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EFFECT OF TWO TYPES OF HELIUM CIRCULATORS ON THE PERFORMANCE OF A SUBSONIC NUCLEAR-POWERED AIRPLANE

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16. Abstract

Two types of helium circulators are analytically compared on the bases of their influence on airplane payload and on propulsion system variables. One type of circulator is driven by the turbofan engines with power takeoff shafting while the other, a turbocirculator, is powered by a turbine placed in the helium loop between the nuclear reactor and the helium-to-air heat exchangers inside the engines. Typical results show that the turbocirculator yields more payload for circulator efficiencies greater than 0.82. Optimum engine and heat exchanger temperatures and pressures are significantly lower in the turbocirculator case compared to the engine-driven circulator scheme.

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EFFECT OF TWO TYPES OF HELIUM CIRCULATORS ON THE PERFORMANCE OF A SUBSONIC NUCLEAR-POWERED AIRPLANE

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SUMMARY

Two types of helium circulators are analytically compared on the basis of their influence on airplane payload capacity. One type is driven by the propulsion engines through mechanical shafting while the other, a turbocirculator, is powered by a helium turbine placed between the nuclear reactor and the helium-to-air heat exchangers inside the engines. A 453 600 kilogram, subsonic cargo-type airplane is assumed with a helium-cooled thermal reactor supplying energy to six turbofans.

Results show that a turbocirculator yields more payload than engine-driven circulators only for circulator efficiencies greater than 0.82. In the 0.80 to 0.85 efficiency range there is not enough payload difference between the two schemes to decide which is better. This choice depends on many other factors such as seal problems, fabricability, reliability, cost, operating characteristics, and auxiliary equipment requirements. Helium leakage past the rotating shaft seals of engine-driven circulators, in particular, could be severe enough to cause a large replenishment supply of helium to be carried aboard the airplane. No attempt is made to assess a payload penalty for this circumstance.

The turbocirculator scheme calls for the use of relatively low pressure and temperature engines and heat exchangers compared to the engine-driven scheme. Overall engine pressure ratios around 10 instead of 19 and turbine inlet temperatures about 900 instead of 1040 K are typical for an airplane designed to fly at Mach 0.8 and at a 11-kilometer altitude. Likewise, the helium inlet temperature to the heat exchangers drops from about 1300 to 1130 K and the pressure drops from 12 to 8.7 meganewtons per square meter.

Both the payload and the system design variables are rather sensitive to turbocirculator efficiency because of the substantial energy transmitted by this subsystem. This sensitivity is in sharp contrast with the relatively low-power engine-driven circulators.

INTRODUCTION

Practically unlimited endurance and range are unique capabilities offered by a nuclear-powered aircraft. Recent studies (refs. 1 to 3) of a large subsonic nuclear airplane concern the effects of various factors such as the choice of design point, off-design operation, temperatures, and component lifetimes on payload capability and technology requirements. All of these previous studies assume that helium is used to transfer heat from the reactor to the turbofan engines. The helium is assumed to be circulated by means of a power takeoff from the engines. This study considers the alternative of using a separate helium turbine to drive the helium pump. These two schemes are diagrammed in figures 1 and 2. The engine-driven circulator (fig. 1) requires a power train to transfer power from the inner turbines of the engines to the axial flow helium compressors. Since the engine efficiency is of the order of 25 percent, the amount of reactor power required for circulating the helium is about four times the pumping power. This scheme has at least one serious drawback; namely, the high pressure, high temperature helium gas loop is penetrated by the drive shafting. This requires an extremely effective and reliable rotating seal to prevent excessive helium leakage. Without such a seal, an excessively large supply of helium would have to be carried aboard the airplane to replenish the helium circuit. It is probable that such a seal could be manufactured, although this needs to be verified. It is also probable that the seals would be quite complicated and involve significant reliability and weight penalties - but again, this needs to be clarified through demonstration.

The alternative turbocirculator scheme (fig. 2) employs a helium turbine placed between the reactor and the heat exchangers to drive the helium compressor. This arrangement permits a common shaft for the helium turbine and compressor and avoids the rotating seal problem. A rotating seal is required between the turbine and compressor casings but it is not particularly critical since the leakage gas does not escape the helium circuit. Another advantage of the turbocirculator is that it lowers the heat exchanger temperature and pressure levels (assuming constant reactor outlet conditions) since the helium turbine extracts energy from the working fluid. This alleviates a difficult construction problem for these high temperature, high pressure components.

Two disadvantages of the turbocirculator are the following:

(1) It requires a lubrication system within the helium loop that would require special attention to prevent the lubricant from contaminating the helium.

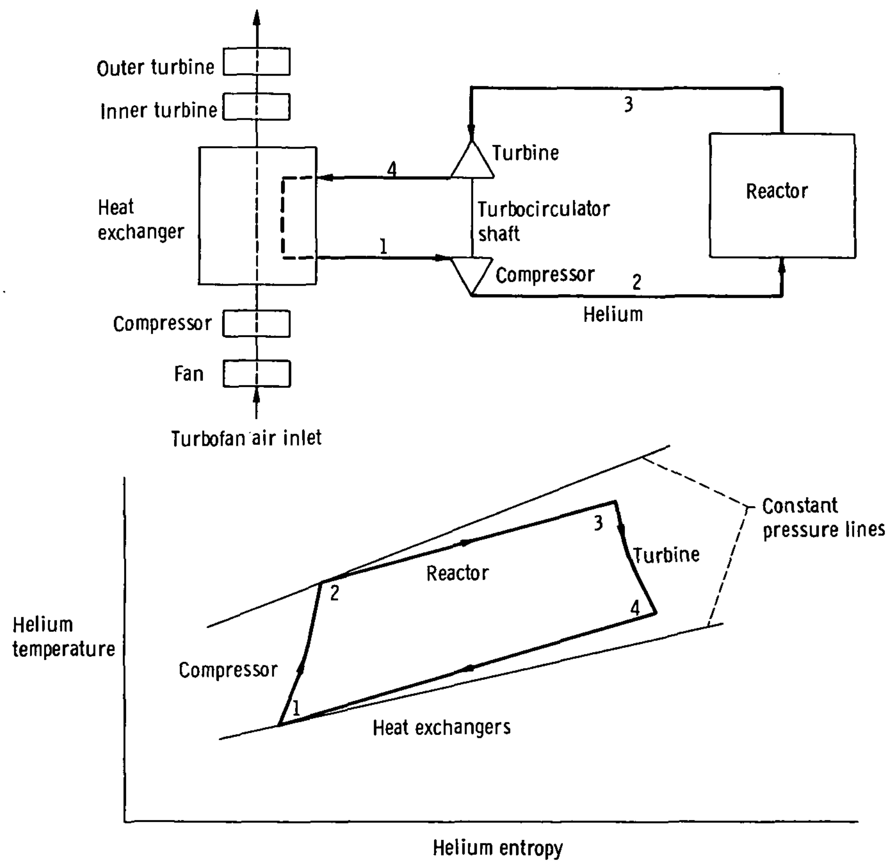
(2) It is neither self-starting nor fully self-sustaining after reactor shutdown; hence, auxiliary equipment is needed for these functions which could lead to other problems.

