture are shown in figure 10 for the alloy Inconel 550. Because

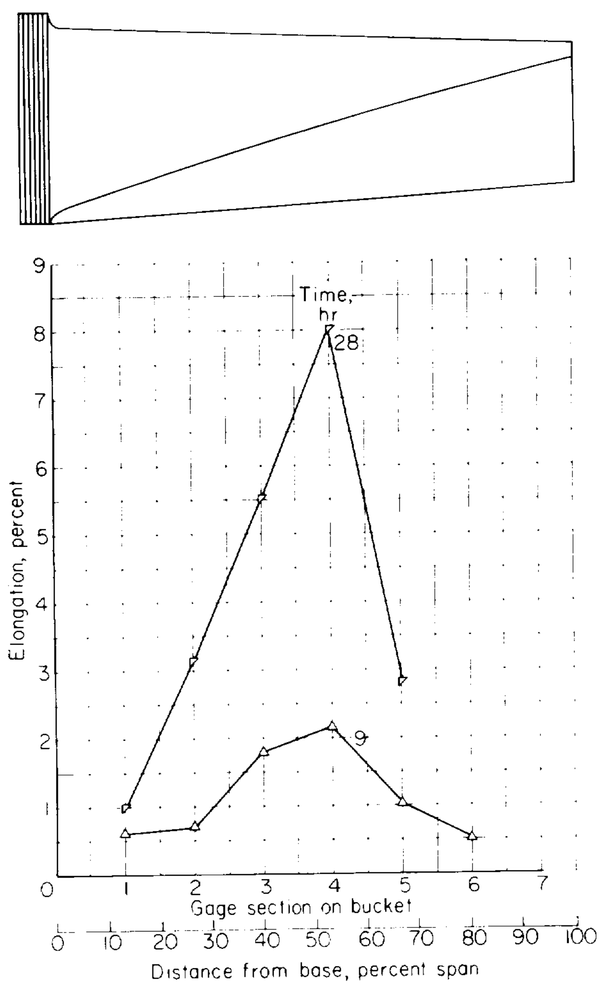


FIGURE 8.—Elongation along bucket. Engine C–1; Hastelloy B alloy. (Data from ref. 14.)

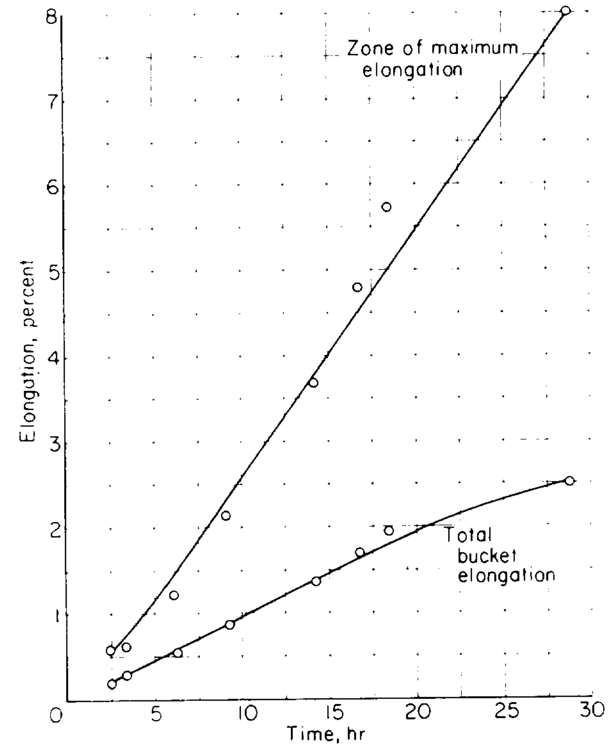


FIGURE 9.—Comparison of total bucket elongation with ½-inch zone of maximum elongation.

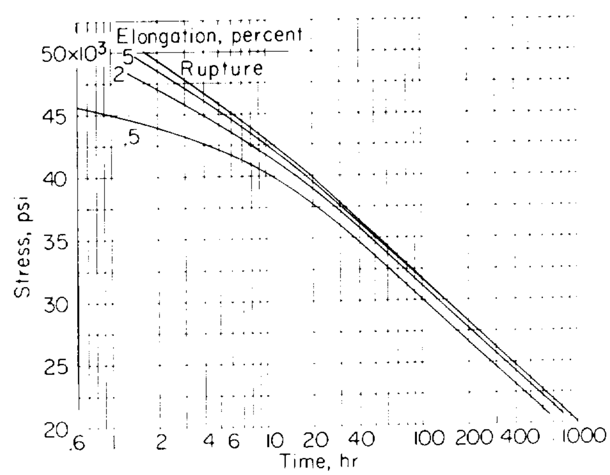


FIGURE 10.—Typical design curves for Inconel 550 at temperature of 1,500° F. (Data from ref. 15.)

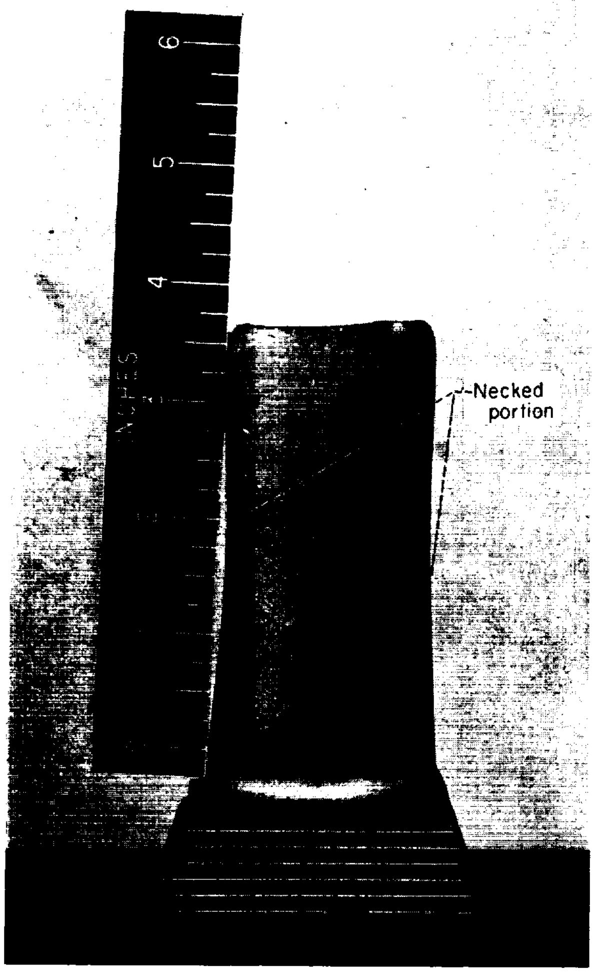


FIGURE 11.—Necked bucket.

the strain in a bucket is highly localized, total elongation is usually small; stress-rupture fracture usually occurs before the allowable total elongation is exceeded. Thus, stress-rupture usually becomes the more important design criterion.

Creep appears as bucket stretch, bucket-shroud rubbing, or bucket necking (local reduction in area). Figure 11 shows a necked bucket. Necking of this degree is not common; furthermore, necking cannot be relied upon to indicate incipient bucket fracture because the time between its first appearance and fracture is usually too short and unpredictable.

BUCKET ELONGATION AS REPLACEMENT CRITERION

If the alloy has large elongation to fracture and if the buckets fracture because of stress-rupture, elongation or creep may be used as a basis for bucket removal to avoid fracture in flight. Because the strain that might indicate impending fracture is highly localized, it may be desirable to place scribe marks on several of the airfoils to permit making elongation measurements in the specific important area. If one or two reached a specified elongation, all buckets would be replaced. This scribing technique has been used for several years in test-cell evaluation of materials, and no bucket fractures have initiated at the scribe marks.

Several practical problems arise if bucket elongation is to be used as a bucket replacement criterion. One problem is the variability of elongation of buckets within one heat, among heats, and with variation in engine operation history. The heat from which a group of buckets is produced is usually not isolated, and any one wheel may contain buckets from several heats. Figure 12 shows elongation curves and times to rupture for stress-rupture specimens cut from six buckets from an engine picked at random from Air Force stock. The specimens were tested in stress-rupture machines at 23,600 psi and 1500° F. The data obtained showed little scatter; total elongation ranged from 15 to 22 percent, and time to failure from 65 to 130 hours. If data such as

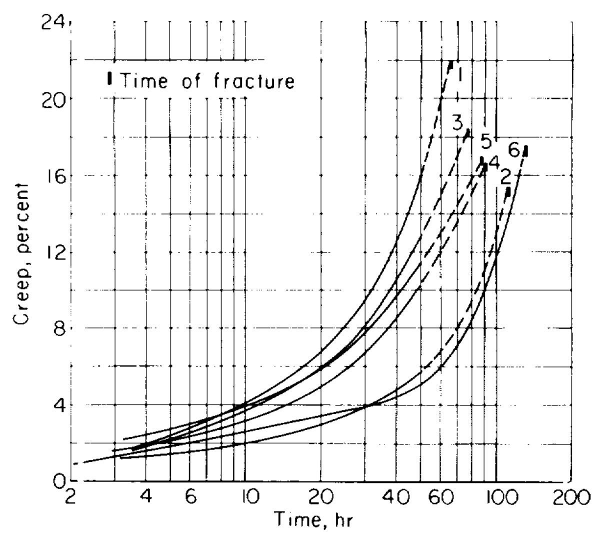


FIGURE 12.—Variation in elongation of specimens cut from random buckets of Air Force stock tested in laboratory stress-rupture machine. Temperature, 1,500° F; stress, 23,600 psi.

these are available for a bucket material, one may select, as a basis for replacements, the minimum total elongation that might be expected of a sample, in this case, 15 percent (specimen 2). In addition, an allowance should be made to ensure that a bucket failure will not occur between inspection periods. For example, if it were assumed that an engine might operate 20 hours at full power between inspections, the buckets should be removed at the minimum elongation obtained 20 hours prior to fracture. For the specimens tested, the buckets should be removed when 11 percent elongation had been reached (20 hr before the fracture of specimen 2).

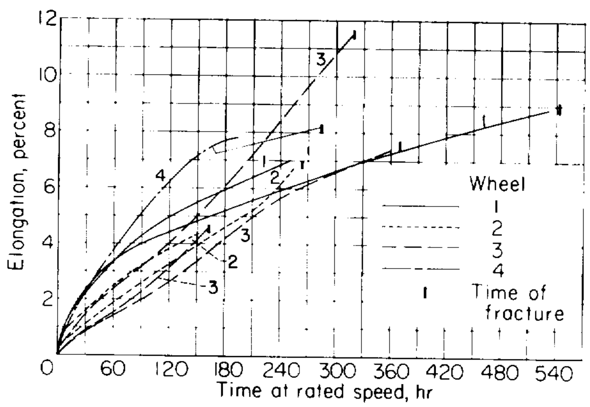


FIGURE 13.—Elongations of S–816 buckets run in different wheels.

In the engine, variation in creep rate of the buckets may be even greater than among these specimens. Figure 13 shows elongation of the 1-inch zone having maximum elongation for eight S–816 buckets operated in a code C engine. Buckets with the same number were operated in the same wheel at the same time. Solid curves are drawn to the time at which the last data point was taken; the short vertical line beyond each curve indicates the time when the bucket fractured. The final elongation at time of fracture was not known. Apparently because of the scatter of elongations even in one engine (for example, engine 1 or 2) bucket elongations for removal would have to be set quite low if all turbine bucket failures were to be avoided. From the data herein, all failures could have been prevented if buckets had been removed after 4 percent elongation. The amount of useful life that may be thrown away can be reduced by reducing production variability and by assembling buckets of only one heat or lot on one wheel.

The usefulness of elongation as a replacement criterion may be reduced by several factors; one of them is use of a bucket alloy having low elongation to fracture. Inconel 550 buckets were operated along with S–816 buckets in both engines 1 and 2. Although they ran much longer than those of S–816, with minimum failure times of 450 hours, the last elongations measured prior to fracture were less than 0.3 percent in the critical zone on any of these buckets. Obviously, elongation could not be used to predict failure of these buckets. Later it will be shown that creep measurements cannot predict bucket failure if the buckets fracture by mechanical fatigue or by thermal fatigue, since elongation before fracture can be negligible.

In summary, the use of bucket elongation to predict failure is limited to those buckets of alloys that have large elongation to fracture and in which the cause of fracture is stress-rupture. If a low-elongation alloy is used, or if the life of the bucket is decreased from the stress-rupture life because of thermal or vibratory fatigue, elongation measurements will be of little use except to warn that a severe combination of stress and overtemperature has been encountered.

VIBRATORY STRESS OR FATIGUE

The most important source of excitation of bucket vibrations is the impulse given to the buckets when they pass through the wakes of the nozzle vanes. The flow from individual combustion chambers likewise introduces irregularities into the gas flow impinging on the buckets and may also excite vibrations. The frequency of vibratory impulse imparted to the bucket is thus a function of the number of nozzle partitions, or combustion chambers, and of engine speed. The buckets can vibrate (fig. 14) in simple bending, in torsion, or in complex combinations. Different patterns or modes of vibration are induced, depending upon the frequency of the excitation (fig. 15 and ref. 1). Whenever the natural frequency of the bucket is the same as or a harmonic of the excitation frequency (a condition of resonance) the amplitudes and the

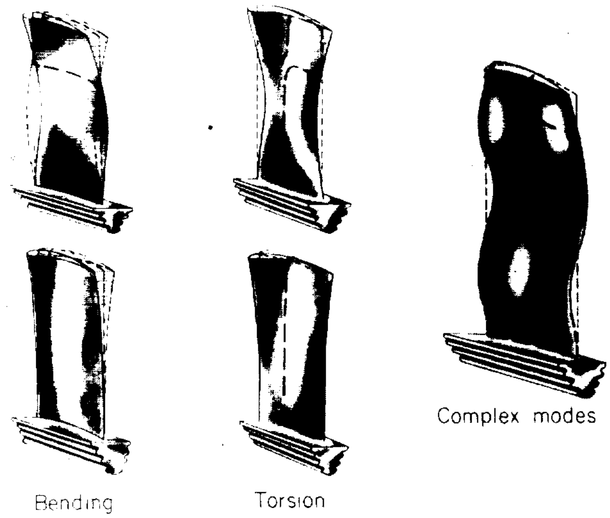


FIGURE 14.—Modes of turbine-bucket vibration.

induced stresses may become very large. Bucket vibrations at natural frequencies in the range from 1000 to 10,000 cycles per second have been measured in operating engines.

Typical fatigue curves for a high-temperature alloy (N-155) at room temperature and at elevated temperature are shown in figure 16. At room temperature, the curve becomes horizontal at a stress of 53,000 psi (the endurance limit). Thus the material can stand a vibratory stress less than 53,000 psi indefinitely without fracture. At elevated temperatures materials do not show an endurance limit, at least for times or cycles to failure of practical interest; that is, with a particular applied stress there is a corresponding time to fracture. The elongation at fracture in pure fatigue, either from bending or from tension, is negligible.

Typical fatigue fractures are shown in figure 17. Fatigue cracks are usually found singly as shown for the transverse cracks of figures 17(a) and (b) and the radial tip crack of figure 17(c), as contrasted with stress-rupture cracks, which are usually many in number. Fatigue cracks may be found almost anywhere on the airfoil, whereas stress-rupture cracks are confined to the previously defined "critical zone." Fatigue cracks are

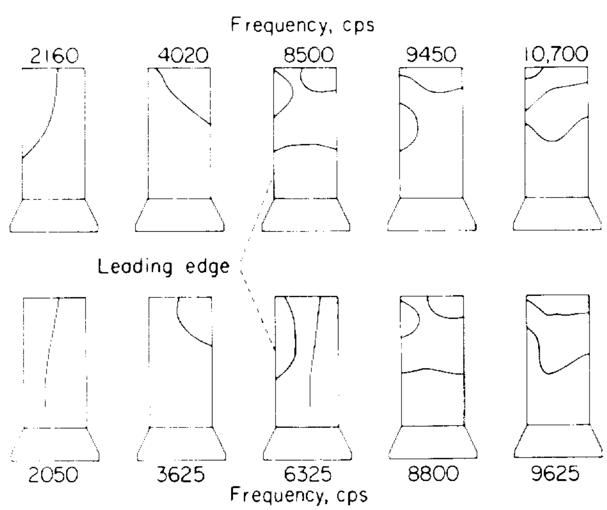


FIGURE 15.—Nodal patterns for various frequencies of vibration. Fundamental frequency, 1,270 cycles per second. (From ref. 1.)

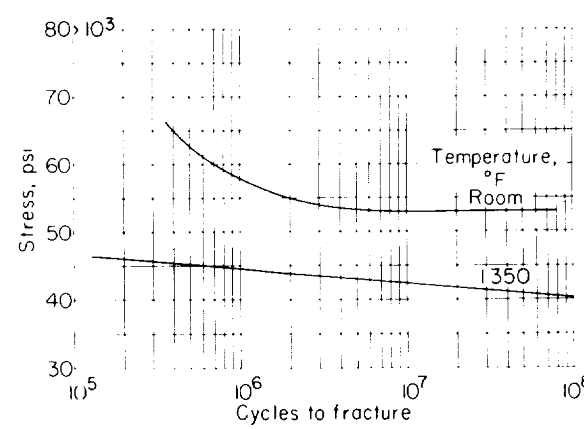
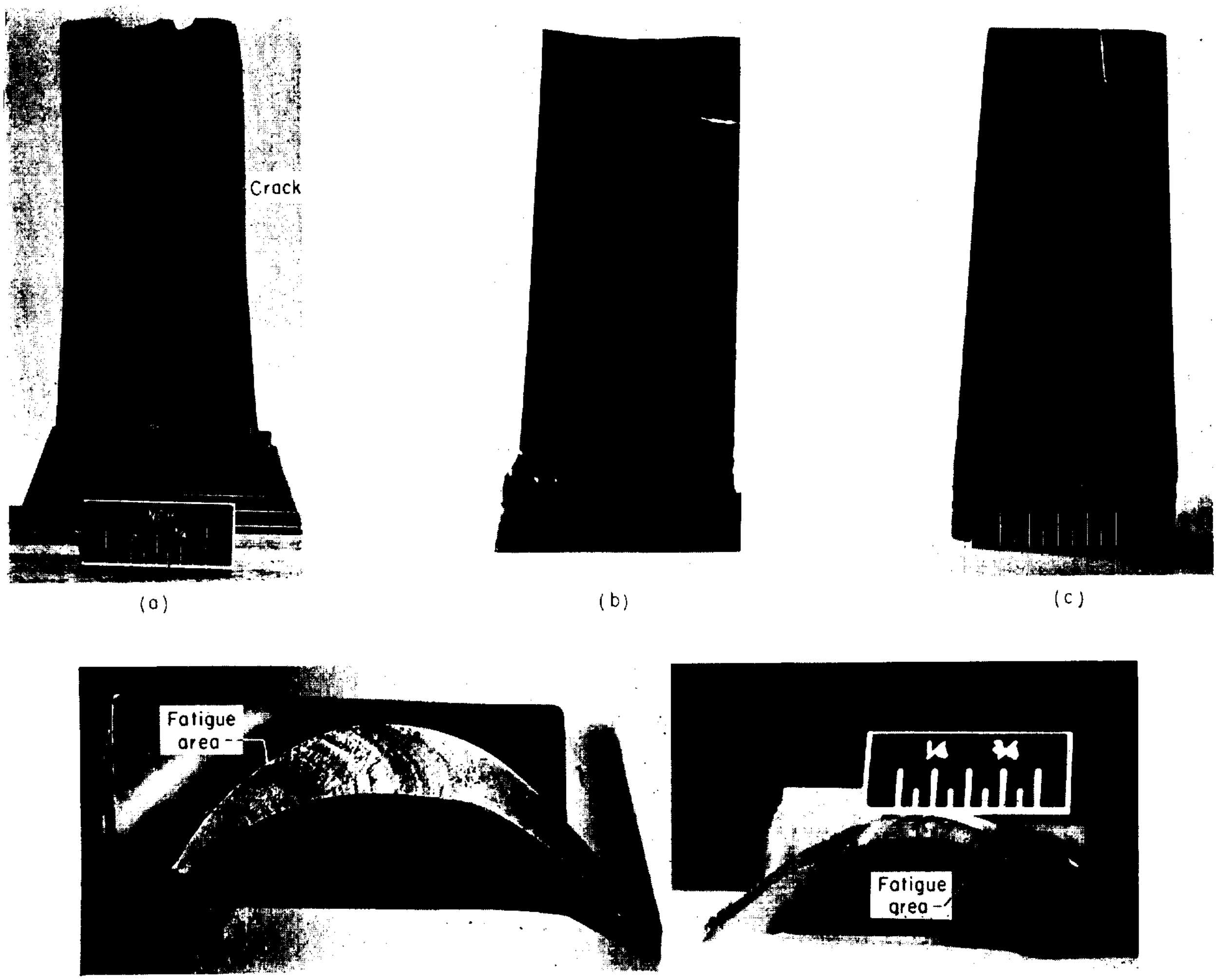


FIGURE 16.—Typical fatigue curves for high-temperature alloy (N-155). (Data from ref. 16.)



(a) Transverse cracks. (b) Transverse tip cracks. (c) Radial tip cracks. (d) Two typical surfaces.

FIGURE 17.—Typical fatigue failures.

transgranular and an area of the fracture surface will be relatively smooth (fig. 17(d)). The smooth area frequently will have been subdivided by semicircular bands called progression rings.

In many cases the vibratory stress is superimposed upon the centrifugal stress. The result of this superimposed vibratory stress is to reduce stress-rupture life and to reduce the elongation at fracture.

The effect of superimposed vibratory stress upon stress-rupture life is shown in figure 18. Here, stress ratio (vibratory stress divided by mean stress) is plotted against life for three levels of mean stress for the alloy S-816 at 1500° F. If a turbine bucket were operating at 1500° F and a stress of 18,600 psi without any vibratory stress, the bucket fracture by stress-rupture could be expected at 1000 hours. If, however, a vibratory stress of 60 percent of the centrifugal stress were superimposed (point a), the expected life of the bucket would be reduced to 440 hours. The magnitude of this effect varies with mean stress, temperature, and alloy.

An example of how a vibratory load superimposed on the centrifugal load reduces the elongation at fracture is shown in figure 19.

When vibratory stress is superimposed on the centrifugal stress the appearance of the frac-

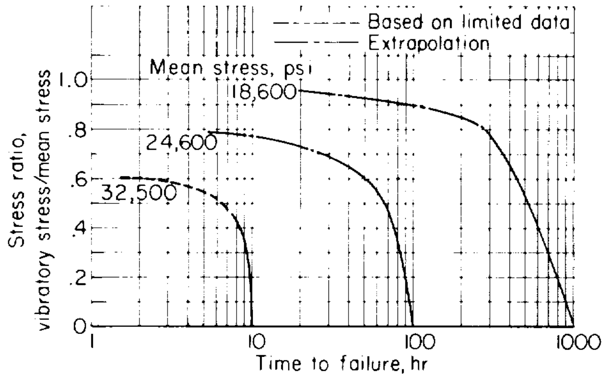


FIGURE 18.—Effect of vibratory stress on time to rupture of wrought S-816 at 1,500° F.

tures is dependent upon the magnitude of the vibratory stress relative to the mean tensile stress. This dependency is shown in figure 20. If vibratory stress is high relative to mean tensile stress, the fracture shows typical fatigue characteristics, a smooth transgranular area on the fracture surface, and no additional cracks on the airfoil surface near the point where the failure started. With no vibratory stress, the specimens, of course, have only stress-rupture failure characteristics. With small amounts of vibratory stress the fracture may have characteristics of stress-rupture only, or fatigue only, or a combination depending on the alloy composition and the temperature (ref. 2). In fact, the vibratory stress can be high enough to cause an appreciable decrease in life over the life predicted on the basis of stress-rupture (mean stress) alone, without introducing any indications of fatigue in the fracture. Therefore, even though only stress-rupture characteristics may be visible in the fracture, it cannot be said with certainty that fatigue did not reduce bucket life. In some cases, fractures exhibiting only stress-rupture characteristics occur outside the "critical zone" for stress-rupture fracture. Such fractures by virtue of their location alone suggest that fatigue may have contributed to the failure (ref. 3).

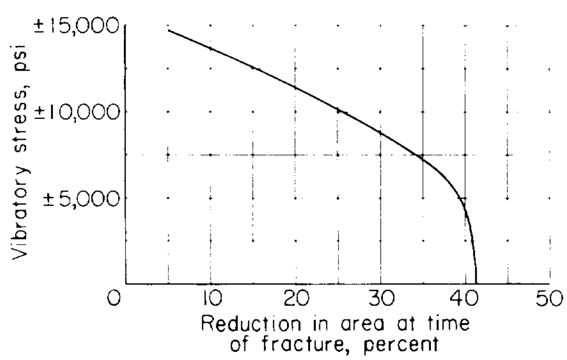


FIGURE 19.—Effect of vibratory stress on specimen elongation. Heat-treated wrought S-816; temperature, 1,500° F; mean stress, 25,000 psi.

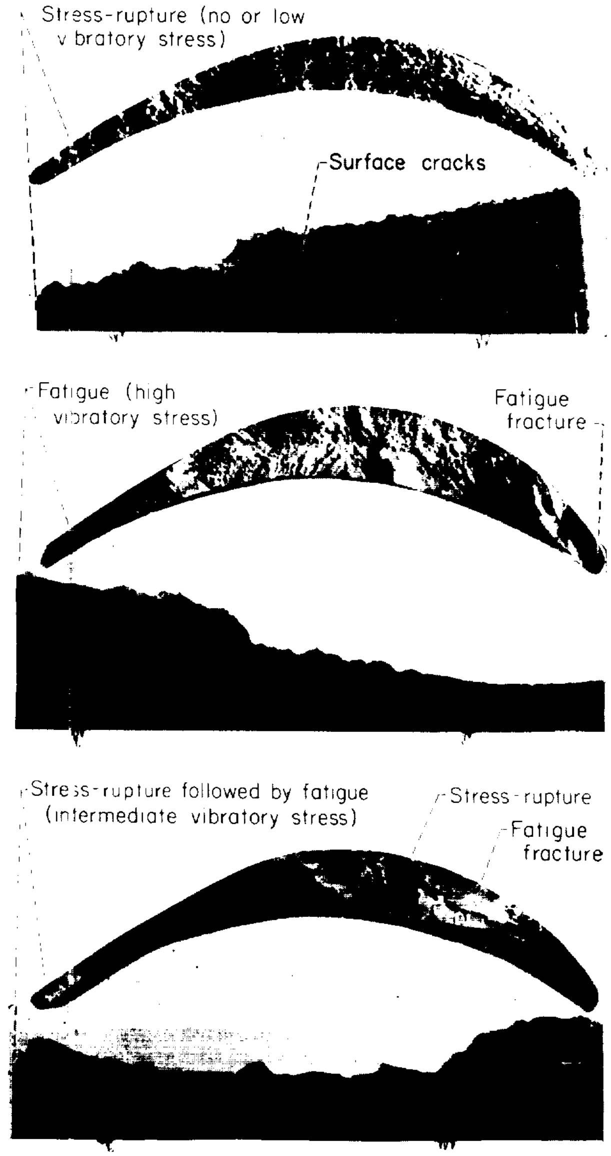


FIGURE 20.—Typical bucket failures.

The use of any crack as a replacement criterion to avoid complete bucket fracture in flight depends upon the rate of propagation of the crack to complete fracture. If one could be sure that a crack would not propagate to complete fracture before the next inspection, the bucket may be satisfactory. For a fatigue crack, the rate of propagation may depend largely upon its location on the bucket airfoil. Radial tip cracks (fig. 17(c)) have been found in at least one engine study to propagate very slowly; cracked buckets could be found and rejected during inspections. Fatigue cracks across the chord (figs. 17(a) and (b) and 20) have a centrifugal load superimposed upon them (increasing as the crack is located nearer the base) and may be expected to propagate more rapidly.

The combination of high vibratory stress and high frequency that may occur at resonance can fracture a bucket in a very few minutes. Thus, the avoidance of resonant conditions that result in high vibratory stresses is very important.

In designing an engine it is impossible to predict with certainty that high vibratory stresses will not be encountered. If high vibratory stresses are found, or if fatigue fractures are encountered, several changes can be tried to reduce the vibratory stresses. Some of these are as follows:

(1) The sources of excitation can be altered by methods such as

(a) Changing the number of nozzle vanes to change the excitation frequency or changing the spacing of the nozzle vanes to break up the regularity of the impulses

(b) Changing the number of combustors or using an annular combustor

(c) Changing the distance of the wheel from the nozzle vanes (ref. 4)

(2) The bucket vibration characteristics can be altered by methods such as

(a) Changing the bucket geometry

(b) Stiffening the bucket by thickening or by shrouding the tips

(c) Damping vibrations (by methods similar to those discussed in ch. IV)

(3) The ability of the buckets to withstand vibration may be improved by methods such as

(a) Changing surface-stress state by polishing or shot-peening

(b) Changing the material to one having better fatigue properties

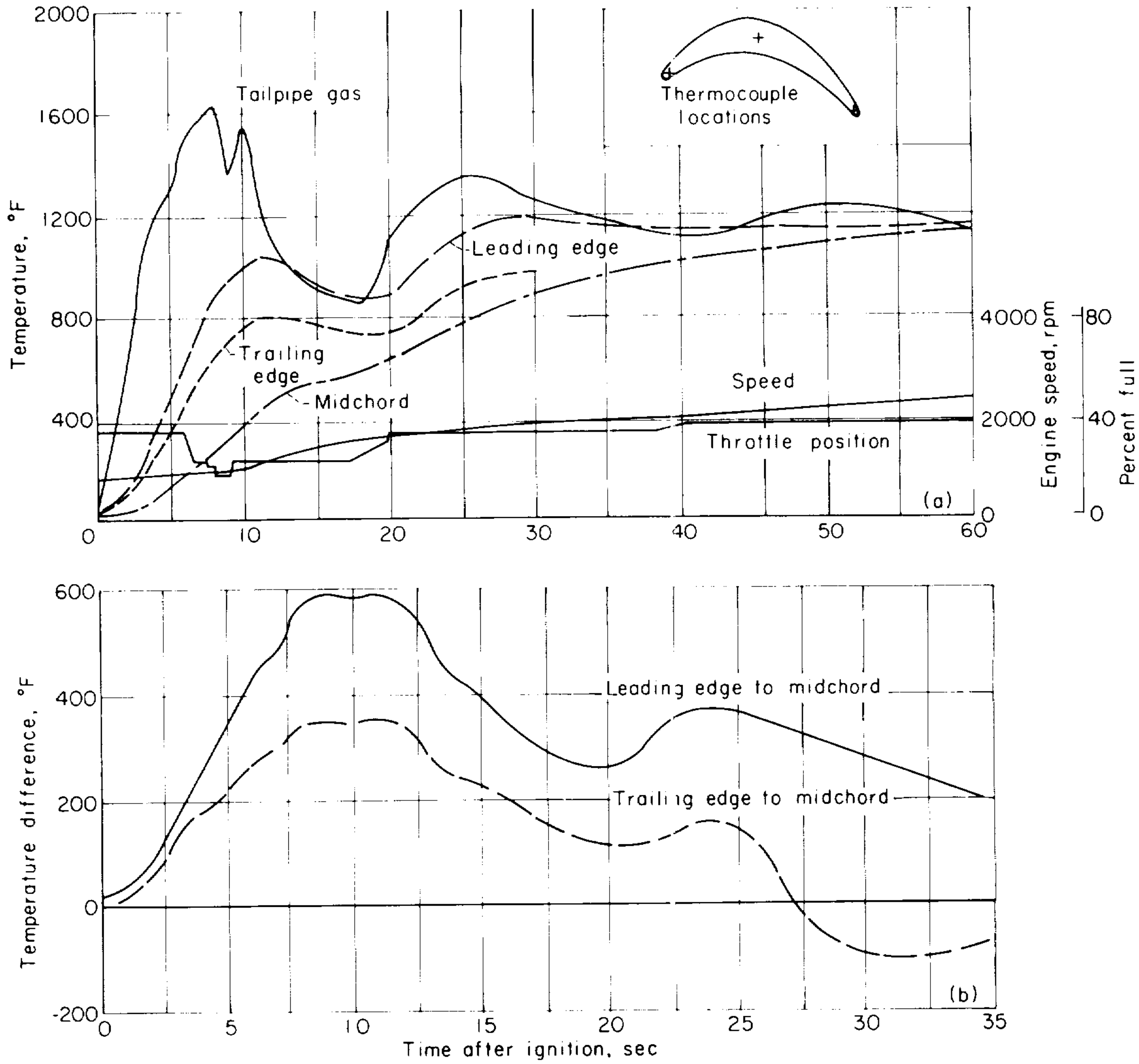
In some cases, it may be worthwhile to instrument the turbine buckets with strain gages and determine the engine speeds that cause resonance. These critical speeds can then be avoided in future engine operation, if they do not occur at important operating conditions such as at cruise or at rated power.

THERMAL STRESS FATIGUE

Since the leading and trailing edges of turbine buckets heat and cool much more rapidly than the thick midchord region, chordwise temperature differences occur in the buckets whenever the engine is operated through transients, such as during starting, stopping, accelerating, and decelerating. Some of these temperature differences have been measured with thermocouples and are shown for the turbine bucket of the engine B-9 in figures 21 to 23.

Figure 21 shows that for this engine during a normal start temperature differences between the leading edge and the midchord (a distance of about 1 in.) are of the order of 600° F. In a hot start, the temperature differences will be greater; in the case shown in figure 22 they were of the order of 800° F.

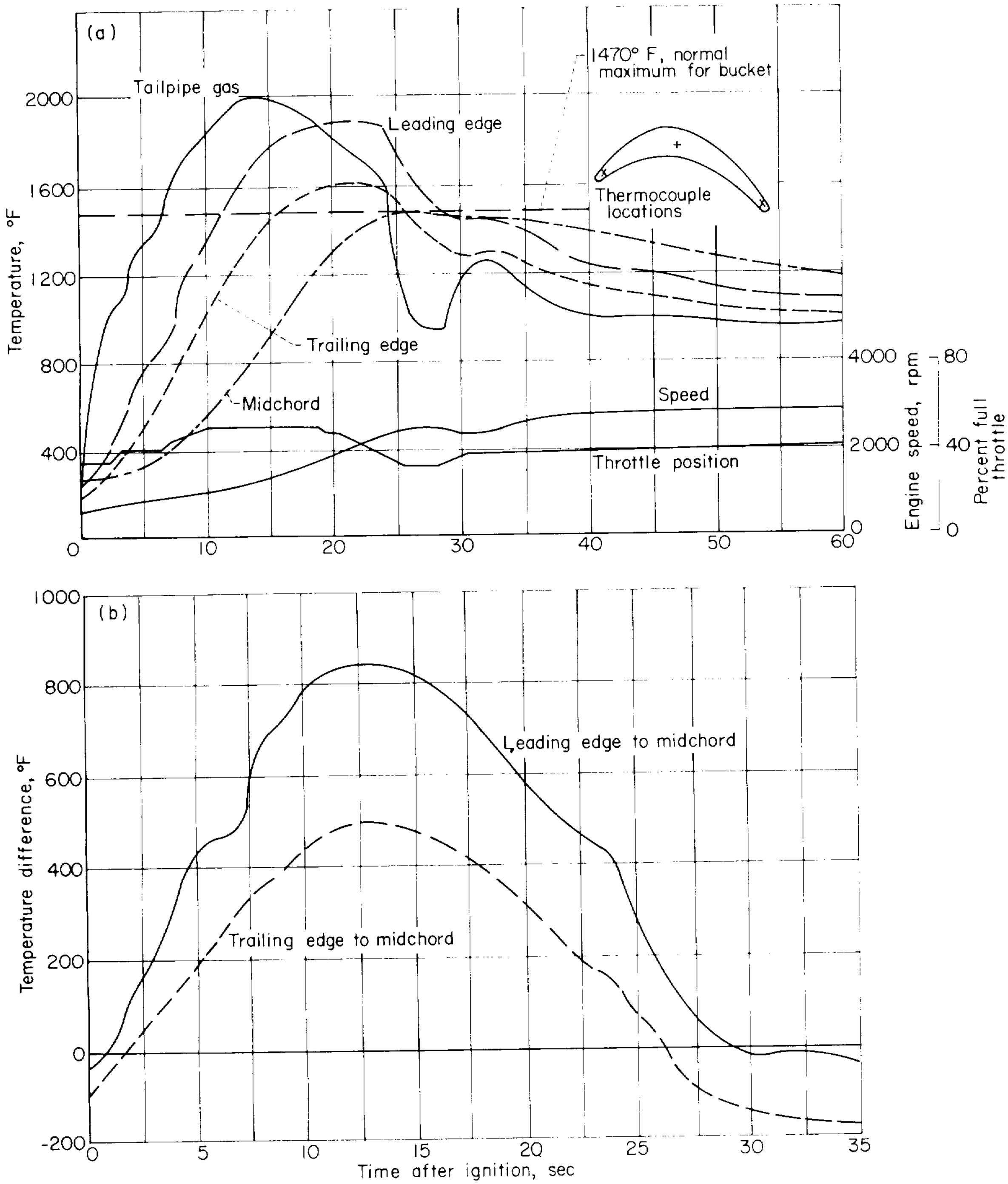
Bucket temperatures measured during a normal engine acceleration from idle to full power are shown for engine B-9 in figure 23; the temperature difference between leading edge and midchord was about 185° F. Temperature differences measured during a shutdown from full power to stop (fuel suddenly shut off when running at full power) have been observed to be of the order of 350° F. These temperature differences will vary among engine designs.



(a) Temperatures.

(b) Turbine-bucket temperature differences.

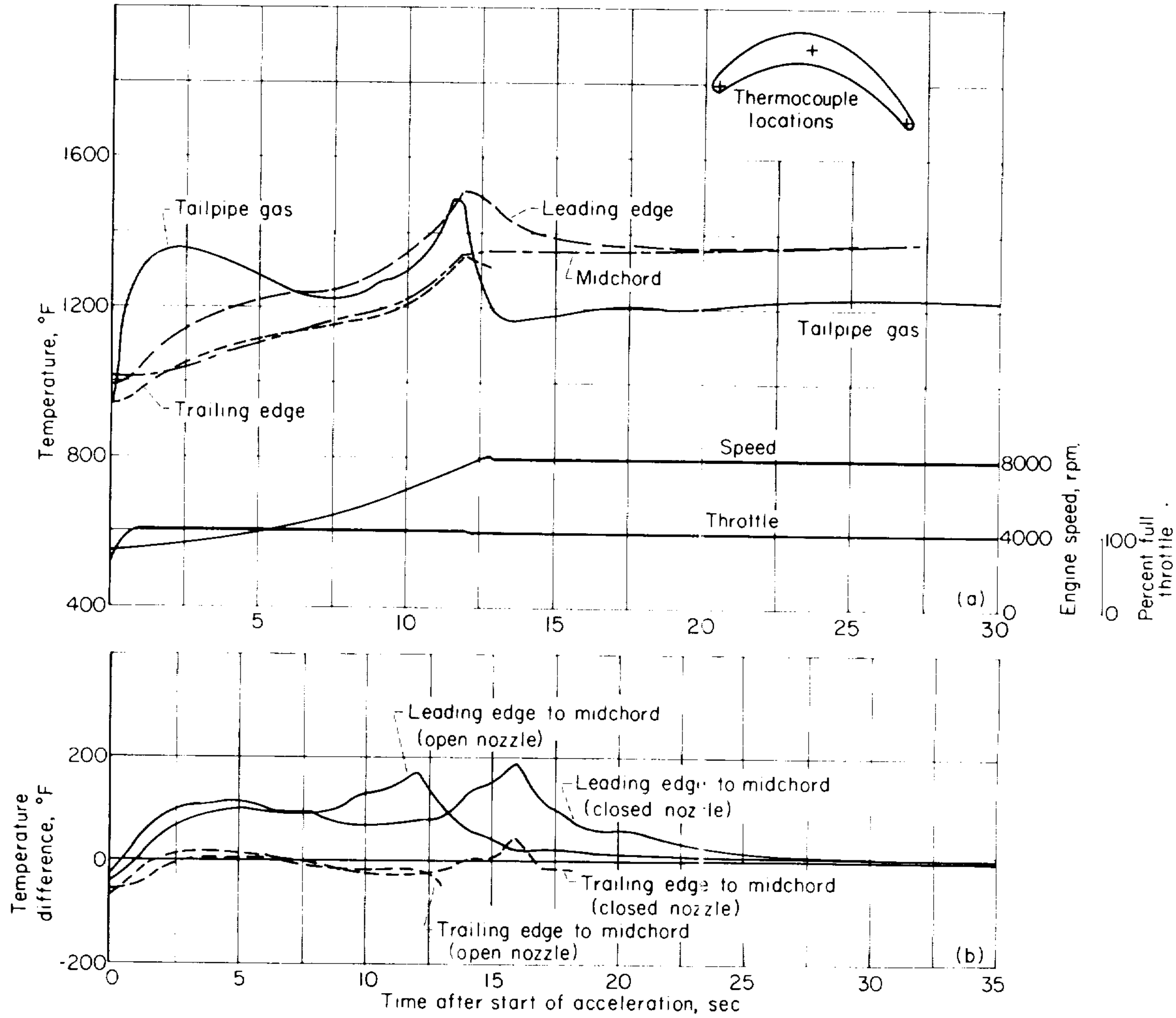
FIGURE 21.—Temperature conditions during "normal start." (Data from ref. 17.)



(a) Temperatures.

(b) Turbine-bucket temperature differences.

FIGURE 22.—Temperature conditions during "hot start." (Data from ref. 17.)



(a) Temperatures.

(b) Turbine-bucket temperature differences.

FIGURE 23.—Temperature conditions during acceleration from idle speed (3,000 rpm) to rated speed (7,950 rpm). (Data from ref. 17.)

