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EXPERIENCE WITH LOW COST JET ENGINES

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ABSTRACT

A summary is given of the results of a NASA program for reducing the cost of turbojet and turbofan engines. The design, construction, and testing of a simple turbojet, designed for use in missiles, is described. Low cost axial stage fabrication, the design of a fan jet engine, suitable for propulsion of light aircraft, and application of such engines to provide higher flight speeds, are discussed.

INTRODUCTION

The purpose of this paper is to discuss the major results of a program for investigation of low cost jet engines which has been conducted by the NASA Lewis Research Center with assistance from contractors. This program has covered a number of problem areas. These are: (1) Engine cycle analysis and airplane performance studies relating to the cost-performance trade-off question; (2) Engine configurations and design techniques; (3) Engine operation simulation and control design study; (4) Fabrication development programs and spin testing intended to establish some possible new compressor and turbine construction methods; (5) Construction of experimental models and tests on a low cost annular combustor; (6) Design and testing of a new type of hydromechanical fuel control; (7) Work on low cost accessories; and (8) Construction and test of a complete turbojet engine.

ENGINE DESIGN DISCUSSION

The results of the engine performance analysis portion of this work may be summarized as follows: If lower cost gas turbine engines are to be obtained, we must be willing to give up some performance; limit the design temperature to a level low enough to obtain reliable machinery requiring only low cost materials; limit the pressure ratio to reduce the tip speeds, the stress levels, and the number of expensive stages.

Following this approach we have limited the design turbine inlet temperature to 1500^{o}F and the pressure ratio to 4.0 to 1 for our simple turbojet engine. The engine designed on this basis is shown in Figure 1. This engine has been designed for 650 lbs. sea level static thrust, and it has been built and is currently being tested in cooperation with the U.S. Navy in a program to determine the applicability of low cost turbojet engines for missiles and drones. This engine uses a four-stage cast compressor and a single-stage cast turbine. The front bearing support, the compressor casing, and the rear bearing support, are all one piece castings. The combustor housing and liner are simple sheet metal assemblies. These components will be shown in more detail later.

This engine is expected to have a fuel consumption of 1.3 lbs/hr per lb. at a flight speed of 570 mph. It has an external diameter of $11\frac{1}{2}$ inches and is expected to weigh approximately 100 lbs. A view of this engine, assembled and mounted on its thrust stand, is shown in Figure 2. Note that the stator mounting attachments on the side of the compressor housing are for the test engine only and would be eliminated and replaced by single piece stator rings in later models. This turbojet has been designed and built using low cost components and fabrication methods, throughout. It is intended to demonstrate

the technology of low cost engines for unmanned use but the experience and knowledge gained will also apply to jet engines which would be suitable for light aircraft propulsion. It is being tested in a Lewis Research Center sea level test stand. Further discussion of the fabrication, component testing, and complete engine testing phases of this program will be given later.

Various configurations for low cost fan jet engines are also being evaluated. One configuration being studied is the geared front fan, as shown in Figure 3. Here a single stage fan rotor is driven by a single spool core engine through a gear box having a speed reduction ratio of about 2 to 1. This type of engine takes advantage of the added fan stage to operate at a higher overall pressure ratio and propulsive mass flow. Both of these factors improve its thrust and fuel consumption, compared to a simple turbojet. At an overall pressure ratio of 6.0 to 1 and a fan bypass ratio in the range from 2.5 to 4.0 this engine will have a specific fuel consumption of from about .70 to .90 lbs/hr per lb. in the flight speed range of interest for advanced light aircraft.

Using a core engine having an 8 inch diameter four or five stage axial compressor and a 10 inch diameter, two stage axial turbine this engine would have a sea level static thrust of about 1100 lbs. The core engine for this thrust level would thus be of the same size as the turbojet engine, previously discussed. For the bypass ratio range of from 2.5 to 4.0, thought to be of chief interest, the fan tip diameter would be of from 15.0 to $16\frac{1}{2}$ inches.

The geared fan configuration provides the fuel economy of the fan jet engine and the low noise level of a low tip speed fan, while still retaining a single shaft, two-bearing design for the core engine. It is therefore of major interest for a low cost fanjet. It should also be noted, however, that designs with a separate turbine and a coaxial drive shaft to the fan, or to a speed reducing gear box, would provide certain technical advantages as well as cost competition and are also still being actively considered.

In this fanjet design drawing (Figure 3) may also be seen the thin disks for the compressor and turbine rotors which is typical of sheet metal construction. This type of construction will be further discussed later.

For light aircraft, the fan jet will provide greater range, greater takeoff thrust, and much lower noise levels than the turbojet and is therefore the preferred type engine. Interest is also centered on the fan from a cost standpoint, since the core engine is the component containing the greatest number of expensive high temperature parts whose cost is greatly influenced by their size. The thrust of a given core engine can be nearly doubled by adding the fan rotor. If properly done, therefore, it may be expected that the fan jet engine can provide greater economy in dollars per pound of thrust.

A further word may be in order concerning the relative cost of the all axial flow engine as shown and the engine types which employ a radial flow compressor. Both types of configurations were evaluated in

detail during our program. The radial compressor has advantages in reducing the number of compressor components. This advantage is offset however, by the considerably larger diameter of the radial compressor and its diffuser and cost comparisons of these two types of compressors made early in our program indicated a near stand off. However, the axial compressor, having the advantage of being smaller, is also more directly suitable for fabrication by low cost methods, such as casting or stamping. It was therefore decided that all design and fabrication work on our project would be concentrated on this type.