It is not the intent of this study to explore all of these aspects or cost of the two circulator schemes. Reference 4 contains detailed discussions of turbocirculator aerodynamics, mechanical design, seals, lubrication, assesory system, and stability and control for a stationary powerplant. Some of the problems peculiar to the aircraft application are discussed in reference 5. The only concern here is what advantage, if any, a turbocirculator offers in terms of airplane performance (i. e., payload capacity) and how it would alter key design variables such as maximum heat exchanger temperature and turbine inlet temperature of the engines.

ANALYSIS

Turbocirculator

The turbocirculator operates on the Brayton cycle as diagrammed in sketch (a). Helium is compressed in the compressor along line 1-2; it is heated in the reactor along



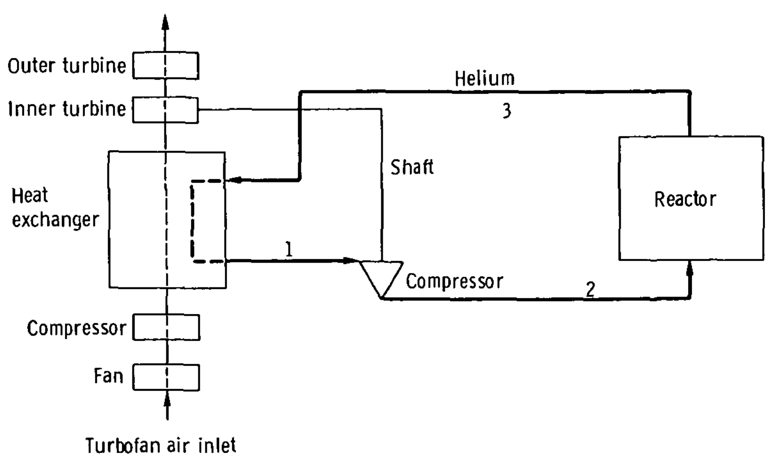
(a)

line 2-3; it expands through the turbine, line 3-4; and it is cooled in the heat exchangers, line 4-1. Ideally, the compression and expansion processes are isentropic and the heating and cooling processes are done at constant pressure. Realistically, of course, these ideals are not achieved. However, since lines of constant pressure diverge on the temperature - entropy diagram as temperature is increased, and the turbine inlet temperature is higher than the compressor inlet temperature, the turbine pressure ratio is less than the compressor pressure ratio for compressor work equal to the turbine work and realistic efficiencies. The excess compressor pressure ratio is used to overcome the other pressure drops in the helium circuit. The pressure drops occur in the reactor, the shield, the heat exchangers, and the helium ducts connecting the various components.

One of the variables free for optimization in this scheme is the heat exchanger helium outlet temperature T_1. There is, of course, a lower limit on T_1 equal to the engine air compressor discharge temperature. In practice, though, a high value of T_1 is desirable (assuming the reactor power and heat exchanger inlet temperature are fixed) from the standpoint of reducing heat exchanger size, even though the helium flow rate increases with T_1. There is, however, an upper temperature limit that depends on the reactor outlet temperature, the overall helium loop pressure drop (exclusive of the helium turbine), and the component efficiencies. This is shown in the thermodynamic analysis in appendix A. The value of T_1 that maximized payload was invariably found to be the same as this upper temperature limit (eq. (11)).

Engine-Driven Circulator

The engine-driven helium circulator is driven by the airplane propulsion engines through mechanical shafting as shown in sketch (b). Each circulator consists of a helium



(b)

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compressor and the shafting that transmits work from the inner turbine of an engine to the helium compressor. Six engines were assumed so that there are also six engine-driven helium circulators. The work required by the helium compressor is added to the work required by the air compressor to determine the enthalpy drop across the inner turbine of the engine.

The helium temperature entering the heat exchangers is taken as the reactor outlet temperature. The helium temperature leaving the heat exchangers is, as in the case of the turbocirculator, free for optimization. The upper temperature limit is the reactor outlet temperature. The helium compressor pressure ratio is obviously smaller than that for a turbocirculator since no helium turbine is required - only the fluid frictional resistance need be overcome. The thermodynamic relations are contained in appendix B.

Comparison Ground Rules

Most of the data were calculated for the following conditions:

(1) Design altitude, 11 kilometers

(2) Design flight Mach number, 0.8

(3) Airplane gross mass, 453 600 kilograms

(4) Reactor and heat exchanger lifetime, 5000 hours

(5) Design reactor wall temperature, 1420 K

(6) Maximum helium pressure, 12 meganewtons per square meter

The 11-kilometer altitude condition allows the airplane to be operated above the frequently unfavorable weather conditions in the troposphere. Peak payload occurs at Mach numbers near 0.63, while maximum delivery rate occurs near 0.75 (ref. 2). But Mach 0.8 was chosen here because it is representative of current large cargo airplane designs. The 453 600 kilogram gross mass baseline is at the low end of the sensible range for nuclear aircraft especially when it is coupled to 5000-hour reactor and heat exchanger lifetimes. This combination yields payload ratios near 10 percent for the assumed flight conditions (ref. 3) which is only about one-third of that of similar chemically fueled airplanes. But these stringent ground rules also cause the airplane performance, in terms of payload, to be relatively sensitive to design modification. Hence the performance differences between the two circulator schemes should be more pronounced than if larger airplanes or shorter lifetimes were considered.

The reactor is assumed to be the pin and tube type described in references 1 and 3. The 1420 K design reactor wall temperature and 12 meganewtons per square meter maximum helium pressure are baseline values consistent with the design philosophy of reference 2. The unit radiation shield is composed of uranium and water layers. The shield weight equations are given in reference 2. Nuclear power is used only for cruise flight so that a chemical fuel supply is required for takeoff, climb, letdown, landing, and emergency cruise. No pump power is extracted from the engines (engine-driven case only) during chemical flight although such a provision is necessary for afterheat removal. The airplane, aerodynamic, structural, and engine assumptions are identical with those of reference 2. The heat exchanger material is N-155 alloy and the turbocirculator turbine material is assumed to be molybdenum. These materials are considered to be within the present state-of-the-art for the turbocirculator application. However, the stress-rupture curve used in the heat exchanger design is definitely optimistic in the engine-drive pump application. This is because weldments subjected to very high temperatures cause substantial strength reductions that are ignored in the calculations.

The adibatic efficiency of the circulator is treated as a parametric variable covering the range 0.75 to 0.90. The efficiency of the turbocirculator turbine is arbitrarily set equal to the circulator efficiency plus 0.02. Admittedly, there is no justification for this assumption, but it does keep the turbine and compressor efficiencies at about the same technology level and it avoids introducing another parametric variable. The turbocirculator turbine is exposed to reactor outlet temperatures near 1250 K and is therefore assumed to be made from molybdenum alloy. The allowable turbine stress determines the blade speed. The details of this consideration and the remaining assumptions concerning the mass of the two circulator subsystems are presented in appendix C.

The airplane's payload (the single performance criterion in this comparison) is maximized with a simple univariate search program by optimizing the following free variables for fixed gross mass, cruising Mach number, and altitude:

Engine variables:

(1) Overall pressure ratio
(2) Fan pressure ratio
(3) Bypass ratio
(4) Turbine inlet temperature during nuclear cruise
(5) Turbine inlet temperature during chemical flight
(6) Air-side heat exchanger pressure drop

Airplane variables:

(1) Aspect ratio
(2) Cruise lift coefficient

Reactor loop variables:

(1) Reactor outlet temperature
(2) Helium-side heat exchanger outlet temperature
(3) Helium-side heat exchanger pressure drop
(4) Reactor pressure drop
(5) Ducting pressure drop

This program as well as the details of the calculations are presented in reference 2.

