May 15, 1962 H. V. WHITE 3,034,298

TURBINE COOLING SYSTEM

Filed June 12, 1958 3 Sheets-Sheet 1

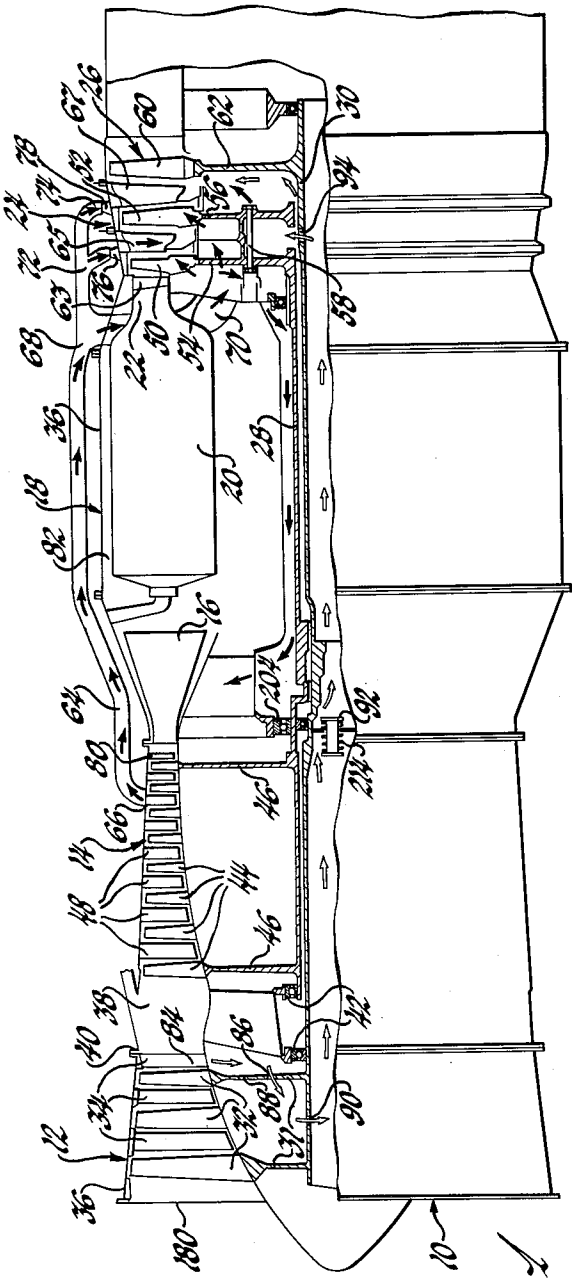


Fig. 1

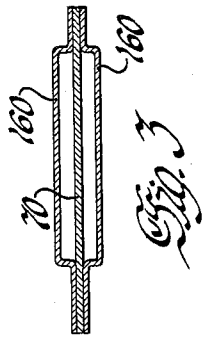


Fig. 3

