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NUCLEAR ENGINE DESIGN CONSIDERATIONS

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The design of a nuclear rocket engine encompasses a wide variety of technologies and scientific disciplines. It is the purpose of this paper to describe a typical nuclear rocket engine, defining its characteristics so that an appreciation of the over-all design problem may be realized. To understand the depth of the design considerations which must be examined, it is perhaps best to assume a typical design requirement for a possible space mission application, then to examine possible solutions to that requirement with a view toward selection of preliminary design criteria for the system and its associated components. Following this selection, optimization studies such as those described herein would delineate the initial design criteria of the engine system. Subsequent analyses, design studies, experiments, and even development work may dictate further re-examination of and modifications to the engine criteria. It should be pointed out here that in proceeding through the description and discussions of a typical nuclear rocket engine, the specific values and parameters of the assumed engine system and its components bear no direct relation to any existing or planned nuclear rocket engine. These parameters are totally imaginary, but are perhaps reasonable values for such a hypothetical system.

Figure 1 is a schematic diagram of a typical nuclear rocket engine which employs a turbine bleed cycle. The engine system basically contains a liquid-hydrogen pump, an associated turbine to drive the pump, a regeneratively cooled nozzle, a reactor thrust chamber assembly, and the necessary reactor and turbine power controls. The engine operates in the following manner. Liquid hydrogen flowing

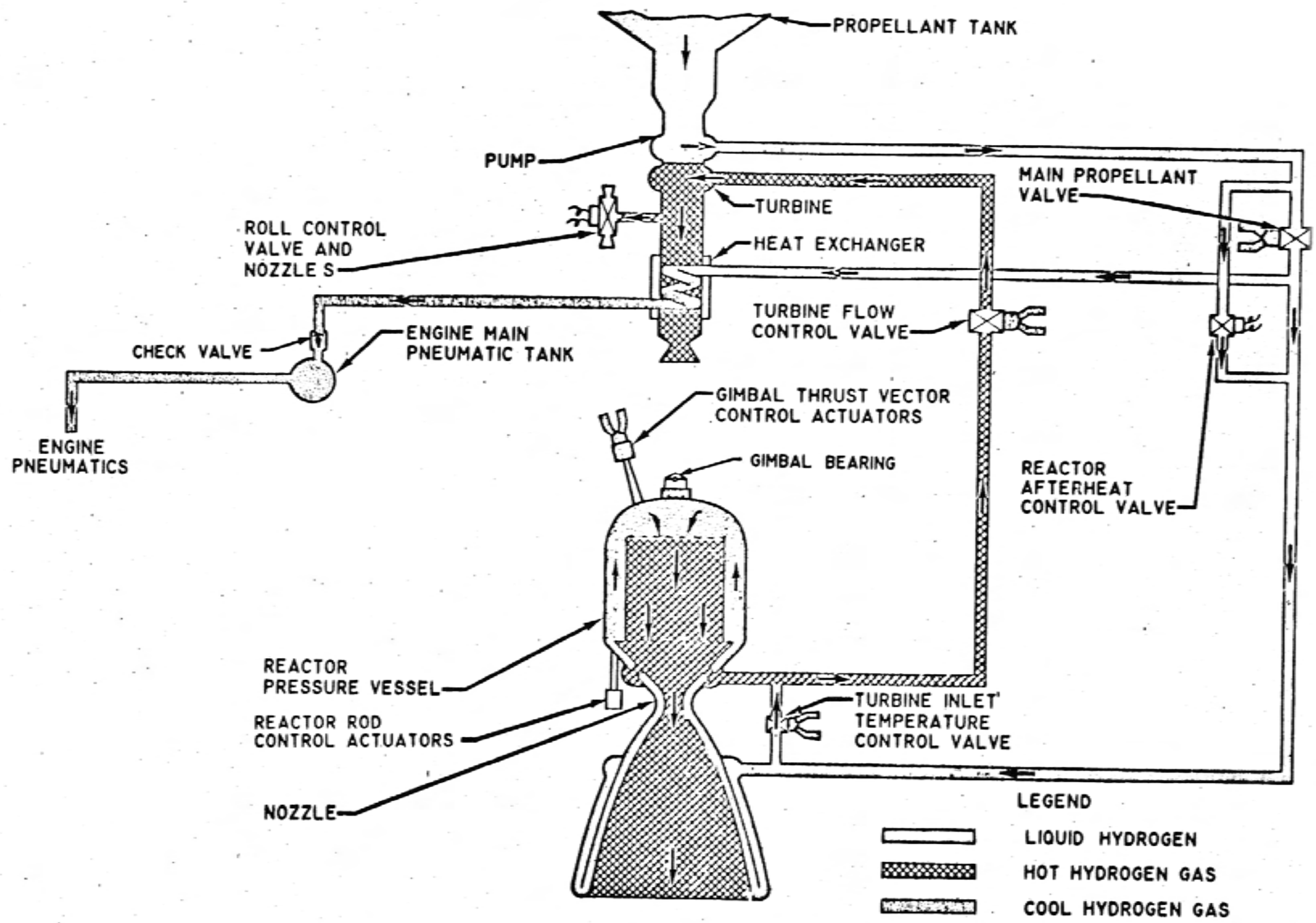

Figure 1. Typical Bleed Cycle Engine Schematic

from the vehicle propellant tank is boosted in pressure and delivered to the inlet of the cooling jacket of the regeneratively cooled nozzle. The propellant flows through the tubes of the nozzle, cooling them, and is then delivered to an annular reflector/preheater which surrounds the reactor core. After cooling the reflector/preheater, the hydrogen flows in a gaseous state through the top shield of the reactor assembly, continues down through the core, where it is heated to a temperature of perhaps 5000 R, then expands outward through the nozzle to produce the desired reactive thrust. At some convenient point in the reactor/thrust chamber assembly, heated gases are withdrawn and ducted to the turbine inlet where they expand across the turbine wheels to produce the power to drive the pump. Reactor control elements and a turbine throttle valve are employed to balance the energy being produced by the reactor with the energy being carried off by the propellant, so that the desired thrust and performance are obtained from the engine.