RESULTS AND DISCUSSION

Effect of Circulator Efficiency

Payload. - The most influential variable bearing on the comparison of the two circulator schemes is the circulator efficiency. Its effect on payload for both engine-driven and turbo-type circulators is shown in figure 3. Especially significant is the difference in slopes of the two curves. The turbocirculator performance is quite sensitive to circulator efficiency - the payload increases from 33 200 to 56 500 kilograms when the efficiency is increased from 0.75 to 0.90. The payload of the engine-driven circulator scheme, however, is rather insensitive to efficiency - only rising from 47 400 to 49 000 kilograms over the same efficiency range. The two curves cross at a circulator efficiency of 0.82, with the turbocirculator offering less payload below this value but more above it. Hence, unlike the engine-driven circulator, a development program for a turbocirculator should concentrate heavily on the attainment of high component efficiencies. This presumes a constant, high value of turbine efficiency in the engine-driven scheme (e.g., 90 percent).

The turbocirculator scheme is much more sensitive to circulator efficiency for two reasons: (1) there are actually two component efficiencies (turbine as well as compressor) being changed simultaneously in the turbocirculator case, and payload varies with the product of these two efficiencies (eq. (5)), and (2) the turbocirculator handles three to six times as much work as do the engine-driven circulators. The idea of two component efficiencies changing simultaneously is in keeping with the ground-rule assumption that an improvment in turbocirculator technology would most likely arise from an improvement of both components. The much greater workload of the turbocirculator is due to the large pressure drop across its additional component - the helium turbine. Hence, the temperature change across the helium compressor is roughly 120 K for the turbocirculator instead of about 27 K for the engine-driven circulators. This difference allows the turbocirculator component efficiencies to exert a relatively powerful influence on the various helium temperatures and pressure drops and ultimately on the turbofan variables through the heat exchanger coupling. A complete set of details concerning these interactions is complicated by the large number (13) of independent variables involved. It is perhaps sufficient to simply recognize the source of the sensitivity.

Design variables. - Several of the more important variables are plotted against circulator efficiency in figure 4. For comparison purposes, consider the case where the two schemes yield the same payload - 0.82 circulator efficiency (and 0.84 helium turbine efficiency). The helium-side heat exchanger inlet temperature is 160 K lower (from 1290 to 1130 K) for the turbocirculator than for the engine-driven circulators. This is because the pump work is extracted from the helium before it reaches the heat exchang-

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ers. The helium pressure is also lowered - by 28 percent. Both of these changes are beneficial from the standpoint of heat exchanger design parameters (ref. 1). On the other hand, the turbine inlet temperature of the turbofan engines is decreased (from 1040 to 900 K) and this in turn results in a lower overall compression ratio (18.7 compared to 10) and a lower bypass ratio (4.7 compared to 3.85). These changes result in a less efficient overall propulsion system (efficiency drops from 0.272 to 0.268) that in turn causes a slight increase in design reactor power (264 instead of 262 MW). Attempting to alleviate this situation by raising the reactor outlet temperature and thereby raising all downstream temperatures is countered by increased design problems and weight penalties of the reactor, heat exchangers, and circulators (ref. 3).

Table I contains a more complete set of comparison details for several values of circulator efficiency. Both table I and figure 4 show that some major changes in design are to be considered when comparing the two circulator schemes. The use of a turbocirculator requires relatively low pressure and low temperature turbofans and heat exchangers. And if its components can achieve high efficiency, then the helium-side pressure drops should be considerably higher.

Mass breakdown. - A mass breakdown by major subsystems is given in figure 5. The potential performance advantage of the turbocirculator at high efficiency is derived from mass reductions in the nuclear system (reactor, shield, and ducts) as well as in the heat exchangers. The decreased nuclear system mass results from the lower power levels required by the more efficient total system (fig. 4). The heat exchangers become lighter because of the lower temperatures and pressures involved and also because of the reduced power levels. The engines, however, and the chemical fuel that they consume for the noncruise portions of flight are always heavier in the turbocirculator case because of their relatively low turbine inlet temperature and bypass and compression ratios. There is very little difference in structure mass.

The difference in the masses of the two circulator systems is due to the much greater helium compressor power needed in the turbocirculator scheme and the lower blade speed limit. Lower blade speeds are required in the turbocirculator scheme since the turbine must operate at the very high temperature of the reactor outlet stream. Thus, even with a molybdenum alloy the allowable blade stress is comparatively low, and this, coupled with the much higher helium compressor pressure ratio and low molecular weight of helium, causes many compressor and turbine stages to be needed (details in appendix C). Table I shows that about 12 turbine and 43 compressor stages are needed for the turbocirculator instead of just 2 compressor stages for the engine-driven circulators. The additional shafting mass required by the engine-driven circulators only partly offsets this disadvantage. The net difference in circulator masses is about a $2\frac{1}{2}$ to 1 advantage in favor of the engine-driven circulators. However, since only a fraction of 1 percent of the gross airplane mass is attributable to the circulator system,

[redacted]

this mass difference is not very important in an overall sense. Furthermore, even though the mass estimates (in appendix C) for these components may be in considerable error, the very small mass of the circulator subsystem obviates the need for refinement at this stage and minimizes the possibility of significant changes in these results.

Effect of Flight Speed

The effect of varying the design cruising speed of the airplane on payload is shown in figure 6. The comparison between the two circulator schemes is practically unaltered over the Mach number range of 0.4 to 0.8. For the particular case illustrated (0.80 circulator efficiency), the turbocirculator yields 3 to 6 percent less payload than the engine-driven circulators. Flight Mach number does, of course, influence the absolute payload strongly. The maximum payload occurs near Mach 0.6 where there is a 60-percent payload gain compared to Mach 0.8. Even at Mach 0.7 there is a 50-percent payload gain.

Effect of Reactor and Heat Exchanger Lifetime

The effect of shortening the reactor and heat exchanger lifetimes from the baseline value of 5000 to 1000 hours is shown in figure 7. As discussed in reference 3, the payload can be substantially increased in this manner, but, of course, at the disadvantage of more frequent replacements. In the particular case illustrated here the payload doubles when the lifetimes are decreased from 5000 to 1000 hours - primarily due to a substantial decrease in the reactor subsystem mass. Such dramatic changes would be lessened though if cases with higher payload fractions were used as baselines, as, for instance, in the case of larger airplane sizes. In any event, the circulator comparison is not affected very much by lifetime variations - showing almost no difference at 1000 hours and only a 6-percent difference in favor of the engine-driven circulators at 5000 hours. In this example, as in the flight speed comparison, the circulator efficiency is assumed to be 0.8.

CONCLUDING REMARKS

Although the results given here indicate that an efficient turbocirculator outperforms the engine-driven circulators, it must be emphasized that many important, and probably overriding, considerations are left unresolved. Among these are off-design performance, stability and control, startup and shutdown auxiliaries, lubrication, fabrication, reliability, and rotating seals - all concerning primarily the circulator subsystem alone. Additional considerations involve some of the other major systems such as the engines and heat exchangers.

Perhaps the first issue to settle is the helium leakage rate for engine-driven circulators. If this proves to be prohibitively large, then this scheme can be eliminated as a useful contender. Otherwise, it should be determined if either system can be ruled out on the basis of stability and control problems. Startup and shutdown (including afterheat removal) in the turbocirculator case requires significantly more auxiliary equipment than is needed with engine-driven circulators. The best solution to this problem should be sought including some kind of quantitative assessment of the penalty involved. The potential lubrication problem (contamination of the helium loop with lubricant) of the turbocirculator should be handled similarly.

