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COMPREHENSIVE TECHNICAL REPORT

DIRECT AIR CYCLE | AIRCRAFT NUCLEAR PROPULSION PROGRAM

APEX-910

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AIRCRAFT NUCLEAR PROPULSION APPLICATION STUDIES

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COMPREHENSIVE TECHNICAL REPORT
GENERAL ELECTRIC DIRECT-AIR-CYCLE
AIRCRAFT NUCLEAR PROPULSION PROGRAM

AIRCRAFT NUCLEAR PROPULSION APPLICATION STUDIES

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AEC RESEARCH AND DEVELOPMENT REPORT

GENERAL ELECTRIC
NUCLEAR MATERIALS AND PROPULSION OPERATION
(Formerly Aircraft Nuclear Propulsion Department)
FLIGHT PROPULSION LABORATORY DEPARTMENT
Cincinnati 15, Ohio

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ABSTRACT

This is one of twenty-one volumes summarizing the Aircraft Nuclear Propulsion Program of the General Electric Company. This portion describes the studies of advanced applications of nuclear reactors that were performed, including various types of aircraft, missiles, space vehicles, ships, and portable power plants. In part, the studies are based on the advanced power plants described in APEX-909, "Aircraft Nuclear Propulsion Systems Studies." Although most of the work was concerned with open-cycle, gas-cooled systems, other systems were also investigated, such as closed gas cycle, indirect liquid metal cycle, gas fission, and liquid circulating fuel systems. Air, helium, hydrogen, and neon were considered as coolants for the gas-cooled systems. Except for a portable nuclear system for power generation, all the studies were concerned with propulsion applications.

The application studies show the feasibility both of using reactors developed during the ANP program in advanced vehicles, and of the use of advanced reactors in various types of systems. Performance data, configurations, development program elements and schedules, and estimated costs are included in this summary of the results of the studies.

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PREFACE

In mid-1951, the General Electric Company, under contract to the United States Atomic Energy Commission and the United States Air Force, undertook the early development of a militarily useful nuclear propulsion system for aircraft of unlimited range. This research and development challenge to meet the stringent requirements of aircraft applications was unique. New reactor and power-plant designs, new materials, and new fabrication and testing techniques were required in fields of technology that were, and still are, advancing very rapidly. The scope of the program encompassed simultaneous advancement in reactor, shield, controls, turbomachinery, remote handling, and related nuclear and high-temperature technologies.

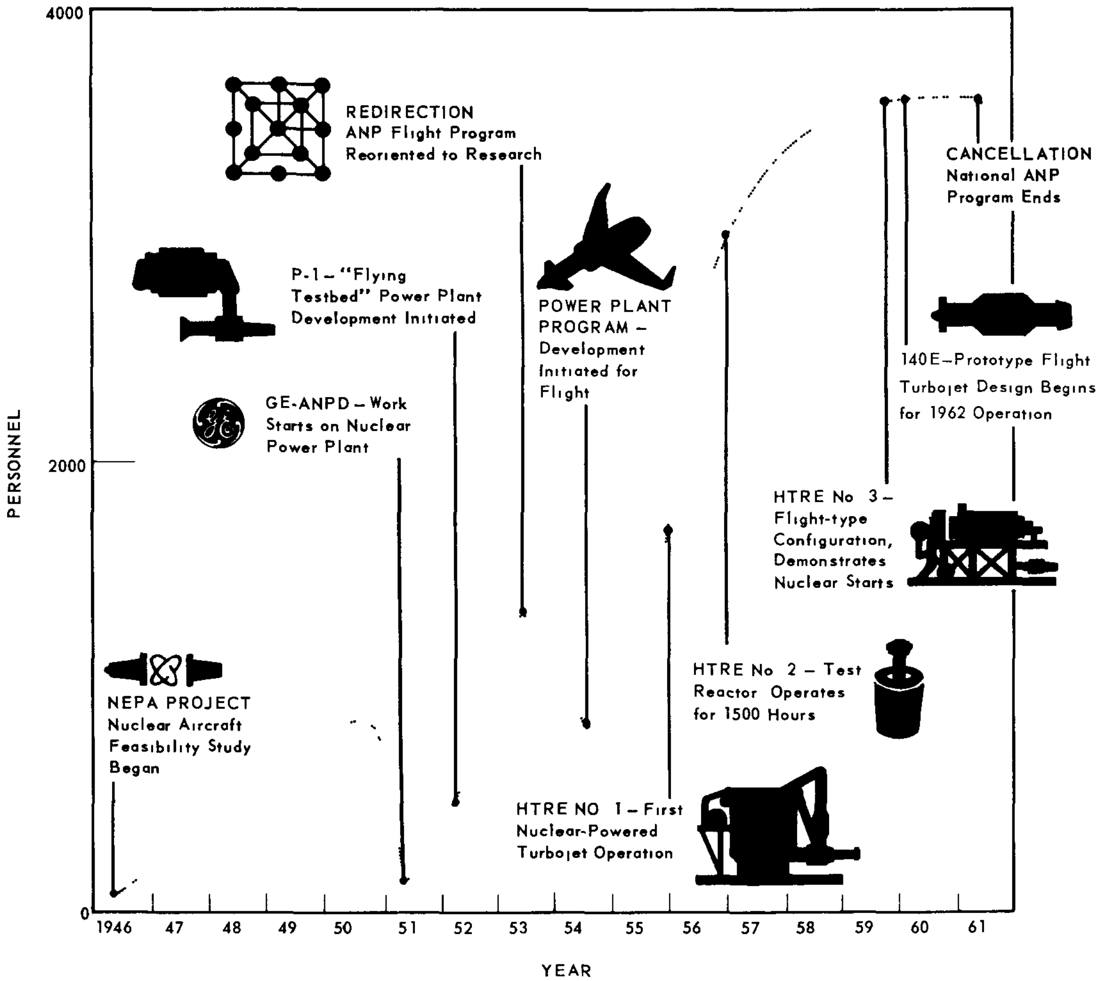
The power-plant design concept selected for development by the General Electric Company was the direct air cycle turbojet. Air is the only working fluid in this type of system. The reactor receives air from the jet engine compressor, heats it directly, and delivers it to the turbine. The high-temperature air then generates the forward thrust as it exhausts through the engine nozzle. The direct air cycle concept was selected on the basis of studies indicating that it would provide a relatively simple, dependable, and serviceable power plant with high-performance potential.

The decision to proceed with the nuclear-powered-flight program was based on the 1951 recommendations of the NEPA (Nuclear Energy for the Propulsion of Aircraft) project. Conducted by the Fairchild Engine and Airplane Corporation under contract to the USAF, the five-year NEPA project was a study and research effort culminating in the proposal for active development of nuclear propulsion for manned aircraft.

In the ensuing ten years, General Electric's Aircraft Nuclear Propulsion Department carried on the direct air cycle development until notification by the USAF and USAEC, early in 1961, of the cancellation of the national ANP program. The principal results of the ten-year effort are described in this and other volumes listed inside the front cover of the Comprehensive Technical Report of the General Electric Direct Air Cycle Aircraft Nuclear Propulsion Program.

Although the GE-ANPD effort was devoted primarily to achieving nuclear aircraft power-plant objectives (described mainly in APEX-902 through APEX-909), substantial contributions were made to all aspects of gas-cooled reactor technology and other promising nuclear propulsion systems (described mainly in APEX-910 through APEX-921). The Program Summary (APEX-901) presents a detailed description of the historical, programmatic, and technical background of the ten years covered by the program. A graphic summary of these events is shown on the next page.

Each portion of the Comprehensive Report, through extensive annotation and referencing of a large body of technical information, now makes accessible significant technical data, analyses, and descriptions generated by GE-ANPD. The references are grouped by subject and the complete reference list is contained in the Program Summary, APEX-901. This listing should facilitate rapid access by a researcher to specific interest areas or



Summary of events – General Electric Aircraft Nuclear Propulsion Program*

*Detailed history and chronology is provided in Program Summary APEX-901 Chronology information extracted from *Aircraft Nuclear Propulsion Program hearing before the Subcommittee on Research and Development of the Joint Committee on Atomic Energy* 86th Congress of The United States First Session July 23 1959 United States Government Printing Office Washington 1959 sources of data. Each portion of the Comprehensive Report discusses an aspect of the Program not covered in other portions. Therefore, details of power plants can be found in the power-plant volumes and details of the technologies used in the power plants can be found in the other volumes. The referenced documents and reports, as well as other GE-ANPD technical information not covered by the Comprehensive Report, are available through the United States Atomic Energy Commission, Division of Technical Information Extension, Oak Ridge, Tennessee.

The Report is directed to Engineering Management and assumes that the reader is generally familiar with basic reactor and turbojet engine principles; has a technical understanding of the related disciplines and technologies necessary for their development and design; and, particularly in APEX-910 through APEX-921, has an understanding of the related computer and computative techniques.

The achievements of General Electric's Aircraft Nuclear Propulsion Program were the result of the efforts of many officers, managers, scientists, technicians, and administrative personnel in both government and industry. Most of them must remain anonymous, but particular mention should be made of Generals Donald J. Keirn and Irving L. Branch of the Joint USAF-USAEC Aircraft Nuclear Propulsion Office (ANPO) and their staffs; Messrs. Edmund M. Velten, Harry H. Gorman, and John L. Wilson of the USAF-USAEC Operations Office and their staffs; and Messrs. D. Roy Shoults, Samuel J. Levine, and David F. Shaw, GE-ANPD Managers and their staffs.

This Comprehensive Technical Report represents the efforts of the USAEC, USAF, and GE-ANPD managers, writers, authors, reviewers, and editors working within the Nuclear Materials and Propulsion Operation (formerly the Aircraft Nuclear Propulsion Department). The local representatives of the AEC-USAF team, the Lockland Aircraft Reactors Operations Office (LAROO), gave valuable guidance during manuscript preparation, and special appreciation is accorded J. L. Wilson, Manager, LAROO, and members of his staff. In addition to the authors listed in each volume, some of those in the General Electric Company who made significant contributions were: W. H. Long, Manager, Nuclear Materials and Propulsion Operation; V. P. Calkins, E. B. Delson, J. P. Kearns, M. C. Leverett, L. Lomen, H. F. Matthiesen, J. D. Selby, and G. Thornton, managers and reviewers; and C. L. Chase, D. W. Patrick, and J. W. Stephenson and their editorial, art, and production staffs. Their time and energy are gratefully acknowledged.

November 8, 1961

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1. INTRODUCTION AND SUMMARY

This volume consists of summaries of advanced application studies of nuclear power plants performed by the Aircraft Nuclear Propulsion Department of the General Electric Company. The term "advanced application" is used with two connotations. The first refers to the application of nuclear power plants developed during the ANP program in vehicles or systems other than those for which they were originally designed. The second meaning refers to variations of the basic ANP power plants, or completely new ones, as applied to various vehicles or systems. The field of advanced applications, therefore, was in addition to, or beyond, the primary ANP program effort at any given time. For example, when the primary effort of the ANP program was on the XNJ140E-1 power plant for the Convair NX-2 airplane, the study of this power plant in a B-52 aircraft was considered an "advanced application." Similarly, studies of the P140Y2 power plant, a high-performance, supersonic version of the XNJ140E-1, in any vehicle were also "advanced applications."

The purpose of the work was to provide a basis for planning future nuclear weapons systems for the national defense, and to advance the technology of nuclear reactors by finding useful applications in various fields. The applications work was closely coordinated with the developmental phases of the ANP program: preliminary design, conceptual design, and materials development effort. Some of the early studies on nuclear ramjets and rockets, summarized in this volume, influenced later government programs such as ROVER and PLUTO. Early applications studies also influenced the direction of the manned nuclear aircraft program. Many of the studies utilized advanced power plants described more fully in APEX-909, "Aircraft Nuclear Propulsion Systems Studies," of this Report.

The applications studies were conducted throughout the duration of the ANP program and covered a broad range of vehicle applications and types of reactors. They are diverse in nature because of the great variation in weapons systems considered during the ten years of the program, and the multitude of requests for analyses and information. For this reason, the studies are presented essentially by the type of application rather than chronologically.

The first section of this volume summarizes subsonic aircraft applications studies. Both existing and parametric or "paper" aircraft were investigated. Turbojets, turbofans, and turboprops were considered, utilizing direct-air, indirect-liquid-metal, and liquid-circulating-fuel cycles.

Studies of supersonic aircraft powered by direct-air-cycle turbojet engines are presented in the second section. Both high-speed, high-altitude systems and the hunter/killer mission were investigated and are described.

The next section covers nuclear ramjet, turbojet, and rocket missile applications. A study of the feasibility of applying nuclear turbojet power plants to the "Snark" missile is included. The nuclear ICBM application is reported here because its mission places it more logically in the missile than in the rocket category.

The section on nuclear rocket applications includes work done in 1956 on heat-transfer rockets and more recent studies of the NERVA-Phoebus system. It also presents a concept of a gas-fission rocket propulsion system studied in 1956.

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Subsequent sections cover a study on nuclear-powered helicopters, nuclear hydrofoil propulsion, an examination of gas-cooled reactors for portable power plants in accordance with Army requirements, an analysis of helium-cooled, closed-cycle power plants for turbojets, compressor-jets, and turboprops, and a study of nuclear turboprop airship application.

Following these sections, several of the land, sea, undersea, and space applications of the 601 series of power plants are discussed. These are compact, integral, neon-cooled, closed-cycle, fast-reactor power plants that illustrate the strides made in gas-cooled nuclear reactor technology from the beginning to the end of the ANP program.

The final section of the report is devoted to a concept of aerospace nuclear propulsion that would permit flight from the Earth to Mars, inspection of that planet, and return, by re-entry, to Earth. The basic concept for this mission is the utilization of a single nuclear system that operates as a turbojet in dense atmospheres, as a ramjet in rarified atmospheres, and as a pure rocket in the vacuum of space.

2. SUBSONIC AIRCRAFT

2.1 CONVERSION OF C-99 CARGO AIRCRAFT

A study was made in 1953 of the feasibility of converting a chemically fueled C-99 cargo aircraft to a nuclear-fueled cargo plane or tanker.[1*] As originally designed, the C-99 was powered by six R-4360 engines, had a maximum take-off weight of 357,500 pounds, and could accommodate 100,000 pounds of cargo. The study indicated that the nuclear-powered C-99 could carry 100,000 pounds of cargo with acceptable performance.

The following major modifications would have been necessary for the nuclear version:

1. The six reciprocating engines and accessory systems removed from the wing.
2. One nuclear-fueled power-plant package installed in the lower aft cargo compartment.
3. A shielded three-man crew compartment installed in the vicinity of the existing crew compartment.
4. Two chemically fueled J77 turbojet engines installed in the wings for augmented thrust.

The resulting airplane configuration is shown in Figure 2.1.

The AC-2, the nuclear power plant for this airplane, consists of a tubular-type reactor with a wire-screen fuel element, powering two turbojet engines mounted on either side of the reactor-shield assembly. The power plant weight is 65,000 pounds, and each turbojet engine has an airflow of 300 pounds per second at sea level static. At a turbine inlet temperature of 1650°F, the power plant delivers 35,500 pounds of static thrust at sea level. The two chemically fueled J77 turbojet engines installed in the wings are to provide additional thrust during take-off and at off-design conditions.

The lead and polyethylene-plastic crew shield, weighing 20,000 pounds, limits the radiation dose to the crew to about 0.5 roentgen per hour.

The weight breakdown of the system is shown in Table 2.1.

2.2 LIQUID-CIRCULATING-FUEL-REACTOR POWER PLANTS

2.2.1 INTRODUCTION AND SUMMARY

A design study was made in 1953 of a series of nuclear power plants for manned turbojet bombers utilizing liquid-circulating-fuel reactors and ternary heat-transfer systems.[2] The temperature of the liquid fuel at the reactor discharge was assumed to be limited to 1500°F by the properties of the Inconel structural material. The study indicated that the power-plant weight per pound of thrust is strongly dependent upon the attainable power density in the liquid-circulating-fuel reactor. Insufficient data were available at the time

*Superscripts refer to the reference lists at the end of each section.

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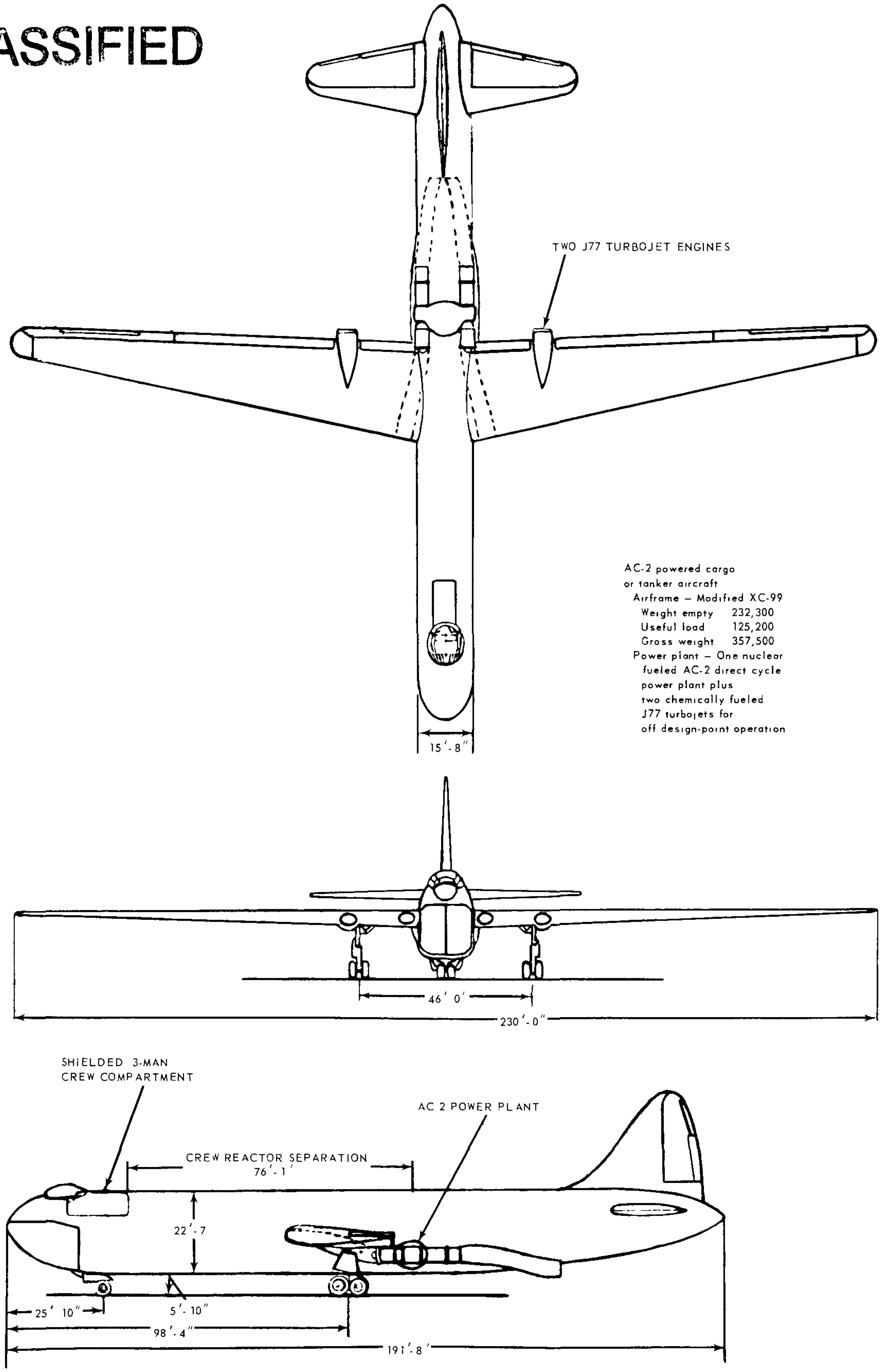

Fig. 2.1 – C-99 airplane – nuclear version

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TABLE 2.1

WEIGHT BREAKDOWN OF NUCLEAR-POWERED C-99 CARGO AIRCRAFT

Component	Weight, lb
Airframe	109, 750
Power plant (including nuclear and chemical engines, moderator cooling system, and all shielding)	122, 550
Total empty weight	232, 300
Useful load	125, 200
Chemical fuel	12, 500
Take-off gross weight	370, 000

to predict accurately the power density that might be attainable. Therefore, two types of power plants were considered: (1) those using a "probably attainable" average reactor power density of 1.7 to 2.5 kilowatts per cubic centimeter, and (2) those using a "less-probably attainable" average reactor power density of 3 to 5 kilowatts per cubic centimeter. The group of "probable" power plants includes the following three examples that utilize 8, 6, or 4 modified General Electric J73 turbojet engines:

1. The basic power plant design utilizes eight modified J73 turbojet engines equipped with NaK-to-air radiators as well as chemical combustion chambers for take-off and off-design-point operation. This power plant, designated the LF-2, was designed to propel a heavy bomber of the B-52 type at 35, 000 feet and Mach 0.75, with a reactor power of approximately 100 megawatts (see Figures 2.2 and 2.3).
2. An alternative power plant, the LF-1, utilizing the same reactor but only six engines, can power a medium bomber similar to the B-47 at 35, 000 feet and Mach 0.75, the reactor generating approximately 76 megawatts.
3. A third power plant (LF-3), based on the same reactor but with only four modified J73 engines, appears suitable for flying a sea-level "hedge-hopper"-type aircraft at speeds up to Mach 0.90, with a reactor power of approximately 115 megawatts.

The group of "less probable" power plants includes three configurations utilizing 8, 6, or 4 General Electric 7E-XJ53-GE-X21 turbojet engines. These power plants have a lower weight-to-thrust ratio than the power plants with a lower reactor power density.

The shield for each of the foregoing power plants is of a quasi-unit design. The radiation dosage to the crew is limited to one roentgen per hour. The radiation dosage at the shield surface of the reactor is limited to 600 r/hour at full power or 6 r/hour at one percent of full power. Therefore, it is possible for ground personnel to carry out visual inspection and minor tune-up work with the reactor operating at from one to ten percent of design power, the turbojet engines being turned at reduced speed by a continuous starter.

The estimated thrust-to-weight ratio for the liquid-fuel power plants utilizing quasi-unit shields appears to be nearly competitive with the thrust-to-weight ratio for the AC-series direct-cycle power plants with divided shields, for operation at 35, 000 feet at high-subsonic speeds. For sea-level operation, the AC-series power plants appear to be definitely superior to the liquid-circulating-fuel power plants on the basis of thrust

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per pound of power-plant weight. The liquid-circulating-fuel power plants are more vulnerable to damage by enemy action, primarily because of the fire hazard associated with the use of NaK as a secondary coolant. Furthermore, the reliability of a liquid system, based on existing technology in the handling of liquid metals, is probably lower than for an air-cycle power plant. An accurate appraisal of the potential reliability of liquid-circulating-fuel systems would have required extensive testing of heat exchangers, radiators, pumps, and other components specifically designed for aircraft use, in addition to the operation of liquid-fuel reactors and eventually a full-power ground-test reactor.

2.2.2 POWER-PLANT DESIGN AND PERFORMANCE

A circuit schematic is shown in Figure 2.4 of a liquid-circulating-fuel-reactor power plant. Figure 2.5 shows the detail of the reactor-shield assembly and Figure 2.6 is a layout of the modified J73 engine used. Data are given in Table 2.2 for the three power plants previously described, the LF-1, LF-2, and LF-3.

Preliminary estimates were made for three additional power plants, designated LF-4, LF-5, and LF-6, that utilize 7E-XJ53-GE-X21 engines and require considerably higher reactor power densities than the LF-1, LF-2, and LF-3. The design of the reactor core and intermediate heat exchanger of all six power plants is identical, but the shield diameter and the number and type of turbojet engines are different.

2.2.3 RESULTS

A breakdown of the estimated component weights for each power plant is given in Table 2.3. Estimated weights for aircraft similar to the B-47 and B-52 are given in Table 2.4. The turbojet engines are assumed to be mounted in the fuselage as close to the reactor as possible. Extensive modifications would have had to be carried out on either airplane to accommodate this type of power plant, which constitutes an extremely concentrated load with engines or fuel in the wings, to relieve wing bending moments. Modification in the empennage would also have been necessary to avoid damage by the turbojet exhaust.

2.2.4 CONCLUSIONS

The conclusions of this study were:

1. Power plants based on existing information and utilizing liquid-circulating-fuel reactors are nearly competitive with the AC-series direct-cycle power plants from the standpoint of power plant weight per pound of thrust at a design altitude of 35,000 feet and a speed of Mach 0.75.
2. For performance at sea level and a speed of Mach 0.90, the liquid-fuel power plants with a quasi-unit shield weigh 50 to 75 percent more than the direct-cycle AC-series power plants with a divided shield in the range of total thrusts up to about 30,000 pounds.
3. There is evidence that the liquid-fuel power plants compare more favorably with the direct-cycle at very high thrust ratings, particularly for high-altitude performance.
4. The total U^{235} investment for the liquid-circulating-fuel reactors is about 125 pounds compared to about 150 pounds for the R-1 reactor, 50 pounds for the AC-1, and 100 pounds for the two-reactor AC-1B. For equivalent thrust ratings, there appears to be no significant advantage, with respect to uranium investment per power plant, for the liquid-circulating-fuel reactor compared to advanced-design direct-cycle reactors. However, no comparison was made of over-all U^{235} inventory, holdup in chemical processing, fabrication, etc., during this study.
5. It was concluded that a ternary system, using NaK as the secondary coolant, is the most feasible method of energy transfer from a circulating-fuel reactor to turbojet engines when a quasi-unit shield is employed. However, such a system constitutes a serious fire hazard for use in aircraft.