There are two fundamental differences between nuclear rocket engines of the type just described, and conventional, liquid-propellant rocket engines. First, in a liquid-propellant engine, the energy release is determined both by the chemical reactions which occur between the oxidizer and the fuel, and by the total quantity of propellants being delivered to the combustion chamber. Hence, for any given propellant delivery conditions, the energy release, and, hence, the total thrust capability of the system, are determined by the turbopump power which in many instances is self-limiting. In a nuclear engine, however, as pointed out above, the power generation process is fundamentally independent from what may be going on in the rest of the engine system and is normally controlled only by the reactor control elements. Thus, it is necessary that the propellant (or working fluid) be delivered to the thrust chamber in just the right amount to carry off the energy being produced in the reactor.

The second fundamental difference involves the environment produced by the nuclear reactor. This environment is comprised of very high fluxes of high-energy neutrons and gamma rays which interact with the other components of the engine, the propellant, and the rest of the vehicle to produce induced heating, radiation damage, and component activation problems.

To gain a better appreciation of the tasks confronting the nuclear engine designer, it is useful to assume a typical nuclear rocket application for which a set of hypothetical propulsion requirements may be specified. For the purposes of this paper, the application will be the delivery of 100,000 pounds of payload to an orbit about the moon.

Simple vehicle performance and trajectory calculations can suggest that a Saturn C-5 booster with a suitable, nuclear-powered second stage should be capable of performing the given mission, provided that the nuclear engine can be employed in the following manner. First, upon booster-stage burnout, the nuclear, second-stage engine will be started and utilized to propel the second stage and its payload into a low-altitude, earth orbit. Then, after a short hold of approximately 15 minutes in the earth parking orbit, the engine will be restarted to propel the vehicle to earth-escape velocity. Finally, after approximately 2-1/2 days, the engine will be restarted again and used to decelerate the vehicle into a lunar orbit.

The assumed basic design requirements of the engine for the proposed operational mode are given below:

Thrust	600,000 pounds (13,000 megw)
Specific Impulse	850 seconds

Total Operating Time	22 minutes
Number of Operating Cycles	3
Thrust Vector and Roll Control	10 pounds at 5 degrees/sec

Consideration of possible solutions to the foregoing propulsion requirements must start with the specific impulse requirement. To obtain an 850-second engine specific impulse means that the average temperature of the gases leaving the core may have to be of the order of 4800 R, depending on the type of engine cycle, nozzle configurations, and auxiliary thrust requirements. Assuming reasonable temperature differentials to effect the energy transfer into the hydrogen gas, it is apparent that portions of the core would have to operate at temperatures ranging to 5000 R.

The number of fuel-element base materials which might be considered for operation at these temperatures is quite limited. Graphite, the refractory metals, and carbides of these metals have been mentioned as possible fuel-element base materials in the open literature.

It has been reported that graphite exhibits peak strength at temperatures around 5000 R; in addition, with both its relatively low atomic weight and neutron absorption cross-section, it is generally considered one of the better moderators. With a propellant flowrate requirement of approximately 700 lb/sec, as determined by the thrust and specific impulse requirements, and with reasonable assumptions for reactor solidity and operating pressures, the required core volume will probably be somewhat in excess of 100 cubic feet. The geometric buckling of a core of this size should be sufficient to ensure that the core configuration will be dictated by heat-transfer considerations, and not by criticality considerations. Consequently, despite the limited uranium investment capability of graphite, the utilization of a graphite-based core would not introduce any meaningful size penalty and, therefore, should be substantially lighter than a refractory metal and/or metallic carbide-based core.

Before a preliminary selection of the fuel-element base material can be made, it is necessary to consider the total operating time and number of operating cycles required for the reference mission. The cumulative effects of corrosion/erosion of the walls of the coolant passages in the core by the propellant, the migration of fissionable and moderating material out of the fuel element/core structure, the buildup of neutron poisons within the core, or the excess reactivity required for each complete operating cycle can limit the useful operating life and the core's capability of being recycled. For the purposes of this paper, it will be assumed that a graphite-based core has the capability of meeting the total operating time and number of cycle requirements specified for the reference mission.

The selection of the type of engine cycle affects attainment of the performance requirements in the following way. In a bleed cycle (Fig. 1), a small portion of the total propellant flow is withdrawn from the reactor/thrust chamber assembly and mixed with unheated propellant. This cooled mixture is utilized to drive the turbine. After expanding across the turbine, these gases may be ducted to auxiliary exhaust nozzles where they produce additional thrust. The specific impulse of the turbine-drive gases, utilized in the above manner, is substantially less than the specific impulse of the main exhaust jet because the turbine exhaust temperature is much cooler than that of the gases leaving the core. As a consequence, the effective specific impulse of the engine is degraded by the use of bleed gases to supply the turbodrive power. The over-all performance loss is quite small, however, ranging from 2 to 3 percent of the specific impulse of the main jet, depending on the temperature of the bleed gases.