Circulator fabrication difficulty should be minimal in the case of the engine-driven scheme. The 750 K compressor environment should permit conventional techniques and materials to be used. Fabrication of a turbocirculator would be more difficult. The turbine is exposed to 1250 K which definitely requires the use of one of the high-strength alloys, such as molybdenum, as assumed in this study. In addition, a very large number of stages are required (12 for the turbine and 43 for the compressor). It is possible to reduce the number of stages by using an even higher strength turbine material, such as a tungsten alloy, so that the blade speed could be raised. If a tungsten alloy is used, the number of stages could be reduced by a factor of 5 or 6, but it must be remembered that such an application for tungsten is not within the present state-of-the-art. Cooling the turbine with a small amount of the compressor outlet helium (around 720 K) appears to be an attractive alternative. Sufficient cooling could substantially reduce the number of stages and, perhaps, allow the use of a nickel alloy instead of a molybdenum alloy. Yet another approach is to arbitrarily reduce the number of stages by increasing the stage loading and accepting a decrease in efficiency. A detailed circulator design study is really needed to find the best solution to these problems.

In any case, whatever fabrication difficulty there is for the turbocirculator is at least partially offset by the reduced fabrication difficulty of the heat exchangers and engines. The 160 K lower heat exchanger temperature is particularly helpful in that it allows the use of N-155 alloy (readily welded, worked, and machined) instead of a more difficult to fabricate high temperature, oxidation resistant alloy. Also, the engines need fewer stages in the turbocirculator case since the engine pressure ratios are significantly reduced.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, October 16, 1970,
126-15.

APPENDIX A

SYMBOLS

c_p — constant pressure specific heat

D — diameter

g — gravitational constant

K — scaling factor

M — component mass

N — number of stages

p — pressure

R — gas constant

$\mathcal{R}$ — hub-to-tip radius ratio

s — stress

T — temperature

U — mean tangential blade speed

V — gas axial velocity

w — mass flow rate

z — ratio of inlet to exit axial kinetic energies

α — stator exit angle

γ — specific heat ratio

λ — pressure drop parameter defined by eq. (B6); also, stage speed-work parameter

η — component efficiency

ψ — head coefficient

ρ — density

θ — defined by eq. (B6)

ω — defined by eq. (B6)

Subscripts:

b — blade

c — compressor

t — turbine

min — minimum

max — maximum

1, 2, 3, 4 — station numbers appearing in sketches

APPENDIX B

THERMODYNAMICS OF CIRCULATORS

Turbocirculator Scheme

There are certain relations between the system temperatures, pressures, and efficiencies that must be satisfied as a result of the direct coupling between the compressor and turbine. In particular, the compressor work is

$$\text{Compressor work} = wc_p(T_2 - T_1) = wc_pT_1\left[\left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma} - 1\right]\frac{1}{\eta_c} \tag{B1}$$

where w is the helium gas flow rate, c_p the constant pressure specific heat, T the absolute temperature of the helium, p the helium pressure, η_c the adibatic compressor efficiency, γ the specific heat ratio, and numerical subscripts refer to the stations indicated in sketch (a). Likewise the turbine work is

$$\text{Turbine work} = wc_p(T_3 - T_4) = wc_pT_3\left[1 - \frac{1}{\left(\frac{p_3}{p_4}\right)^{(\gamma-1)/\gamma}}\right]\eta_t \tag{B2}$$

where η_t is the adibatic turbine efficiency. Equating compressor work to turbine work and assuming c_p and γ are constants give

$$T_1\left[\left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma} - 1\right] = \eta_c\eta_tT_3\left[1 - \frac{1}{\left(\frac{p_3}{p_4}\right)^{(\gamma-1)/\gamma}}\right] \tag{B3}$$

But,

$$\frac{p_3}{p_4} = \frac{p_2}{p_1}\frac{p_3}{p_2}\frac{p_1}{p_4} \tag{B4}$$

And, using equation (B4) in equation (B3) gives

$$\eta_c \eta_t \left(\frac{T_3}{T_1}\right) = \frac{\left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma}\left[\left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma} - 1\right]}{\left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma} - \left(\frac{p_2}{p_3}\frac{p_4}{p_1}\right)^{(\gamma-1)/\gamma}} \tag{B5}$$

It is understood that duct pressure losses are included in the product $(p_2/p_3)(p_4/p_1)$ in these expressions. The form of equation (B5) can be simplified by letting

$$\lambda \equiv \left(\frac{p_2}{p_3}\frac{p_4}{p_1}\right)^{(\gamma-1)/\gamma} \tag{B6a}$$

$$\theta \equiv \eta_c \eta_t \frac{T_3}{T_1} \tag{B6b}$$

$$\omega \equiv \left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma} \tag{B6c}$$

With these definitions substituted into equation (B5) we get

$$\theta = \frac{\omega(\omega - 1)}{\omega - \lambda} \tag{B7}$$

This relation is plotted in figure 8. It is clear that for any given value of the pressure loss parameter λ there is a minimum value of the temperature ratio parameter θ. The compressor pressure ratio at which this occurs may be determined by differentiating equation (B7) with respect to ω:

$$\left[\omega = \lambda\left(1 + \sqrt{1 - \frac{1}{\lambda}}\right)\right]_{\theta=\theta_{min}} \tag{B8}$$

Solving equation (B7) for λ and substituting the resultant expression into equation (B8) give the equation for the minimum value of θ in terms of the pressure ratio parameter ω:

$$\theta_{min} = 2\omega - 1 \tag{B9}$$

This relation is represented by the dotted straight line in figure 8. Finally, substituting equation (B8) into equation (B9) gives the minimum value of θ in terms of λ:

$$\theta_{min} = 2\lambda\left(1 + \sqrt{1 - \frac{1}{\lambda}}\right) - 1 \tag{B10}$$

These expressions are used to determine the operating temperatures and pressures in the helium loop. In particular, suppose that the reactor outlet temperature T_3 is fixed as are the various pressure drops and component efficiencies. Then a question arises as to what heat exchanger outlet temperature T_1 should be used. According to equation (B7), or figure 8, T_1 may take on many different values but the highest it can be is determined by equation (B10), which rewritten with the use of equation (B6) is

$$\left(T_1\right)_{max} = \frac{\eta_c \eta_t T_3}{2\left(\frac{p_2}{p_3}\frac{p_4}{p_1}\right)^{(\gamma-1)/\gamma}\left[1 + \sqrt{1 - \left(\frac{p_2}{p_3}\frac{p_4}{p_1}\right)^{(1-\gamma)/\gamma}}\right] - 1} \tag{B11}$$

This value of T_1 results in the smallest heat exchanger size and was found to yield maximum payload.

Engine-Driven Circulators

In this case, a small temperature rise takes place in each compressor:

$$T_2 - T_1 = T_1\left[\left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma} - 1\right]\frac{1}{\eta_c} \tag{B12}$$

where the pressure ratio p_2/p_1 is just equal to the fluid frictional resistance $(p_2/p_3) \times (p_3/p_1)$ of the reactor and heat exchangers including intercomponent ducting (see sketch (b)). The work required by all the compressors is given by equation (B1).