Temperature differences induce stresses in the buckets because of differential thermal expansion. The thin leading or trailing edge heats and cools more rapidly than the midchord and is restrained from expansion or contraction by the thick midchord section, and thus high compression stresses (during heating) or tensile stresses (during cooling) are induced in the edges. Temperature gradients of the magnitude described above are not sufficient to cause fracture in 1 cycle, but with repeated cycles the bucket can fail by what is called thermal fatigue. Of first-order importance in failures by thermal fatigue is the number of cycles to which the blade has been exposed rather than time of operation. Time is undoubtedly of importance, in that time at high temperature will permit some relaxation between cycles and thus affect the initial stress condition during subsequent cycles. The buckets shown in figure 24 were removed from an engine of the type in which the transient data were obtained. The buckets had been operated through 451 cycles comprising 5 minutes at idle and 15 minutes at full power (a total of 112 hr at full power)

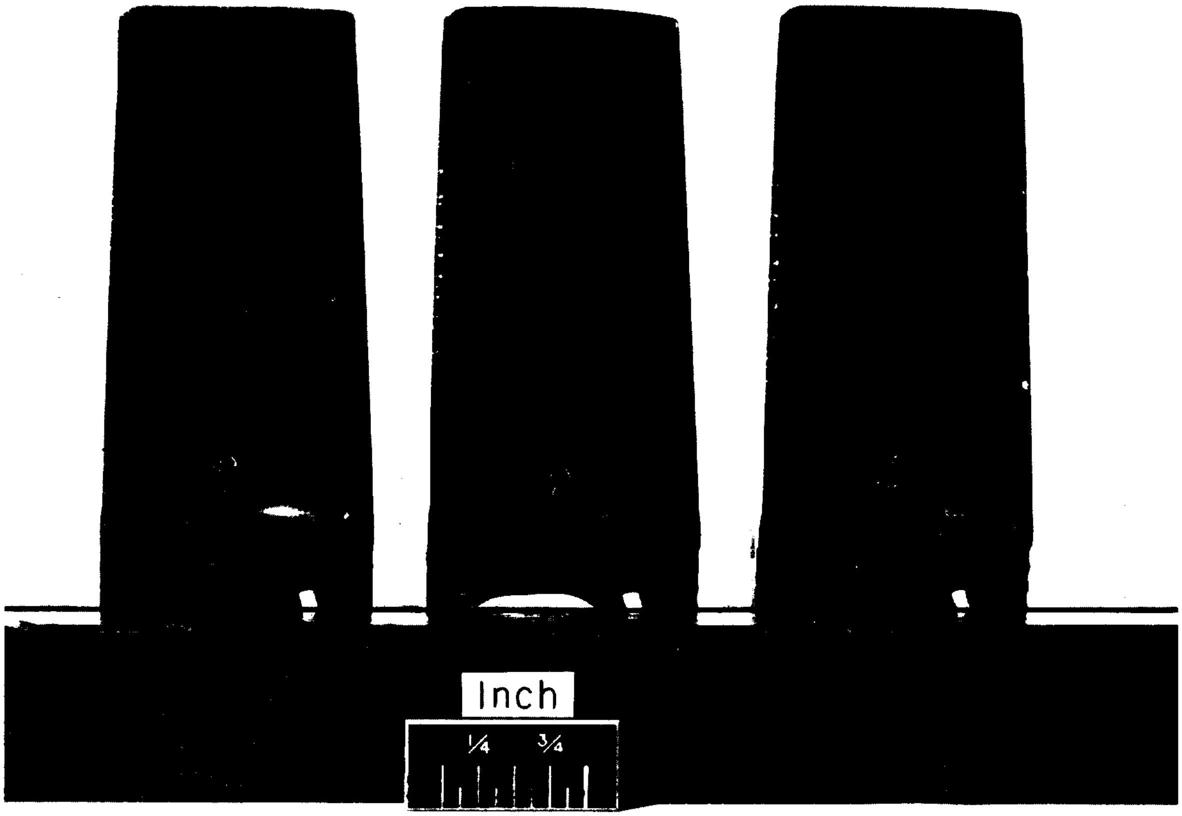


FIGURE 24.—Leading-edge cracking from repeated thermal gradients.

As shown in figure 24, thermal cycling damage manifests itself as warping or cracking along the edges of the bucket, predominately along the leading edge. Usually, many fine cracks are present instead of the single crack typical of mechanical fatigue. The cracks can (and usually do) extend outside the stress-rupture "critical zone" as defined by the centrifugal stress and temperature. In the buckets shown, there are cracks along two-thirds of the leading edge. Even if the buckets do not crack directly as a result of thermal stress, it is possible that the repeated working of the material by thermal stresses may lower stress-rupture or fatigue strength so that buckets fail early by these mechanisms (refs. 5 and 6).

Thermal stress cracking tends to relieve the thermal stress, and such cracks will not propagate to complete fracture without additional applied stress. The propagation to complete fracture depends on other stresses, and the rate of propagation will vary with engine design. The rate depends largely upon the stress-rupture safety factor designed into the bucket and upon the notch sensitivity of the alloy. In the case of the low-stress engine of figure 3, a sample of buckets cracked in the manner shown in figure 24 ran 300 hours without complete fracture. The cracks could be detected in regular engine inspections if the inspections were made carefully. If such cracks were to occur in the buckets of a high-stress engine, they might be expected to nucleate fatigue or to act as a notch; for some materials, notching greatly decreases stress-rupture life. In such an engine, thermal fatigue cracks may very quickly propagate to cause complete fracture.

Unfortunately, the material factors that influence the susceptibility of a material to cracking due to thermal cycling are not understood. A study of this problem may lead to clues to reduction in the tendency for thermal cracking through control of the metallurgical and the processing factors. In the absence of this under-

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standing there is still the possibility of reducing the tendency for thermal cracking by decreasing the temperature gradient through control of the rate of temperature change during transient operation. In the engine studied, the control of gas temperature during starting (from lightoff to idle) was manual, and the rate of gas temperature change was, therefore, a function of the skill of the operator. During acceleration, the fuel flow was scheduled against compressor-outlet pressure. Gas temperature was not sensed by the engine control. It was found that the thermal fatigue cracking of the buckets of this engine was associated with the gradients induced during starting of the engine and that the cracking could be greatly delayed by starting the engine less rapidly (refs. 7 and 8).

Another approach to reducing the stress in the bucket is to change the bucket factors that cause the gradients and the resultant stress. For a particular rate of change of gas temperature, the gradients set up in the bucket are a function of heat transfer to the bucket, bucket shape, and thermal conductivity, and the conversion of this gradient to stress is a function of bucket shape, modulus of elasticity, and coefficient of expansion of the material. An example of a change of this type that may be investigated would be surface coating on the bucket that would reduce the heat transfer during transients.

CORROSION

Corrosion of the high-temperature alloys used for turbine buckets has not been a problem of concern except where leaded fuels have been used. In this case, if the surface temperature of the bucket is above the melting point of the deposited lead oxides, the molten film of lead oxide tends to react with the bucket surface and cause a rapid "washing away" or erosion of the surface. Solutions to this problem have not been found. Leaded fuels (gasoline) have been used for logistic reasons, but since they are no asset to the turbojet engine, their use should be avoided if possible.

EFFECTS OF OVERSPEED, OVERTEMPERATURE, AND IMPACT

OVERSPEED AND OVERTEMPERATURE

A review of Unsatisfactory Reports and DIR's indicates that engine overspeeds of 4 to 6 percent are occasionally reported. According to present technical orders, if an overspeed of the order of 2 to 4 percent occurs, depending on engine model, the engine must be removed for overhaul because serious damage may have occurred to the rotating parts.

An overspeed occurs most frequently because the control has allowed excess fuel to be injected; thus overspeed is usually accompanied by overtemperature and can very drastically reduce bucket stress-rupture life. It was calculated for one engine that if fuel flow were increased to give a 5-percent overspeed (10-percent increase in stress) the temperature would increase by about 200° R. Continuous operation at this point would reduce the bucket stress-rupture life from 30,000 to about 10 hours. With present knowledge it is not possible to state with certainty the effect of momentary overspeed and overtemperature on the stress-rupture life of buckets of this engine. The same increase in speed and temperature would be more serious in an engine which is designed with a lower stress-rupture life.

Typical occasions when overtemperature might occur without overspeed are as follows:

(1) During a hot start; in most current engines control of the engine from start to idle is dependent on pilot care

(2) During acceleration; the compressor may stall, or fuel flow may be increased too rapidly

(3) At full engine speed

(a) During afterburner ignition, since turbine back pressure can reduce engine speed and cause the fuel regulator to increase fuel flow before the tailpipe can open

(b) During drift or malfunction of thermocouples

(c) At very high altitudes, primary because of a reduction in compressor efficiency (see ch. X)

Overtemperature may thus occur when the buckets are at almost any level of centrifugal stress between the negligible stress of "lightoff" speed and maximum stress at full engine speed. Although some conditions of overtemperature are of long duration (e.g., control thermocouple drift), most are of short duration, usually only seconds. Table III summarizes some of these conditions to give a qualitative indication of the stresses, temperatures, and duration. Service data indicate that overtemperature most commonly oc-

TABLE III. OPERATING CONDITIONS DURING WHICH OVERTEMPERATURES MAY BE ENCOUNTERED

Condition	Temperature difference above normal maximum	Probable stress	Duration
Hot start	Can be very great	Very low	Short (few sec)
Acceleration, compressor stall	Can be very great	Low	Short (few sec)
Fuel flow increased too rapidly	Moderate to great	Low to high	Short (few sec)
Afterburner ignition	Can be great	High	Short (few sec)
Drift or malfunction of thermocouples	Moderate	High	Can be long
Very high altitudes (primarily reduced compressor efficiency)	Moderate to great	High	Moderate
Overspeed	Moderate to great	High	Short

curs during hot starts, when the stresses are low, and duration is short. Because most overtemperature is of short duration and because the bucket edges follow changes in gas temperature more closely than the middle of the bucket does, the edges will suffer most damage. As indicated in figure 22, it is probable that because of the lag of the midchord section only the edges will be overheated in transient overtemperatures that occur when the bucket is initially at low temperature.

The effects of overtemperature on bucket materials are

(1) In combination with high stress, to use up a large part of the life

(2) To damage the metallurgical structure (regardless of stress level)

(3) To induce thermal stresses if the overtemperature is a rapid transient

The magnitude of the damage is, of course, a function of the stress level during the overtemperature, the magnitude and duration of the overtemperature, and the bucket alloy used. With the present knowledge and data on this subject, however, it is not certain what damage to the buckets has resulted from a particular overtemperature condition, nor is it possible by examination of a bucket to state that neither overtemperature nor damage has occurred (ref. 9).

The sensitivity of stress-rupture life to temperature described earlier is indicative of the damage that may result when overtemperature occurs while the bucket is at high stress levels.

Overtemperature that occurs even when stresses are low, as during a hot start or stall while accelerating, may cause damage that will shorten bucket life. Many of the alloys used for turbine buckets achieve their properties by a heat treatment consisting of a solution treatment at temperatures of the order of 1950° to 2150° F followed by an aging at temperatures of 1350° to 1600° F. The aging forms a precipitate in the microstructure that hardens and strengthens the alloy. Heating above the aging temperature causes the precipitate to be agglomerated (overaged) or redissolved, and, in many cases, the alloy to be weakened. The amount of overaging or re-solution increases with increases in temperature and time and alloys differ in their susceptibility to damage by overaging or re-solution. Studies are underway in several laboratories to gain further insight into this problem.

RECOGNITION OF OVERTEMPERATURE DAMAGE

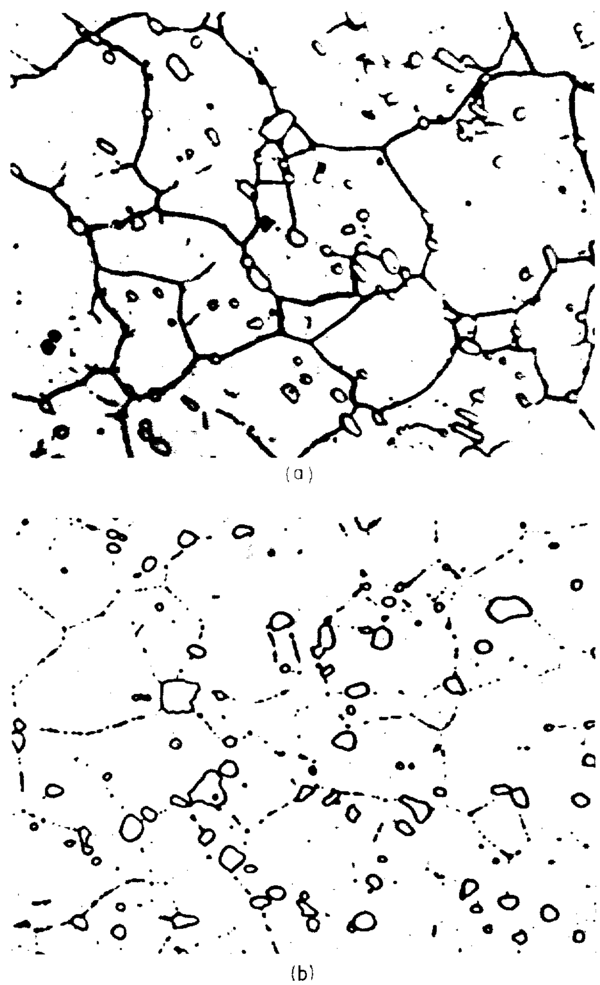
Overtemperature can be so severe as to melt or burn the buckets, or to cause warping, cracking, obvious stretch, or fracture. For some alloys, the occurrence of overtemperature causes changes in the microstructure. This is shown for overtemperatured S-816 in figure 25(b), which shows that grain boundary precipitates have been partially redissolved, compared with an undamaged structure in figure 25(a). For some alloys, hardness is changed markedly by re-solution, and such a change may indicate damage.

It is possible, however, that damage may occur and not be revealed by any known test except cutting test specimens from sample buckets and running laboratory tests such as stress-rupture (refs. 9 to 11). Because of this difficulty in determining whether overtemperature has occurred, it is desirable that all engines be equipped with instruments to record engine speed and temperature as a function of time.

Severity of the overtemperature problem will be reduced if buckets are designed for very long stress-rupture life and may be reduced by changes in alloys, but controls should be developed that will avoid the situations where dangerous overtemperatures occur.

IMPACT

Failure statistics have revealed that buckets are frequently damaged by impact. The sources are foreign objects, parts of other engine components (such as the combustion liner and nozzle vanes) that have failed, or parts of buckets that have failed. Alloys currently used may suffer impact severe enough to cause fracture (figs. 1 and 2). More commonly the result is dents, gouges, or nicks on the airfoil that act as notches in the surface causing stress concentrations that accelerate fatigue or stress-rupture failure. Depth and sharpness are important notch variables. As was discussed for thermal-stress cracks, the rate of propagation to complete failure (from a notch) depends upon the safety factor in the bucket, the magnitude of vibratory stresses, and notch-rupture properties of the bucket alloy.



(a) Standard heat treatment: 1 hour at 2,150° F, water-quenching; 16 hours at 1,400° F, air-cooling.

(b) Severely overtemperatured (15 min at 2,000° F) following standard heat treatment.

FIGURE 25.- Effect of overtemperature on microstructure of S-816. X1000.

The subject of notch-rupture, mentioned in chapter VIII, is described in some detail in reference 12. The effects of notches on fatigue are well known. The high probability of impact damage to rotating buckets makes notch-rupture and notch-fatigue properties important in choosing bucket alloys.

In general, notches will not cause immediate failure. Buckets must be inspected regularly for damage, and notched buckets must be reworked (to reduce notch sharpness) or replaced. Inspection limits that establish standards for bucket rework and replacement must be set.

VARIABILITY OF BUCKET LIFE

Turbine buckets of a single wheel will differ considerably in time to failure. This is demonstrated in figure 26. A total of 46 turbine buckets of the same material for a given engine, chosen at random from Air Force stock, were installed in a turbine wheel. The engine with this wheel was operated continuously at full engine power until a bucket failure occurred. Operation was continued, with each failed bucket being replaced, until all original buckets had failed. Figure 26 is a plot of number of original buckets failed against time at full engine power. The first failure occurred after 180 hours and the last after 540 hours, a spread ratio of 3:1.

The data of figure 26 also indicate that the failure distribution of these turbine buckets is of the "wear-out" type discussed in chapter II. Thus, after the first bucket had failed, additional failures followed shortly with the rate of failure increasing with time. For buckets of this kind, after the first bucket failure has occurred, additional bucket failures can be expected at very

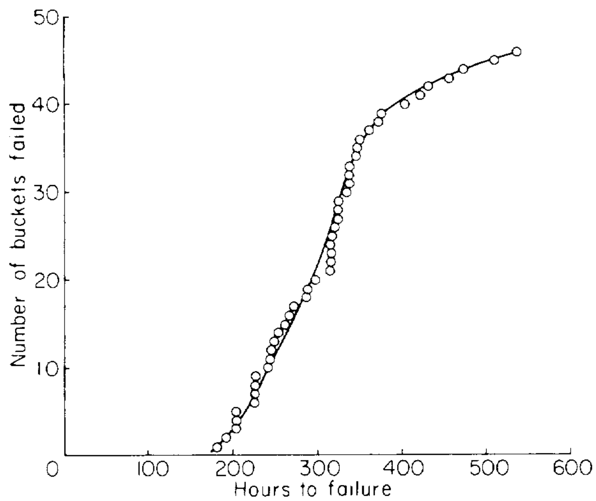


FIGURE 26.—Typical distribution of turbine-bucket failures. Forty-six buckets in one wheel. (Data from ref. 18.)

short time intervals and the wheel should be equipped with all new buckets.

SOME FACTORS AFFECTING RELIABILITY

Consideration of several factors in design, production, operation, field inspection, and overhaul may help to avoid bucket failures, or at least to reduce their frequency.

DESIGN

It is important that turbine buckets be designed for very long stress-rupture life, for at least two reasons. One is that impending stress-rupture failure cannot be detected by inspection methods (with the exception that elongation may aid in some rare cases); thus assurance that stress-rupture will not occur must rest on the design. The second is that long stress-rupture life will reduce susceptibility to fracture from cracks introduced by other causes such as impact, thermal fatigue, and mechanical fatigue and susceptibility to damage by overspeed and overtemperature.

Long stress-rupture life can be achieved by lowering temperature or stress, of course, or by using a stronger alloy. Cooling of the bucket would be beneficial if used to gain a large safety factor. For example, if an engine were designed with a centrifugal stress in a hollow S-816 bucket of 20,000 psi and a temperature of 1500° F, the time to rupture should be about 410 hours (fig. 4). If this bucket were cooled only 200° F, a tremendous safety factor in stress-rupture would be realized. If, however, the gas temperature and bucket stresses were such that the bucket must be cooled to achieve satisfactory life, mechanical failures of the cooling system may lower reliability.

Every effort should be made to reduce vibratory stresses to safe levels. Since the designer cannot be sure that serious vibrations will not be encountered, it is important that the new designs be given long endurance runs (several hundred hours at expected service operating conditions) in test stands to see if failure occurs and to indicate typical failure mechanisms. Endurance tests of buckets on an accelerated cycle (5 min at idle and 15 min at full thrust) in NACA sea-level test stands run on four models of two engines duplicated the major mechanisms of failure found in engines in service. When a new engine is introduced into service, it should be inspected at short intervals for indications of failure. If failures are encountered, immediate steps such as those indicated in the section on fatigue should be taken to reduce the vibratory stresses or to improve the ability of the buckets to withstand the stresses. Usually, fatigue cracks will propagate slowly if the bucket has been designed for long stress-rupture life.

In the case of thermal-fatigue cracking, research is required to provide design criteria. If stress-rupture life is very long and vibratory stresses are low, however, thermal-stress cracks will propagate slowly, if at all, and may be found in scheduled inspection.

Controls to avoid overtemperature and too rapid temperature changes should be investigated and installed.

If bucket failure occurs, it is less dangerous in a single-stage turbine than in a multistage turbine. In general, however, for the same application it is usually possible to achieve lower stresses in a multistage turbine, and the probability for a bucket failure may be smaller.

An examination of accident records indicated that serious damage occurs when a failed bucket fragment becomes jammed between the shroud band and the tips of the other buckets (as was the case for the test engine shown in fig. 1). Provisions to permit the fragments of failed buckets to be thrown clear radially so they will not become jammed in this manner or cause damage downstream in multistage turbines should be studied and installed if possible. In NACA tests of bucket materials, it has been found necessary to reduce the turbine shroud thickness to a point where bucket fragments will easily pass through so the remainder of the test buckets on a wheel are not severely damaged. Shrouded buckets may be an advantage in this respect because they do not need a heavy shroud band to maintain tip clearance. In an airplane, failed buckets must be captured in an area beyond the tips to protect the airplane from flying fragments.

MANUFACTURE

Many of the steps used in the manufacture of turbine buckets can cause variability and influence properties of the material. For example, chemical composition and forging and heat-treating temperatures influence stress-rupture properties. Procedures used to finish leading and trailing edges can affect the amount of residual stresses in them and may possibly change bucket resistance to cracking by thermal fatigue. It is apparent that before a new manufacturing process is adopted, a number of test buckets should be fabricated and carefully evaluated to ensure that the process will produce buckets with the required properties. After a process appears satisfactory, close control of the process and quality is necessary. Even minor changes in the manufacturing process should be carefully evaluated to ensure that they do not impair quality. Measurements for the control of quality must be tailored to the method of bucket fabrication, that is, whether the buckets are forged or cast. In either case, control begins with the raw materials used in producing the alloy composition and ends with measurements on the finished bucket.

Discussion of the problem of inspection of materials and finished buckets is beyond the scope of this paper. Based on a knowledge of how buckets fail, however, some suggestions are offered as to the kinds of strength tests that might be included in inspection.

Because properties of buckets may vary with time of production (i.e., from day to day) as well as with alloy heat from which they are made, finished buckets logically should be sampled on both bases to ensure quality. It would be desirable to run on the sampled buckets all tests, such as stress-rupture, mechanical and thermal fatigue, and notch rupture, known to be important to bucket life for the particular engine. Such extensive tests may not be practical, however, because they are too difficult and time consuming. A minimum requirement might be a stress rupture test of a specimen taken from the airfoil. (A sample from the root cannot be expected to represent critical airfoil properties.) The test temperature should be close to the service environment. Usually relatively short-time tests are desired to cut testing time, but they should be used only after it has been proved that passing the short-time test will ensure needed long-time life. One test cannot be expected to measure properties all across the airfoil and additional checks must be conducted. For example, the remains of the bucket after removing the test specimen can be checked across the chord and along the span of the airfoil for grain size, microstructure, internal defects, and hardness.

If an engine is encountering a particular type of failure, such as leading-edge thermal fatigue, additional tests of this property should be introduced to ensure maintaining the best possible quality in this critical area. Completely satisfactory tests of resistance to thermal fatigue short of operation of the bucket at conditions exactly duplicating engine conditions are not now known and considerable study will be required to set up such a test.

The evaluation of creep-rupture properties from a specimen cut from cast-bucket airfoils is usually of questionable value because of the inherently large grain size of castings. If a test specimen has 15 to 20 grains per cross section, the specimen may be representative of the bucket-material properties, but the small specimen that might be cut from a cast bucket airfoil frequently may have only one or two grains in its cross section. A standard practice is to cast a test bar in each mold with the buckets and to test the properties of this specimen as representative of bucket properties. Occasionally this bar is placed in a position in the mold (usually the center) where it does not cool at the same rate as the buckets. Thus, the grain size and structure can differ from those of the buckets. Changes in the casting process may affect the bucket properties without affecting those of the test specimen. In some cases, a thin wire of the casting material is cast in a mold along with the buckets. This wire must bend through a specified angle to indicate acceptable ductility of the buckets, but it may not be representative of the buckets. For example, the mold design can be such that directional cooling of the wire causes it to be a single crystal with high ductility that does not represent bucket properties. Careful laboratory studies are necessary to be sure that test bars made by a particular casting procedure and mold design are representative of the buckets produced before the bar is used in quality control. If the mold design, casting process, or alloy composition is changed, this study must be repeated.

Variations will occur on a time basis in any production process. A minimum of scatter might be expected from parts made in succession. To minimize scatter of bucket lives on any one wheel, it would be logical to assemble buckets onto any one wheel in the chronological order of their production; at the least, it would be preferable not to mix heats of alloy on wheels.

OPERATION

Bucket life has been shown to be related to various factors in engine operation and therefore may be improved by a control of some of these factors. Some suggestions follow:

(1) Increased efforts should be made to ensure avoidance of hot starts through pilot training, development of improved starting methods, or development and installation of automatic controls.

(2) Unnecessarily rapid accelerations and decelerations should be avoided.

(3) Full power should be used only when necessary.

(4) Causes of overtemperature should be isolated and these operating conditions should be avoided or controls to avoid their occurrence should be developed and installed.

Instrumentation should be provided to make continuous records of engine speed and gas temperature against time. Important operation variables would then be known. These records would show the amount of time at full power, the amount of time and the temperature at overstress and overtemperature. They would also show the number and the severity of acceleration and deceleration cycles, starts, and stops, and could be used as a basis for scheduling replacements and special inspections. (All these data plus a history of bucket failures and replacements would be very valuable as an improved basis for design criteria.)

Until such an instrument is provided, the pilot should keep records of time and temperature at maximum power, the number and severity of overtemperatures and overstresses, and number of flights (assuming a correlation between a number of accelerations and decelerations and number of flights).

Adequate instrumentation should be provided to enable the pilot to monitor the temperature and engine speed set by the automatic controls at maximum power setting. This provision will permit him to make corrections if the automatic control drifts or otherwise permits operation at too high speed or temperature.

FIELD INSPECTION AND OVERHAUL

Buckets should be inspected at regular intervals for cracks, warpage, melting or severe scaling, excessive elongation, nicks, and dents.

Although at some later date it may become evident that some cracks (e.g., those resulting from thermal fatigue in a particular engine) will never propagate to failure in the normal life of the engine, present practice should be to immediately replace buckets having any of these indications except nicks and dents. In some cases, nicked and dented buckets may be reworked to increase radius in lieu of replacement. Simple visual inspection for cracks is inadequate, and aids such as fluorescent penetrant fluids and powders should be used. The frequency of inspection will depend upon the severity of operating conditions and the past records of failures with the buckets.

In some cases, cracks propagate so rapidly to complete fracture that replacement of cracked buckets during regular inspections will not forestall bucket fracture in flight. One such case is bucket fracture by stress rupture. The current practice in regard to stress-rupture failures is to replace each bucket as it fails. This practice should be refined by replacing all buckets on the wheel when the first fails because, as indicated by figure 26, when the first of a sample of buckets fails by stress-rupture, failures of the remainder will follow shortly. It is not desirable to allow even one bucket to fracture by stress-rupture, however, because this results in a high risk of flight accident, particularly since a piece of the bucket of the order of one-half the airfoil has been released. Although this procedure has been used in single-stage turbines, it is not permissible for multistage turbines because of the greatly increased likelihood of a serious accident.

If the buckets show large elongation before fracture, periodic measurement of elongation may be helpful as a replacement criterion.

In addition, a system for scheduled replacements based on the time at various operating conditions and the stress-rupture properties of the buckets should be worked out.

The additive effects of stress-rupture conditions are not clearly understood. For example, if 50 percent of the life expected at one stress and temperature has elapsed, the percent of life remaining at some other stress and temperature is not known. On the basis of stress-rupture life, however, the time at maximum power is so much more important than a corresponding time at cruise that the time at cruise is not of serious concern. A slight adjustment in time at maximum power would probably compensate adequately. For example, the buckets of one engine (C, fig. 5) should have a life of 900 hours at full engine power. If the engine were continuously operated at cruise the buckets should have, because of lowered stress and temperature, a stress-rupture life of 46,000 hours or 50 times as great. An approximate formula for replacement time might therefore be worked out using time at full power less the utilized fraction of life used at cruise (if desired) plus a safety factor to take into account overtemperature, overstress, and variability of bucket properties. To apply this formula for replacement of buckets from considerations of stress-rupture, it would be desirable to have a device to record engine speed and temperature against time as previously described. Another practice that may aid is removal of a bucket from the engine, machining a specimen from the airfoil, and stress-rupture testing it to indicate magnitude of life remaining. The remainder of the bucket can be checked for indications of surface damage (severe intergranular corrosion, etc.).

In cases of mechanical fatigue where the cracks propagate slowly (e.g., radial tip cracks in some cases), the buckets may be replaced using the presence of cracks as a criterion. If, as is more commonly the case, they propagate rapidly or inconsistently, it is imperative that the vibratory stresses be reduced; no sound basis for working out replacement schedules exists. Flight failure cannot be avoided by inspection alone.

In the case of thermal-fatigue cracks, if the engine is designed for long stress-rupture life and if vibratory stresses are low, the cracks will propagate slowly and replacement on the basis of cracks alone will be satisfactory. If they propagate very rapidly, it will be because of superimposed centrifugal or mechanical-fatigue load. In this case, as soon as one cracked bucket is found, all buckets on the wheel should be replaced to reduce likelihood of accident. Efforts should be directed toward reducing the centrifugal or vibratory stresses in order to slow the propagation rate, or to eliminate the thermal-fatigue cracking since, again, a sound basis for scheduling replacements is not available.

In overhaul, the practice of pooling or mixing buckets from several engines should be avoided, since this mixes buckets having long life remaining with those having short life remaining. Because the first bucket failure on each wheel may cause all buckets to be replaced or damage the airplane, mixing buckets tends to make all engines as bad as the worst.

BENEFITS OF ENGINE DERATING

The extreme sensitivity of bucket stress-rupture life to stress and temperature (figs. 4 and 6) suggests that if the bucket stress-rupture life is too short, it may be increased by lowering stress or temperature or both, that is, by derating the engine.

It is important to determine what sacrifices in engine thrust might result from derating, and how specific fuel consumption might be affected. Calculations made for a specific engine are shown in figure 27. Tailpipe area was varied to optimize each condition. Curves shown for three

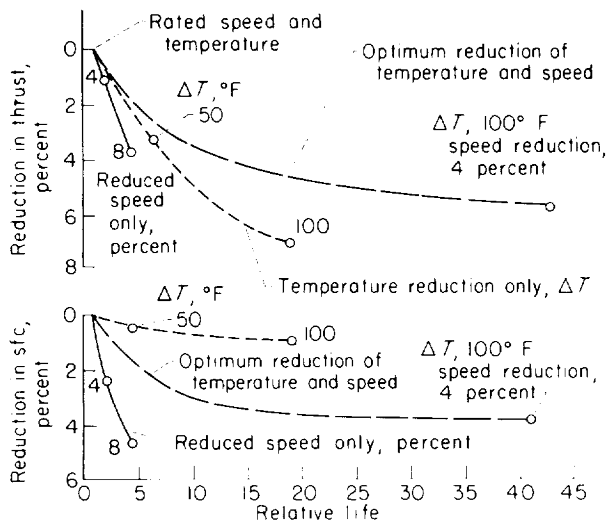


FIGURE 27.—Effect of engine derating on thrust, specific fuel consumption, and bucket stress-rupture life.

methods of derating in the upper part of this figure indicate that, if the engine speed is reduced with temperature held constant, an 8-percent reduction in speed reduces thrust about 4 percent, and quadruples stress-rupture life. If temperature is decreased with speed held constant, a 100° F drop in temperature increases life about 19 times, but thrust is reduced by 7 percent.

The curves of this series show that it is preferable to reduce both temperature and speed if this engine is to be derated. If temperature is reduced 100° F and speed 4 percent, bucket stress-rupture life would increase 44 times, while thrust would be reduced 6 percent.

The data of the lower part of the figure show the reduction in specific fuel consumption for the derating conditions just discussed. Since a reduction in specific fuel consumption is desirable, all derating conditions investigated were beneficial. The condition where temperature was reduced 100° F and speed 4 percent would improve specific fuel consumption by 4 percent.

The curves show that, in general, derating of this engine could achieve greatly increased bucket stress-rupture life with some improvement in fuel consumption and with moderate reductions in thrust. The same effect is gained by limiting operating time at full power; that is, running more of the time at cruise condition.

A word of caution is needed at this point. This section has shown that bucket stress-rupture life might be improved by derating. If buckets are failing by stress-rupture, these data illustrate gains that might be made. If stress-rupture life is improved, the buckets may then fail by another mechanism, and all the expected gains may not be achieved. Also, buckets of some engines do not fail by stress-rupture, and for these engines, an increase in life cannot be predicated from stress-rupture data as has been done in this example. Each engine must be studied separately.

ADDITIONAL INFORMATION NEEDED FOR IMPROVEMENT OF OPERATIONAL RELIABILITY

Turbine-bucket reliability will, of course, be benefitted by the current efforts that are devoted towards improving materials, methods of fabrication, etc.

Listed in the following paragraphs is information that will aid directly in improving turbine-bucket reliability by providing specific data or an improved insight or understanding of practical problems revealed by service experience with the turbojet engine:

(1) Additional information is needed on the effect of various amounts and durations of overtemperature on the life of the material when the bucket is subjected to various levels and types of loading. The influence of bucket-life period at which overtemperature occurs might be studied to determine, for instance, the relative importance of overtemperatures that occur during the first or second stages of creep. Early studies should duplicate conditions encountered in service insofar as is possible.

The possibility that heat treatments may restore material-property deficiencies caused by overtemperature should be studied.

(2) Information is needed to determine whether periods at various levels of stress and temperature in stress-rupture are additive; for example, if 60 percent of the bucket life is used up at maximum power conditions of stress and temperature, does 40 percent remain to be used at some other condition, say cruise? Development of methods which will permit the calculation of the percent of life used at each operating condition will lead to a rational basis for scheduling of bucket replacement to avoid stress-rupture failures.

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(3) Studies are needed that will alleviate bucket cracking caused by repeated thermal stresses. Reducing the rate of change in engine gas temperature is an obvious step, but it is desirable that engines be able to accelerate very rapidly. More knowledge is needed on the effect of bucket material and bucket shape and an understanding of the relative influence of engine cycle variables is needed.

(4) Studies are needed to isolate causes of variability in bucket properties, for example, whether bucket fabrication methods are the prime cause or whether the variation stems from the bar stock or from metal heats.

(5) As a guide in scheduling replacements, studies are needed on the rate of propagation of various types of cracks to complete fracture in various stress and temperature environments.

(6) Nondestructive methods are needed that will indicate (a) impending bucket failure, particularly in stress-rupture and fatigue, and (b) occurrence of damage from overtemperatures. Properties such as electrical resistivity and internal damping might be studied.

CONCLUDING REMARKS

Air Force service records indicate that with several jet-engine models, frequent turbine bucket replacements are made because of bucket cracking or fracture. Buckets occasionally fracture during flight. In engines with single-stage turbines, the fragments from fractured buckets are often ejected through the discharge nozzle with no further damage and with little loss in thrust. There is a risk associated with permitting bucket fracture in flight, however; in 1953 out of 205 Air Force flight accidents attributed to jet-engine malfunction 16 were traced to turbine-bucket failure. In engines with multistage turbines, the probability of catastrophe from failure of a turbine bucket is much greater. A failed bucket in an early stage may wipe out the buckets in successive stages and stop or destroy the engine.

Turbine buckets are subject to a combination of centrifugal stress, vibratory stress, high and rapidly changing temperature, and a corrosive atmosphere. Turbine buckets can fracture by stress-rupture or by fatigue, or a combination. Fracture can be accelerated by damage from overtemperature or overstress, damage or cracks resulting from thermal fatigue or perhaps from corrosion, or by nicks caused by solid objects in the gas stream.

In general, buckets should be designed for very long stress-rupture life, since no reliable warning for stress-rupture fracture exists. Long stress-rupture life also provides a safety factor that may reduce the rate of propogation of cracks from other causes. If bucket stress-rupture life is too short, it can be increased by reducing bucket stress or temperature or both, that is, by engine derating.

When excessive vibrational stress exists, it should be reduced to a safe value by design changes such as changes in nozzle geometry and bucket stiffening (including bucket shrouding) and by bucket vibration damping.

With a very long stress-rupture life and low vibratory stress, buckets can be replaced when dangerous cracks or nicks are found in a scheduled inspection or when the bucket has run a designated replacement time based on stress-rupture life with adequate safety factors (reductions) for variability in bucket properties and operational history.

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CHAPTER VIII

TURBINE DISKS

By G. C. Deutsch and R. H. Kemp

SUMMARY

Disk failure in an airplane turbine can cause immediate destruction of the airplane; of the major airplane accidents caused by the engine in 1953, 8 percent were due to failure of turbine disks. Evidence of incipient failure is frequently found in the disks of engines being overhauled for other reasons.

The stress and temperature environment in which the disk operates is reviewed. A discussion based on this review is made of the possible mechanisms of failure. Service experience is reviewed to show which of these mechanisms are known to have resulted in failures. The disk failures encountered in service were found to have resutled from (1) tension in the serration area; (2) radial cracking of the rim, particularly at bucket-retaining pin-holes; (3) fracture at the weld in composite disks; or (4) failure of the shaft.

Two suggestions for avoiding flight failures are made. They are for design improvements in the disk-cooling system and for scheduled inspections to detect cracks, abnormal growth, and distortion in the bucket-root area. The inspection schedule for each disk type should be based on service experience with the specific disk.

INTRODUCTION

The U.S. Air Force jet accident summary for 1953 (ref. 1) attributes 17 major airplane accidents to failure of the turbine disk and shaft. Of these, 10 destroyed the airplane and 3 resulted in fatalities.

In order to recognize fully the effect of turbine-disk reliability on over-all engine reliability, it is first necessary to learn the types and frequencies of the difficulties that have occurred in current operational practice. If the reasons for the difficulties can be learned, it should be possible to improve reliability of the disk. The causes of disk failures will be presented by discussion of

(1) Mechanisms by which disks fail (In addition to fragmentation, failure is considered to have occurred when defects arise that make further operation imprudent.)

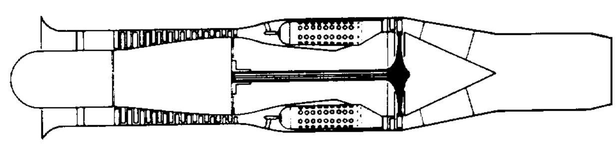
(2) Effect of disk environment (stress, temperature, corrosive media, etc.) on disk life during both normal and above-normal operating conditions

(3) Influence of manufacturing processes

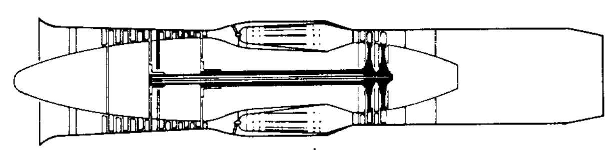
(4) Influence of maintenance

The available knowledge of the disk life being achieved in military operation is reviewed and the mechanisms of failure are discussed. Suggestions for achieving improved operational reliability are also presented.

Most military experience has been with single-stage turbines; therefore this discussion will deal primarily with problems that have been encountered in single-stage turbine disks. However, it is recognized that current design trends indicate increased use of multistage turbines. The mechanisms of failure that will be discussed apply also to the disks in multistage engines; these engines have the additional problem of being more difficult to inspect. Typical single-stage and two-stage turbines are shown in figure 1.



(a)



(b)

(a) Single-stage turbine engine.

(b) Two-stage turbine engine.

Figure 1.—Location of turbine disks in turbojet engines.

IDENTIFICATION OF TURBINE-DISK PROBLEMS

SOURCES OF INFORMATION

The service experience with disks used in currently operating turbojet engines is of interest as an initial step in a study of disk reliability. This experience, being solely military, may be different from that which would prevail if the engine were to be used commercially. Therefore, extension of the available information to nonmilitary usages must be made cautiously.

The available sources (see ch. II) indicate that turbine-disk malfunction is only rarely responsible for sending the engine to overhaul. However, upon disassembly during overhaul, as many as 76 percent of the disks in one engine model were found to have defects that could ultimately lead to failure. It had been hoped that through a study of the information sources it would be possible to ascertain the present disk reliability and also to establish a distribution curve for disk life. This could not be done, however, since operating bases make many unrecorded disk changes, and there are no data to indicate whether disks were replaced at prior overhauls.

The sources were very helpful, however, in indicating that a disk problem existed and in determining the various modes of disk failure. The gravity of certain types of disk failure is illustrated in figure 2. In this case, the disk burst while the airplane was in flight, completely severing the tail section from the rest of the airplane. The figure illustrates the manner in which the disk fragments tore the aft fueslage section, in the plane of the turbine wheel. The remainder of the airplane crashed and burned. This example emphasizes the importance of the disk in jet-engine reliability and indicates that even a small percentage of disk bursts is not tolerable. Every reasonable effort must be made to ensure that such failures do not occur, regardless of the number of engines in operation.

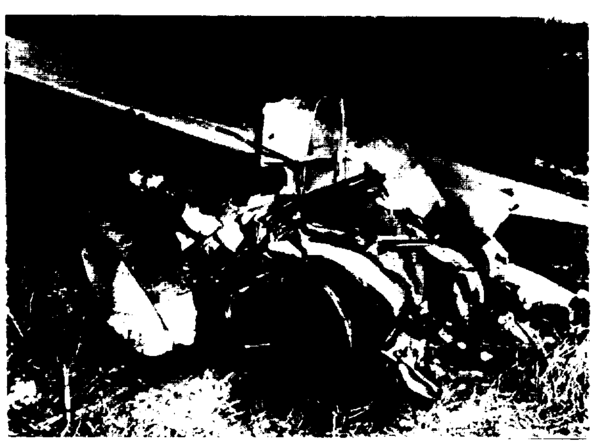


FIGURE 2.—Tail section of airplane that crashed as result of turbine-disk failure.