FABRICATION METHOD DISCUSSION

Reduction in manufacturing cost is the second vital factor for obtaining a low cost gas turbine engine. Numerous techniques have therefore been investigated for achieving such cost reduction. The general approach is to use manufacturing methods in which the engine components can be quickly manufactured, using relatively inexpensive machinery and equipment, and keeping the man-hours of labor to a minimum. One procedure meeting these requirements is investment casting of complete disk and blade assemblies for the compressors and the turbine. The cast compressor and turbine rotors which have been fabricated for the Navy Ordnance turbojet engine are shown in Figure 4. Each of the four axial compressor stages were investment cast in a single piece using 17-4PH, a relatively inexpensive material which may be readily cast and which has a simple heat-treat cycle. Very sound castings with accurate blade profiles have been obtained, and these have been spin-tested to a speed 50% greater than their rated speed. Also shown in the figure is the single piece turbine stator assembly which has been investment cast using Haynes Stellite 31, and the turbine rotor cast from Inconel 713LC. Both of these castings have also proven to be very sound and accurate, and the rotor has also been spin-tested to 30% above its rated speed.

To make the complete rotor, the four compressor rotor castings are joined together by circumferential electron beam welding, as shown in Figure 5. The compressor assembly is then electron beam welded to a cast shaft. This assembly technique has provided very sound and accurate joining of the components, and it is also very rapid. Only about one minute of actual welding time is needed for each of the five welds. The turbine is then attached to the rotor by bolts. This approach facilitates engine assembly and eliminates a weld between dissimilar materials.

An alternate approach for obtaining low-cost axial stage rotors which is also being investigated is illustrated in Figure 6. Combined disk and blade assemblies are stamped out of sheet metal, and the blade profiles are formed by coining. To obtain the total number of blades required, a pair of bladed disk assemblies is fitted together. As illustrated in Figure 7, these are then fitted into mating rings to form a complete rotor.

The rotor shown is one which has been built for spin testing. Several of these sheet metal compressor rotors have been built and tested in our spin-test rig. During these tests, strain and clearance measurements were taken to compare with the design values. The measured values correspond very well with calculated ones and certain versions of this rotor have been tested to 25 percent above their design speed, indicating a good strength margin.

Sheet metal construction techniques, similar to those shown for the compressor rotor, are also being used to build fan rotors and axial-flow turbine rotors. An experimental fan rotor built with removable stamped sheet metal blades is shown in Figure 8. The hollow sheet metal blades have proven to be very strong, and they have a high resonant vibration frequency. A sheet metal turbine is also being investigated. A model illustrating a turbine which would use this construction is shown in Figure 9. In general, the results achieved during tests of both the compressor and the fan rotors have agreed with expectations and indicate that the sheet metal construction technique is attractive. The analysis also indicated that sheet metal construction will be suitable for turbines. In addition, aerodynamic tests have been performed on thin bladed turbines, typical of the sheet metal type. These tests have shown very good efficiency.

Fabrication development and performance testing have also been performed on a low-cost annular combustor, as shown in Figure 10. This combustor is constructed from rolled cylinders with a simple pattern of punched holes. Commercial perforated sheet is used and this is oriented in a special way to provide a cooling airflow layer which keeps the liner substantially below the gas-stream temperature. The combustor has achieved combustion efficiency above 95% and an adequate temperature-variation pattern factor of below .3. In addition, the low metal temperatures, well below the gas temperature, permit low-alloy materials to be used, and this combustor is therefore considered to be very promising for application to low-cost gas turbine engines.

ACCESSORIES AND CONTROL

Accessories are also very important to the overall cost and size of small jet engines, since they tend to be large and expensive, when we desire just the reverse. The most critical accessory is the engine speed and fuel control which must provide capability for rapid thrust response without surging the compressor or exceeding the allowable turbine temperature. The control is a key to the safety and reliability of the entire engine and is furthermore critical to the cost. The control adds 20% to the cost of the engine. We have, therefore, extensively investigated the fuel control and have developed a new type of hydromechanical control based upon use of a parameter consisting of fuel flow, divided by speed and by a compressor inlet pressure correction factor. Using this parameter as a function of compressor

ratio, a fuel schedule is established which provides the correct steady-state fuel flow over the complete range of rotational speed, altitude, and flight Mach number. This fuel schedule is then modulated by a speed error signal from a speed governor to control the speed of the engine. Compressor surge is avoided, during an acceleration, and blowout, during a deceleration, by keeping the fuel-speed parameter between proper limits. The technique used to obtain the speed signal and the governing action is to use a small fuel pump, of the type known as a zero gradient pump. The arrangement for this control is shown in Figure 11. Operation is as follows: A small speed sensing positive displacement gear pump is driven by the engine. The pressure difference across this pump is kept at zero by a pressure regulating valve which bypasses flow from the main fuel pump. Since no pressure rise occurs across the small pump, its output flow rate is directly proportional to the engine speed. This flow quantity is then passed through a fixed area orifice to generate a pressure signal which is directly proportional to the quare of the engine speed. The actual speed is then compared with the pilot's speed command setting to generate a speed error signal, and the area of an orifice that bypasses the speed sensing pump is controlled by the compressor inlet and discharge pressures and also modulated by a speed error signal to control or govern the engine speed. Thus, when the throttle is advanced, the control provides additional fuel flow to increase the engine speed. During such speed transients, however, the changes in fuel flow are limited by the acceleration and deceleration schedules.

This simple hydraulic circuit provides all of the required functions of the jet engine control. The working parts required to construct such a control are shown in Figure 12. The small gear pump which provides the speed signal, the fuel bypass valve which controls the zero pressure gradient, and the valve which adjusts the flow according to the speed error signal are shown. In addition, two valves operated by compressor inlet and discharge pressures, which provide the controlled area orifice for the fuel schedule are pictured. This control has an aluminum housing and stainless steel pump and valve elements. Close fits are not required in the metering pump because of the zero pressure gradient nor are they required in the valve assemblies because the maximum valve pressure drop is only 25 psi. Only two spring bias adjustments are required at assembly, one which sets the deceleration limits, and the other the acceleration limit.

This control has been extensively analyzed and its operation simulated on the analog computer. It was then built and tested. It performed successfully in actual operation on a J85 engine. During these tests, it provided all the expected control functions, and it also produced a very smooth acceleration of the engine with short response time.

In addition to the work on the fuel control, a number of other accessory areas have been considered. The more important results are summarized in the following paragraphs.