APPENDIX C

CIRCULATOR MASS ASSUMPTIONS

Turbocirculator

The turbocirculator mass was taken to be the sum of the compressor and turbine masses:

$$M_c = K_c D_c^{2.2} N_c^{1.2} \tag{C1}$$

$$M_t = K_t D_t^{2.5} U^{0.6} N_t \tag{C2}$$

where subscripts c and t refer to compressor and turbine, respectively, M is the component mass including casing, D the mean blade diameter, U the mean blade speed determined at D, N the number of stages, and K a scaling factor dependent on application. These equations are based on unpublished results (obtained from Mr. Jonas Sturas, Lewis Research Center) correlating conventional turbojet engine component weights. For M in kilograms, D in meters, and U in meters per second, K_c was assumed to be 50 and K_t was assumed to be 9, which correspond to compressors and turbines slightly heavier than those used in typical turbojet cruise engines.

The number of compressor stages was estimated by

$$N_c = \frac{c_p g(T_2 - T_1)}{\psi U^2} \tag{C3}$$

where g is the universal gravitational constant and ψ is the head coefficient, assumed to be 0.3. The number of turbine stages was estimated with the aid of information given in reference 6:

$$N_t = \frac{g\gamma R(T_3 - T_4)\lambda}{(\gamma - 1)U^2} \tag{C4}$$

where R is the helium gas constant and λ is the stage speed-work parameter, assumed to be 0.5. The mean blade speed U in these equations is based on blade centrifugal stress s_b (ref. 7):

[redacted]

$$s_b = \frac{2\rho_b U^2(1 - \mathcal{R})}{g(1 + \mathcal{R})} \tag{C5}$$

If it is assumed the blade density ρ_b equals the disk density ρ_d, the blade stress s_b equals 0.6 of the disk stress s_d, and the hub-to-tip radius ratio $\mathcal{R}$ is 0.818 (ref. 6), equation (C5) when solved for U^2 gives

$$U^2 = 3g\frac{s_d}{\rho_d} \tag{C6}$$

The allowable disk stress s_d is determined from figure 9 of reference 6, assuming the disk material to be molybdenum alloy and to have a 10 000-hour lifetime:

$$\ln(2s_d) = -0.002758\,(T_4 - 100) + 16.83 \tag{C7}$$

where T_4 is in degrees Rankine and s_d is in pounds per square inch.

The mean turbine blade diameter is calculated from continuity considerations (ref. 6):

$$D_t = \left[\frac{wRT_4}{\pi p_4 V_4}\left(\frac{1 + \mathcal{R}}{1 - \mathcal{R}}\right)\right]^{1/2} \tag{C8}$$

where w is the helium mass flow rate, p_4 the turbine exit pressure, and V_4 the exit axial velocity:

$$V_4 = \sqrt{z}\,\frac{U}{\lambda}\cot\alpha \tag{C9}$$

Here, z is the ratio of inlet to exit axial kinetic energies (assumed to be 1.5), and α is the stator exit angle (assumed to be 70°). The mean compressor blade diameter is assumed to be the same as the mean turbine blade diameter.

Engine-driven circulators. - The total mass in this case consists of six sets of compressors and drive transmissions - one set for each engine. The compressor mass portion is estimated with equations (C1) and (C3) where U is assumed to be 460 meters per second (ref. 7), and the compressor diameter is calculated by

[redacted]

$$D_c = \left[\frac{wRT_1}{\pi p_1 V_1}\left(\frac{1 + \mathcal{R}}{1 - \mathcal{R}}\right)\right]^{1/2} \tag{C10}$$

From reference 6, V_1 is assumed to be 150 meters per second and $\mathcal{R}$ equals 0.8. The helium flow rate w is one-sixth the total rate. The shafting mass in kilograms is estimated (ref. 8) to be 0.061 times the compressor power in kilowatts.

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REFERENCES

1. Rom, Frank E.: Subsonic Nuclear Aircraft Study. NASA TM X-1626, 1968.
2. Allen, John L.; Fishbach, Laurence H.; and Strack, William C.: Preliminary Study of a Subsonic Nuclear Cruise Airplane Having a Helium-Cooled Thermal Reactor. NASA TM X-1667, 1968.
3. Fishbach, L. H.; Allen, J. L.; and Strack, W. C.: Temperature and Lifetime Effects on the Performance of a Nuclear-Powered Airplane. NASA TM X-52432, 1968.
4. Yampolsky, J.; Barbat, V.; Berman, H.; Cavallaro, L.; and Ross, F.: Investigation of Turbocirculator for TARGET, a 1000-Mw(e) Reactor. Rep. GA-5559, General Atomics Div., General Dynamics Corp., Sept. 28, 1964.
5. Anon.: Study of Gas Cooled Fast Reactor Technology. Rep. GE-NMP-67-505, General Electric Co. (NASA CR-72321), Oct. 20, 1967.
6. Kofskey, Milton G.; and Glassman, Arthur J.: Turbomachinery Characteristics of Brayton Cycle Space-Power-Generation Systems. Presented at the ASME Ninth Annual Gas Turbine Conference and Products Show, Houston, Texas, Mar. 1-5, 1964.
7. Cavicchi, R. H.; Ellerbrock, H. H.; Hall, E. W.; Heppler, H. J.; Livingood, J. N. B.; and Schwenk, F. C.: Design Analysis of a Subsonic Nuclear-Powered Logistic Airplane with Helium-Cooled Reactor. NASA TM X-28, 1959.
8. Johnson, Paul G.; and Smith, Rober L.: Comparison of Three Nuclear Propulsion Systems for a Logistic-Type Aircraft. NASA Memo 3-16-59E, 1959.

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TABLE I. - DESIGN VARIABLES FOR NUCLEAR-POWERED AIRPLANE

[Mach number, 0.8; altitude, 11 km; reactor and heat exchanger lifetimes, 5000 hr.]

	Circulator compressor efficiency/turbine efficiency							
	0.75	0.75/0.77	0.80	0.80/0.32	0.85	0.85/0.87	0.90	0.90/0.92
	Engine-driven circulator	Turbo-circulator	Engine-driven circulator	Turbo-circulator	Engine-driven circulator	Turbo-circulator	Engine-driven circulator	Turbo-circulator
Engine								
Overall pressure ratio	18.7	8.40	18.7	9.55	18.7	10.8	18.8	12.3
Fan pressure ratio	1.42	1.42	1.41	1.41	1.41	1.42	1.42	1.43
Bypass ratio	4.64	3.35	4.69	3.74	4.74	3.90	4.72	4.02
Turbine inlet temperature, nuclear cruise, K	1042	857	1042	885	1043	909	1047	930
Turbine inlet temperature, chemical fuel, K	1022	886	1025	908	1025	930	1028	947
Airplane								
Aspect ratio	5.48	5.84	5.46	5.63	5.43	5.47	5.44	5.38
Cruise lift coefficient	0.408	0.431	0.408	0.410	0.409	0.408	0.408	0.407
Heat exchanger								
Air-side inlet temperature, K	611	481	610	500	610	519	612	540
Air-side outlet temperature, K	1042	857	1042	885	1043	909	1047	930
Air-side pressure drop	0.0554	0.0472	0.0552	0.0454	0.0559	0.0487	0.0553	0.0449
Helium-side inlet temperature, K	1291	1156	1291	1135	1285	1123	1290	1122
Helium-side outlet temperature, K	775	538	757	574	759	606	762	644
Helium-side inlet pressure, MN/m^2	12	8.3	12	8.70	12	8.70	12	8.91
Helium-side pressure drop	0.0082	0.0044	0.0083	0.0069	0.0088	0.0094	0.0089	0.0116
Maximum metal temperature, K	1236	1086	1236	1080	1233	1079	1235	1084
Reactor loop								
Reactor inlet temperature, K	784	653	784	700	786	740	787	775
Reactor outlet temperature, K	1291	1270	1291	1261	1285	1259	1290	1252
Reactor inlet pressure, MN/m^2	12	12	12	12	12	12	12	12
Reactor pressure drop	0.0285	0.0214	0.0301	0.0312	0.0312	0.0439	0.0335	0.0509
Shield pressure drop	0.0198	0.0249	0.0199	0.0290	0.0204	0.0326	0.0200	0.0349
Ducting pressure drop	0.0113	0.0080	0.0122	0.0120	0.0127	0.0161	0.0122	0.0199
Number of circulator compressor stages	2	40	2	42	2	44	2	43
Number of circulator turbine stages	------	12	------	13	------	13	------	13
Helium flow rate, kg/sec	98.5	84.7	99.0	91.0	100	96.8	99.5	102
Pumping power, MW	14.9	50.8	1460	59.9	14.1	67.7	13.0	69.7
Reactor design power, MW	263	273	262	266	262	261	261	255
Overall powerplant efficiency	0.271	0.255	0.271	0.265	0.272	0.272	0.273	0.280