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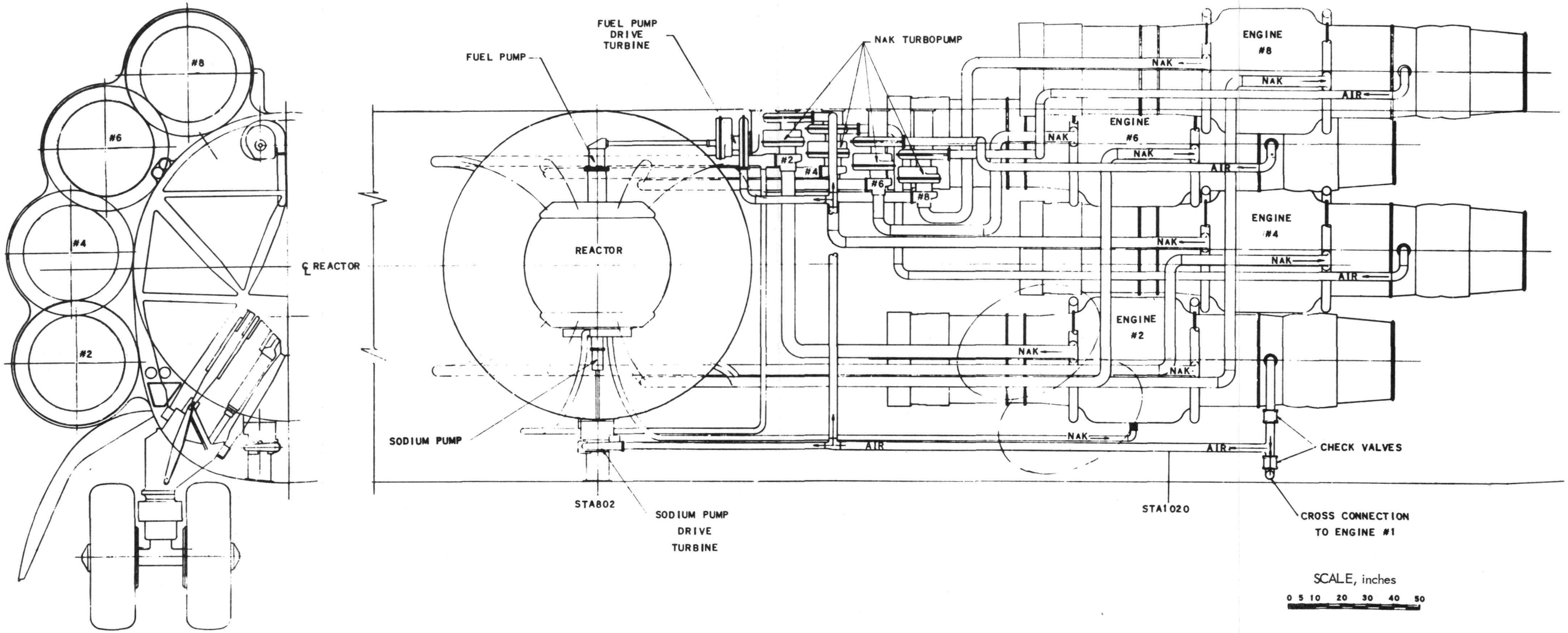

Fig. 2.2 – Typical LF-2, 8-engine circuit, B-52 fuselage installation

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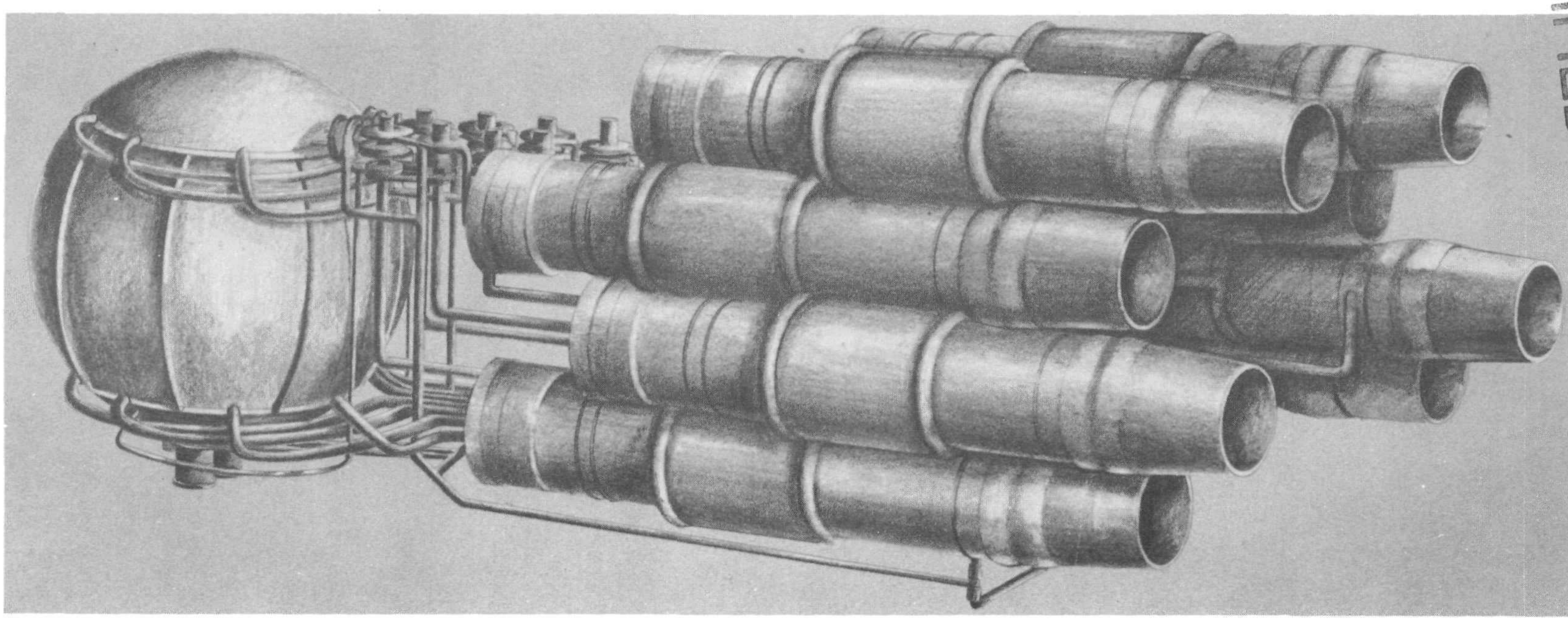

Fig. 2.3 – LF-2 power plant perspective, B-52 fuselage installation

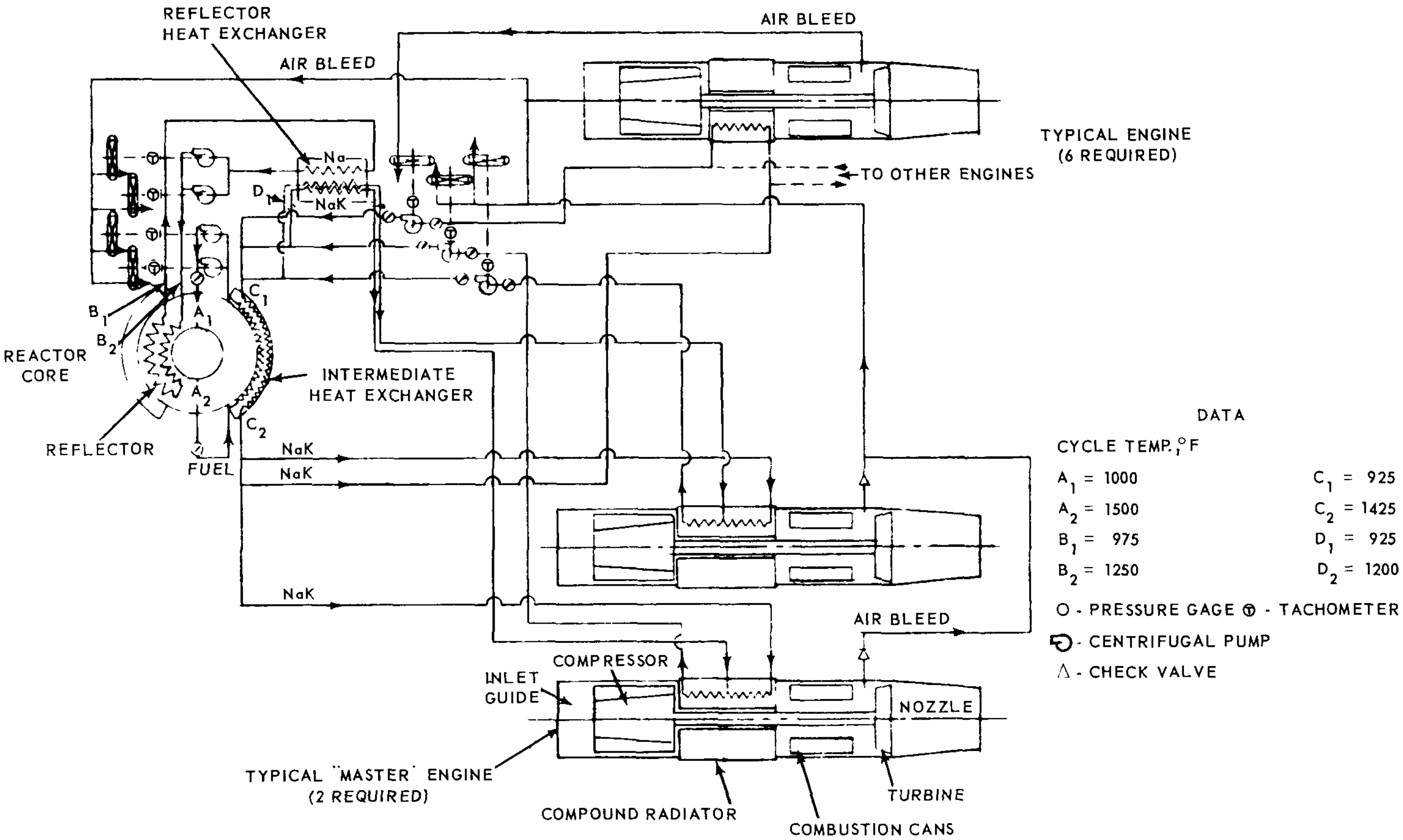

Fig. 2.4 – LCFR circuit schematic

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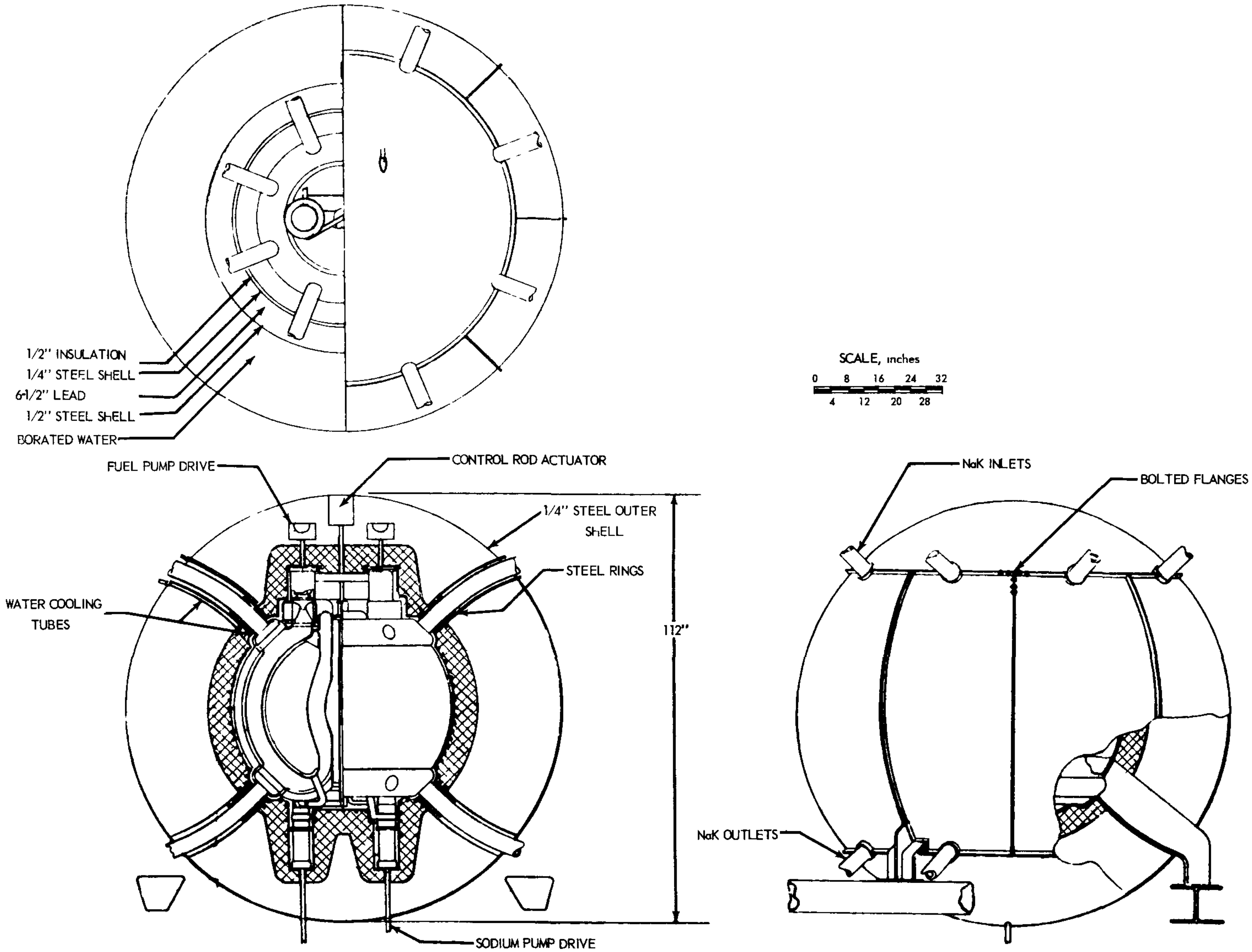

Fig. 2.5 – Reactor shield assembly, LF-2 power plant

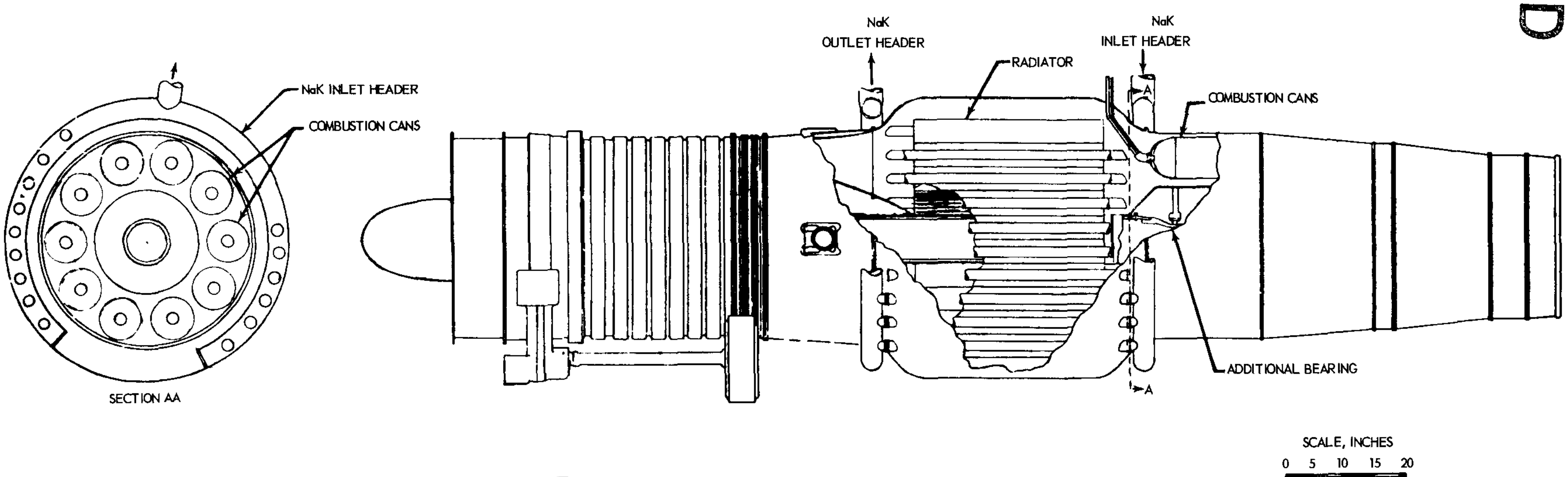

Fig. 2.6 – Preliminary layout, modified J73-GE-3

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TABLE 2.2

SUMMARY OF DESIGN DATA

	Power Plant		
	LF-1	LF-2	LF-3
Aircraft type (similar to)	B-47C	B-52	"Hedge-hopper"
Number of modified J73 engines	6	8	4
Design altitude, ft	35,000	35,000	sea level
Design speed, Mach No.	0.75	0.75	0.90
Design speed, knots	500	500	550
Reactor Data (Design Point)			
Reactor power, Btu/sec	72,000	96,000	109,000
Reactor power, mw	76	101	115
Fuel inlet temperature, ^{o}F	1,000	1,000	1,000
Fuel outlet temperature, ^{o}F	1,500	1,500	1,500
Total fuel flow rate, ft^3/sec	2.87	3.83	4.35
Total fuel flow rate, gpm	1,290	1,720	1,950
Total fuel flow rate, lb/sec	360	480	545
Total fuel pressure drop, psi	14	25	32
Pump head required, ft	17	30	38
Total pump power required, hp	19	45	66
Beryllium Cooling Data			
Na inlet temperature, ^{o}F		925	
Na outlet temperature, ^{o}F		1250	
Power dissipated in island, kw		800	
Na flow rate in island, lb/sec		7.8	
Power dissipated in external reflector, kw		4200	
Na flow rate in external reflector, lb/sec		41	
Turbojet Data (Modified J73)			
Compressor pressure ratio	8.05	8.05	5.45
Turbine inlet temperature, ^{o}F	1,270	1,270	1,110
Diffuser efficiency (assumed)	1.00	1.00	1.00
Compressor efficiency	0.75	0.75	0.82
Turbine efficiency	0.90	0.90	0.90
Nozzle efficiency	0.95	0.95	0.95
Ratio of turbine inlet to compressor discharge total pressures	0.94	0.94	0.97
Net thrust per engine, lb	2300	2300	4320
Total thrust, lb	13,800	18,400	17,200
Airflow per engine, lb/sec	56.6	56.6	204
Specific impulse, lb_F-sec/lb_M	41	41	21
Main Radiator Data			
NaK inlet temperature, ^{o}F	1,425	1,425	1,425
NaK outlet temperature, ^{o}F	925	925	925
Total NaK flow, ft^3/sec	12.3	16.4	18.6
Total NaK flow, gpm	5,500	7,350	8,350
NaK flow per engine, ft^3/sec	2.05	2.05	4.65
NaK flow per engine, gpm	920	920	2,080
Total NaK flow, lb/sec	576	770	872
Total NaK pressure drop, psi	35	49	130
Pump head required, ft	108	150	400
Pump power required, hp/engine	31	44	265
Reactor Shield Data (lead and borated water)			
Over-all diameter, in.	109	112	114
Dosages at full power,			
50 feet from reactor (except in front cone), r/hr	6	6	6
Crew compartment, r/hr	1	1	1
16 feet from reactor (except in front cone), r/hr	60	60	60
Shield surface (except in front cone), r/hr	600	600	600

[redacted]

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TABLE 2.3

COMPARISON OF LIQUID-CIRCULATING-FUEL AND DIRECT-CYCLE POWER PLANTS

	Power Plant							
	LF-1	LF-2	LF-3	LF-4	LF-5	LF-6	AC-1B	AC-1
Number of engines	6-J73's	8-J73's	4-J73's	6-X21's	8-X21's	4-X21's	2	1
Design reactor power, mw	76	101	115	135	180	232		
Component Weight, lb								
Reactor shield assembly	74,000	75,000	76,000	77,000	81,000	84,000	66,000[a]	35,000[a]
Turbojet engines and radiators[b]	45,000	60,000	30,000	65,700	87,600	43,800		
Piping and insulation, dry[c]	3,000	4,000	2,000	5,400	7,200	3,600		
NaK in piping[d]	1,500	2,000	1,000	2,700	3,600	1,800		
Crew shield	3,600	3,600	3,600	3,600	3,600	3,600	20,000	20,000
Shield-moderator cooling system	300	300	300	600	600	600	10,000	5,000
Total accessories	2,400	3,100	1,800	4,200	5,600	3,200		
Miscellaneous	5,000	5,000	5,000	5,000	5,000	5,000	5,000	5,000
Total power-plant weight	134,800	153,000	119,700	164,200	193,600	145,600	101,000	65,000
Total thrust at 35,000 ft, M = 0.75; lb	13,800	18,400	9,200	24,000	32,000	16,000	14,000	7,000
Weight per pound of thrust at altitude	9.8	8.3	13.0	6.8	6.0	9.1	7.2	9.3
Total thrust at sea level, M = 0.90; lb			17,200			29,200	31,600	15,800
Weight per pound of thrust at sea level			7.0			5.0	3.2	4.1

[a]Including turbojet engine/s
[b]7500 lb/engine for J73 and10,950lb/engine for X21
[c]500 lb/engine for J73 and 900 lb/engine for X21
[d]250 lb/engine for J73 and 450 lb/engine for X21

TABLE 2.4

AIRCRAFT WEIGHT ESTIMATES

	Aircraft Type	
	B-47C	B-52
Power-plant type	LF-1	LF-2
Number of J73 engines	6	8
Weights, lb		
Power plant, including reactor shield	131,200	149,400
Crew shield	3,600	3,600
Aircraft structure	45,770[a]	105,600[b]
Fixed equipment	12,640[a]	12,700[b]
Miscellaneous equipment and crew	2,270[a]	3,000
Chemical fuel	10,000	20,000
Bomb load	20,000	20,000
Gross weight	225,480	314,300
Maximum lift/drag ratio	18	20
Thrust required, lb	12,500	15,700
Thrust available at 35,000 ft, M = 0.75; lb	~13,800	~18,400

[a]Reference: WASH-24.
[b]Reference: NEPA-1639.

[redacted]

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2.3 NAVY NUCLEAR SEAPLANE

2.3.1 INTRODUCTION AND SUMMARY

Naval requirements for a low-level attack-bomber seaplane were the basis for a study[3] of a compact-core reactor (CCR) of the type proposed by the Nuclear Development Corporation of America. In this type of power plant, a liquid-metal system supplies turbojet engines with heat from a liquid-metal-cooled reactor with a solid-fuel source. Reactor information used in the study was provided by the Nuclear Development Corporation.

The study was conducted in 1955 on the basis of the requirements and the additional assumptions shown in Tables 2.5 and 2.6. A review of the aircraft performance was conducted with both the Martin Company and Convair-San Diego, who were making independent studies at the request of the Navy. The design conditions selected for nuclear cruise were 30,000 feet altitude and a speed of Mach 0.9. It was decided that the engines must be based upon the existing technology and that either two or four engines should be used.

The auxiliary power system requirements for this type of power plant are unique in that, in addition to normal aircraft requirements, a large amount of power is necessary at all times for the operation of the power plant itself. Because of this large constant power requirement, the auxiliary power unit should be capable of operation on nuclear as well as chemical heat sources. Again, because of the dependence of the power plant upon the auxiliary power unit, an adequate emergency backup must be provided. An additional duty of the auxiliary system is to provide a heat sink for the removal of after-heat after reactor shutdown, and a heat source for maintaining the metal coolants in a liquid state when necessary.

It was believed that the pilot should have a single-stick control that will provide thrust upon demand. The flight engineer, however, must be permitted to override the automatic control manually when necessary and to select nuclear or chemical power or combinations of the two as desired. Because this is in effect a single-engine power plant, the reactor must not be permitted to "scram" completely, but instead must "set back" to a safe power level. Under these conditions, chemical interburning must be provided in order to maintain thrust sufficient for flight.

A 1000-hour life was desired for the power plant (this life, however, was not required of the fuel elements). In order to take full advantage of the nuclear power plant, aircraft operation must not be limited except by crew dose and, in the case of some operational conditions, by chemical fuel supply. Basically, this means that there should be no temperature or atmospheric-condition limitations on the operation of the aircraft. The power plant should be designed to permit inspection and service. Special techniques, however, are required for the inspection and service of any type of nuclear power plant.

Because of radiation hazards and the extreme difficulty in connecting and disconnecting liquid-metal lines, the installation design should permit its removal and replacement as a unit. It was not recommended that any part of the power plant be disassembled while still in the aircraft.

A brief study was made of the possibility of placing the liquid-metal-to-air radiator in the fuselage external to the engines. It was determined that because of the size of the ducting (up to 30 inches in diameter) and the resultant pressure drop from the diffusers, headers, and turns (approximately 14 percent with a 5 percent radiator pressure drop), as well as the complexity of the installation, the in-line radiator is more practical.

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TABLE 2.5

BASIC REQUIREMENT FOR A NAVAL NUCLEAR SEAPLANE

	Desired	Minimum or Maximum
Speed - sea level dash	Mach 0.9	Mach 0.87
Cruise speed at 25,000 ft (min)	440 knots	440 knots
Minimum glide speed	Mach 1.20	Mach 0.95, Min
Rate of climb, sea level - nuclear	1200 ft/min	300 - 600 ft/min
Take-off time	30 sec	Under 60 sec
Reactor power	120 mw	160 mw Max
Radius	8,000 naut mi	8,000 naut mi
Dash radius	100 naut mi (20,000 lb armament)	50 naut mi (20,000 lb armament) 100 naut mi (8,000 lb armament)
Payload	30,000 lb Min	30,000 lb
Armament	12,000 lb Min 20,000 lb Max	8,000 lb Min
Gross weight	180,000 lb	300,000 lb Max
Number of engines	4	2
Minimum nuclear altitude	Sea level	Sea level
Average flight dose	0.030 r/hr	0.1 r/hr
Average ground dose	0.016 r/hr	Safe for reasonable operation - lab tolerance
Date	1961	Soon as possible
Reactor life (fuel life)	500 hr	200 hr (5 missions)
Take-off speed	100 knots	140 knots
Temperature - NaK	1500^{o}F	
- Direct air cycle	1800^{o}F-turbine-inlet temperature	

TABLE 2.6

ADDITIONAL ASSUMPTIONS FOR A NAVAL NUCLEAR SEAPLANE

Reactor power: 120 mw
Design conditions:
  30,000 ft, Mach 0.9 on nuclear power
  Sea level, Mach 0.4 on nuclear power
  Sea level, Mach 0.9 on nuclear plus chemical power
Engines (two or four): present state of the art
Auxiliary power system:
  Provide power plant and aircraft power requirements
  Operate on nuclear power, chemical power, or both
  Provide for a 60 percent backup for power supply
  Provide heat sink for afterheat, or heat supply, as required
Control:
  Pilot to have single-stick control
  Manual overrides to be provided
  "Setback" with automatic chemical topping instead of "scram"
General:
  1000-hour life - except for fuel elements
  Freedom of aircraft operation limited only by crew dose and chemical fuel
  Routine inspection and service permissible
  Unit power package replaceable at sea
  Capable of operation in all climates and under standard cold-to-hot temperature range
  Radiator to be within engine structure

[redacted]

2.3.2 POWER-PLANT DESCRIPTION

As designed, the power plant uses a solid-fuel-element, liquid-sodium-cooled, beryllium-moderated and reflected reactor. Reactor control is achieved by means of rotating reflector drums. A lead and water shield is utilized, with provision for draining part of the water and replacing it with mercury when shield augmentation is required. The reactor power is 120 megawatts, and the shield is designed for a dose of 50 rem per hour at 50 feet. The shield is of an isotropic design.[4]

The primary coolant, liquid sodium flowing at a rate of 6700 gpm at full power, leaves the reactor at 1550°F and enters the intermediate heat exchanger. The sodium leaves the heat exchanger at 1050°F and flows back to four pumps, each of which has a 2000-gpm total capacity. Under normal conditions of full-power operation, each pump puts out 1675 gallons per minute. The entire sodium system, including the heat exchanger, is within the shield shell.

The NaK, flowing at 10,000 gpm, enters the heat exchanger at 1000°F, and its temperature is raised to 1500°F. The heated NaK flows to the radiators of the main engines and the auxiliary power units. At full-power operation at design point the NaK is cooled to 1000°F in the main-engine radiators. It then flows to four 3000-gpm full-rating pumps. Under normal full-power operation each pump delivers 2500 gpm.

In operation at other power levels, the sodium and NaK temperature are maintained at a constant level by means of radiator bypasses and by variation of flow rates.

The liquid-metal system that transmits the NaK requires lines 14 inches in diameter for combined flow and 10 inches in diameter for the flow for each engine. Four-inch lines must be provided for the auxiliary power units. On-off, bypass, check, and throttling valves are provided as required.

The main engine radiator is mounted between the turbojet compressor and the combustion section. At design-point operation (30,000 feet, Mach 0.9, 120 mw), the air is heated to a maximum of 1400°F as it leaves the radiator. The proposed heat-transfer surface is a wavy fin - plate fin type, because it provides the smallest, lightest unit. It was believed that the state of development of this unit, however, might eliminate it from consideration for the first power plants.

The turbojet engine was designed for a 456-pound-per-second SLS airflow and an 8-to-1 compressor pressure ratio. Its design was based on conventional components and on the technology developed in the J79 engine. The engine includes an interburner between the radiator and the turbine for chemical or combined chemical and nuclear operation. Normal sea level and altitude cruise are performed on nuclear power only. The chemical interburner must be used for the sea level sprint. A subsonic afterburner is provided for use only during take-off.

The auxiliary power system provides two auxiliary power units that drive the eight liquid-metal pumps. The NaK pumps require 730 horsepower, and the sodium pumps require 390 horsepower. Aircraft accessory power is provided under normal conditions by turbines driven by air bled from the main engines. Under emergency conditions, the auxiliary power units also will provide sufficient bleed air at altitude for partial accessory power, and at sea level for full accessory power. Upon shutdown of the main engines, the auxiliary power units serve as a sink for afterheat removal. These units can operate on nuclear or chemical power, or a combination of both.

Installation of the power plant in a typical seaplane is shown in Figure 2.7. In this artist's concept, the engine has been cut in half to show the radiator and one auxiliary power unit removed.

[redacted]

2. 3. 3 POWER-PLANT PERFORMANCE

The complete power plant with all required accessories weighs approximately 106, 000 pounds, assuming an isotropic shield. It was the opinion of scientists of the Nuclear Development Corporation of America that the shield weight can be reduced somewhat by proper shaping; this, however, would increase the side and rear dose. Performance of this power plant in proposed aircraft of different gross weights is shown in Table 2. 7. The performances indicated are based on estimated drag curves and are therefore presented only as typical examples. Total sea-level static thrust with full reheat is 90, 800 pounds for the two engines with a specific fuel consumption of 1. 62 pounds per hour per pound.

2. 3. 4 TURBOJET ENGINE

The study engine, the X310, is a turbojet engine composed of the normal engine components plus a liquid-metal-to-air heat exchanger which is placed between the compressor outlet and the combustion section. The engine was designed to operate on heat from the liquid metal in the radiator, from chemical fuel burned in the combustors (in this design called the interburner), or on combinations of both. The design power level is 60 megawatts of power from the liquid-metal system at 30, 000 feet and Mach 0. 9. The X310 also has an afterburner that is used for thrust augmentation on take-off.

The engine will operate on nuclear power during cruise conditions and use chemical augmentation in the interburner and afterburner when an increase in thrust is desired.