The startup technique for the engine has been extensively considered. After evaluation of numerous possible startup techniques, it was concluded

that, since a generator and battery will be needed for flight, the overall best economy will be achieved by using these same components for engine cranking. Since cost of starter-generators rise very rapidly with power output, it is evident that cranking must be achieved with the very minimum size unit possible. This type of starter-generator has, therefore, been tested to verify its capabilities in both modes of operation and to investigate possible mechanical simplifications which may permit cost reduction. The conclusion was reached that a 150 amp starter-generator, operating at 24 volts, should be adequate for cranking of the 1000 lb. thrust engine. This unit was moderate in cost, having an estimated production price of $350.

A design study has also been undertaken on the fan drive speed reduction gearing. The result of this study was favorable on both a technical and a cost basis. The gearing system selected provided an overall speed ratio of 28 to 15 and had the advantages of a coaxial output shaft and a small space envelope. It also had low gear tooth loads and bearing loads. Very good life was indicated with the small, low cost bearings and gears so that both a low fabrication cost and a long life appeared possible.

Additional accessories and auxiliaries which have been designed and investigated include (1) a fan duct flow blockage system, which permits reduction of fan torque and elimination of fan thrust at idle speeds, (2) low cost engine mounts and dampers, (3) a simple accessory drive gear box for driving the engine starter-generator and the lubrication and scavenge pumps, and (4) a hydraulic drive system to allow remote placement of accessories. In addition, attention has been given to arrangements for fitting accessories within the engine mounting pylon to permit preserving the low frontal area of the basic fanjet engine.

TEST EXPERIENCE WITH THE TURBOJET ENGINE

The 650 pound thrust turbojet has proven to be very practical in fabrication and assembly. A complete set of engine parts was detail designed and parts were procured, finished machined, and assembled in approximately one year from the date of the decision to go ahead. This included procurement of all the tooling required for limited production fabrication of all major parts. It also included spin test qualifying of all rotor components and of the electron beam welding of the rotor. In addition, verification of the rotor dynamic stability and bearing lubrication system was made in a dynamic simulation test rig. The engine was then installed in a sea level test stand as shown in Figure 13, and operated successfully in its first scheduled test in December, 1971.

A few comments concerning the results of the spin testing and dynamic qualification may be of interest. A first pertinent factor is that the design philosophy of using low stress levels to obtain a high safety factor in the cast parts was at least partically shown to

be successful by performing spin testing on all compressor rotor components to a speed of 150% of rated speed. All compressor parts have thus been shown to operate at stress levels more than twice as great as required in service. The cast turbine was also qualified to a speed of 30% above rated speed and calculations show that it should also be capable of operating at stress levels twice as great as design levels.

Another interesting development is the bearings for the turbojet engine which are lubricated with an air atomized oil mist and cooled by conduction and by compressor bleed air. Both the front and rear bearings are soft mounted. The front has a grease damped rubber mount and the rear a metal spring. This bearing lubrication and mounting system has performed well in the dynamic test rig and in the engine. Both bearings should operate at below 300°F in service, to permit low cost bearings to be used, and the mounting system completely damps out the rotor rigid body critical motions and reduces transmission of rotor vibration to the engine housing. The engine design also features a very stiff shaft which gives a high bending critical frequency and makes the requirements for rotor balancing much less stringent. The operating engine has shaft and blade tip clearance probes and vibration pickups installed and measurements from these have shown that the overall shaft mechanical design and mounting system is very satisfactory.

The engine operating in the sea level test stand has accummulated over 18 hours of hot operation at the time of this writing. This test time has been occupied to date with the task of obtaining individual stage performance and overall performance on the compressor and the turbine. Since neither component has independently developed in individual test rigs, the proper compressor stator vane settings and turbine stator and jet nozzle area trimming is being performed from measurements taken in the operating engine. In order to permit performing this adjustment the first operating prototype engine has been provided with adjustable stator blade angles and a variable area jet nozzle. By the use of ram air from the test facility, the engine can be operated over a wide range of speed, corrected weight flow, and simulated flight Mach number.

At the present time, the engine has been successfully operated up to its full design speed of 35,200 rpm. The four axial compressor stages are not yet fully matched, however, and incipient compressor surge limits the engine operation over the entire speed range. Adjustments to the compressor rotor and stator blade setting angles and flow areas are being investigated to eliminate this problem. At the time of this writing compressor efficiencies of 75% have been attained and it is expected that further trimming will allow the design goal of 83% to be achieved. The data already indicates that the turbine design efficiency goal of 88% has been reached. On several tests, the design turbine inlet temperature of 1560°F has been reached and subsequent tear-down and inspection has shown that the combustor liner, the turbine stator, and rotor, and the rear bearing support structure are all free from major distortion or heat damage.

In interest of cost economy, the compressor design has avoided use of interstage bleed, variable guide vanes, or variable angle compressor stators. As previously discussed, the current problem for the engine is compressor surging, arising from this lack of variable geometry. The current corrective action is finding stator angle settings and similiar compressor geometry changes which will allow the engine to be self-sustaining and free from compressor surge over its entire speed range.

The full investigation of startup capability on the engine has not been completed, but it has shown its ability to start and self-sustain at speeds of from 25% to 40% of design. An impingment air starter installed has also shown that it is capable of producing rotor speeds of 20%. It is therefore expected that the engine will be capable of windage starting at ram Mach numbers of from .40 to .80 and starting with gas impingement alone at sea level static conditions.

ENGINE PERFORMANCE ESTIMATES AND APPLICATIONS

The requirements stated by the U.S. Naval Weapons Center for the ordnance-type turbojet engine are shown in Figure 14. The engine size and weight requirements, the windmill start capability, and the flight duration requirements can all be met by the simple, low pressure ratio, moderate temperature engine design which we have been investigating. Work is currently underway to demonstrate the sea level static thrust and the altitude cruise thrust capability will next be obtained in the Lewis PSL engine test facility. The requirement for specific fuel consumption of below 1.8 should be easily met with the estimated performance being an SFC of 1.3 at .8 Mach number.