DEFINITION OF TURBINE-DISK REGIONS

For the subsequent discussion in this report, it is convenient to establish a standard nomenclature for the various disk parts. The terminology used is shown in figure 3. The terms are as follows:

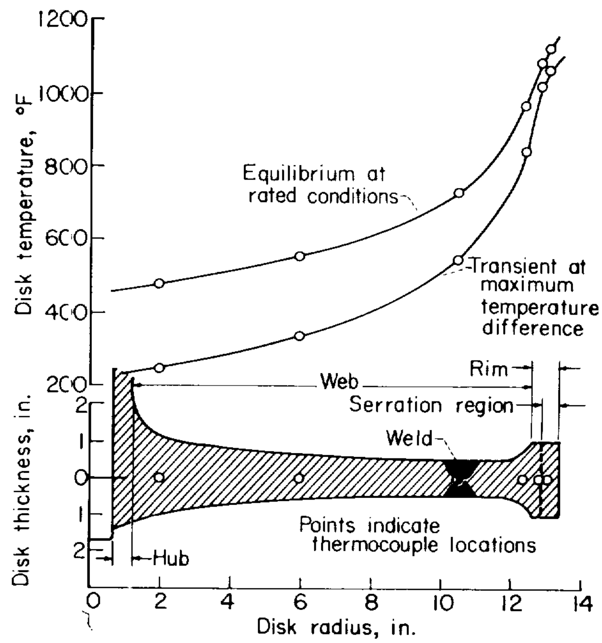


FIGURE 3. Turbine-disk temperature at transient and equilibrium conditions.

Serration region: The serration region is that portion of the disk beyond the largest radius at which the disk is crcumferentially continuous. The serrations support the turbine buckets and are also called the bucket-fastening region of the disk.

Rim region: The rim region contains the bucket-fastening region and extends radially inward to where the thickness of the disk substantially diminishes.

Web region: The web, usually thinner than the rim, extends from the hub to the rim. In composite disks, the web contains the weld area.

Hub: The hub is the central portion of the disk. In integral disks, the hub is the axial projection of the shaft through the disk. In demountable disks, the hub contains the bolting circle, mounting flanges, and so forth.

The term "turbine wheel" refers to the turbine disk and the bucket assembly.

TURBINE-DISK OPERATING TEMPERATURES

The turbine disk is heated by the hot gases flowing over the rim, by conduction from the turbine buckets, and by radiation from adjacent hot parts such as the tailcone bullet.

Cooling air may be bled to the central areas of the disk from the high-pressure stages of the compressor (fig. 4). This air exhausts through the seals and baffles that prevent the combustion gases from inadvertently spilling over the disk sides. In addition, the cooling air from the upstream face of the disk flows over the disk rim and cools the bucket-fastening region.

Typical radial temperature distributions in the central plane of a disk cooled on both sides are shown in figure 3. The upper curve represents conditions at rated speed and tailpipe temperature, after sufficient time has elapsed to allow the disk temperatures to become stabilized. There is a temperature difference between the hub and the rim of about 620° F. This temperature difference and the shape of the curve are important in determining the stress state to which the disk is subjected; in general, the greater the temperature difference, the greater are the thermal stresses. This relation is discussed in greater detail in a subsequent section of this paper.

The maximum difference between temperature at the hub and at the rim generally occurs shortly after the engine is started: rim temperature rises rapidly to very near the operating value, while the hub heats slowly as shown by the lower curve of figure 3. A typical example of this relation is shown in figure 5. The engine was accelerated at a rate considered normal for jet-engine operation; rated speed was attained 1 minute after ignition. The maximum temperature difference of 790° F, 170° F greater than that at equilibrium, was reached 5 minutes after ignition. The equilibrium temperature difference was reached about 22 minutes after ignition.

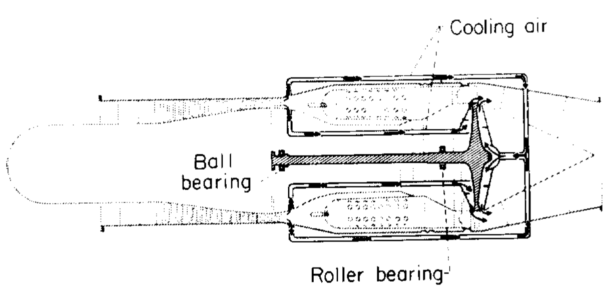


FIGURE 4.—Schematic diagram of disk cooling-air system.

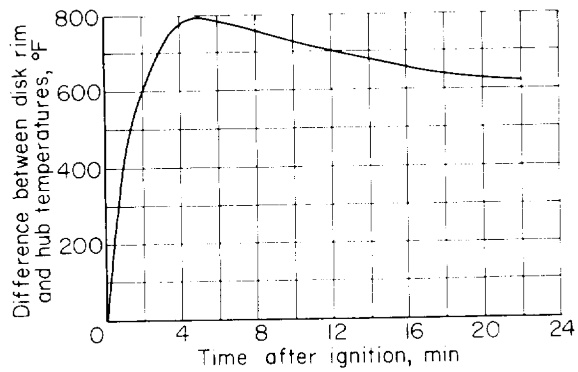


FIGURE 5. Temperature difference between hub and rim of disk as function of time after ignition.

In addition to the radial variation in temperature of the turbine disk, there also may be substantial axial variations. Typical axial gradients in a disk cooled on both faces are shown in figure 6. These gradients were measured at rated speed and tailpipe temperature after equilibrium had been reached. Midway in the rim, a temperature difference of 120° F between the faces of the disk can be noted. Such axial temperature gradients are troublesome because there are no established procedures for calculating the stresses they induce; some uncertainty is thereby introduced in computed stress values.

Thus far, disk temperatures during normal starting and operation have been considered. There are several conditions for which the temperature gradients may be higher than these. These conditions include

(1) Engine-starting at very high altitude. The engines are frequently very cold and the temperature gradients are more severe.

(2) Overtemperature operation of the engine. Severe overtemperatures would be likely to cause severe overheating of the bucket-fastening region and possibly failure in this region even before the temperature rose significantly in the web. Prolonged moderate overtemperatures could result in excessive overheating of the web area.

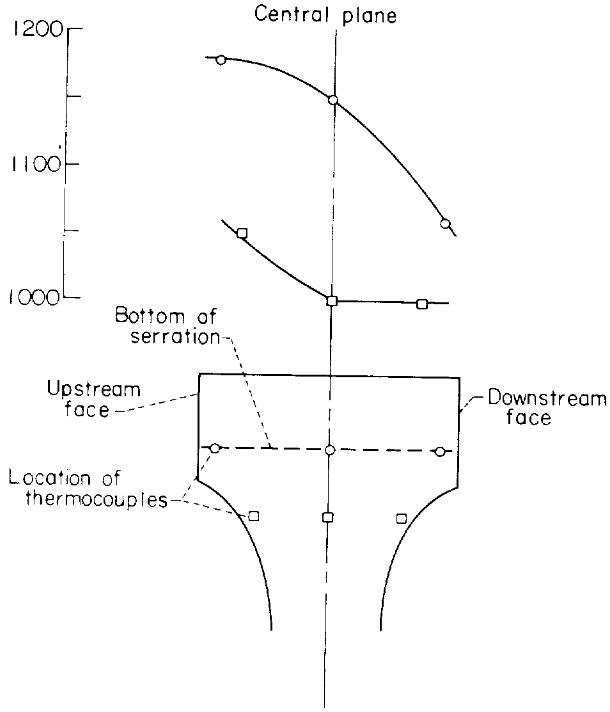


FIGURE 6. Typical axial temperature gradients in turbine-disk rim.

(3) Operation of the engine with malfunction of the seals and baffles intended to prevent hot gases from flowing over the disk faces. These seals are difficult to design because they must allow for the large unpredictable expansions that occur in the turbine section. Part of the difficulty arises because, in most engines, the ball bearing that locates the disk relative to the frame of the engine is located at the forward end of the turbine shaft, as shown in figure 4. The axial clearance between the seals and the disk is therefore changed by the difference in axial expansions of the turbine shaft and of the engine frame. In one engine, this axial clearance change was found to be approximately 3/8 inch. Also, the normal cyclic operation of the engine causes a gradual distortion, or warpage, of the baffles and further decreases their effectiveness. This warpage may merely decrease sealing effectiveness and allow the hot gases to flow over the faces of the disk, or in an extreme case it may actually direct the hot gases over the faces of the disk.

Additional turbine disk temperature information is presented in references 2 and 3.

DISK STRESSES AND RELATED PROBLEMS

HUB AND WEB REGIONS

Centrifugal stresses.—Although one of the primary purposes of the turbine disk is to transmit torque to the compressor, the stresses produced in the disk proper in performing this function are influenced only slightly by torque. The major stresses are caused by other types of loading. Of these, the primary ones are centrifugal forces, the thermal gradients just discussed, and the locked-in internal loadings resulting from certain steps in the fabrication process and from the preceding engine operation. Measurement of these internal loadings is difficult and not much is known about them.

The centrifugal forces acting on the turbine buckets are transmitted through the bucket fastening mechanism to the disk proper. In determining the resulting stresses produced in the disk, it is generally assumed that these forces are evenly distributed on a fictitious rim that corresponds to the continuous rim at the base of the serrations. The rim loading is determined by dividing the total centrifugal forces on the buckets and disk segments by the circumferential area of the fictitious rim. The problem is thus reduced to one of a simple disk with a smooth and continuous rim having an evenly distributed load acting in a radial direction. In addition to the rim loading, rotation of the disk also causes centrifugal forces that act on each element of the disk in direct proportion to its mass and its distance from the center of rotation and in proportion to the square of the angular velocity.

The assumption is generally made that, at any given disk radius, centrifugal stress will be constant from one face of the disk to the other. In other words, in the computation of the stresses, a smoothly faired disk profile is assumed that will permit even distribution of the stresses in the axial direction.

Since axial effects are assumed negligible, the resulting computed stress state in the disk proper is biaxial with the principal stresses occurring in the radial and the tangential directions. For illustrative purposes, the computed stresses due only to the centrifugal force field for a typical turbine disk operating at its rated speed are shown by the dashed lines of figure 7. The disk

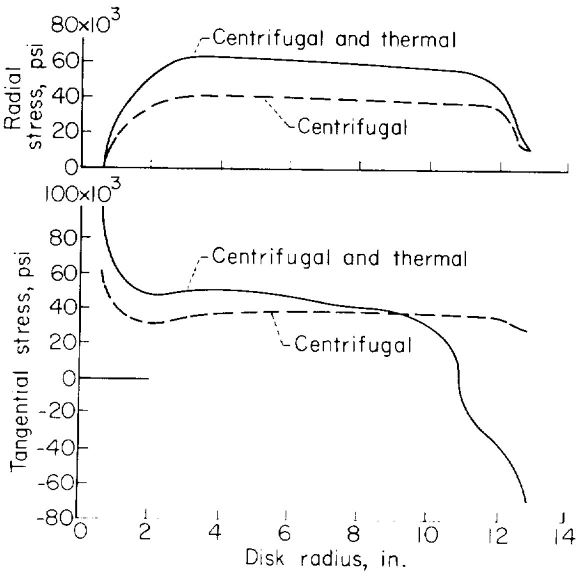


FIGURE 7.—Thermal and centrifugal stresses in typical turbine disk.

chosen had a rim loading of 11,000 psi and had a 1.31-inch-diameter hole in the center. Radial stress is zero at the hole and rises to a maximum of 40,000 psi in the web region of the disk. Radial stress at the rim is the rim loading of 11,000 psi. Tangential stress at the hole is 62,000 psi and decreases rapidly with distance from the hole. In the web region of the disk, tangential stress is 38,000 psi and decreases to 28,000 psi at the rim. If the disk were solid instead of having a central hole, the radial and the tangential stresses would have been equal at the center with a value of about 35,000 psi.

Thermal stresses.—During normal engine operation, the turbine disk is subjected to a radial temperature distribution as described previously for the illustrative case shown as the upper curve of figure 3. At equilibrium conditions, the rim is 620° F hotter than the hub. The rim material tends to expand to a new radius, but is restrained from doing so by the cooler hub material. The rim is therefore forced into compression, while the center or hub is forced into tension. These effects are superimposed on the stress state created by the centrifugal forces, and the combined result is that shown by the solid lines of figure 7. The tangential stress at the hole has increased from 62,000 to 98,000 psi, while at the rim it has been changed from a tensile stress of 28,000 psi to a compressive stress of 68,000 psi. Radial stresses in the web region of the disk have increased from 40,000 to 60,000 psi with no change at the hole or at the rim. At a radius of 10.5 to 11.0 inches (the weld region) there is now a rather sharp break in the tangential stress curve. This break is caused by differences in the coefficients of expansion and Young's modulus for two materials of which this disk is fabricated.

The tangential stress at the hole of the disk is of the order of 100,000 psi. The approach of this stress to the yield point of the material does not necessarily constitute a dangerous operating condition. Materials that provide a substantial amount of center ductility are chosen so that plastic flow can occur and can result in a redistribution of the stresses. Trouble may be encountered, however, if flaws exist at the center, since these would reduce the ability of the material to flow and could result in a disk failure. The effect of ductility on the strength of disks is discussed more fully in reference 4.

The stresses described can be computed by the method described in reference 5 for the case where the yield point of the disk material is not exceeded, or reference 6 for the case where the stresses do exceed the yield and plastic flow takes place. An extension of the method of reference 6 can also be used to determine the residual stresses that will be locked in the disk after engine operation. Semiquantitative evidence of these residual stresses has been noted during cutting of the disks along a diameter after engine operation. As the cut reaches approximately 80 percent of the diameter, the cut at the starting point has been observed to open up as much as 3/8 inch. These residual stresses combine with the other stresses in the disk to produce some regions of higher than normal operating stresses and some lower. The over-all influence on the reliability of the disk is not clear, since no correlation with disk failures has been established.

Center bursts.—Since the hub region operates at temperatures of about 400° F, design is based on the short-time physical properties of the material rather than on the time-dependent stress-rupture and creep properties. Yield point of the material is therefore used to determine the proper profile for the hub region of the disk. However,

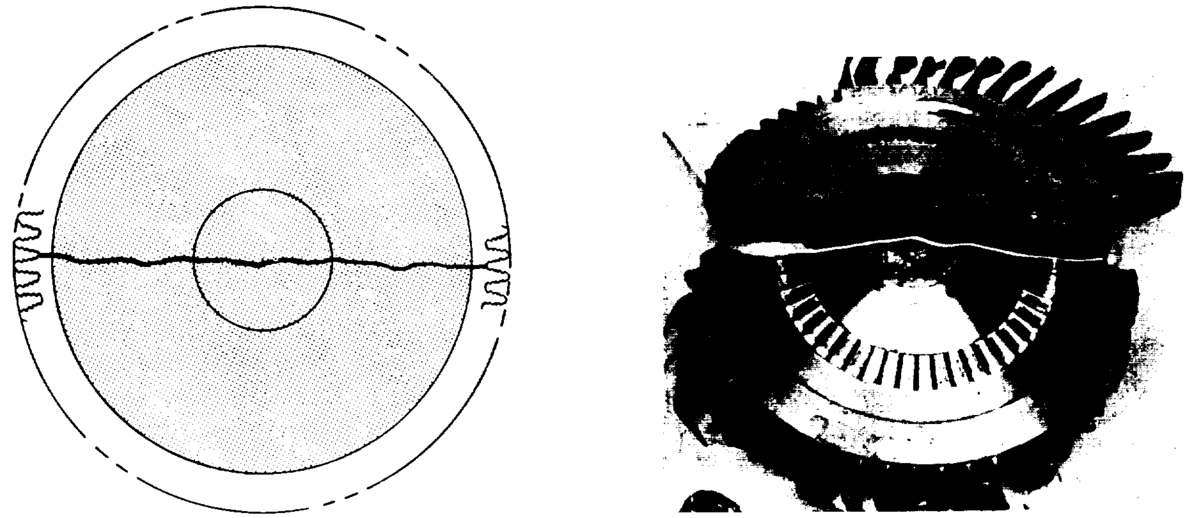


FIGURE 8.—Typical turbine-disk burst.

because engine weight is influenced to a large extent by the weight of the turbine disk, an attempt is made to obtain the profile that will give the lightest possible disk and yet be safe under all operating conditions. Every possible precaution must therefore be observed to ensure that the center material has the design strength. Since the center is generally the thickest part of the disk, it often does not receive much mechanical working during the forging process; it may therefore contain defects such as segregations and porosity. These defects, which reduce material ductility, may cause a center burst with catastrophic results to the airplane. Figure 8 shows a center burst caused by excessive segregations in the hub area. The disk was split into two almost equal parts.

Because center bursts occur without warning, this type of failure can only be avoided by tighter controls during the fabrication processes and by improved preoperation inspection procedures. Improved X-ray, ultrasonic, and penetrant oil inspection procedures in addition to destructive examinations on a sampling basis can do much to eliminate the hazards of a center burst. As a final check before acceptance, it is general practice to cold-spin the bucketed disks to a speed somewhat above the rated engine speed. This practice assumes that the centrifugal stresses induced by the overspeed will in effect take the place of the combined centrifugal and thermal stresses when operating in an engine and thereby proof test the disk. However (referring again to the stresses in the typical disk shown in fig. 7), it will be noted that it would be necessary to increase the centrifugal tangential stress from 62,000 to approximately 100,000 psi to simulate the combined thermal and centrifugal conditions at the disk center. This increase would require an overspeed of approximately 27 percent, which would probably cause a failure in the outer regions of the disk, perhaps in the bucket-fastening area. Cold-spin testing is therefore performed at overspeeds of the order of 5 to 10 percent and does not simulate actual operating conditions. A cold-spin test should be considered only as adjunct to the other inspection procedures and not as a final proof of reliability of a disk. It is entirely possible that overspeed testing can do more harm than good. For example, too great an overspeed may cause localized plastic flow near the rim or at stress concentrations near the center because of defects passed by other inspection methods. The spin test may therefore induce residual stresses in the disk or cause existing defects to be enlarged. In either case, severe overspeed spin-testing could reduce the reliability of the disk during subsequent engine operation.

Field and overhaul maintenance procedures that may influence reliability of the disk are discussed in a later section of this report.

Web necking and burst.—The web region of the disk operates at higher temperatures than the hub (fig. 3). For the typical disk shown, the temperature is about 750° F at the weld location and about 1150° F at the rim. In the design of the web, it is therefore necessary to determine for each radius whether the short-time physical properties or the stress-rupture and creep properties are the controlling design criteria. In addition, a substantial margin of safety must be incorporated to allow for disk temperatures higher than normal because of inadvertent overtemperature operation of the engine or such malfunctions of the disk-cooling mechanism as were previously discussed.

If substantial overtemperature operation is encountered, the web may be subjected to creep that may lead to severe necking with an accompanying increase in the over-all diameter of the disk. This condition is shown in figure 9. A composite disk is shown, although the difficulty could also occur with disks fabricated from a single alloy. The dashed line shows the original contours of the disk and the solid line the contour after necking has taken place. In this case, considerable deformation took place in the weld area. A photograph of a necked disk is also shown in figure 9. The face of the disk was originally a straight-line contour; the light shining through

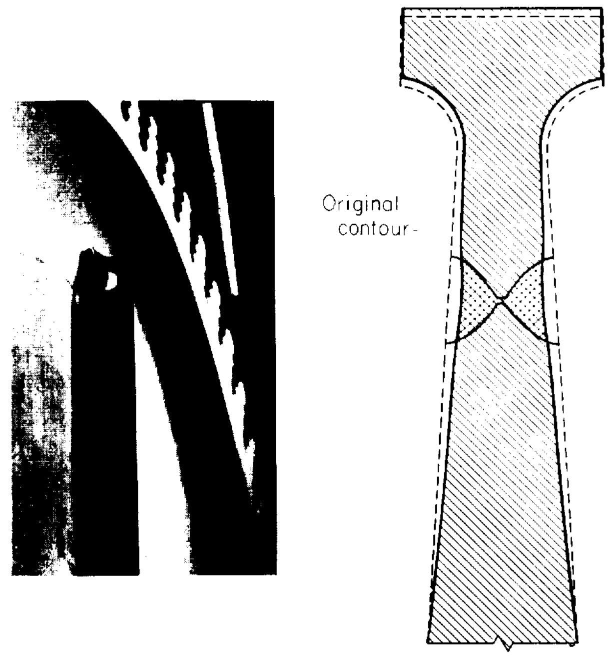


FIGURE 9.—Growth of welded turbine disk.

between the straight edge and the disk indicates that necking has taken place.

If allowed to proceed, the deformation or necking mechanism can lead to a burst such as that shown in figure 10. The photograph shows a disk fabricated of one material that had been operated in an engine having warped cooling-air baffles. This type of failure can be as catastrophic as a center burst.

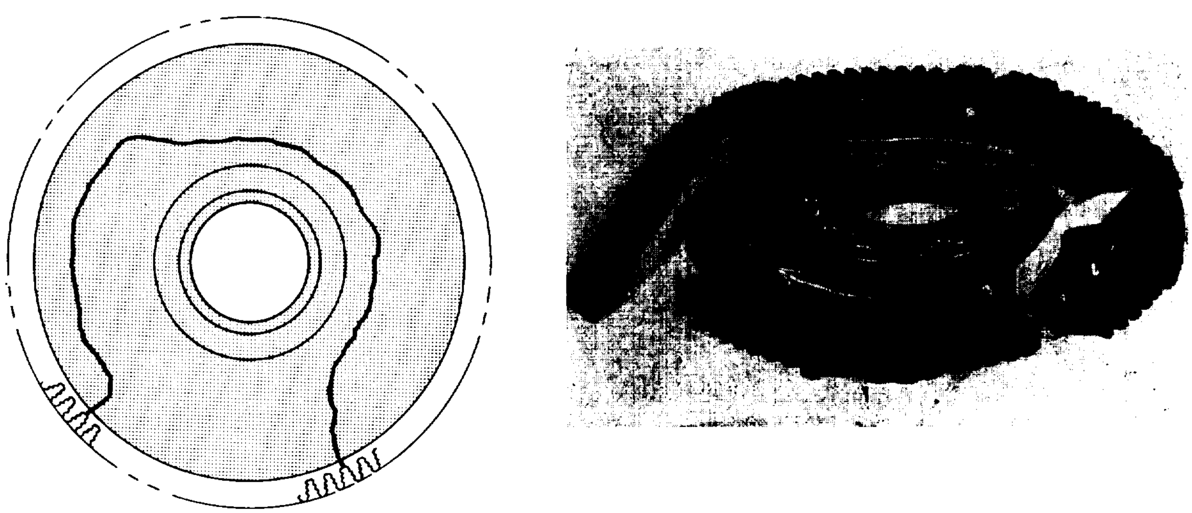


FIGURE 10.—Turbine-disk burst as result of excessive growth of web.

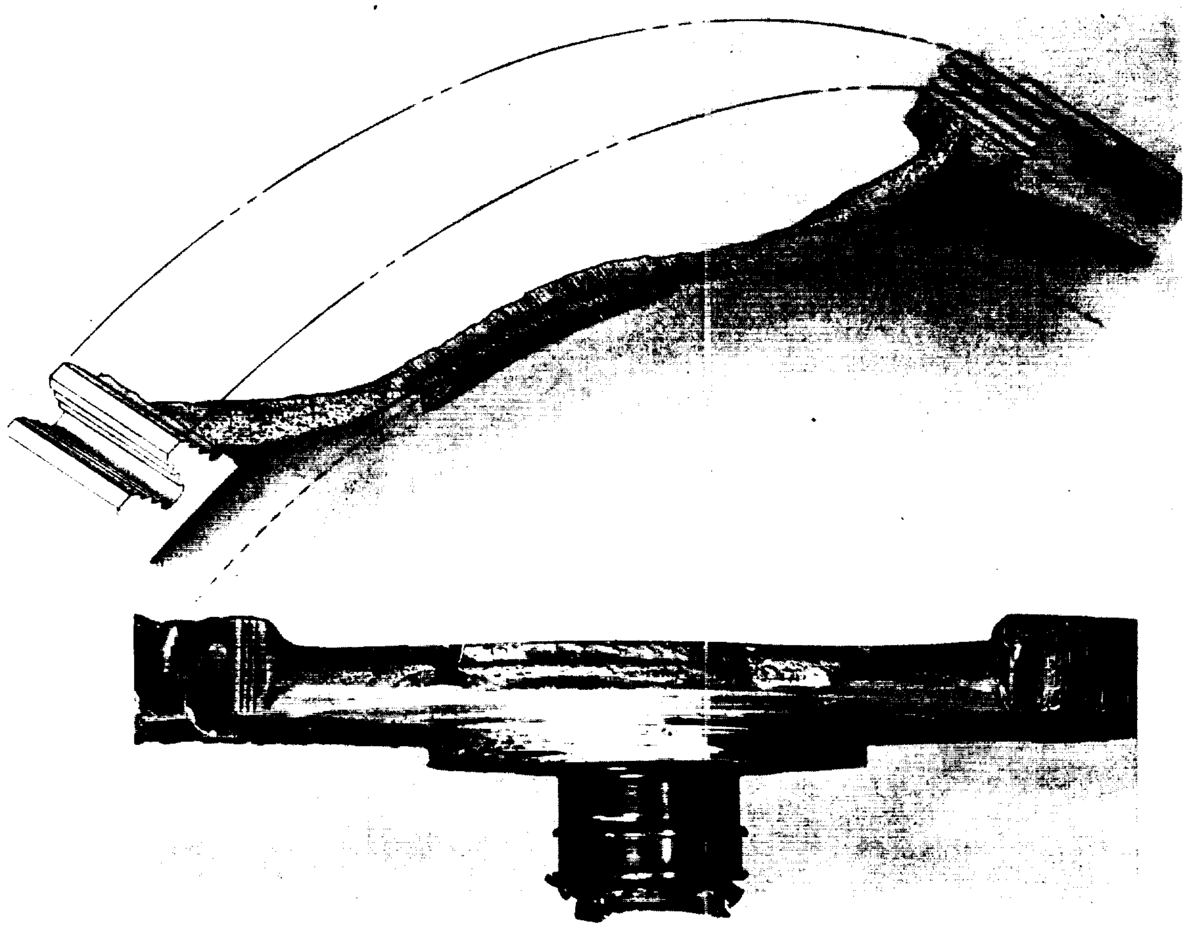


FIGURE 11. Weld failure.

In another case, deformation caused a failure at a weld defect in a composite disk (fig. 11). The defect was caused by lack of fusion between the weld metal and the hub material and the presence of slag and porosity in the weld. The engine was damaged beyond repair.

The possibility of a web burst can be reduced in several ways. First, it is necessary to ensure that the web material has the proper physical properties to withstand the design stresses. This requirement is particularly to be emphasized for composite disks that employ welds, since these are susceptible to a number of different types of serious defects. X-ray, ultrasonic, and penetrant-oil inspections should be used in addition to destructive testing on a sampling basis. Second, it is necessary to avoid engine operation at conditions that will permit disk temperatures to rise above their design level. Excessive web temperatures can occur as a result of (1) prolonged operation at excessive turbine-inlet temperatures, (2) warping, distortion, or improper installation of cooling-air baffles and seals, and (3) the failure of any portion of the ducts that supply cooling air to the disk. Obviously, the improvement of any part of the engine, such as the controls or the cooling-air system, which will tend to ensure proper disk temperature will improve reliability of the disk. Since faulty installation of the cooling system can lead to overheating of the disk, a device for warning of excessive disk temperatures or loss of cooling airflow would increase reliability. Third, particular attention should be given to the design of the various structures adjacent to the faces of the disk to minimize possibility of their coming loose and moving against the rotating disk. Several disk failures in the service records are attributed to this cause.

RADIAL CRACKS IN RIM REGION

Thermal-fatigue cracks.—As stated previously, the combined effects of centrifugal loading and thermal gradients induce high compressive stresses in the rim area of the turbine disk at rated-speed operation. During starting, the thermal gradients in the disk are even more severe; as a consequence, the stresses are further increased to the point where they frequently exceed the yield strength of the alloy. This induces an immediate compressive plastic flow. In some cases further

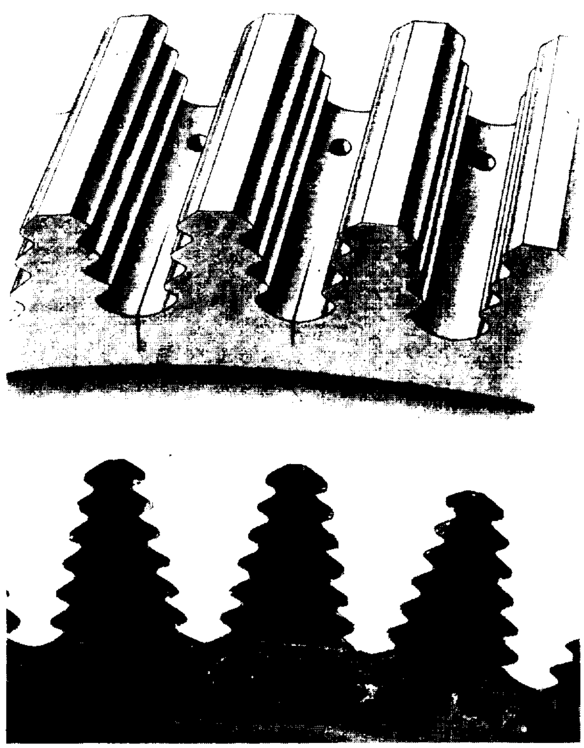


FIGURE 12. Thermal-fatigue cracks.

plastic flow can take place (at a much reduced rate) by a creep mechanism. This creep can also occur when the stresses induced by the temperature gradient are insufficient to cause immediate plastic flow. In either case, on engine shutdown the plastic flow that occurred at high temperature will induce a residual tensile stress; this stress may also exceed the yield strength of the material and cause plastic flow in tension. Repetition of this cycle in successive starts and stops eventually consumes the capacity of the alloy to deform and the rim cracks. This mechanism of failure, termed thermal fatigue, is described in reference 7. Figure 12 shows a disk cracked by thermal fatigue. This type of failure is cycle dependent and generally occurs late in the engine's life. In one engine, the cracks were not noted until after approximately 400 hours of operation. Although these cracks have been known to progress to failure (ref. 8), this appears to happen only rarely. Consequently, it can be concluded that the rate of progression is sufficiently low that the cracks can be noted and the disks rejected at normal overhaul periods.

In some of the newer engines, which utilize very thin disks, the deformation per cycle appears to be much greater; cracks have been noted after as little as 1 hour of operation. One possible reason for this rapid deterioration is overtemperatures during starting.

As disk development advances and design defects that cause early failures are eliminated, thermal-fatigue cracking will probably become a more predominant mode of disk failure. At present, there does not appear to be sufficient background information on the thermal-fatigue properties of alloys to permit the designer to predict the thermal-fatigue life of a disk. He must rely on frequent inspection so that cracked disks can be detected and removed from service before the cracks progress to catastrophic failures. Since thermal-fatigue cracks are cycle dependent, inspections based on the logged number of engine starts would be desirable. Because thermal-fatigue cracking is made more likely by overtemperatures, additional inspections after hot starts would be desirable. For this purpose, an instrument that records time-temperature history of the engine and speed of the turbine would be of considerable assistance.

Pin-hole cracks.—The pin-hole crack (fig. 13) is a special case of radial rim cracking. These cracks originate in the vicinity of the bucket-retaining pin hole and progress to both the forward and the aft faces of the disk. They can be readily detected by a sensitive surface-inspection technique that employs fluorescent oil, which is emulsified after application. The causes of the cracking appear to be complex; several factors that are believed to contribute to the failures are

(1) Stress corrosion in the area cold-worked by the broaching operation and cold-worked by scratching during the removal of turbine buckets

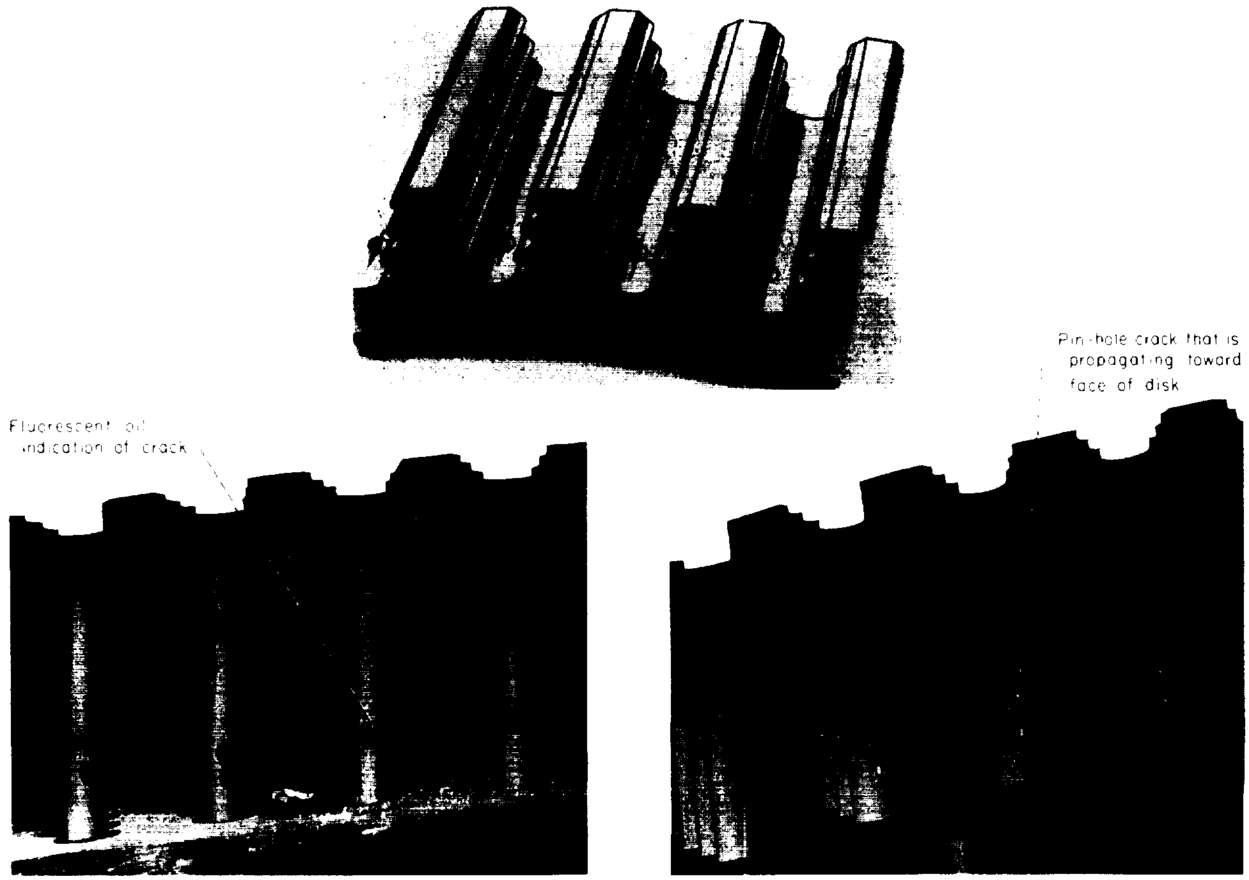


FIGURE 13. Pin-hole cracks.

(Fig. 14, which is from ref. 9, shows the intergrannular corrosion which accompanies the cracking.)

(2) Concentration of stresses or scratches and tool marks at the pin hole itself

(3) Stressing by the previously described thermal-fatigue mechanism

As in the case of the thermal-fatigue cracks, pin-hole cracks can lead to destruction of the turbine disk. The frequency of occurrence of this type of cracking varies with engine type. In one engine model, 76 percent of the disks in engines coming to overhaul (for causes other than disk difficulties) were rejected because of pin-hole cracks. In this engine the cracks seldom occur in only one pin hole. Usually, when they are first detected about five pin holes are found to contain cracks (refs. 10 and 11). The number increases rapidly with additional operating time until most or all of the pin holes contain cracks. Although the average time for the first appearance of the cracks is quite long (256 hr), they have been noted in as few as 22 hours. The cracks seem to progress rather slowly; in one test (ref. 10) 30 minutes at 4-percent overspeed and at slight

FIGURE 14.—Intergranular corrosion and crack at pin hole. Unetched; X100; from reference 9.

overtemperature did not increase the size or length of the cracks. Crack propagation to the aft face of the turbine disk (the face nearest the pin hole) does not appear to be harmful. In a run in which saw cuts simulated cracks aft of each pin hole, no deleterious effects were detected after 50 hours of an accelerated test schedule. Cracks have progressed from the pin hole to the forward face of the disk in about 400 hours of engine operation.

Of all the sources of failure, the pin-hole crack appears most amenable to elimination. Many of the newer engines have eliminated the pin hole and have substituted "cotter pin" type bucket-retaining devices. This change eliminates not only the stress-concentrating effect of the hole but also the scratching sometimes caused by a sheared pin during bucket removal. In substituting the cotter pin for the retaining pin, the cotter-pin slot machined into the bottom of the serrations should have large fillets. Otherwise, as described in reference 10, cracks may propagate from the sharp corners.

SERRATION REGION

The principal centrifugal and thermal stresses acting on the turbine disk are discussed previously. In addition to these stresses, there are several other important stresses which act locally in the serration region of the turbine disk that may lead to disk failure. These are the tensile, shear, and bearing stresses, which will be discussed separately.

In some respects, the serration region is the most critical part of the turbine disk because

(1) The serration region is at the highest temperature and the life is very sensitive to temperature. This relation is discussed in greater detail in a subsequent section of this report.

(2) The serrations are nearest to the hot gases and are subject to frequent and drastic temperature changes.

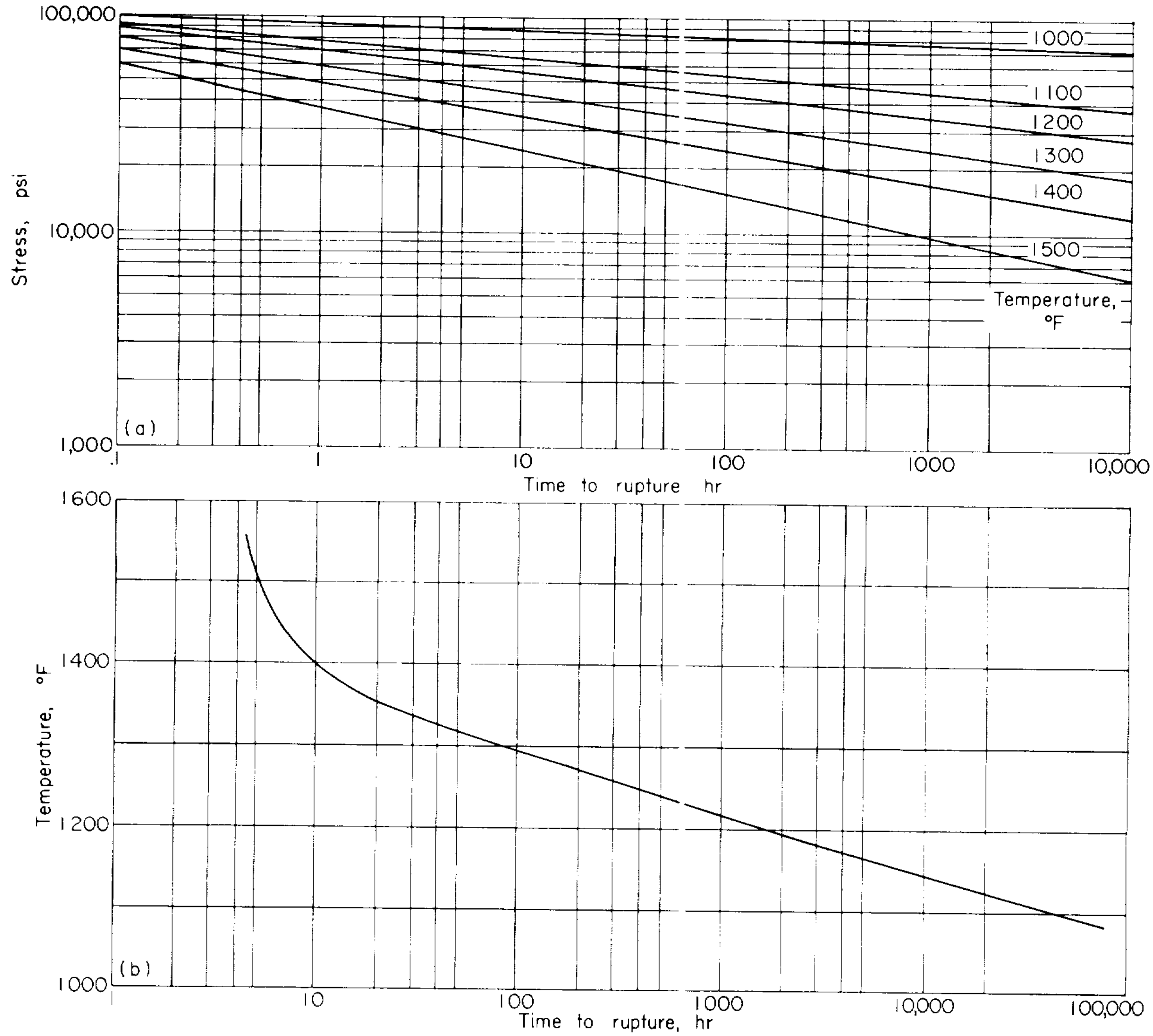
(3) Axial-temperature gradients (see fig. 6), which are difficult to consider in design, impose added stresses.

(4) The serrations have a complex shape involving many small radii.

Serration tensile failures.—Tensile failures in the serration region may result from inability of the serrations to withstand centrifugal pull of the blades. If failures occurred at the terminus of the stress-rupture life of the alloy, they would be regarded as normal wearout and the removal of disks from service could be scheduled on the basis of stress-rupture life, with allowances for the scatter band. However, stress-rupture life of the rim material is generally considerably greater than the time for the first appearance of serration cracks. The failures are attributed to one or more of the following reasons: (1) The temperatures exceed those anticipated in design; (2) the alloy does not have the expected life; (3) the stresses are not those anticipated in design; and (4) vibratory stress is present.