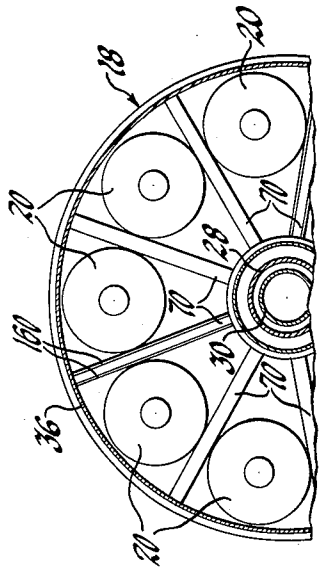


Fig. 4

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TURBINE COOLING SYSTEM

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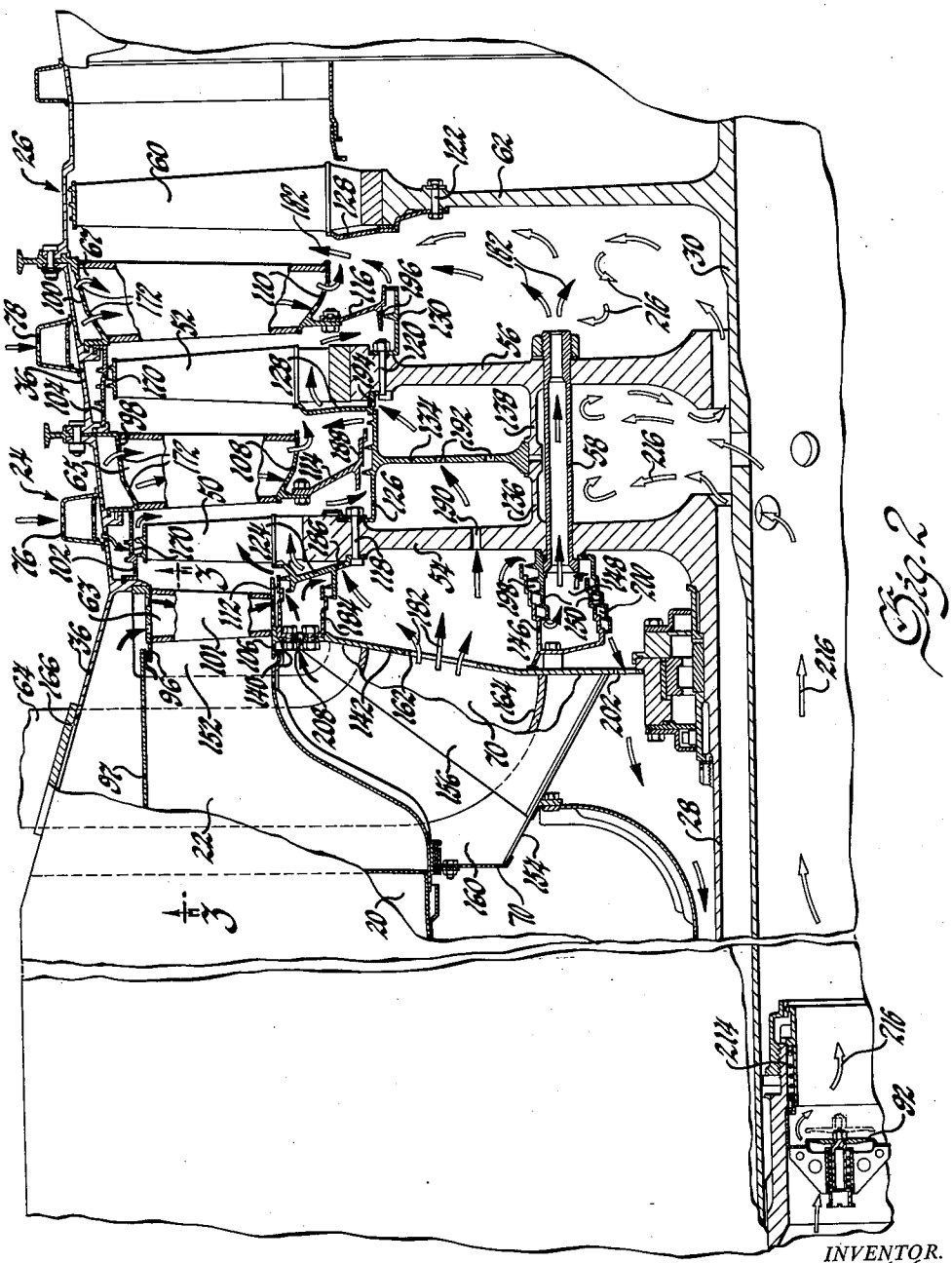