It is expected that this small turbojet could also be further developed for application to man-rated service. At a Mach .65 flight speed (450 mph) at 25,000 feet which has been used as an advanced light airplane design point, this engine could be operated at a continuous cruise inlet temperature of 1300°F and would still produce approximately 240 lbs. thrust at an SFC of 1.22. This thrust level would provide a mean flight weight of 2040 lbs. at a mean lift-drag ratio of 8.5. A two-place sport plane having an empty weight of 1330 lbs., a passenger and luggage allowance of 460 lbs., and a useable fuel capacity of 500 lbs., would thus have a take-off gross weight of 2290 lbs. and a range of 770 statute miles. This performance could be obtained at a fuel cost of only 4¢ per mile. Obtaining such a flight speed and cruise altitude capability within these gross weight and L/D limits may well represent a considerable challenge to aircraft designer. These weight and drag limits are sufficiently close to the characteristics of current aircraft, however, to make this a meaningful illustration of the performance improvements obtainable by use of the jet engines.

A second illustration of the performance capability offered by the fan jet type engine is given in Figure 15. Here are given performance estimates which were made for a 6,000 lb. gross weight,

twin engine plane, designed to cruise at 450 mph at 25,000 ft. This plane has a 1000 lb. payload at full fuel weight and would provide a generous four passenger capability. The fan engines considered have a 2.5 to 1 bypass ratio. This plane has a typical fuel-to-gross-weight ratio and fuel reserve, and a wing loading low enough to provide the low takeoff and landing speed of 80 mph. For this airplane the two 1000 lb. takeoff thrust engines are capable to achieving a low takeoff distance of only 1050 feet. The very useful range of over 1000 miles is also obtained at an operating fuel cost level less than piston engines of equivalent thrust horsepower. These cost estimates, of course, take into account the lower cost of jet fuel, compared to aviation gasoline.

Interest has also centered recently in evaluating the capability of the fan jet engine for light plane propulsion at flight speeds lower than 450 mph and for smaller aircraft, closely similar to present-day light twins and single engine types. For application to lower flight speeds, higher fan bypass ratios are of interest since they improve the low speed fuel consumption. This approach appears to be cost competitive since sheetmetal fan blades, of the type previously shown, could be used in longer blade lengths without adding appreciably to total engine cost. This approach gives more thrust per lb/sec. of core engine airflow, and improves the fuel consumption to permit longer aircraft range.

It is also of interest to evaluate the performance achievable from a core engine of the same size as the ordnance turbojet. The size and performance parameters which would result for such an engine are given in Figure 16. As may be seen, a core engine of the same size as the current turbojet, using a single stage fan of 16.6 inches diameter, could attain specific fuel consumption of from .69 to .82 for the flight speed range of from 250 to 450 mph at 25,000 ft. altitude. Thrust available at the cruise turbine inlet temperature of the current turbojet, 1500°, would range from 420 to about 400 lbs. over the same speed range. Using this fanjet engine, a four-place single engine aircraft, having 800 lbs. passenger payload, a 3/4 hour fuel reserve, an empty weight of 2000 lbs., an L/D of 8.5, and a takeoff gross weight range of approximately 3700 lbs., could achieve a 1000 mile flight range at 450 mph. At this speed the fuel cost of this plane would be less for the fan jet, than for either a turboprop or a turbocharged piston engine. The thrust level of the turbofan also varies in direct proportion to gas density over a wide range of altitude. It would thus permit the light plane pilot to choose an altitude for maximum L/D over a wide range of flight speeds. This characteristic will therefore permit the fanjet powered airplane to achieve good range and economical operation even down to lower flight speeds. Due to its reduced engine frontal area and elimination of cooling drag, the fanjet powered aircraft requires less thrust than a piston engine aircraft. Considering this factor as well as the lower cost of jet fuel (compared to aviation gasoline), and the elimination of interest charges and maintenance costs on a large variable pitch prop, prop control, and low output speed gear box, it is expected that the fanjet will be found

competitive in operating cost with the piston engine at speeds as low as 300 mph. At flight speeds in the neighborhood of 350 mph, it should be competitive with the turboprop if the prop and gear box cost savings are again considered.

In order to permit comparison of the thrust power produced by the turbofan engine with that required for an equivalent piston engine or turboprop, the last column of Figure 16 lists the thrust horsepower produced over the range of flight speeds. These horsepower ratings would have to further be increased by the propeller and gear box inefficiency factors to arrive at a shaft power rating for an equivalent engine. When all pertinent factors are considered, such as weight, size, noise level, atmospheric pollution and maintenance requirements for example, it is expected that the fanjet engine will be found attractive and competitive at flight speeds of only 100 miles per hour above the speeds of current general aviation aircraft. It also has the potential of extending these flight speeds to at least 450 mph.

CONCLUDING REMARKS

The foregoing sections of this paper have summarized the work in progress which is intended to provide construction techniques and designs for gas turbine engines which could be both reliable and low in cost. This paper has been intended to provide a review of the program and a progress report on recent experience and, consequently, is not intended to provide major conclusions and recommendations.

It does appear evident, as the foregoing discussion has presented, that the simple, low pressure ratio, low temperature, fan jet and turbojet engines described will provide adequate performance for both general aviation aircraft and for service missiles and drones. Experimental confirmation of these predicted performances as well as more information on reliability and operating problems will be obtained upon completion of the test programs now in progress and planned.

The planned tests for the turbojet engine include the following: (1) completion of the current component matching tests and then measuring engine performance in the sea level test rig over a range of simulated flight Mach numbers; (2) demonstration of the overall engine performance and operation with the new type of hydromechanical control system previously described; (3) testing of the engine and control in an altitude test chamber over a wide range of simulated altitudes and flight Mach numbers. Following these tests, it is expected that the engine will be operated and evaluated in an unmanned flight program to be conducted by the Naval Weapons Center at China Lake, California.

In addition to these tests, work will also continue on the sheet-metal axial compressors, fan rotors, and turbines. Fabrication feasibility of this type of construction, its comparative strength, and its aerodynamic and mechanical performance in operating engines, will be

established. Further work is also anticipated in reducing the cost of casting the compressor and turbine rotors by use of lower cost materials and different processes. Design work on complete fanjet engines for application to general aviation will also continue. Development testing will be performed on critical components for this type of engine.