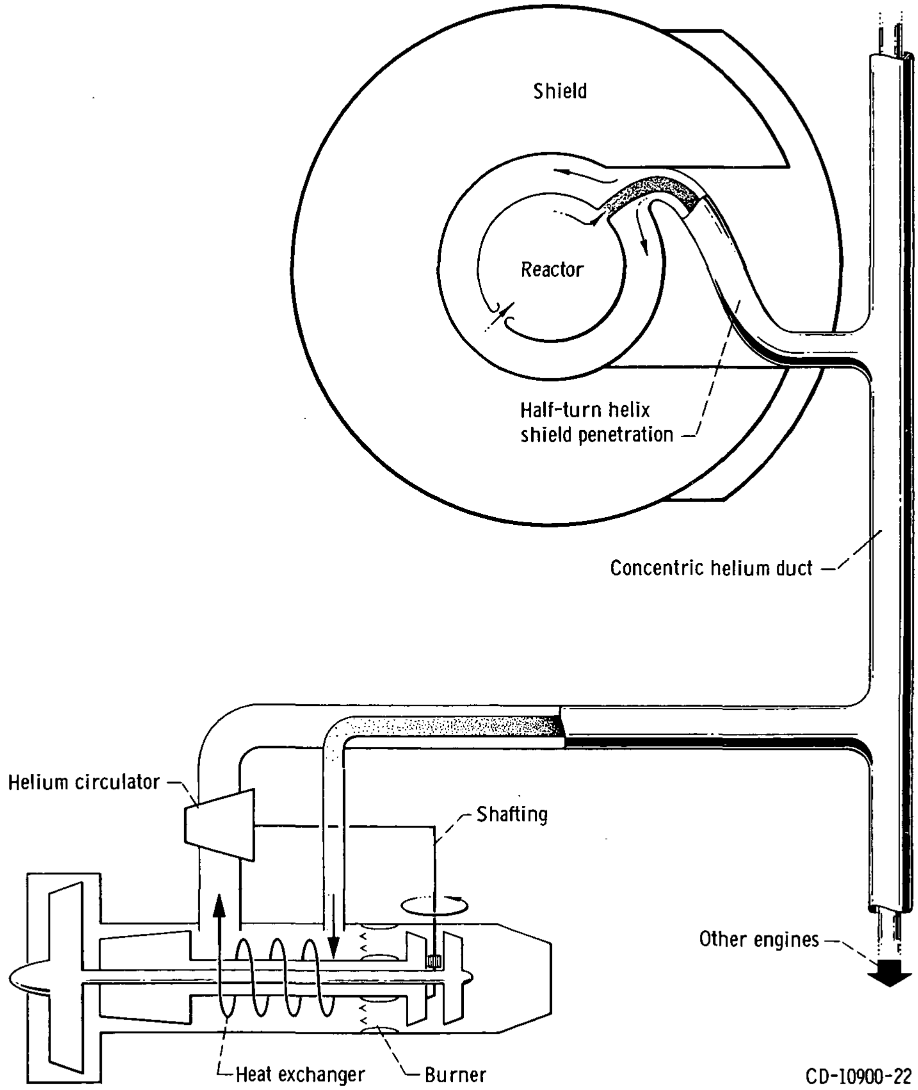

Figure 1. - Engine-driven helium circulator scheme for nuclear-powered airplane.

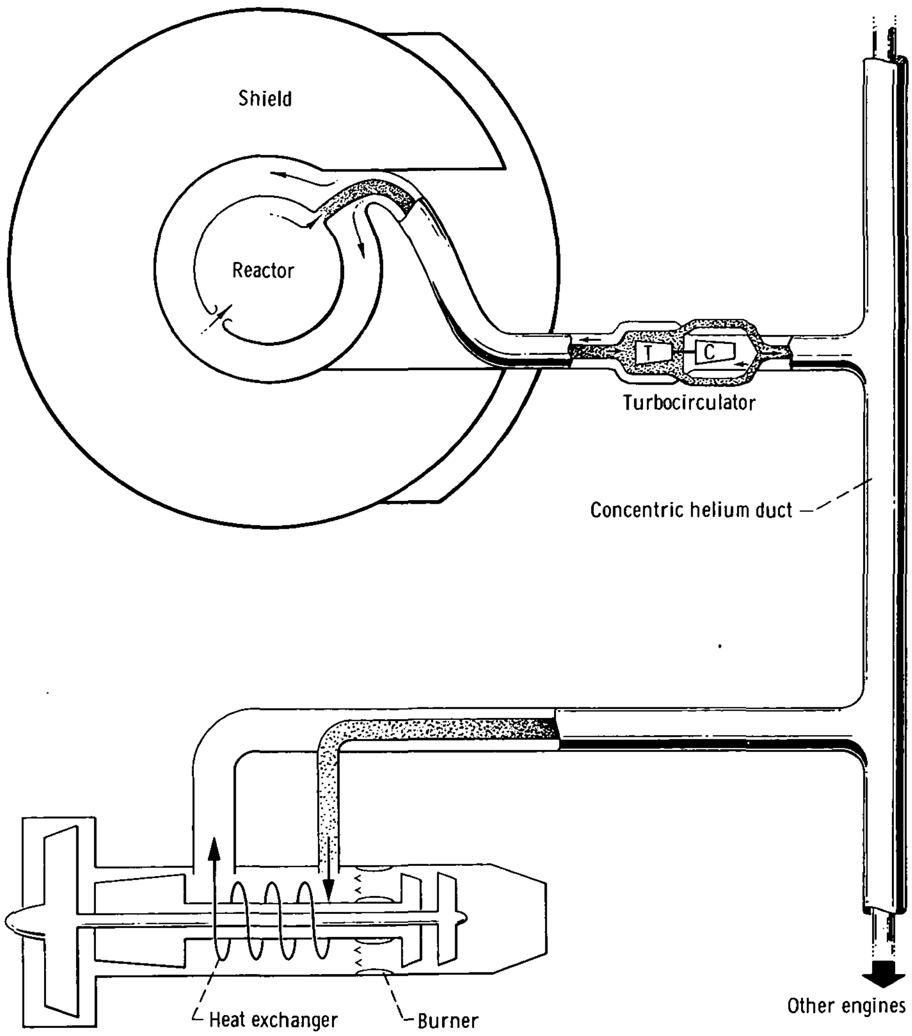

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Figure 2. - Turbocirculator scheme for nuclear-powered airplane.