In a topping cycle (Fig. 2), all of the propellant flow is partially heated in passing through the annular reflector/preheater section of the reactor assembly. At this point, the propellant flow is split. Most of the propellant is directed to an expander turbine. The remaining propellant that bypasses the turbine is ducted to the reactor core where it again joins the fluid that has been exhausted from the turbine after supplying the requisite turbine power. The combined stream then flows through the reactor core in which all the propellant is heated to full core exit temperatures. The turbine bypass flow allows for more stable engine operation and may be adjusted, as required, to control engine thrust. Thus, in the topping cycle, the engine suffers no performance degradation because a portion of the working fluid was exhausted at a lower temperature, as in the bleed cycle.

To meet the assumed specific impulse requirement of 850 seconds, the engine designer has the choice of: (1) for a topping cycle, requiring a core-exit gas temperature of approximately 4600 R, and (2) for a bleed cycle, utilizing turbine drive gases of around 1500 R, requiring a core-exit gas temperature of 4800 R. Therefore, an important factor in the selection of the engine cycle to be employed is the capability of the reactor core to operate, for the required number of cycles and total duration, at the fuel element temperatures associated with the core-exit gas temperatures mentioned above.

Although the obvious advantage of reduction of core-operating temperature is to be gained from the use of a topping cycle, it is accompanied by certain inherent disadvantages from the standpoint of subsystem

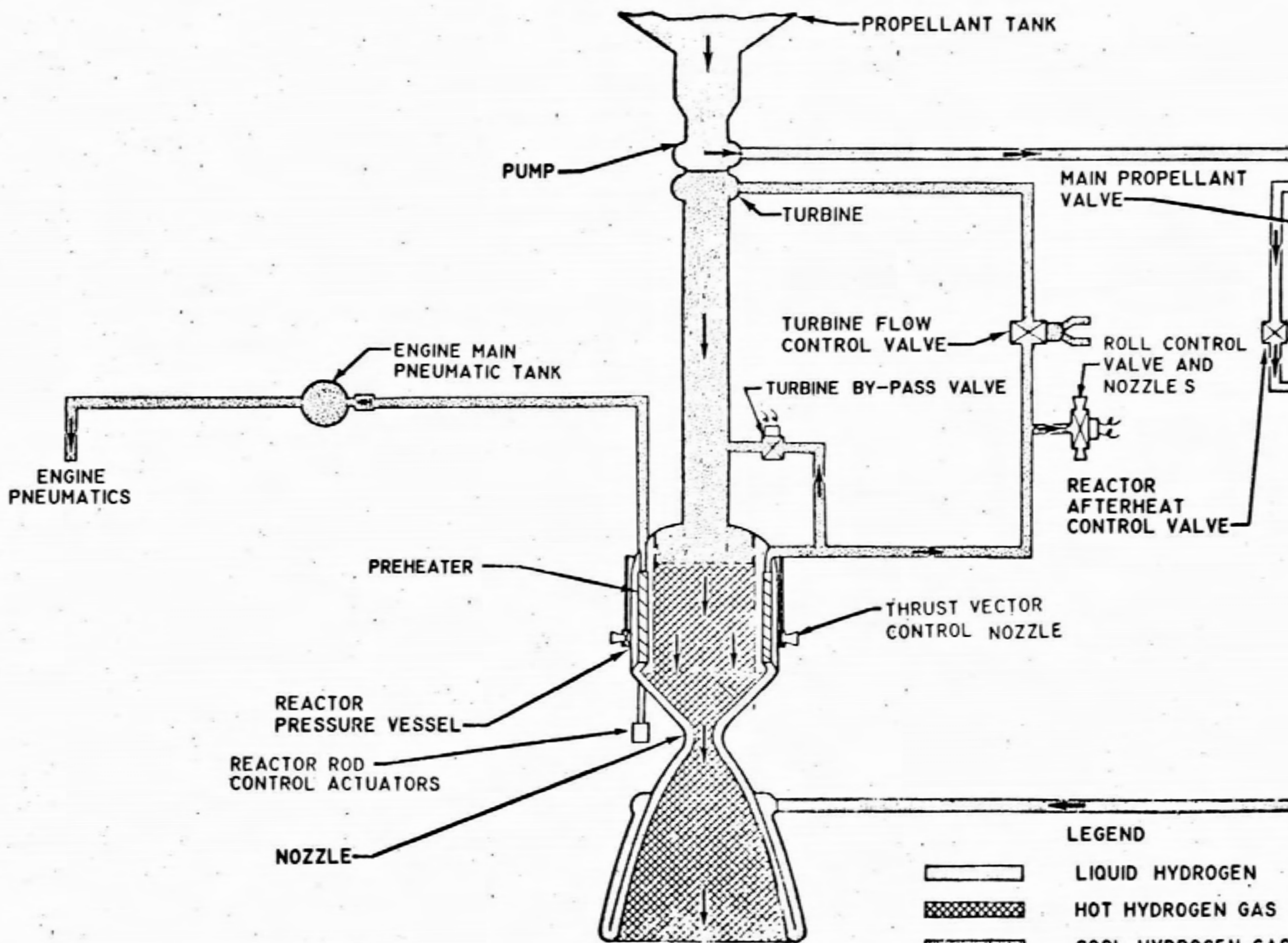

Figure 2. Typical Topping Cycle Engine Schematic

and component design considerations. With regard to the reflector/preheater, provision must be made to provide sufficient heating to the hydrogen prior to its introduction to the turbine to permit reasonable turbine weights. The required energy release in the preheater can be provided by loading this portion of the reactor assembly with fissionable material; this provision, however, will affect the neutronics and the distribution of energy release within the reactor. If the reactor design has been initially specified, it may not be possible to adopt such a design feature.