The life of the disk is very sensitive to operating temperature; if the temperature is exceeded, the life is drastically reduced. This is illustrated for the typical disk alloy 16–25–6 in figure 15. In figure 15(a) are the conventional stress-rupture curves for the alloy, and, for illustration purposes, a cross plot at a constant stress of 35,000 psi is given in figure 15(b). It can be noted that, if the disk-rim temperature is allowed to increase either through malfunctioning of the engine or through the inadequacy of the design estimate by even 50° F, the life is severely curtailed. For example, at 1200° F (see fig. 15(b)) and a stress of 35,000 psi, the expected life is 1000 hours. If the operating temperature is increased to 1250° F, the life is reduced to 200 hours. Conversely, if through the readjustment of disk to gas-baffle clearances or through an increase in cooling air to the disk faces the temperature could be reduced 50° to 1150° F, the life could be prolonged sixfold and an added margin of safety attained.

In addition to reducing stress-rupture life, overtemperature can cause metallurgical changes in the alloy that reduce the strength so that the failure can occur during subsequent operation at normal temperatures. This effect is similar to that of improper heat treatment of the alloy during processing, which can also cause premature tensile failure in the serration region.



(a) Stress-rupture life of 16-25-6 alloy.

(b) Sensitivity of stress-rupture life to temperature at 35,000 psi.

FIGURE 15.—Effect of temperature on life of 16-25-6 alloy.

An example of a structural change that reduces life is given for 16–25–6 alloy in figure 16. The heavy precipitate shown was produced by overheating test bars in the laboratory; similar structures have been reported present in turbine disks having serration tensile failures. This change greatly embrittles the alloy.

A situation in which the strength may not be that relied upon by the designer is one in which the notch-rupture properties predominate in controlling life. Reference 12 shows that at 1200° F and a mean stress of 40,000 psi, the life of 16–25–6 alloy is reduced from 400 to 15 hours by the presence of an 0.012-inch-radius notch of the depth used in the serration of a current turbine disk.

The notch-rupture behavior of an alloy may also change under conditions that cause failures at long times. An example of this is afforded by the alloy 17–22A(S) which is currently being considered for turbine disks (ref. 13). Figure 17 shows the rupture strength for both smooth and notched bars. At stresses that cause failure in

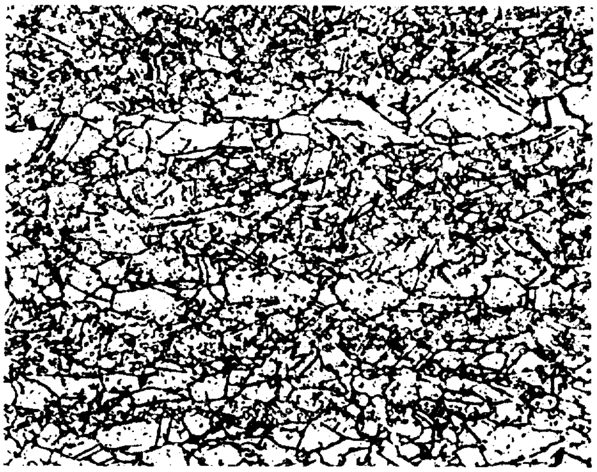


FIGURE 16. Structure of 16-25-6 alloy after aging. Etchant, aqueous solution of 90 percent hydrochloric acid plus 10 percent nitric acid; X500.

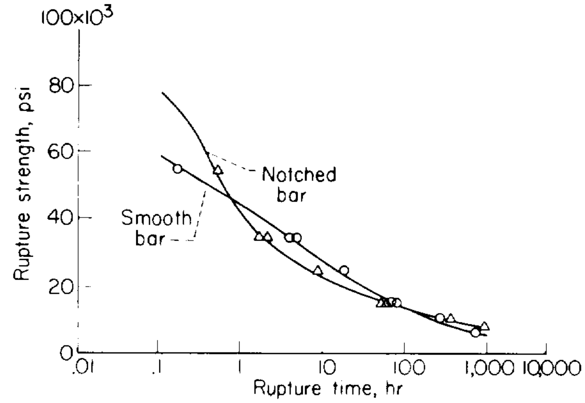


FIGURE 17. Stress-rupture properties of smooth and notched bars of 17–22A(S) at 1200° F.

short times, the notched bar is stronger than the smooth bar. In this condition the alloy is said to be "notch ductile." At the longer times, this situation reverses and the alloy becomes "notch brittle." This reversal is attributed to a precipitation reaction in the alloy and when, at the longest times shown, overaging has taken place, the situation again reverses and the alloy is again notch ductile. If the design stress is in the notch-brittle region, serration difficulties may develop.

Figure 18 shows a type of failure that results from inadequate strength in the serration region. Cracks in the fillets between the serration teeth portend incipient failure. The cracks can be detected by the sensitive postemulsion penetrant-oil surface inspection method. They generally originate in the axial center of the disk and occur principally in the radius below the bottom tooth but may occasionally occur in one of the upper

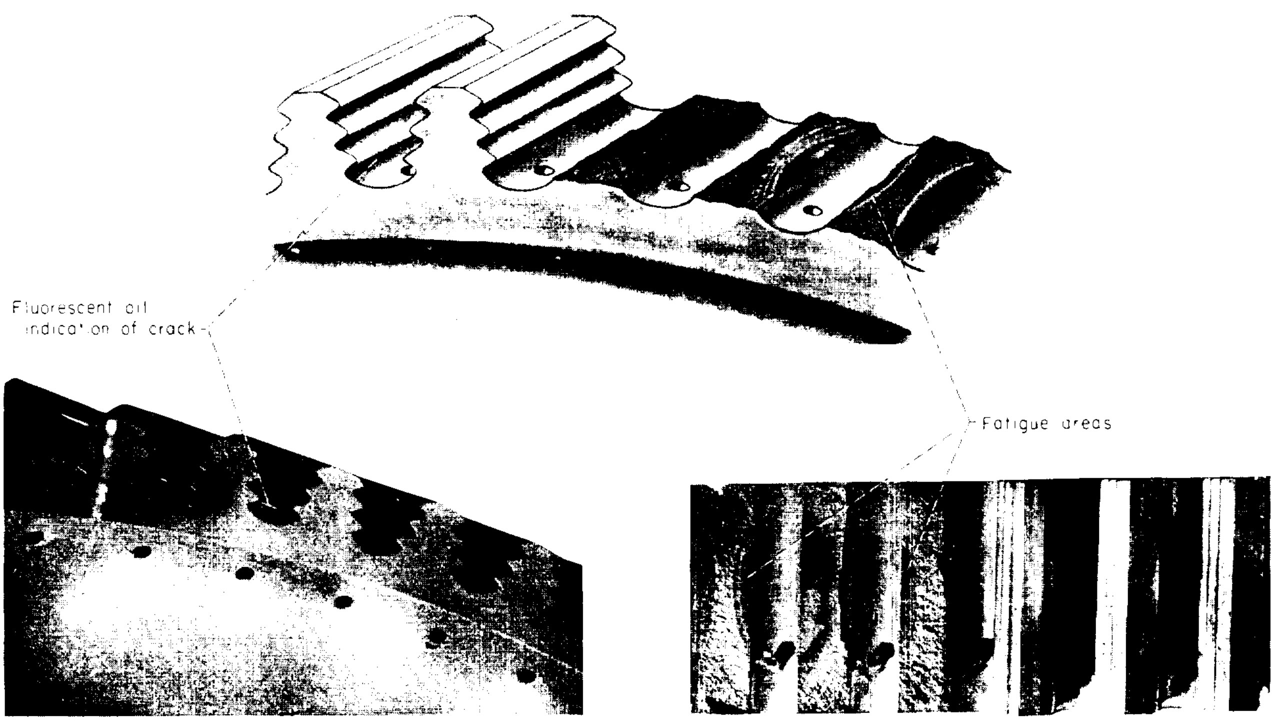


FIGURE 18. Serration failure.

teeth. Propagation of the cracks occurs by a fatigue mechanism; the radiating lines and semicircles that are indicative of fatigue can be seen in figure 18. The fractures seldom occur in only one segment; the usual case involves three to five serrations, but as many as seventeen have been noted.

In one engine type, this mode of failure was particularly troublesome: 60 such failures occurred in service, with approximately 45 percent of the engines coming to overhaul giving evidence of incipient failure. Time for the failure has varied from about 23 to 300 hours; there does not appear to be any correlation with running time. The rate of crack propagation from first appearance to failure has not been established. However, the fact that many disks are found during overhaul inspections to contain cracks implies a slow rate of propagation.

In a special form of serration tensile cracking, the cracks occur on only one side of the serration. Such cracks are shown in figure 19. If a disk is in a condition in which serration tensile cracks from any of the previously mentioned mechanisms are imminent, a small nonuniform load could cause the failure to occur initially in one side of the serration. Factors that might impose such a nonuniform load are (1) machining errors that shift the position of the root with respect to the airfoil, (2) inadequate compensation for gas bending loads, (3) warpage of the airfoil that tends to shift the center of gravity.

This type of cracking can occur at any tooth. In one engine type, it frequently resulted in loss of the outermost tooth without destroying the ability of the disk to hold the buckets. For this engine, cracks in the top serration are not a cause for disk rejection (ref. 14), and tolerance limits have been established for cracks in other serrations. Since the rate of propagation of the cracks is unknown, this latter procedure appears hazardous.

Shear failures in serration region.—There has been no evidence of shear failures occurring in service. This type of failure is included in this discussion, however, because shear failures have been encountered in spin-pit studies of turbine disks, and because studies of the shear strength of turbine-disk alloys indicate increasing likelihood of this type of failure as operating temperature is increased, or in the event of severe overtemperature. In one study (ref. 12) an example is given of a current disk design that has an expected life at 1200° F of about 200 hours in tension and about

Figure 19.—Serration tensile failures resulting from nonuniform loading.

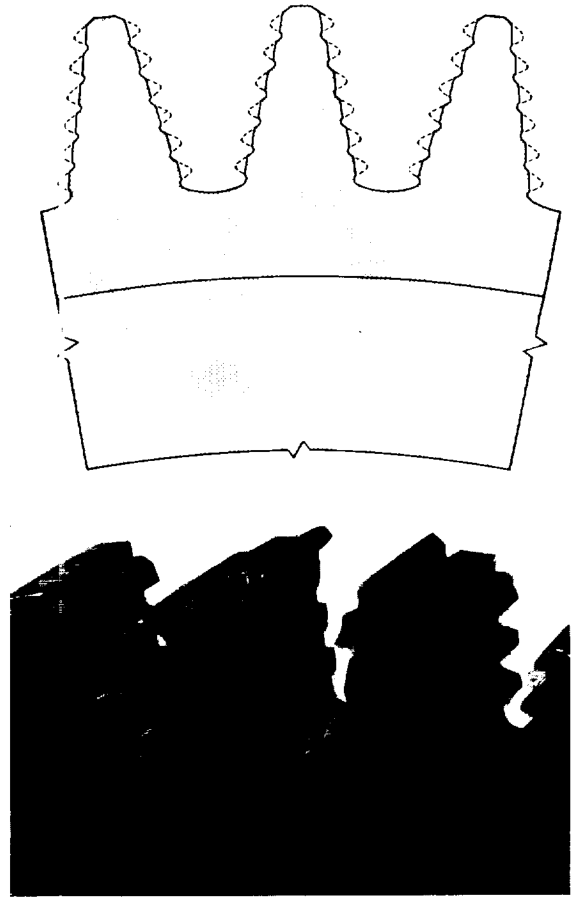


Figure 20.—Serration shear failure.

1500 hours in shear. This disk could be expected to fail by tensile stress-rupture. If the temperature were raised to 1350° F, both the tension life and the shear life would be decreased; but shear life now controls, and a failure in shear could be expected in about 16 hours.

Figure 20 shows a spin-pit disk that has suffered a shear failure.

Bearing failures in the serration region.—Stress analysis of the serration region indicates that the teeth are subject to substantial bearing stresses, although service experience does not record any failures attributed to them. A bearing-stress failure would be manifest by spalling and checking along the line of contact between the teeth and the turbine blades.

TURBINE-SHAFT FAILURES

A typical turbine-shaft failure is represented schematically in figure 21. Failure generally occurs at stress-concentration points such as those at the bearing-retainer threads. Since this region of the disk operates at relatively low stress, the failures are attributed to unusual operating conditions. Several possible causes are (1) torsional vibration of the shaft, (2) overheating of the shaft caused by faulty bearing operation, (3) failure of the cooling-air supply to the bearing, (4) sudden unbalance loads imposed by bucket failures, and (5) gyroscopic loads.

There are no data to indicate whether the failure goes through a detectable incipient stage, but the current practice of inspecting the shaft during overhaul by magnetic means or penetrant oil is undoubtedly sound. Shaft failure, while rare, is extremely destructive.

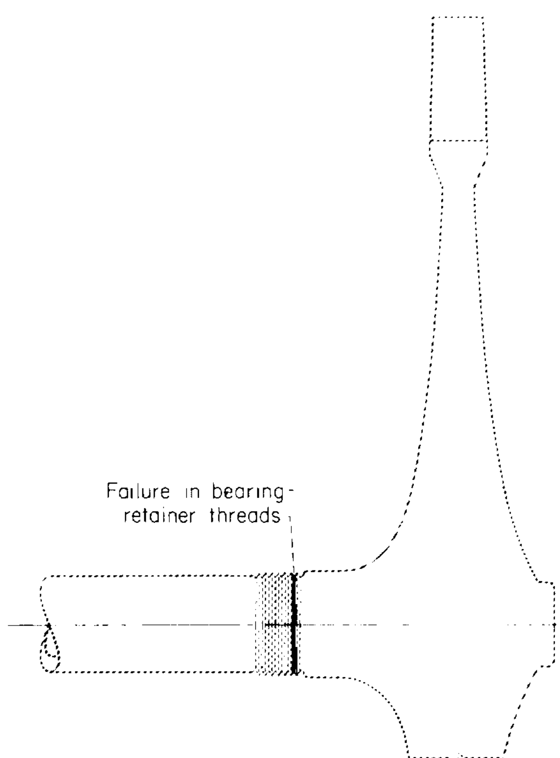


FIGURE 21.—Turbine shaft failure.

INFLUENCE OF MANUFACTURING PROCESS ON RELIABILITY

QUALITY CONTROL BY ALLOY PRODUCER

Billets to be forged into turbine disks are purchased by the fabricator on specifications of the individual engine producer. These specifications are elaborations of the standard Aircraft Material Specifications (AMS) and include standards for such items as grain size, homogeneity, and cleanliness that are not included in the AMS. Present practice is to preserve the identity of the heat, ingot, and ingot position; the engine producer records this information with the disk serial number. Should disk difficulties then develop, the production history of the disk can be readily constructed.

This backlog of information could be a fruitful field for statistical studies of the effects of processing variables, if accurate service histories for the disks were also available.

The alloy specifications cover routine physical properties and soundness of the disk alloy. The infrequency of catastrophic failures indicates that the specifications essentially accomplish these purposes; however, there are other failure mechanisms being encountered in service for which current inspections are believed to be inadequate. These include thermal fatigue, stress-corrosion, embrittlement, and resistance to notches. Adequate tests can be readily devised for determining the resistance of the alloy to stress-corrosion, embrittlement, and notch properties, but the limits for specification purposes would have to be determined. In the case of thermal fatigue, considerable research is required to devise an adequate test method.

QUALITY CONTROL BY FABRICATOR

In making the disk, the fabricator must ensure that it meets the reliability standards of the engine producer. The specification ensuring reliability appears to be determined, at present, primarily by the engine producer. He is, of course, well aware of the reputation that he must uphold in providing a reliable product; from this consideration, he would be justified in requiring very closely controlled fabrication and inspection processes. On the other hand, the engine producer realizes that there are certain limits of control and inspection beyond which it is not reasonable to proceed, because of the effects on production ability and economy; he does not feel justified in requiring additional inspection unless he is certain of the significance of the added requirement to performance. The final inspection specifications are usually a compromise between what the engine producer would prefer and what the disk fabricator can reasonably be expected to perform. An example of disk fabrication inspection procedure is as follows:

(1) The rough forging of the billet is specified with regard to temperatures used, time at temperature, and amount of working of the material.

(2) The rough forging is heat-treated according to a given procedure, and straightening of the shaft is accomplished at this point, if applicable.

(3) Acceptance of the rough forging is made on the basis of one of several methods. Two of the methods in current use are:

(a) Samples of each forging are provided from which test specimens can be machined. Measurements are made of such short-time and long-time physical properties as ultimate strength, yield strength, ductility, hardness, stress-rupture strength, and notch-rupture strength.

(b) Samples for test specimens are obtained by cutting up typical forgings and testing as described in (a).

(4) The forging is rough-machined and inspected by X-ray, ultrasonic-reflection, and magnetic-particle methods.

(5) The forging is finish-machined and inspected by a penetrant-oil method, particularly in the bucket-fastening region.

(6) The disk is balanced, bucketed, and spin-tested at overspeed. Outside diameter is measured before and after the spin test, and the entire wheel is inspected by magnetic-particle and penetrant oil methods.

(7) Corrosion protections are applied.

For composite disks it is also necessary to include adequate specifications covering the welding operation, weld inspection, and weld repair.

Many of the necessary compromises in the specifications arise from an inadequate knowledge of what limits may be tolerated in the various inspection processes. Reliability of a disk inspected by a given procedure may therefore depend to a large extent on the arbitrary selection of these limits. In addition, the effectiveness of the inspection processes is greatly influenced by the degree of their sensitivity and by human error factors involved in carrying them out. The effect of human errors is partly nullified by the limited cross check that some of the inspections make on each other. Considerable research is required to improve the reliability of inspection procedures.

INFLUENCE OF MAINTENANCE ON DISK RELIABILITY

PRESENT PRACTICES

Current procedures for turbine-disk inspection include inspection:

(1) At scheduled periods at the operating base (In many cases, inspections are required at 25 and 100 hr on both overhauled and new engines.)

(2) At major overhaul

(3) After the engine operating limits of temperature or speed have been exceeded

The inspection performed at the operating base is generally only visual, and, as far as the disk is concerned, is severely limited for two reasons. First, the buckets are not removed; as a result, the serration areas cannot be properly examined. Second, the wheels are not removed from the engine; consequently, only the downstream face of the disk can be viewed. In multistage turbines, the forward stages cannot be examined at all. The shaft, including the journal areas, is not visible in any case. Visual examination of the accessible areas is not entirely satisfactory, since incipient failure conditions are apt to escape detection. Gross defects such as rubbing of adjacent parts or perhaps severe weld necking may, however, be discovered. In some cases, turbine-blade tip diameter is measured at the operating base.

If growth has occurred, it is generally presumed to have taken place in the buckets. In general, the operating base examination is focused much more on the buckets than on the disk.

Inspections performed at the time of major overhaul are much more complete. In addition to visual examination of the entire disk and the shaft areas, measurements are made to determine disk growth, and penetrant-oil and magnetic-particle inspections for cracks are performed. In some cases, the disks are again subjected to an overspeed spin test. The advisability of this practice has not been definitely established. As stated previously, the possibility exists that certain incipient failure conditions may be aggravated by the overstressing.

The criteria for an inspection made because of excessive turbine speed or temperature varies considerably from engine to engine. Such inspections are intended primarily as a protection for the turbine buckets and the other components that are directly in the hot gas stream. The disk itself is not in this stream, and hence is affected by large momentary overtemperatures only through delayed conduction from the bucket bases. Overtemperature of the disk itself is therefore unlikely to occur unless the overtemperature operation continues over a substantial period of time. The scheduled inspections are necessarily based on the needs of the turbine buckets and nozzle vanes and are conservative with respect to the disk. Examples of overtemperature operation for which inspections are specified for some engines are as follows:

(1) After a single hot start in which tailpipe temperature exceeds 2000° F for at least 30 seconds (ref. 15)

(2) After five hot starts during which tailpipe temperature exceeds 1500° F for at least 1 minute (ref. 16)

(3) After five 20-second periods during which tailpipe temperature exceeds 1320° F (ref. 17)

(4) After five periods in which tailpipe temperature reaches or exceeds 1600° F momentarily (ref. 17)

(5) Wheel is replaced if tailpipe temperature reaches 1832° F once or after 10 times at 1600° F (ref. 17)

In certain cases, if the operation is sufficiently severe, the engine is sent to overhaul without inspection. Examples of overspeed limits are 2 percent for one engine (ref. 16) and 4 percent for another (ref. 17).

SUGGESTIONS FOR IMPROVING DISK INSPECTIONS

The current incidence of disk failures indicates that more rigid inspections are desirable to improve reliability. The inspections must be practical, however, and efforts should be made during the design of an engine to provide maximum accessibility of the turbine disk. In addition, inspection procedures must be such that no damage will be incurred as a result of the inspection itself. This is a problem at the present time with certain disks in which the buckets are held in position by means of a pin. When the buckets are removed for inspection of the serrations, the pin is sheared; the sheared pin makes deep scratches in highly stressed areas of the disk. A redesign of the bucket retaining device is, of course, in order.

As stated previously, inspections are required after certain limits of time, temperature, or speed have been exceeded. It would be desirable to base the inspection requirement on the record of a speed-temperature-time recorder permanently attached to the engine. The pilot is not always aware of exceeding limits or he may misjudge the amount by which the limits have been exceeded.

Improved operational reliability may result if additional inspections are made:

(1) Inspect disk serrations whenever blades are removed at the operating base. This inspection should be made by the postemulsion penetrant-oil method. If cracks are found, the disk should be sent to overhaul.

(2) Portable hardness testing equipment is available and could be used at the operating base to determine the extent of structural deterioration. The specification ranges for hardness are generally quite broad; and if a disk is initially near one end of the range, it can have undergone considerable structural changes and still be within the specifications. To make the hardness a meaningful inspection tool, it would be desirable to establish the rim hardness of each new disk. This could be done by making hardness tests on the sides of several serrations. Hardness tests made during inspections can then be compared with the original hardness and changes evaluated. At

the present time, there does not appear to be sufficient data to judge what degree of hardness change is dangerous; however, it is known that increases in hardness tend to indicate greater notch sensitivity. Research may establish what change in hardness should require disk removal.

(3) A record should be kept of the diameter of the disk itself and the depth of the serrations to provide knowledge concerning possible growth. For a welded disk, scribe marks on either side of the weld could be used for additional dimensional checks. Necking in the web region should also be looked for.

(4) Microstructures of the disk could be determined by polishing and etching the sides of one or two serrations to detect dangerous metallurgical changes that may have taken place during engine operation.

ADDITIONAL INFORMATION NEEDED FOR IMPROVEMENT OF OPERATIONAL RELIABILITY

Information in the following areas may improve the reliability of turbine disks:

(1) Studies of material properties.

(a) Notch sensitivity: There is a scarcity of information about the sensitivity of the tensile and stress-rupture properties of disk alloys to notches. Some work in this field is currently underway.

(b) Thermal fatigue: The behavior of materials under conditions of thermal fatigue should be studied to determine the factors which influence thermal-fatigue life.

(c) Effects of combined shear and tensile stress for long times at high temperature: Extensive information is available on the stress-rupture properties of disk alloys. The serration region of a turbine disk, however, is subjected to a combination of tensile and shear stresses. A study to determine the effects of combinations of stresses approximating those in an engine would be of considerable value.

(d) The effect of variations in temperature on the stress-rupture life of disk alloys: In practice, the disk is subjected to brief periods at overtemperature conditions; the effects of these on the stress-rupture life may be important.

(2) Studies of design procedures: Information is needed on design methods that consider such factors as thermal fatigue and the stresses due to axial temperature gradients, particularly those in the serration region.

(3) Improved instrumentation.

(a) The development of a light, rugged flight recorder (time-speed-temperature) would be useful for scheduling inspections as well as for the accumulation of superior statistical data.

(b) A disk-growth indicating device would aid the pilot in avoiding web failures and weld failures.

(4) Alloy quality and inspection.

While there is little evidence that recent disk failures have been due to defective alloy, information on the service life of a batch of disks produced under drastically tightened quality control specification might indicate that substantial improvement can be made in this direction.

(5) Field inspection.

Inspection methods that will readily detect changes in alloys are urgently needed. Such methods could encompass hardness, microstructure, and electrical properties as possible inspection tools.

CONCLUDING REMARKS

Failure of a turbine disk can cause immediate destruction of an engine. Turbine disks are inspected in manufacture for internal and surface flaws and are usually given overspeed spin tests to check their short-time strength. Difficulties with turbine disks in service can develop through design defects, overheating, thermal cycling, and creep. The principal disk failure mechanisms are summarized in figure 22 and table I.

Design defects can best be eliminated in an adequate developmental program on a specific engine type.

Overheating can be caused by failure of the disk-cooling system, improper installation of the system, or excessive combustion temperatures. Overheating in combination with the stresses in the disk caused by centrifugal force can cause excessive growth of the disk rim. Because rapid and catastrophic failure of disks can occur by severe overtemperature, a device for warning of overtemperature of the disk is desirable. Possible devices that might be developed are radiation thermocouples that scan the disk rim, elec-

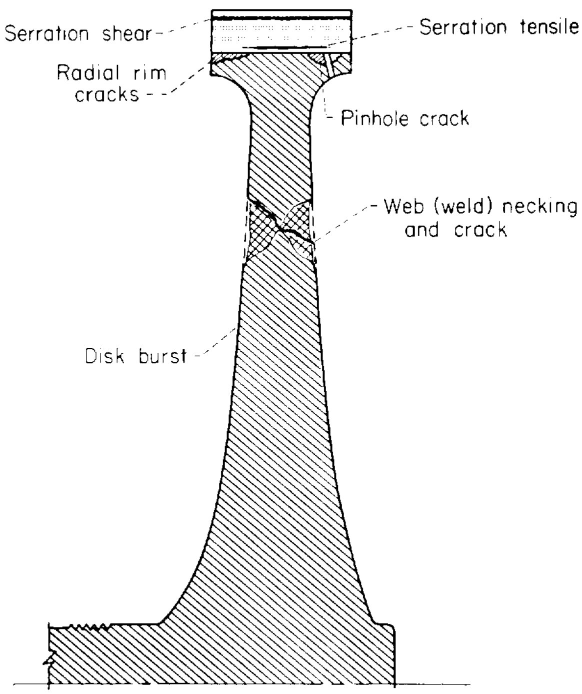


FIGURE 22.—Locations of disk failures.

trical contact devices that warn of excessive disk growth, or a device to detect a reduction in cooling airflow.

Thermal cycling associated with starting, accelerating, and stopping of the engine causes plastic working of the disk material and eventually produces cracks in the disk rim. These cracks may originate in scratches caused by shearing of the bucket retaining pin during removal of the bucket from the disk. In most cases, these cracks progress slowly enough that inspection can be delayed to a suitable inspection period. Inspection of the disk for cracks at the base of the serrated area can be made when buckets are replaced or during a scheduled inspection period.

The centrifugal force exerted by the buckets may eventually cause shearing of the fir-tree serrations in the disk or stress-rupture failure across the fir-tree segment. Very few failures of the shear type have occurred; however, numerous stress-rupture failures are noted in the literature. There is no fundamental reason why the serration segment cannot be designed for long life. The short stress-rupture life that has been encountered in some engines was suspected to be caused either by overtemperature of the disk rim, embrittlement induced by overtemperature operation, or undesirable metallurgical practices in manufacture.

In commercial transport service, much longer disk operating life will be required than in military service; time-dependent failures may then be more frequent. Difficulties are often indicated by deformation or cracking of the serration sector, but an accurate indication of incipient failure is not always available. Failure of one serrated sector results in the loss of two blades and sometimes unbalances the disk sufficiently to be catastrophic.

Further, the failure of a serration segment usually triggers the failure of additional segments: generally, more than two fail at one time, and as many as seventeen have been known to fail at the same time. The unbalance caused by failures of this kind may destroy an engine. In a multistage turbine, failed parts will probably destroy the downstream stages and stop or destroy the engine. The possibility of failures of this type can be minimized by a replacement schedule set up by the user with the advice of the manufacturer. The replacement schedule should make adequate allowance for the scatterband in disk properties and for unusual operating conditions.

In summarizing, the following practice in service with regard to the turbine disk is suggested:

(1) Schedule inspections for radial and tangential rim cracks, rim growth, deformation of the serration region, and evidences of embrittlement. The inspection interval should be based on both operating time and the number of starts and stops to which the engine is subjected.

(2) Inspect disk rim for cracks whenever a bucket is replaced.

(3) Provide a device warning of overtemperature of the disk.

(4) Schedule replacement time for the disk, based on an estimate of its life, with an adequate factor of safety for the scatterband and other uncertainties.

TABLE I. SUMMARY OF IMPORTANT DISK FAILURE MECHANISMS.

Failure type	Cause	Rate of propagation	Inspection to determine incipient failure	Suggested frequency for inspection	Special warning measures	Design remedy
Serration shear	Shear-rupture	Time dependent	Determine tooth distortion	At blade replacement	Time-temperature-rpm recorder	(1) Decrease stress or temperature (2) Controls to avoid overtemperature and overspeed (3) Reduce stress concentration for serration tensile
Serration tensile	(a) Stress-rupture (b) Material embrittlement	Time dependent	Surface inspection for cracks			
Radial rim cracks	Thermal fatigue	Cycle dependent	Surface inspection for cracks	At blade replacement	Log of number of starts	(1) Improved rim cooling (2) Reduce temperature gradient
Pinhole crack	(a) Stress concentration (b) Stress corrosion (c) Thermal fatigue	Cycle dependent	Surface inspection for cracks	At blade replacement	Log of number of starts	Use other method for retaining blades
Web (weld) necking and crack	(1) Defective weld (2) Overtemperature	Probably time dependent	(a) Dimensional (b) Weld inspection	After evidence of overtemperature	Time-temperature-rpm recorder	Improved inspection in manufacture
Disk burst	Defective disk	Instantaneous				

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CHAPTER IX

ROLLING-CONTACT BEARINGS

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SUMMARY

The various types of bearing failure, the mechanisms of failure of these various types, and possible methods of improving bearing reliability are discussed. Fatigue life, which is of primary importance to ball thrust bearings, can be increased by reducing load $\left(life \propto \frac{1}{load^3}\right)$. *At present, it appears necessary to schedule bearing replacements on the basis of expected fatigue life. An excessive wear rate indicates the need for redesign. Under conditions of normal wear, bearing replacement can be based on measured clearances in the bearing at inspection periods. For extreme boundary lubrication failures, when excessive metal transfer occurs at the sliding surfaces, material change and redesign are indicated.*

Bearing reliability can be improved by (1) additional information, (2) better failure detection devices, and (3) better maintenance practice. Bearing temperature acceleration seems to be a parameter that can be used to warn of imminent bearing failure.

Improvements in manufacturing standards and in quality control should result in improved bearing reliability.

INTRODUCTION

One of the factors related directly to reliability of the aircraft gas-turbine engine is the reliability of the engine bearings. The influence of bearing failures on engine reliability, the factors that influence bearing reliability, and measures that may improve bearing reliability are discussed in this chapter.

Most bearings of aircraft gas-turbine engines are of the rolling-contact type, and, therefore, this report discusses only this type. Typical examples of rolling-contact bearings showing the various components are presented in figure 1; typical turbine-engine bearing arrangements are shown in figure 2. Rolling-contact bearings are preferred (ref. 1) over hydrodynamic (sleeve) bearings because they

(1) Require less starting torque

(2) Are much less sensitive to oil-flow interruptions

(3) Need less oil flow

(4) Impose a lower cooling load

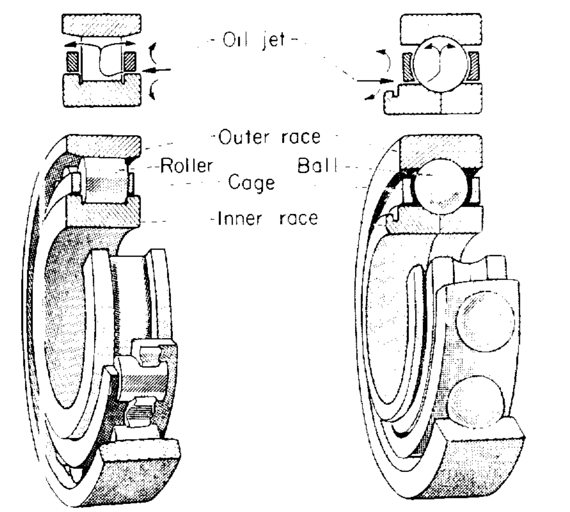
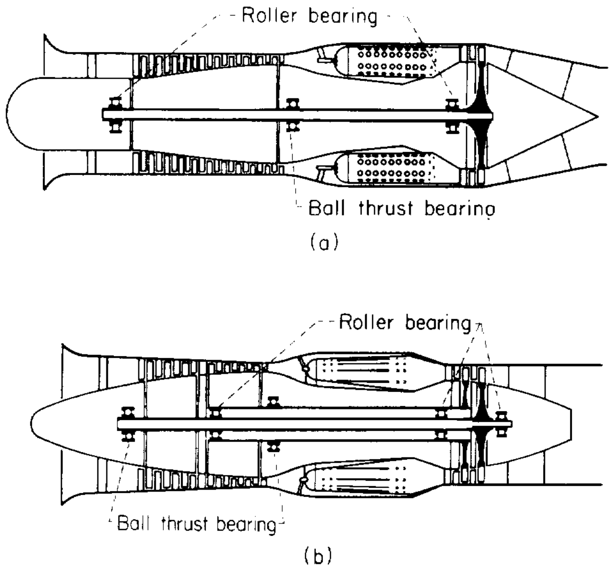


FIGURE 1.—Typical high-speed bearings for turbine engines.

The frequency and the severity of engine damage caused by bearing failures are discussed with respect to the available turbojet-engine bearing-failure data. Bearing reliability is affected by (1) bearing quality, (2) severity of application, and (3) type of care and handling a bearing gets throughout its life. Factors that determine bearing quality include material quality control, material inspection methods, tolerances, and dimen-



(a) Single-rotor engine.

(b) Two-spool engine (from ref. 1, appendix C).

FIGURE 2.—Illustrative turbine-engine bearing arrangements.

sional inspection methods. Factors relevant to severity of application include operating conditions such as speed, temperature, amount of lubrication at starting, and continuity of oil flow. Factors of importance to care and handling include overhaul and maintenance practices such as inspection methods as well as bearing replacement and removal practices. These factors are discussed herein.

BEARING FAILURES IN TURBOJET ENGINES

Analysis was made of U.S. Air Force Disassembly Inspection Report (DIR's) (for Aug. to Nov., 1953) on overhaul of 769 turbojet engines. The statistical data resulting from the analysis (discussed in considerable detail in ch. II) show considerable variation in bearing difficulty between different engine types, different models of the same engine, and even in different installations of the same engine model. The variation is illustrated by the following statistics on four engine types. Of these engines, the percentage that had main bearings replaced at overhaul varied from 24 for one engine type to 60 for another engine type; replacement of accessory bearings varied from 2 to 64 percent.

Of the same engines, those sent to overhaul because of main-bearing failures varied from 0 to 7 percent for the several types. While the number is low, the overhaul life of the average engine was also quite low (115 to 180 hr). Longer running time may have considerable effect on main-bearing failures, as is discussed later.

Bearing failure frequencies for eight groups of engines (identified by code) are shown in table I. As noted on this table, both primary and total bearing failure columns include main and accessory bearings.

Further statistics on reasons for sending J33 and J35 engines to overhaul are given in ref-

TABLE I.—BEARING FAILURE FREQUENCY DATA FOR VARIOUS ENGINES

[Based on USAF Disassembly Inspection Reports.]

Engine	Number of engines	Maximum running time, hr	Percent of engines in which bearing failures were found ([a])	
			Primary	Total (secondary + primary)
C-6	39	650	18	92
C-7	211	885	8	64
B-3	73	634	0	66
B-4	32	493	6	59
B-7	87	165	9	72
B-8	93	390	6	17
B-9	75	447	12	36
B-10	159	301	4	30

[a] Includes main and accessory bearings.

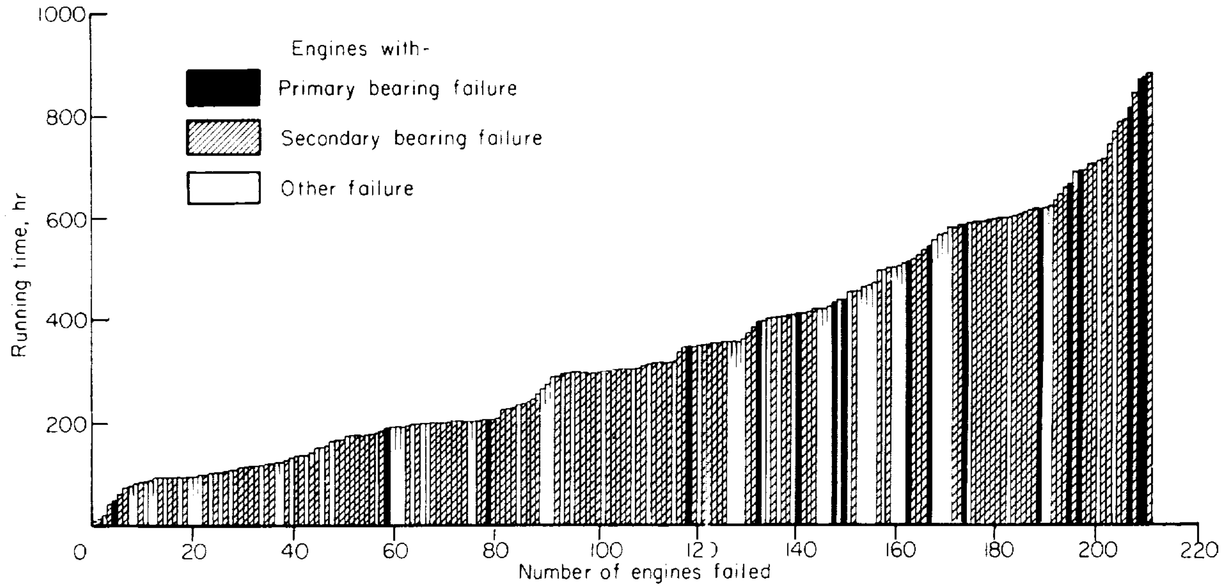


FIGURE 3.—Bearing-failure data for group of 211 engines (code, C-7). Data from USAF Disassembly Inspection Reports (Aug. to Nov., 1953).

erence 2; these data show that bearings have been an important reason.

EFFECT OF RUNNING TIME ON BEARING RELIABILITY

Engine bearing failures sustained by 211 engines (code, C-7) are shown by failure type in the bar graph of figure 3. All engines were new prior to the service shown, so that all bearings were new at indicated zero running time. The data of figure 3 are shown plotted in figure 4 as primary and total (primary plus secondary) failures against engine running time; these terms

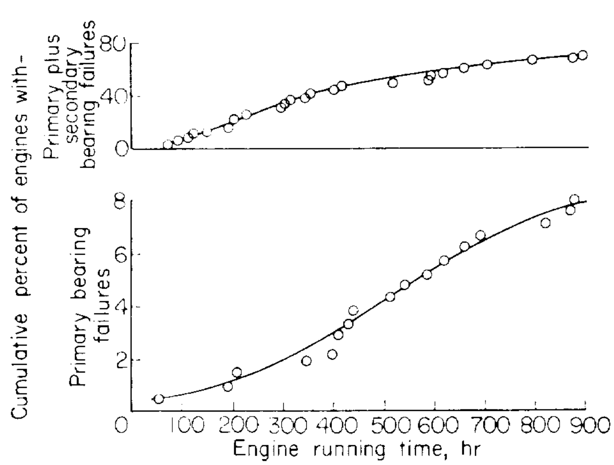


FIGURE 4.—Bearing failures found in overhaul of 211 engines (code, C-7). Data from USAF Disassembly Inspection Reports (Aug. to Nov., 1953).

are defined as follows: (1) Primary bearing failures are those that caused the engine to be sent to overhaul; (2) secondary bearing failures are bearing conditions found at overhaul that are judged to require bearing replacement. (Such conditions are not, however, the primary reason for overhaul of the engine.)

The curves of figure 4 show that the total of primary plus secondary failures is quite high for engines with long running times.

Because of weaknesses in the data, no further statistical analysis of these data is made. These weaknesses are:

(1) Bearing history is not always known with certainty; bearings may be replaced in the field without the replacement being noted on the engine record.

(2) The most serious bearing failures (that is, those that cause destruction of an engine and a crash) would not appear in the DIR data. DIR's are written only for engines that can be overhauled.

(3) Engine failures occur from a variety of causes. It is difficult to isolate bearing statistics for analysis, and when bearing failures are separated, the resulting sample size is small.

Besides these weaknesses in data for new engines, an additional weakness exists in similar data for overhauled engines; frequently the data do not account for the many bearings replaced at previous overhauls. Removal of lightly damaged bearings undoubtedly prevents many primary bearing failures. Because of the gravity of accidents caused by bearing failures and because of the low bearing cost relative to overhaul cost, present practice is to replace bearings at the slightest evidence of deterioration. Hence, a large percentage of bearings are replaced at overhaul.

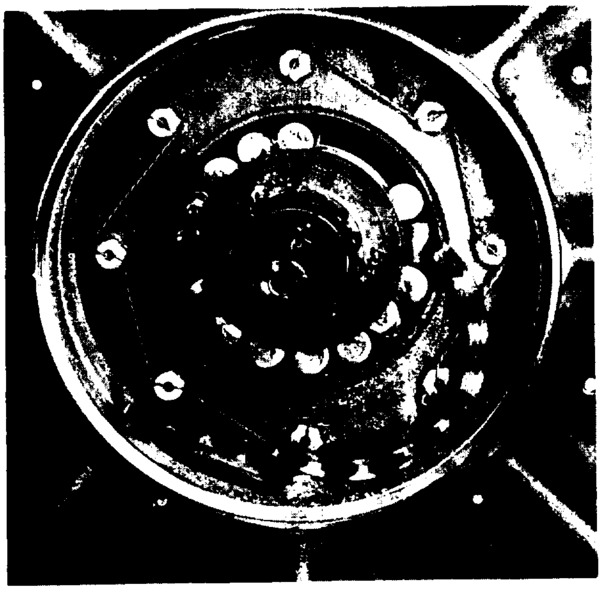


FIGURE 5.—Bearing failure resulting from oil interruption. Engine with jet lubrication of bearings. Photograph shows cage has failed completely. Failure occurred 17 minutes after oil interruption (from ref. 1, appendix B). Other photographs of this engine in figures 6 and 7.