Fig. 2

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TURBINE COOLING SYSTEM

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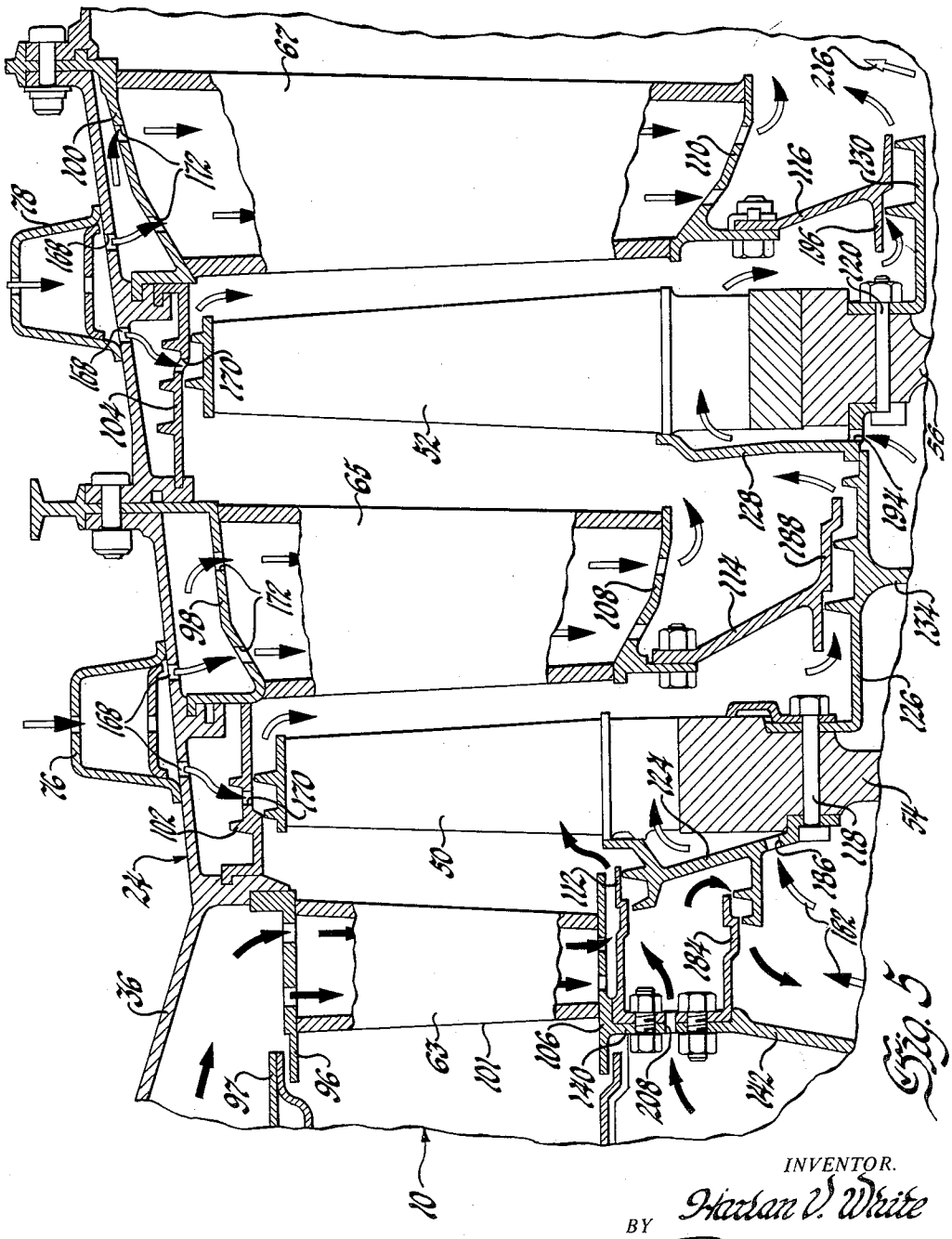


Fig. 5

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TURBINE COOLING SYSTEM

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Filed June 12, 1958, Ser. No. 742,466
5 Claims. (Cl. 60—39.66)

This invention relates to means for cooling the turbine of a gas turbine engine.

More specifically, this invention relates to cooling the turbine by passing lower temperatured air therethrough from the compressor. For turbine cooling purposes in an engine having multi-stage axial flow compressors and turbines, it is desirable to use air that is as cool as possible but having sufficient pressure to force its passage through the higher pressure turbine stages. At low altitude, low speed operation, the later stages of the compressor are the earliest stages having sufficient pressure to accomplish this. In the operation at higher Mach number flight speeds, however, the pressure gain because of ram effect in the intermediate stages relative to these later stages make the cooler air from these intermediate stages useful to cool the turbine. At high altitudes and high flight speed operation, the air from these latter stages becomes relatively hot, and it therefore becomes desirable to use the cooler air from an early stage of the compressor to aid in cooling the turbine.

Attempts have been made in the past to cool the turbine by air from the compressor, resulting in a compromise, with the use of the air from a single stage of the compressor to cool the turbine. This has been unsatisfactory as will be clear from a consideration of the problems mentioned above, since the use of air from a single stage or even two stages at some time during the operation either has insufficient pressure to force passage through the turbine, or is not cool enough to accomplish its purpose. This results in overheating and malfunction of the engine, with a resultant dangerous operating condition of the turbine.

Therefore, it is an object of this invention to provide means permitting cooling the turbine of a gas turbine engine by passing air therethrough from various stages of the compressor at the necessary pressure and temperature level, according to the operating conditions.

Other features, advantages and objects will become apparent by reference to the detailed description of the invention and to the drawings wherein there is shown a preferred embodiment of this invention.

FIGURE 1 is a diagrammatic view of a gas turbine engine embodying this invention, with parts broken away,

FIGURE 2 is an enlarged view of one-half of the turbine section of the engine of FIGURE 1 with parts broken away and in section,

FIGURE 3 is a cross-sectional view of a detail obtained by passing a plane through the FIGURE 2 construction as shown, and viewed in the direction of the arrows 3—3,

FIGURE 4 is a diagrammatic cross-sectional view of a detail obtained by passing a plane through a portion of the combustion chamber, and

FIGURE 5 is an enlargement of a portion of FIGURE 2.