The most critical factor on a low cost engine is, of course, the price. Previous papers have discussed the preliminary price estimates which indicated that manufacturing prices of $5 per pound of thrust should be attained for the missile-type engines and $10 per pound for the general aviation quality turbofan engine. These papers have emphasized that three factors are essential in achieving such low costs. These are: (1) Use simple, moderate pressure ratio designs, like those previously described; (2) Use fabrication techniques suitable for rapid, quantity production; (3) Actually produce large quantities.

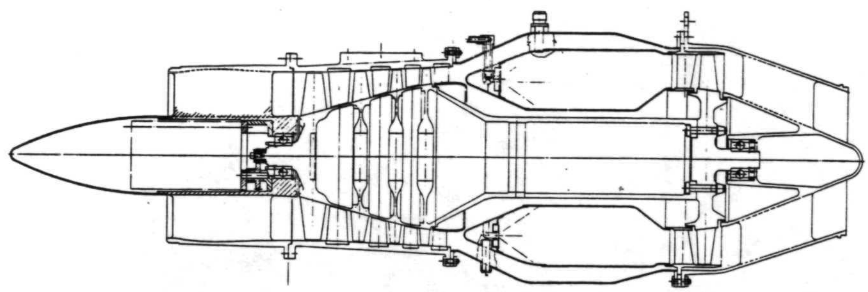
The success of applying these factors for achievement of low specific cost (dollars per pound thrust) has been shown by numerous production engines. For example, the J-33, J-35, J-44, J-47, J-57, and the J-69 engines were all produced to sell at less than $10 per pound of thrust. The J-33, which sold for $4 per pound thrust in the fifties would still be attractively priced in 1972 dollars. The J-69-T29 engine, which sold for $9 per pound of thrust in the late sixties, demonstrates the value of applying the above factors and also the value of using designs having a high thrust per unit frontal area. Here the thrust increase provided by adding the transonic supercharging stage to the basic J-69 engine also resulted in a very substantial reduction in specific cost.

Further verification of the cost potential of the designs currently being investigated in our program cannot as yet be made, since the quantities produced are very limited. It is expected, however, that further analysis of the component prices and the machining time requirements will allow our cost estimates to be refined and extended as our prototype engine assembly and testing program proceeds. It is, therefore, expected that in the near future the engine test results and refined cost estimates will allow defining both the performance and the cost expectations for gas turbine engines which would be suitable for providing substantial performance improvements in small business and general aviation aircraft.