EFFECT OF BEARING FAILURES ON ENGINE FAILURE

Figures 5 to 7 (taken from ref. 1, appendix B) show engine damage that occurred when bearing failures progressed beyond the initial stage. These photographs are of an engine tested to determine the time an engine can run after complete stoppage of the oil supply. Two engines run in the tests described in appendix B of reference 1 ran 17 and 26 minutes before bearing failure and subsequent engine failure. The engines were completely destroyed, and the test stands in which they were mounted were damaged.

BEARING-FAILURE TYPES

Both the exact nature of a bearing failure and the symptoms that accompany it vary widely with the type and the stage of the failure. When a

FIGURE 6.—Compressor damage resulting from bearing failure in oil-interruption test (from ref. 1, appendix B).

bearing does not function as well as when new, but still allows the mechanism of which it is a component to operate satisfactorily, the terms "early failure" or "incipient failure" can be used. The presence of a fatigue pit on a race or a rolling element, or the transfer of cage material to the cage-locating surface, or heavy cage wear would be classed as incipient failures. When a bearing seizes or disintegrates, the failure is total. In the case of a total bearing failure, wreckage of the engine may occur.

Bearing failures may be divided into two general groups: time-dependent and time-independent. The time-dependent failure types include:

(1) Fatigue

(2) Wear

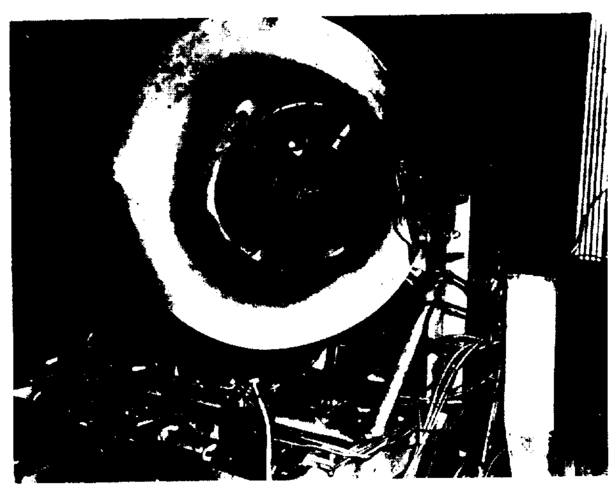


FIGURE 7.—Engine torn from test mount as result of engine stoppage by bearing failure in oil-interruption tests (from ref. 1, appendix B).

The time-independent failure types include:
(1) Extreme boundary lubrication
(2) Brinelling
(3) Misalinement

Corrosion is a special failure type; it can be either time-dependent or time-independent depending on circumstances. For gas-turbine engines, corrosion has not been a major difficulty. Corrosion damage may result from presence of water or corrosive materials, such as the oxidation products of the lubricant. Corrosion of the bearing materials results in surface pitting that can accelerate bearing failure.

Failure of bearings because of dirt is also time-independent. In particular, precision bearings are extremely sensitive to dirt and other contaminants. NACA inspection of a small sample of condemned bearings from an Air Force overhaul base showed that dirt was a major cause of bearing damage; the damage appeared as scratched rolling elements and races. For this sample of bearings, however, the average running time was low.

Time-dependent failures alone would probably follow an approximately normal or "wear-out" (Gaussian) distribution. The time-independent failures, which can be influenced by environment, might be expected to follow a chance law. Poor designs, materials, or assembly can accelerate time-independent as well as time-dependent failures.

TIME-DEPENDENT FAILURES

Fatigue.—Fatigue life of a bearing is defined as the number of revolutions that it makes before a fatigue pit first develops in the bearing material. Bearing-life ratings are based on the 10-percent-failure (90-percent-survival) point. A fatigue-failure distribution curve for a group of ball bearings is shown in figure 8 (data from ref. 3). Fatigue-failure distributions for other groups of rolling-contact bearings will be similar to this curve.

Fatigue in a rolling-contact bearing appears as a pit or spalled area in the track of either race or on a rolling element. A typical fatigue pit is also shown in figure 8. In most ball bearing designs the inner race is most susceptible to fatigue because an elementary volume of material in this race is stressed at a higher frequency

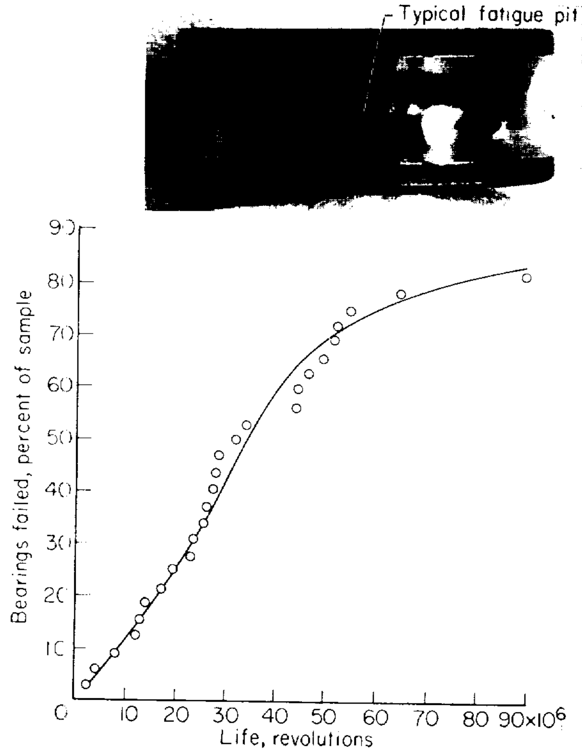
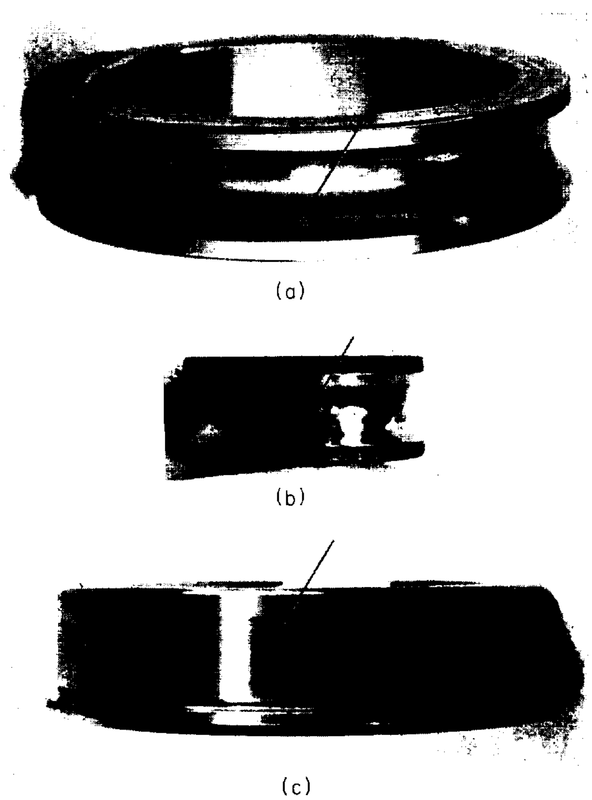


FIGURE 8.—Fatigue failure curve for bearings (6309 size) of SAE 52100 steel. Radial load, 4240 pounds; speed, 1500 rpm; grease lubrication at room temperature (data from ref. 3).

than are similar elements of the outer race or of a ball or roller.

For practical loads, ball thrust bearings do not have an endurance limit; this is in contrast to other types of fatigue where endurance limits exist. Even if all other failure modes were eliminated, fatigue failures would still occur. There is, however, an inverse cubic relation between load and life $\left(\text{life} \propto \frac{1}{\text{load}^3}\right)$ so that life increases markedly with a relatively small decrease in load.

In a roller bearing there is essentially line contact between roller and race, and in a ball bearing there is essentially point contact between ball and race. Hence, for the same bearing load, the ball bearing has a higher contact stress. Because of the line contact and the small external loads on roller bearings in turbine engines, the contact stresses are low and fatigue life should be quite long. Hence, fatigue should not be a major problem in roller bearings. The point



(a) Early stage of flaking on inner raceway of deep-groove ball bearing caused by fatigue.

(b) Later stage of flaking on inner raceway of deep-groove ball bearing caused by fatigue.

(c) Early stage of flaking on inner raceway of cylindrical roller bearing caused by fatigue.

FIGURE 9.—Typical rolling-contact-bearing fatigue failures (from ref. 15).

contact and high external loads of ball bearings in turbine engines, however, produce contact stresses that are high, and fatigue life can become of major importance. For example, a typical turbojet engine roller bearing (75-mm bore, 10,000 rpm, and 1000-lb radial load) would have a theoretical fatigue life of 10,000 hours. A typical turbojet engine ball bearing (110-mm bore, 10,000 rpm, and 5000-lb thrust load) would have a theoretical fatigue life of 110 hours. Fatigue life will probably be shorter with the newer engines because of the trend to higher thrust loads (ref. 1, appendix C).

Fatigue failures in various stages are shown in figure 9. While a fatigue failure is not always catastrophic, there is very little information on the time from first appearance of fatigue failure, as indicated by pits such as shown in figure 9(a), and final failure. Until more information is available, replacement must be scheduled on the basis of curves such as that of figure 8.

Wear.—The wear of a rolling-contact bearing seems to be influenced strongly by speed and only slightly by load. Excessive wear may be caused by poor bearing design, by use of an incompatible material combination, or by inadequate lubrication (which may be caused by poor design).

Under normal conditions, little wear of the races and rolling elements of rolling-contact bearings occurs; wear is light primarily because of low sliding velocities. The critical wear areas in a rolling-contact bearing are the cage-locating surface and the cage pockets (refs. 1 and 4); at these locations the surfaces are in pure sliding motion, usually at high sliding velocities.

The cage-locating surfaces are, in reality, plain journal bearings of very small length in comparison to their diameter (ref. 1, appendix A); journal bearings of low length-to-diameter ratio have very low load capacities. In consequence, the cage-locating surfaces are subjected to boundary lubrication conditions at all times; occasionally they are subjected to extreme boundary lubrication conditions. The factors involved are shown in the curves and sketches of figure 10. There are plotted, for a typical journal bearing,

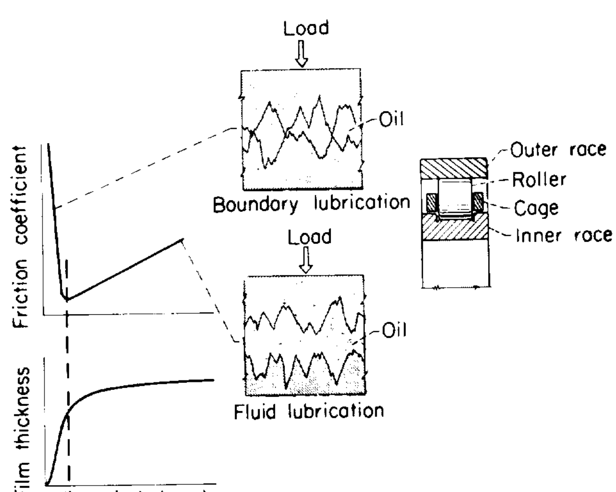


FIGURE 10.—Boundary and fluid lubrication.

friction coefficient and film thickness against the parameter ZN/P, which is viscosity multiplied by speed and divided by load. Boundary lubrication exists to the left of the vertical line and hydrodynamic or thick-film lubrication to the right. In hydrodynamic lubrication, the fluid film is thick enough that no surface contact takes place, as is shown in the lower sketch. In boundary lubrication, the lubricant film is so thin that the surface asperities make contact through the film, as shown in the upper sketch. Film thicknesses for boundary and for hydrodynamic conditions are shown in the lower curve. Thus, in hydrodynamic lubrication the load is supported on the lubricant film and the properties of the lubricant are important; in boundary lubrication, the load is supported primarily by solid contacts and the properties of the solids are important in determining the wear and friction characteristics.

Thus, for the components in which sliding takes place, it is important to use material combinations with good sliding friction characteristics, that is, with good "antiweld" properties (ref. 5). Good antiweld properties are defined as the ability to wear without violent welding and adhesion at the surfaces under boundary lubrication conditions. It is possible, however, that even materials which have adequate antiweld properties under boundary lubrication conditions may show excessive welding and adhesion under extreme boundary-lubrication conditions.

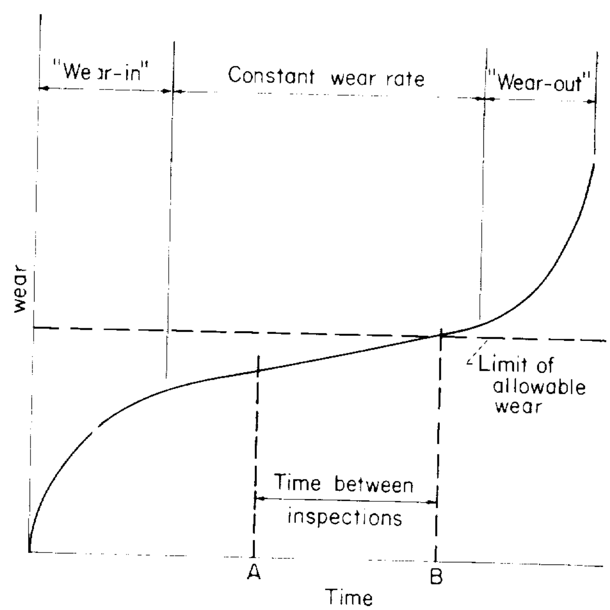
Under abnormal conditions, wear of the races and rolling elements can become excessive. For example, excessive cage slip can occur when bearings are operated at high speeds and light loads (ref. 6); roller or ball skidding is also possible under these conditions. Since the radial loads are usually small in the gas-turbine engine, the bearings carrying only radial loads are sometimes subject to these conditions.

Data showing that cage slip in high-speed roller bearings produces severe roller and cage wear are included in reference 7. The effect of rolling-element wear would be to increase bearing radial clearance. In an engine, this increase would allow radial displacement of the shaft, which could lead to serious engine damage and possible stoppage. There is also the chance that excessive rolling-element wear can cause bearing failure. Excessive wear also allows wear particles to accumulate in the lubricant and to be circulated throughout the engine; these particles can cause damage to other bearings, splines, or gears.

Presence of wear particles in the oil and oil filter might be used as an indication of excessive wear rate in the bearings.

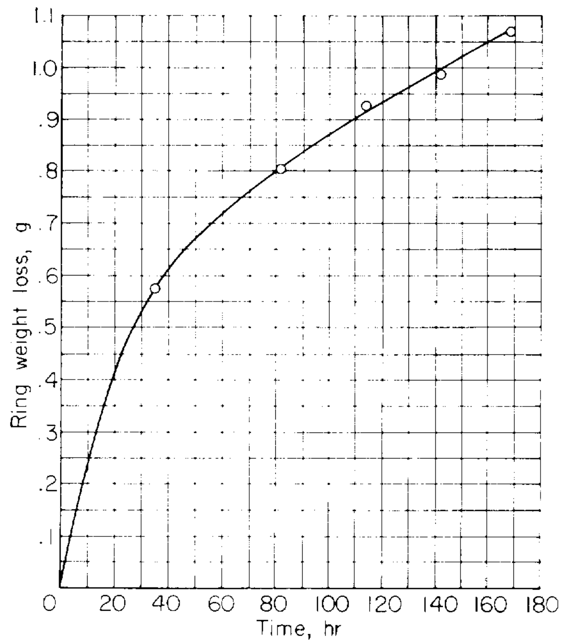
Since the major wear difficulties occur at the cage-locating surface, corrective measures involve improving the conditions at this location. Redesign and material change can improve these conditions. Oil flow can be improved or cage contact stresses decreased by redesign. Proper choice of materials can reduce wear by reducing welding and adhesion at the contacting surfaces.

While wear is a complex phenomenon, a simplified illustrative wear curve for a pair of rubbing surfaces is shown in figure 11(a) to illustrate a number of points. Under ordinary conditions, the initial operation of wearing surfaces can be termed a "wear-in" or "run-in" period. After the wear-in period is complete, a constant wear rate obtains for a period of time. At some later time, the wear rate increases rapidly and the period following the change in rate is termed "wear-out." Also included in figure



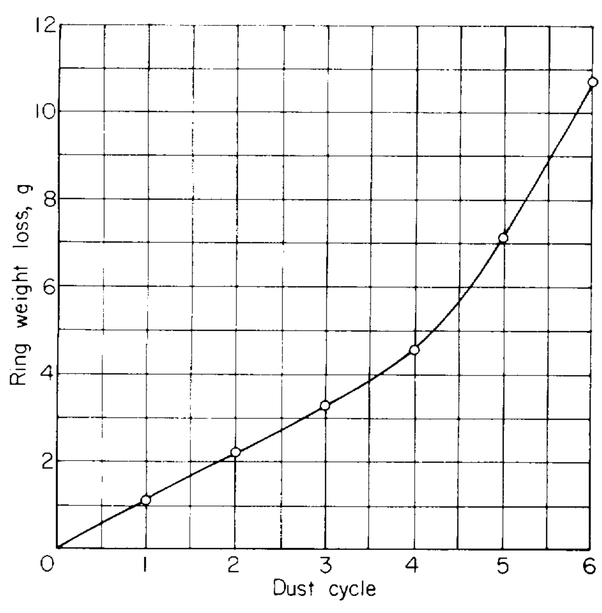
(a) Illustrative wear curve showing three periods.

FIGURE 11.—Illustrative and actual curves of wear plotted against time.



(b) Wear of nitrided piston rings in tests at high power output; this curve shows the "wear-in" and "constant wear rate" periods (data from ref. 20).

FIGURE 11.—Continued. Illustrative and actual curves of wear plotted against time.



(c) Wear of nitrided piston rings under abrasive conditions (dust tests); this curve shows "constant wear rate" and "wear-out" periods (data from ref. 20).

FIGURE 11.—Concluded. Illustrative and actual curves of wear plotted against time.

11(a) for illustrative purposes is a line representing a limit of allowable wear. The limit of allowable wear occurs at time B. In scheduling bearing replacement, account must be taken of the wear that will occur between inspections, so that bearing replacement must be made at or before time A in figure 11(a). This will ensure that the limit of allowable wear will not be reached between inspections.

Actual wear curves for reciprocating-engine piston rings are included in figures 11(b) and (c) to illustrate the various stages of wear. Because many factors (such as temperature, speed, load, and lubrication conditions) influence the time at which wear-out begins, prediction of bearing wear must be based on experience with the bearings in a specific application.

It is recognized that it may be difficult to base bearing replacement on wear rate because of the following reasons:

(1) Critical wear may be occurring at some point inaccessible without complete disassembly of the bearing.

(2) An establishment of bearing wear rate is dependent on an exact knowledge of the original bearing clearances because tolerances on cage diametral clearance, for example, are fairly liberal.

(3) There may be difficulties in determining the limit of allowable wear.

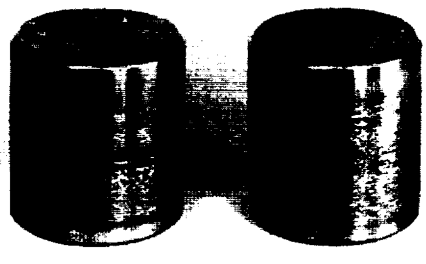
TIME-INDEPENDENT FAILURES

Extreme boundary lubrication.—Under conditions of extreme boundary lubrication, average oil-film thickness is even less than in boundary lubrication (fig. 10); metal-to-metal contact is therefore more severe, and severe surface damage more likely. This surface damage appears as violent welding and adhesion, as well as excessive wear.

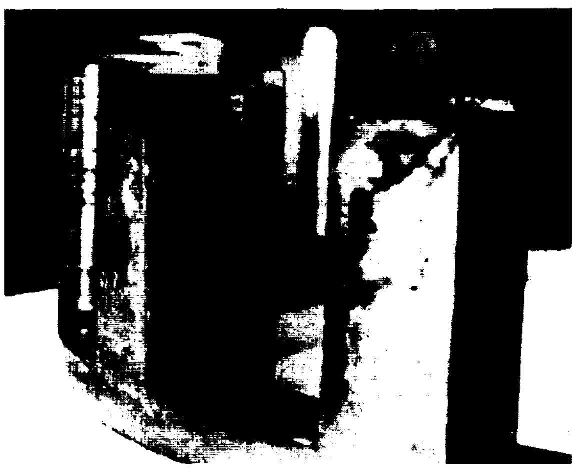
The conditions leading to extreme boundary lubrication are those which decrease the value of the parameter ZN/P. These conditions are as follows:

(1) High loads
(2) High temperatures
(3) Low viscosity
(4) Lack of adequate lubricant supply

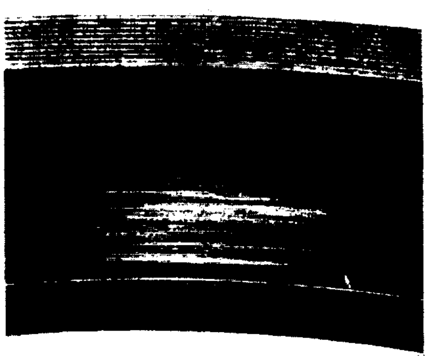
A typical example of smearing of the cage material on the rollers, the outer race, and the cage pockets that resulted from inadequate lubrication is shown in figure 12. Extreme boundary lubrication may result in either heavy cage wear or in excessive metal transfer between the cage and the surfaces against which the cage rubs (its locating race and the rolling elements).



(a)



(b)



(c)

(a) Smearing on cylindrical rollers.

(b) Smearing on cage pockets.

(c) Smearing on outer raceway.

FIGURE 12.—Typical cylindrical-roller-bearing failure caused by inadequate lubrication between rollers and cage pockets (from ref. 15).

Brinelling.—True brinelling or denting of the races is produced by pressure or impact and results in an indentation because of plastic flow. Loads above the static capacity of the bearing (the load above which plastic flow occurs) must be applied to produce true brinelling. Unless the bearing is improperly chosen or handled, true brinelling will not occur.

"False brinelling" is a special case of fretting (fretting corrosion, friction oxidation). Fretting is the surface failure that occurs when closely fitting metal surfaces (such as a rolling element and race) experience slight relative motion over long periods of time. False brinelling of engine bearings usually occurs when the engine is subjected to vibration during shipping or when

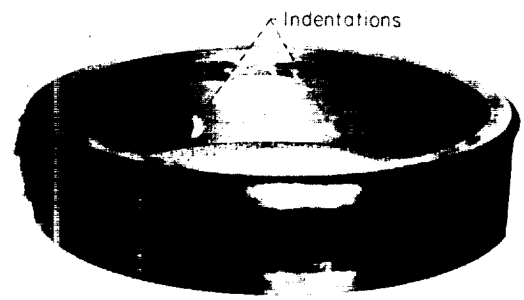


FIGURE 13.—False brinelling of outer race of double-row self-alining ball bearing caused by vibration without bearing rotation (from ref. 15).

standing on a vibrating platform. Such damage can usually be prevented by eliminating or appreciably decreasing the vibration at the bearings. This type of damage can take place under loads far below the static-load capacity of the surfaces. False brinelling in a double-row, self-alining ball bearing is shown in figure 13.

Misalinement.—Bearing races may be misalined with each other because of (1) poor assembly, (2) manufacturing deviations, or (3) distortion difficulties. These may be caused, respectively, by (1) improper mounting because of dirt; (2) out-of-square shaft shoulders or housings; and (3) mechanical and thermal stresses in the shaft, bearing housing, or engine frame.

Misalinement may cause several types of bearing failure. Parts of a roller bearing that failed because of misalinement are shown in figure 14; this bearing shows evidence that misalinement

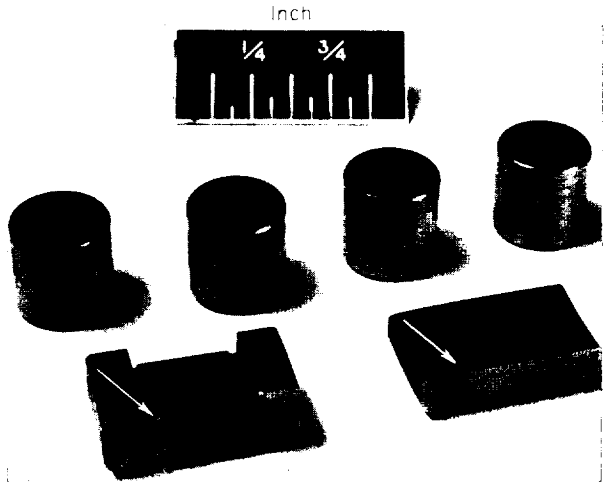


FIGURE 14.—Cylindrical roller bearing that failed because of misalined load.

altered the load from a fairly uniform distribution across the length of the rollers to a very heavy load near one end of the rollers. This caused premature surface failure of both races in the plane of the magnified load. Both races are flaked in this region.

The time between incipient failure and total failure, from misalinement causes, is probably long enough to permit total failures of this type to be forestalled by scheduled inspections. Vibration measurements may possibly be used to detect failures of this type.

OPERATING PROBLEMS INFLUENCING RELIABILITY

There are a number of operating conditions that strongly influence bearing reliability. These conditions and a number of suggested checks to make on the engines (and bearings) in aircraft are discussed in the following sections.

HIGH SPEEDS

Operation at the high rotative speeds of aircraft gas-turbine engines imposes a general lubrication problem in the use of rolling-contact bearings. Bearings in current engines run at DN values (bearing bore in mm times shaft speed in rpm) up to about 1.5 million. Few operational data are available at higher speeds, although some very limited experimental data are available at DN values up to 2.2 million (ref. 7). Adequate lubrication at the cage-locating surface, necessary because of the high sliding velocity, is difficult to achieve under the operating conditions imposed on aircraft gas-turbine bearings, and is one of the important reasons for the high replacement rate of some engine bearings. Centrifugal forces acting on the oil within the bearing tend to throw it radially outward, away from the cage-locating surface in bearings equipped with inner-race-riding cages.

HIGH TEMPERATURES

In engines of old design in present operation, maximum bearing temperature is approximately 350° F (ref. 5). At this temperature, SAE 52100 steel and silver-plated bronze are generally the bearing and cage materials, respectively. In engines of more recent design, bearing temperatures have increased to 500° F (ref. 5). New materials are necessary and tool steels or intermediate high-temperature steels are being utilized. Silver-plated bronze is still used as the cage material. These materials may not be completely satisfactory at 500° F. Research is in progress on both race and cage materials.

For engines of future design, bearing temperatures as high as 750° F are anticipated (ref. 5). For both the 500° and 750° F temperature levels, the molybdenum tool steels are being studied. The molybdenum tool steels have adequate hardness and dimensional stability at these temperatures. Molybdenum tool steels produced by ordinary melting methods have not, in general, shown adequate fatigue life (ref. 8). It is shown in reference 8 that it is necessary to use vacuum-melted tool steels to obtain a load capacity (or life) equivalent to that of SAE 52100 bearings. The data of reference 8 are summarized in table II. The bearings made of consumable electrode vacuum-melted M–50 alloy had a load capacity greater than standard SAE 52100 bearings at room temperature and a load capacity (50-percent failure point) at 450° F equivalent to standard SAE 52100 bearings at room temperature (an AFBMA load rating of 100 percent). It must be borne in mind that commercial grades of the standard bearing steel, SAE 52100, are satisfactory only because of many years of metallurgical development. Figure 15 shows the marked improvement in fatigue life of SAE 52100 steel bearings made during a 24-year period. During this period, metallurgical and

TABLE II. LOAD CAPACITIES FOR BEARINGS FABRICATED FROM TOOL STEELS

[Data from ref. 8.]

Melting practice	Material	Temperature	Load capacity, percent AFBMA basic load rating (a): 10-Percent failure point	Load capacity, percent AFBMA basic load rating (a): 50-Percent failure point
Basic arc	M-10	Room	77	70
	M-10	450° F	64	64
	M-1	Room	83	91
	M-1	Room	76	78
	M-1	450° F	60	54
Induction furnace vacuum melted	M-1	Room	132	122
	M-1	450° F	64	77
Consumable electrode vacuum melted	M-50	Room	144	170
	M-50	450° F	93	100

a AFBMA (Anti-Friction Bearing Manufacturers Association) basic load rating for SAE 52100 bearings is 100 percent.

manufacturing improvements resulted in a tenfold increase in fatigue life. It appears that the M-type tool steels, with refinements in melting techniques, will produce bearings with satisfactory fatigue life, but the bearings will cost more because of increased material costs and machining times (appendix D, ref. 1).

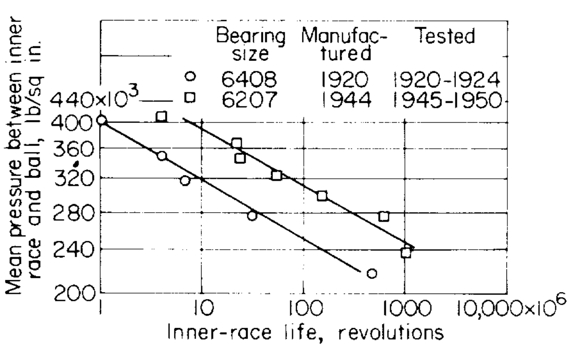


FIGURE 15.—Improvement in fatigue life of SAE 52100 steel bearings from 1920 to 1944 (from ref. 9).

Most cage materials now in use have satisfactory strength at present operating temperatures but may not have at expected temperatures (ref. 1); ferrous or nickel alloys may be necessary at the higher temperature levels.

Because of their high volatility, mineral oil lubricants do not appear promising as lubricants for future high-temperature bearings (refs. 5 and 9). Low-temperature lubrication (−65° F) requires low-viscosity lubricants; high-temperature lubrication requires low-volatility lubricants. Petroleum oils of low viscosity evaporate excessively fast at the bearing temperatures expected in future engines. The petroleum and chemical industries are developing synthetic lubricants for use at high operating temperatures; synthetic lubricants are now available that have acceptably low volatility at bearing temperatures of 500° F and also meet the low-temperature starting requirement. Properties of some synthetic lubricants are listed in table III; military specification (MIL-L-7808) requirements as well as properties of a petroleum lubricant are also listed for comparison. The petroleum and the diester are used in current jet engines. It appears unlikely that a liquid lubricant can be devised that will be satisfactory at bulk lubricant temperatures much above 500° F (refs. 5 and 9). For subsonic aircraft, bulk lubricant temperatures are expected to be lower than 400° F; for supersonic aircraft, bulk lubricant temperatures are expected to exceed 400° F. The alternatives to developing a liquid lubricant with the required stability at high temperature are (1) provision of greater bearing cooling by means other than the lubricant; (2) use of a solid lubricant (ref. 10); (3) use of a gaseous lubricant (ref. 11); or (4) use of a gas bearing, such as an externally pressurized air bearing (ref. 5). Each of these alternatives undoubtedly creates problems that have not as yet been completely evaluated.

TABLE III.—PROPERTIES OF TYPICAL SYNTHETIC LUBRICANTS

Fluid	Viscosity, centistokes, at—				A. S. T. M. pour point, ° F	C. O. C. flash point, ° F	C. O. C. fire point, ° F
	−65° F	−40° F	100° F	210° F			
MIL–L–7808 Specification	13, 000 (max)	----	----	3. 0 (min)	−75	385	----
Petroleum: [a]							
MIL–O–6081A (grade 1010)	40, 000	----	9. 95	2. 47	−70	300	----
Diesters:							
Di(2-ethylhexyl)sebacate	8297	----	12. 78	3. 32	−70	440	472
Di(2-ethylhexyl)sebacate plus additives	16, 000	2700	20. 8	5. 3	−75	450	475
Polyalkylene glycols (designated by viscosity at 100° F):							
Water soluble, 8.9 centistokes	--------	1800	8. 9	2. 4	−85	260	285
Water insoluble, 7.4 centistokes	4587	905	8	2. 52	−70	270	335

[a] For comparison purposes.

LACK OF LUBRICATION AT STARTING

When an engine is started, the bearings are, for a time, lubricated only by the lubricant left in the bearing from the previous operation. In present engines, "soak-back" of heat from the rotor evaporates and bakes any lubricant left on the bearings at shutdown (ref. 12, p. 184); in consequence, lubrication at starting is almost nonexistent. This phenomenon has caused a number of bearing failures. It was found (ref. 12) that silver-plated cages are less subject to this type of failure. The use of lower volatility lubricants, such as the synthetics, has also improved bearing lubrication during engine starting where operating and soak-back temperatures are not too high; newer engines, however, have operating temperatures so high that the dry bearings may still be a problem (ref. 5).

A number of modifications to engine design and operation might help to alleviate the soak-back problem in future engines. Insulation of bearings to decrease the flow of heat to the bearing from external sources should lower the soak-back temperature. Auxiliary cooling of the turbine wheel or continued circulation of the lubricant after shutdown should also lower it. The possibility of bearing failure during dry starts after soak-back can be decreased by improving cage wear characteristics; this improvement can be obtained by use of cage materials which have inherent antiweld properties.

OIL INTERRUPTION

In several Korean actions, engines in battle-damaged airplanes operated long enough after lubrication failure to save both the pilot and the aircraft (ref. 1, appendix B). Other battle-damaged airplanes were lost; there is a possibility that some of these could have been saved if the engines could have operated for 15 minutes without oil flow to the bearings. This possibility has prompted the U.S. Air Force to consider operation without oil flow as a "necessary attribute for a fully acceptable turbojet engine" (ref. 1, appendix B). Rolling-contact bearings require very little lubricant for adequate lubrication; this fact may account for their good performance under oil-interruption conditions.

Oil-interruption tests (ref. 1, appendix B) showed that two engines operated longer than 15 minutes. Routine bearing requirements are, however, becoming more severe. Future engines of higher power will be less likely to survive such an interruption without special precautions. One possible solution to this problem is use of an emergency lubrication system or an accumulator that could lubricate engine bearings to ensure their survival for sufficient time for the pilot to take emergency measures. For single-engine aircraft, measures of this type are imperative.

In multiengine commercial aircraft, the danger of battle damage is not present, but a lubrication system may fail. For such installations, an auxiliary lubrication system might be feasible, or an engine with lubrication failure could be shut down in flight.

COOLING OF BEARINGS AND LUBRICANTS

Engine temperatures higher than current values will increase heat flow to the engine bearings. This higher heat flow will increase the cooling load on both the lubricant and the atmosphere around the bearing. At supersonic speeds, ram air or air bled from the compressor will not be able to provide adequate cooling (ref. 1). Supersonic speeds also cause a rise in bulk fuel temperature because of higher skin temperatures. The fuel can then absorb less heat from the lubricant (in a lubricant-fuel heat exchanger) before it begins to vaporize or decompose. Unless a bearing-lubricant combination capable of operating at the high temperatures obtained in supersonic flight can be developed, lubricant refrigeration will be necessary. Many aircraft employ refrigeration systems for cabin and equipment cooling, but their enlargement to include lubricant cooling would result in additional weight, complexity, and cost.

IMPROVING RELIABILITY IN DESIGN AND MANUFACTURE

IMPROVED MATERIALS

For materials used in the races or the rolling elements, an improvement in the quality of the steel (such as by vacuum melting) can greatly improve the load capacity or equivalently the fatigue life of the bearing (refs. 8 and 9 and fig. 15). As indicated in reference 5, the materials for cages of bearings to be used in higher temperature engines must be improved in strength, oxidation resistance, and sliding-friction compatibility with the race material.

IMPROVED DESIGN

Improved cage designs afford some opportunity to improve turbojet-engine bearing performance. One goal of bearing-cage research should be hydrodynamic lubrication between the cage and its locating surface and between the cage pockets and the rolling elements. Full achievement of this goal is unlikely, but even a partial achievement would result in an improved bearing. Research showed that roller bearings designed to improve lubrication and cooling (by providing less resistance to lubricant flow through the bearing) gave better performance than conventional bearings (ref. 7). Schematic sectional views of two conventional and one experimental roller bearing illustrate patterns of lubricant flow through the bearings (fig. 16). Because of the restriction to oil flow into the bearing, the conventional bearing with inner-race-riding cage is poorly cooled and the cage-locating surfaces are poorly lubricated. The conventional bearing with outer-race-riding cage restricts oil flow out of the bearing; churning of the trapped oil therefore takes place and considerable heat is generated at high speeds. This design, however, shows improved lubrication of the cage-locating surfaces because oil is forced to flow over the surfaces requiring lubrication. The data therefore show (fig. 16) little difference in operating temperature but some difference in limiting speed. Limiting speed is defined as the maximum speed to which the bearing can be operated with an equilibrium temperature.

The experimental bearing had little restriction to flow of oil through the bearing. It was equipped with a straight-through outer race and a number of cut-outs on either side of the cage to provide free flow of oil out of the bearing.

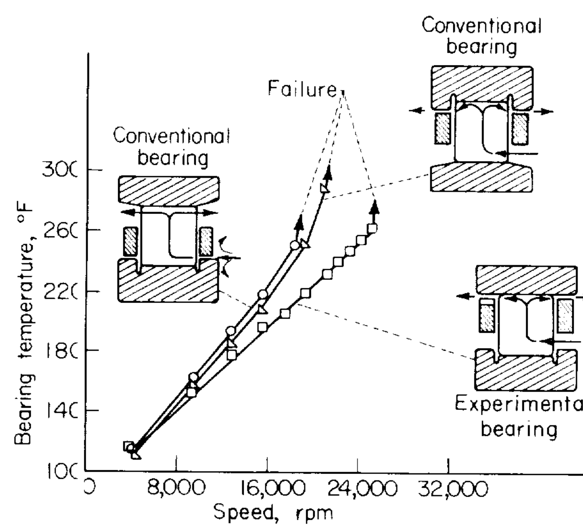


FIGURE 16.—Data showing lower temperatures and higher failure speeds permissible with experimental bearing designed for improved cooling and lubrication. Bearing designed for easy flow of lubricant through bearing and for forced flow of lubricant over cage locating surfaces (data from ref. 7).

Some oil is, however, still forced to flow over the surfaces requiring lubrication (cage locating surfaces). The results for this bearing (bottom curve of fig. 16) show *both* an appreciable reduction in bearing operating temperature for any one speed and an appreciable increase in limiting speed.

Redesign can also help solve the fatigue problem in bearings. The bearings can be redesigned either to increase load capacity of the single bearing (space permitting) or to decrease the load per bearing by use of a tandem arrangement of multiple bearings.

INSPECTION AND QUALITY CONTROL

Inspection and quality control during bearing manufacture are extremely important in bearing performance. Present day aircraft bearings are manufactured to extremely small tolerances. Further reduction in the tolerances of bearing parts could be achieved only with difficulty and at great expense. Present inspection and quality control methods and difficulties of improving them are discussed in the appendix.

The weakest link in bearing quality control may be the race and rolling-element material. Inclusions are usually detrimental to fatigue life, but little is known about the effect of inclusion characteristics (size, type, shape, distribution) on fatigue life under rolling-contact stresses. Research programs are being set up to obtain the answers to these questions. Figure 17 illustrates the scatter in fatigue life of ball bearing inner races (size 6207) at different stress levels. Until a group of fatigue failures with less scatter is observed, the role of dimensional tolerances in determining life scatter cannot be accurately evaluated.

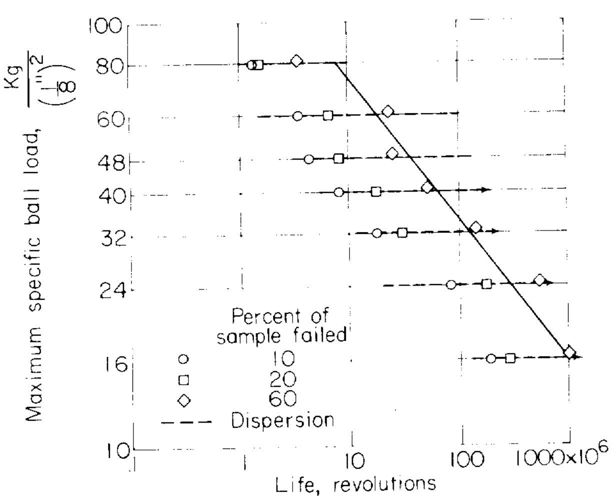


FIGURE 17.—Scatter in fatigue life of ball-bearing inner races (6207 size at different stress levels) (data from ref. 9).

OVERHAUL AND MAINTENANCE METHODS FOR IMPROVING RELIABILITY

BEARING INSPECTION AND REPLACEMENT

The frequency of bearing inspection would depend on the past history of the bearing in that particular application and on the loss of life or property that a failure of the bearing would entail. Specifically, in a new engine type, all bearings should be inspected at short intervals until enough running time is accumulated to establish a valid estimate of probable bearing life. During the development and early application stages of an engine, most bearing failures will be caused by faults in the bearing environment. These failures, however, should be fully correctable. It is during the period of normal use after sufficient experience has been accumulated to correct early faults that time-dependent bearing failures such as wear and fatigue appear. Proper engine maintenance should include inspection of the lubrication system for wear debris and chips and careful inspection of all bearings at overhaul.

When a bearing is removed for inspection, it should be visually checked for flaking, pitting, scuff marks, roughness in the races, brinelling of the balls or races, metal transfer to and from the cage, and excessive wear; the bearing should then be rejected or passed on the basis of established specifications on these items. After a thorough cleaning, a noise test should be given to the bearing.

In order to minimize fatigue failures in flight, it is necessary to schedule replacement based initially on expected fatigue life and later on service experience in the partcular application. At practical load levels, bearings have a finite life; also the scatter of data may be quite broad. Complete avoidance of fatigue failures by a replacement schedule may, therefore, be impossible. Rather, the probability of fatigue failure in a specific application will be governed by the frequency of replacement. Differences in service history must be considered in setting up the replacement schedule. Records of bearing service time must be kept to apply this schedule. Since fatigue is of importance only in the ball thrust bearing, time records need be kept only for these bearings.

It may be possible to reduce the incidence of bearing failures due to excessive wear by bearing replacement based on measured clearances within the bearing at inspection periods. Allowance must be made for the wear that will occur before the next scheduled inspection, and bearings that do not have sufficient wear life left to run until the next inspection period must be replaced.