Referring now to the drawings, and more particularly to FIGURE 1, there is shown therein diagrammatically a dual-spool gas turbine engine **10** illustrating the preferred embodiment of this invention. The engine includes a low pressure compressor section **12** and a high pressure compressor section **14** axially aligned therewith for the passage of air therethrough, the air being discharged therefrom through a diffuser section **16** into the combustion section **18**, where the air is properly mixed with fuel to be ignited in a combustion can **20**. While

only one combustion can is shown for clarity, it will be clear that normally six or more combustion cans annularly arranged and equally spaced from each other are provided, with the ignition therein of the fuel being accomplished through the use of suitable conventional crossover tubes (not shown). The extremely hot gases in the combustion cans are then delivered through a transition section **22** to the high and low pressure turbine sections **24** and **26** respectively, to drive corresponding turbine shafts **28** and **30**, respectively.

The low pressure compressor section **12** comprises a three stage axial flow compressor having three annular rows of rotor blades **32** connected to the low pressure turbine shaft **30** by disks **31**. Separating the rotor blades **32** in the conventional manner and completing the three stages of the low pressure compressor section are the three rows of stator vane members **34** connected to and supported by the outer engine case **36**, the case being connected to the compressor stage separating section **38** by suitable flange means **40**. The separating portion **38** rotatably supports the two turbine shafts **28** and **30** through suitable bearing means **42**. The high pressure compressor section **14** comprises an eight stage axial flow compressor having suitable rows of rotor blades **44** mounted on disks **46** connected to the high pressure turbine shaft **28** for rotation therewith. Suitably positioned between the rows of rotor blades **44** are an equal number of rows of stator vanes **48** secured to the outer engine case **36**.

Referring now to the turbine section illustrated in FIGURE 1, the high pressure turbine section comprises a two stage axial flow turbine having two rows of rotor blades **50** and **52** mounted on disks **54** and **56**, the disks being connected together by a number of circumferentially spaced bolts **58** (only one shown). The disk **54** is either formed integrally with the high pressure turbine shaft **28**, or fixed thereto for rotation together by any suitable means. The low pressure turbine section **26** comprises a single stage axial flow turbine having a single row of annular turbine blades **60** supported on a disk **62** in turn formed integrally with or suitably splined to the low pressure turbine shaft **30**.

Suitably positioned between the turbine blade rows of the high and low pressure turbine sections are the rows of the hollow stator vane assemblies **63**, **65** and **67** suitably supported by and connected to the outer engine case **36**.

Since the engine depicted in FIGURE 1 is merely illustrative of an engine to which this invention may be applied, further details of the engine are believed unnecessary at this time. While a somewhat more detailed description of the turbine structure will be given later in connection with the cooling thereof, additional details of the turbine section are described and shown in more detail in Serial Number 746,051, filed July 1,1958, Turbine Mounting Construction by F. G. Koziura.

With respect to the cooling system for cooling the turbine illustrated in FIGURE 1, air from the compressor is delivered through a number of suitable exterior conduits **64**, preferably three spaced 120° apart, each having one end connected at **66** through a 60° circumferential inlet to receive a portion of the 9th stage compressor air discharged thereinto, with its other end being divided at **68**, one portion of the air being delivered through a number of struts **70**, preferably three, to the interior of the turbine, and other portions **72** and **74** being connected to annular manifolds **76** and **78** fixed to the turbine case and opening into communication with the rotor blade shrouds and hollow stator vanes therein. As will be seen from FIGURE 1, the air discharged from the 11th or final stage **80** of the compressor passes into the jacket space **82** between the combustion can **20** and outer engine case **36** as well as into the can, thereby supplying the

interior of the turbine with air from the 11th stage. Also, as seen in FIGURE 1, a portion of the air from the third stage 84 of the compressor is bled internally of the compressor through suitable openings 86 in the third stage rotor wheel 88, and through suitable openings 90 in the hollow low pressure turbine shaft 30 to pass therethrough and through a one-way check valve 92 and suitable openings 94 at the opposite end of the shaft 30 into the interior of the turbine section. A more detailed explanation of this cooling system and its operation will be given presently upon a consideration of the details of FIGURES 2 through 5.