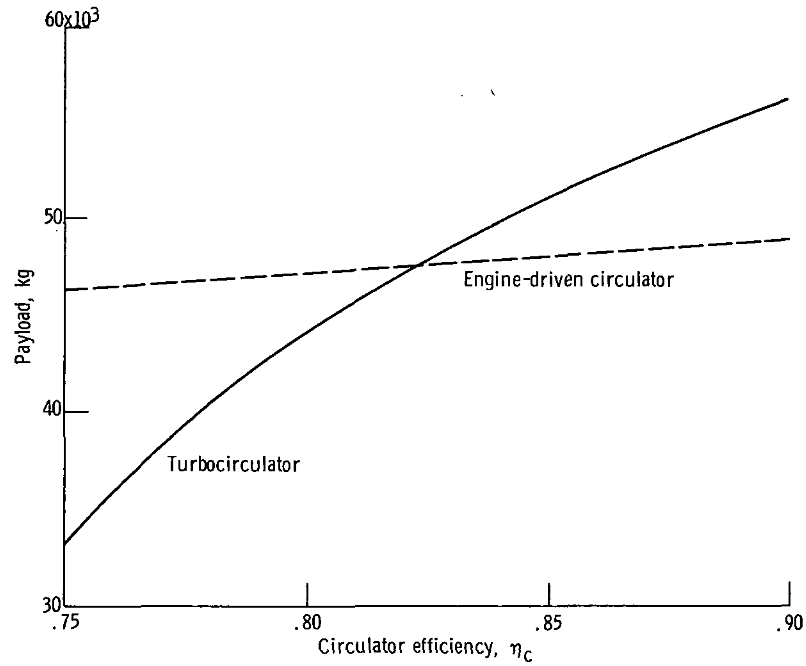

Figure 3. - Effect of circulator efficiency on payload of 453 600 kilogram nuclear-powered airplane. Flight Mach number, 0.8; altitude, 11 kilometers; reactor and heat exchanger lifetimes, 5000 hours; helium turbine efficiency, $\eta_t = \eta_c + 0.02$.

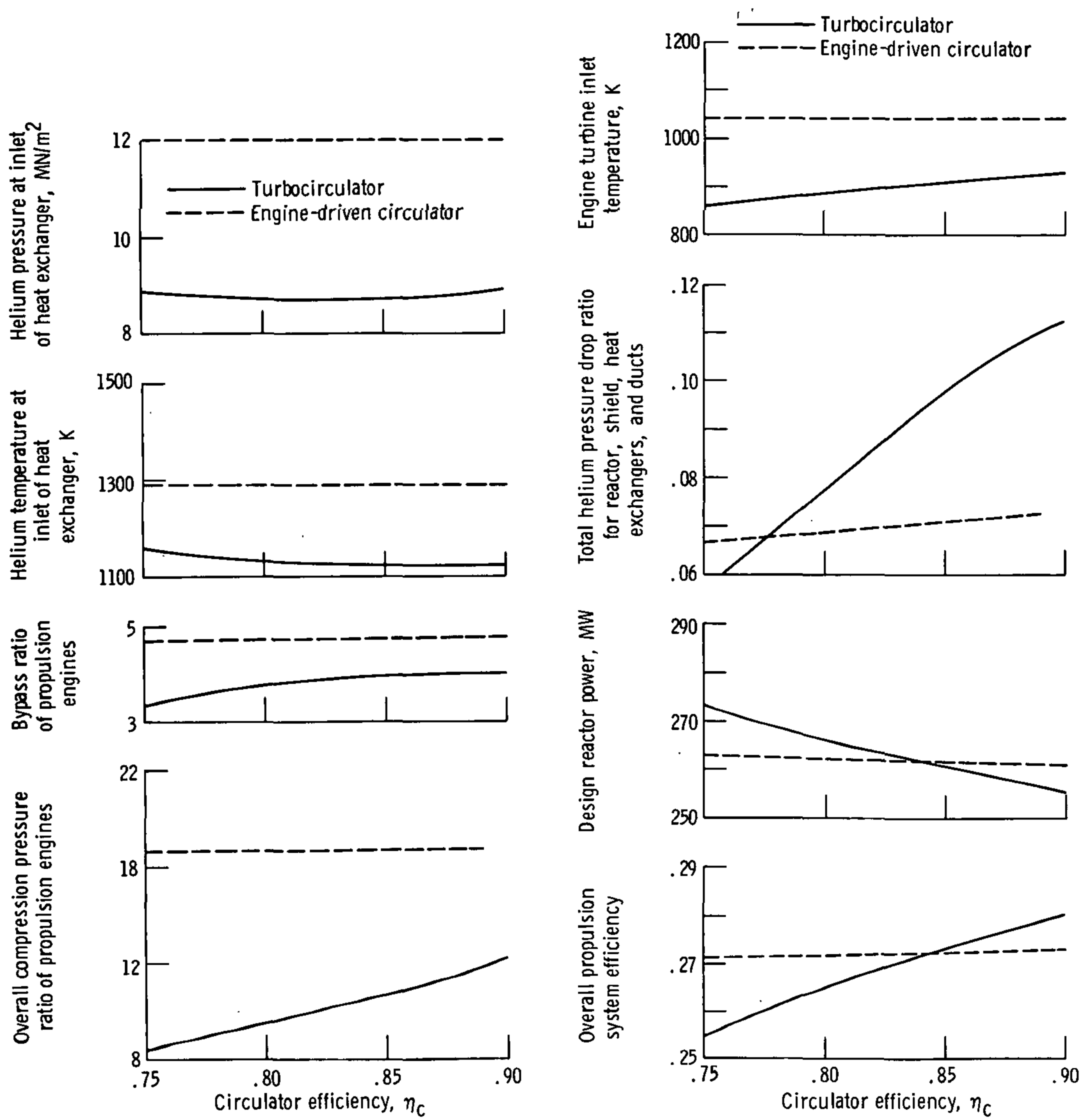

Figure 4. - Effect of circulator efficiency on various system variables of 453 600 kilogram nuclear-powered airplane. Flight Mach number, 0.8; altitude, 11 kilometers; reactor and heat exchanger lifetimes, 5000 hours; helium turbine efficiency, $\eta_t = \eta_c + 0.02$.

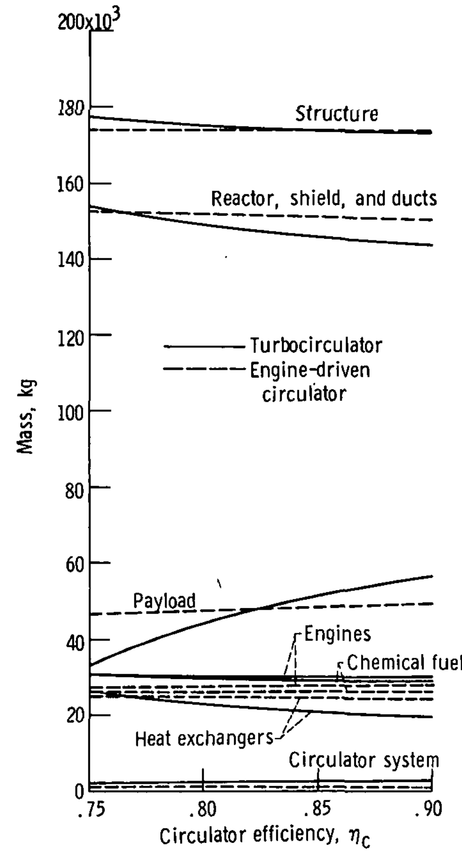

Figure 5. - Subsystem mass breakdown of a 453 600 kilogram nuclear-powered airplane. Flight Mach number, 0.8; altitude, 11 kilometers; reactor and heat exchanger lifetimes, 5000 hours; helium turbine efficiency, $\eta_t = \eta_c + 0.02$.