There are three weaknesses which are prone to arise in maintenance methods for bearings.

The first is a possible lack of uniform training among people who are expected to pass judgment on a bearing. The same bearing might pass inspection at one station and be rejected at another (ref. 13). This clearly emphasizes the need for better training. Training courses conducted by competent instructors using reliable manuals on antifriction bearing maintenance and trouble detection such as references 14 to 17 would be valuable. It must be borne in mind, however, that the determination of the serviceability of a bearing is a difficult task at best and frequently a nearly impossible one. For this reason, research is being conducted in an effort to develop precise techniques for determining bearing serviceability criteria. Reference 18 is a progress report in this program.

A second common weakness in maintenance methods for bearings is a lack of cleanliness in the bearing inspection and assembly areas. Bearings are assembled and inspected at the manufacturing plant under ideally clean conditions to ensure freedom from dirt and other contaminants. In order to perform satisfactorily, they must be given the best possible care in handling and installation throughout their lives.

TIME RECORDS

A third weakness in the maintenance methods for bearings is the lack of comprehensive bearing time and life records. Time records would result in more adequate statistics and enable improvement in bearing reliability by ensuring against normal fatigue failures. In order to ensure against normal fatigue failures, it is necessary to keep running time records only for the ball thrust bearing. Serviceable bearings removed from discarded engines should be tagged with their running time and placed in the parts pool.

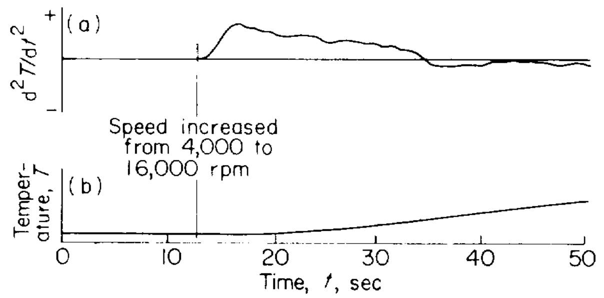
Bearing distress may manifest itself in several ways. Any one or more of the following bearing characteristics may indicate bearing trouble:

(1) Increased noise or vibration level
(2) Higher operating temperature
(3) Increased friction torque
(4) Wear

In an engine installation, bearing noise and friction torque cannot be used because they could not be detected. Any increase in bearing noise level would be lost in the engine noise, and increased friction torque could not be detected because it represents only a small fraction of total engine power.

A number of checks of engines and bearings in aircraft might be of considerable help toward improving reliability of both engines and bearings. These include

(1) Temperature and temperature acceleration. Thermocouples placed in bearing housings and read in the cockpit have been used extensively with new engines in aircraft. After sufficient engine time has been accumulated to remove design faults, however, it has been general practice to remove these thermocouples. Temperature level may not be sufficiently sensitive, however, to indicate bearing trouble. Preliminary research has shown (ref. 19) that incipient bearing failures can be detected by measuring bearing temperature acceleration. The response of temperature acceleration to changing operating conditions is much more rapid than that of temperature Figure 18 illustrates typical tempera- and temperature-acceleration patterns that occur during an increase in severity of operating conditions. After an increase in severity of operating conditions, temperature acceleration is positive. If an equilibrium condition is being approached, it becomes negative after a time interval. During an incipient failure, however, temperature acceleration remains positive unless the cause of the imminent failure is removed. Figure 19 shows a temperature-acceleration pattern during an incipient failure produced by shutting off the oil flow. Restoration of the oil



(a) Temperature acceleration.

(b) Temperature.

FIGURE 18.—Temperature and temperature acceleration patterns during increase in severity of operating conditions on 75-millimeter-bore cylindrical roller bearing (data from ref. 19).

flow prevented a total failure and caused the temperature acceleration to become negative.

(2) Vibration. The use of vibration level to indicate bearing trouble might be feasible, although increased vibration might indicate trouble in any one of many components. An accelerometer and a vibration meter might be used to measure vibration amplitude. It is standard practice to measure engine vibration levels in test stands, but measurement in flight would involve a development program in each specific installation. Vibration would vary with flight conditions and perhaps from airplane to airplane. In a multiengine airplane, wing-mounted engines would present additional problems.

(3) Wear. Excessive wear might be detected by regular examination of the lubrication system for foreign material or wear debris in filters. An instrument to detect wear in flight does not appear worthwhile, because a wear check can easily be made between flights.

One way of obtaining a quantitative measure of wear would be to use a radioactive cage material. Such a scheme, however, might be unduly complicated.

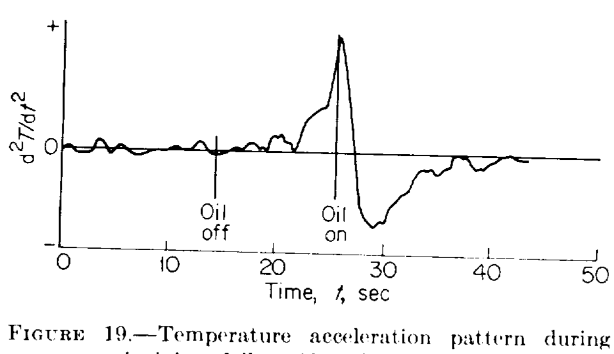


FIGURE 19.—Temperature acceleration pattern during incipient failure (data from ref. 19).

(4) Lubrication system. Ground checks might be made for wear debris as in item (3), and devices for detection of blockages or leaks in the system should be studied. Flowmeters and pressure gages, or simple warning lights, to indicate loss of oil pressure or flow might be installed in the cockpit.

ADDITIONAL INFORMATION NEEDED FOR IMPROVEMENT OF OPERATIONAL RELIABILITY

Studies to obtain information on improved bearing materials, bearing designs, and lubricants are needed, but this report discusses only information required to improve reliability in the sense of avoiding serious bearing failures.

Bearing reliability could be improved if additional information could be obtained on the following factors:

(1) Methods of indicating incipient bearing failure during operation

(2) Methods of detecting lubrication system malfunction during operation

(3) Time lapse between incipient and total failure, for the various failure types

(4) Relation between material properties and fatigue life

(5) Mechanism of failure during oil interruption

(6) Methods of extending bearing time to failure, following oil interruption

CONCLUDING REMARKS

Data on bearing failure and life (from service experience of turbojet engines) are insufficient to draw general conclusions concerning bearing reliability.

Table IV shows a compilation of some bearing failure types and some of the possible corrective measures. Fatigue life, which is of primary importance to the ball thrust bearings, can be increased by reducing load; bearing tests have established that life is inversely proportional to the cube of load. At practical load levels, bearings have a finite life; also the scatter of data may be quite broad. Complete avoidance of fatigue failures by a replacement schedule may, therefore, be impossible. However, the probability of fatigue failure can be kept low by a replacement schedule that is based on service

TABLE IV.—BEARING-FAILURE TYPES AND CORRECTIVE MEASURES

Failure cause	Corrective measures
Fatigue (thrust bearing, primarily)	(a) Low fatigue life can be increased by reducing load. Life $\propto \frac{1}{\text{load}^3}$. (b) There should be scheduled replacements based on experience.
Wear	(a) Excessive wear rate indicates need for redesign. (b) There should be scheduled inspections and replacement at a specified wear.
Extreme boundary lubrication	(a) When excessive metal transfer occurs at sliding surfaces, material change and redesign are indicated.

experience and that takes into account differences in service history. An excessive wear rate indicates a need for redesign. Under conditions of normal wear, bearing replacement can be based on measured clearances within the bearing at inspection periods. Bearings that do not have sufficient wear life left to run until the next inspection period must be replaced. For extreme boundary-lubrication failures, indicated by excessive metal transfer at the sliding surfaces, material change and bearing redesign are indicated.

Bearing performance and reliability can be improved by

(1) Additional information
(2) Better failure detection methods
(3) Better maintenance practice

Studies designed to eliminate bearing failures would be benefited if more complete and accurate information on the causes, modes, and frequencies of bearing failures were compiled at the maintenance and overhaul bases. For this purpose, the inspectors must be trained with regard to the information needed.

The number of catastrophic bearing failures could probably be reduced if better methods of bearing- and lubrication-system failure detection were developed.

APPENDIX

INSPECTION AND QUALITY CONTROL

The number of bearings in the lot being inspected determines whether 100 percent or a statistical sample is inspected. Statistical inspection methods are never employed with lots smaller than 300 and usually not with lots smaller than 500. At present, practically all rotor bearings for turbine engines are made in quantities under 300 so each bearing is inspected. This means that very precise dimensional measurements together with magnaflux and etch inspection of races and balls (used to detect surface flaws, cracks, and material defects) are made.

Many types of special gages are used in the production line and final inspection of precision bearings.

Ball sphericity is checked and balls are size matched for specific bearings. A vibration meter is used to check the smoothness and vibration level of an assembled bearing.

All turbine engine rotor bearings are given a very thorough visual inspection, and definite limits regarding pits, dents, and scratches have been established. Races not acceptable in the visual inspection are reground and reinspected. Final inspection of bearings is done in areas where temperature, humidity, and dust content are closely controlled. Most of the gages used in the final inspection of bearings require precise temperature control and skilled operators to give meaningful results.

The improvements in bearing performance that result from smaller tolerances must be evaluated experimentally and must be weighed against the increased cost of manufacture. Even for perfect alinement, stress differences will exist in the various balls because of the size differences. Closer matching of balls would probably improve bearing fatigue life. This closer matching would, however, accentuate the problem of reproducibility of measurement; when measurements to a few millionths of an inch are attempted on a set of balls, it is often found that these measurements cannot be reproduced at a later date.

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CHAPTER X

ENGINE FUEL CONTROLS

By JOHN C. SANDERS and HAROLD GOLD

SUMMARY

The reliability of fuel controls on turbojet engines is reviewed on the basis of available failure statistics. In general, these statistics indicate that fuel-control systems that incorporate many functions are more subject to failure than those incorporating fewer functions.

Air Force accident records indicate that one-third of the flight accidents occurring with turbojet aircraft are caused by fuel-control malfunction. An evaluation of the performance of current fuel controls from Air Force records indicates further that nearly three-fourths of the fuel-control malfunctions are attributable to mechanical failure of control and fuel-system components and that one-fourth are due to inadequacy of the control method. Therefore, the possibility exists that a very significant improvement in turbojet-aircraft reliability could be achieved by improvements in the details of mechanical design of control and fuel system components.[1]

The effects of fuel-control malfunctions on engine performance are examined, and the factors that lead to control complexity are evaluated. The evaluation indicates that fuel-control complexity results from complex requirements imposed by the dominant requirement of operational reliability of the turbojet aircraft.

INTRODUCTION

The fuel control on the turbojet engine, as considered in this paper, performs all the functions of automatic regulation of the engine variables such as speed, temperature, acceleration, and thrust by means of the control variable, fuel flow. In addition, fuel-injection equipment will be considered to be a component of engine fuel controls.

Failure reports compiled by the Air Force over the past several years show that component failure in control and fuel-system equipment used on jet-propelled aircraft occurs with serious frequency. Air Force records indicate that fuel-control malfunctions cause one-third of engine failures that lead to flight accidents. This represents the largest category in the Air Force compilation.

The Air Force records of control-system failure show that simple controls are more reliable than complex ones. The problem of turbojet-engine control reliability cannot, however, be solved simply through reduction in control complexity. Control complexity results from complex-control requirements imposed by the dominant requirement of operational reliability of the turbojet aircraft.

The object of this study is to examine the effect of control- and fuel-system-component reliability on the overall reliability of the turbojet engine and to evaluate the factors that lead to control complexity. First, a review will be made of failure statistics. Then an investigation of the modes of failure possible by inadequate control will be presented. With this information, criticism will be made of several methods of control- and fuel-system-component configurations.

STATISTICS ON CONTROL FAILURE

During the past decade of use of the turbojet engine in aircraft, considerable experience has been obtained with various types and methods of control. The U.S. Air Force has collected Unsatisfactory Reports of control and engine fail-

[1] Subsequent to the period in 1953 covered by the statistics and subsequent to the original writing of this report, considerable improvement has been made in mechanical reliability of control components. The basic difficulties in temperature and acceleration control remain.

TABLE I.—DISTRIBUTION OF ENGINE AND CONTROL MALFUNCTIONS ATTRIBUTED TO CONTROL-COMPONENT FAILURE OBTAINED FROM UNSATISFACTORY REPORTS

Nature of control difficulty reported	Total reports, percent
Flameout	5.4
Overspeed	5.7
Compressor surge	6.9
Overtemperature	7.0
Loss of control	13.4
Inability to obtain rated speed	29.6
Fuel-system oscillation	32.0
Total	100.0

ures during this period. (For a description of the Unsatisfactory Reports, see ch. II.)

The distribution of engine and control malfunctions attributed to control-component failure (as obtained from the Unsatisfactory Reports) is given in table I. The percentages given in the table are based on the total number of reports on three different engines. The tabulation represents a total of 1420 reports convering an operating time of 87,765 hours. The basic characteristics of the control systems employed on the engines are given in table II.

Engines B and C incorporate more control functions than engine A. Engines B and C may be considered to be of approximately equal complexity and of greater complexity than engine A. The frequency of Unsatisfactory Reports appears to be considerably lower for the simplest engine and control system (engine A) than for the more complex engines (engines B and C). Engines B and C appear to show about the same records of reliability in spite of considerable differences in the design of control and fuel-system components.

ANALYSIS

The records (as contained in the Unsatisfactory Reports) on the reasons for the control difficulties are incomplete in most cases. The statements in the reports are often vague and are therefore subject to several possible interpretations. Nevertheless, there is sufficient information on the possible modes of action of both engines and control systems to make a rational interpretation in nearly all cases. On this basis, an attempt will be made to analyze the statistics in table I to indicate the probable causes for failure.

Fuel-system oscillation and loss of control.—Mechanical failure of fuel-system components represents the greatest source of control malfunctions in current Air Force turbojet engines. Unsatisfactory Reports list control breakdowns due to such causes as rupture of material and faulty assembly. Parts that have failed because of rupture of material include diaphragms (fabric), bellows (metal), governor shafts, and pump drives. Diaphragms failed many more times than did pump drives. The failure of a dia-

TABLE II.—BASIC CHARACTERISTICS OF THREE CONTROL SYSTEMS COVERED IN UNSATISFACTORY REPORTS

Engine	Control computer	Computer fluid	Overspeed protection	Surge protection	Blowout protection
A	Hydraulic and mechanical	Oil	Yes	No	No
B	Hydraulic and mechanical	Fuel	Yes	Yes	Yes
C	Hydraulic and mechanical	Oil	Yes	Yes	Yes

phragm on one engine control system accounted for nearly all the cases of "Loss of control" listed in table I.

A great many faults are associated with flow-control and shutoff valves. Valves in flow dividers for duplex fuel nozzles have been found to stick, causing both loss of power and overtemperature in starting. Pressure-regulating valves have been found to oscillate. Fuel-system oscillation represents the most frequently reported malfunction (table I). It is even likely that some of the reports of surge were actually cases of fuel-system oscillation. Fuel-system oscillation is attributable to loss of frictional damping as regulator valve elements wear. This type of valve instability may not appear in every control model and thereby may escape detection in the development of the control system.

Overspeed and inability to obtain rated speed.—Current speed controls are capable of regulating engine speed to well within 1 percent at rated speed. Thus, in the case of the properly functioning speed controls, there is little loss of available thrust or reduction in turbine life due to control error. As shown in table I, overspeed was reported in 5.7 percent of the Unsatisfactory Reports. These reports give overspeeds ranging from 103 to 111 percent of rated speed. This magnitude of overspeed does not lead to immediate destruction but may simultaneously cause a severe overtemperature, the total result of which materially shortens turbine blade life. The records examined showed no indication of immediate destructive overspeed. However, this difficulty can lead to immediate destruction of the aircraft; and, hence, the records may be incomplete.

The largest number of speed-control difficulties is associated with the inability to obtain rated speed. As shown in table I, this malfunction was reported in 29.6 percent of the Unsatisfactory Reports. Speed-governor malfunctions are usually associated with loss of adjustment or frictional effects. It is therefore likely that, on the average, governor malfunctions will result in as many underspeeds as overspeeds. Thus, probably 5.7 percent of the total reported malfunctions were underspeeds due to governor failure, and 23.9 percent were cases of inability to obtain rated speed because of engine deterioration or other control-component failure. On this basis, 11.4 percent of the malfunctions are attributable to speed-control failure. These failures are attributable to mechanical defects rather than inadequacy of the control method.

The frequency of cases of engine deterioration or of control-component failure (other than speed control) causing inability to attain rated speed appears to be related to the complexity of the engine and control system. Table III shows the distribution of speed-control malfunctions for the three engines included in the Unsatisfactory Reports. In the case of the simplest engine (engine A), the reports are nearly equally divided between overspeeds and low-speed limiting. In the case of the more complex engines (engines B and C), the low-speed limiting very definitely predominates.

TABLE III.—DISTRIBUTION OF SPEED-CONTROL MALFUNCTIONS

Engine	Speed-control malfunctions related to overspeed, percent	Speed-control malfunctions related to low-speed limiting, percent
A	45	55
B	30	70
C	20	80

Overtemperature.—Overtemperature was reported in 7.0 percent of the Unsatisfactory Reports involving control difficulties (table I). Many of the overtemperatures are connected with overspeeds. The overtemperatures reported indicate tailpipe temperatures between 1800° and 1900° F.

Current-production temperature controls do not utilize direct temperature sensing. An attempt is made in these systems to keep the temperature within safe limits by speed and acceleration control. This control of temperature by secondary variables does not adequately protect the engine over all its operating conditions. For example, it does not give adequate protection during starting. This lack of temperature control during starting is believed to cause much more engine damage from overtemperature than is indicated from the Unsatisfactory Reports.

Reports obtained by Lewis laboratory personnel during visits to Air Force overhaul depots indicate that a very large percentage of engines (possibly greater than 50 percent) is subject to severe overtemperatures between overhaul periods. The overhaul depots have on hand an extremely large number of turbine blades which have been discarded because of overtemperature. These discards are suspected to have suffered the overtemperature damage chiefly during starting. This overtemperature damage can be attributed to the inadequacy of the temperature-control methods now in use.

Surge control.—According to the Unsatisfactory Reports, compressor surge occurred in 6.9 percent of the total malfunctions (table I). A large number of these cases probably resulted from drift or improper adjustment of the schedule-type controls in current use. Drift or loss of adjustment is caused by mechanical failure and may be eliminated with refinement in design; however, environmental changes can cause a deviation of the engine surge characteristics from a fixed-design schedule. It can therefore be assumed that the surge-control malfunctions resulted both from mechanical breakdown and from inadequacy of current control methods to correct for all environmental changes.

In addition to the reports of surge in the Unsatisfactory Reports, there are a significant number of cases in Air Force accident records in which compressor stall is thought to have caused flight acicdents during landing. It is likely that these cases of surge resulted from the lowering of the surge limits (below the normal schedule) due to the successive accelerations that often occur in landing. It is not feasible to lower the acceleration schedule to avoid this condition of surge entirely because of the severe penalty that would be imposed on engine acceleration. New methods of surge prevention or incipient-surge detection will have to be developed before the hazards from surge can be completely eliminated.

Flameout.—Flameout from all causes is reported in 5.4 percent of the Unsatisfactory Reports relating to control difficulties (table I). The reported flameouts are associated with (1) drift of the flameout controls, (2) deterioration of the engine or of the fuel system, and (3) the inadequacy of the control to handle the specific operating conditions encountered. Flameouts have occurred more frequently at low and medium altitudes than at high altitudes. This indicates that deterioration of the engine or fuel system or flameout-control inadequacy caused more flameouts than did flameout-control drift (mechanical failure).

Flameout caused by fuel-control malfunction is listed in Air Force accident records as the cause of a significant number of flight accidents. Because these flameouts occurred at low altitudes, it is likely that many of the reported flameouts occurred during acceleration or deceleration. This type of failure is attributable to inadequacy of the method of flameout prevention that is currently employed.

CONCLUSIONS

On the basis of the analysis of the failure statistics, it is apparent that mechanical breakdown is the major cause of engine fuel-control malfunction. It has been indicated that through mechanical failure the various control components contributed to fuel-control failure as follows: speed control, 11.4 percent; regulating valves in the fuel system, 32 percent; ruptured diaphragms, 13.4 percent; and surge control, 3.4 percent. Of the malfunctions resulting in inability to attain rated speed that were attributed to engine or control deterioration, it is likely that about one-half were cases of unidentified mechanical failures in the engine fuel control. Thus, unidentified mechanical failures can be considered to contribute 12 percent of the control malfunctions. The total percentage of malfunctions of engine fuel controls attributable to mechanical breakdown is then 74.1 percent. The remaining 25.9 percent of fuel-control malfunctions is attributable to inadequacy of the control methods currently in use.

Currently, control designs are changing rapidly, and a given design is not retained for sufficient time for complete elimination of mechanical defects. When control designs are stabilized, it can be expected that mechanical difficulties will eventually be eliminated by improvement in details of the equipment.

A large number of controls of various designs are now under development. The Lewis laboratory has not made a design analysis of these controls and hence does not attempt in this paper to suggest improvements to the mechanism of any specified control. Instead, an attempt is made to provide an insight into the ways an engine can fail through inadequate control and the problems associated with providing adequate control over the wide range of flight conditions the engine is expected to encounter.

MODES OF ENGINE FAILURE

Engine damage due to improper or inadequate control can result from the following:

(1) Overspeed

(2) Overtemperature of engine parts

(3) Vibratory stress and overtemperature produced by compressor stall or surge

(4) Flameout

(5) Thermal shock and overtemperature during starting or quick thrust change

OVERSPEED

High overspeed can cause immediate rupture of rotating parts, such as compressor and turbine disks. The conservation of the design and the age of the rotating parts both influence the margin of overspeed before such catastrophic failure occurs.

Overspeeds less than those needed to cause instant failure produce a damage that accumulates with time. Stress-rupture and associated creep are such causes of this deterioration, particularly in the turbine blades. The effect of speed (and stress) on the stress-rupture life of a representative turbine blade material is shown in figure 1. The abscissa is the square root of material stress and hence is proportional to engine speed. For the stress-rupture-life characteristics shown in figure 1, an increase in speed (and hence stress) of 10 percent above an abscissa value of 150 will decrease the stress-rupture life from 1000 to 350 hours at 1433° F. This represents a decrease in stress-rupture life to one-third of the life at rated speed.

Frequently the overspeed is accompanied by overtemperature, particularly in engines with a fixed nozzle area. For a representative case a 10-percent overspeed is accompanied by a 200° F overtemperature and a stress-rupture life of less than 1 hour, or 1/1000 the life at rated speed.

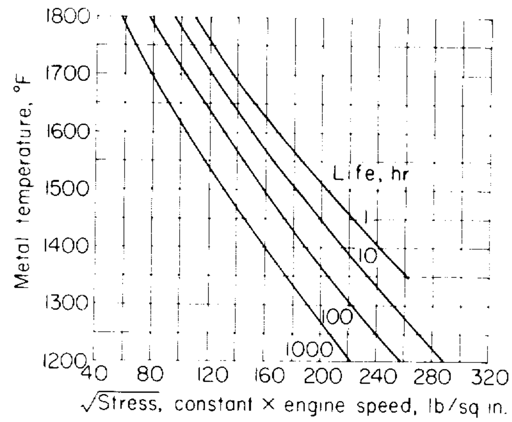


FIGURE 1.—Stress-rupture characteristics of representative turbine blade alloy.

An overspeed control is subject to error in the direction below the desired limit as well as above the limit. When the control limits engine speed below the design maximum, the effect is a reduction in engine power. In the case of engines with fixed exhaust nozzles, thrust falls off about 4 percent for a 1-percent reduction in speed from rated speed. This high rate of thrust loss imposes a stringent requirement of a high degree of accuracy and reliability on speed limiting or regulating controls.

OVERTEMPERATURE

At high temperature the life of turbine blade materials changes drastically with changes in operating temperature. Consequently, it may be expected that the life of the turbine blades will be sensitive to the accuracy of the temperature control system. Figure 1 shows the stress-rupture characteristics of a representative turbine blade alloy as a function of engine speed and material temperature. At a stress of 22,500 pounds per square inch (square root of stress = 150) and a metal temperature of 1500° F it may be seen that an increase in temperature of 100° F reduces the life from 200 to 10 hours, or to $\frac{1}{20}$ of the life at the lower temperature.

In the case of the fixed-exhaust-nozzle turbojet engine the equilibrium gas temperatures are essentially functions of engine speed up to a critical altitude. Above the critical altitude the equilibrium turbine-discharge temperature at rated speed becomes considerably higher than at lower altitudes. This variation is shown in figure 2. From the data it may be seen that the critical

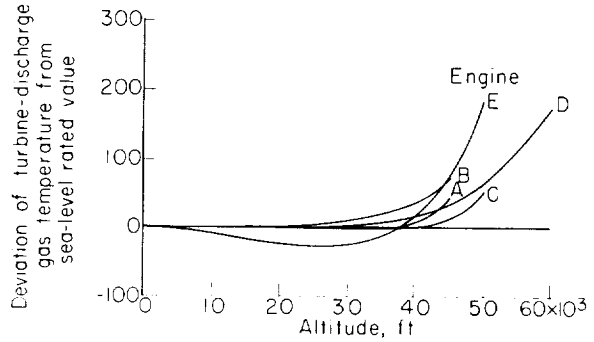


FIGURE 2.—Effect of altitude on turbine gas temperature.

altitude is approximately 40,000 feet. For aircraft flying below 40,000 feet it has therefore been practical to consider the overspeed limit as an adequate overtemperature limit for equilibrium running of the engine. However, during starting and during acceleration of the engine, gas material temperatures rise markedly above equilibrium values. Hence direct overtemperature protection is desirable even for low-altitude aircraft.

The temperature-control problem is similar to the speed-control problem in that small deviations in the controlled variable (temperature) result in large variations in thrust. The relation of thrust to measured exhaust-gas temperature is shown in figure 3. It may be seen that approximately an 8-percent loss in thrust results

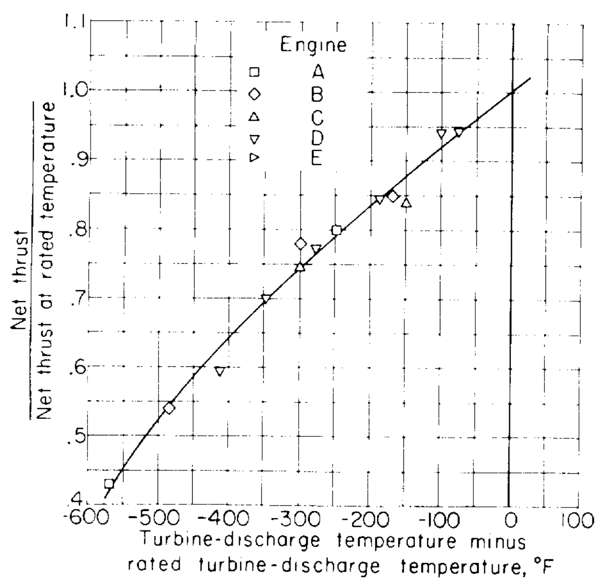


FIGURE 3.—Thrust sensitivity to turbine-discharge temperature at rated speed.

from a 100° F reduction in temperature. Because of this high-thrust sensitivity, it is necessary to operate engines near the maximum allowable turbine blade temperature in order not to sacrifice a large measure of the potential thrust of the engine. The dilemma involved herein is demonstrated in figure 1, where it may be seen that a variation of 100° F could result in a 20:1 variation in material life.

There are two basic difficulties in obtaining sufficiently accurate temperature measurements. The first of these difficulties is simply that of obtaining a measuring sensor that will have the necessary life, sensitivity, and accuracy at temperatures above 1500° F. While a truly satisfactory sensor does not yet exist, it it reasonable to expect that one may be developed in time. The second difficulty in obtaining the temperature measurement results from the irregular gas temperature profiles that exist from the burners to the exhaust nozzles. Figure 4 shows that the radial temperature distribution in the turbine annulus may not only be irregular but may vary appreciably with altitude. Because of the irregular profiles, the material temperature may differ widely from the gas temperature measurement at a given point in the engine. Therefore, a sensor located at one radial station cannot be relied upon for accurate control of turbine bucket temperature. Where temperature control is employed in current practice, single-point sensors at several stations around the turbine annulus are generally used. This practice is inadequate, but it is a compromise with the inadequacy of available temperature sensors. Chief reliance for overtemperature protection is still placed on indirect means of control, such as speed and acceleration control. Improvements in engine design may result in more uniform temperature distributions in future engines. This will greatly ease the temperature-control problem and will be a major step toward greater engine reliability.

During starting, excessive temperature results from the delayed ignition of liquid fuel that has collected in the burners during the initial phase of the starting process (ref. 1). This collection of liquid fuel can readily occur, because the fuel-flow rate required to obtain ignition is large compared with the flow rate required to accelerate the engine from cranking speed. It is

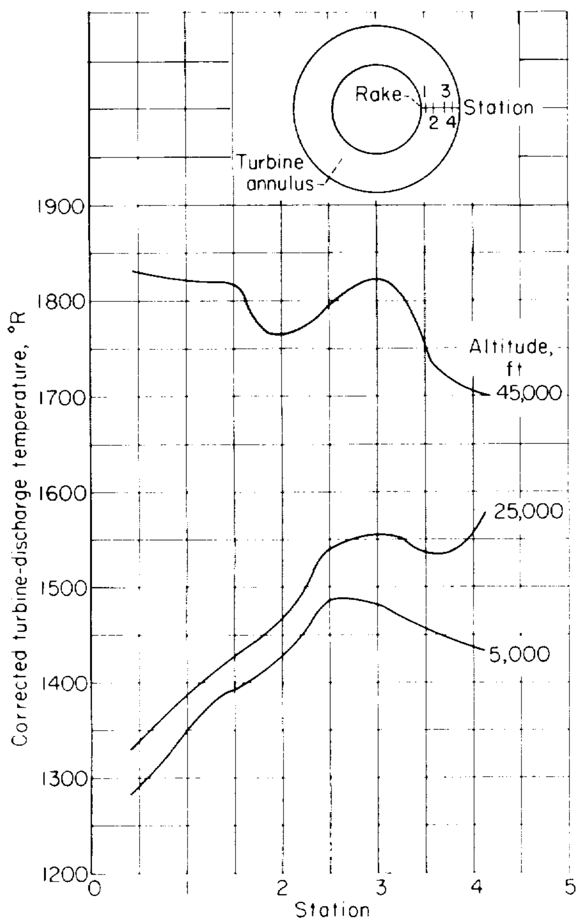


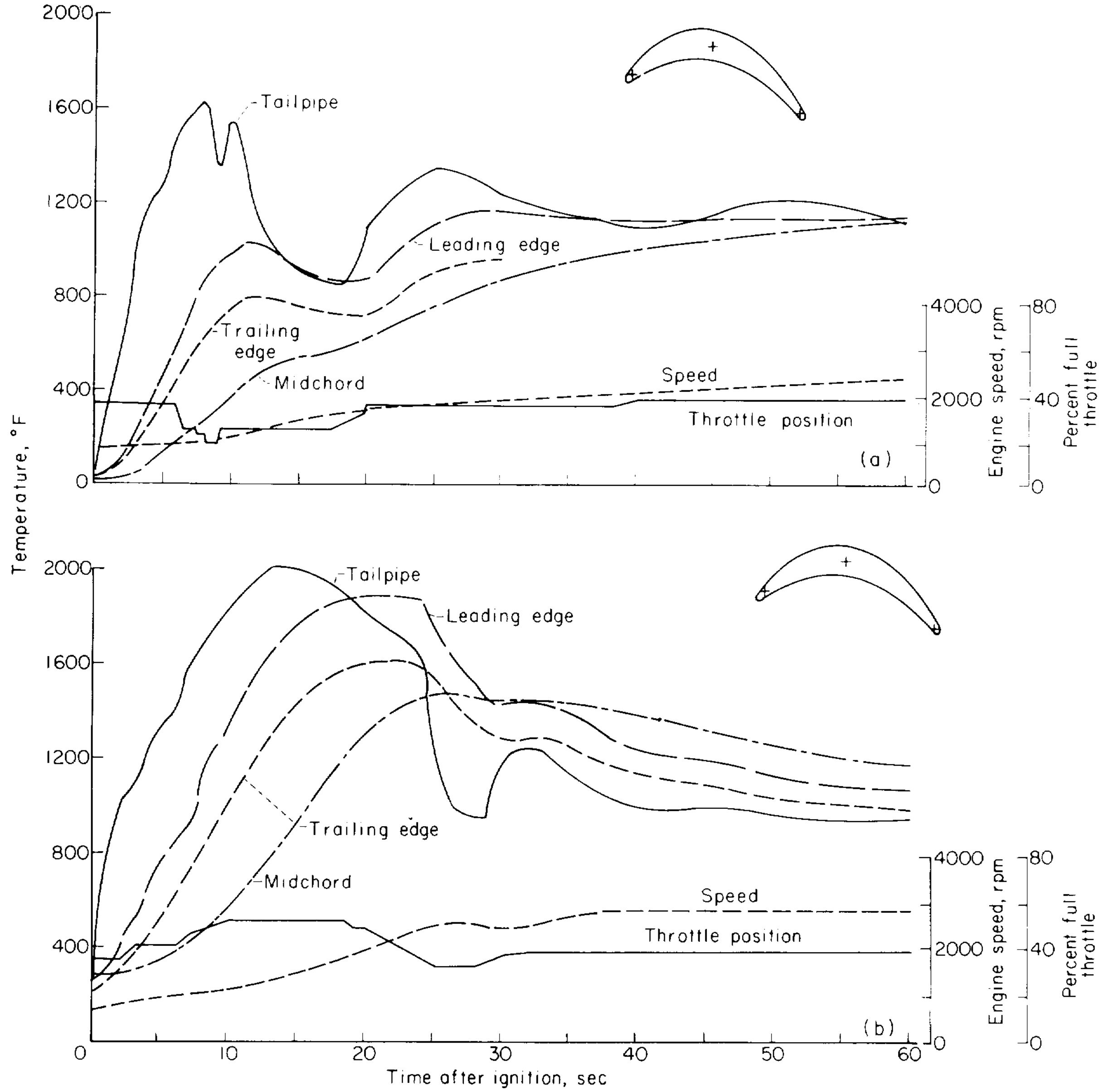
FIGURE 4.—Effect of altitude on temperature distribution at turbine discharge.

therefore necessary to avoid both ignition delay and delay in reducing fuel flow after ignition has occurred. In manual starting, the skill of the operator in evaluating the ignition delay and in manipulating the throttle after ignition has occurred is a very important factor in the avoidance of overtemperature during starting. Figure 5 shows the variation of blade temperature during two engine starts (ref. 2). Figure 5(a) shows a start during which proper manipulation of the throttle has kept material temperature within safe limits. Figure 5(b) demonstrates the manner in which incorrect throttle manipulation results in dangerously high engine temperature.

The two causes of overtemperature during starting are related to inadequate fuel atomization at low flow rates. Engine tests have shown that with a finely atomized fuel spray, engine starting at a fixed fuel-flow rate is possible. Starting at a fixed flow rate eliminates the dangers that result from throttle manipulation. Tests have also shown that with proper atomization overtemperature during starting does not occur in spite of unlimited ignition delay (ref. 3). Engine damage incurred during starting may be largely eliminated as improvements in fuel-atomization components are made.

During engine acceleration the rise in turbine-inlet temperature above equilibrium running values is necessary to establish the accelerating torque within the engine. However, the turbine-inlet temperatures that are required to induce rapid acceleration of the engine are not generally above the equilibrium value at rated speed. Therefore, engine damage from overtemperature during acceleration from normal operating speeds does not represent a significant factor affecting engine life. The principal source of damage associated with the temperature rise during acceleration is the creation of thermal stresses in the turbine. These stresses result from the sudden change in gas temperature and therefore occur even when the gas temperature does not rise above operational limits. Thermal stresses can be reduced by retarding the rate of gas temperature change. This could be accomplished by automatic control but only at the expense of the increased operational hazard due to the slow response of thrust to throttle command. A more desirable solution to the thermal-shock problem lies in improvements in material and design of turbine buckets.

Overtemperature is also encountered at maximum speed when the temperature control does not possess sufficient speed of response to adjust quickly to a disturbance. Such a case is illustrated in figure 6, which shows a time history of turbine-discharge gas temperature following afterburner ignition. In this instance the engine was equipped with a variable exhaust nozzle. Temperature control was obtained by automatic variation of the exhaust nozzle. It may be seen that an overtemperature of approximately 200° F existed for about 2 seconds following afterburner ignition and that this period corresponds to the time required for the nozzle area to change.



(a) Safe-temperature acceleration.

(b) Overtemperature acceleration.

FIGURE 5.—Histories of overtemperature acceleration and safe-temperature acceleration during starting (ref. 2).

More rapid actuation of the nozzle area can reduce such temperature deviations to negligible values.

COMPRESSOR STALL AND SURGE

At the present state of development of axial-flow compressors, it is a practical necessity, in the case of engines utilizing such compressors, to incorporate elements in the control system to prevent operation that will lead to stall. Compressor stall occurs when the pressure rise through the compressor exceeds a critical value at a given engine speed. Figure 7 shows the characteristic variation of pressure ratio with airflow at various rotational speeds in axial-flow compressors. The surge line represents the highest attainable pressure ratio at a given speed or airflow. Also

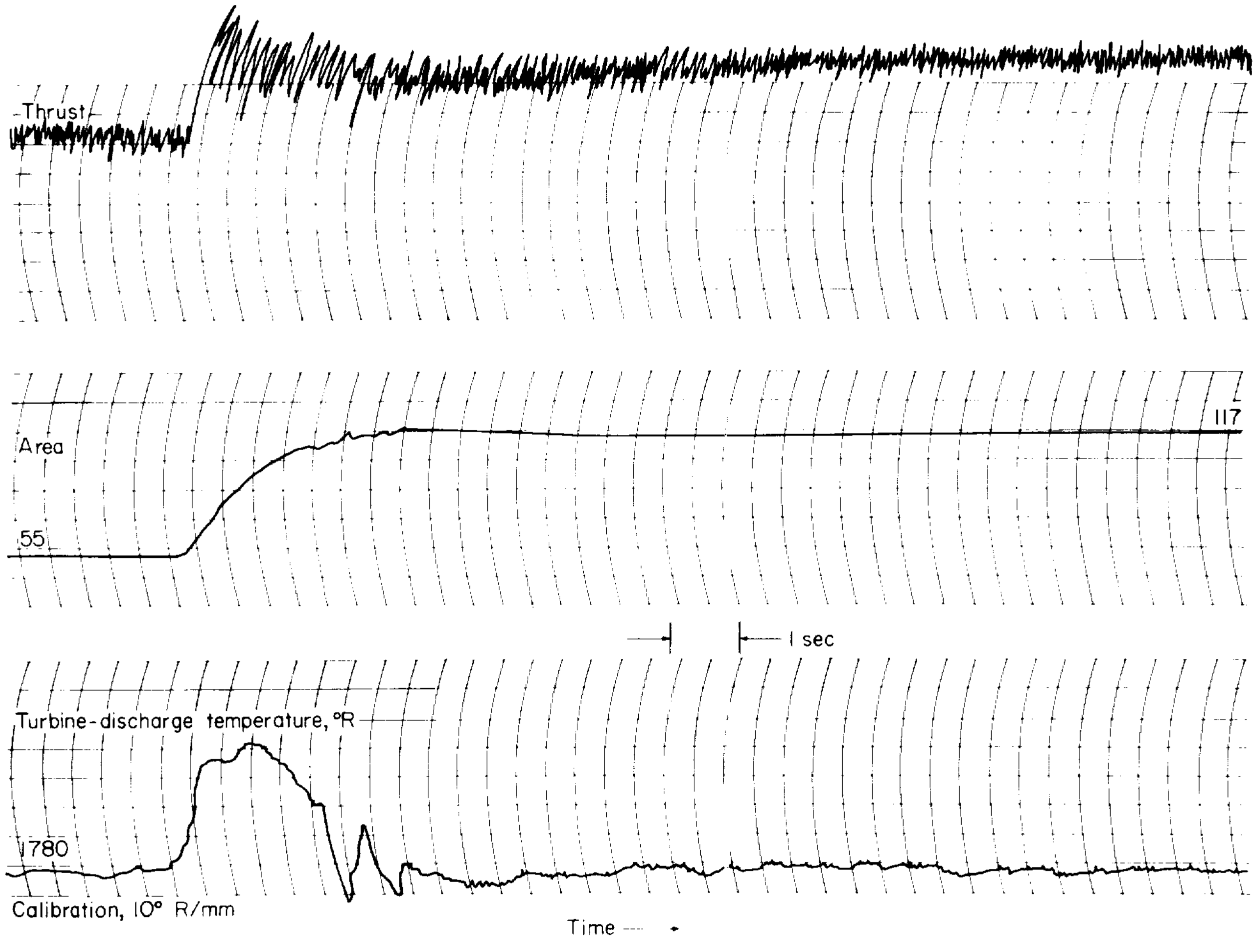


FIGURE 6.—Overtemperature incurred during ignition of afterburner. Temperature controlled by variation of discharge nozzle area.

superimposed on the map are a typical engine equilibrium operating line and lines of constant engine flow. It may be seen that the equilibrium line roughly parallels the surge line up to rated speed and that the lines converge in the overspeed region. The area between the equilibrium line and the surge line is the engine acceleration region. When acceleration is attempted, the increased temperature above steady state initially forces the compressor pressure upward along a constant-speed line, moving the operating point in the map above the steady-state operating line and toward stall or surge. Some engines show a severe deviation from the parallel relation of the surge and operating lines at speeds below maximum. In this case, critical points of operation exist where the margin between the lines is small.