Referring now to FIGURE 2, the turbine section is shown enclosed by the outer engine case 36 having a suitable configuration for supporting the individual stator vane assemblies, rotor blade shroud rings, and interstage stationary seal portions. As seen in this figure, the stator vane assemblies 63, 65 and 67 are each provided, respectively, with outer annular shroud portions 96, 98 and 100 connected to the outer engine case by suitable flanges and bolts, and axially spaced from each other by abutting annular rotor blade shroud rings 102 and 104 also connected to the outer engine case 36. The stator vane assemblies include rows of hollow stator vanes having inner annular shroud portions 106, 108 and 110, respectively, supporting stationary annular portions 112, 114 and 116, respectively, of stage labyrinth seal means. Each of the row of rotor blades 50 and 52 of the high pressure turbine assembly 24 supports rotating portions of the annular labyrinth seal means 124, 126 and 130 for cooperation with the stationary portions of the seal to prevent the escape of the hot gases from the turbine nozzle into communication with the turbine bearings and support members. These rotating portions of the seals are fixed to the turbine disks by a number of circumferentially spaced bolts 118, 120, and 122. The bolts also secure to the disks a number of heat insulating baffles or shields 128 on the upstream side of the disks 56 and 62 between the disks and blade platforms. As shown, the rotating seal portion 126 is formed centrally thereof with an annular stiffener 134 splined or otherwise connected to the axial extensions 136 and 138 of the disks 54 and 56 of the first and second stages of the high pressure turbine 24.

As seen in this figure, the stationary double seal portion 112 of the first stage stator vane assembly 63 is connected at 140 to an annular bulkhead 142 extending radially inwardly towards the engine axis for connection with the front or forward turbine bearing supporting the high pressure turbine shaft 28. Also connected to the bulkhead 142 is the stationary portion 146 of a double acting labyrinth seal 148 cooperating with the two rotating portions 150 of the seal secured to disk 54 by the hollow tie bolts 58.

At the left portion of FIGS. 2 and 5, the converging transition section 22 of the combustion cans 20 is shown having an appropriate shape and outlet 152 cooperating with the turbine nozzle defined by the stator vanes of assembly 63. The combustion cans and transition section are supported at this point by suitable annular stiffener means 154 connected at one edge to the combustion can and at its other edge to the bulkhead 142, with a number of gussets 156 attached both to the stiffener 154 and bulkhead 142.

FIGURE 4 illustrates schematically a partial cross-sectional view of the combustion section. As seen in this figure, and in FIGURE 2, the combustion cans 20 are separated from each other by suitable struts 70 secured at one end to the engine case adjacent the connection to air conduit 64, and at their opposite ends to the bulkhead 142 and the annular stiffener 154. As seen in FIGURES 2 and 3, a sheet metal hat section 160 is suitably fastened to strut 70 on opposite sides thereof for the passage of cooling air therethrough from the compressor to the interior of the turbine. While struts are shown in FIGURE 4 on each side of the combustion cans, only

three need by provided with the hat sections for cooling purposes to connect with the three conduits 64 (only one shown). Each cooling strut is curved at its radially inward portion adjacent the bulkhead 142 to connect the hollow chambers 164 formed by the hat sections 160 and strut 70 with the interior of the turbine through openings 162 in the bulkhead. The outer case 36 is apertured at 166 directly below the connection of conduit 64 to the case to connect the 9th stage compressor air in conduit 64 to the chambers 164. Thus, 9th stage air is delivered to the interior of the turbine through the struts 70. 9th stage air also passes through manifolds 76 and 78, openings 168 in the engine case 36, openings 170 in the rotor blade shroud rings 102 and 104, openings 172 in the hollow stator vane shroud rings 98 and 100, and through the hollow stator vanes of assemblies 65 and 67 to cool the turbine parts.

As seen in FIGURE 2, the shape of the transition section liner conforms to the shape of the turbine nozzle defined by the stator vane assembly 63, thereby providing a space between the liner 97 and case 36 for the passage of 11th stage air from the compressor to the hollow stator vanes 101. Also, since 11th stage air surrounds the cans 20, the liner 97 cooperates with the inner shroud 106 of vane assembly 63 and bulkhead 142 to provide an additional chamber or passage therebetween for the flow of 11th stage air through throttling orifices 208 in the bulkhead into the turbine for a purpose to be described.

Operation

Referring now to the cooling system as a whole, arrows have been provided in FIGURE 2 to illustrate the approximate direction of movement of the cooling fluid, the solid arrows indicating the movement of 11th stage compressor air, the partly solid arrows representing the direction of movement of the 9th stage air, and the hollow arrows indicating the direction of movement of the third stage air.