For a topping turbine, the back pressure on the turbine is essentially the reactor core inlet pressure. Since the capability of the turbine to deliver sufficient power to drive the pump is dependent upon the turbine pressure ratio, the available turbine horsepower is somewhat dependent upon the pump discharge pressure. Thus, a topping cycle demands a more stringent turbopump design. Typical discharge pressures for a topping cycle engine might range as high as 3000 psig; for a competitive bleed cycle engine, however, these pressures might be of the order of 2000 psig. Although the additional head requirements associated with the topping cycle are such that they do not preclude the possibility of design based upon present day technology, their attainment will not be realized without an accompanying increase in weight over a turbopump designed for use in a bleed cycle.

As a result of the high pump discharge pressure, the pressure shell must also be designed to accommodate higher operating pressures. If the pressure shell is exposed to the full pressure of the propellant leaving the coolant passages in the nozzle, the required wall thickness and weight of the shell may be excessive. The utilization of tubular "pass-through," running the full height of the reflector/preheater, can alleviate this problem, but the presence of the required tubes will influence the neutronics in this area of the reactor and may complicate the reactor control design problem. Therefore, the selection of the type of cycle to be used is dependent upon a judicious balance between the advantages to be gained from the slightly reduced core operating temperatures and the disadvantages from the standpoint of additional hardware weight and possible increased complexity.

To achieve an optimum design, the selection of the cycle cannot be considered independently of aspects of the propulsion system, e.g., propellant tankage, chamber pressure, nozzle expansion ratio, reactor tank distance, and shield attenuation factors. For, after all, the engine designer's goal is to achieve, consistent with reliability requirements, a minimum over-all propulsion system weight, which includes engine, tankage, and propellant, to supply the required propulsive energy. Some of the other factors mentioned and how they may influence propulsion system design and the results of an optimization study will be considered in the following paragraphs.

The operation of a nuclear rocket reactor at power levels typified by the assumed design requirement, generates an extremely powerful radiation field, comprising high-energy gamma rays and neutrons. Leakage fluxes of the order of 10^{16} gammas/sec cm^2 and 10^{15} neutrons/sec cm^2 are to be expected. The effects of this radiation on the propulsion system include induced heating of any material exposed to the radiation and possible radiation damage. For example, a cubic inch of steel, exposed to these radiations, would be heated to its melting point in about 1 minute, if provisions were not made to cool it. The utilization of shielding in a nuclear rocket engine system affords a means of meeting this design problem, but if too great a reliance is placed upon its use, severe weight penalties will result. The two areas of the propulsion system most in need of protection are the components and structure located close to the reactor, and the propellant in the vehicle's tanks.

The propellant heating problem will be examined first, because its resolution will then fix the environment to which the components of the engine system will be exposed.

For a fixed reactor power level and power density, the temperature rise of the propellant entering the pump, due to the radiation reactions with the propellant in a given tank, is a function of the spectrum of the flux field, the shielding attenuation factor, and the reactor-propellant tank separation distance. The first two factors are a function of both the shield materials and the composite shield thickness. Since the vapor pressure of liquid hydrogen increases with temperature, it is desirable to minimize propellant heating so that the tank pressure requirements, to meet the pump NPSH requirements, are not excessive. Obviously, propellant heating may be minimized if the tanks are located a sufficient distance from the reactor, but this solution leads to increased thrust structure and interstage weights. If a shield is employed, increased shield thickness will reduce reactor-tank separation distances for a given allowable heat generation rate in the propellant. Therefore, the solution to the problem involves establishing an optimum combination between the pump NPSH requirements, as influenced by the design of a pump's booststage or inducer, the reactor-tank separation distance, and the reactor shield thickness. The optimum combination of shielding distance and attenuation factor will be that combination where the sum total weights of turbopump, propellant tank, pressurization gas and subsystem, start tank, thrust and interstage structures, and composite (top and side) shield yield a minimum. The optimum combination will be different depending upon the mission or application.

As mentioned, the radiation environment affects the components of the engine system, both by induced heating and by possible radiation damage. Some of the components may operate satisfactorily in this environment, because they are adequately cooled by the propellant flowing through them and because the materials of which they are fabricated can tolerate rather high integrated radiation doses. Other components may suffer from self-absorption-induced asymmetrical heating, which may cause distortion problems in close-tolerance machinery, and from differential expansion of materials, brought on by different thermal and nuclear radiation absorption coefficients, which can frustrate the designer's intention for proper structural operation and induced heat removal. Still other components may contain parts, such as plastics and electronic elements, which are particularly vulnerable to nuclear radiations. For the solutions of these problems, provisions may be required for utilization of local shielding and use of special cooling loops. In the case of oils, for lubricating bearings and gears, and hydraulic fluids, for operating servo systems and valves, the designer is forced to consider the use of propellant-cooled bearings and gears and the utilization of pneumatically-operated actuators.

Selection of the design point chamber pressure and nozzle expansion ratio for a nuclear rocket engine system is dependent not only upon the application for which the system is intended, but also upon the maximum permissible pressure drop across the reactor core and the nozzle cooling requirements. When optimizing these two parameters to achieve a minimum propulsion system weight, one cannot be considered without simultaneously considering the other. For space applications, where ambient pressure is, for all practical purposes, zero, the larger the expansion ratio, the higher is the specific impulse achievable. For a given chamber pressure, as expansion ratio is increased, the vacuum thrust coefficient and specific impulse increase, but the nozzle length and nozzle exit area also increase. Thus, an upper limit on expansion ratio exists from envelope limitations.