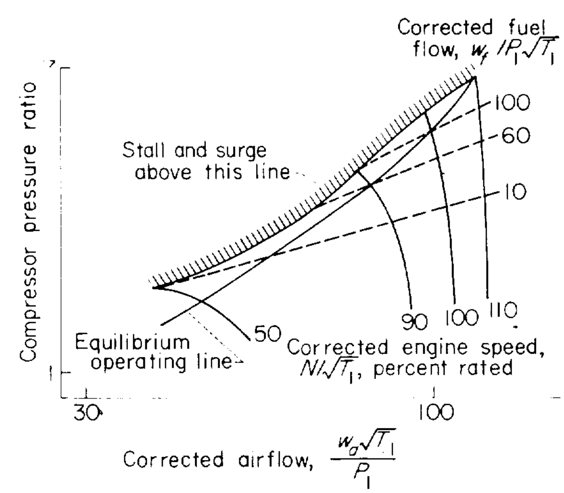


FIGURE 7.—Typical characteristics of axial-flow compressor. N, engine speed; P_1, inlet pressure; T_1, inlet temperature; w_a, airflow; w_f, fuel flow.

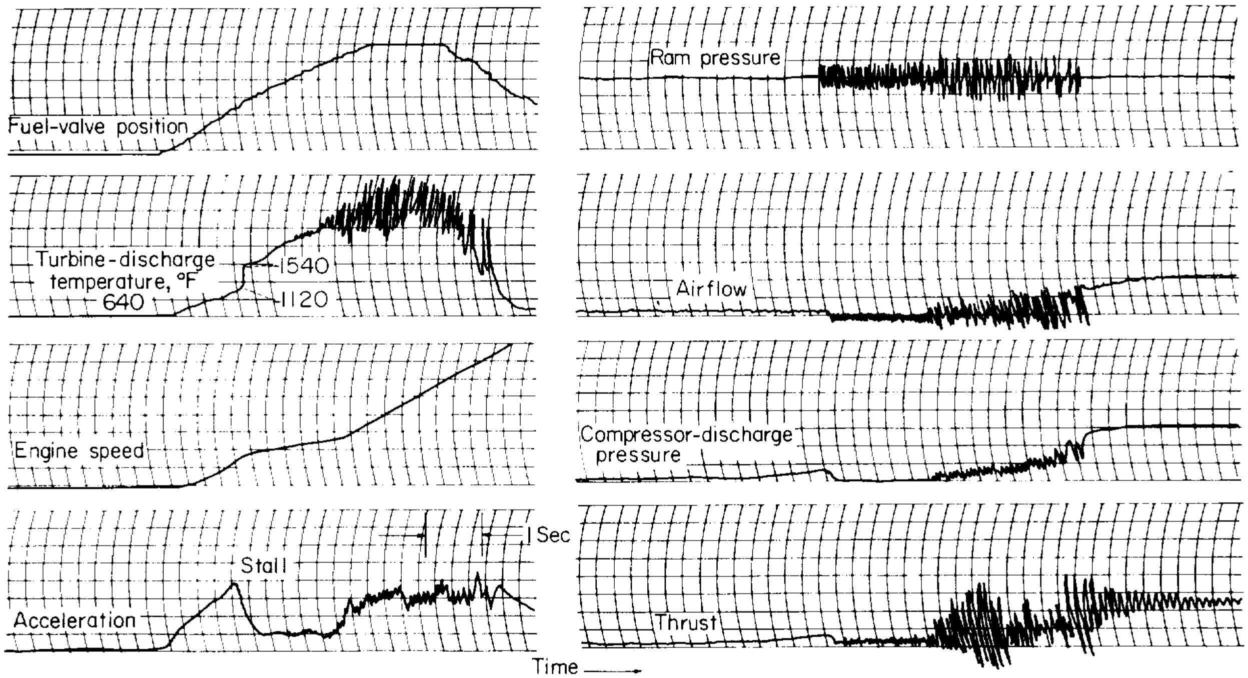


FIGURE 8.—Record of stall and surge during acceleration.

A record of a rapid acceleration in which stall was encountered is shown in figure 8. The acceleration was induced by steadily opening the fuel valve, thus increasing the fuel flow to the engine. It may be seen that, after the fuel flow had been increased beyond a certain value, the compressor-discharge pressure suddenly decreased, the acceleration fell to a low value, and the turbine-discharge temperature shot up to a higher value. A condition of poor acceleration accompanied by fluctuating flows and temperature patterns existed. This is a condition of stall and surge. Thus, stall induces the detrimental conditions of high operating temperature and high vibratory loadings accompanied by very low acceleration.

FLAMEOUT

Loss of combustion can result from either too rich a mixture, too lean a mixture, or from insufficient atomization of the fuel. The overrich mixture can occur during acceleration from very low engine speeds. The overlean mixture can occur during a sudden reduction in fuel flow at high engine speeds. Flameout from overrich or overlean mixtures can be prevented by limiting the magnitude of sudden changes in fuel flow. In order not to impose sluggish engine response, this limit on fuel flow must be varied with altitude, flight speed, and engine speed.

Flameout caused by insufficient atomization imposes an altitude limit on engine operation. Because atomization is a function of absolute fuel-flow rate, the limiting altitude is lowered as engine speed is reduced. Thus an engine operating at rated speed at high altitude is subject to flameout when the speed is reduced (even if slowly reduced) for descent of the aircraft. The relighting of an engine at high altitude and high flight speed is difficult. Explosions have been experienced under such starting conditions.

A minimum fuel-flow limit can be utilized for altitude flameout prevention. However, this provision imposes simultaneously a minimum thrust limit. The minimum thrust limit, in turn, imposes a maximum angle of descent on the aircraft at high altitudes. If the maximum angle of descent is exceeded, the engine thrust can drive the aircraft to a destructive airspeed.

METHODS OF CONTROL

This discussion of turbojet-engine reliability has been limited principally to modes of engine damage possible with inadequate control. This section presents a discussion of basic methods whereby control is achieved. Four control systems are described. The first system discussed is the system having the minimum number of automatic components with which it is practical to fly a turbojet engine. The remaining three systems discussed are the more complex systems that offer more complete protection.

MANUAL THROTTLE WITH OVERSPEED PROTECTION

One of the simplest control systems is shown in figure 9. It is essentially a hand-operated fuel valve with an overriding governor to prevent

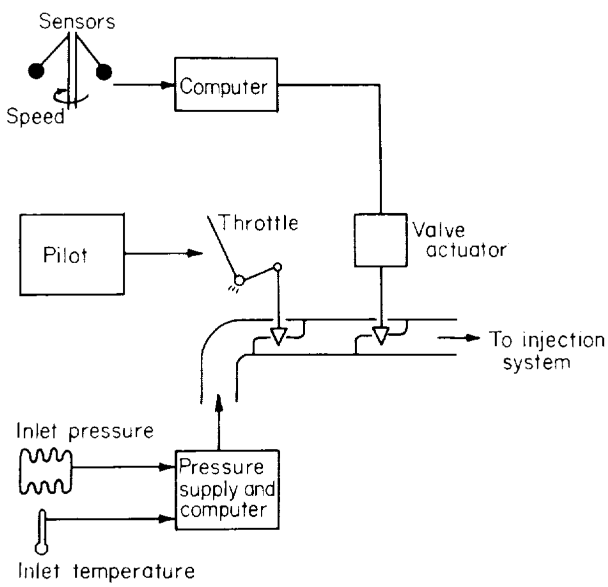


FIGURE 9.—Turbojet-engine control system employing manual fuel throttle with altitude-compensated supply pressure and automatic overspeed protection.

overspeed. The fuel-supply pressure is made proportional to altitude pressure to maintain a substantially constant engine speed at a fixed throttle setting.

This type of control system was used on engine A of table II and has a better record of reliability than the more complex systems. However, the control fails to provide acceleration control with protection against compressor stall and surge. Its record for reliability was accumulated in service on centrifugal engines in which compressor stall is less frequently encountered and is of less serious nature than in axial turbojet engines.

SPEED CONTROL WITH PRESSURE-SCHEDULED ACCELERATION LIMIT

This method of control provides surge and stall protection by limiting the maximum fuel-flow rate in relation to compressor-discharge pressure and compressor-inlet pressure and temperature.

These measurements are simple to obtain, but the schedule based on these parameters imposes an unnecessarily severe limit on engine acceleration under many conditions of operation. The reason for this is that the stall limit is a nonlinear function of pressure ratio and is therefore a nonlinear function of compressor-discharge pressure as well. Furthermore, the deviation of the stall limit from a linear relation with compressor-discharge pressure varies with the inlet pressure and hence with the altitude. As shown in figure 10, a linear schedule based on compressor-discharge pressure imposes a severe penalty on acceleration at low and high values of compressor pressure ratio.

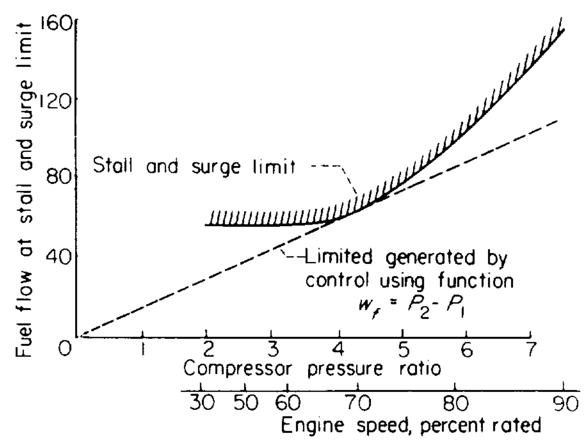


FIGURE 10.—Turbojet-engine stall-limit correlations. w_f, fuel flow; P_1, inlet pressure; P_2, compressor-discharge pressure.

The use of a maximum-fuel-flow limit based on compressor-discharge pressure creates a problem with regard to engine starting at high altitude. At high altitude the compressor-discharge pressure may be sufficiently low to cause the limiting of fuel flow below that required to obtain ignition. In this case, the engine could not be started following flameout at high altitude. As a consequence, it is necessary to add an additional control component to provide a minimum-

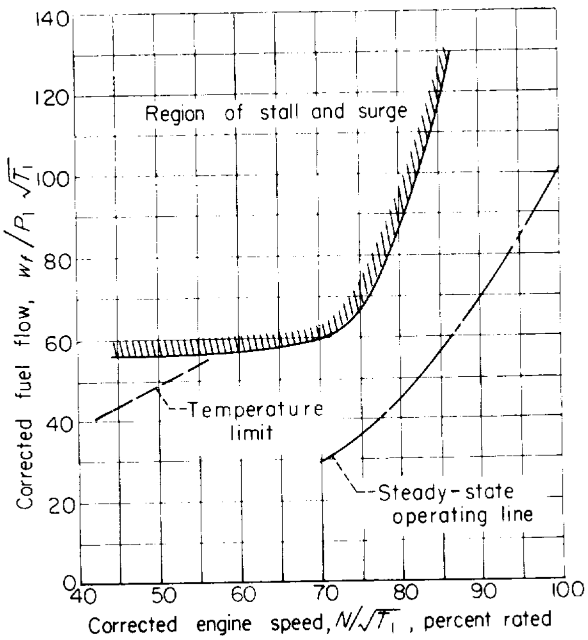


FIGURE 11.—Generalized map of stall- and surge-limited fuel flow. w_f, fuel flow; N, engine speed; P_1, inlet pressure; T_1, inlet temperature.

fuel-flow limit which overrides the surge control at very high altitude. The minimum-fuel-flow limiter, in turn, imposes a new complication in that it could, through malfunction or misadjustment, cause overspeed at high altitude. An engine caught between coinciding or overlapping maximum- and minimum-fuel-flow limits cannot be controlled by the pilot except by emergency means. Numerous cases of this occurrence appear in the Air Force records.

GENERALIZED ACCELERATION SCHEDULE

The auxiliary scale drawn in figure 10 shows that the surge limit can be correlated with engine speed. A characteristic form of this function is given in figure 11. As shown in the figure, a temperature limit exists which is lower than the surge limit in the lower-speed range. A control that operates on a combined schedule, as shown in figure 11, can provide both overtemperature and surge protection during acceleration. Such a control is more complex in practice than the control based on compressor-discharge pressure but provides protection with less loss of potential engine acceleration. Figure 12 shows a schematic diagram of this type of control system.

The generalized correlation of surge-limited fuel flow with engine speed is seriously affected by the distribution of the air velocity at the engine inlet. Distortion of the inlet-air velocity profile reduces the surge-limited fuel flow below

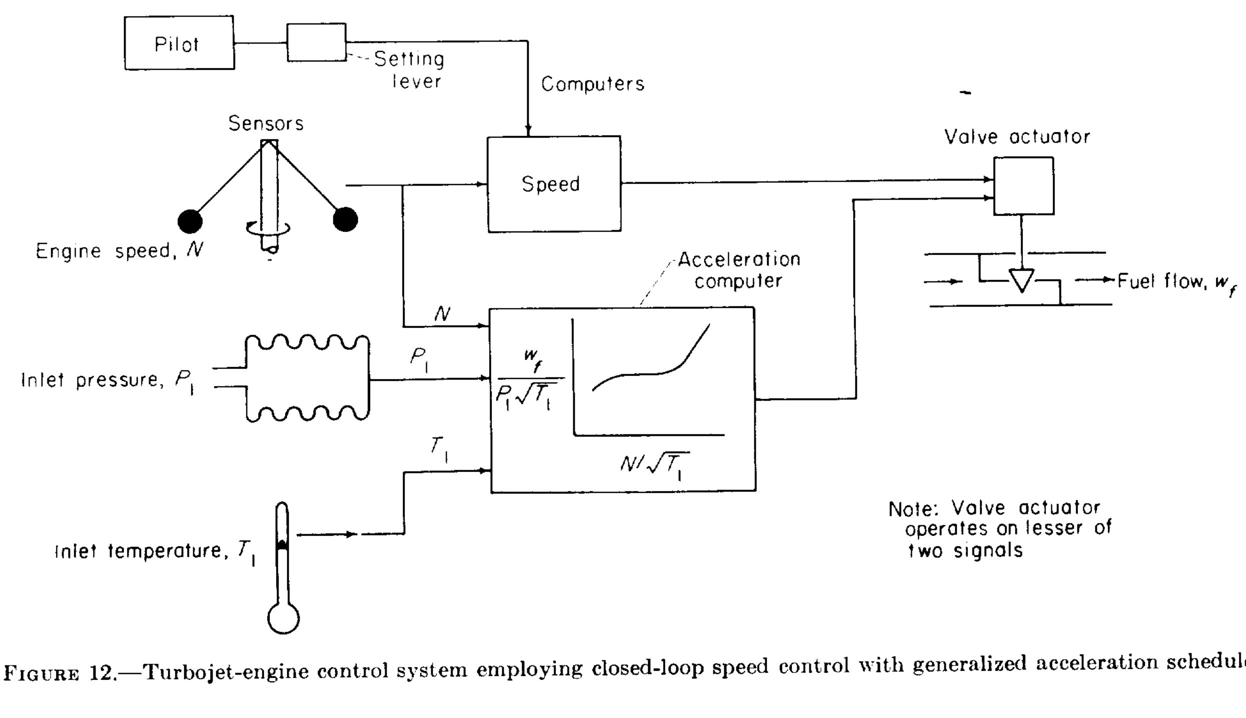


FIGURE 12.—Turbojet-engine control system employing closed-loop speed control with generalized acceleration schedule.

the values described by the generalized correlation. Surveys of engine installations in several aircraft have revealed many cases of inlet distortion of sufficient magnitude to cause a reduction in the excess fuel flow for acceleration, as limited by surge, by 50 percent. A characteristic variation of this inlet-air-distortion effect is shown in figure 13.

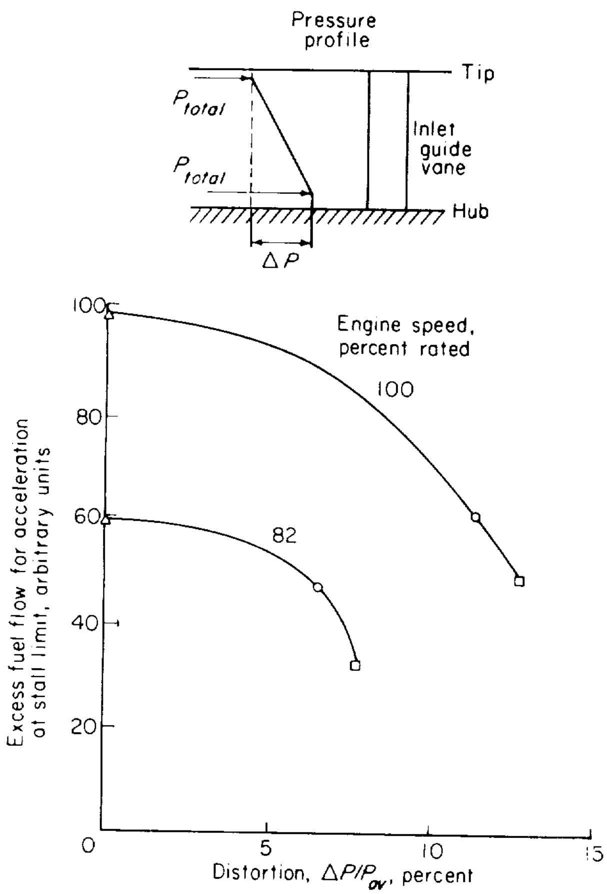


FIGURE 13.—Effect of inlet-air distortion on acceleration fuel-flow limit.

Inlet-air distortion is created by the ducting from the nacelle opening to the first compressor stage. It may therefore be presumed that the problem is one that may be largely solved by improvements in ducting design. This may be realizable in slower aircraft such as transport aircraft. However, in high-speed aircraft inlet-duct airflow is very sensitive to flight speed and angle of attack. Therefore, it is very difficult to eliminate inlet-air distortion by duct design for all flight conditions.

Another factor that seriously affects the correlation shown in figure 11 is the period of time between deceleration and acceleration. Engines which are cycled through several successive accelerations and decelerations show a reduction in the excess fuel flow for acceleration as limited by surge by as much as 50 percent. This cycling can occur during landing or flight refueling maneuvers. The effect does not take place if the period between the deceleration and the acceleration is longer than approximately 15 seconds.

Inlet-air distortion and acceleration cycling are factors which are random in nature. It is therefore not possible to correlate the effects to measurable parameters. In order to avoid stall and surge it is therefore necessary to limit fuel flow to account for the worst possible conditions. This procedure imposes a loss in available engine acceleration when the hazardous conditions do not exist.

Because of these difficulties with schedule-type acceleration controls, attempts have been made to devise methods whereby the threshold of stall or surge may be detected. If this can be achieved, acceleration controls can be devised which do not unnecessarily limit acceleration under any condition (ref. 4).

TEMPERATURE CONTROL

In the section OVERTEMPERATURE it is shown that, in the case of the fixed-exhaust-nozzle engine flying below 40,000 feet, control of engine speed simultaneously controlled turbine-discharge temperature during equilibrium running. The addition of surge-prevention schedules to speed control can simultaneously provide overtemperature protection during acceleration. These two factors have made practical the use of engine controls which do not directly measure or control turbine-discharge temperature. The widespread use of speed controls with scheduled-type protections against surge and overtemperature has probably discouraged the development of direct control of temperature. Nevertheless, the direct control of temperature provides the only positive means of providing protection against overtemperature damage under all conditions of operation.

A surge-limit correlation exists between generalized turbine-discharge temperature and engine speed (ref. 5). The characteristic form of this

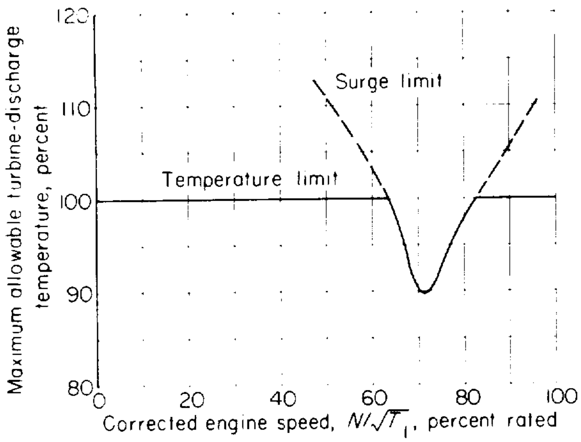


FIGURE 14.—Characteristic temperature- and surge-limit correlation.

correlation is shown in figure 14. It therefore appears possible to obtain surge prevention through temperature control according to a correlation curve such as figure 14. This factor combined with the factor of positive protection against overtemperature will make the development of temperature controls an important step toward improved engine reliability.

It is not practical to consider speed control by control of turbine-discharge temperature because of two factors: (1) the narrow margin between maximum rated speed and stress-rupture speed, and (2) the insensitivity of turbine-discharge temperature to speed changes at low engine speeds. The combination of direct overspeed protection with surge and overtemperature protection through temperature control appears to be a very desirable control method. However, the development of controls based on this method must await the development of an adequate temperature sensor.

BASIC RELIABILITY OF CONTROL METHOD

Automatic control systems can be divided into two general groups, that is, open-loop systems and closed-loop systems. Both types of control systems and combinations of the two types have been employed in turbojet-engine service. Figure 9 shows a control that operates as an open-loop control at speeds less than maximum speed. In this system the altitude-compensated pressure supply varies the throttle inlet pressure in response to air pressure and temperature parameters relating altitude and ram pressure. The variation is made in accordance with a preset schedule to maintain engine speed at a substantially fixed value at fixed throttle setting. Factors such as nonlinear schedule requirements, engine deterioration, and internal friction in the control elements introduce large inaccuracies in speed control by this method. Because of the extreme importance of speed control in the turbojet engine, such controls must be constantly trimmed by the pilot and must be protected by an overspeed governor.

In the closed-loop control method a particular variable, such as engine speed, is continuously measured and a controlling variable, such as fuel flow, is automatically varied to maintain a value of engine speed that is set by the pilot. Such a system is shown in figure 12. The closed-loop control can be extremely accurate and is not affected by nonlinear fuel-flow requirements, engine deterioration, or altitude and ram-pressure effects. The closed-loop system is, however, subject to complete breakdown in a phenomenon known as unstable oscillation. The susceptibility of the control to oscillation is often increased by steps taken to improve control accuracy over a wide range of flight conditions. Nevertheless, proper design of control-system and fuel-system components to obtain the necessary dynamic responses can make this type of malfunction extremely rare.

In the closed-loop system, overspeed protection is obtained by simply incorporating a maximum limit on set speed. An open-loop system being controlled by the overspeed governer is essentially a closed-loop system with a fixed value of set speed.

FUEL-SYSTEM-COMPONENT CONFIGURATIONS

The control components and the fuel-system components make up what is generally considered as the fuel system of the turbojet engine. In its broadest scope a discussion of fuel-system components could include such components as pumps, valves, fittings, tubing, and filters in addition to fuel-injection components. Aside from the fuel-injection components there is no component in the turbojet-engine fuel system that is not frequently found in other hydraulic systems. For this reason the following discussion is limited to fuel-injection components.

With regard to both inherent performance and reliability, the problems associated with fuel-injection components on the turbojet engine stem from flow range. Operation of a turbojet engine over the range of conditions extending from high-speed flight at low altitudes to gliding descent at high altitudes requires control of fuel-flow rate over a range of approximately 100:1. The successful operation of an engine over this range of conditions is, in a great measure, dependent upon the ability of the fuel-injection components to deliver a properly atomized fuel spray to the burners and to maintain equal flow rates among the several burners.

Fixed-area fuel nozzles.—The problems associated with liquid atomization over wide flow ranges have led to several different types of fuel-atomizing nozzle. The simplest nozzle is the fixed-area swirl nozzle, which is shown schematically in figure 15.

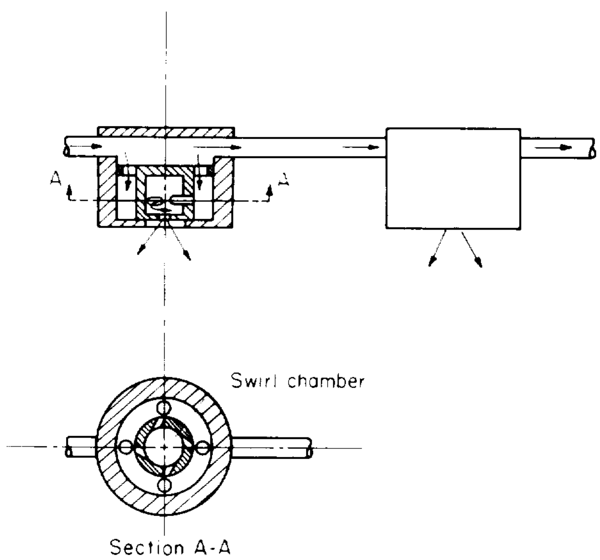


FIGURE 15.—Schematic diagram of fixed-area swirl-nozzle injection system.

The nozzles are distributed around the engine burner system and are connected by a common tube. Fuel is metered into a swirl chamber through tangential passages. The rotating fuel then discharges through a circular opening to form a hollow conical sheet of spray. In order to develop the conical spray, a fuel pressure drop of at least 10 pounds per square inch is required. Because of this minimum pressure requirement and because the pressure drop through the nozzle increases as the square of the flow rate, the maximum flow range is limited to about 10:1. The fixed-area nozzle is a very reliable device, but because of its limited flow range cannot be utilized in high-altitude service. The fixed-area fuel nozzle imposes another price for its simplicity in the form of difficult starting. For these two reasons the fixed-area fuel nozzle has been abandoned in turbojet-engine practice.

Multiple-entry nozzles.—A number of designs of multiple-entry fuel nozzles have been developed which overcome, in various degrees, the shortcomings of the fixed-area nozzles with regard to both range and starting. A form of a double-entry nozzle widely used is shown schematically in figure 16.

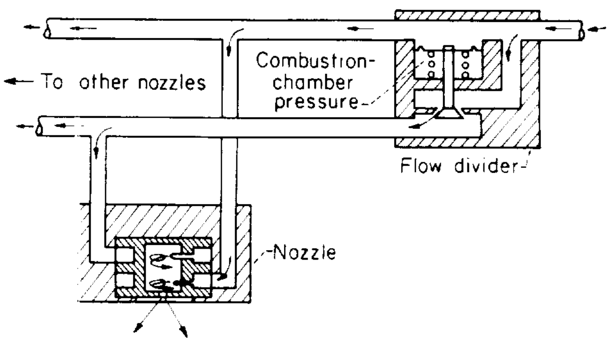


FIGURE 16.—Schematic diagram of double-entry swirl-nozzle injection system.

As may be seen, this system requires an additional component known as the flow divider and an additional manifold and associated pipe connections. The fuel nozzles have no moving parts. At low flow rates the flow divider transmits flow to the small tangential openings in the swirl chamber. When the flow is increased beyond a preset value, the fuel pressure opens the flow-divider valve and fuel is transmitted to the large tangential openings. By this means a high swirl velocity is obtained at low flow rates, and excessive pressures are avoided at high flow rates.

The flow divider is actuated by fuel pressure, and therefore the rate of flow discharged into the engine at a given inlet pressure is subject to the functioning of the flow divider. Malfunctioning of the flow divider can be very serious. If the malfunction is such that the flow is high for a given pressure, fuel atomization becomes poor with the consequent loss of fuel economy. Under certain conditions blowout may result. If the malfunction of the flow divider is such that the flow is low at a given inlet pressure, it may not be possible to attain rated engine speed at low altitudes.

Uniformity of fuel distribution among the several nozzles in the system is obtained by maintaining close manufacturing tolerances on the nozzle dimensions and by periodic flow bench calibration of the individual nozzles. A serious error in distribution can develop in this system if the small tangential openings of one or more of the nozzles become partially or entirely clogged with particles carried by the fuel. As shown in figure 16, the pressure developed on the walls of the swirl chambers is transmitted backward to the manifold. If differences exist in the pressures at the walls of the swirl chambers, a flow will result in the manifold. By this action, fuel metered into one swirl chamber can be discharged from another. If a small tangential opening is seriously clogged, there will be little pressure developed on the chamber walls and the nozzle will run very rich at the expense of the others. Burner damage is very likely under such conditions.

A method of preventing interflow in double-entry nozzle systems that has been adopted by one engine manufacturer employs separate flow dividers for each nozzle. The flow divider is built into the nozzle housing. This system eliminates one manifold and associated fittings but involves a considerable increase in the number of moving parts. Flow-divider characteristics must be carefully matched for uniform distribution.

OPERATING CONSIDERATIONS

EMERGENCY PROVISIONS

It is general practice in military turbojet aircraft to provide means for emergency operation of the engine in the event of control or fuel-system-component failure. There is little question that this practice is justified, but the degree to which it is implemented varies greatly. The sim-

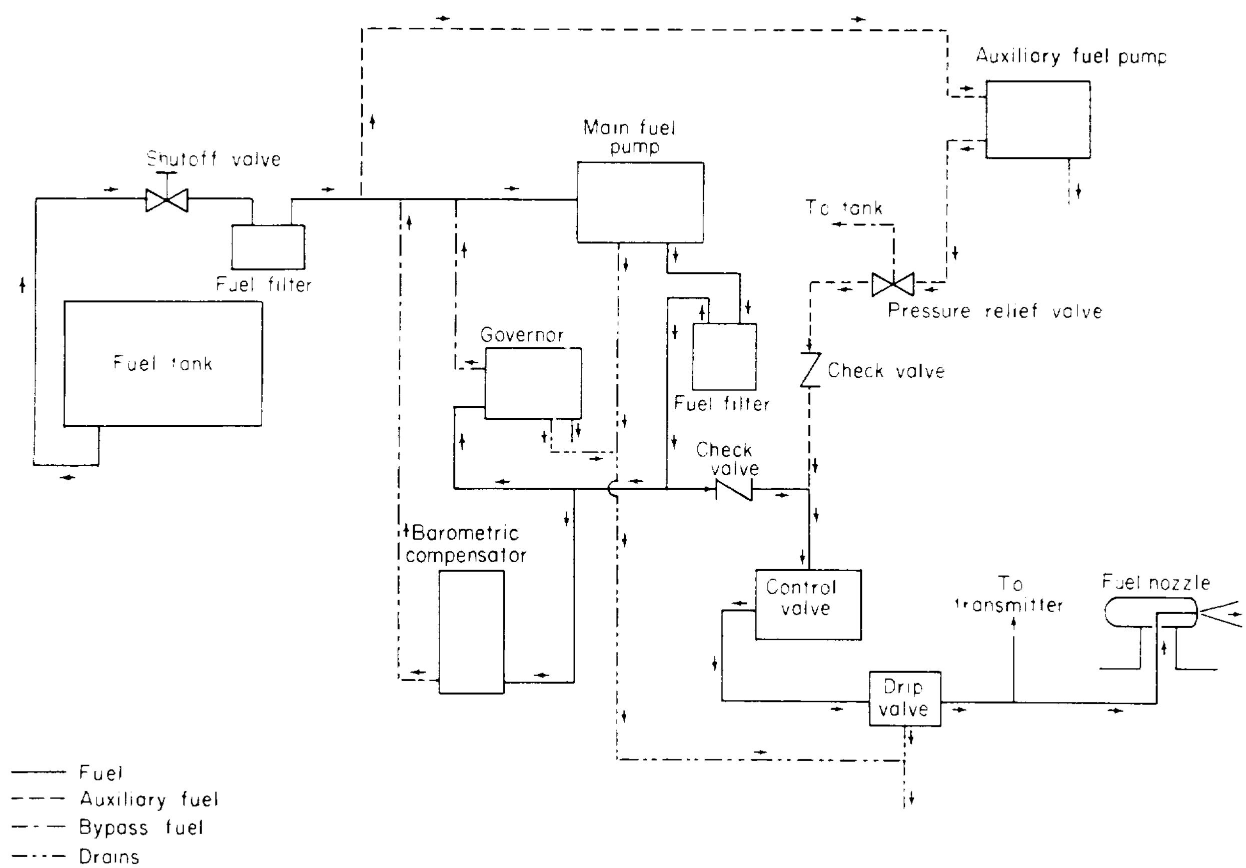


FIGURE 17.—Turbojet-engine fuel system employing emergency fuel pump.

plest emergency control is a manual fuel throttle that completely overrides all other control components. The most complex emergency control system comprises a complete duplication of the primary control system. Emergency fuel pumps are usually carried by military turbojet aircraft, but it is not feasible to install emergency units for other fuel-injection components.

In spite of the obvious advantage of the existence of emergency controls in an instance of failure, there are two objections to the practice. The first objection relates to the increased complexity of the control system created by the emergency controls. The second objection relates to the need for transfer valves, check valves, and the like required to put the emergency system into operation. Figure 17 shows an example of this type of complication which occurs in the case of a fuel system that employs an emergency fuel pump. In the figure, the components added by the emergency system are the following: (1) auxiliary fuel pump, (2) pressure-relief valve for auxiliary pump, (3) check valves, (4) porting in control valve for selecting main or auxiliary pump, and (5) required piping and connection.

Emergency controls can be effective only if failure of the primary control does not cause destructive damage to the engine. Destructive damage through control failure can usually be avoided if the control is designed to reduce engine power and temperatures in the event of the most likely failures or malfunctions. This "fail-safe" provision is commonly provided in present turbojet-engine controls.

Another means of avoiding destructive damage through control malfunction is through the use of sufficient instrumentation or by means of a warning system whereby the pilot can detect impending control failure.

A second requirement for effective use of emergency controls is pilot training for operation with emergency controls. This is especially true when the emergency control system does not carry out as many functions as the primary control system.

GROUND MAINTENANCE

Ground maintenance of engine-control components and fuel-injection components involves two functions. The first of these functions is inspection; the second function is adjustment.

The purpose of inspection is to prevent failure by replacement of worn or weakened parts. Aside from the main fuel pump, there appear to be few fuel-system or control components that are subject to wear-out failure. For this reason, techniques must be developed for detection of components in which failure is impending.

The detection of impending failure can often be accomplished by means of bench testing. Test benches are in use for inspection and adjustment of fuel nozzles, overspeed governors, surge-prevention components, and barometric scheduling devices. A discussion of bench testing of turbojet-engine fuel controls is given in reference 6.

In most cases, bench tests of control-system components are concerned with steady-state calibration. This is an important and necessary test but is not in itself a complete test. On the engine, the manner in which the control components respond to rapid changes in power level and flight conditions is also extremely important. For example, in order to ensure proper performance of such components as surge-protection devices, it is necessary to duplicate engine acceleration on the test bench. A test bench for dynamic testing of turbojet-engine controls has been recently developed and is reported in reference 7. In the case of closed-loop control systems, dynamic bench testing can be used to check control stability.

Most fuel-injection and control components are affected by solid particles in the fuel. Effective filtration and regular replacement of filters are therefore a very necessary maintenance activity. Impending failure due to dirt accumulation can very often be detected in the process of bench testing.

PRETAKEOFF INSPECTION

In spite of the most rigid ground maintenance and inspection, pretakeoff inspection by the pilot remains as a very important activity. Many control and engine malfunctions can be detected with relatively simple instrumentation during ground operation of the engine. Table IV shows the types of malfunction that can be detected and the means by which the detection is made. The malfunctions listed in table IV are limited to those of a general nature. In the case of a specific control, it is likely that many other indications can be used to detect other specific malfunctions.

TABLE IV.—MALFUNCTIONS INDICATED DURING GROUND OPERATION

Malfunction	Indication
Governor	High top speed
Main control	Low maximum fuel flow
	Slow acceleration to top speed
	Surge during acceleration
High-pressure fuel leak	High fuel flow at top speed
Fuel-injection component	High fuel flow at top speed
	High fuel-pump discharge pressure at top speed
	Low fuel-pump discharge pressure at top speed
	High tailpipe temperature at top speed
	Irregular circumferential temperature distribution in tailpipe

ADDITIONAL INFORMATION NEEDED FOR IMPROVEMENT OF OPERATIONAL RELIABILITY

At the present time the need for improved means of acceleration control is the most pressing engine fuel-control problem. Because of this, engine and control manufacturers and associated research laboratories are intensively investigating surge and stall. As was pointed out in the discussion of surge and stall prevention, present acceleration controls operate on the basis of fuel-flow-limiting schedules. Because of the many factors that affect this schedule, a considerable margin of safety must be employed. This safety margin unnecessarily limits the acceleration rate under many conditions of operation. Limited acceleration during combat or landing wave-off can result in destruction of the aircraft. It is generally believed that basic information on the fundamental nature of surge and stall, now being obtained, will permit the design of acceleration controls which will permit maximum acceleration rates at all conditions.

The need for direct control of temperature is not as pressing as the need for improved acceleration control, but there is little question that the development of a high-temperature sensor having both long life and good dynamic response will lead to methods of control that will result in improved operational reliability.

The general field of automatic control is now developing very rapidly. It is very likely that improved analytical techniques in control design and improved design and construction of basic automatic control components, which will be developed, will lead to greater reliability and improved performance of the turbojet engine.

CONCLUDING REMARKS

The results of this study of the effect of engine fuel-control reliability on the operational reliability of the turbojet engine have shown that control malfunction can cause either immediate destruction of the engine, serious reduction in engine power, or reduced life of engine components. Statistics show that at the present time the frequency with which control malfunctions occur is serious.

At the present time the need for improved means of acceleration control is the most pressing control problem. Loss in engine power due to improper fuel-flow limiting by inadequate or malfunctioning acceleration controls is as serious a problem as direct damage through control malfunctions.

Improved control in the fields of fuel atomization, starting, flameout protection, and temperature control can result in significant increases in turbine and burner life. However, while improved and expanded control operation can yield improvements in both engine performance and component life, increased control complexity increases the probability of control failure. Increased control complexity must therefore be offset by improved reliability of control components, and facilities must be provided for maintenance and inspection of complex fuel controls.

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CHAPTER XI

SUMMARY DISCUSSION

By Benjamin Pinkel

SUMMARY

This chapter summarizes the results of the studies in the other chapters of this report. An attempt is made to treat under common headings the difficulties in the various components of the turbojet engine encountered from similar causes and the methods of handling these difficulties that are similar.

The reliability of the turbojet engine can be improved by increasing the endurance of its components and by providing for the removal of critical components from the engine before their failure jeopardizes the engine. The basis for the removal of engine components must come from an understanding of their failure causes and characteristics. The methods of increasing engine reliability are discussed, and the additional information needed to more effectively handle the reliability problem is summarized.

INTRODUCTION

Examination of the individual studies indicates that the causes for component failures come under the following headings:

(1) Underdesign and defects in materials and manufacture
(2) Wear
(3) Creep
(4) Fatigue
(5) Foreign-object damage
(6) Overtemperature and overstress
(7) Thermal distortion and thermal fatigue
(8) Corrosion and stress-corrosion
(9) Fuel-control malfunctions

Some component failures result from weaknesses that can be corrected by application of standard methods in design and manufacture, and they should eventually be eliminated. These failures fall under item (1). The next three items, wear, creep, and fatigue, can cause progressive deterioration in the strength of some of the engine components and eventual failure of these components. The times to failure by these mechanisms can be roughly predicted from the data on the affected components when they are operated under well controlled conditions. The remaining items listed, and in some cases fatigue also, can cause unpredictable and early failure of engine components. The various causes of failure listed and the components involved are discussed. Failure of engines by environmental causes, such as icing and dust erosion, may occur. However, these problems are not studied in this report.

Methods for the improvement of operational reliability are discussed under the following headings:

(1) Design
(2) Inspection in manufacture
(3) Inspection in service
  (a) Scheduled by time
  (b) Scheduled on basis of contingency
  (a) Flight checks and warning devices
(4) Replacement schedules
(5) Records
(6) Operational and repair practices

The additional information needed to improve operational reliability presented in the individual papers is summarized.

REVIEW OF CAUSES FOR ENGINE AND COMPONENT FAILURES

Before the review of the causes of engine failure, the findings presented in chapter II relating to component failures experienced in military service are briefly summarized.

STATISTICAL STUDY

Disassembly Inspection Reports are prepared by inspectors at overhaul bases and report (1) the primary reason for the engine going to overhaul

TABLE I.—REASONS FOR OVERHAUL OF ENGINES

[Aug. to Oct., 1953]

Failures associated with—	Percent of engines overhauled				
	Engine code				
	C-7	A-7	B-3	B-9	B-10
Compressor	8	9	12	4	5
Blades	.5	5.9	2.7	0	0
Stators	1.4	.7	4.1	0	.6
Disks	0	.7		1.4	1.2
Casing	4.3		--	0	0
Miscellaneous	1.4	1.5	5.5	2.7	3.1
Main bearings	7	5	0	7	4
Accessory bearings	1	0	0	8	1
Combustor assembly [a]	5	0	10	0	1
Outer shell	3.8	0	0	0	0
Inner shell	.5	0	0	0	0
Transition liner	0	0	9.6	0	.6
Fuel nozzle	.5	0	0	0	0
Miscellaneous	0	0	0	0	0
Turbine section [a]	11	1	16	0	3
Nozzle diaphragm	9.5	.7	15.1	0	.6
Buckets	.5	.7	1.4	0	1.9
Disks	0	0	0	0	0
Inner gas baffle	0	0	0	0	0
Miscellaneous	1.4	0	0	0	.6
Accessories	11	1	7	3	7
Foreign objects [b]	30	57	26	59	42
Other causes [c]	28	27	29	19	39
Mean time to overhaul, hr	305	160	247	55	88
Sample size	210	136	73	73	161

[a] Does not include foreign-object damage.
[b] Damage was mainly in the compressor.
[c] Technical-order compliance, overtemperature, overspeed, crash, and accident.

and (2) the additional components requiring repair or replacement discovered during overhaul. The primary reasons for overhaul and the operating time on the average engine in overhaul are shown in table I.

Additional failures found on disassembly are shown in table II. An extremely large percentage of failures is attributed to foreign-object damage. The large number of parts that require replacement after relatively short operating times is also apparent.

Because of the present policy of repairing hot-section components in the field, they are infrequent causes for engine overhaul. The frequency of engine repair precipitated by hot-section component failure is indicated by data obtained from the Field Maintenance and Repairs Summaries (table III). Turbine shaft bearings are also inspected and replaced in field repair.