The three levels of operation - nuclear, dry, and reheat - are defined as follows:

1. Nuclear operation is defined as operation with heat transfer to the air in the heat exchanger only.

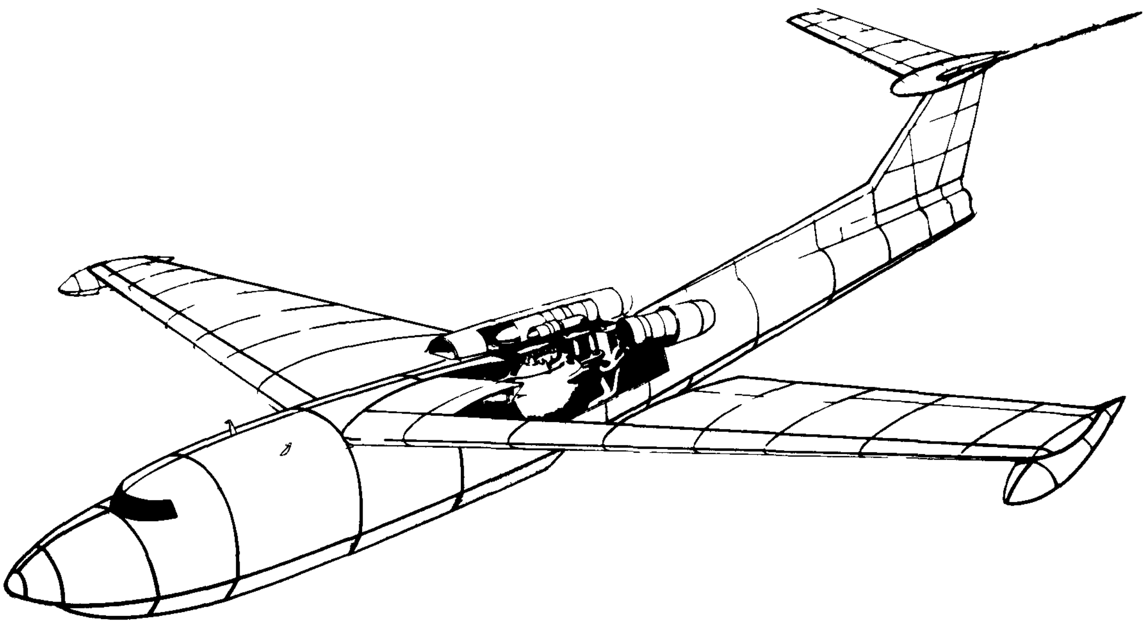

Fig. 2.7 – Power plant installed in a seaplane

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TABLE 2.7

MAXIMUM FLIGHT PERFORMANCE

Type of Operation	Altitude	Speed, Mach No.	Specific Fuel Consumption, lb/hr-lb
215,000-pound-gross seaplane			
Nuclear	Sea level	0.55	
	25,000 ft	0.88	
	30,000 ft	0.88	
Nuclear plus chemical interburning	Sea level	0.92	0.976
250,000-pound-gross seaplane			
Nuclear	Sea level	0.47	
	25,000 ft	0.82	
	30,000 ft	0.82	
Nuclear plus chemical interburning	Sea level	0.88	0.955

2. Dry operation combines nuclear operation with chemical fuel combustion in the interburner.
3. Reheat operation implies chemical combustion in the afterburner in conjunction with nuclear and interburner power delivery.

Cycle optimization was accomplished by the development of an IBM design-point cycle-study program. Flexibility in the program was necessary in order to permit variations in the cycle parameters such as efficiencies and pressure drops. Engine performance was calculated and plotted against compressor pressure ratio on a per-pound-of-airflow basis. Calculations of this type were performed for operation at 30,000 feet and Mach 0.9, at sea level and Mach 0.9, and at SLS. Curves of net thrust, airflow, and specific chemical fuel consumption against cycle pressure ratio were then developed for performance at 30,000 feet and Mach 0.9. A state-of-the-art compressor flow-speed characteristic was used to relate the airflow at Mach 0.9 and 30,000 feet with that at other flight conditions. In addition, the pressure drop from compressor discharge to turbine inlet was varied between 10 and 30 percent.

The third study phase involved an optimization to determine the best operating compressor pressure ratio for this installation. The most important parameters selected for optimization were: net thrust per pound of airflow, net thrust per pound of engine weight, and net thrust per unit of frontal area. Investigations of two- and four-engine configurations were made for this optimization. The curves of thrust per pound of engine weight indicate better results with operation at low cycle pressure ratios, whereas curves of thrust per unit of frontal area indicate better results with engine operation at higher pressure ratios. For two reasons, the thrust per pound of engine weight and thrust per pound of airflow are parameters of greater importance for this application than the thrust per unit of frontal area: (1) the engines are mounted on the aircraft wings or fuselage in a partially submerged position, and (2) they are mounted with the jet nozzle canted downward and with an air scoop above the wing surface. Therefore, considerably less engine area is exposed in comparison with a normal nacelle installation.

The optimization studies indicated that an 8-to-1 compressor pressure ratio at sea level static is best suited for this type of engine; this is the pressure ratio selected. Also, a twin-engine configuration was selected for this application.

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The heat exchanger liquid-metal-inlet temperature is constant at 1500°F; the air discharge temperature is at a maximum of 1400°F at the design point and varies with flight conditions. The over-all pressure drop from compressor discharge to turbine inlet is 12-1/2 percent at the design point; 5 percent of this occurs in the heat exchanger.

The maximum turbine inlet temperature is 1800°F for military power with chemical interburning. For maximum continuous operation, the maximum temperature is 1650°F.

The X310 compressor is a front-variable-stator, 11-stage compressor. The pressure ratio at the design point (Mach 0.9, 30,000 feet) is 8.6 to 1. The tip diameter, which is constant throughout the axial length, is 34.25 inches. The variable stators are used through the first four stages. The design rotational speed (100 percent rpm) is 4675 rpm with a tip speed of 1110 feet per second. The sea level static airflow is 456 pounds per second.

The interburner, used when thrust increases are desired, is of the triple-annular type, 25 inches long. This design was selected to reduce weight and length for this component.

The two-stage turbine incorporates a conical-flow-path design. It was designed to withstand a maximum inlet temperature of 1800°F for short periods and 1650°F for extended periods of time.

A conventional-type afterburner is used in the X310 engine. The afterburner was designed for reheat operation at take-off only. However, only slight modification (with slight weight increase) is required for reheat operation at other flight conditions.

The X310 jet nozzle is of the converging type; area variation is achieved by a hydraulically operated shroud which causes nozzle-finger movement.

The control schedule for the X310 engine was established to provide a single control for the pilot. However, a switch is provided for the flight engineer to convert from nuclear operation alone to nuclear plus chemical or all-chemical operation. Control schedules for each of these operating conditions were determined.

A brief summary of engine performance at various points of interest is presented in Table 2.8. It should be noted that the operation of the engine at sea level on nuclear power is limited to less than 100 percent rpm. Maximum thrust at the slower speeds is obtained at 90 percent rpm, because the air entering the heat exchanger is at a density, and therefore a mass flow, considerably higher than at the altitude design point. In this type of power plant, the heat output of the nuclear system is limited by the liquid-metal temperatures and flows, and therefore by the power established for design-point operation. The power plant was designed for 120 megawatts, based on the requirements at 30,000 feet, Mach 0.9. At sea level operation, therefore, each heat exchanger can still supply only 60 megawatts to the air, whereas a much higher power input is required to maintain a reasonable turbine-inlet temperature. At 100 percent rpm at sea level, the turbine-inlet temperature becomes too low to maintain engine operation. Reducing the airflow raises the turbine inlet temperature so that maximum thrust at sea level up to Mach 0.8 is obtained at 90 percent rpm. At Mach 0.9 it is necessary to drop back to 85 percent rpm.

Three versions of the engine were studied. The first uses a plate-and-fin-type radiator and a triple-annular interburner. The weight of this engine without the radiator is 11,330 pounds. With the radiator, the weight is 13,700 pounds. The second version uses a helical-fin - round-tube radiator with a triple-annular interburner, and weighs 12,750 pounds without and 16,080 pounds with the radiator. The third version has the helical-fin - round-tube radiator with a single-annular interburner, and weighs 15,100 pounds without and 18,430 pounds with the radiator.

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[redacted]

TABLE 2.8

X310 ENGINE PERFORMANCE SUMMARY

Operation	Altitude	Speed, Mach No.	Power Setting	Engine rpm, %	Net Thrust, lb	Specific Fuel Consumption, lb/hr-lb
Nuclear	Sea level	0.4	Cruise	90	8,400	
		0.6	Cruise	90	6,950	
Nuclear plus chemical		0	Max reheat	100	45,400	1.62
		0.8	Military	100	27,400	0.916
		0.8	Max continuous	100	24,500	0.845
		0.9	Military	100	27,900	0.965
		0.9	Max continuous	100	24,800	0.895
Nuclear	25,000 ft	0.8	Cruise	100	9,460	
		0.9	Cruise	100	9,080	
Nuclear plus chemical		0.9	Military	100	15,500	0.525
		0.9	Max continuous	100	14,000	0.423
Nuclear	30,000 ft	0.8	Cruise	100	9,200	
		0.9	Cruise	100	9,360	
Nuclear plus chemical		0.9	Military	100	13,200	0.421
		0.9	Max continuous	100	12,000	0.291

2.3.5 INSTALLATION

After determination of the component sizes and system assembly, an installation study was made. For this, a layout was made of a complete power plant installation in a typical seaplane. The power-plant arrangement is shown in Figure 2.8 and its installation in Figure 2.9. The reactor-shield assembly is forward of and below the main engines, placing it well within the fuselage and permitting the location of all components and plumbing within the fuselage, except for the main engines and the auxiliary power units. The NaK pumps are located directly aft of the reactor-shield assembly and are grouped below the auxiliary power unit. This permits location of the drive gear box for the NaK pump directly at the take-off shaft from the auxiliary power unit. The drive shafts from the four sodium pumps within the shield pressure shell extend vertically above this assembly into a gear box similar to that provided for the NaK pumps. The drive gear box for the sodium pump is driven by shafts extending directly from the NaK pump gear box.

Possible locations of accessory equipment, such as air conditioning, boundary-layer control, and electric and hydraulic packs, are indicated in Figure 2.9.

Except for the engines, the whole system falls within an envelope of 190 by 150 by 110 inches. The engine with the wavy fin-plate fin radiator is 230 inches long.

The assembly is designed as a power package to be installed and removed from the aircraft as unit. It was not contemplated that any disassembly of the power plant would be accomplished while installed in the aircraft. Because of the necessity for breaking liquid-metal lines to disassemble components, liquid metal must first be drained from the system. This may be more easily and safely accomplished with the power plant outside of the airframe. In addition, it is possible to use proper remote handling equipment, and parts are more readily accessible in a hot shop.

The installation was designed to maintain the aircraft center of gravity at the one-fourth mean aerodynamic chord. The engine centerlines are canted down by 30 degrees and out by 2 degrees to provide a clear exhaust path.

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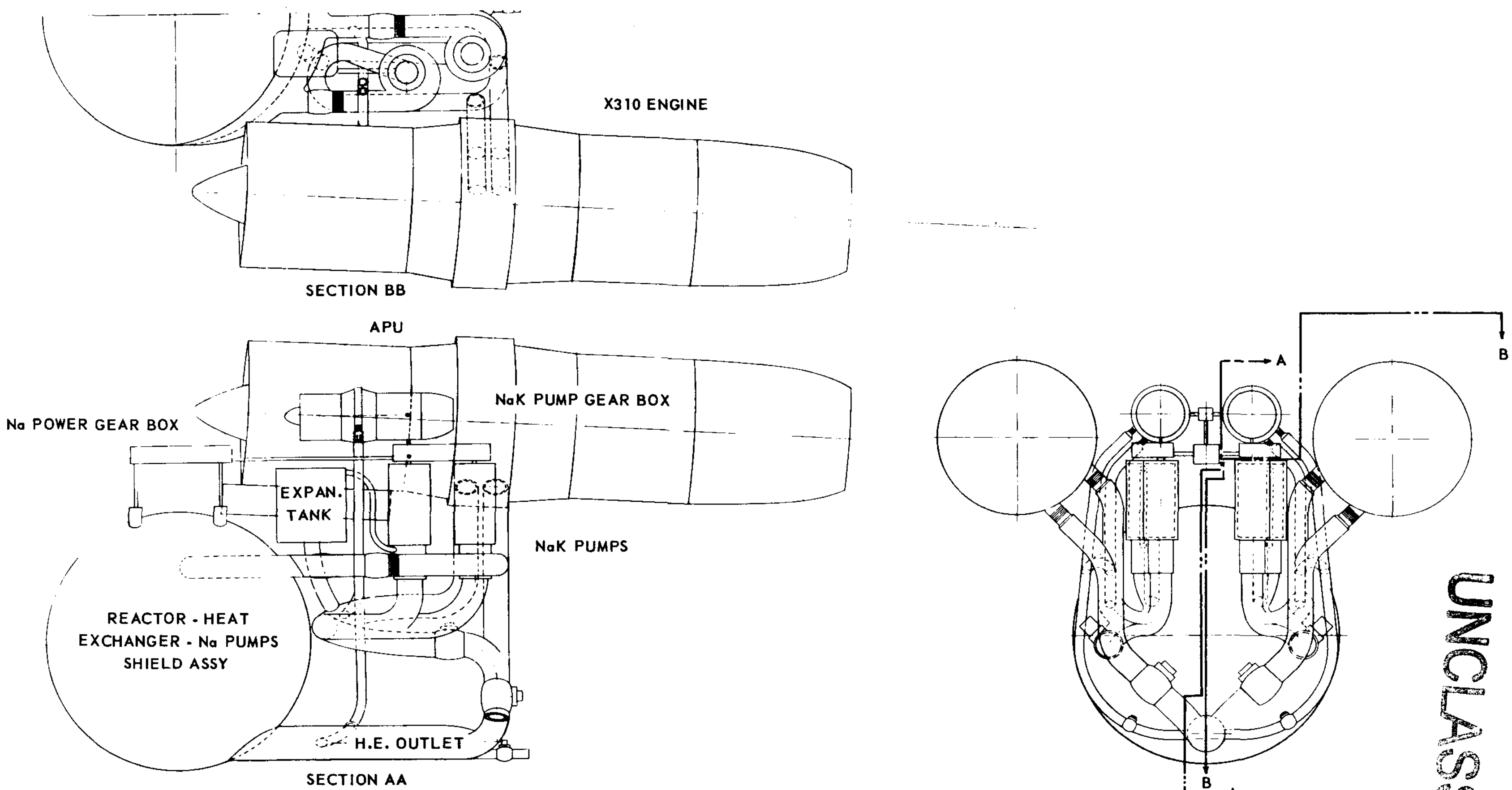

Fig. 2.8 – CCR power-plant arrangement

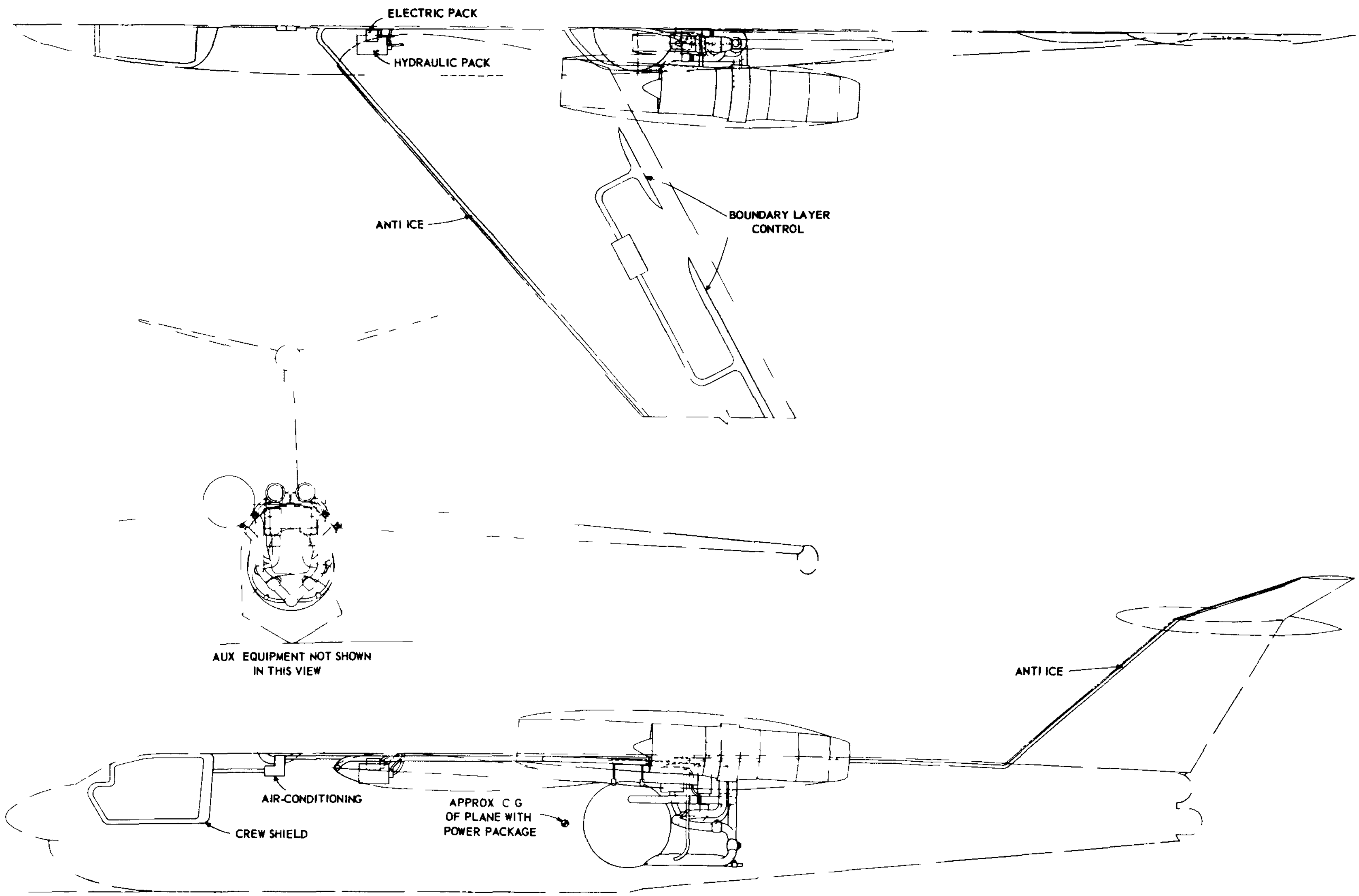

Fig. 2.9 – CCR power-plant installation

2.3.6 INSTALLATION OF OTHER POWER PLANTS

2.3.6.1 AC-110* Power Plant

A modified AC-110-type reactor-shield assembly coupled to two X211 engines, as shown in Figure 2.10, was also studied for installation in the same aircraft.[6] The airframe accessories were assumed to be identical to those in the CCR installation. Other weight assumptions were: (1) reactor-shield assembly, 52,000 pounds; (2) engines, 13,100 pounds each; and (3) crew shield, 35,000 pounds.

The power plant installed in the P6M airframe is shown in Figure 2.11. The modifications of the airframe contours are generally the same as those described in the CCR installation except that the large diameter of the reactor-shield assembly requires a "bulging" of the fuselage aft of the rear wing spar. This alters the hydrodynamic characteristics of the airplane and was considered extremely undesirable.

2.3.6.2 AC-107 Power Plant

Two AC-107-type power plants were also examined for installation in the P6M airplane.[6] An artist's concept of a possible arrangement is shown in Figure 2.12. The weight of an AC-107 for this study was assumed to be 39,500 pounds, with a crew-shield weight of 35,000 pounds.

2.3.7 CONCLUSIONS

As a result of this study, the following conclusions were reached.

The CCR power plant described in this study, with successful development and proper design, could perform the Navy mission. That is, it could power a seaplane on nuclear power at low subsonic speeds at sea level, and at more than 440 knots at 25,000 feet. The required cruise range can easily be met on nuclear power. With proper aircraft design, the sea-level sprint at Mach 0.9, which must be performed with chemical augmentation, can also be met.

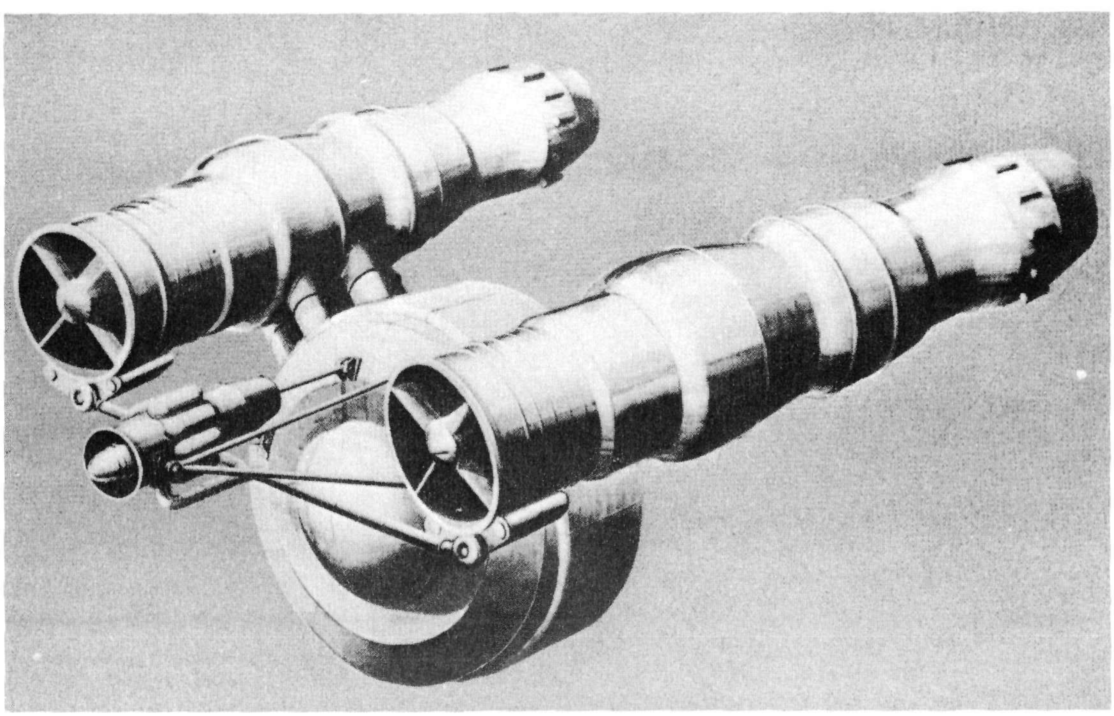

Fig. 2.10 – Modified AC-110 power plant

*Early version of XMA-1 power plant described in APEX-907, "XMA-1 Nuclear Turbojet," of this Summary Report.

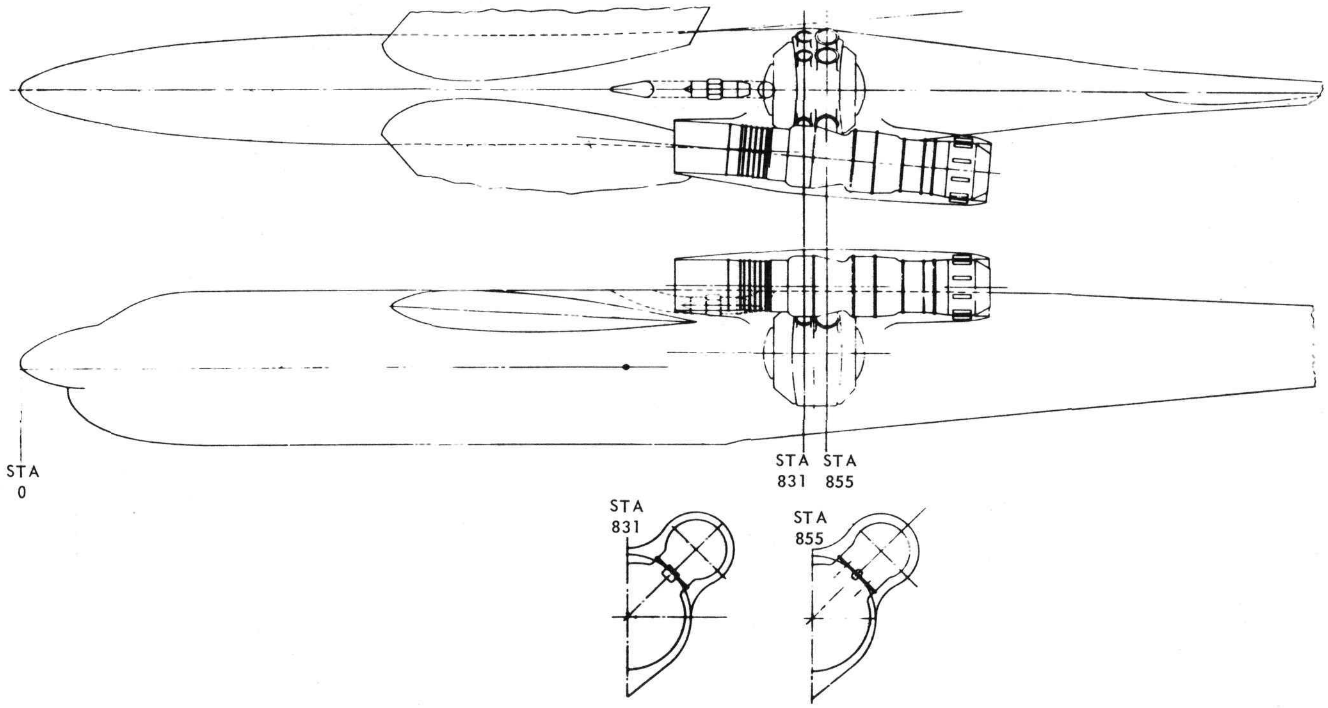

Fig. 2.11 – Modified AC-110 power-plant installation

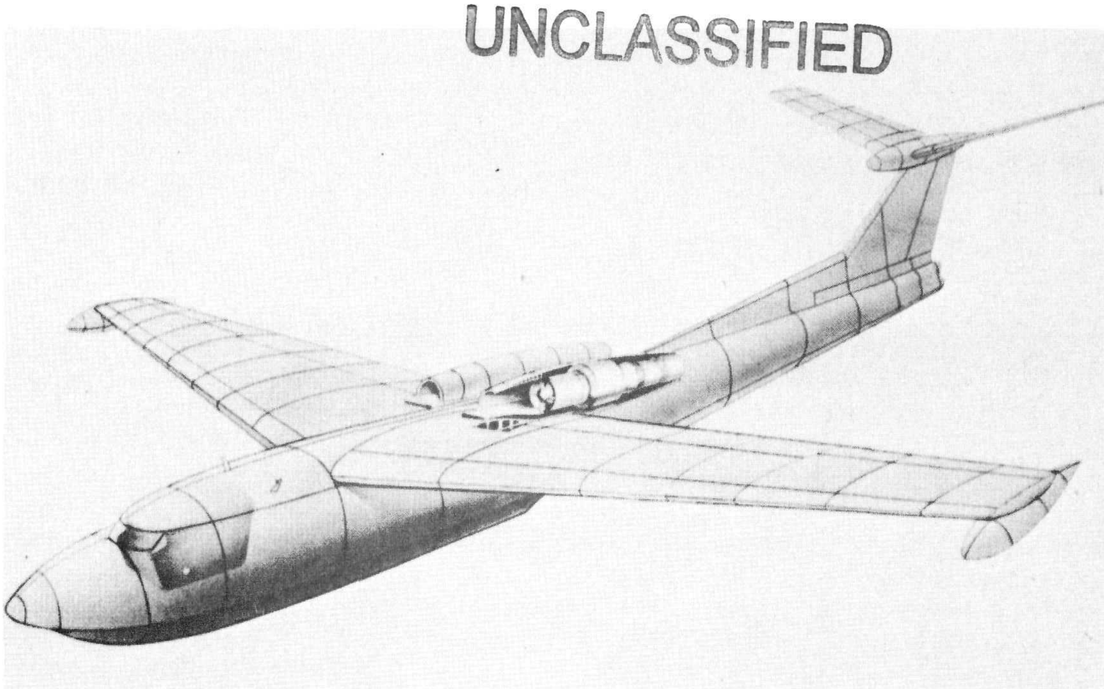

Fig. 2.12 – AC-107 installation

The over-all weight of this power plant, which has a maximum power output of 120 megawatts, is about 106,000 pounds without any provision for chemical fuel for the auxiliary power unit. Adding a crew shield, which would weigh approximately 22,000 pounds for a dose of 0.1 rem per hour or 25,000 pounds for 0.03 rem per hour, results in a power-plant plus crew-shield weight of 128,000 to 131,000 pounds. This power plant is too heavy for an aircraft with a gross weight of 215,000 pounds. It would require an aircraft gross weight in the 250,000-pound range.

Performance curves of study aircraft in the 250,000-pound-gross class indicate a maximum sea level sprint with nuclear plus chemical interburning of slightly less than Mach 0.9. It was felt, however, that with proper aircraft design, the Mach 0.9 speed could be achieved.

The development difficulties of this type of system are those that are inherent in its basic concept: i.e., the use of liquid metal as a coolant. Excluding the problems of the reactor itself, the major development items are the engine radiator and the liquid-metal pumps and system. The radiator type that promises optimum liquid-metal-to-air heat-transfer performance is in very preliminary stages of development. Even those units that provide poor performance require considerable development. No liquid-metal radiators for aircraft have yet been built and tested. Only very small models have been built at all, and very few of these have been tested with liquid metal. Even fewer have proved successful in the model test.

Liquid-metal centrifugal pumps for ground-test-loop operation have been built in various sizes and operated successfully for relatively short periods of time. No pumps of an aircraft type have been built and tested to full temperature with liquid metal, and none of the size required for this unit have yet been designed.

Commercial-test-loop type valves are available, but they must be completely redesigned for aircraft use; however, this is not considered a major problem.

Although much has been done to eliminate mass-transfer problems in liquid-metal circuits, only small laboratory loops have been operated successfully for reasonable periods of time. The problems in fabricating, joining, completely sealing, and cleaning a full aircraft liquid-metal system in order to assure the coolant purity required for elimination of mass transfer, are many orders of magnitude greater than those met in a laboratory loop. This transition from laboratory to operational systems has yet to be accomplished.