The stages of the axial flow low and high pressure compressors shown in FIGURE 1 progressively increase the pressure and temperature of the air as it progresses from the inlet 180 to the last stage 80, while the stages of the high and low pressure turbine sections progressively decrease the pressures and temperatures of the motive fluid therein from the turbine nozzle to the last stage. The choice as to which stage or zone of the compressor the air will be taken from to cool the turbine will depend upon the particular area or zone of the turbine to be cooled. It is desirable to use the coolest air possible for cooling the turbine; however, this air must be at a pressure sufficient to force its passage through the particular turbine section, as mentioned previously. For example, during sea level, low flight speed operation, to cool the turbine nozzle portion, which is surrounded by motive fluid at the highest pressure, cooling air having a higher pressure must be used to effect circulation of the cooling air through this portion of the turbine. Therefore, the high pressured yet lower temperatured 11th stage air would be used at this point in the turbine and surrounding points at the same pressures. In other zones of the turbine where the pressures and temperatures are lower, combinations of 9th and 11th stages air or 9th stage alone would have the sufficient pressure to effect circulation of cooling air to cool these parts of the turbine. It will be seen therefore that the cooling air to be used will depend upon the pressures in the zones of the turbine to be cooled, with the use of the coolest air possible having the sufficient pressure to effect circulation.

During higher flight speed operations, 11th and 9th stages air, while still having the sufficient pressure, become hotter and their effectiveness in cooling the turbine zones to which they are distributed is less. Therefore it would be desirable to use air from a cooler stage of the compressor to cool these particular zones of the turbine. In this stage of operation, the ram pressure of the air in the inlet to the compressor has increased the pressure in

the third stage of compression, for example, to a point where it has a pressure sufficiently higher than some portions of the turbine so that use can be now made of this cooler air.

The overall operation of the cooling system therefore selectively makes use of cooling air from the various stages of the compressor depending upon the particular pressure and temperature conditions of the turbine zones to be cooled and the pressure and temperature conditions of the stages of the compressor from which the cooling air is withdrawn.

Referring now to FIGS. 2 and 5 for a detailed study of the flow of the cooling air, it will be seen that a portion of the air from the 9th stage will flow into conduits 64 and into the struts 70 and manifolds 76 and 78 to communicate with the interior of the turbine section. As shown by the arrows 182, 9th stage air entering through the struts 70 and bulkhead 142 flows to the labyrinth seal 184, through an aperture 186 in seal member 124 to cool the portions between the platforms of the blades 50 and the disk 54, and through the labyrinth seal 188 to flow as indicated and cool the adjacent parts. At the same time, 9th stage air entering through the bulkhead 142 passes through suitable apertures 190, 192 and 194 in the disk 54, stiffeners 134 and baffle 128, respectively, to cool the turbine elements in the intermediate pressure area and then forces passage through the labyrinth seal 196 to return to the main stream of motive fluid. 9th stage air also passes through the labyrinth seal 198 into the hollow tie bolts 58 to aid in cooling the low pressure stator vane assembly 67 and the assembly for the rotor blades 60 along with disks 56 and 62. 9th stage air further passes through the labyrinth seal 200 and an aperture 202 in the bulkhead to pass back along the turbine shaft 28 as shown in FIGURE 1 to cool the same and the mid-frame or rear compressor bearing 204 supporting the turbine shaft 28. As seen also in FIGURES 2 and 5, the 9th stage air entering the manifolds 76 and 78 flows through the holes 168 and 170 in the engine case and the stationary portions of the turbine blade shroud rings to cool the rotating portions of the shroud rings and the outer peripheral portions of the turbine rotor blades. The cooling air from these manifolds also passes through the holes 172 in the two stator vane shroud rings 98 and 100 to enter the hollow stator vanes, passing therethrough to cool the inner portions thereof and the rotor blade assemblies in the intermediate and low pressure areas as indicated. This flow as indicated by the partly solid arrows in FIGURES 2 and 5 is possible because of the higher pressures of the 9th stage compressor air compared to that of the turbine air in the zones under consideration.