Table III refers specifically to engines sent to field repair. Most engines in service receive field

TABLE II. ENGINE PARTS REPLACED DURING OVERHAUL

[Aug. to Oct., 1953]

Engine part replaced	Percent of engines overhauled				
	Engine code				
	C-7	A-7	B-3	B-9	B-10
Compressor [a]	22	12	15	5	7
Blades	.5	6.6	4	1.4	0
Stators	9	.7	4	0	.6
Disks	0	1.5	0	1.4	1.9
Casing	8.1	0	1.4	0	1.2
Miscellaneous	6.2	4.4	6.9	2.7	3.7
Main bearings	39	31	60	29	24
Accessory bearings	64	2	29	11	2
Combustor assembly [a]	44	57	26	3	25
Outer shell	5.2	2.2	0	0	.6
Inner shell	43.3	55.9	16.4	2.7	23
Transition liner	0	.7	16.4	0	1.2
Fuel nozzle	.5	2.9	0	0	0
Miscellaneous	0	.7	0	0	.6
Turbine section [a]	87	49	48	4	14
Nozzle diaphragm	85.7	42.7	35.6	1.4	8.7
Buckets	1.4	7.4	28.8	2.7	5.6
Disks	0	.7	0	0	0
Inner gas baffle	3.8	0	0	0	0
Miscellaneous	11	1.5	2.7	0	1.2
Accessories	14	9	8	10	12
Foreign objects	84	85	33	73	66
Compressor [b]	50.3	81.1	31.5	69.7	62.7
Blades	49.3	83.4	28.8	69.7	62.1
Stators	32.3	72.1	28.8	67.6	55.9
Casings	1.9	1.4	1.4	0	1.8
Miscellaneous	24.2	14.7	2.7	0	1.8
Turbine section [b]	75.4	52.4	16.4	43.4	43.5
Nozzle diaphragm	66.8	22.4	2.7	34.2	31.1
Buckets	70.6	49.6	16.4	43.4	40.4
Mean time to overhaul, hr	305	160	247	55	88
Sample size	210	136	73	73	161

[a] Does not include foreign-object damage.
[b] Foreign-object damage.

TABLE III.—ENGINE COMPONENT REPLACEMENT IN FIELD

[April to June, 1953]

Engine component replaced	Percent of engines in field repair — Engine code A	B	C
Nozzle diaphragm	55	38	31
Turbine bucket	22	10	25
Turbine disk	36	8	17
Inner liner	16	55	34
Transition liner	----	3	33
No. 3 bearing	22	3	3
No. 4 bearing	36	3	6
No. 4 oil seal	13	1	7
Compressor rework [a]	1	18	19
Sample size	322	340	1, 261

[a] Such as "stoning out" of minor nicks in compressor blades.

repair at least once, and some engines are repaired many times between overhauls.

In 1953, 205 airplane accidents were attributed to engine malfunction or failure, of which approximately 50 percent resulted in destruction of the airplane. The causes for these accidents are listed in table IV. These data were taken from statistics prepared by the USAF Directorate of Flight Safety Research. Although no failures are attributed to foreign-object damage, the compressor is charged with a high percentage of the failures. It is suspected that the principal cause for compressor failure is foreign-object damage.

TABLE IV.—CAUSES OF ENGINE MALFUNCTION OR FAILURE THAT RESULTED IN AIRCRAFT ACCIDENTS

[During 1953]

Failure	Number of accidents	Percent of accidents
Fuel controls	68	33
Compressor	54	26
Turbine bucket	16	8
Turbine disk	14	7
Bearings	10	5
Miscellaneous (16 different causes)	20	10
Undetermined	23	11
Total	205	100

UNDERDESIGN AND DEFECTS IN MATERIALS AND MANUFACTURE

In an effort to obtain engines of low weight per unit thrust, components are designed with small margins of safety. Heterogeneous failures are usually experienced in the early models and are generally eliminated by improvements in design and manufacturing methods. In this category are failures from inadequate design, defective materials, improper heat treatment, flaws in castings, incorrect forging practice, improper welds, and inclusions in bearing materials. These problems have been under extensive study by the engine manufacturers, but they have received little study at the NASA and are discussed only to a limited extent in this report.

WEAR

Wear, in this discussion, refers to the deterioration of rubbing surfaces. Although wear may be a problem in some of the control equipment and fuel-system components, discussion herein is limited to the bearings of the main rotor. Wear occurs principally on the surfaces of the cage in contact with the race and the rolling elements.

Because of the relatively low cost of bearings and the possibility of catastrophies resulting from bearing failure, bearings are replaced in field repair and overhaul on indication of only minor flaws. Hence, in current engines, bearings do not usually run for sufficient time to require replacement for normal wear.

Abnormally high wear rates have been encountered when bearing surfaces are overloaded, the flow of lubricant to the surfaces has been interrupted, or dirt and other abrasives have entered the lubricant. These difficulties should be eliminated by design changes and better handling of the lubricant.

When other bearing difficulties are eliminated, and when in the interest of economy it is desired to operate bearings for longer times, then replacement of bearings for normal wear will be required. Bearings can be inspected for wear on the basis of measured changes in clearance (see ch. IX).

CREEP AND STRESS-RUPTURE

Under a steady load at high temperatures, materials tend to creep or deform with time and eventually to break. The time to failure is called the stress-rupture life of the material. It depends on the material, the stress, and the temperature. Creep and stress-rupture life are of concern in the turbine disk and buckets, which are exposed to a combination of centrifugal force and high temperature. The attendant problems are discussed in the chapters on these components.

In some engines, many failures of turbine buckets by stress-rupture have been encountered in service. A variation in stress-rupture life, of as much as 3:1 between the longest-lived and shortest-lived bucket on the same turbine, has been encountered.

This is attributed to small variations in material composition, structure, and fabrication history, which occur in spite of present efforts to control these factors.

Stress-rupture life is very sensitive to temperature and stress; small reductions in these factors can cause large increases in stress-rupture life. The stress-rupture life is the longest life the buckets can have. The actual life can be reduced by such factors as overtemperature, thermal and mechanical fatigue, impact by foreign objects, and vibration.

In the case of the turbine disk, the rims are generally cooled by air bled from the compressor in order to achieve desired low creep rates and high stress-rupture lives. Indications of creep under the influence of the high centrifugal forces encountered are increases in rim diameter and deformation in the serrations which hold the turbine buckets. Failure by stress-rupture has rarely been observed in disks which experienced no damage from other causes. However, failures have occurred when the disk was overheated, and also when the disk material was weakened by improper manufacture. Failures by stress-rupture under normal conditions may become a problem when attempts are made to obtain longer operating times from a given disk than is the current practice in military service.

FATIGUE

Periodic forces resulting principally from regular disturbances in the gas flow through the engine cause vibration of such items as turbine buckets, compressor vanes and blades, and various sheet-metal parts exposed to the gas flow. When the frequency of the vibratory force is in resonance with a natural frequency of a component and the damping losses are small, high stresses can result which may cause early failure of the component. Because of the difficulty of computing the magnitude of these factors in the initial design, the magnitude of the resultant vibratory stresses cannot be predicted. The vibratory stresses are usually investigated during the development program on an engine.

The most serious cause of vibration in compressor blades is rotating stall. The mechanism whereby rotating stall is set up is discussed in chapter IV. Rotating stall occurs at engine speeds below 70 percent of rated speed at low flight Mach numbers and is usually encountered during acceleration of the engine through this speed range. It can occur at higher rotative speeds at higher flight Mach numbers. Rotating stall has been the cause of a number of compressor failures in service.

Disturbances in the gas flow caused by the wakes of the nozzle vanes or by the spatial variation in the velocity of the gases issuing from the individual burners impose periodic forces on a given turbine bucket as it passes repeatedly through these disturbances. In some engines the vibratory forces are large and cause failure of the blades in a small fraction of their design life. Sheet-metal parts and other parts of the engine are likewise susceptible to failure by vibration. Even less is known about predicting the vibratory stresses of these parts than of compressor and turbine blades.

The rotor thrust bearing, which is a ball bearing, is also subject to fatigue. Material in the vicinity of the surfaces of the balls and races of the bearing is periodically subjected to a high force during rotation of the bearing. Since the balls and races are in nearly point contact, the stresses are very high. Damage from fatigue originates as a small spalled area on the ball or race. It usually progresses to complete failure of the bearing in times considerably less than any practical time between bearing inspections. The failure of a bearing can be catastrophic. Hence, protection against this type of failure requires a replacement schedule and is discussed later.

FOREIGN-OBJECT DAMAGE

Foreign-object damage is found in chapter II to be the principal cause for sending engines to overhaul. The foreign objects comprised mainly debris drawn into the engine inlet from the airport surface, debris left in the engine, and other objects which have broken loose within the engine (see ch. III). These objects, when passing through the engine, nick or break compressor and turbine blades. Nicks act as nuclei for fatigue failure and can shorten the lives of compressor blades or turbine buckets. Broken compressor blades usually cause a chain reaction of compressor blade failures in passing through successive compressor stages. Compressor blade failure is one of the most important causes of flight accident. Broken turbine buckets in a single-stage turbine usually pass out the engine exhaust nozzle without any appreciable damage to the engine, although there are cases where a broken turbine bucket has caused flight accidents. In chapter II it is pointed out that out of 205 accidents, which resulted from engine malfunction, 16 were attributed to failure of turbine buckets. Turbine bucket failure in a multistage turbine is much more serious than in a single-stage turbine. A broken bucket in one of the early stages of a multistage turbine can destroy the buckets in the later stages and cause engine stoppage.

The statistics quoted on foreign-object damage were obtained on engines that for the most part were equipped with inlet screens. The need is thus indicated for more effective screens or other devices for eliminating foreign objects. Some of the engines were equipped with retractable screens, which dumped collected debris into the inlet when retracted.

OVERTEMPERATURE AND OVERSTRESS

Overtemperatures have occurred during starting as a result of inadequate control by the pilot or failure of the automatic control system. Under these conditions, overtemperature lasts for only a short period of time and affects mainly the combustor liner and turbine nozzle vanes and buckets. Overtemperature can contribute to the deterioration of these components through buckling and thermal fatigue by increasing the temperature gradients and, also, in some materials through detrimental changes in the metallurgical structure. The Air Force Technical Orders call for inspection and overhaul of the turbine after a prescribed number of overtemperature events.

Overtemperature can also occur during acceleration of the engine if the automatic control is not functioning properly, particularly, if compressor surge is encountered.

At maximum engine speed, overtemperature is especially serious because it occurs simultaneously with high stresses in the turbine. As pointed out in chapter VII, a small amount of overtemperature can drastically reduce the design stress-rupture life of the turbine bucket. Overtemperature at this condition has been caused by drift of the automatic-control setting or by improper setting. In most controls, a maximum speed limit is used for regulating the maximum-thrust condition on the assumption that, when the maximum engine speed is held constant, the maximum temperature also remains constant. Chapter X shows that at altitudes above 35,000 feet the gas temperature of engines increased with altitude in spite of the fact that the engine speed was held constant. The variation of maximum temperature with altitude and flight speed for constant engine speed should, therefore, be determined for the specific engine under consideration before relying solely on maximum engine speed as a performance-limiting control.

When drift of the engine speed control occurs in an engine with a constant discharge nozzle area, then both overtemperature and overstress occur which shorten the life of turbine buckets and disks more drastically than overtemperature alone.

Local overtemperature has been caused by improper performance of the fuel nozzle or by blockage of the fuel spray by a carbon formation in the combustor. Local overtemperature hastens the failure principally of the combustor liner and nozzle diaphragm.

A very serious form of overtemperature results from failure of the turbine disk cooling system. Catastrophic failure of a turbine disk has been experienced, for example, as a result of overtemperature caused by warpage of the disk cooling baffles. The pilot has no warning of difficulty from this source.

THERMAL DISTORTION AND THERMAL FATIGUE

Thermal distortion and thermal fatigue are problems in such components as the turbine nozzles, buckets, and disks, combustor liners, and sheet-metal parts in the hot end of the engine.

Under steady operating conditions, large temperature gradients exist in the combustor liner which cause buckling of the liner and plastic flow. Repeated cycles of engine operation involving starting, accelerating, and stopping of the engine cause repeated plastic working of these areas and eventually cracking. Often these cracks start at a point of stress concentration such as a louver or an air intake hole in the combustor liner. Heat treatment to eliminate residual stresses introduced during fabrication will reduce the tendency for initiating cracks. Fortunately, cracks in the combustor liner progress slowly enough that pieces do not break out before it goes to repair, where cracked liners are often replaced. In only a few of the cases studied have liners come to overhaul with pieces broken out. In these cases the pieces passed through the turbine without breaking a turbine bucket. There was, however, some evidence of nicking of the buckets, which probably resulted in reduction of life. Distortion of the flame pattern in the combustor, either through malfunctioning of the fuel system or as a result of accumulation of the carbon on the nozzle, ignitor, or liner surface, creates hot spots and accelerates the thermal fatigue of the liner. These problems are discussed in detail in chapter V.

In some engines the nozzle diaphragm consists of two concentric rings with each nozzle vane welded at each end to these rings. Differences in temperature on this assembly, because of the nonuniform temperature at the combustor outlet, cause differential expansion and distortions. On repeated cycles of operation, cracks appear in the nozzle assembly because of the distortions. In an effort to provide for thermal expansion, the nozzle vanes in some designs are attached to one ring, and they slide in airfoil-shaped slots into the other ring. In nozzle diaphragms of this type, cracking of the rings has been encountered because of differential expansion between the ring and its support on the engine frame. These cracks usually originate at the trailing edge of the slot for the nozzle vane. At this point the slot has a sharp radius, which creates a stress concentration. Differential expansion in the engine frame can shift the support for the inner ring with respect to that for the outer ring. This causes mechanical stress in the nozzle diaphragm, which can hasten failure. Fortunately, the cracks in the nozzle assembly progress sufficiently slowly that they are usually detected in overhaul or on inspection, and they have been a negligible cause for accidents.

In the turbine disk under steady-state conditions, a temperature difference of the order of 600° F may exist between the rim and the hub sections. Expansion of the rim is prevented by the colder hub and web sections with the result that plastic flow occurs in the rim. During cooling of the engine at shutdown, the direction of the thermal stresses in the rim is reversed. With repeated cycles of engine operation the rim material eventually hardens and cracks. The cracks usually progress slowly and have been found on inspection of the disks in overhaul. The inception of rim cracks is hastened if scratches exist in the rim. Scratches have been made in the rim by shearing of the bucket retaining pins on removal of buckets from the turbine.

Large temperature gradients occur in turbine nozzle vanes and buckets during transient conditions such as startup, acceleration, and shutdown. The leading and trailing edges of the turbine nozzle vanes and buckets follow the gas-temperature variations more rapidly than do the bodies of these objects with the result that large temperature differences can exist momentarily. For example, in a normal start on one engine, a difference of 600° F was measured between the leading edge and the center of a turbine bucket about 3 seconds after ignition. The heavier body section of the bucket restrains the expansion of the leading and trailing edges with the result that plastic flow may occur in these areas. When the body eventually attains the equilibrium temperature, the direction of the thermal stress in the leading and trailing edges which have been plastically deformed is reversed. This phenomenon is repeated during successive cycles of engine operation and has caused warping and cracking in the leading and trailing edges of some nozzle vanes and buckets. These cracks do not cause nozzle vanes to break, and they are found in inspection during overhaul. Cracks of the type described have been found during inspection on turbine buckets in an engine in which the design stress-rupture life of the bucket was extremely long. In buckets having short stress-rupture lives, cracks of this type may cause rapid failure of the bucket.

CORROSION AND STRESS-CORROSION

The corrosive action of hot gases on components in the high-temperature end of the engine has forced the use of high-alloy materials. These materials are sufficiently resistant to corrosion that ordinary corrosion is not an important reliability problem. In the interest of using lower-alloy materials for combustor liners and other sheet-metal parts, coatings for protecting these components against corrosion are being investigated. In naval applications, corrosion by salt water is still an important problem. When leaded fuels are used, nozzle vanes and possibly turbine buckets are susceptible to corrosion by lead compounds.

Compressor blades made of AISI type 403 stainless steel, which contains approximately 12 percent chromium, have been found in some engines to crack at the leading edges by a stress-corrosion mechanism. Stress-corrosion cracks can form nuclei for fatigue failures and can shorten the fatigue life of the compressor blades. The tendency toward the formation of stress-corrosion cracks can be reduced by changes in the heat treatment of the material—unfortunately, with some sacrifice in strength of the material. Cracking of rims of turbine disks has been accelerated by the stress-corrosion mechanism. These problems are discussed in chapters IV and VIII.

FUEL-CONTROL MALFUNCTIONS

Of 205 flight accidents in 1953 attributed to turbojet-engine malfunction, the fuel control was charged with 68. By fuel control is meant the fuel and control systems. The principal difficulties associated with fuel controls in engines are (see ch. X)

(1) For engines in which the limiting of the fuel flow in the starting operation is in the hands of the pilot, overtemperatures are experienced in starting. The Air Force Technical Orders prescribe limits on the number of permissible overtemperatures. An appreciable number of engines are sent to overhaul because these limits are exceeded. A reliable method of gas-temperature sensing is needed for the development of automatic control of temperature.

(2) Current controls do not accurately follow the surge limits during acceleration. Several factors that make this difficult are: (a) The limiting fuel flow is a nonlinear function of engine speed and varies with compressor-inlet temperature and pressure; (b) the surge limit on the compressor is reduced by distortion of the air velocity profile at the compressor inlet; (c) the amount of inlet-flow distortion changes with angle of attack, flight speed, and altitude; (d) rapid and repeated accelerations and decelerations of the engine that occur when the controls are rapidly manipulated in some landing maneuvers likewise reduce the surge limit on the compressor. Too restrictive a limit on fuel rate during acceleration reduces the allowable acceleration rate. Too liberal limits result in surge, which in some instances has led to flight accidents. The alleviation of this problem involves improvement in the control methods, the engine, and the engine installation.

(3) Drift or improper adjustment of the control at the maximum-thrust condition has permitted engine overspeed and overtemperature. Engines have been sent to overhaul because of damage caused by overspeed and overtemperature. Also, some flight accidents have resulted from these causes. Engines are currently limited in maximum thrust by an engine speed governor. On a number of current engines studied, the gas temperature remained nearly constant with constant engine speed up to an altitude of approximately 35,000 feet and then increased rapidly with further increase in altitude. Hence, overtemperature at high altitude can occur with a control of this type even if the control is functioning as designed. The need for a practical temperature sensor and a maximum temperature control is indicated.

(4) Flameout during rapid deceleration of the engine in a landing maneuver may occur and can cause accidents. Flameouts can also occur

at high altitudes. Relighting of the engine at these conditions is very difficult, and in some cases where it has been accomplished explosions resulted which destroyed the engine. Improvements in fuel-nozzle design that would provide good atomization at low fuel rates would improve the blowout limits of the engine. More accurate duplication of these limits by the control for the wide variety of conditions in which flameout can occur is likewise required.

(5) Failures of control and fuel-system components by wearout, fracture, sticking, and clogging are an important cause of engine malfunction. Approximately 75 percent of the fuel-control malfunctions are attributed to component failure rather than inadequacy of the control method.

These difficulties should eventually disappear with improvements in design and manufacture of the fuel-system components. To protect the engine from such troubles, an emergency mechanical override control is provided; however, difficulties have been experienced with interference of the failed automatic control with the emergency control. The transition from automatic to manual control has also been a source of some trouble.

The responsibility for keeping the engine out of these difficulties is presently shared by the pilot and the automatic control. For example, in the simplest type of control, only a maximum engine speed limiting device is used, which automatically reduces the fuel flow whenever the speed exceeds a specified value; the other control functions are in the hands of the pilot. Controls are now in use which, in addition to controlling maximum speed, limit fuel flow during acceleration to avoid compressor stall and possess rudimentary flameout control features. Development is in progress on controls that also limit maximum temperature and that have more accurate flameout prevention features.

Additional difficulties with the fuel system discussed in the chapter on combustors (ch. V) are

(1) Changes in the fuel discharge rate to a combustor by erosion of the discharge orifice or clogging of the orifice or filter by carbon or other debris can cause uneven heating of the combustors with the result that some of the combustors are overheated.

(2) Distortion of the fuel-flow pattern in a combustor as a result of carbon formation on the combustor near the fuel nozzle can cause hot spots on the fuel liners and turbine nozzles.

METHODS FOR IMPROVING RELIABILITY

The difficulties mentioned in the previous section are the collected problems of a variety of engines. Some of the problems appear in some engines and have been eliminated in others. Some of the problems are common to all the engines investigated. They have all been listed to indicate the kinds of difficulties that must be considered. In the present section an attempt is made to discuss methods for handling these problems. These methods include improvement of the initial product and avoidance of failures in operation through application of inspections, replacement schedules, and warning devices.

The reliability problems and methods discussed are subject to modification with time. New engine designs may reveal failure modes not discussed in these papers. Applications that involve long engine lives, such as commercial transport service, may bring out additional types of failure.

Considerable improvement in reliability can result from an intensive study of design and manufacturing methods. The Lewis laboratory has not made such a study; therefore, the remarks herein are limited only to some reflections on these subjects drawn from the studies on the failure mechanisms of the components discussed.

The small amount of available documented evidence supports the prevalent impression that pod installations of engines are less prone to cause aircraft accidents than engines buried in the aircraft structure, and four-engine aircraft obviously are more reliable than single-engine airplanes. The problems of engine installation are not discussed.

DESIGN

The current high-speed airplane is the result of the development of engines of high thrust per unit weight and frontal area. The low-engine-weight requirement forces design toward small margins of safety. Because of the continual competition for flight performance, design and material improvements have been utilized primarily to obtain engines of higher thrust per unit weight and size. At each stage in the development of turbojet engines, the designer must weigh the conflicting requirements of high specific thrust and high reliability and arrive at a practical compromise. Raising the design temperatures and stresses, for example, allows the attainment of engines of higher specific thrust, but at the sacrifice in life of some of the critical components in the hot end of the engine.

The trend in engine type has also reflected the emphasis on performance rather than reliability. The axial-flow compressor has displaced the centrifugal compressor, in spite of its much greater probability for catastrophic failure from foreign-object damage or vibration, for the benefit of greater thrust per unit frontal area and efficiency. In the single-stage turbine, loss of a turbine bucket usually does not cause extensive damage or loss of thrust. In the newer engines equipped with multistage turbines, however, the loss of a bucket in an early turbine stage can cause extensive damage to the engine. The final choice between performance and reliability is in the province of the user. His choice, however, is limited to engine types commercially available in the thrust range desired.

Small reductions in operating temperature and stress lead to large increases in design stress-rupture lives of turbine buckets. As an illustration, it is pointed out in chapter VII that reductions in operating temperature of 100° F and in engine speed of 4 percent in a current-production-model engine result in a 44-fold increase in stress-rupture life of the buckets at the cost of a 6-percent loss in thrust, but with the benefit of a 4-percent reduction in specific fuel consumption. The effects on bucket life due to reductions in temperature and engine speed are different if the buckets fail by a mechanism other than stress-rupture.

The actual bucket life can be considerably less than the design stress-rupture life if damage by thermal cycling, foreign objects, or corrosion occurs or if severe vibration is encountered. The reductions in life from these causes are difficult to anticipate. Foreign-object damage, for example, is a chance phenomenon and can occur at any time; similarly, the damage done by vibration depends on the length of time the engine is held at a speed at which a bucket is in resonance with a strong vibratory force. Therefore, a condition favorable to high operational reliability is obtained if the bucket is designed to have (1) long stress-rupture life, considerably longer than the acceptable discard time for the bucket, and (2) low vibrational stress. Vibrational stress can be reduced by such methods as reducing the exciting force, designing the blade to have no resonance with the exciting force at the important engine operating speeds, and providing vibration damping such as shrouds or root damping devices. (See chs. IV and VII.) Under these conditions, damage initiated by foreign objects or thermal cycling will progress through the bucket slowly and can be detected in a scheduled inspection of the buckets. Reliability is further benefited if the intensity of the cyclic thermal stresses is reduced.

If the designer departs from the conditions outlined in the foregoing paragraphs toward more severe operating conditions in the interest of obtaining greater thrust performance, the user must attempt to obtain good operational reliability by more frequent inspections and replacements. In some recent high-performance engines the buckets have been designed for long stress-rupture life and low vibratory stress.

It is likewise easier to obtain high operational reliability on the turbine disk if the critical areas of the disk, namely, the rim and serrated section, are designed for considerably longer stress-rupture life than the anticipated operating life of the disk. This is done, of course, by choosing the proper combination of material, design stresses, and material temperatures. The material temperature can be decreased by decreasing the combustion-gas temperature and also by improving the cooling of the disk. If the latter method is used, the disk cooling system must be carefully designed so that the probability of failure of the system or of improper installation is very small. Until this is assured, operational reliability would be improved if a device for either warning of overtemperature of the wheel rim or loss of wheel cooling airflow were provided.

The disk rims, turbine buckets, nozzle vanes, and combustor liners can be cracked by thermal cycling. The tendency for thermal cracking of these components would be reduced if combustion temperatures and rate of temperature change in transient operations were decreased. The rate of temperature rise during ignition, for example, can be decreased if atomization at low fuel rates and ignitor performance are improved to permit lightoff with a smaller amount of fuel initially in the combustion chamber. Starting the engine on one burner and then gradually igniting additional burners will reduce the rate of temperature rise of the turbine buckets. Reduction of engine acceleration rates and avoidance of surge will, likewise, reduce the rate of temperature change. Rapid temperature drops occur during deceleration and shutdown. Methods for obtaining a more gradual temperature change during shutdown should be studied. The tendency for thermal cracking can be decreased if the temperature of the component is decreased by improved cooling. When reliance is placed on cooling, adequate performance of the cooling system at the transient conditions of interest must be assured.

Considerable increase in the lives of the nozzle diaphragm and the combustor liner would result if they were designed to minimize the thermal stresses induced by the large temperature gradients on them. For example, the nozzle diaphragm would benefit if the vanes and outer and inner rings were permitted to expand independently, if regions of stress concentration such as the sharp trailing-edge corners in the vane slots in the inner ring were eliminated, and if provision for differential expansion between the rings and their points of support on the engine frame were made. Improvements in design and materials that tend to reduce the temperature differentials in these structures would also be beneficial.

The compressor blade should be designed with a large strength margin so that damage from foreign objects or stress-corrosion would progress slowly enough to be found in an inspection. Increased thickness in both compressor blades and turbine buckets increases the resistance of these components to impact.

Reduction in the severity of vibrational stress by the methods described in chapter IV will increase the blade reliability. Excessive rotating-stall forces should be reduced by design changes. These are likewise discussed in chapter IV.

The fatigue life of thrust bearings must likewise be carefully chosen to provide a reasonable replacement time. The fatigue life should be sufficiently high so that, after allowance is made for the scatter in bearing properties and other uncertainties, a practical replacement time results. The fatigue life of bearings varies inversely as an exponential function of the stress and can, therefore, be appreciably increased by decreasing the stress. The stress, of course, can be decreased by designing more rugged thrust bearings or by distributing the thrust load among several bearings.

Failure of the lubrication system can precipitate bearing failure and catastrophic failure of the engine. Evidence obtained on engines in which the lubrication system was damaged by gunfire indicated that the bearings on some engines continued to function for sufficient time to permit the airplane to return safely to its base. Bearing and lubricant systems should be designed to provide this reserve in life to permit the engine to operate for sufficient time for adequate emergency measures to be taken.

In the design of controls for improved reliability, there are two opposing features that require consideration:

(1) The more elaborate controls, when they work properly, reduce the hazard of pilot error, particularly when he is preoccupied with flight operational problems.

(2) The more elaborate controls have a higher probability of control component failure.

At every stage in the development of the art of controls, an optimum division of responsibility between pilot and automatic control for maximum reliability exists. The more advanced the state of the art, the greater the responsibility that can be placed on the automatic control. The choice of this division of responsibility depends on the type of airplane service involved. For example, in transport operation, where the flight plans are routine and performance requirements more relaxed, more of the responsibility can be placed in the hands of the pilot than in the case of the fighter airplane, where rapid accelerations may be required and where the pilot may be involved in combat at a time when engine conditions are changing rapidly because of the combat maneuvers. The current trend is toward the more elaborate automatic controls to relieve the pilot of much of the control burden.

Reliability of the engine would be improved by the following improvements in the control functions:

(1) More accurate acceleration control. The improvement of acceleration control requires, not only more accurate correspondence between the control adjustment and the compressor surge limits, but also improvement in the design of the engine and engine installation. In view of the complex relation between surge limits and engine and flight conditions, the problem of acceleration control would be greatly simplified if a method of directly sensing imminent surge were developed.

(2) Addition of a reliable maximum-temperature-limit function to the control.

(3) More accurate flameout control.

INSPECTION

The frequency of inspection and replacement, and hence the cost of maintenance, can be decreased by reduction in the severity of operating conditions. In any practical compromise between maintenance cost and engine thrust, the need for considerable inspection can be anticipated. The engine and installation should be designed to facilitate these inspections.

Inspection in manufacture.—Because of the great cost of modern jet engines and the possibility that a component failure can cause loss, not only of the engine but also of the airplane and passengers, thorough testing of the critical components of each engine is justified.

Discussion of the complex problem of the inspection of materials and finished products is beyond the scope of this report. Tests indicative of the resistance of the component to the important failures discussed should be included in the specifications and manufacturing procedures. Examples are given in chapters VI, VII, and VIII.

By good inspection to eliminate imperfect material and by good control of the fabrication processes, the number of defective parts of the engine may be held to a very low value. However, a finite percentage of flaws will probably pass undetected. Furthermore, difficulties such as misalinement of components can be introduced by improper engine assembly. Therefore, every engine should be subjected to a trial-run period in which some of these defects may be found and corrected. The trial run should include test conditions and maneuvers that will bring out performance difficulties such as severe rotating stall and flameout, and also the adequacy of the control.

The delay of flight application of an engine to a time when it is completely developed is not feasible. Furthermore, the different aircraft applications of the same engine have their special problems. Hence, a period of "debugging" during the early flight history of a new engine, or of an old engine type in a new application, can be anticipated. During this period, it is important that the manufacturer and user treat the early flight operations as part of the development program, and that they set up procedures to expedite this program with minimum hazard of flight accident. This requires close cooperation between the manufacturer and user.

Inspection in service.—The inspection procedures that can be set up to prevent a flight accident by failure of a component depend on (1) the cause of failure of the component, (2) the speed with which the component proceeds from indication of incipient failure to final failure, and (3) the seriousness of such a failure with regard to its tendency to cause a flight accident.

Inspections scheduled on time basis.—For some of the failures the grace time from first indication of failure to final failure can be sufficiently long that inspections may be scheduled to detect the difficulties. In this category come such items as wear of bearings and thermal cracking of combustor liners and transition pieces, turbine disk rims, nozzles, and buckets. If, for cases that fall in this category, the grace time is too short for a practical inspection period, this condition should be corrected by changes in design or operating conditions. Since inspection of these components requires appreciable disassembly of the engine, their inspection requirements should be studied and an integrated procedure devised. Scheduling of this major inspection to coincide with desired major overhaul times would, of course, be highly desirable. Whether this can be done depends on the quality of the engine and the severity of the service operating conditions. During the scheduled inspection, examination should also be made for foreign-object damage. However, because the time of occurrence of foreign-object damage is unpredictable, and because the consequences may be catastrophic, every effort should be made to minimize the danger of foreign-object damage and to check for evidence of foreign-object damage on a preflight basis. When extensive inspection of the compressor blades for foreign-object damage is impractical, a limited inspection, involving the screen and the early compressor stator and rotor stages, may be feasible without disassembly and may greatly reduce the probability of a failure from this source.

Inspections scheduled on basis of contingency.—In this category come failures which may be considered accidental. Very rapid deterioration of the life of the components in the hot end of the engine can result from overtemperature or overspeed. Inspection of the parts in the hot end of the engine should be made when overtemperatures and overspeeds of prescribed intensity and number have been experienced, as is the practice in military service. The reporting of these incidents is currently left to the pilot. An automatic recorder which gives the temperature and engine speed as a function of time would provide a much more accurate basis for scheduling inspections. Not only the mean combustion gas temperature but also abnormal distributions in temperature should be indicated.

Inspection of the turbine disk should also be made when there is indication of difficulty with the turbine wheel cooling system. Evidence of excessive rim growth and changes in hardness and microstructure may be used as indications of deterioration. Inspection of the bearings and the lubrication system should be made when there is evidence of lubrication-system failure and when excessive amounts of metallic particles are detected in the oil filters.

Flight checks and warning devices.—Failure of the control system in a crucial maneuver may cause a flight accident. For example, failure of the acceleration control during a landing or a flight refueling operation when the throttle may be "jockeyed" at reduced engine speed may permit the engine to experience destructive compressor surge. When reliability of the control is a problem in any specific engine, the pilot can, prior to entering into a critical maneuver, check the operation of the control in the expected engine condition.

Monitoring instrumentation in the control, fuel, lubricant, and turbine cooling systems would warn the pilot of trouble in these systems.

REPLACEMENT SCHEDULES

Replacements are, of course, made when inspections indicate irreparable damage to any of the components. However, there are some components for which the time between the appearance of incipient damage and the final failure of the component is much less than the time between scheduled inspections. In these cases it is necessary to provide a replacement schedule in order to avoid failure of these components in flight. Components that fail by stress-rupture or fatigue mechanisms come under this heading. Turbine disks, turbine buckets, and thrust bearings are specific items that must be considered. If the stress-rupture lives of the turbine buckets and disks and the fatigue life of the thrust bearings are so great that these items can be expected to fail by other causes long before they reach their life limits, then the setting of these life limits is not important. However, should these other causes of failure be largely eliminated so that some of these components begin to approach the stress-rupture and fatigue life limits, then it is important to set up a replacement schedule based upon the best available experimental data on the lives. The removal times should make allowance for the scatter in performance of these components and should provide sufficient margins of safety for uncertainties in the operational conditions encountered by the individual engines. When a replacement schedule is set up for a component, a system for keeping track of the operating time on the component is necessary.

ENGINE RECORDS

Records that would assist in scheduling inspections and replacements are

(1) Operating time on thrust bearings for scheduling replacement for fatigue

(2) Operating time on turbine disks and buckets for scheduling replacement for stress-rupture

(3) Extent and duration of overtemperature and overspeed for special inspections and replacements

(4) Number of starts, accelerations, and stops for scheduling inspection of components subject to thermal cracking

A continuous record of gas temperature and engine speed against time would be extremely valuable in providing the preceding data. Some devices to do this are under development at this time. If reliance is placed on pilot reports, as is the prevalent practice in military service, accurate gas-temperature and speed indicators should be provided.

The problem of keeping records on component lives is simplified if components are reassembled into the same engine in overhaul.

OPERATION AND REPAIR PRACTICES

There is a natural tendency to try to fit turbojet-engine operation and maintenance practices into the pattern established for the reciprocating engine. Some fundamental differences between these two engine types exist, however, which must be considered in setting these practices. The turbojet engines are much more susceptible to unpredictable, and in some cases catastrophic, failure from such sources as foreign-object damage, overtemperature, overstress, and thermal and mechanical fatigue.

The frequency of foreign-object damage can be reduced by screens or other protective devices, by removal of debris from airport runways, and by careful maintenance practices. The frequency of overtemperature, overstress, and surge can be reduced by improved controls and by more extensive pilot training. Creep and thermal-fatigue difficulties can usually be reduced on a given engine by reducing operating temperatures and stresses.

The military services press for high thrust performance, and therefore many of the hot-end components experience short lives. The Air Force has obtained a large reduction in maintenance cost and "engine-out" time by permitting considerable maintenance at the operating base. The operating base is now permitted to replace all hot-end components and bearings, and consideration has been given to including compressor blade replacement.

It would be highly desirable in commercial operation to achieve overhaul times comparable with those obtained on reciprocating engines. The user will have to determine for each engine type the best compromise between thrust and component life. When this compromise involves component lives less than desired times to overhaul, a study should be made to establish minor repair procedures that would result in reduced cost and engine-out time.

ADDITIONAL INFORMATION NEEDED FOR IMPROVEMENT OF RELIABILITY

Extensive research efforts are being made to improve the strength and performance of all the critical engine components, which should contribute to improvement in reliability. These research programs are not discussed. In each chapter of this report the need is pointed out for special information associated with the particular problems of reliability discussed and this is summarized here. Information is needed on

(1) Failure mechanisms. Some of the failure mechanisms for several of the components and the influence of the important variables on time to failure are not understood. For example,

(a) In the failure by thermal cycling of such components as combustor liners and turbine disks, buckets, and nozzle vanes, the influence of the temperature-stress history, the object shape, and the material properties on the number of cycles to failure is not sufficiently understood. This information would lead to remedial measures and to procedures for avoiding failure from this difficulty in flight.

(b) More information is needed on the effects of overtemperature and overstress on the stress-rupture life of turbine buckets and disks at normal conditions. This information would permit setting more realistic criteria for removal of buckets and disks for overtemperature and overstress.

(2) Time from first indication of failure to final failure. A knowledge of this grace time is needed to set up inspections and parts replacement schedules for each engine type of interest. Under this heading come such items as

(a) The time from the first appearance of cracks produced by thermal cycling to failure of turbine buckets and disks.

(b) The time from the first appearance of a fatigue pit or other damage in a bearing to final bearing failure.

(3) Defect tolerance. Information is needed on the magnitude of the following component

defects that may be tolerated before safe operation is jeopardized:

(a) Compressor blade and turbine bucket nicks.

(b) Bearing scratches.

(4) Methods of detecting damage. Damage by such mechanisms as stress-rupture or fatigue is not now detectable until the subject is very close to failure. Hence, replacement of these objects cannot be based on evidence of damage in a scheduled inspection. These components could be placed on an inspection schedule if methods were available for indicating the amount of life remaining in the component.

(5) Warning devices. The failure of a component in the fuel, control, lubrication, or turbine cooling systems can precipitate a catastrophic failure. Warning devices in these systems would permit the pilot to take preventive action. In particular, information on methods of sensing combustion-gas temperature would help in the solution of the problem of overtemperature control. Information on a method of detecting imminent surge would assist in the development of better acceleration controls.

(6) Methods of preventing a component failure from causing engine failure. Under this heading come such items as

(a) Methods of extending operating time of bearing, after failure of lubricating system, to permit emergency measures to be taken.

(b) Methods of disposing of broken compressor and turbine blades to reduce the probability of a chain reaction of failures in successive stages of these components.

(c) Methods of localizing the effects of failure of a control-system component in order to avoid endangering the engine.

(d) Methods of reducing the probability of damage to the airplane structure and adjacent engines by a turbine disk burst.

(7) Methods of reducing foreign-object damage. Additional information is needed on the design of screens and other devices for improving protection against foreign-object damage.

(8) Methods of designing large structures which are subject to temperature difference to avoid distortion and cracking. This information would permit improvement in the life of such items as combustor liners, nozzle diaphragms, and engine frames.

CONCLUDING REMARKS

The statistics on the overhaul of older models of turbojet engines reveal that a large number of parts were replaced at times less than desired times for major overhaul. This situation may be improved in engines of newer design. However, the designer must make compromises between the conflicting requirements of high thrust performance and reliability. Hence, it is reasonable to expect that there will be a residue of items with lives shorter than desired or with small margins of safety in strength. The designer must contend with such problems as the finite stress-rupture life of turbine buckets and disks and the finite fatigue life of thrust bearings and compressor blades.

The strength of the engine components can be unpredictably reduced by such incidents as (1) foreign-object damage, which may cause catastrophic failure of compressor blades and turbine buckets; (2) thermal cycling associated with starting, accelerating, and stopping of the engine, which can cause cracks to appear in such items as combustor liners and turbine nozzle vanes, buckets, and disks; (3) vibration of compressor blades and turbine buckets caused by holding the engine at a speed at which strong vibratory forces and resonances exist; and (4) overtemperature and overspeed of the engine by malfunctioning or inadequacy of the automatic controls, or mishandling of the engine, which can drastically shorten the lives of combustor liners turbine buckets and disks, and nozzle vanes.

The turbine buckets and disks should be designed for long stress-rupture life in order that defects from causes such as thermal cracking will progress slowly enough to permit detection. Large vibratory stresses would tend to accelerate failure greatly, when defects occur, and, hence, should be avoided by proper design.

Even when the engine is conservatively designed, the improvement of operational reliability requires scheduled inspections and replacements, warning devices, and instrumentation.

Scheduled inspections are required for such items as thermal cracking of turbine disks and buckets, nozzle vanes, and combustor liners; foreign-object damage of compressor blades and turbine buckets; bearing wear; and malfunctioning of fuel, lubrication, and control systems. The engine and installation should be designed to facilitate the inspection. The frequency of inspection will, of course, depend on the specific engine design and the severity of operation.

There are some items for which an inspection schedule is not adequate, because the time between the first indication of deterioration and final failure is usually shorter than any reasonable time between inspections. This category includes fatigue failure of thrust bearings and stress-rupture failure of turbine buckets and disks. In an engine designed primarily for reliability, the lives of these components can be considerably longer than the practical discard times, and they impose no scheduling problem. However, if these components are pressed by thrust performance requirements to operate at conditions which give shorter lives, they require a replacement schedule which is based on statistics on their lives. A system for recording the operating times of these components is required in order to apply the replacement schedule.

Failure of systems such as the turbine disk cooling system, the lubrication system, the fuel system, and the control system can cause catastrophic failure of the engine. Devices which will warn the pilot of difficulty in these systems will permit him to take preventive measures.

Compressor rotating stall encountered during acceleration of the engine has been responsible for the very rapid fatigue failure of compressor blades. The severity of vibrations from rotating stall can be reduced by design changes that have been discussed.

Overtemperature and overspeed have been responsible for the failure of the various components in the hot end of the engine, such as the turbine buckets and disks. Difficulty has also been experienced from surge during acceleration. The automatic controls should be improved to handle more effectively the conditions in which overtemperature and surge may occur.

The records presented showed that failure of the fuel system and control system is the largest cause for accidents in flight. Therefore, it is essential, in addition to design improvement, that a method of preflight and inflight checking of these systems be worked out and that warning devices be developed.

For improvement in operational reliability, additional information is required on such items as failure mechanisms, methods of detecting incipient failure, warning devices for malfunctioning in the systems, methods of preventing component failures from becoming catastrophic, and methods of reducing foreign-object damage.