2.4 "PRINCESS" FLYING BOAT

2.4.1 BASIS OF THE STUDY

In 1958, the U.S. Navy sponsored a study of the application of nuclear propulsion to the Saunders-Roe "Princess" flying boat. The companies participating in the study were Saunders-Roe, the Martin Company, and Convair - San Diego for the aircraft, and General Electric and Pratt and Whitney for the nuclear power plants. The purposes of this program [5] were (1) to develop an early, simplified aircraft system to demonstrate the feasibility of a nuclear-powered aircraft; and (2) to produce, by successive, well-planned stages of development, operational aircraft to fulfill the Navy's assigned missions. The following criteria were to be used in making engineering decisions, listed in the order of their importance:

1. Certainty of success; i.e., over-design where necessary and possible.
2. Simplicity and reliability.
3. Minimum cost, including time, manpower, and facilities.
4. Mission potential.
5. Performance and efficiency.

The study was based on existing GE-ANP technology and employed, wherever practical, existing components and subsystems. Emphasis was placed on problems unique to the application of nuclear power to aircraft rather than associated technical problems. This approach led to the study of a low-powered, quasi-unit-shielded reactor supplying cruise power to a relatively low-performance seaplane fitted with turboprop engines.

2.4.2 FLIGHT TEST OPERATIONS

The plan for flight test operations was made to minimize the requirements upon the nuclear system. A seaplane was selected in order to increase the safety of test operations and reduce take-off and base facility problems. All water operations, take-off, climb, maneuvering, and landing are performed on chemical fuel. The nuclear system is used only for the nominal cruise condition, thus reducing the reactor's power requirements. The flight operation concept includes:

1. Take-off, climb, and cruise (up to one hour on chemical fuel alone).
2. Cruise, on nuclear heat only, at 10,000 feet and 220 knots on a standard day, with the chemical engines feathered.
3. Cruise at 10,000 feet and 220 knots on chemical engines, with the nuclear engines feathered.
4. After nuclear operation, cruise up to one hour, descend, and land on chemical engines.

2.4.3 DESCRIPTION OF THE AIRCRAFT

The aircraft selected as most suitable for this program was the Saunders-Roe Princess, Figure 2.13. Several of these flying boats had been built in England for trans-Atlantic passenger service but then had been moth-balled. It was planned to refit the aircraft with four T-34 chemical turboprop power plants in addition to the nuclear propulsion system.

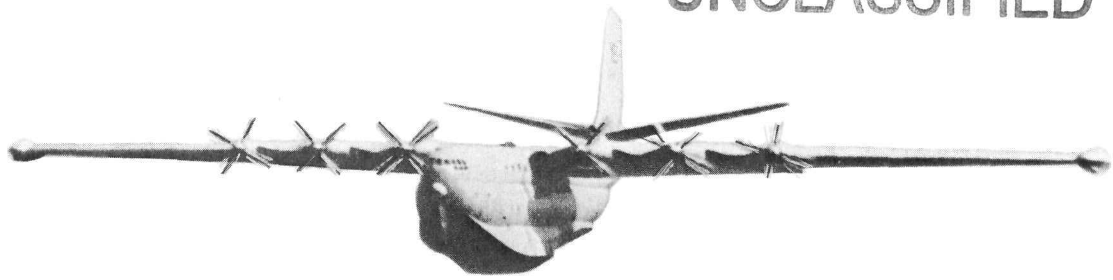

Fig. 2.13 – Saunders-Roe "Princess"

The basic Princess aircraft weight, including structure, structural modifications, fixed equipment, and chemical engine mounts and cowls, was targeted at 115,000 pounds. For take-off and landing, the maximum weight was set at 345,000 pounds, leaving the difference for the nuclear system and the chemical power plants and fuel.

2.4.4 DESCRIPTION OF POWER PLANTS

Two direct air cycle nuclear power-plant systems were considered in the application studies performed by the Martin Company and Convair - San Diego. Both systems employed the same basic reactor-shield assembly proposed by General Electric. The system, suggested by the Navy, utilizes T-57 nuclear engines. The other system, recommended by General Electric and designated the P302A, uses a derated X211 gas generator and bleed turbines to drive the propellers. The latter system was proposed because of the installation and control advantages of a bleed system and the growth potential of the X211 gas generator. Details of the design and performance of the power plants are covered in APEX-909, "Aircraft Nuclear Propulsion Systems Studies" and in reference 7.

2.4.5 CONCLUSIONS

2.4.5.1 Martin Company Studies

The results of the Martin Company's application studies of both direct air cycle power plants are reported in references 8 and 9. No fuel is placed around the reactor for shielding, but 6700 pounds of fuel is placed between the reactor and crew, forming a shadow shield that eliminates most of the direct radiation and results in a crew dose rate of 0.25 rem per hour. By eliminating the liquid-shield cooling system, on the assumption that it would not be required if the reactor were operated for five hours or less, the weight is reduced by approximately 1500 pounds.

Four T-34 chemical turboprop engines, with propellers 15 feet 2 inches in diameter, are used with both nuclear power plants. Twenty-foot-diameter propellers are used on the nuclear T-57 engines and the bleed turbines of the P302A system. The T-57 installation is shown in Figure 2.14 and the P302A installation in Figure 2.15. The group weight statement for each of the two studies is shown in Table 2.9. It should be noted that the

[redacted]

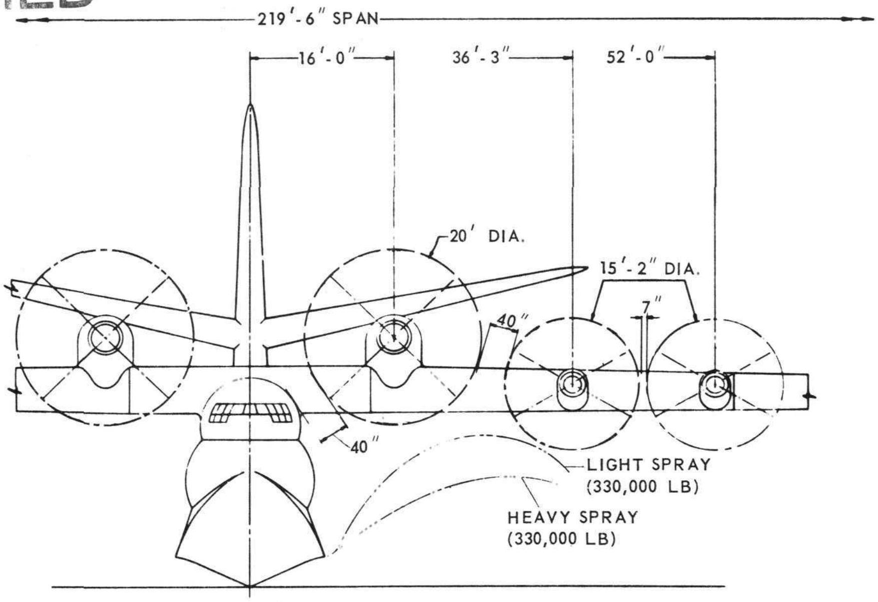

Fig. 2.14 – T-57 system configuration

nuclear turbomachinery weight used by the Martin Company is 3000 pounds greater than that quoted by the General Electric Company in reference 10.

A comparison of the performance of the two systems, both operating with a 1400°F turbine-inlet temperature and 30 percent pressure drop from compressor to turbine, is shown in Table 2.10.

If the nuclear turbomachinery weight recommended by General Electric had been used in the study, the performance of the two systems would be more nearly comparable.

2.4.5.2 Convair - San Diego Studies

Convair - San Diego proposed to use two J-75 turbojet engines as the chemical power plants on the Princess aircraft, and to make use of the chemical fuel for reactor shielding. However, Convair concentrated on the T-57 configuration of the nuclear propulsion system and only made general comments about the P302A bleed turbine configuration. An artist's concept of the Princess modification proposed by Convair is shown in Figure 2.16.

The standard-day nuclear-cruise performance for this aircraft is 224 knots at 10,000 feet with a cruise weight of 309,310 pounds.[11] This performance is based on 1400°F turbine-inlet temperature and a 30 percent pressure drop from compressor to turbine. Therefore, through saving weight by using chemical turbojet engines, Convair was able to show better performance than the 210 knots at 10,000 feet specified by the Navy.

The Convair report states that using 27,625 pounds of fuel as reactor shielding, for a total reactor-shield-assembly weight of 140,740 pounds, reduces the crew dose rate to 0.1 rem per hour. However, General Electric data in reference 7 shows a dose rate of about 0.25 rem per hour for that weight.

The group weight statement for this version of the Princess is given in Table 2.11. Supplementary data on this study are presented in references 12 through 19.

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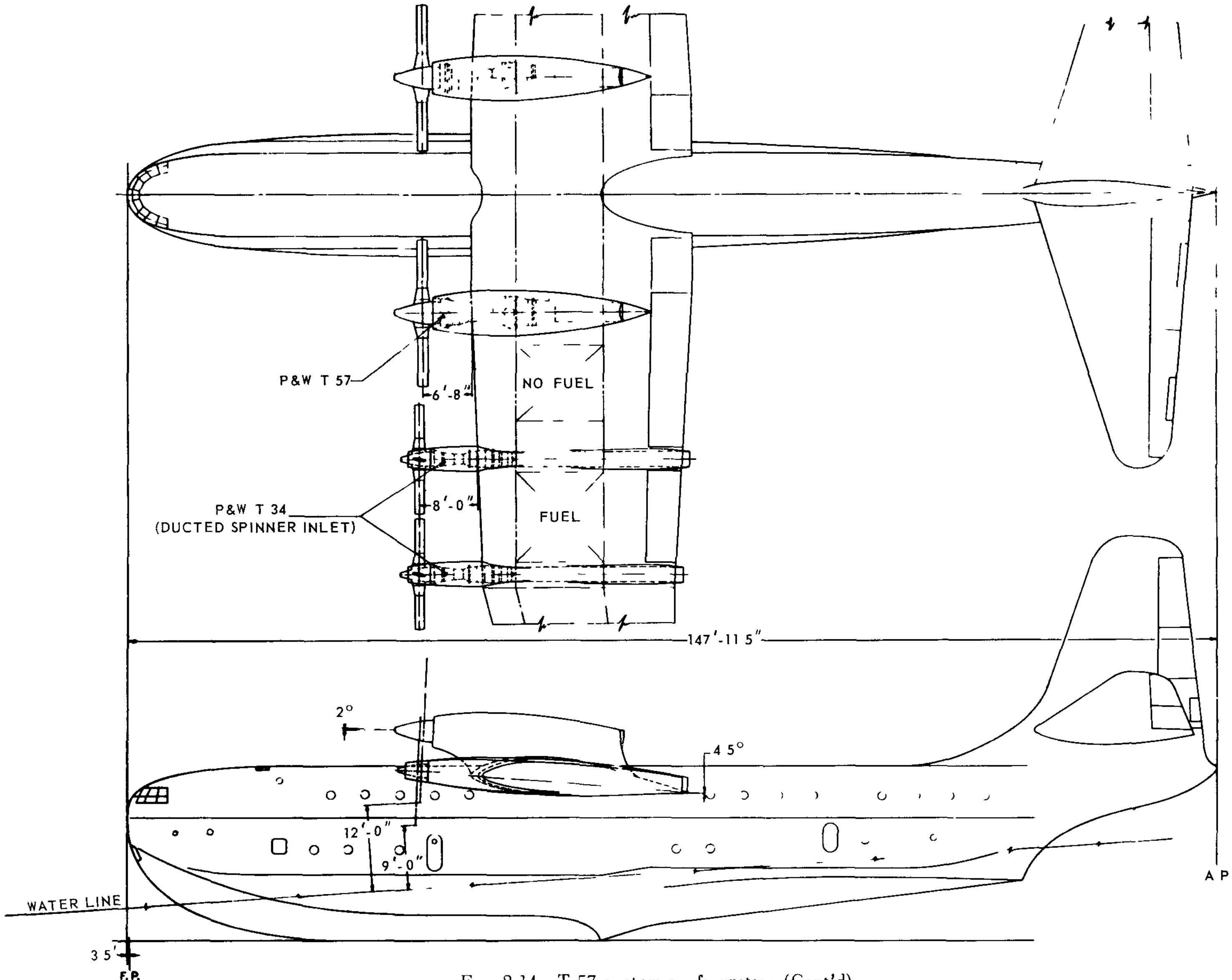

Fig. 2.14 – T-57 system configuration (Cont'd)

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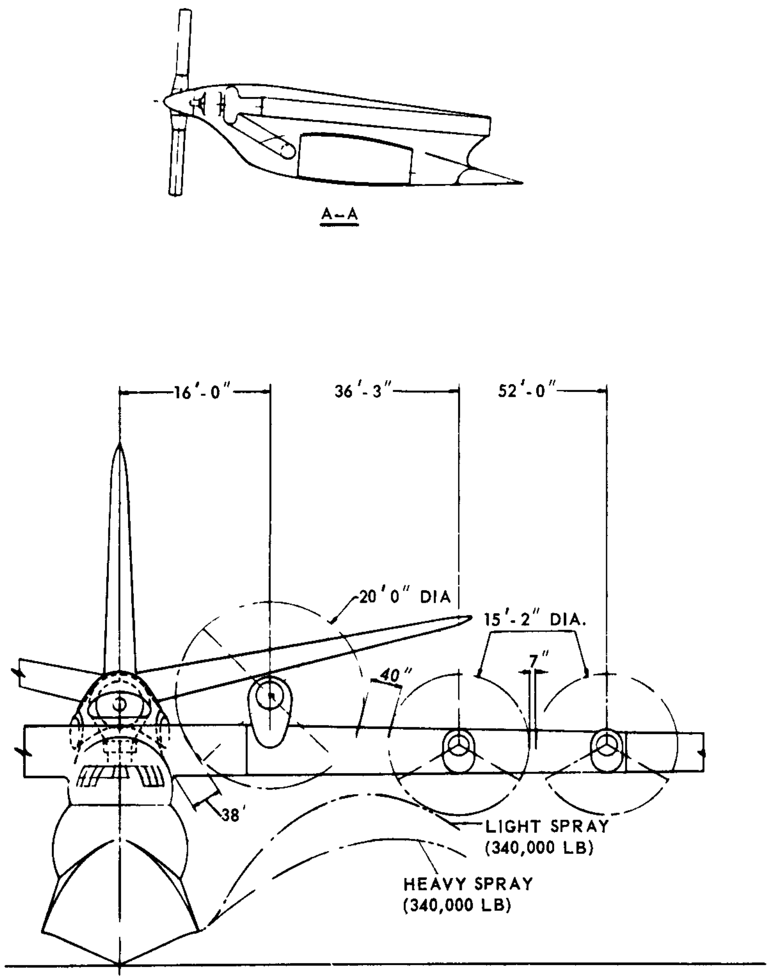

Fig. 2.15 – P302A configuration

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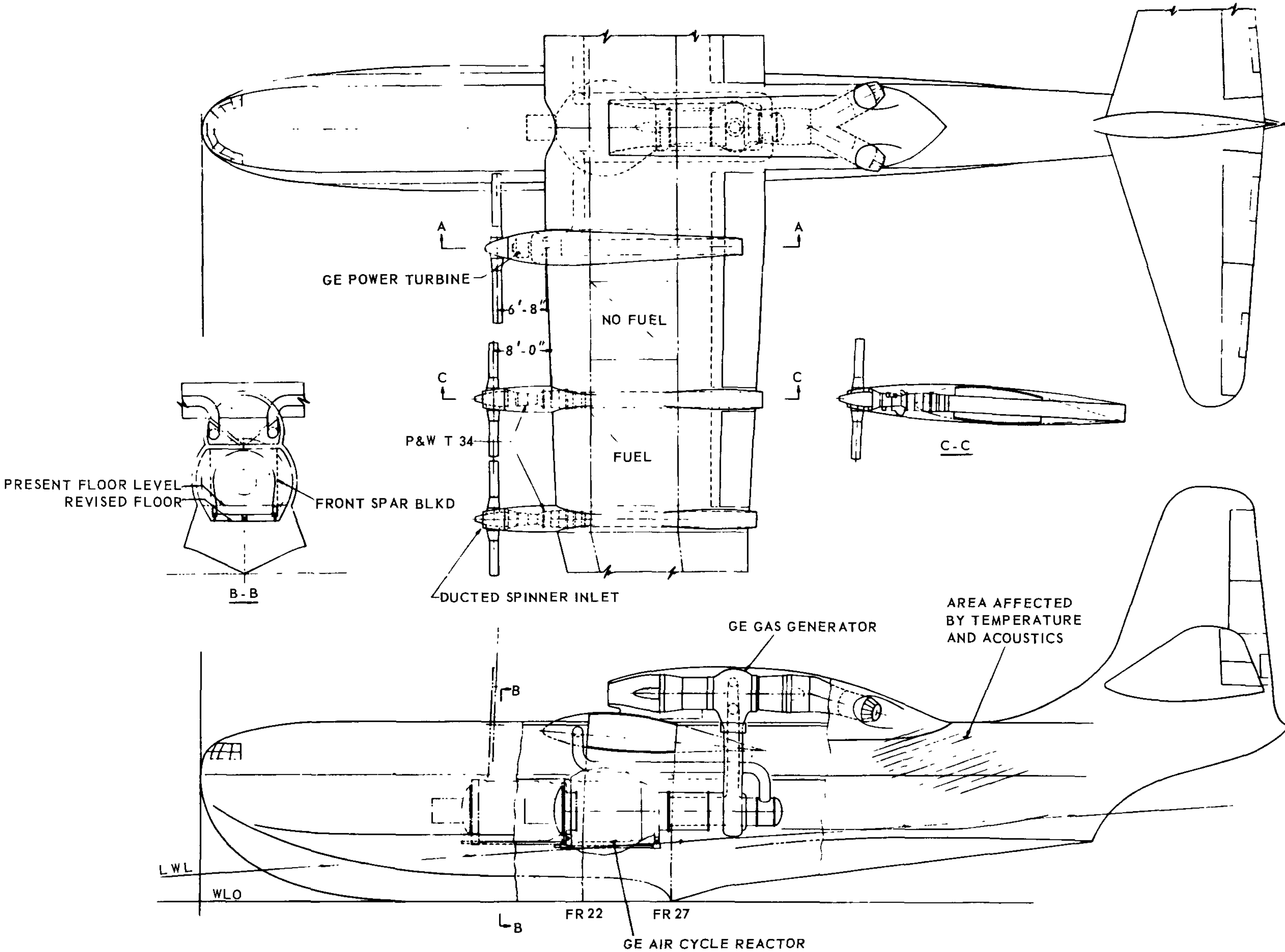

Fig. 2.15 – P302A configuration (Cont'd)

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TABLE 2.9

MARTIN COMPANY STUDY: GROUP WEIGHT STATEMENT OF AIR CYCLE CONFIGURATIONS

Item	Weight, lb	
	T-57 System	P302A System
Structure	91,800	91,800
Propulsion system		
T-34 engines, nacelles, and propellers	22,335	22,335
Nuclear turbomachinery and propellers	21,880	29,820
Nacelles and mounts	3,590	4,945
Reactor installation including ducts	137,000	137,960
Fuel system	2,725	2,725
Power system	4,910	5,170
Electronics	810	810
Furnishings, equipment, and crew	15,414	15,414
Empty weight	300,464	310,979
Fuel	31,600	31,600
Water	960	960
Gross weight	333,024	343,539

TABLE 2.10

COMPARATIVE SYSTEM PERFORMANCE

	T-57 System	P302A System
Probable nuclear cruise weight, lb	315,000	325,000
True air speed, kn	208	207
Service ceiling, ft (100 fpm R/C)	8,600	5,600

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Fig. 2.16 – Convair – San Diego modification of the Princess aircraft

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TABLE 2.11

CONVAIR STUDY: GROUP WEIGHT STATEMENT FOR TWO T-57's AND TWO J-75's

Item	Weight, lb
Structure	88,750
J-75 engines and nacelles	14,390
T-57 engines and propellers	22,290
T-57 nacelles and mounts	4,000
Reactor installation, including ducts	125,215
Fuel system	1,950
Power system	7,625
Furnishings, equipment, and crew	15,270
Empty weight	279,490
Fuel - hull shield	27,625
wing, A. P. U., and residual	31,570
Gross weight	338,685

2.5 B-52 AIRPLANE STUDIES

2.5.1 INTRODUCTION

A flying test bed is of vital importance in the early stages of the development of an aircraft engine, since it is the only means by which the engine can be exposed to its operational environment well in advance of its use as an actual propulsion device. This was particularly true in the case of nuclear propulsion which involved entirely new technologies in several fields.

Accordingly, a flight-test-bed program was proposed[20] for the XNJ140E-1* nuclear engine, providing for a B-52G aircraft modified to carry the engine in an external nacelle, side-mounted on the aft fuselage as in the Sabreliner and Caravelle configurations. The intent of the program was to provide the means of operating the nuclear engine throughout the entire range of altitudes and speeds expected of the Convair NX-2 airplane in the systems-evaluation phase of the work.

The first phase of the proposed flight program was to be conducted with a chemically-powered version of the power plant. Following the chemical-operation phase, a single nuclear power plant was to be installed, and nuclear flight testing initiated. Installation of a second XNJ140E-1 power plant on the other side of the fuselage, to create a twin-pod configuration, was assessed and the aircraft stressed for this condition. A performance calculation based on a configuration of two XNJ140E-1 nuclear power plants and eight J57 chemical engines showed that the modified B-52 aircraft is capable of demonstrating all-nuclear flight.

An artist's sketch of the proposed aircraft is shown in Figure 2.17. Details of the study may be found in references 20 through 24.

2.5.2 TEST-BED AIRCRAFT EVALUATION

Aircraft potentially suitable as flying test beds for a single-engine XNJ140E-1 installation were screened and evaluated at the General Electric Flight Test Center at Edwards

*Described in APEX-908, "XNJ140E Nuclear Turbojet," of this Summary Report.

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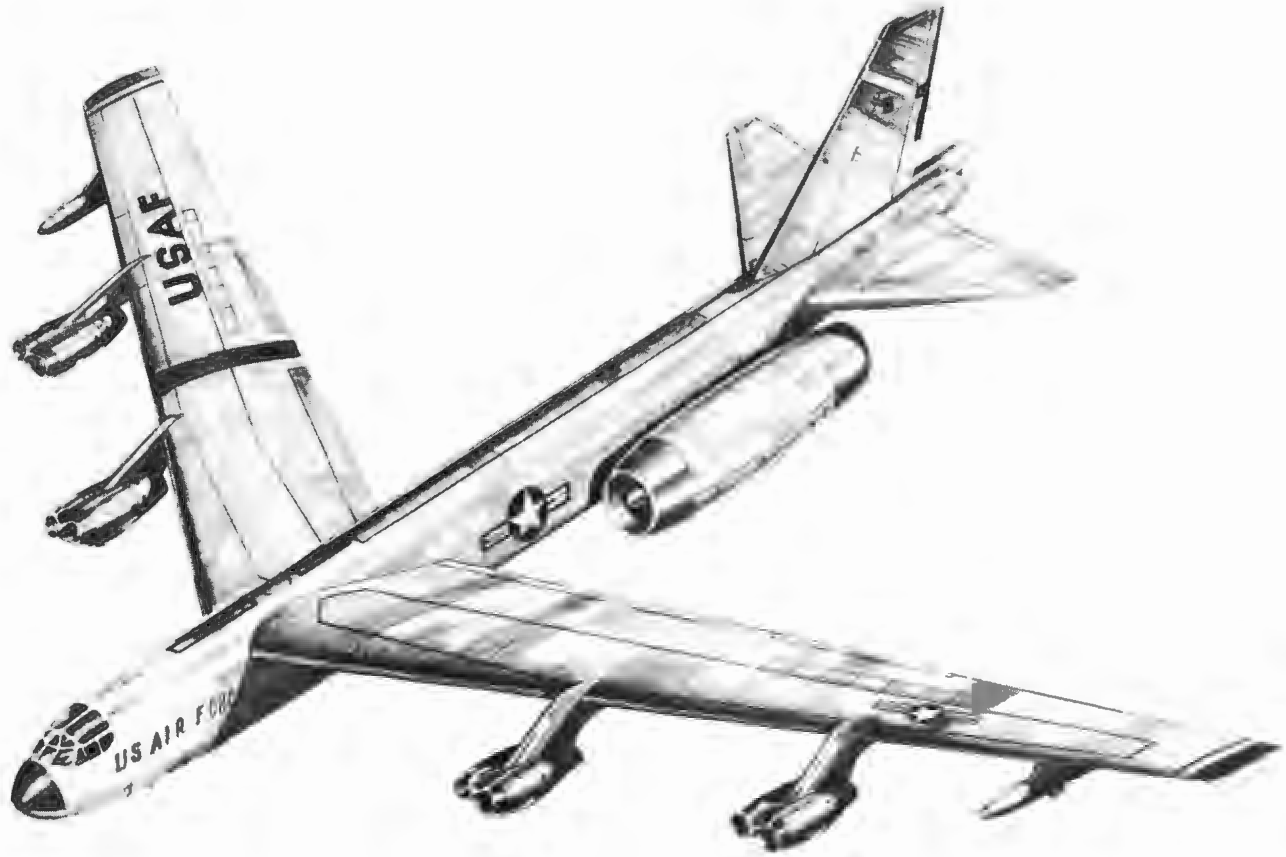

Fig. 2.17 -- B-52 flying test bed

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Air Force Base. The B-52G aircraft was selected because it meets the load-carrying and space requirements for testing the XNJ140E-1 power plant, and its performance characteristics permit testing under the expected NX-2 flight conditions. The following criteria were used in the evaluation:

1. Performance with the XNJ140E-1 power plant installed and operating to be no less than Mach 0.8 at 35,000 feet.
2. Endurance capability of 7 hours on chemical fuel only.
3. Weight of the XNJ140E-1 power plant to be 60,000 pounds, and the crew-shield weight 30,000 pounds.

In addition to the B-52G, the following aircraft were considered but were limited in potential as indicated:

C-133A - Internal installation is practical within weight limitations, but the speed is limited to Mach 0.62 and the endurance to 4 to 5 hours.
KC-135 - Internal installation appears feasible, but the endurance is limited.
B-58 - Internal installation of the engine is not possible because of inadequate clearance.

Other large aircraft such as the C-130, B-36, R-3Y, and P6M do not meet the load-carrying or the performance specifications. Large foreign aircraft appear to be limited in the same ways, although these aircraft were not exhaustively surveyed.

2.5.3 CONFIGURATION STUDY

Three configurations seemed feasible for a single-engine installation in the B-52G: the side-mounted nacelle ("Caravelle"), the internally mounted engine, and the top-mounted nacelle. The side-mounted Caravelle configuration was recommended.

2.5.4 NUCLEAR CONSIDERATIONS

The radiation from nuclear power plants in manned aircraft makes mandatory the use of protective shielding for personnel. The criteria for providing flight-crew protection are the dose rate and the integrated dosage received by the crew over a period of time. On the other hand, the radiation criteria for aircraft equipment are based on the required operational lifetimes in the environments of the nuclear aircraft. In addition to the added radiation fields, these environments include specific ranges of temperature and humidity, vibration, and shock. As a result, radiation shielding is generally not provided for specific equipment in the flying test bed. Rather, slight design modifications and changes in location are made so that the equipment will perform in the nuclear environment for the lifetimes specified.

The total time of full-nuclear operation was less than 1000 hours in the proposed three phases of the flight-test program. Nonmetallic component materials, other than Teflon located in the high-radiation field, have a satisfactory life for use in most locations in the B-52G nuclear test bed. The elastomer components affected by the radiation are gaskets and flexible connecting tubes and other components that must have elastic properties. Items such as tires, hydraulic fluid, oil, seals, and hoses are normally replaced after several flights or during power plant removal even during non-nuclear engine-flight-testing operations. Therefore, many of the more vulnerable components located in accessible places can easily be replaced periodically if necessary.

The data-acquisition system and other electronic equipment containing semiconductors (transistors) can be protected from radiation by local shielding or installation within the shielded crew compartment. Since the total amount of integrated neutron radiation is not large for a given flight, many of these sensitive components can be replaced after several flights.

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The oxygen systems must be located in low radiation fields to prevent the accumulation of toxic ozone in the crew air supply.

The parameter proposed by Convair for the evaluation of radiation damage to materials is the threshold of damage, the minimum amount of radiation that will change the properties of a material to the specification limit. The approximate thresholds of some of the more common aircraft-system materials are listed in Table 2.12. In certain regions it may be desirable to replace Teflon with polyethylene, which has a functional threshold about 3000 times greater than Teflon. Polyurethane was also recommended as a replacement for Buna-N.

Except for semiconductors, gamma radiation causes most of the incipient damage to the components. The neutron field causes only a small percentage of the damage. The approximate isodose pattern of a 70-megawatt nuclear system is shown in Figure 2.18. The operating lifetime of a component in this nuclear environment, designated by zones, is calculated by dividing the functional threshold of the material by the radiation field in the respective zones. The isodose patterns do not include air scattering, structural scattering, nor absorption in the airframe structure.