Simultaneously with the flow of 9th stage air to the turbine, a portion of the 11th stage air will be fed through the hollow stator vanes 101 to the seal 184 to cool the adjacent parts in the high pressure area. 11th stage air also is bled through the metered openings 208 in the bulkhead to the opposite sides of seal 184. Even though the pressure of the 11th stage air is reduced by being metered through openings 208, it is still higher than 9th stage air, i.e., say 115 p.s.i. for 11th stage air, as compared to 110 p.s.i. for 9th stage air, for example. Therefore, 11th stage air leaks out between the stationary and rotating sections of both parts of labyrinth seal 184 as indicated by the solid arrows in FIGURES 2 and 5 to circulate as shown thereby preventing the hot motive fluid from flowing radially inwardly, while at the same time cooling the adjacent portions of the stator vane assembly 63 and the turbine rotor assembly 50 as well as bulkhead 142, disk 54 and surrounding elements. The pressure differential between 9th and 11th stage air will be dissipated by flow losses upon leakage of the air through the seals thereby preventing the flow of 11th stage air through the 9th stage air conducting means. As seen in FIGURES 2 and 5, the 11th stage air flowing through seal 184 is continuously mixed with the 9th stage air entering through the bulkhead 142. As stated, at low flight speeds, the pressure differential between 11th and 9th stages air is large. As the flight speed increases, this pressure differential decreases rapidly, i.e., the gain in 9th stage air pressure due to ram effect with an increase in Mach number flight speeds is such that at high Mach number speeds, 9th stage air pressure has attained a value close to that of the 11th stage pressure. Therefore, the 9th stage air has gained sufficient pressure relative to the 11th stage air pressure gain to almost equal that of the metered 11th stage air at the seals 184; or, in fact, the gain may be sufficient to reverse the direction of leakage of air through the seals 184 to substantially eliminate 11th stage air from the cooling circuit. Therefore, the mixture of 9th and 11th stages air in this zone of the turbine will vary with the decrease in the pressure differential between the two stages. The proportion of the cooler 9th stage air in this mixture therefore increases with an increase in flight speed to a condition where the mixture contains practically all 9th stage air. Since the 9th stage air may be 100° cooler, for example, than 11th stage air, this large proportion of 9th stage air is desirable. To illustrate the change in the mixture, in the engine under consideration, during low flight speeds, low altitude operation, the mixture might consist of 50% 11th stage air and 50% 9th stage air; while at high Mach number flight speeds, the mixture might consist of approximately 10% 11th stage air and 90% 9th stage air or even all 9th stage air.

At high altitude, higher flight speed or higher Mach number operation, the 11th and 9th stages air become fairly hot and are not as desirable to use for lowering the temperature in the lower temperatured portions of the turbine as under low speed operating conditions. It therefore becomes desirable to utilize a cooler source of cooling air such as that from the earlier stages of the compressor, i.e., the 3rd stage. Under low speed operation, the pressure of the air in the 3rd stage is lower than the low pressure zone of the turbine where the air is to be introduced. For this reason, a one-way check valve 92 is provided for preventing the flow of higher pressured air from the turbine to the compressor under low speed operation, while permitting flow of the cooler higher pressured compressor air to the turbine under high speed operation. The one-way check valve is of a conventional type having a spring 214 normally closing the check valve and opening upon sufficient pressure of the compressor 3rd stage air. Under high altitude, high operating speeds, the pressure of the ram air entering the inlet 180 of the compressor is sufficient by the time it passes through the third stage of compression 84 to force its passage into the appropriate lower pressured turbine sections through openings 86 in disk 88, openings 90 in shaft 30, the check valve 92 and shaft openings 94 to mix as indicated by the hollow arrows 216 with the 11th and 9th stage air to cool the turbine parts as indicated. The temperature of the air from the third stage is cooler than any of the motive fluid in the turbine, and therefore will maintain the turbine cool at this time.

It will therefore be seen that this invention provides a cooling system whereby the turbine is cooled by the passage of air therethrough from various cooler stages of the compressor, 9th and 11th compressor stage air being utilized continuously throughout the entire operating range of the engine and supplemented with cooler third stage compressor air during higher altitude, higher speed operations. Circulation of cooling fluid is therefore always provided in the turbine section because of the use of the higher pressured and lower temperatured air from the various compressor stages flowing through the various turbine stage seals, hollow stator vanes and the turbine shaft to cool the turbine elements and bearing members. This invention therefore insures that the turbine section will not be overheated and that no injurious operation of the engine will be permitted.

While the invention has been illustrated in its preferred

form in connection with a dual-spool gas turbine engine, it will be clear that many modifications can be made therefrom by one skilled in the arts to which this invention pertains without departing from the scope of the invention.

I claim:

1. A jet engine of the reaction motor type having a compressor subject to ram air effects and a turbine axially aligned and each of the high pressure multi-stage axial flow type, conduit means connected between stages of said turbine and compressor for cooling the several stages of said turbine by the flow thereto of cooler higher pressure fluid from several stages of said compressor, means at one portion of said turbine to intermix the higher pressure fluids from several compressor stages to maintain the one portion cool, means to reduce the cooling fluid pressure differential between the several compressor stages upstream of said intermixing means, the pressure gain due to ram air effects of each of the several compressor stages upon increase in engine flight speed decreasing progressively from the lower to the higher pressure ends of said compressor to progressively further decrease the pressure differential between the fluids in the mixture to thereby vary the relative proportion of each of the fluids in the mixture.