E-6985

650 LB. SLS THRUST

NAVY ORDNANCE ENGINE



CS-56759

Figure 1

NASA ORDNANCE ENGINE—MOUNTED ON THRUST STAND

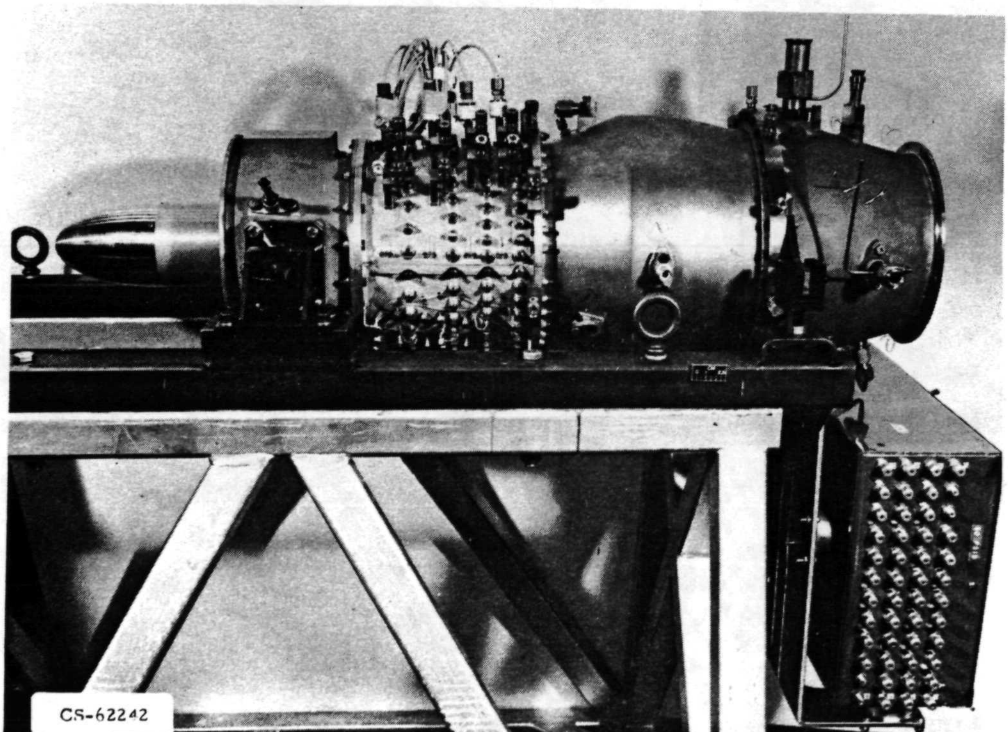


Figure 2

GEARED FAN JET CONFIGURATION

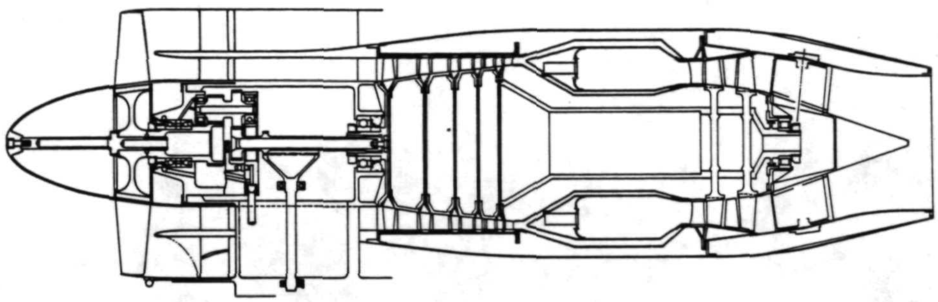


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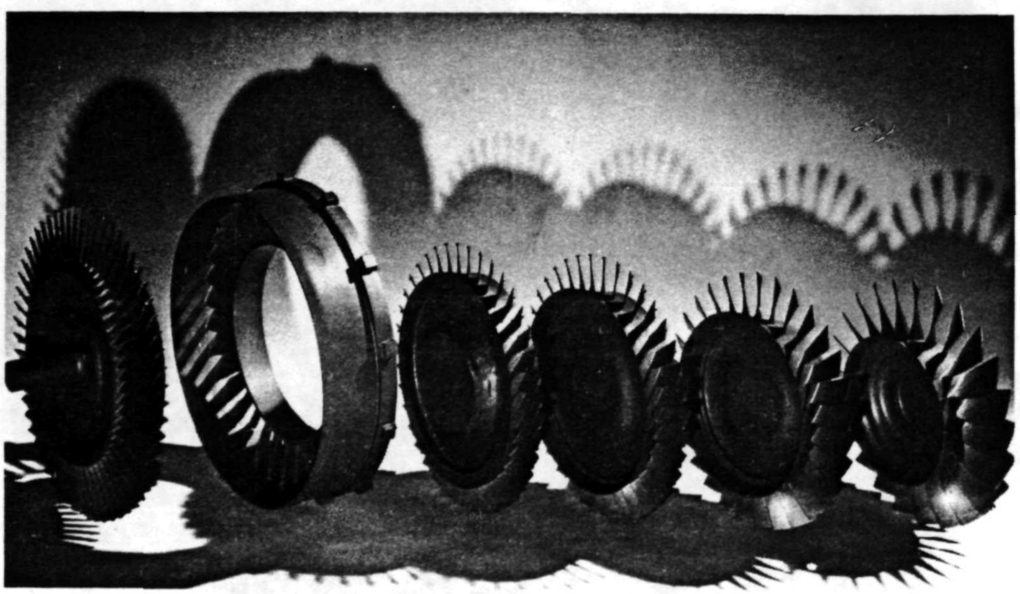
Figure 3

NASA ORDNANCE ENGINE COMPRESSOR ROTOR AND TURBINE CASTINGS



CS-62243

Figure 4

E-6985

CAST COMPRESSOR AND TURBINE ROTOR

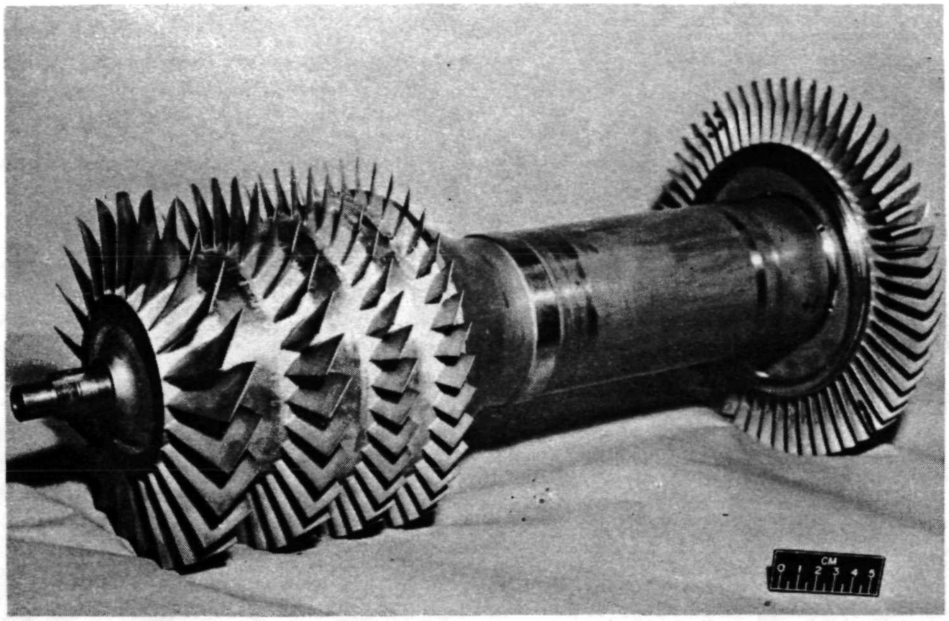


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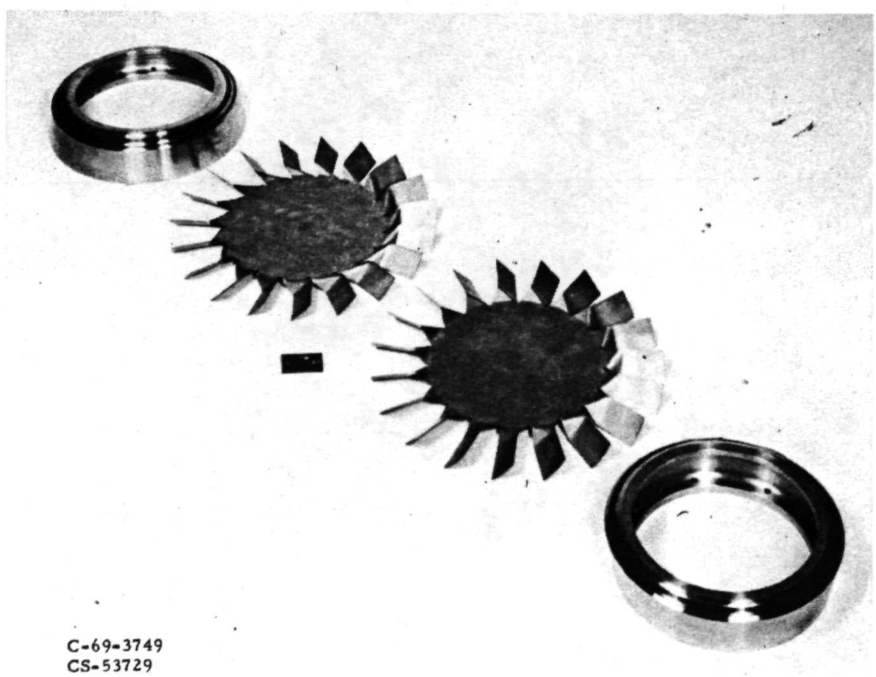
Figure 5