TABLE 2.12

FUNCTIONAL THRESHOLDS OF DAMAGE FOR COMMON AIRCRAFT-SYSTEM MATERIALS

System	Material	Use	Functional Threshold, ergs/gram (carbon)
Hydraulic	MIL-H-5606	Fluid	5×10^9
	Buna-N coated fabric	Hose	4×10^8
	Buna-N	Fluid seal	1×10^{10}
	Teflon	Backup rings	4.3×10^9
Lubrication	Oil		
	MIL-L-7820	Corrosion prevention	4×10^{10}
	Grease		
	MIL-G-3228	General lubricants	9×10^9
	MIL-G-7421	Low-temperature lubricant	9×10^9
	MIL-G-2118	Gear lubricant	9×10^9
Electrical	Polyvinyl chloride	Insulation	1.1×10^{10}
	Nylon	Connector insert and insulation	5×10^9
	Diallyl phthalate	Connector insert	1×10^{10}
	Neoprene	Connector insert	8×10^9
	Teflon	Insulation	3.4×10^6
Fuel	Nylon	Clamps and inserts	2×10^9
	Kel-F	O-rings and seals	2.4×10^9
	Neoprene - Buna-N	Fuel seal	5×10^9
	Sealant (PR-1422)	Integral fuel tank	$3.5 - 9.9 \times 10^9$
Aft landing gear	GR-S + Natural Buna (radiation resistant)	Tires	8.4×10^9

2.5.5 AIRCRAFT MODIFICATION STUDIES

The necessary modifications of the B-52G aircraft in the Caravelle configuration are shown in Figure 2.19.

2.5.6 B-52G FLIGHT TESTS

2.5.6.1 Summary

The primary objectives of the B-52 flight tests were:

1. In-flight operation of the nuclear aircraft engine.
2. Identification of engine and reactor deficiencies and operational problems under actual flight conditions.
3. Correction of engine and reactor deficiencies and solution of operational problems prior to, and separate from, flight testing of the prototype airframe.

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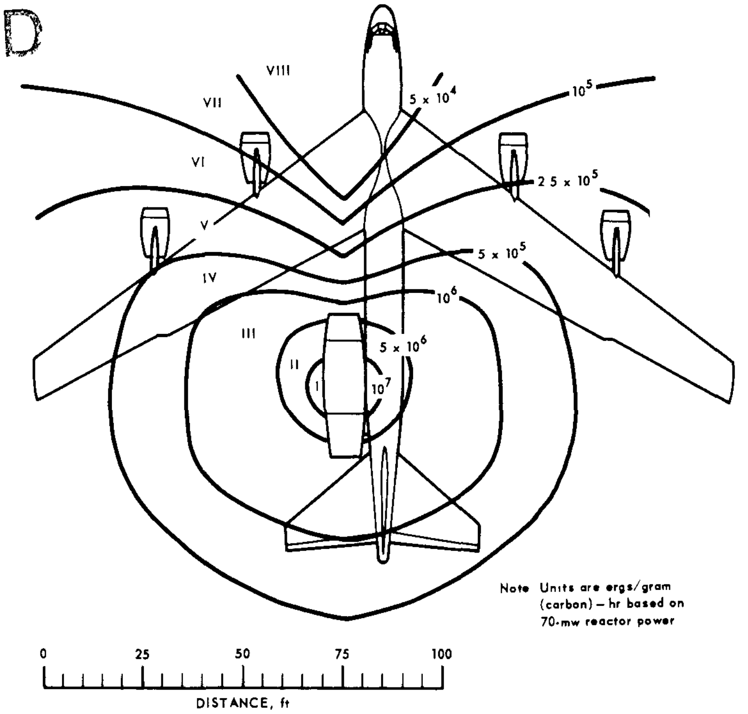

Fig. 2.18 – Approximate isodose pattern around the flying test bed

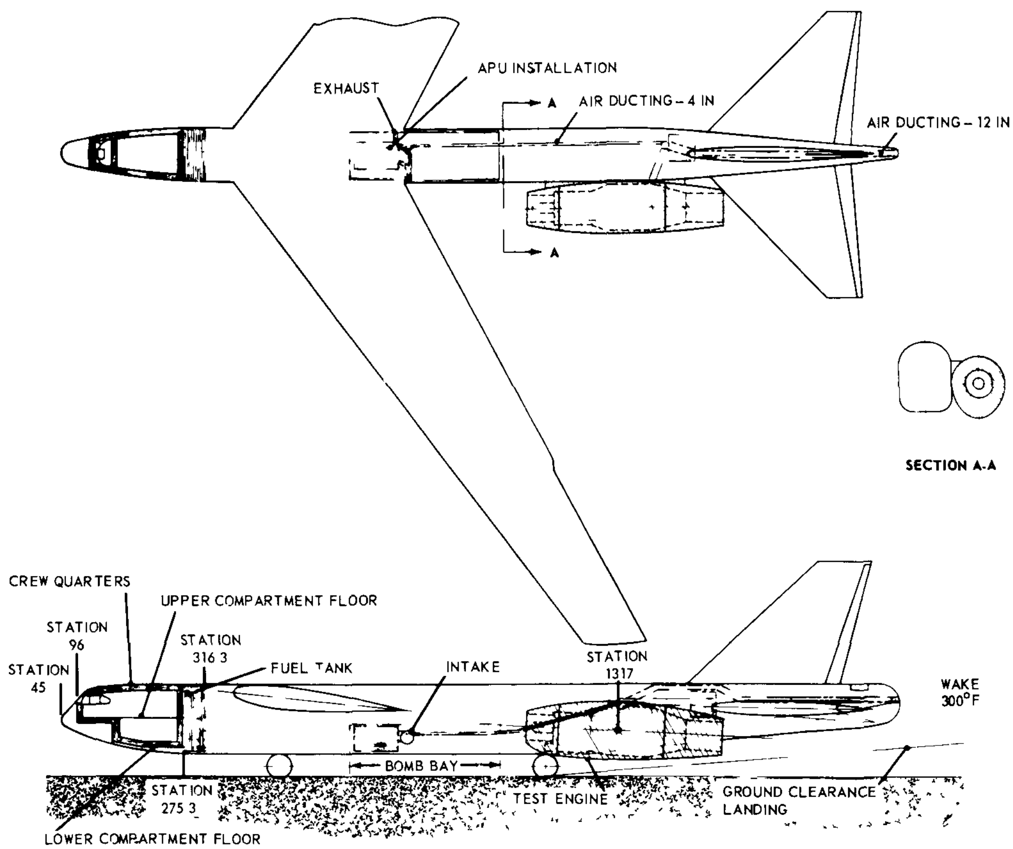

Fig. 2.19 – General arrangement of B-52 aircraft with XNJ140E-1 power plant

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4. Development of equipment, techniques, and procedures for handling and maintenance of a nuclear engine.

The anticipated contributions to the program were:

1. Expedite development of the prototype airframe by the earlier availability of a thoroughly flight-tested propulsion system.
2. Lower cost per flight hour for early in-flight engine development than for the nuclear prototype airframe.
3. Flight testing of engine improvements while independent development of the prototype airframe continues.

2.5.7 AIRCRAFT PERFORMANCE

The adaptability of the B-52G airplane to flight test three different combinations of nuclear XNJ140E-1 and chemical J57 power plants was studied.[23]

2.5.7.1 One XNJ140E-1 and Eight J57 Engines

One of the flight test systems studied consisted of the chemical B-52G aircraft modified to include one XNJ140E-1 nuclear power plant while retaining all eight J57 chemical power plants. The purpose was to test the operation of a nuclear propulsion system at various power conditions while maintaining flight with the chemical engines. The nuclear XNJ140E-1 engine was not to be operative during take-off or landing. The take-off gross weight of this configuration is 450,000 pounds, including 157,000 pounds of chemical fuel. The ground run during take-off is 12,200 feet based on Air Force Hot Day (AFHD) conditions. The flight performance capability is Mach 0.6 at 30,000 feet, with an endurance of 7.0 hours at these flight and altitude conditions.

2.5.7.2 One XNJ140E-1, One Chemical X211, and Eight J57 Engines

The second configuration consisted of the chemical B-52 modified to include one XNJ-140E-1 nuclear power plant and one chemical X211 engine while retaining all eight J57 engines. The purpose of this system was to test and compare the operation of the nuclear engine and its chemical counterpart, the X211 engine, under both take-off and flight conditions. Flight was to be maintained with all eight chemical J57 engines although, during take-off, both the XNJ140E-1 and the X211 were to provide additional thrust. Since additional thrust is obtained during take-off, the allowable take-off gross weight is increased to 480,000 pounds, including 155,000 pounds of chemical fuel. The ground run during take-off is 9040 feet based on AFHD conditions. The flight performance capability is Mach 0.7 at 30,000 feet for 8.0 hours at these flight and altitude conditions.

2.5.7.3 Two XNJ140E-1 and Four J57 Engines

The third configuration studied consisted of two XNJ140E-1 nuclear power plants and four chemical J57 power plants. The purpose of this system was to explore the feasibility of all-nuclear flight. The four J57 engines were retained to provide additional thrust during take-off and to provide an emergency chemical range capability with one nuclear engine inoperative.

All-nuclear flight is possible with this configuration. The only limitation that exists on range is determined by the crew radiation-dose tolerances. The take-off gross weight of this configuration is 400,000 pounds, including 57,000 pounds of fuel. This gross weight is based on a ground run without afterburning from the two nuclear engines. The ground run, at 490,000 pounds take-off gross weight, is 10,400 feet under AFHD conditions. The flight performance capability is Mach 0.64 at 25,000 feet; the endurance is limited only by crew dose. The emergency chemical range with one of the nuclear engines inoperative

[redacted]

and with the four chemical engines operating is 1150 nautical miles at Mach 0. 4 at 5000 feet. A complete weight breakdown for each of the three configurations of the modified B-52G is included in reference 23.

2.5.7.4 High-Speed, High-Altitude B-52G Performance

A study was performed of the high-speed, high-altitude performance capability of the B-52G airplane modified to include either one XNJ140E-1 nuclear engine and eight chemical J57 engines (Case 1), or with two XNJ140E-1 engines and four J57 engines (Case 2).[24] While the take-off gross weights of these configurations are the same as in the previously mentioned studies (450, 000 and 400, 000 pounds respectively), the weights at the beginning of the high-speed, high-altitude cruise are 328, 000 pounds (including 30, 000 pounds of fuel), and 376, 000 pounds (including 30, 000 pounds of fuel), respectively. With these initial weights and with the nuclear power plants operating at normal continuous power and the chemical engines operating at maximum cruise power, a flight capability of Mach 0. 84 at 44, 000 feet can be demonstrated by the B-52G airplane in Case 1 for 0. 8 hours with the consumption of 20, 000 pounds of fuel; similarly, a flight capability of Mach 0. 84 at 37, 800 feet can be demonstrated for 1. 7 hours by the B-52G in Case 2.

In addition to the capability for high-speed, high-altitude testing at the initial test conditions, a low-speed, lower-altitude cruise capability of Mach 0. 6 at 30, 000 feet with an endurance of 5. 8 hours can be performed prior to the high performance tests in Case 1; similarly, a cruise capability of Mach 0. 64 at 25, 000 feet with an endurance limited only by the crew dose tolerance can be demonstrated in Case 2.

A complete speed-altitude profile and a weight and performance summary for each configuration discussed above is given in Figure 2. 20 and Table 2. 13.

2.6 HEAVY-PAYLOAD AIRCRAFT

The payload-carrying capability of nuclear turbojet, turboaft-fan, and turboprop applications was investigated.[25] A summary of this work follows.

2. 6. 1 TURBOJET

The studied turbojet applications involve three-, four-, and five-engine configurations designed for cruise altitudes ranging from 20, 000 to 40, 000 feet at flight speeds from Mach 0. 5 to 0. 9. Auxiliary chemical engines are employed in the three-engine nuclear turbojet applications designed for speeds from Mach 0. 5 to 0. 7. These assist during take-off and, in the event of an engine failure during take-off, provide the necessary performance margin required during an assumed 30-nautical-mile emergency go-around. These auxiliary power plants are not always necessary in the three-engine aircraft if the objective is merely to fly, regardless of payload. Auxiliary engines become necessary, however, when attempting to provide the largest possible payload.

2. 6. 2 TURBOAFT-FAN

The turboaft-fan applications also include three-, four-, and five-engine configurations and have been optimized for maximum payload for the same range of cruise conditions established for the turbojet applications. Again, the three-engine aircraft designed for the lower cruise speeds employs two chemical turbojet engines to provide auxiliary power during both normal and emergency take-off conditions.

2. 6. 3 BLEED TURBOPROP

The bleed-turboprop-aircraft applications investigated are powered by two or three gas generators and are designed for essentially the same flight spectrum as the other two sys-

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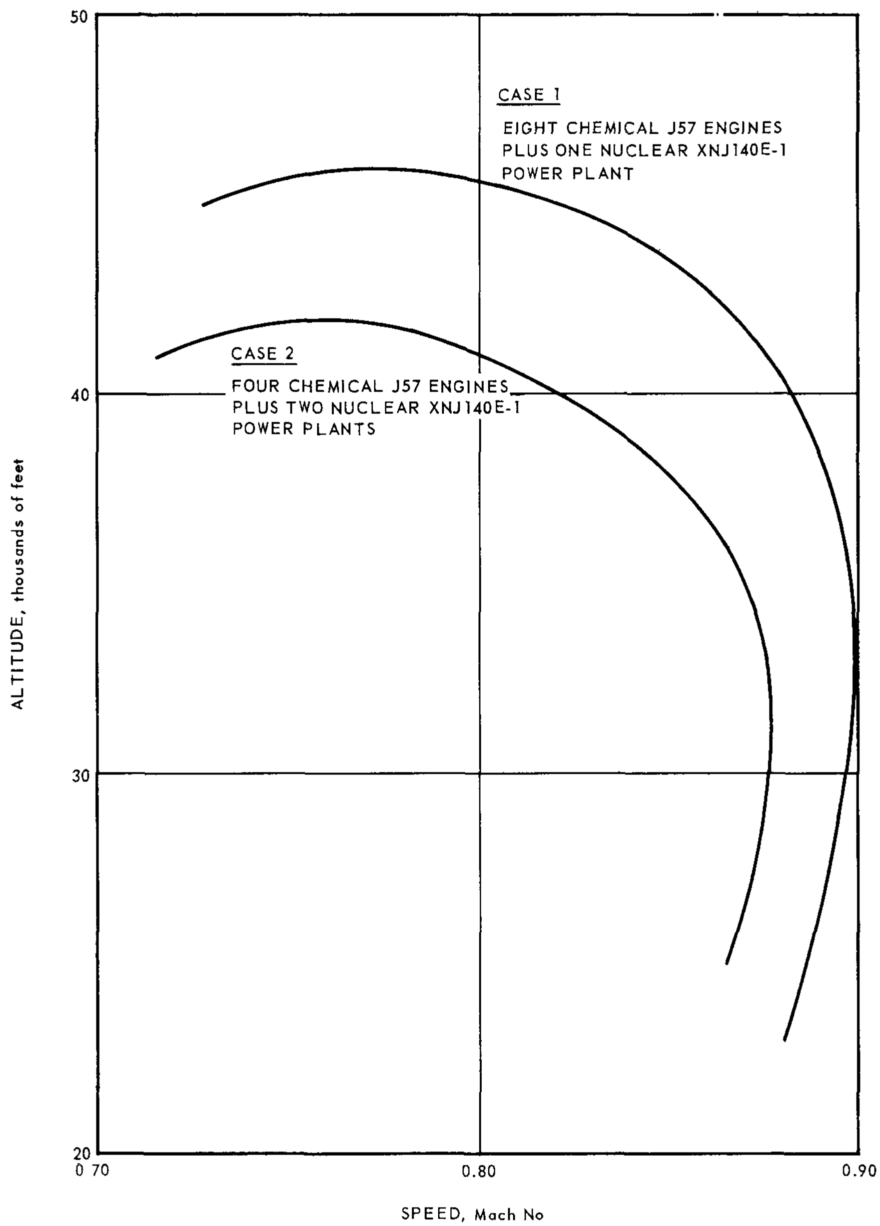

Fig. 2.20 – B-52G flight test aircraft high-speed, high-altitude test profiles

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TABLE 2.13

B-52G FLIGHT TEST AIRCRAFT WEIGHT AND PERFORMANCE SUMMARY

Case	1	2
Number chemical J57 engines	8	4
Number nuclear XNJ140E-1 engines	1	2
Crew shield weight, lb	58,500	65,000
Dose rate at 20,000 feet	0.1	0.1
Gross weight (less fuel), lb	298,000	346,000
Fuel weight, lb		
Reserve	10,000	10,000
High Mach-altitude cruise	20,000	20,000
Take-off and climb	16,000	24,000[a]
Low Mach-altitude cruise	106,000	
Total fuel weight, lb	152,000	54,000
Take-off gross weight, lb	450,000	400,000
Gross weight at start of high-speed, high-altitude cruise, lb	328,000	376,000
Endurance at low-speed cruise, hr	5.8	Unlimited
Low-speed performance, Mach No./altitude	0.6/30,000	0.64/25,000
Endurance for high-speed, high-altitude tests, hr	0.8	1.7

[a]Includes fuel for simulated emergency conditions.

tems. Mach 0.8 was substituted for Mach 0.9 for the high-speed cruise condition because preliminary studies indicated that undesirable results would occur at Mach 0.9.

2.6.4 BASIC FACTORS AND ASSUMPTIONS

The basic power plant associated with the study was an advanced design of the XNJ140E-1 based on the S23A cycle.[25, 26] The turbine inlet temperature at the normal continuous power level is 2000°F. At the military power rating, turbine inlet temperature is 2100°F. The airflow is 500 pounds per second at 100 percent corrected speed. Power-plant performance is given in Table 2.14.

All of the aircraft investigated for this study were the result of an aircraft optimization technique described in reference 27. Pertinent aircraft component-weight equations, aerodynamic-performance relationships, and definitions and assumptions relating to the various optimizations available are given in that report. For the studies under discussion, the aircraft were optimized to provide maximum payloads. The factors influencing the design and performance of a particular application (including payload capability) are the power-plant weight and performance capabilities, the cruise conditions assumed, the emergency performance required, and the desired take-off characteristics. In particular, the emergency (engine out) cruise and emergency go-around after take-off each require the aircraft to maintain a 100-foot-per-minute rate of climb capability at 5000 feet altitude under AFHD conditions with payload on board. The take-off restraints include a maximum

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TABLE 2.14

SUBSONIC X211 TURBOJET STUDY PERFORMANCE AT ARDC PRESSURE ATMOSPHERE, 100% RAM RECOVERY, NO BLEED-AIR OR POWER EXTRACTION

Altitude, ft	M_p	% rpm	F_n guar.	SFC guar.	Q, mw	P_o, psia	P_3, psia	T_3, °F	WHE, lb/sec	P_4/P_3	T_4, °F	P_6, psia	T_6, °F	W_6, lb/sec
	Normal Continuous, Standard Day													
20,000	0.5	98	16,400	0	102	6.753	111.8	632	260.1	0.740	2000	20.79	1317	290.1
	0.7	98	17,000	0	112	6.753	125.2	659	291.5	0.740	2000	22.88	1313	325.2
	0.9	98	17,900	0	124	6.753	142.0	690	331.2	0.741	2000	25.73	1313	369.4
30,000	0.5	98	11,900	0	72	4.364	77.4	583	178.7	0.735	2000	14.79	1324	199.3
	0.7	98	12,600	0	80	4 364	87.5	610	202.5	0.736	2000	16.43	1320	225.9
	0.9	98	13,600	0	91	4.364	101.6	645	235.6	0.737	2000	18.60	1314	262.9
40,000	0.7	98	8,380	0	53	2.720	57.2	582	131.0	0.729	2000	10.75	1321	146.1
	0.9	98	9,210	0	61	2.720	66.9	615	153.5	0.730	2000	12.31	1316	171.3
	Military, Hot Day													
5,000	0.5	100	21,000	0	146	12.228	165.1	759	378.1	0.743	2100	30.30	1394	421.8
	0.7	100	20,600	0	155	12.228	176.9	776	405.2	0.744	2100	33.10	1402	452.1
	0.9	100	20,300	0	164	12.228	189.9	795	435.1	0.744	2100	36.84	1417	485.4
10,000	0.5	100	19,200	0	130	10.106	145.5	743	332.5	0.741	2100	26.42	1389	370.9
	0.7	100	19,100	0	139	10.106	157.3	762	359.8	0.742	2100	28.89	1395	401.4
	0.9	100	19,100	0	149	10.106	170.9	783	391.0	0.743	2100	32.28	1407	436.2
20,000	0.5	100	15,200	0	99	6.753	107.8	702	245.1	0.737	2100	19.60	1387	273.5
	0.7	100	15,600	0	107	6.753	119.4	728	271.9	0.738	2100	21.53	1386	303.3
	0.9	100	16,100	0	118	6.753	133.1	755	303.4	0.739	2100	24.14	1391	338.5
	Maximum A/B, Hot Day, T_8 = 3040°F													
5,000	0.0	100	36,400	1.39	136	12.228	151.0	736	345.6	0.742	2100	27.47	1389	385.5
	0.0	103	37,700	1.42	140	12.228	158.2	759	361.8	0.742	2100	27.04	1369	403.7

Cycle

W_a = 500 lb/sec at $100\%N/\sqrt{\theta_2}$
P_3/P_2 = 13.3
P_4/P_3 = 0.74 (S23A burner out)
FF_4 = 158
Mil T_4 = 2100°F at 100%N
N. C. T_4 = 2000°F at 98%N
A/B T_8 = 3240°F
103%N for emergency take-off.

take-off velocity of 170 knots and a maximum critical field length of 15,000 feet, also assuming an AFHD atmosphere at 5000 feet altitude. The cruise altitudes and flight speeds are two of the major parameters involved in the evaluation. As previously indicated, these values range from 20,000 to 40,000 feet and from Mach 0.5 to 0.9 respectively.

The various data points that make up the working curves used in the study represent unique, optimized aircraft, and as such they are the best compromise between power-plant performance, aircraft characteristics, payload, and crew-shield design. This compromise indicates that the highest payload capabilities are associated with medium-subsonic flight speeds at altitudes ranging from 20,000 to 30,000 feet.

2.6.5 COMPARATIVE RESULTS

The comparative statistics of the three types of power plants (turbojet, turboaft-fan, and bleed turboprop) are indicated in Table 2.15. The cruise altitudes represent the altitude that corresponds to the best payload for the given aircraft configuration and flight speed.

The applications studied are very large aircraft. Some exceed 1,000,000 pounds in gross weight. The payload potential is also very high however, approaching 500,000 pounds.

[redacted]

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TABLE 2.15

COMPARISON OF NUCLEAR TURBOJET, TURBOAFT-FAN, AND TURBOPROP AIRCRAFT CHARACTERISTICS

	Three-Engine Turbojet[a]	Three-Engine Turboaft-fan[b]	Three-Gas-Generator Turboprop[a]
Gross weight, lb	844,000	973,300	1,340,000
Cruise weight, lb	829,000	958,300	1,327,000
Payload, lb	225,900	304,800	424,400
Crew shield weight, lb	100,000	87,000	100,000
Propulsion system weight, lb:			
Nuclear	188,200	193,500	188,200
Chemical auxiliary	17,000	17,000	
Bleed prop assembly and accessories			150,000
Fuel and oil weight, lb	22,400	22,400	18,000
Equipment weight, lb	37,270	39,850	46,430
Crew weight, lb	900	900	900
Structural fraction, percent	29.7	31	31

[a]Design point: Mach 0.7 at 20,000 feet.
[b]Design point: Mach 0.7 at 25,000 feet.

2.7 COMPARISON STUDIES

2.7.1 TURBOJET, TURBOPROP, AND TURBOAFT-FAN

Many studies were made comparing the effect of engine type on the performance of nuclear-powered aircraft. The conclusions derived from each study depend upon the aspect of performance being compared and upon the assumptions made for the analyses.

The results of one such study[28] are shown in Table 2.16. In this study, the engine performance assumptions were:

1. Engine primary air 300 lb/sec at SLS.
2. Compressor pressure ratio at SLS 14:1
3. Turbine inlet temperature 1700°F.
4. Compressor-to-turbine pressure ratio 0.70.

The power plants used water-moderated reactors with metallic fuel elements. A water and lead shield was assumed at the reactor and polyethylene plastic and lead shield at the crew compartment. Two wing-mounted reactor-shield assemblies are used for each airplane.

The turboprop payload of 270,000 pounds is 11 percent greater than that of the turboaft-fan and 4 percent greater than the turbojet payload. To power an aircraft of the same gross weight as the turboprop aircraft at the selected design point, the other engine types require more installed airflow and reactor power, which in turn involves greater power-plant weights. Consequently, the turboprop aircraft carries 67 percent more payload per pound per second of sea level static installed airflow than the turboaft-fan aircraft, and 336 percent more than the turbojet.

A similar study[29] was made which, in addition to the parameters of the previous study, also compares the "productivity" of the aircraft of the three different types ("productivity"

[redacted]

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TABLE 2.16

COMPARISON OF AIRCRAFT PERFORMANCE BY ENGINE TYPE

	Turbojet	Turboaft-fan	Turboprop
Altitude, ft.	20,000	20,000	20,000
Speed, mph	353	353	353
Aircraft gross weight, lb	715,000	715,000	715,000
Aircraft L/D	18.5	18.5	18.5
Installed power-plant weight, lb	314,000	257,600	227,000
Payload, lb	183,000	242,900	270,000
$W_{A_{SLS}}$ per reactor, lb/sec	685	450	300
Payload/$W_{A_{SLS}}$ total, $\frac{\text{lb}}{\text{lb/sec SLS}}$	134	270	450
Reactor power, mw	126	87.7	58.5
Payload ratio[a]	1.476	1.11	1.0
Power-plant weight ratio[a]	0.724	0.884	1.0
Reactor power ratio[a]	0.465	0.666	1.0
Reactor airflow ratio[a]	0.438	0.666	1.0
Payload/$W_{A_{SLS}}$ ratio[a]	3.36	1.67	1.0
Thrust at sea level and 70% take-off speed (dry)	66,800	78,000	108,000
Take-off thrust ratio (dry)[a]	0.618	0.722	1.0

[a]All ratios = $\frac{\text{Turboprop}}{\text{Turboaft-fan or Turbojet}}$

is defined as the payload times the cruise speed divided by the take-off weight minus the payload). It is, in other words, the available ton-knots per unit weight of the aircraft. This criterion gives an indication of the relative efficiency of comparable aircraft designs. Figure 2.21 illustrates the effect of design cruise speed on the relative productivity. At all of the design speeds studied, nuclear turboprop aircraft have greater productivity than aircraft using the other two types of engines. For the same reactor power, there is little difference between the propulsion-system weights, yet the turboprop has greater cruise and take-off thrust than the turboaft-fan or turbojet. As a result, aircraft of greater gross weight are possible with turboprop engines, permitting more payload and greater productivity. This is not true for chemically-powered aircraft, primarily because such aircraft vary in weight during flight (due to the consumption of fuel), while the weight of nuclear aircraft is essentially constant.

The basic conclusions of this study may be stated as follows:

1. Where speed is of primary importance (and is greater than Mach 0.8), a nuclear turbojet or turboaft-fan is preferred.
2. If payload and/or low dose rates external to the reactor shield are of major interest, the nuclear-powered turboprop is superior to the turbojet and turboaft-fan.

A study was made at GE-ANPD in 1957 to compare turboprop and turbojet performance for a nuclear-powered low-level subsonic bombing mission.[30] The study considers sea level flight at Mach 0.8 and 0.9. All comparisons are made with design-point engines; no off-design point performance is considered. The comparison is based on a turbine inlet temperature of 1800°F and an aircraft L/D of 7 at Mach 0.9 and 8.58 at Mach 0.8. Reactor-shield-assembly sizing methods, power-plant performance, and airplane assumptions are detailed in the study report. The following conclusions were derived:

1. Power plants using one reactor-shield assembly are lighter in weight than those using two.

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$$\text{PRODUCTIVITY} = \frac{\text{PAYLOAD X AIR SPEED}}{\text{GROSS WEIGHT - PAYLOAD}}$$

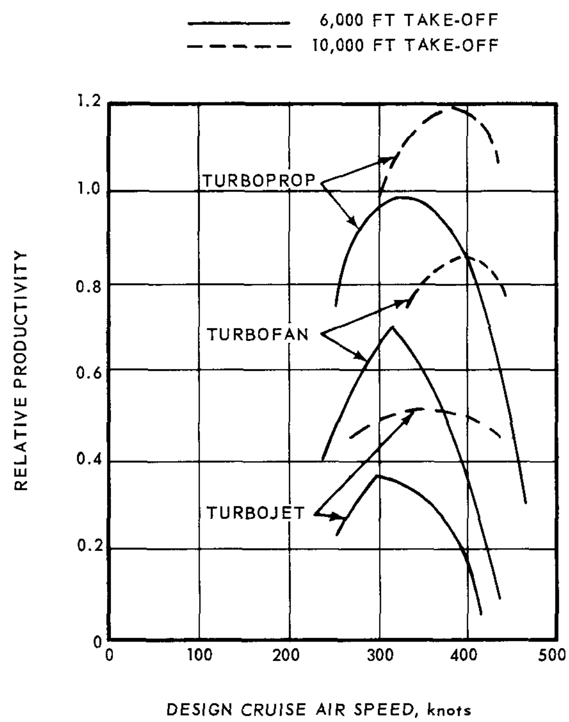

Fig. 2.21 – Effect of engine type on relative productivity of nuclear-fueled aircraft

2. The turbojet engine has a higher thrust-to-weight ratio than the turboprop, at the operating points examined.
3. The gross weight required to carry a given payload at the design point is lower in the case of the turbojet-powered airplane.
4. The total engine airflow and the reactor power required are smaller in the case of the turboprop power plant.
5. All parameters plotted indicate an improvement in the relative performance of the turboprop as the flight speed is decreased.
6. A relative improvement in turboprop performance is indicated when unit shielding is used.
7. The turbojet requires a higher airflow for a given thrust at the design points studied. The larger flow requires a larger core and a heavier reactor-shield assembly. However, for the points considered, the extra weight required for the reactor-shield assembly is not as great as the additional weight required for ducts, propellers, bleed turbines, gear boxes, etc., needed for the turboprop power plant.