2. A jet engine of the reaction motor type having a compressor subject to ram air effects and a turbine axially aligned and each of the high pressure multi-stage axial flow type, conduit means connected between said turbine and compressor for cooling the high and intermediate pressure stages of said turbine by the flow thereto of cooler higher pressure fluid from the later and intermediate stages, respectively, of said compressor, means adjacent the high pressure stage of said turbine to continually mix the higher pressure fluids from the later and intermediate compressor stages to maintain the turbine high pressure stage cool, means to reduce the cooling fluid pressure differential between the later and intermediate compressor stages upstream of said mixing means, the pressure gain due to ram air effects of the compressor intermediate stage upon increase in engine flight speed being greater than the pressure gain of said later compressor stage thereby progressively further decreasing the pressure differential between the fluids in the mixture to thereby vary the relative proportion of each of the fluids in the mixture.

3. A jet engine of the reaction motor type having a compressor subject to ram air effects and a turbine axially aligned and each of the high pressure multi-stage axial flow type, conduit means connected between said turbine and compressor for cooling the high and intermediate and low pressure stages of said turbine by the flow thereto of cooler higher pressure fluid from the final and intermediate and initial stages respectively of said compressor, means adjacent the high pressure stage of said turbine to continually mix the higher pressure fluids from the final and intermediate compressor stages to maintain the turbine high pressure stage cool, means to reduce the cooling fluid pressure differential between the final and intermediate compressor stages upstream of said mixing means, the pressure gain due to ram air effects of the compressor intermediate stage upon increase in engine flight speed being greater than the pressure gain of said final compressor stage thereby progressively further decreasing the pressure differential between the fluids in the mixture to thereby vary the relative proportion of each of the fluids in the mixture, and means in the conduit means between said initial compressor stage and said low pressure turbine stage preventing flow of fluid from said initial stage below a predetermined pressure of the fluid in said initial stage.

4. A jet engine of the reaction motor type having a compressor subject to ram air effects and a turbine axially aligned therewith and each of the high pressure multi-stage axial flow type, conduit means connected between said turbine and compressor for cooling the high and intermediate pressure stages of said turbine by the flow thereto of cooler higher pressure fluid from the later and intermediate stages respectively of said compressor, means adjacent the high pressure stage of said turbine to continually mix the higher pressure fluids from the later and intermediate compressor stages to maintain the turbine high pressure stage cool, means defining a fluid cooling chamber adjacent the turbine high pressure stage having a fluid inlet communicating with the compressor later stage fluid and a passage communicating with the compressor intermediate stage fluid, pressure reducing means in said inlet reducing the pressure of said final stage fluid in said chamber, restrictive fluid bleed means in said passage restrictively communicating said higher pressure later stage fluid with said intermediate stage lower pressure fluid thereby mixing the two, the pressure gain due to ram air effects of the compressor intermediate stage upon increase in engine flight speed being greater than the pressure gain of said later compressor stage thereby progressively decreasing the pressure differential between the fluids in the mixture thereby varying the relative proportion of each of the fluids in the mixture.

5. A jet engine of the reaction motor type having a compressor subject to ram air effects and a turbine axially aligned and each of the high pressure multi-stage axial flow type, conduit means connected between said turbine and compressor for cooling the high and intermediate pressure stages of said turbine by the flow thereto of cooler higher pressure from the later and intermediate stages respectively of said compressor, means adjacent the high pressure stage of said turbine to continually mix the higher pressure fluids from the later and intermediate compressor stages to maintain the turbine high pressure stage cool, means defining a fluid cooling chamber adjacent the turbine high pressure stage having a fluid inlet communicating with the compressor later stage fluid and a passage communicating with the compressor intermediate stage fluid, pressure reducing means in said inlet reducing the pressure of said final stage fluid in said chamber, restrictive fluid labyrinth seal means in said passage permitting the bleed of the said higher pressure later stage fluid out of said chamber to mix with said intermediate stage lower pressure fluid, the pressure gain due to ram air effects of the compressor intermediate stage upon increase in engine flight speed being greater than the pressure gain of said later compressor stage thereby progressively decreasing the pressure differential between the mixed fluids to thereby vary the relative proportion of each of the fluids in the mixture.

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