As has been pointed out, the pressure drop across the core is influenced by the operating chamber pressure. In general, the reactor designer's problem of achieving high power densities is aided by operating at high propellant pressures. Also, an increase in chamber pressure, as one would suspect, will effect a decrease in nozzle dimensions and nozzle weight. However, with increasing chamber pressure, other component operating conditions become more severe, leading to increased component weights and possible increased complexity, particularly nozzle cooling and design problems. Hence, the upper limit on chamber pressure is that point where a noticeable increase in design complexity begins or where the allowable operating limits in pressure and associated nozzle heat transfer conditions begin to be exceeded. The procedure for optimizing chamber pressure and expansion ratio, therefore, will be one where a judicious balance is obtained between minimizing core outlet gas temperature, while meeting the required specific impulse; minimizing over-all propulsion system weight, remaining within the allowable envelope limitations for the engine system; and selecting an operating condition which will not impose undue hardships from the standpoint of individual component design and complexity.

The nozzle operating conditions introduce a formidable design problem. The convective heat load, produced by the very hot hydrogen gases expanding to supersonic velocities, is aggravated by increased chamber pressure. Severity of the resultant heat fluxes is depicted in Fig. 3. For comparison purposes, typical heat transfer rates for liquid rocket thrust chambers using conventional and high-performance chemical propellants are also presented.

In addition to the high convective heat load, the nuclear rocket nozzle is subjected to thermal radiation (from the bottom of the core) and nuclear radiation heat loads. The combined thermal loads will produce very high, gas-side wall temperatures.

If the liquid hydrogen propellant is used as a coolant for the nozzle, some extreme temperature differentials will be created, leading to severe thermal stresses. These stresses can be so high that plastic flow and yielding of the metal walls must occur to relieve the stresses. These conditions obviously tend to limit the design life of the nozzle.

The instrumentation system requirements for a nuclear rocket engine differ from those of a chemical liquid propellant rocket engine in two respects. First, because the reactor power and the turbopump power must be independently controlled, it is essential that reliable measurements be made of critical engine operating parameters. Second, the transducers