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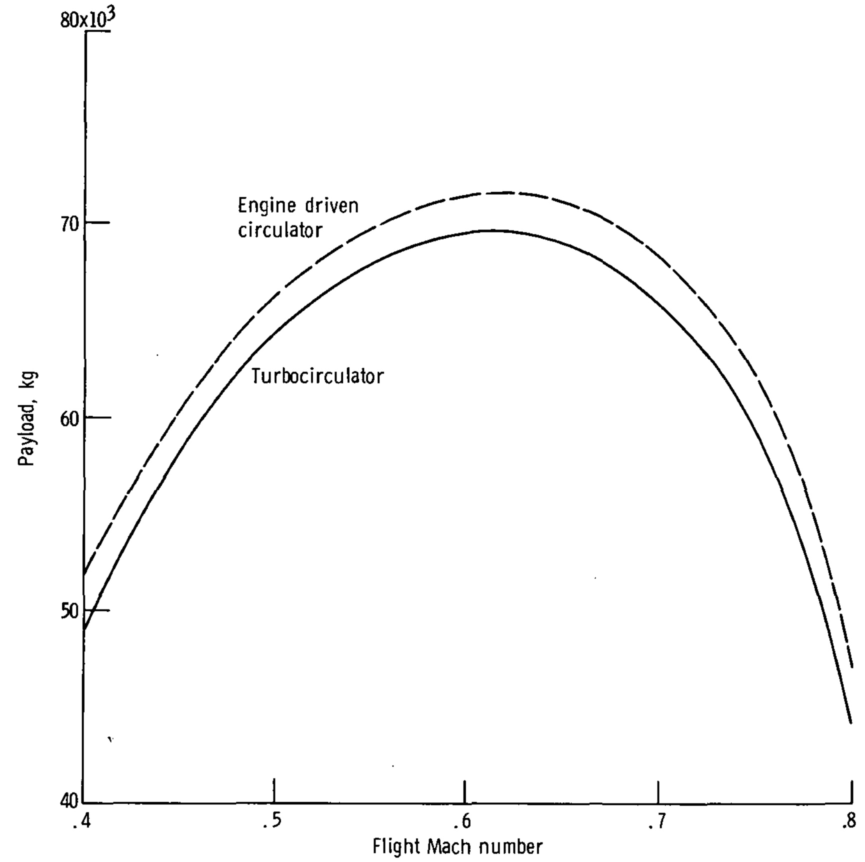

Figure 6. - Effect of flight Mach number on payload of nuclear-powered airplane. Gross mass, 453 600 kilograms; altitude, 11 kilometers; reactor and heat exchanger lifetimes, 5000 hours; helium circulator efficiency, 0.80; helium turbine efficiency, 0.82.

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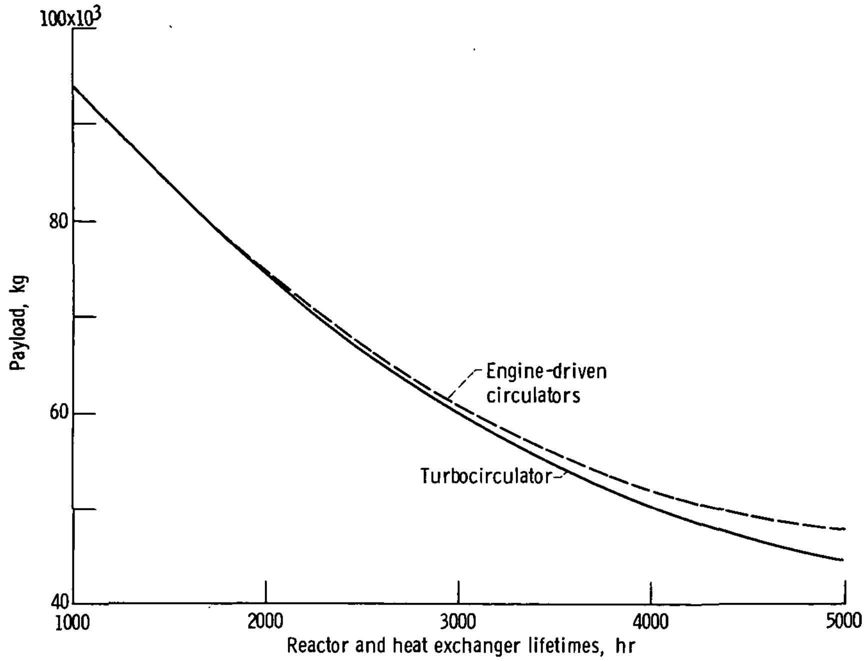

Figure 7. - Effect of reactor and heat exchanger lifetimes on payload of 453 600 kilogram nuclear-powered airplane. Flight Mach number, 0.8; altitude, 11 kilometers; helium circulator efficiency, 0.80; helium turbine efficiency, 0.82.

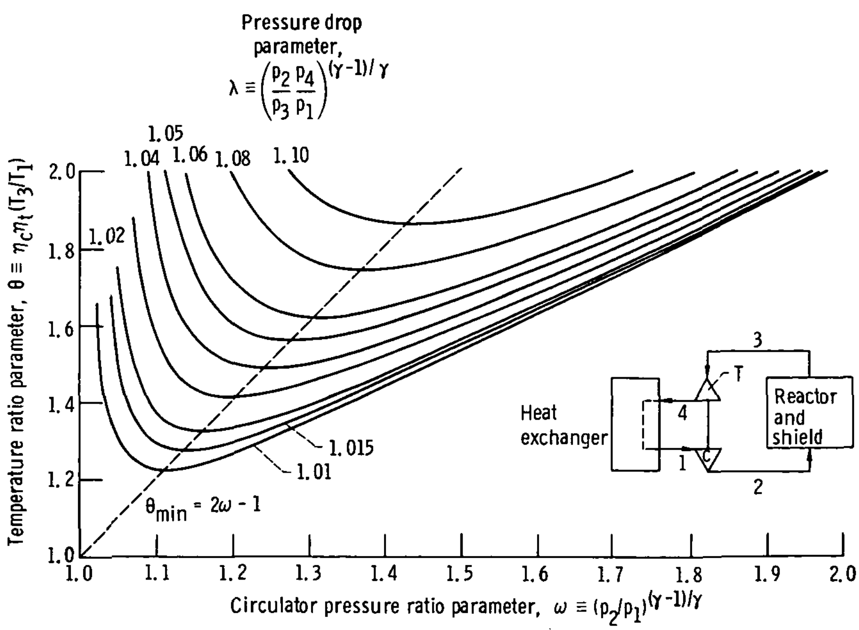

Figure 8. - Thermodynamic relations for turbocirculator.

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