Another interesting comparative study was made of nuclear turboprops and turbojets[31] in the attempt to establish the comparative performance of the two systems throughout a range of flight conditions. The first set of comparisons covered the range of speeds from Mach 0 to 1.0 at 20,000 feet altitude. The turbine inlet temperature was varied from 1300° to 1700°F. A high level of component efficiencies was assumed for one comparison and a low level for another. Also, two propellers were assumed: a subsonic propeller with poor performance above Mach 0.6, and a supersonic propeller with good performance up to Mach 1.0.

[redacted]

[redacted]

The second performance comparison assumed a take-off thrust-to-gross-weight ratio of at least 0.20 (equal to that of the B-47 airplane). The other assumptions were: low component efficiency level, a subsonic propeller, a turbine-inlet temperature of 1500°F, an airplane L/D of 18, take-off with chemical fuel burning at a turbine temperature of 1700°F, and with afterburning for the turbojet.

The following conclusions were obtained from this study:

1. The thrust, or thrust-to-weight ratio, may be equal through a wide range of flight speeds, depending on the turbine-inlet temperature and the efficiency level. At high efficiency, high temperature, and with a supersonic propeller, the turboprop is competitive up to a speed of Mach 1.0. At low efficiency, low temperature, and with a subsonic propeller, the turboprop is not competitive above Mach 0.5 or 0.6.
2. The flight speed for equivalent thrust-to-weight ratio decreases with a decrease in turbine-inlet temperature; also for decreasing component efficiencies.
3. To be competitive above Mach 0.7, the turboprop engine requires a supersonic propeller.
4. A low-efficiency turboprop is not competitive with a high-efficiency turbojet above Mach 0.4.
5. The turboprop airplane has a good load-carrying ability at Mach 0.6 and below. The load-carrying capacity decreases to zero at Mach 0.8 or 0.9. The turboprop has quite good high-altitude flight capability.
6. The turbojet airplane has lower load-carrying ability but can carry its load at higher flight speeds. It can cruise with small loads in the range from Mach 0.8 to 1.0.

2.7.2 DIRECT AND INDIRECT CYCLES

A brief comparison study of several nuclear-propulsion systems for a long-range subsonic cruiser or transport aircraft was made in 1956.[32] The systems compared for this type of mission are the AC-300-1 turboprop, the compact core reactor (CCR) turboprop, and the CCR turbojet. An attempt was made to compare the systems under the same state-of-the-art or development-time periods to meet a possible flight date of 1959 or 1960. Since the study was done in a general manner, the results should be considered as an indicative comparision rather than a final, absolute answer to this type of nuclear mission. Some of the assumptions used in the study are:

1. Hardware is "state-of-the-art." This means the use of designs that were being operated, manufactured, or at least on the drawing board in the final design stage in 1956. (A moderate extrapolation of items was, of course, necessary and permissible.)
2. A cruising speed of Mach 0.57 (350 knots) at 20,000-feet altitude, and comparable short-runway-airport performance, which is necessary for aircraft operating out of of advance military or transport bases.
3. The aircraft are assumed to have the same gross weight and L/D characteristic.
4. Dose rates in the cargo compartment of 5 rem per hour total and in the crew compartment of 0.05 rem per hour.
5. Reactor-shield assembly designed for wing pods (two to an aircraft).
6. Engines of a size under development in 1956.
7. Power-plant life of 500 hours.
8. Lead-water-type reactor shields.
9. Turbine-inlet temperatures of 1700°F for the air cycle and 1385°F for liquid-metal cycles.
10. Final comparison is made on net payload for aircraft of the same gross weight.

Tables 2.17 through 2.20 present the comparative performance data and weights.

[redacted]

The analysis indicated that the AC-300-1 and the CCR turboprop are very similar in payload factor; the selection between these two would have to be done on a comparison of mechanical feasibility. Liquid-metal power plants were not suitable for aircraft consideration at the time (1956). In particular, good, light-weight radiator design is a problem. Judging from the success of about eight major heat-exchanger companies, development of this item is progressing at a slow rate. Because fire is one of the major hazards in aircraft, the liquid-metal power plant requires a nearly perfect design and assembly; there is little margin for error.

One basic point in this comparison favoring the AC-300-1 is the asymmetric shield design (the design of the CCR shield is isotropic). This discrepancy is probably more than balanced by the assumption of aircraft of identical L/D and gross weight. The CCR uses eight engines and the air cycle uses only four; the added nacelle drag of the extra engines would materially affect the L/D characteristic of this aircraft. The CCR engines are also larger in diameter than the AC-300-1 because the radiator determines the major diameter of the engine. Since this is also true in the turbojet version, wing-mounting rather than pod-mounting of this engine is virtually a necessity to cut down frontal area.

It is somewhat invalid to compare the performance of jet aircraft with turboprops at a single point, since the jet may be much better at higher speeds; but since this is a comparison based on "state-of-the-art" assumptions, speeds high enough to make the jet appear advantageous would have to be considered in a later effort.

2.7.3 AIRPLANE COMPARISONS

Studies were also made comparing performance of various chemical airplanes with nuclear power plants installed in them.

Table 2.21 gives the performance of a B-52, a KC-135, and a C-133, as possible vehicles for the XMA-1 power plant for a "quick flight" of a nuclear airplane.[33] This study, made in 1958, also covered schedule and cost estimates as well as performance of new airplanes suggested by Convair and Lockheed.

Another study[34] considered the Princess Flying Boat, the C-133A, and the R3Y as flight test beds for the installation of two X-235A bleed turboprops. The data provided are in the form of graphs showing power available and power required for these modified airplanes at various altitudes and flight speeds.

TABLE 2.17

ALTITUDE PERFORMANCE PER REACTOR

	Power Plant		
	AC-300-1	CCR Turboprop	CCR Turbojet
M_p	0.57	0.57	0.57
Altitude, ft	20,000	20,000	20,000
Number of engines per reactor	2	4	2
Total SLS airflow, lb/sec	300	690	700
Turbine inlet temperature, ^{o}F	1700	1385	1385
Reactor power, mw	55	70	107
Dose rate at 50 feet, rem/hr	5	5	5
Crew compartment dose, rem/hr	0.05	0.05	0.05
Shaft horsepower	20,500	20,300	
Propeller efficiency	0.87	0.87	
Jet thrust, lb	1,030	1,080	17,630
Net thrust, lb	17,630	17,630	17,630

[redacted]

TABLE 2.18

AIRPORT PERFORMANCE PER REACTOR

	Power Plant		
	AC-300-1	CCR Turboprop	CCR Turbojet
Net thrust, lb	65,000	64,080	69,400
Reactor power, mw	83	83	128
SFC-interburner		0.05	0.450
Take-off fuel, lb			
10 minutes, interburner and afterburner		500	17,000
10 minutes, interburner		500	3,550

TABLE 2.19

REACTOR COMPONENT WEIGHTS

	Power Plant		
	AC-300-1	CCR Turboprop	CCR Turbojet
Reactor-shield assembly	80,000	63,800	77,500
SMCS (cooling system)	3,500		
Engines	9,200	18,450	17,960
Propellers and gearbox	6,600	10,500	
Ducting	1,850		
Radiator		5,050	5,120
Pumps		1,200	1,000
Valves		1,500	1,000
Piping (wet)		3,180	1,880
Expansion tank		800	1,000
Radiator, NaK		360	550
A. P. U.		2,500	3,500
Total reactor weight	101,150	107,440	109,510

NOTE: All weights in pounds.

TABLE 2.20

AIRCRAFT PERFORMANCE

	Power Plant		
	AC-300-1	CCR Turboprop	CCR Turbojet
Cruise thrust, lb	35,460	35,460	35,460
L/D	16.5[a]	16.5[a]	16.5[a]
Gross weight, lb	585,000	585,000	585,000
Power plant weight, lb	202,300	214,880	219,020
Crew shield weight, lb	29,800	29,800	29,800
Fuel weight, lb		4,000[b]	82,000[b]
Total power plant weight, lb	232,100	248,680	330,820
Airframe weight, lb	175,000	175,000	175,000
Airframe and power plant weight, lb	407,100	423,680	505,820
Payload, lb	177,900	161,320	79,180
Payload/gross weight	0.304	0.276	0.135

[a]Low-density payload.

[b]Enough chemical fuel was added for initial take-off, and a turn-around take-off from a forward base that may not have a large fuel supply.

[redacted]

TABLE 2.21

AIRPLANE COMPARATIVE PERFORMANCE

		Type of Aircraft		
		B-52	KC-135	C-133
Description of aircraft	Take-off gross weight, lb	331,000	251,000	255,000
	Nuclear power plant	1 XMA-1	1 XMA-1	1 XMA-1
	XMA-1 installation	Top of fuselage	Top of fuselage	In cargo hold
	Chemical power plants	8 J57	4 J57	4 T34
	Chemical fuel weight, lb	45,000	33,000	22,000
Required performance for demonstration of nuclear flight at 10,000 feet	Derated[a] nuclear thrust, %	48.5	42.0	39.2
	Approximate speed, Mach No.	0.4	0.45	0.35
Performance at 100% derated[a] nuclear thrust	Approximate maximum altitude, ft	20,000	25,000	25,000
	Approximate speed, Mach No.	0.5	0.5	
Ultimate capability	Ultimate airframe capability, Mach No.	0.62	0.63	0.42
	Tactically useful	Possibly	No	No

[a]Derated nuclear thrust is defined as 50 percent of the nuclear thrust of a fully developed XMA-1.

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3. SUPERSONIC AIRCRAFT

3.1 EFFECT OF POWER-PLANT WEIGHT ON PERFORMANCE

3.1.1 INTRODUCTION

The achievement of large weight reductions of nuclear power plants is difficult because of the shielding requirements of the reactor. It is desirable therefore, to have a clear understanding of the performance gains that may accrue as a result of any weight reductions. A study was made in 1955[1] to determine the magnitude of the improved aircraft performance (and the decrease in aircraft gross weight) to be expected from progressive improvements in future thrust-to-weight ratios. A wide range of power-plant weights was considered.

The design and construction of a nuclear reactor capable of heating air to a temperature of 2500°F was believed possible within a few years. Likewise, metallurgical and turbine-design developments indicated that turbines might soon be operating at inlet temperatures of 2500°F. To determine the maximum sustained flight speeds that might be anticipated in the near future, the performance of the turbojet engine was calculated for this inlet temperature.

The aircraft used in the study do not represent the most efficient possible designs for the given conditions but are a nearly uniform series for comparing the progressive improvement resulting from the progressive reduction of the power-plant weights.

All of the aerodynamic data apply to the aircraft only in the clean configuration. No attempt is made to estimate take-off and landing performance. The lift-drag ratios should be considered approximate within plus or minus one.

For the purposes of this study, the power plants are represented in terms of the ratios of the total power-plant weights to the weights of the basic turbojet engines used in the power plants. For example, a given power-plant design is designated a 4X-system if the total weight of the power plant is four times the weight of the basic turbojet engines. A system may incorporate one or more engines; i.e., a 2X-system may consist of an air-heating reactor with its associated shielding (including crew shielding if any) plus two, four, or more basic engines. This method of defining the power plants permits the assumption of a uniform series of power plants and corresponding aircraft for which the performance may be computed and compared. The effects of the air-ducting and reactor sizes on the power-plant weights are factored out completely when the systems are compared in this manner.

The ideal nuclear-propulsion system would be one weighing no more than a chemical power plant of equivalent thrust. However, this ideal system could probably not be achieved in the foreseeable future, but a 2X-system might have been obtainable within ten years if a concerted effort were made to develop the art of shielding nuclear reactors (the final GE-ANPD nuclear power-plant designs were approximately 4X-systems, compared to the earlier 9X-systems). The study covers the range of practical interest from 2X- through 6X-systems.

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3.1.2 THE BASIC TURBOJET ENGINE

The basic engine is a turbojet without interburning or afterburning. In this study, four engines are considered the desirable number. However, to keep the size of the basic engine below 450 pounds per second SLS airflow, it is necessary to use six engines in the larger aircraft. Because the aircraft gross weights and thrust requirements vary with the system weights, it is also necessary to vary the size of the basic engine to hold the maximum number of engines to six. As a result, the size and weight of the basic engine are variables. The engine size is determined by the number of engines and the total airflow necessary to produce the thrust required to fly the airplane.

Only aircraft capable of cruising at supersonic speeds on nuclear power alone are assumed and, therefore, the basic engine performance is calculated for a compressor compression ratio of 6.0 at SLS conditions and a turbine-inlet temperature of 2500°F. Under these assumptions, a maximum speed of Mach 3.5 is achieved at altitudes of 35,000 feet and above.

3.1.3 THE AIRCRAFT

The general outline of the aircraft studied is shown in Figure 3.1. The nose of the fuselage is ogival-shaped, capable of enveloping a suitable crew compartment, including shielding if necessary. However, the shielded crew compartment has a larger cross section than the unshielded, necessitating a larger fuselage. The performance of aircraft both with and without crew shielding is calculated.

Previous aircraft application studies show that the maximum diameter of the fuselage is determined by the size of a five-man crew compartment since the development of the tandem-core and solid-moderator reactor systems has reduced the size of the reactor to the point that it does not necessarily set the maximum diameter of the fuselage. On this assumption, the fuselage diameter in this study is determined by the dimensions of the crew compartment.

The five-man crew compartment, with a net inside cross section of 25 square feet, is the basic crew compartment used in this study except for the 2X-system, two-engine aircraft which is based on a one-man crew and a cross section of 13.5 square feet.

Layouts of the aircraft considered in several application studies show the fuselage maximum section to be 2.45 times the inside cross section of the crew compartment when no nuclear shielding is used and the separation distance between the crew compartment and the power plant is not held at 50 feet. When divided shielding is used and the separation distance is held to a minimum of 50 feet, the fuselage maximum section is shown to be 4.74 times the inside cross section of the crew compartment. These ratios will govern if the crew compartment is located well forward yet within the contour of the ogival nose.

All of the aircraft are equipped with a plane wing, tapered in plan form only with 0.388 taper ratio, and 3.1 aspect ratio. The wing has a 3 percent thick biconvex section. This type of wing is chosen because, at the design speed of Mach 3.5, it appears to offer aerodynamic characteristics as good as any other type plus relative structural simplicity.

Tables 3.1, 3.2, and 3.3 present a summary of the aircraft dimensions and the performance of the systems studied.

3.1.4 CONCLUSIONS

On the basis of the methods and assumptions used and the results obtained, the following conclusions were drawn:

1. Gross weight is mainly altitude-limiting and does not necessarily restrict cruising or maximum speed if altitude is not a fixed-design requirement. Because gross weight increases the minimum speed and can create a problem with respect to airport per-

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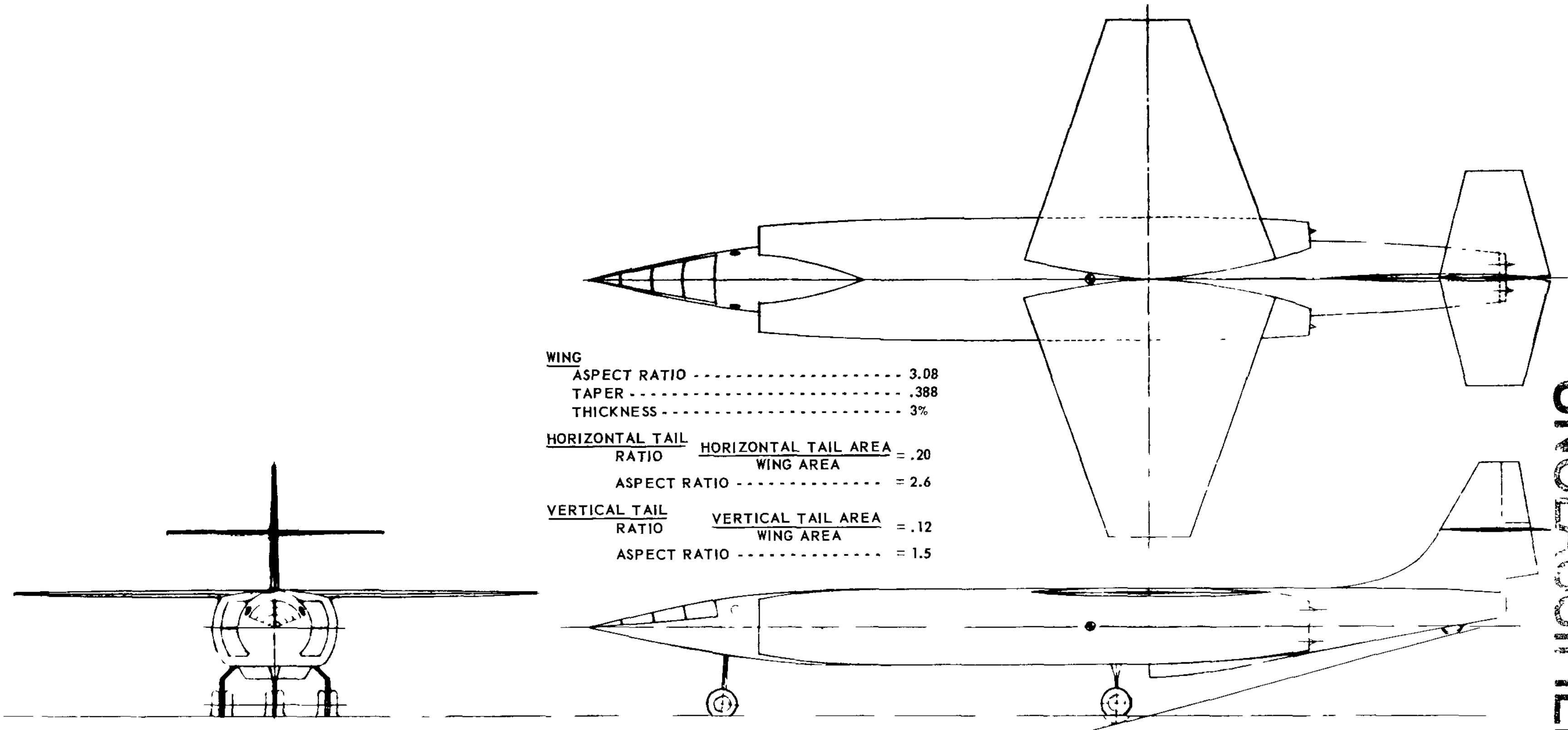

Fig. 3.1 – Supersonic nuclear-powered aircraft

TABLE 3.1

AIRCRAFT DIMENSIONS

System Designation	Number Of Engines	Airplane Gross Weight (W), lb	Total SLS Airflow (W_{a_t}), lb/sec	Wing Area (S), ft^2	W/S lb/ft^2	Wing M. A. C., ft	Fuselage Diameter, ft	Fuselage Frontal (Drag) Area, ft^2	Fuselage Length, ft	Fuselage Wetted Area, ft^2
2X	2	48,200	450	677	71.2	15.8	7.45	43.6	74.5	1,795
2X	4	91,000	900	1,280	71.2	21.75	8.83	82.2	88.3	2,525
3X	4	129,300	1,120	1,815	71.2	25.9	8.83	87.3	88.3	2,525
4X	4	189,000	1,400	2,660	71.2	31.3	8.83	93.9	88.3	2,525
6X	6	422,000	2,400	5,130	82.2	43.5	8.83	117.1	103.3	3,790
4X[a]	6	222,000	1,800	2,970	74.8	33.1	12.3	160.3	144	6,720
6X[a]	6	484,000	2,700	5,360	90.4	44.4	12.3	181.3	144	6,720

[a]Aircraft with crew shielding.

[redacted]

TABLE 3.2

PERFORMANCE SUMMARY - PART 1

System Designation	Number Of Engines	Aircraft Gross Weight, lb	Power Plant Gross Weight, lb	Basic Engine Weight, lb	SLS Airflow Per Engine, lb/sec	Design Altitude, ft	L/D at Design Altitude, M = 3.5	Approximate Absolute Ceiling, ft
2X	2	48,200	13,600	3,400	225	65,000	3.85	69,000
2X	4	91,000	27,200	3,400	225	65,000	3.87	67,500
3X	4	129,300	54,000	4,500	280	65,000	4.27	65,900
4X	4	18,900	96,000	6,000	350	65,000	4.63	65,500
6X	6	422,000	259,000	7,200	400	62,000	5.07	62,400
4X[a]	6	222,000	119,000	4,950	300	64,000	3.93	64,600
6X[a]	6	484,000	302,500	8,400	450	60,000	4.67	60,900

[a]Aircraft with crew shielding.

TABLE 3.3

PERFORMANCE SUMMARY - PART 2

System	Number Of Engines	Acceleration: M.9 To M3.5 Time, Min	Acceleration: M.9 To M3.5 Distance, N. Mi.	Acceleration: M.9 To M_{max} Time, Min	Acceleration: M.9 To M_{max} Distance, N. Mi.	M_{max}	Time To Climb To Design Altitude: SL To 35,000 ft, Min	Time To Climb To Design Altitude: 35,000 ft Design Altitude, Min	Design Altitude, ft
2X - 2 Engines	2	5.55	97.0	7.41	167	4.07	1.86	1.67	65,000
2X - 4 Engines	4	4.50	79.1	6.74	165	4.17	1.69	0.97	65,000
3X	4	5.48	106	7.70	183	4.2	1.87	1.14	65,000
4X	4	7.14	120	9.53	211	4.15	2.28	2.03	65,000
6X	6	11.5	185	15.1	323	4.20	3.25	1.98	62,000
4X[a]	6	6.31	108	9.94	246	4.18	2.17	3.37	64,000
6X[a]	6	13.0	208	16.2	333	4.12	3.46	2.52	60,000

[a]Aircraft with crew shielding.

formance, it is desirable to keep the gross weight of the power plant and aircraft as low as possible.

2. Large reduction of power-plant weights may very nearly double the rate-of-climb capability of a nuclear-powered airplane.
3. Very high initial rates of climb may be expected for high-supersonic aircraft because of the high forward velocity and excess thrust available in such aircraft designed to fly at high altitudes.
4. Low compressor pressure ratios are necessary if very high supersonic speeds are to be obtained. Speeds above Mach 2.2 are impossible for the aircraft studied if ratios greater than 6 to 1 are used. The power-plant thrust drops rapidly to zero above Mach 2.2 if pressure ratios greater than 6-to-1 are used.

   While at the time of this study (1955), the low values of compressor pressure ratio resulted in large reactors and increased ducting sizes, it was necessary to design eventually for the lower figures. For this reason, great strides were necessary to improve shielding theory and its application to aircraft reactor design to effect substantial reductions of the power-plant weights.
5. It is necessary to achieve very high turbine-inlet temperatures if the nuclear-powered aircraft are to fly at speeds of Mach 3.5.

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3.2 HIGH-SPEED, HIGH-ALTITUDE AIRCRAFT

3.2.1 GE X211 - P140Y2 ENGINE APPLICATIONS

An evaluation was made of the supersonic capabilities of an airplane of the B-70 type utilizing five X211-P140Y2[2] nuclear power plants.[3] Two basic configurations were studied.

1. "Dry" Configuration - The first study was the "Dry" configuration; i.e., no chemical fuel was to be carried to achieve supersonic cruise conditions. This configuration was to obtain all of its thrust from the five nuclear power plants operating at military power with no chemical afterburning. The take-off gross weight of 525,000 pounds was calculated as indicated in Table 3.4. The subtotal of the basic component weights was assumed to account for 69 percent of the take-off gross weight, with the structural weight comprising the remaining 31 percent.

TABLE 3.4

"DRY" CONFIGURATION
COMPONENT WEIGHTS

Item	Weight, lb
Power plants (five X211-P140Y2)	248,000
Crew shield	60,000
Equipment	30,000
Payload	20,000
Useful load	4,000
Subtotal	362,000
Structure	163,000
Take-off gross weight	525,000

Lift-to-drag ratios, obtained from discussions with airframe manufacturers and the Aircraft Nuclear Propulsion Office (ANPO), were applied to this take-off gross weight to determine the thrusts required at various altitudes and speeds. These L/D values are shown in Table 3.5.

Correlation of the required thrusts with the installed thrust produced by the five X211-P140Y2 nuclear power plants operating at military power yielded the supersonic performance shown in Table 3.6. This configuration does not have the 1000-nautical-mile sprint capability at Mach 3.0 and 65,000 feet altitude.

2. "Wet" Configuration - To determine the additional weight of fuel required to provide the Mach 3.0 sprint capability, a similar study was conducted on a "Wet" configuration of the B-70 type, again utilizing five X211-P140Y2 nuclear power plants. The fuel required at maximum chemical afterburning for acceleration and climb to Mach 3.0 at 65,000 feet from Mach 2.7 at 45,000 feet, and the fuel required at modulated chemical afterburning for maintaining Mach 3.0 at 65,000 feet for 1000 nautical miles were computed and included in the weight summary, Table 3.7.

The assumed basic component weights are also tabulated. As in the analysis of the "Dry" configuration, the subtotal of these weights was assumed to account for 69 percent of the take-off gross weight with the remaining 31 percent allocated to structural weight.

[redacted]

TABLE 3.5

LIFT-TO-DRAG RATIOS FOR A "DRY" CONFIGURATION SUPERSONIC AIRPLANE

Altitude, ft	Speed, Mach No.	L/D
45,000	2.5	7.0
45,000	3.0	6.0
65,000	3.0	7.5

TABLE 3.6

SUPERSONIC PERFORMANCE SUMMARY FOR THE "DRY" CONFIGURATION

Speed, Mach No.	Altitude, ft
3.0	45,200
2.8	47,300
2.6	49,400
2.5	50,400

TABLE 3.7

COMPONENT WEIGHTS OF "WET" CONFIGURATION

Item	Weight, lb
Power plants (five X211-P140Y2)	248,000
Crew shield (including 12,000 pounds of fuel)	60,000
Equipment	30,000
Payload	20,000
Useful load	4,000
Fuel for climb and acceleration from Mach 2.7 at 45,000 feet to Mach 3.0 at 65,000 feet, maximum afterburning	7,300
Fuel for sprint (1,000 nautical miles, Mach 3.0, 65,000 feet) with modulated afterburning	33,700
Subtotal	403,000
Structure	181,000
Take-off gross weight	584,000

[redacted]

Utilizing the same lift-to-drag ratios referred to in the discussion of the "Dry" configuration, correlation of the thrusts required with the installed thrust available from the nuclear power plants operating at military power yielded the nuclear-power-only performance shown in Table 3.8.

Comparison of the all-nuclear-power supersonic cruise altitudes at corresponding speeds for each configuration indicates that the additional weight of 59,000 pounds of the "Wet" configuration results in a reduction in nuclear cruise altitude of about 2,200 feet. However, the "Wet" configuration can achieve Mach 3.0 at 65,000 feet for 1,000 nautical miles and the "Dry" configuration cannot.

TABLE 3.8

SUPERSONIC PERFORMANCE SUMMARY
FOR THE "WET" CONFIGURATION

Speed, Mach No.	Altitude, ft
3.0	43,000
2.8	45,100
2.6	47,200
2.5	48,300

Work was also performed at North American Aviation on this subject.[4] The conclusion reached by both General Electric and North American Aviation was that supersonic nuclear capability appeared to be attainable and that these studies should be continued and supported.

3.2.2 OTHER ENGINE APPLICATION STUDY

Various other studies were conducted to determine the capability of specific power plants to meet the requirements of high-performance aircraft. In addition, parametric studies were made that indicate the supersonic potential of nuclear propulsion. These studies are described below.

3.2.2.1 Nuclear Propulsion Application Screening Study[5]

A study was conducted on the application of nuclear power to the B-70. From the results, it appeared doubtful that the B-70 could maintain the desired performance spectrum with the suggested nuclear configuration.

3.2.2.2 Nuclear Propulsion Application Parametric Study[6]

Another study investigated advanced nuclear power plants assuming turbine-inlet temperatures of 2000°F and 2300°F. The side, front, and rear shield material considered was borated beryllium oxide. This high-temperature material was used because of the high sprint speed desired.[7,8]

Several power plants were sized, each meeting the thrust requirements of the B-70 at Mach 0.9 at 25,000 feet, Mach 1.2 at 25,000 feet, and Mach 3.0 at 60,000 feet. The results of the study are given in the form of power-plant thrust required per pound per second of engine airflow. This permits choice of engine size.

3.2.2.3 Preliminary Power-plant Sizing[9,10]

A preliminary power-plant sizing study investigated single-engine power plants capable of powering a high-performance aircraft at Mach 3.0 and 60,000 feet with no afterburning. The study was based on the use of plutonia-thoria as a core material with a core exit temperature of 2500°F. The shield and reflector materials used were Inconel and thoria.