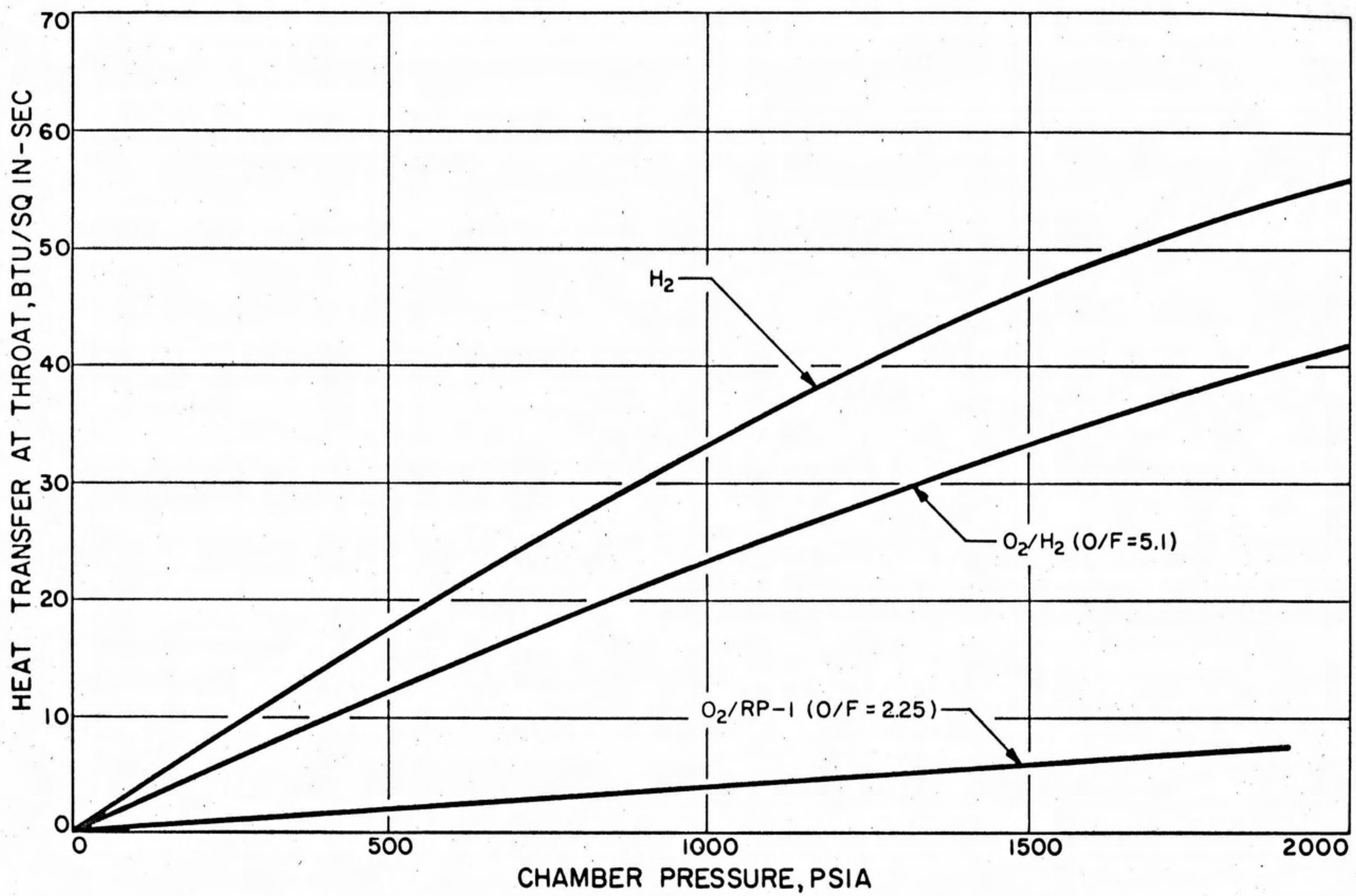

Figure 3. Heat Transfer at Throat vs Chamber Pressure

and transmission systems, employed for controlling and monitoring engine operation, must function reliably and accurately in intense nuclear radiation fields. Therefore, additional effort must be devoted toward the specification and utilization of such items as thermocouples, pressure sensors, strain gages, neutron detectors, and instrumentation cabling which can perform in accordance with requirements of minimum radiation shielding and cooling. In many cases, the requirements of the transducer are more severe than those imposed by the chemical rocket engine, and the need for reliability is greater since failure of a transducer can cause engine shutdown or engine damage.

The situation is aggravated by the neutron-induced activation of engine components. In the development and test phase of a nuclear engine program, transducer replacement is extremely difficult and expensive since the neutron-induced radiation in the components and the test stand will not always permit human access to the engine test stand. Thus, many material and fabrication techniques must be modified in the design of transducers and cables for a nuclear engine system.

Transient performance and control are important areas in the design of a nuclear rocket engine. Thrust buildup rate is limited by the maximum allowable flux rate changes and core thermal rate considerations. Turbopump performance characteristics, as well as reactor neutronics, generally influence selection of startup techniques and require consideration of available turbine power and maximum permissible turbine inlet temperature.

Nuclear engine startup may be divided into three phases: (1) reactor startup, (2) system chilldown, and (3) thrust buildup. These phases will be examined for a typical engine employing a bleed cycle (see Fig. 4).