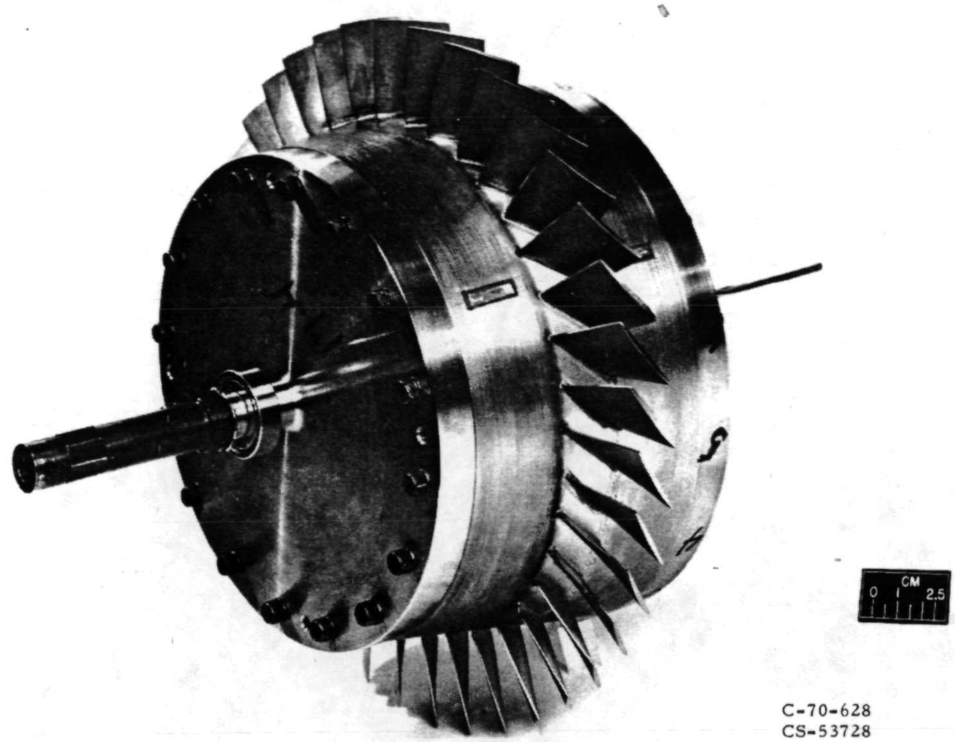
SHEET METAL COMPRESSOR
ROTOR COMPONENTS



C-69-3749
CS-53729

Figure 6

ASSEMBLED SHEET METAL COMPRESSOR ROTOR



C-70-628
CS-53728

Figure 7

SHEET METAL BLADED FAN ROTOR

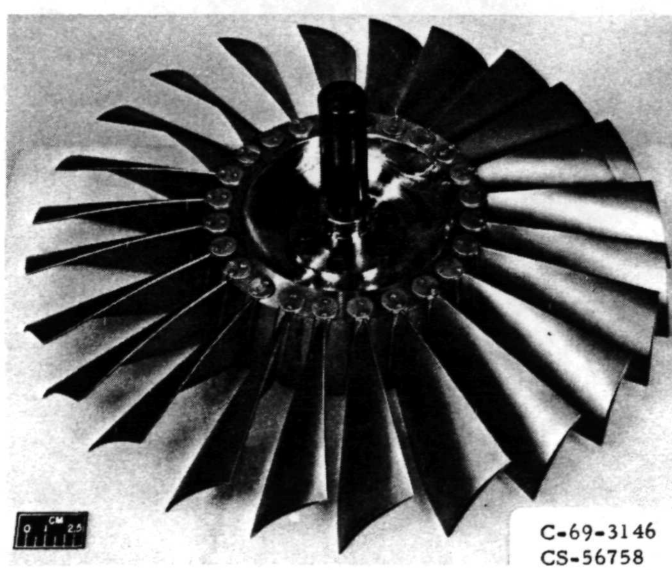


Figure 8

SHEET METAL TURBINE CONSTRUCTION

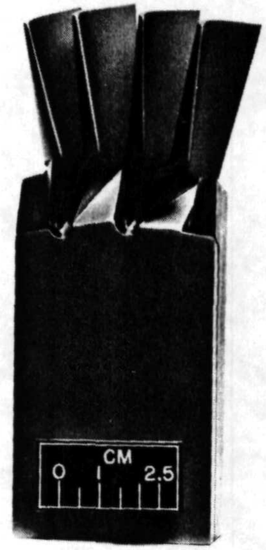


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Figure 9

LOW COST ENGINE COMBUSTOR

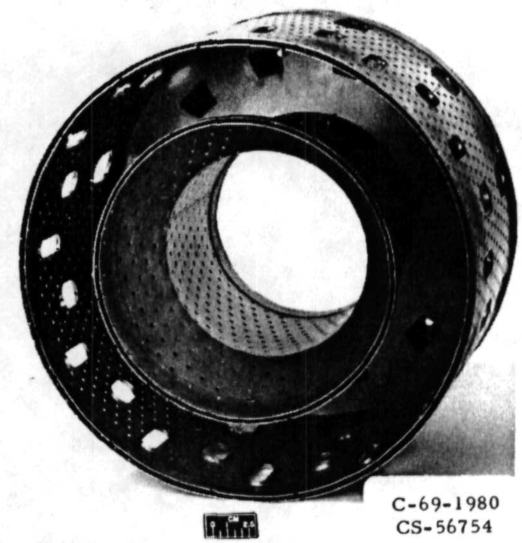

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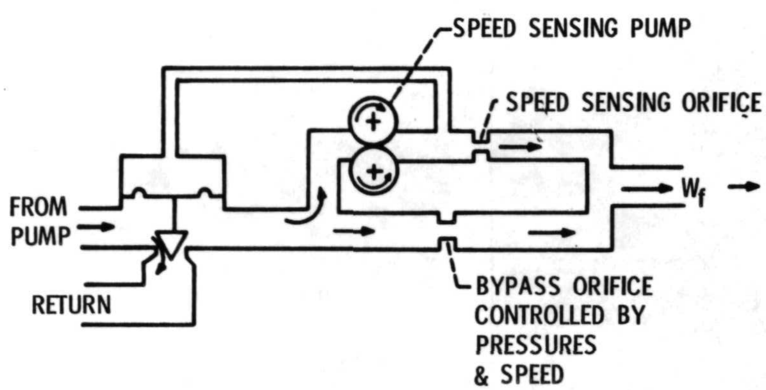
Figure 10