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3.2.2.4 Power-plant Capabilities for the B-70 Bomber[11]

A study of power plants for the B-70 was performed to determine the requirements of the B-70 aircraft operating at a flight speed of Mach 2.3 at 45,000 feet. The engines considered were the J93, the X211, and an engine of the size of the X211 with optimum compressor pressure ratio. In each case, the power-plant was a single engine with the shaft through the core. The engine was assumed to require no chemical interburners and the ratio of turbine-inlet pressure to compressor-discharge pressure was 0.75. Reactor core sizes were determined for a folded-flow design, and reactor-shield assembly weights were based on dose rates contained in reference 12. Turbomachinery weights were based on the original weights of the J93 and X211 engines, and were varied as functions of compressor pressure ratio and turbine-inlet temperature.

The aircraft requirements at the specified flight condition were based on the best available drag data for the chemical B-70 design. Weight and balance studies were not included in the evaluation because of inadequate data. The aircraft was assumed to carry 10,000 pounds of payload and 40,000 pounds of chemical fuel for afterburning operations. The weight of the crew shield was assumed to be 60,900 pounds, of which 9500 pounds was reserve fuel.

This study indicated that for nuclear flight of the B-70 both the J93 and X211 engines are inferior to the optimized-design engine of X211 size. To keep the required turbine-inlet temperatures within reasonable limits, a 5- or 6-engine configuration was necessary. It seemed unlikely that 5 or 6 X211 engines would fit into the chemical B-70 engine compartment without major modification.

3.2.2.5 Visit by North American Personnel to ANP June 10, 1960[13]

This document outlines the various requirements and indicates the type and weight of power plant needed to provide an acceptable supersonic system. The results of studies conducted by North American Aviation mating the P140E power plant with the B-70 are included.

3.3 HUNTER/KILLER AIRPLANE

3.3.1 DESCRIPTION OF THE MISSION

The Hunter/Killer application was one interesting mission proposed for advanced nuclear power plants. Essentially, this is a counterforce system capable of destroying enemy strategic weapons such as ICBM's, long-range aircraft, and their associated bases.

The mission requirements define an airplane with the following basic capabilities:

1. Airborne alert for extended periods of time.
2. Penetrate enemy territory either at sea level or at high altitudes and high speeds.
3. Carry large payloads (50,000 to 100,000 pounds).

3.3.2 AIRPLANE ANALYSIS

To determine the applicability of nuclear power for the Hunter/Killer mission, several of the advanced nuclear power plants discussed in previous sections of this report were applied to airplanes designed for this mission.

3.3.2.1 Assumptions

The supersonic lift-to-drag ratios used in the airplane studies are shown in Figure 3.2. These are design-point values and are based on a fuselage of the fixed B-70 type. The effect of design speed on the ratio of structural weight to gross weight is shown in Figure

[redacted]

[redacted]

3.3. The data are based on a study[14] that includes such factors as aerodynamic heating, wing loadings, and wing design. The results agree quite well with present or proposed aircraft designs. Typical wing loadings to be expected for supersonic airplanes are given in Table 3.9.

All of the power plants considered were assumed to have no interburners and only enough chemical fuel on board for use in the afterburners. The cruise dose rate was set at 0.02 rem per hour at the subsonic loiter condition.

3.3.3 RESULTS

The performance of the resulting airplane, for payloads of 50,000 and 100,000 pounds, is given in Tables 3.10 and 3.11. One of the advanced power plants, the A115D,[13] has the potential of flying at Mach 3.0 even with 100,000 pounds of payload. The maximum speed of all the airplanes at sea level was limited to Mach 0.95 because of a power-plant limitation. Several of the advanced power plants could probably attain supersonic speeds on the deck if certain design changes were incorporated.

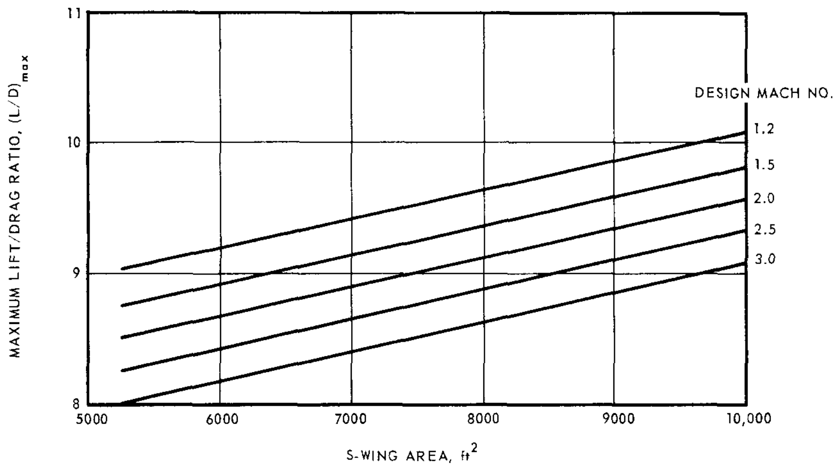

Fig. 3.2 – Maximum design lift-to-drag ratios

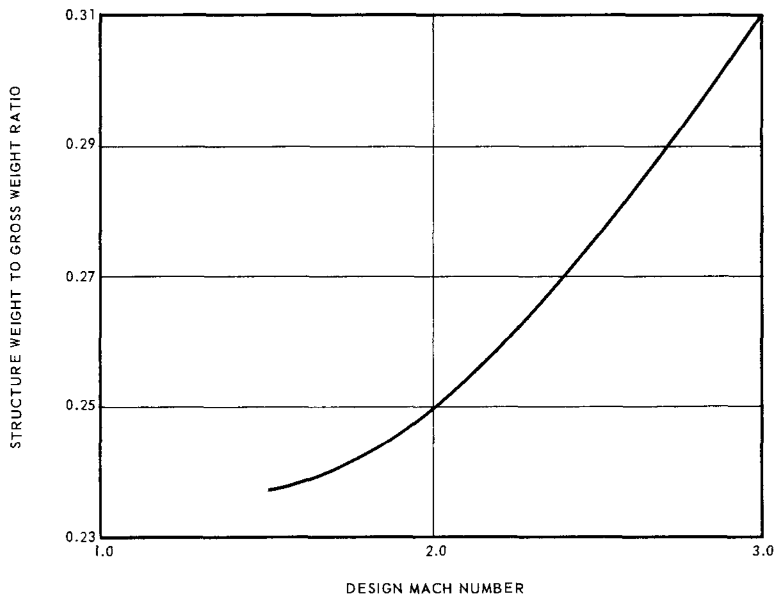

Fig. 3.3 – Effect of design Mach number on airframe structure weight

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TABLE 3.9

WING LOADINGS FOR SUPERSONIC AIRPLANES

Design Speed, Mach No.	Wing Loading, lb/ft^2
1.5	100
2.0	94
2.5	87
3.0	79

TABLE 3.10

HUNTER/KILLER AIRPLANE PERFORMANCE WITH 50,000-POUND PAYLOAD

Power Plant	Number Of Power Plants	Gross Weight, lb	Sea Level Speed, Mach No.	Speed At Altitude	
				Mach No.	Altitude, ft
438 lb/sec (T_4 = 1740°F)	4	550,000	0.89	0.85	32,000
XNJ140E	4	578,000	0.91	0.85	35,000
XNJ140E (T_4 = 2100°F, Mil.)	5	654,000	0.9 - 0.95	1.5	35,000
P122C3 (see ref. 15)	4	548,000	0.95	2.5	40,000
	5	645,000	0.95	2.7	40,000
A115D	4	587,000	0.95	3.1	58,000

Note: T_4 = turbine-inlet temperature.
Mil = military power setting.
Payload = 50,000 pounds.

TABLE 3.11

HUNTER/KILLER AIRPLANE PERFORMANCE WITH 100,000-POUND PAYLOAD

Power Plant	Number Of Power Plants	Gross Weight, lb	Sea Level Speed, Mach No.	Speed At Altitude	
				Mach No.	Altitude, ft
438 lb/sec 70,000 lb Payload (T_4 = 1740°F)	4	580,000	0.88	0.85	29,500
XNJ140E (T_4 = 2100°F Mil.)	4	638,000	0.88	0.85	31,500
P122C3	4	580,000	0.9 - 0.95	1.2	25,000
	5	698,000	0.95	2.5	40,000
A115D	4	660,000	0.95	3.0	55,000

Note: T_4 = turbine inlet temperature.
Mil = military power setting.
Payload = 100,000 pounds

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4. MISSILES

4.1 RAMJET MISSILE

4. 1. 1 MISSION

A nuclear-powered ramjet could meet the requirement for a low altitude, supersonic, high-payload missile with long endurance. Such a missile is shown in Figure 4. 1.

A possible flight profile for the nuclear ramjet could consist of initial propulsion by a solid-propellant booster to a flight speed of Mach 3. 5 at 35, 000 feet. During this phase, the nuclear power plant is started and brought to full power. Nuclear cruise is used at these flight conditions over friendly territory and oceans. When the point of possible enemy detection and interception is reached, the missile descends to 1000 feet and penetrates at Mach 2. 8. The use of a ramjet engine and a solid-propellant booster makes possible a long-term standby capability without deterioration of parts. Nuclear power provides ample range to allow evasive tactics and indirect approach to enemy territory.

The missile should have an around-the-world range, requiring a maximum endurance of approximately 11 hours at speeds from Mach 2. 7 to 3. 5. A normal mission is assumed to be about 5. 5 hours. Maximum altitude of the projected mission is 40, 000 feet. The over-all weight of the missile is less than 50, 000 pounds.

4. 1. 2 POWER-PLANT DESCRIPTION

The nuclear ramjet power plant consists of a supersonic inlet, a subsonic diffuser, a nuclear reactor with its associated controls, and a fixed convergent-divergent jet nozzle.[1, 2, 3, 4] A typical arrangement is shown in Figure 4. 2.

The supersonic inlet contains a variable spike for control of the throat area during start-up. Of the external-internal compression type, the inlet is designed essentially for operation at Mach 2. 7 at 1000 feet altitude, AFHD. The subsonic diffuser is an annular duct between the throat of the inlet and the face of the reactor core.

The ceramic-tube-type reactor has an over-all diameter of 58. 5 inches, an active core diameter of 50 inches, a core length of 50 inches, and an over-all length of 76. 2 inches. Ram air is used for cooling the control rods, the reflector, and the support structure. The control rods are all contained in a 5-inch-diameter tube in the center of the core. The reactor is described in detail in a later section.

The jet nozzle is of the fixed-geometry, convergent-divergent type, providing for full expansion of the air at the design point to maximize thrust. Ram air is used to cool the nozzle structure and to film-cool the inner surface of the nozzle.

The power-plant configuration proposed for a ground test is shown in Figure 4. 3. The reactor and jet nozzle are designed as described above but the supersonic inlet has been replaced with an inlet duct that feeds air to the reactor-nozzle system at the prescribed conditions of temperature, pressure, and flow rate.

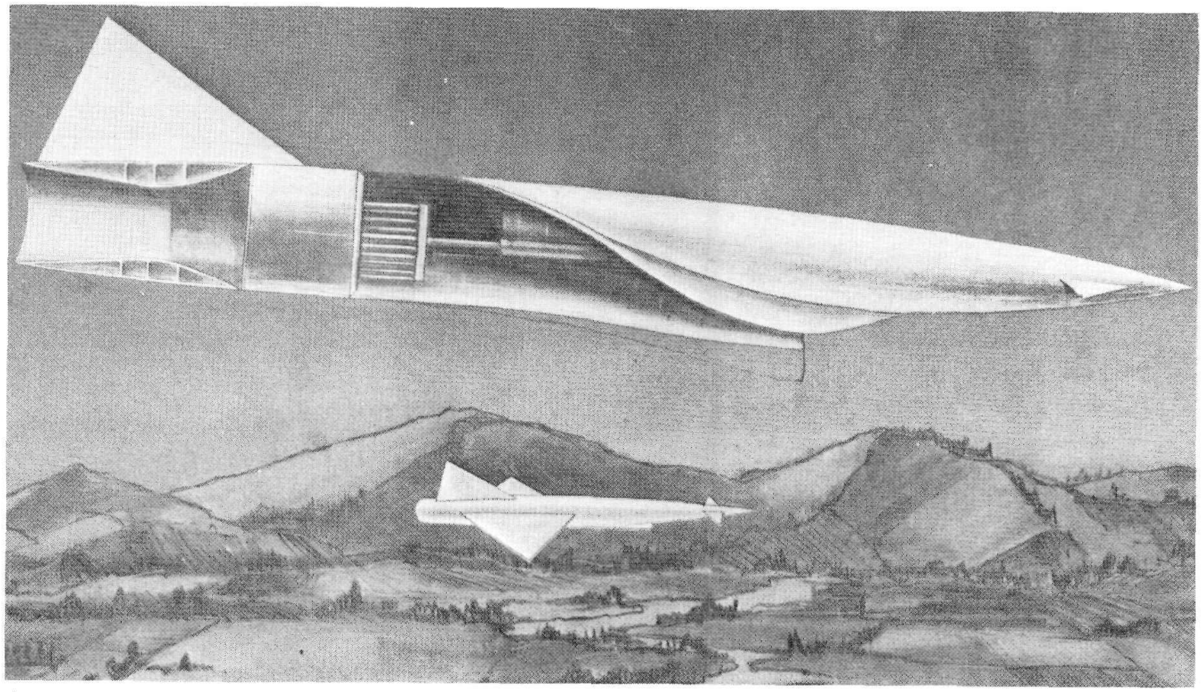

Fig. 4.1—P212 power plant

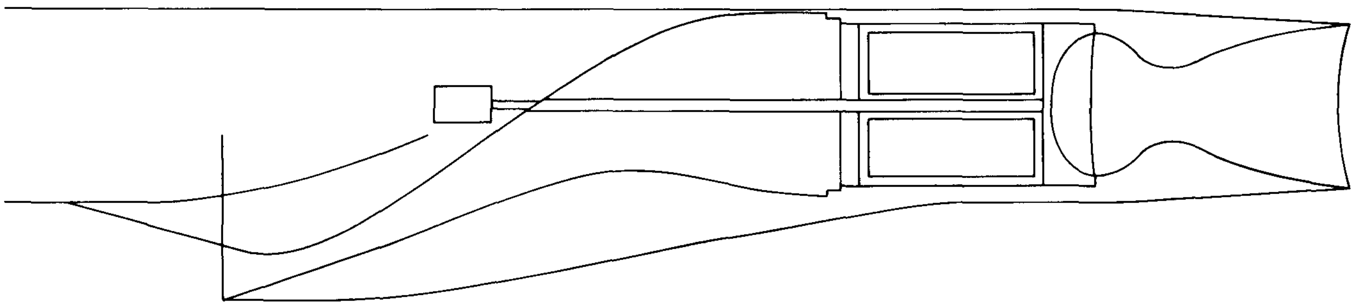

Fig. 4.2 – Ramjet power-plant configuration, bottom inlet

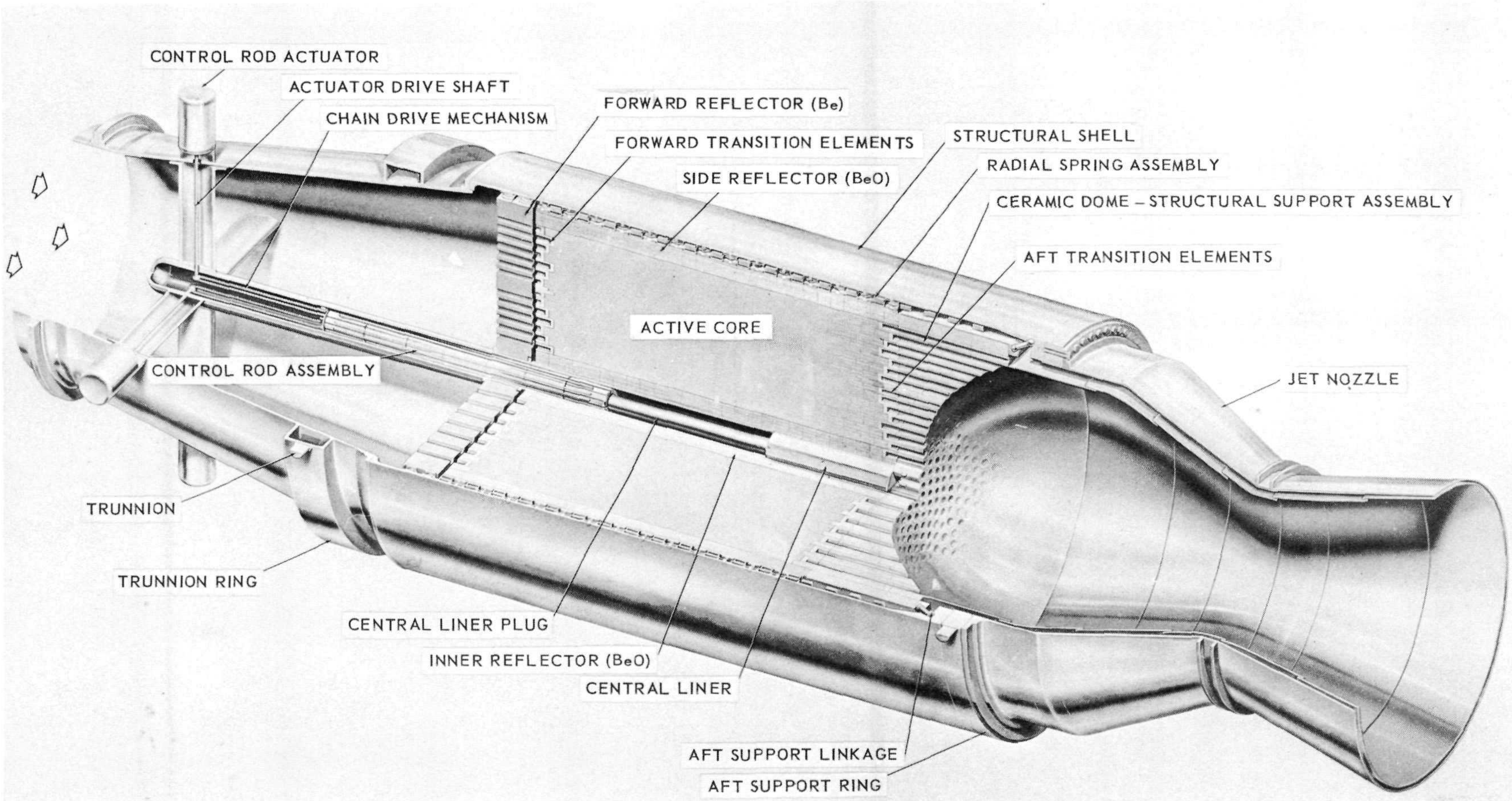

Fig. 4.3 – Nuclear ramjet ground-test prototype

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4.1.2.1 Reactor Description

The reactor for the ramjet power plant is similar to that of the XNJ140E-1 reactor. The design concept of this reactor is the use of simple ceramic shapes as the basic building elements to minimize the operating thermal and mechanical stresses, simplify fabrication procedures, and minimize costs. By comparison, reference 5 is of interest because it is a pioneer conceptual design of a solid-moderated ceramic-tube reactor, published in 1953.

The basic modular ceramic element is a small hexagonal prism, 0.3217 inch across flats and 2 inches long, which is used throughout the radial cross section of the reactor in the central island, active core, and side-reflector regions. The size of the fuel element is derived from thermal-, nuclear-, and mechanical-design considerations. The multitude of uniform hexagonal elements is maintained as a unit assembly be external compression provided by an assembly of springs and an external shell. The configuration of the reactor is shown in Figure 4.4 and dimensional data are presented in Table 4.1. The mechanical arrangement of the ramjet reactor was adopted because of its inherent advantages: (1) removal of structural poison from the active core; (2) increased homogeneity; (3) elimination of low-temperature components from the active core; (4) good radial power distribution due to the effect of the central reflector island; and, (5) the central placement of control rods for maximum poison worth. No alternative arrangement considered offered these advantages.

The ceramic-reactor elements are assembled around an Inconel X support liner in the central island. The inner reflector in the center island consists of hexagonal beryllium oxide elements. Although most of these elements are solid bars, some contain holes for cooling air. The radial thickness of the inner reflector is 1.55 inches.

The active core is made up of fueled BeO hexagonal tubes. The inside diameter of the fuel tubes is 0.2535 inch. The fuel element body is a BeO matrix to which fuel is added in the form of a solid-solution mixture of enriched urania and yttria in the ratio of 45 UO_2 - $55Y_2O_3$. The Y_2O_3 serves as a stabilizer of the UO_2.

The outer reflector is 2.25 inches thick in the radial direction. As in the inner reflector, some of the elements are hollow to provide for the cooling air. The front reflector consists of 5.2 inches of beryllium metal, which also provides the forward support for the ceramic elements, and 2 inches of BeO. The aft reflector structure is composed of the aft transition elements and the aft support assembly, both of BeO. The thickness of the transition elements is 2.0 inches; the minimum thickness of the support assembly is 4.0 inches. Both the front and aft BeO transition elements are shaped to provide a transition section in which the air from 19 fuel tubes is channeled into one air passage. Transition to a larger hole in this manner simplifies the design and fabrication of the forward beryllium reflector and the elements of the aft retainer assembly.

The reactor is held together by the compression of Rene' 41 radial springs bearing against radial pressure pads of cast alloy, Haynes 713C. These springs, in turn, are fastened to the Inconel X structural shell.

Reactivity control in the ramjet reactor is achieved through the insertion and withdrawal of control rods within the central island. The control rods are made of EuO dispersed in a nickel matrix and clad with Nichrome V. The control rods are made of four separate assemblies, each with a separate actuator mechanism. The actuators, adapted from the XNJ140E-1 reactor program, consist of a power-head assembly connected through a drive shaft to a chain mechanism which drives the control rods.

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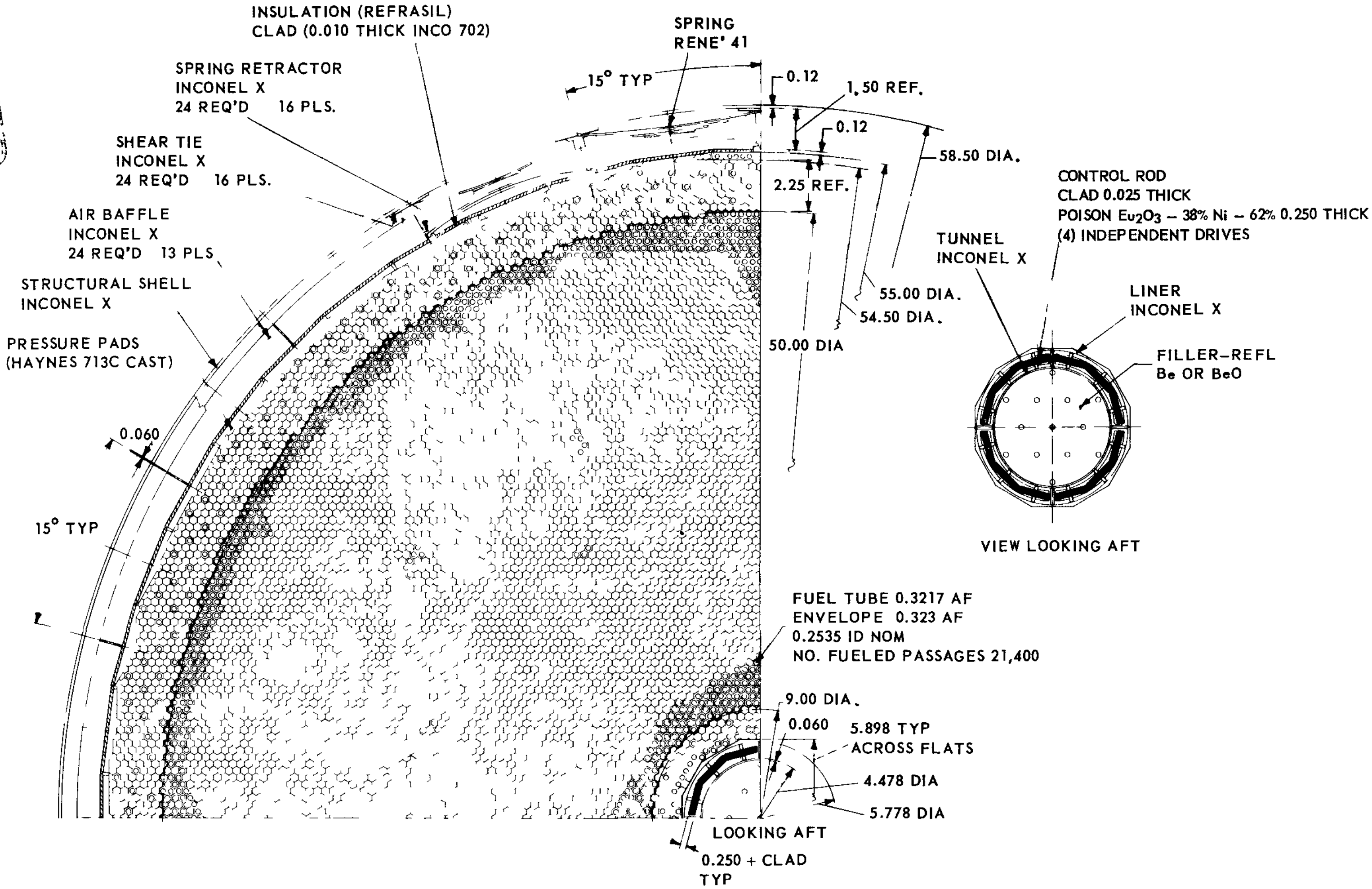

Fig. 4.4 -- Radial section of nuclear-ramjet reactor

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TABLE 4.1

SUMMARY OF CONFIGURATION DATA FOR RAMJET REACTOR

Dimensions, in.	
Over-all reactor diameter	58.5
Over-all length: structural shell	83.95
reactor assembly	76.20
Radial reflector OD	54.5
Active core OD	50.0
Active core ID	9.0
Inner reflector ID	5.9
Center liner ID	5.778
Front reflector thickness: Be	5.2
BeO	2.0
Active core length	50.0
Rear reflector structure thickness: maximum	18.5
minimum	6.0
Fuel Elements	
Average density of fuel elements (% of theoretical)	98
Hydraulic diameter (D_H), in.	0.2535
Width across flats (W_f), in.	0.3217
Number of fuel element channels	21,400
Total weight of reactor assembly, lb	13,170[a]

[a]Less controls and instrumentation.

The aft retainer assembly is one of the most critical parts of the ramjet reactor because it operates at elevated temperatures while supporting the reactor against the high inertial and aerodynamic-drag loads. The classical Roman arch design of the aft retainer assembly provides a simple means of achieving these objectives, since all the components are in direct compression. The assembly is built up of tubular ceramic elements of approximately the same size as transition elements. Shear is transmitted from tube to tube by intersurface friction, which is augmented by mechanical interlocking features. At the perimeter of the assembly, the drag loading and thrust are transmitted to a reinforced ring on the structural shell through linkage bars which permit differential radial expansion between reactor and shell. Under zero-drag-load conditions, the integrity of the structure is maintained by radial springs similar to those in the reactor section.

4.1.2.2 Reactor Performance

The fundamental concept employed in optimizing the reactor performance is that the maximum thrust per unit free-flow area is obtained when the product of specific impulse and reactor flow rate per unit flow area is a maximum. This maximum does not occur when either of the components of the product is a maximum. At the optimum condition, the flow area of the reactor for a required net thrust is a minimum, which translates into the minimum reactor size for a specific fuel loading and selection of reactor materials. Design-point performance data are summarized in Table 4.2. Nuclear and thermal design data are summarized in Table 4.3.

In evaluating the reactor nozzle system at off-design conditions, the maximum-average fuel element temperature was assumed to remain constant at the design-point

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TABLE 4.2

DESIGN-POINT PERFORMANCE DATA

Flight Mach number (hot day)	2.7
Altitude, ft	1000
Net thrust (F_n), lb	45700
Average core-exit-air temperature (T_{T5}), ^oF	2090
Mixed-air nozzle inlet temperature (T_{T6}), ^oF	2040
Reactor airflow (W_T), lb/sec	1675
Active-core airflow (0.92 W_T), lb/sec	1540
Reactor power level, mw	560
Ram pressure recovery (P_{T3}/P_{To})	0.848
Front plenum (straightener) pressure ratio (P_{T4}/P_{T3})	0.985
Reactor inlet pressure (P_{T4}), psia	279
Reactor inlet temperature (T_{T4}), ^oF	887
Reactor pressure ratio (P_{T6}/P_{T4})	0.70
Nozzle velocity coefficient (C_v)	0.985
Average power density in fueled BeO, Btu/in.^{3}-sec	12.88
Maximum power density in fueled BeO, Btu/in.^{3}-sec	16.84
Specific impulse, sec	27.3
Net thrust coefficient (F_n/A_q) (58.5-inch-diameter reactor)	0.2365

TABLE 4.3

SUMMARY OF THERMAL AND NUCLEAR DESIGN DATA

Reactor Temperatures	
Reactor inlet air (T_{T4}), ^oF	887
Reactor exit air (T_{T5}), ^oF	2090
Maximum-average fuel element surface (T_{sm}), ^oF	2550
Maximum fuel element hot spot (T_{mm}), ^oF	2940
Maximum exit air temperature (maximum hot tube), ^oF	2310
Mach Numbers	
Reactor inlet	0.210
Reactor exit	0.447
Pressures	
Ambient air (P_o), psia	14.17
Diffuser exit (P_{T3}), psia	29.3
Reactor inlet (P_{T4}), psia	279
Reactor exit (P_{T5}), psia	196.3
Nozzle inlet (P_{T6}), psia	195.3
Nozzle exit (P_8), psia	14.17
Nuclear Data	
Reactivity (cold, clean)	1.05
Average UO_2 fuel loading, wt %	3.0
Temperature effect (2000^{o}F), % $\Delta k/k$	-3.10
Ten-hour mission xenon requirement (2000^{o}F),[a,b] % $\Delta k/k$	-1.65
Central rod worth (2000^{o}F), % $\Delta k/k$	7.00
Core Volume Fractions	
Fuel element flow area	0.56828
Waste void (stacking tolerances, rounded corners, etc.)	0.01232
Fueled BeO	0.41940
Total	1.00000

[a]Assumed 7-hour cruise plus 3-hour penetration.
[b]Reactivity loss due to fission product poisoning and burnup are negligible compared with xenon and temperature effects.