STARTUP SEQUENCE

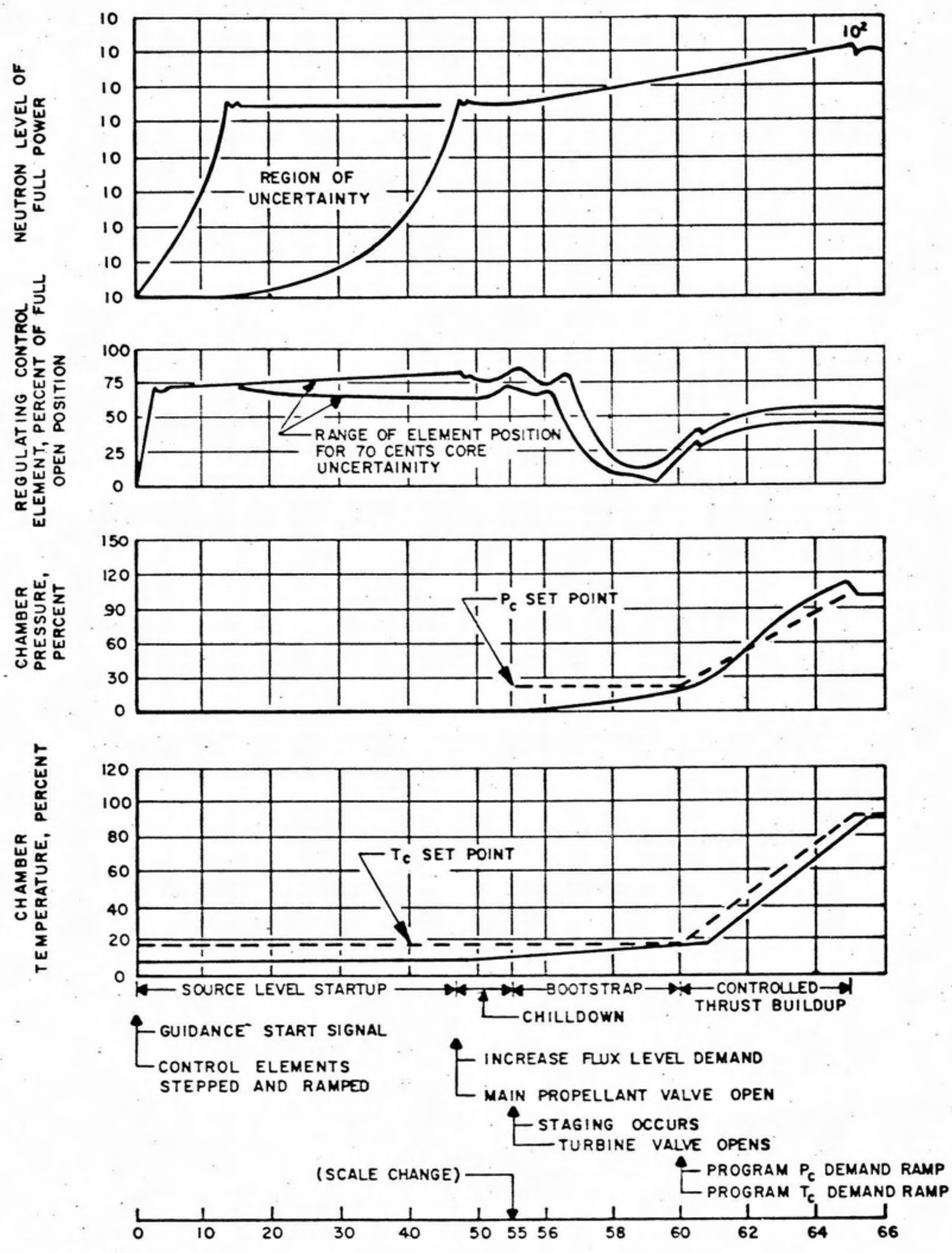

Figure 4. Startup Sequence

At reactor start, the reactor control elements are simultaneously positioned with a programmed signal. The signal first "steps" the elements to a predetermined supercritical position and then "ramps" them at a given reactivity rate. Reactor startup proceeds on an "open loop" or precalibrated basis without any measurement of reactor power level. When reactor power level reaches approximately 0.1-percent of full power, it is assumed to fall within the control range of the system power level nuclear instrumentation and closed loop flux control is initiated to hold the reactor neutron flux at this referenced level.

During the chilldown phase, the temperatures of the feedlines and portions of the nozzle cooling tubes are reduced to their normal operating temperatures so that the system impedance to the flow of a cryogenic fluid does not remain excessive. The chilldown phase is initiated by opening the main propellant valve and programming a reactor power increase. Chilldown flow is produced primarily from tank pressure and varies as the system pressure drop changes. Reactor power is increased during chilldown to provide some core and reflector heating prior to the thrust buildup phase. This action tends to increase the initial thrust buildup rate by providing higher temperature turbine inlet gases.

The primary requirement of the thrust buildup phase is to increase engine thrust to rated level in minimum time and in a reliable and predictable manner. "Bootstrap" is started in less than 10 seconds after initiation of chilldown by opening the turbine throttle valve, with simultaneous controlled increase of reactor power. This action permits a portion of the thrust chamber gases to flow through the turbine and produce torque for turbopump acceleration. As the turbine speed increases, pump flowrate increases which causes the chamber pressure to rise and, in turn, causes increased turbine speed. When the chamber pressure and chamber temperature reach a predetermined value, closed loop control of the turbine throttle is initiated and thrust buildup continues to the maximum design reactor power level.

Consideration must be given to ensuring that the reactor elements do not reach excessive temperatures during the transient buildup stage. In addition, the possibility of pump stall must be avoided; this can occur as a result of excessive system impedance, as previously mentioned, brought about in the feedlines and nozzle cooling tubes, resulting from two-phase flow during the initial chilldown period. Coupled with considerations given to the possible deleterious effect on components, the startup procedure should utilize the propellant so that degradation in propellant performance during this period does not severely reduce over-all vehicle performance.

Engine shutdown contributes problems similar to those encountered during startup. A procedure must be devised whereby the amount of propellant consumed during the shutdown phase is not excessive. A programmed reduction of chamber pressure and temperature must be effected while maintaining a stable operating regime. Coupled with the stable engine performance required during shutdown, compatibility of reactor neutronics with turbopump characteristics is also necessary to prevent the reactor from experiencing excessive temperatures. The problems associated with engine restart are similar to the initial start, with the additional consideration of a wider range of possible core initial temperatures, fuel depletion, fission product poisoning, and required propellant tank pressurization.

Simplified analog or dynamic digital models are usually employed in the analyses required for the solution of the problems mentioned above. In general, the criteria of minimum startup and shutdown time, reproducible thrust transients, and minimum usage of propellant in a low-performance state are the guides in the selection of the control systems for the startup and shutdown phases of engine operation.

After-cooling of the engine will be necessary to maintain a maximum allowable core temperature during the coast period because heat generation following shutdown occurs as the result of fission product inventory which has been built up in the core. Because the amount of cooling required is a function of the past run history, closed loop control of after-cooling propellant flow will be necessary. After-cooling is accomplished with hydrogen propellant and the procedure to be followed must result in a judicious compromise between maintaining temperatures sufficiently low to prevent core or engine system damage, and minimizing total propellant after-cooling usage.

When reactor power decreases sufficiently, a sequence controller closes the main propellant valve and subsequent cooling is supplied by the after-heat valve, arranged to control flow in parallel with the main propellant valve. After-heat coolant is supplied by the bypass valve until coolant requirements are so low that the bypass valve cannot maintain stable control. When this minimum value is reached, the valve will remain in a fixed position and the engine will be over-cooled. Sequence control will then open and close the valve after suitable periods until power generation is so low that thermal radiation alone can cool the reactor/thrust chamber assembly.

The design of a nuclear rocket engine, as for many other space vehicle design tasks, is not a straightforward problem. As was previously mentioned, the selection of engine operating conditions is dependent upon the application for which the engine and vehicle are intended.

To optimize the engine system, information about the over-all vehicle must be known. However, the vehicle designer must know something about the engine system before optimization of the vehicle system can proceed. It is therefore obvious that the two must be in close liaison. Based upon a particular application, the vehicle designer will make certain assumptions regarding the engine and its performance characteristics. This enables the vehicle designer to conduct a first-order trajectory and optimization program. Rough approximations of vehicle gross weight, engine thrust, and duration are established as a result of this initial study.

The engine system designer will then use these results as input to his studies, which permit him to establish optimum conditions resulting in a minimum propulsion system weight, which includes engine, tankage, and propellant, to satisfy the mission requirements. These results, in turn, are fed back to the vehicle system contractor, who substitutes this refined engine information for his initial assumptions. The procedure continues in this manner until both the vehicle system contractor and the engine system contractor converge upon a set of answers that is compatible with the requirements of both. (See Fig. 5.)