ZERO GRADIENT PUMP FUEL CONTROL



CS-56770

Figure 11

FUEL CONTROL COMPONENT PARTS

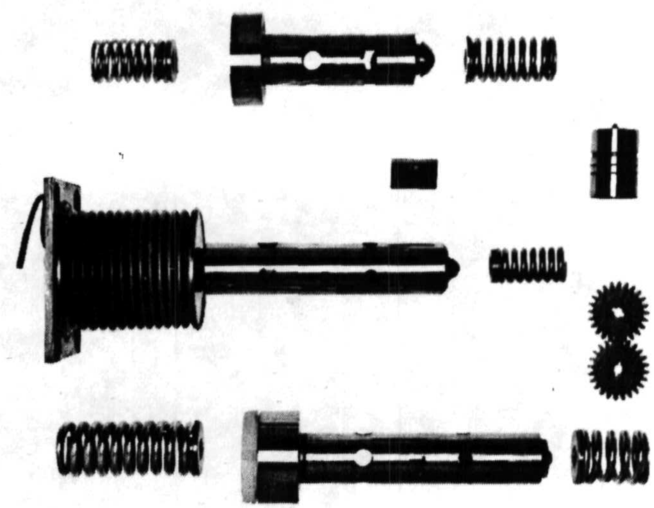


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Figure 12

ENGINE IN TEST STAND

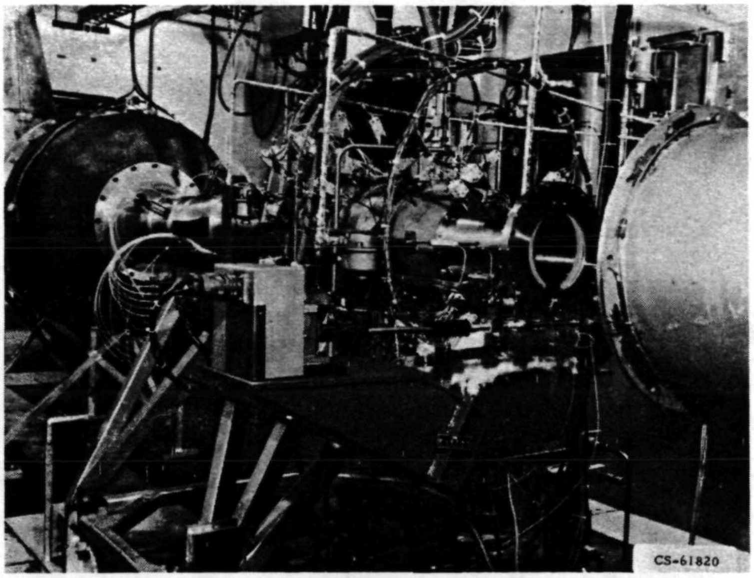

Figure 13

ORDNANCE ENGINE REQUIREMENTS

CRUISE CONDITION M ≈ 0.8 AT 20 000 FT

CRUISE THRUST REQUIRED: 350 LB

SEA LEVEL STATIC THRUST: 650 LB

ENGINE WEIGHT LIMIT: 100 LB

ENGINE DIAM LIMIT: 12 IN.

SPECIFIC FUEL CONSUMPTION <1.8

WINDMILL START AT CRUISE

GAS IMPINGEMENT START AT SLS

FLIGHT DURATION: 15 MIN

CS-56760

Figure 14

E-6985

FANJET POWERED
LIGHT AIRPLANE PERFORMANCE

DESIGN CRUISE: 450 MPH AT 25 000 FT

AIRPLANE CHARACTERISTICS
- 6000-LB GROSS WEIGHT, TWIN ENGINES
- 2000-LB FUEL WEIGHT, INCLUDING 3/4-HR RESERVE
- TAKEOFF THRUST: 1050 LB/ENGINE
- CRUISE THRUST: 350 LB/ENGINE
- TAKEOFF WING LOADING: 40 LB/FT^2 AT C_L = 2.5
- TAKEOFF & LANDING SPEED: APPROX 80 MPH

PERFORMANCE AT CRUISE SFC = 0.90

TAKEOFF DISTANCE, FT	USEFUL RANGE, MI	COMPARATIVE FUEL COSTS (RECIP ENG = 1.0)
1050	1070	0.81

CS-56764

Figure 15

FANJET LIGHT AIRCRAFT ENGINE PERFORMANCE ESTIMATES

ENGINE CHARACTERISTICS

SEA LEVEL STATIC THRUST (APPROX), LB	1100
S. L. CORE ENGINE AIRFLOW, LB/SEC	10.3
FAN BYPASS RATIO	4.0
OVERALL ENGINE PRESSURE RATIO	6.0
FAN STAGE PRESSURE RATIO	1.40
TURBINE INLET TEMP, °F	1500
SINGLE STAGE FAN DIAM, IN.	16.6
CORE ENGINE DIAM, IN.	12.0
ENGINE LENGTH, IN.	42.0
APPROXIMATE ENGINE WEIGHT, LB	200

PERFORMANCE AT 25 000 FT

FLIGHT SPEED, MPH	TSFC, LB/HR/LB	THRUST, LB	THRUST POWER, HP
250	0.693	420	279
300	.734	408	325
350	.770	399	371
400	.806	395	420
450	.822	395	472

CS-62293

Figure 16