A safety system must be incorporated into the engine design to ensure protection against malfunction both on the ground and in flight. A complete system would include surveillance, decision, and action. Sensors and monitoring systems for surveillance would be used in conjunction

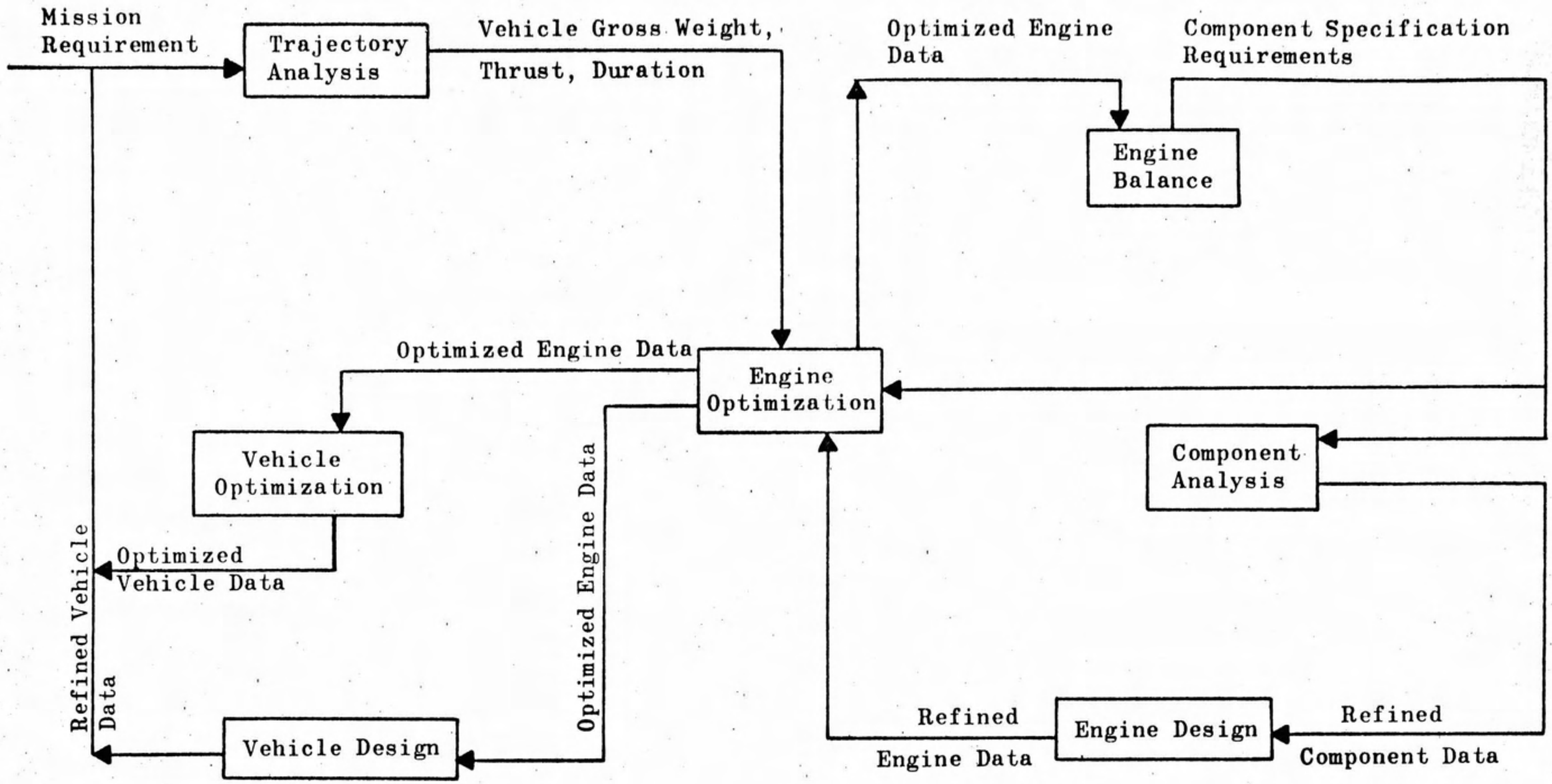

Figure 5 . Engineering Flow Diagram for the Design of a Nuclear Engine-Vehicle System

with the proper action system, depending upon the existing situation. For example, if a reactor control system malfunction occurs on the launch pad, a remotely removable reactor poison must be provided; it must be capable of preventing reactor criticality in the event that effectiveness of all reactor control elements is removed. A reactor fragmentation system, which reduces the core to subcritical masses in either water or liquid hydrogen, might be required in the event of accidental flooding by liquid hydrogen or impact in close, off-shore waters, as a result of a guidance failure. Finally, a reactor pulverization system may be necessary, to accomplish complete dispersal in the event of a malfunction over inhabited areas where fragmentation would result in personnel hazards.

It has not been possible, within the constraints of a paper of reasonable length and security considerations, to develop in detail all of the design considerations for nuclear rocket engines. In addition to the design requirements, as stated or implied in the foregoing discussion, other facets exist which influence the solutions to the design problem, e.g., provisions for remote handling and disassembly, remote replacement of critical components, checkout and monitoring, and trouble shooting at the test or launch stand. These problems must be solved in a practical way. It is hoped, however, that sufficient material has been presented herein to give the reader some appreciation of the designer's job.