[redacted]

value of 2550°F. This was done because fixing it at the design value guarantees maximum off-design performance without exceeding the temperature limitations of the reactor materials. It was also assumed that the inlet always provides sufficient airflow to cause the nozzle throat to remain choked. This is mandatory for the convergent-divergent nozzle under these conditions, since a constant reactor exit Mach number is created because of the fixed-area relationship between the reactor and the nozzle throat for the fixed-area nozzle employed. The off-design performance of the reactor-nozzle system is readily determined for any selected flight point under these conditions. Selection of the flight point immediately fixes the reactor inlet conditions which, together with the fixed fuel element surface temperature, results in a unique combination of airflow rate, temperature increase, and pressure loss that satisfies the reactor exit Mach number requirement and therefore the choked nozzle. The computation of net thrust then follows directly.

4.1.3 PERFORMANCE

A ramjet engine is quite sensitive to the efficiency of the inlet-ram-pressure recovery. A review of the inlet-total-pressure recovery, as well as the design requirements and performance of various types of inlets, led to the selection of an external-internal compression-inlet design. The design point performance was based on the total-pressure-recovery curve shown in Figure 4.5. The inlet recovery for a design speed of Mach 2.7 is approximately 0.845, based on conditions of the air from free stream to the completion of subsonic diffusion.

The pioneer work predicting the performance of nuclear ramjets is given in reference 6 and is of interest in this performance section.

Recent data indicate that, with the use of bleed, substantial increases in ram recovery are possible. To determine design details, a study would have to be conducted, in conjunc-

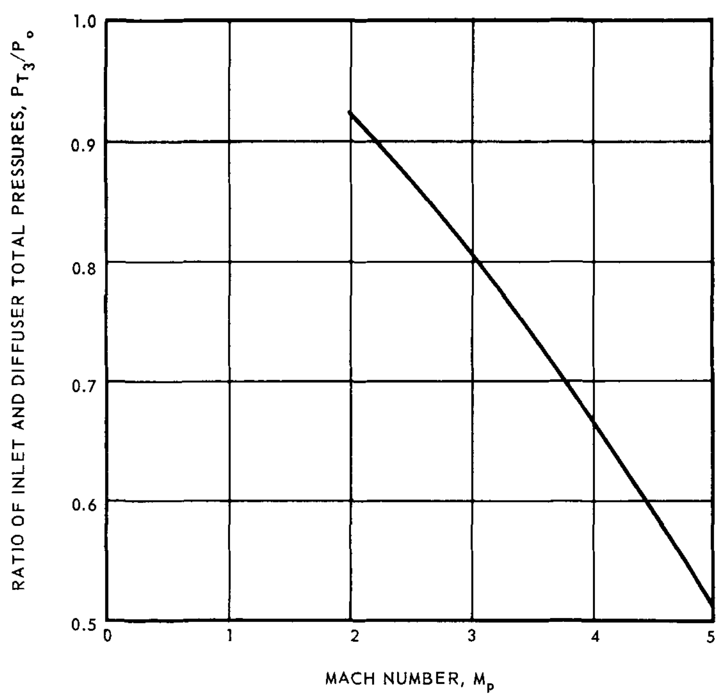

Fig. 4.5 – Assumed ram recovery

tion with an airframe manufacturer, to determine locations of bleed ports, amount of on-design bleed, use and dumping of bleed air, effect of bleed on off-design performance, and methods of bleed control over the flight spectrum. Increasing the recovery from 0.845 to 0.920 would increase the on-design thrust level discussed herein by approximately 5 percent. In this analysis, a total pressure recovery of 0.985 from the end of the diffuser to the reactor face was used; a total pressure recovery at the reactor face of approximately 0.832 resulted.

A portion of the total inlet air is bypassed around the reactor core to cool the reflector, control rods, reactor structure, structural shell, and nozzle. The studied configuration requires a core-discharge temperature about 50°F higher than the design nozzle-inlet temperature at the flight-design point. With a mixed-air temperature of 2040°F entering the nozzle, a design-point thrust level of 45,700 pounds is obtained with a maximum-average temperature of 2550°F at the fuel element surfaces. The nozzle coefficient used in the study was 0.98, which is applicable to a cooled, fixed-nozzle size for the on-design conditions.

Although performance was not analyzed over a large range of off-design speeds and altitudes, preliminary analysis indicates that the design is compatible with flight at altitudes between 30,000 and 35,000 feet and at speeds greater than the design value for low-altitude operation.

The reactor power level at the design point is 560 megawatts. The total reactor airflow is 1675 pounds per second and the specific impulse of the engine is 27.3 seconds. A summary of the on-design and operating conditions, and three off-design conditions, is given in Table 4.4. Of particular significance is the greater thrust (approximately 17 percent) available at Mach 2.8, standard day than at Mach 2.7, hot day.

TABLE 4.4

DESIGN POINT AND OFF-DESIGN PERFORMANCE DATA

Flight Condition	Design Point	Off Design	Off Design	Off Design
Altitude	1,000	1,000	35,000	35,000
Free-stream Mach number	2.7	2.8	3.5	4.0
Atmosphere	Hot Day	Std Day	Std Day	Std Day
Reactor inlet temperature (T_{T4}), °F	887	840	876	1040
Reactor inlet pressure (P_{T4}), psia	279	322	197	353
Average-channel maximum fuel element surface temperature (T_{sm}), °F	2550	2550	2550	2550
Reactor exit temperature (T_{T5}), °F	2090	2045	2120	2095
Reactor exit pressure (P_{T5}), psia	197	218	133	251
Nozzle inlet temperature (T_{T6}), °F	2040	2000	2070	2060
Nozzle inlet pressure (P_{T6}), psia	195.5	220	134.0	248
Nozzle airflow (W_6), lb/sec	1,675	1,900	1,143	2,100
Net thrust[a] (F_n), lb	45,700	53,600	34,100	28,250
Specific impulse (I_{sp}), sec	27.30	28.30	29.80	13.40
Net thrust coefficient[b] (C_F), (F_n/A_q)	0.2365	0.2565	0.4280	0.2715
Reactor power (Q), mw	560	635	395	520

[a]Nozzle flow is fully expanded at design point and under-expanded at all off-design points shown. Plug thrust based upon fixed nozzle exit diameter of 49.2 inches.
[b]Based on maximum reactor diameter of 58.5 inches.

[redacted]

4. 1. 4 FACILITIES AND GROUND SUPPORT

In 1957, a preliminary study[7] was made of the facilities, ground support, and operational-equipment requirements necessary to support a complete test program of a nuclear ramjet missile prior to tactical operational usage. The study considered two nuclear-powered ramjet missile configurations (manned and unmanned), a proposed developmental program, environmental effects of flight operations, a proposed operational base, and requirements of other existing and proposed sites considered necessary to support the proposed power-plant developmental and flight programs. The report is of interest in conjunction with the application study presented in this section.

4.2 NUCLEAR TURBOJET MISSILES

4. 2. 1 PHOTOGRAPHIC-RECONNAISSANCE MISSILE

The first study by GE-ANPD of the application of AC-series nuclear propulsion systems to guided missiles was performed in 1954 for a photographic-reconnaissance type mission.[8] Performance was estimated over a range of gross weights from 37, 000 to 60, 000 pounds, both with and without chemical reheat, and for turbojets of two different pressure ratios. Performance was limited to the supersonic flight condition.

The general configuration of the missile studied is shown in Figure 4. 6. A photograph of a model is shown in Figure 4. 7. The dimensions of four configurations are given in Table 4. 5. Missile weights are presented in Table 4. 6 and performance is tabulated in Table 4. 7.

TABLE 4. 5

MISSILE DIMENSIONS AND PARAMETERS

	Missile Designation		
	ACA-7	ACA-8 ACA-8B	ACA-9
Wing			
Aspect ratio	3. 0	3. 0	3. 0
Taper ratio, C_R/C_T	2. 0	2. 0	2. 0
Thickness-chord ratio	0. 05	0. 05	0. 05
Wing loading, lb/ft^2	125	125	125
Area, ft^2	288	400	324
Sweep angle referred to midchord line, degrees	0	0	0
Span, ft	29. 42	34. 68	31. 20
N. A. C. , ft	10. 16	11. 97	10. 77
Canard control surface			
Aspect ratio	3. 0	3. 0	3. 0
Taper ratio, C_R/C_T	2. 0	2. 0	2. 0
Thickness-chord ratio	0. 05	0. 05	0. 05
Sweep angle referred to midchord line, degrees	0	0	0
Area, ft^2	29. 85	41. 45	33. 58
Vertical stabilizer			
Aspect ratio	1. 5	1. 5	1. 5
Taper ratio, C_R/C_T	2. 0	2. 0	2. 0
Thickness-chord ratio	0. 05	0. 05	0. 05
Sweep angle referred to midchord line, degrees	12. 5	12. 5	12. 5
Area, ft^4	57. 60	80	64. 8
Fuselage			
Maximum length, ft	78. 0	92. 0	82. 8
Maximum depth, ft	7. 66	9. 03	8. 12
Maximum width, ft	5. 09	6. 0	5. 40

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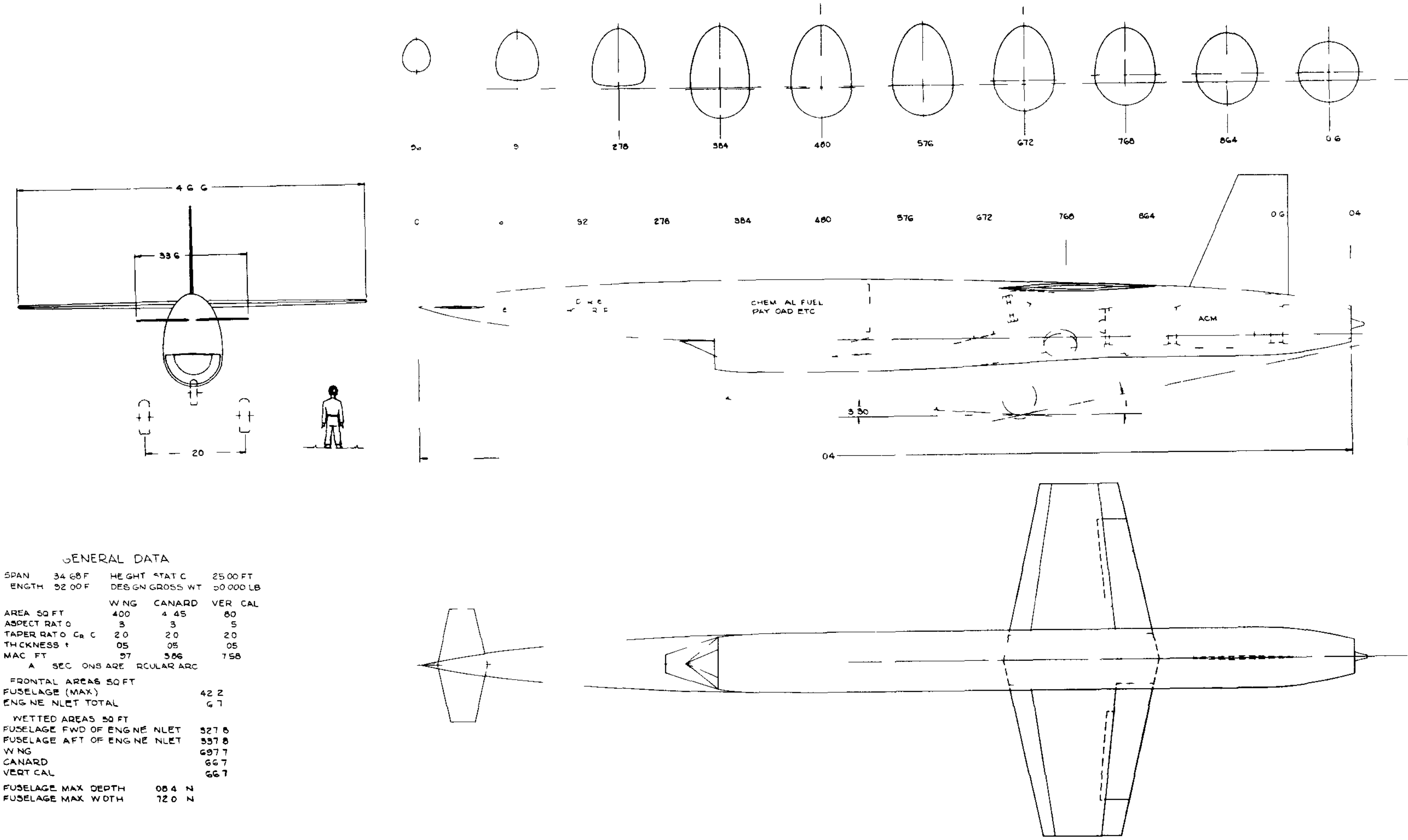

Fig 4.6 – Layout of ACA-8 missile

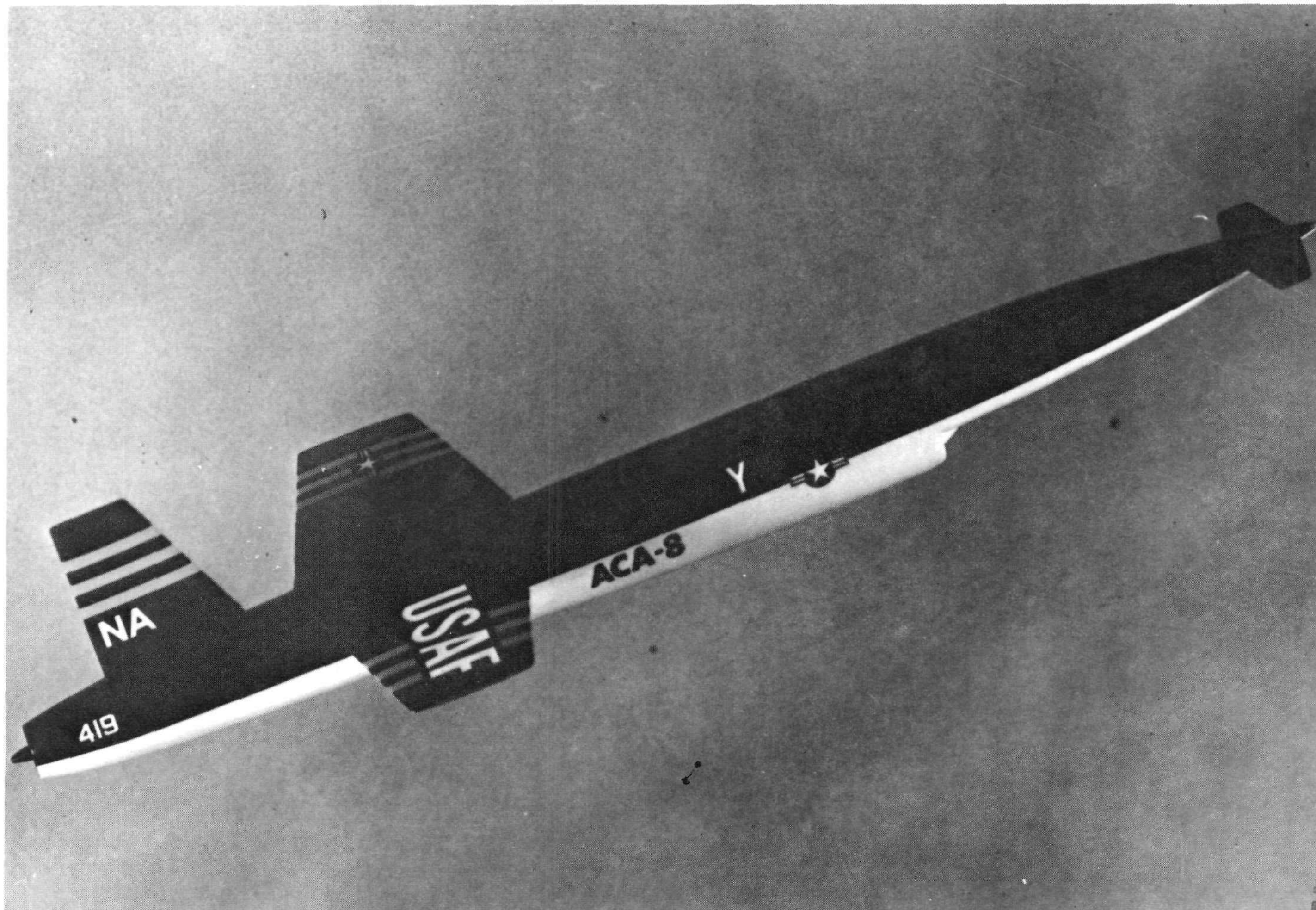

Fig. 4.7 – ACA-8 missile

[redacted]

TABLE 4.6

MISSILE WEIGHTS

	Missile Designation			
	ACA-7	ACA-8	ACA-9	ACA-9B
Airframe				
Wing group	2,700	3,750	3,040	3,750
Tail group	810	1,125	910	1,125
Fuselage group	3,780	5,250	4,250	5,250
Alighting gear	1,710	2,375	1,925	2,375
Total Airframe	9,000	12,500	10,125	12,500
Power plant				
ACM-1C-Mk II[a]	21,243	21,243	—	—
ACM-1C-Mk III[a]	—	—	24,662	24,662
Afterburner	—	650	—	—
Fuel system	—	250	—	—
Fixed equipment				
Guidance and control	2,200	3,000	2,400	3,000
Photographic	3,000	3,000	3,000	3,000
Unaccounted	557	357	313	6,838
Weight empty	36,000	41,000	40,500	50,000
Chemical fuel	—	9,000	—	—
Design gross weight	36,000	50,000	40,500	50,000

[a]Same as AC-6.

The power plants considered in this study are single engine, in-line turbojets with the turbomachinery shaft passing through the reactor, which is the tubular type, liquid moderated, with metallic fuel elements. The core is surrounded by a beryllium reflector. The moderator liquid is cooled by a liquid-to-air radiator system.

The studies indicated that pilotless aircraft may readily use direct-cycle nuclear propulsion systems designed for piloted aircraft. However, much better performance can probably be achieved by power plants designed specifically for missiles.

4.2.2 NUCLEAR SNARK MISSILE

4.2.2.1 Introduction

An advanced design study was made in early 1958 to determine the performance of various nuclear versions of the Northrop Snark missile.[9, 10, 11, 12]

The chemically powered Snark is a subsonic (Mach 0.9 at 30,000 feet) turbojet missile with a gross weight of 39,000 pounds, including the boost system. An outline of a nuclear power plant installed in the missile is shown in Figure 4.8. The missile is approximately 60 feet long; the engine-cavity diameter is 4.5 feet.

The general requirements for the nuclear system were: (1) maximum power-plant weight of 10,000 pounds including control system, engine, and shielding; (2) a speed of Mach 0.9 at 30,000 feet; and (3) endurance of up to 200 hours of uninterrupted design-point operation by prototype models.

[redacted]

[redacted]

TABLE 4.7

PERFORMANCE SUMMARY

Missile	Gross Weight, lb	Turbine Inlet Temperature, °F	At Maximum Speed: Altitude, ft	At Maximum Speed: Speed, Mach No.	At Cruise Ceiling[a]: Altitude, ft	At Cruise Ceiling[a]: Maximum Speed, Mach No.
Nuclear Power Only						
ACA-7	36,000	1700	37,000	2.08	46,000	1.70
		1850	37,000	2.30	49,000	1.80
ACA-9	40,500	1700	38,000	2.17	47,000	1.70
		1850	38,000	2.31	50,000	1.80
ACA-9B	50,000	1700	35,000	1.90	40,000	1.50
		1850	35,000	2.14	44,000	1.70
ACA-8	41,000	1700	37,000	1.70	43,500	1.40
	50,000	1700	35,000	1.57	40,000	1.40
	60,000	1700	32,500	1.48	34,500	1.40
	41,000	1850	38,000	2.00	45,500	1.70
	50,000	1850	36,000	1.97	45,500	1.70
	60,000	1850	35,000	1.84	39,500	1.70
Nuclear Power plus Chemical Reheat to 3200°F						
ACA-8	41,000	1700	45,000	3.60	63,000	2.40
	50,000	1700	45,000	3.60	57,000	2.50
	60,000	1700	45,000	3.56	55,000	2.60

[a]Selected 1000 feet below absolute ceiling.

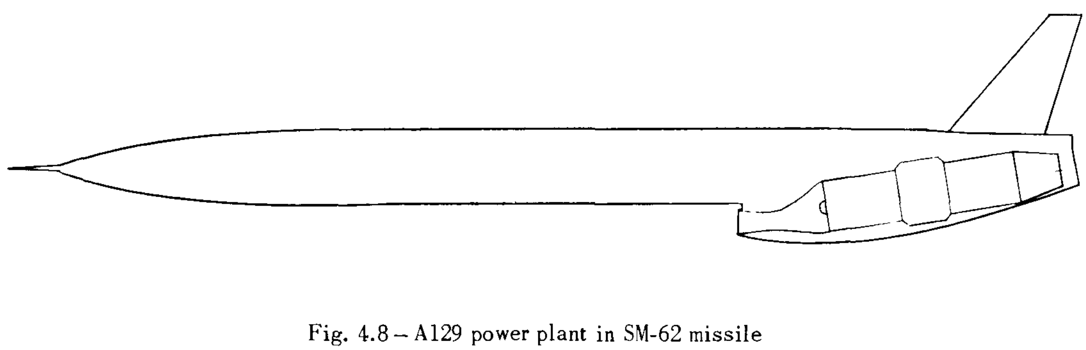

Fig. 4.8 – A129 power plant in SM-62 missile

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[redacted]

In accomplishing the study, three types of reactors and two power-plant arrangements were considered. The types of direct-cycle reactor cores studied were: (1) liquid moderator with metallic fuel elements; (2) metallic moderator with metallic fuel elements; and (3) ceramic moderator with ceramic fuel elements. The types are listed in descending scale of experience and development and ascending scale of sophistication and ultimate level of performance and therefore represent a choice of shorter development time or higher performance. The power-plant arrangements studied were the offset and the in-line. The offset is shown in Figures 4. 9 and 4. 10, and the in-line in Figure 4. 11.

4. 2. 2. 2 Offset Power Plants

The primary advantage of the offset power-plant arrangement over the in-line is that the reactor core can be built without a hole through the center. The core is thus smaller by the amount of the hole area and has better nuclear characteristics because there is less poison and lower neutron leakage. The smaller core is also lighter because less moderator and radial reflector materials are required and because the pressure shell in the vicinity of the engine shaft is less complicated. Another advantage that may be very important from the standpoint of development time is the use of conventional control rods protruding from the compressor end of the reactor and extending out over the compressor itself. Still another advantage is that compressor bleed air is not required for engine-shaft cooling; except during static operation, ram air could probably be used for this purpose.

The main disadvantage of the offset design is in its structural characteristics. Since the reactor and engine (and therefore airflow passages) are not in line, provisions must be made for ducting the air from the compressor around the shaft to the core, and then from the core around the shaft into the turbine. Although experience has been gained in building collector scrolls for this purpose, it is not easy to attain sufficient structural strength in an aerodynamically clean system of flight weight. Another disadvantage is that the offset arrangement requires more space than the in-line.

4. 2. 2. 3 In-Line Power Plants

The principal advantage of the in-line arrangement is the ease of ducting air from and to the reactor while maintaining structural integrity and minimum weight. Although the reactor must be somewhat larger, as discussed above, a slight increase in pressure ratio is also possible, thus achieving a slight thrust increase.

A disadvantage of this system is that conventional control rods cannot be used; a radial reflector-mounted control element may be required. This may not constitute a difficult development problem if a high degree of accuracy is not required to maintain power distribution. Further development and sophistication of the system, particularly better control of power distribution, would require more development time and effort.

The necessity of maintaining constant flat power in certain reactors may be relaxed for prototype power plants. The design assumptions used for the ceramic core are relatively conservative, notably the maximum fuel element surface temperature, thereby enabling some relaxation of the usual rigid restraints on control of power distribution. This, in turn, permits the use of a relatively simple control system. If the higher performance of a more nearly optimum design system is desired, a more sophisticated control system is required. The required degree of sophistication of control system may be the main performance restraint.

Another disadvantage of this system, although probably not very serious, is the requirement for shaft cooling. Secondary, or gamma heating, makes it necessary to provide approximately 4 percent bleed air from the compressor to be used in this power plant.

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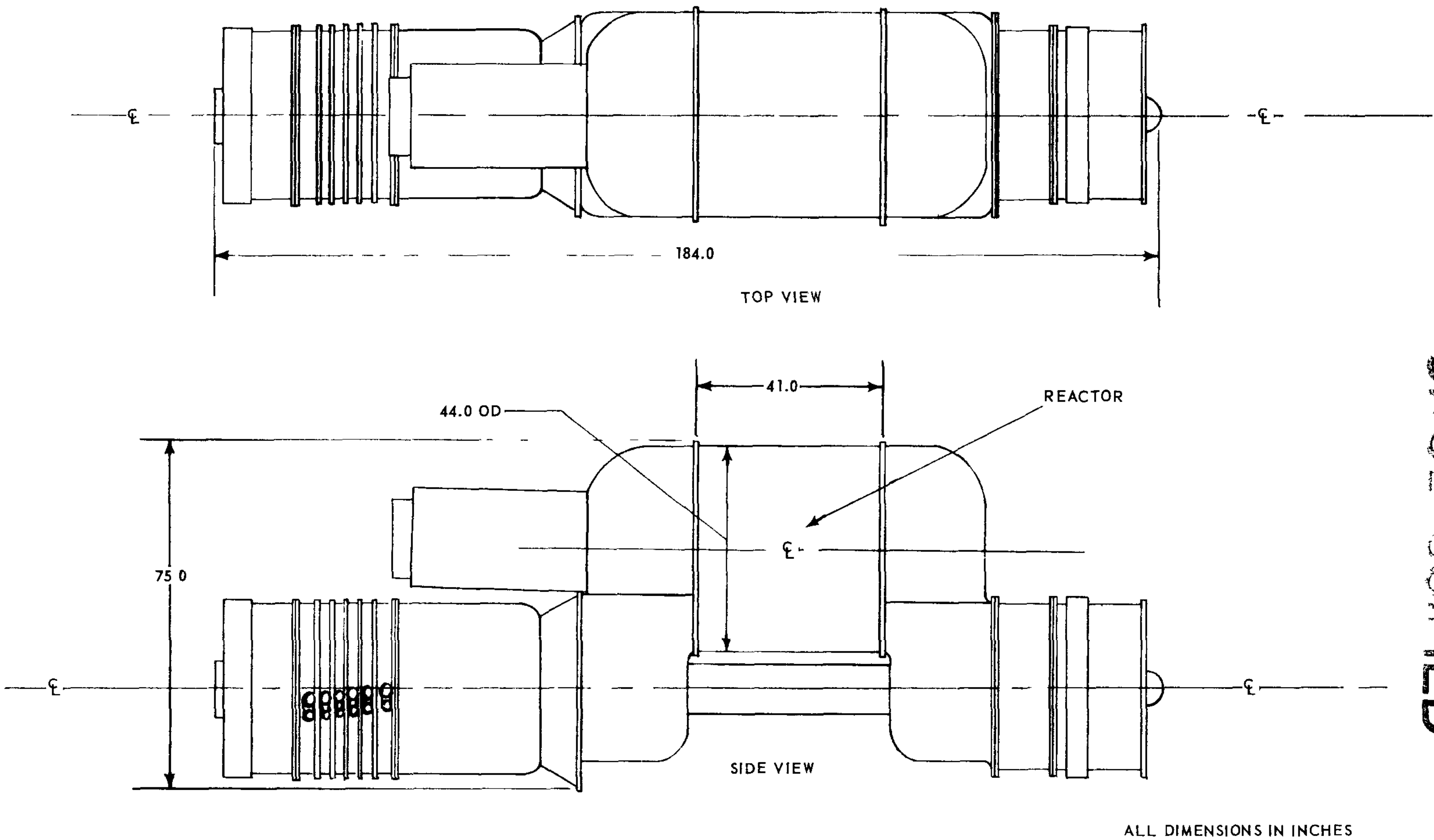

Fig. 4.9 -- Offset H_2O-moderated reactor with modified J79X207 engine

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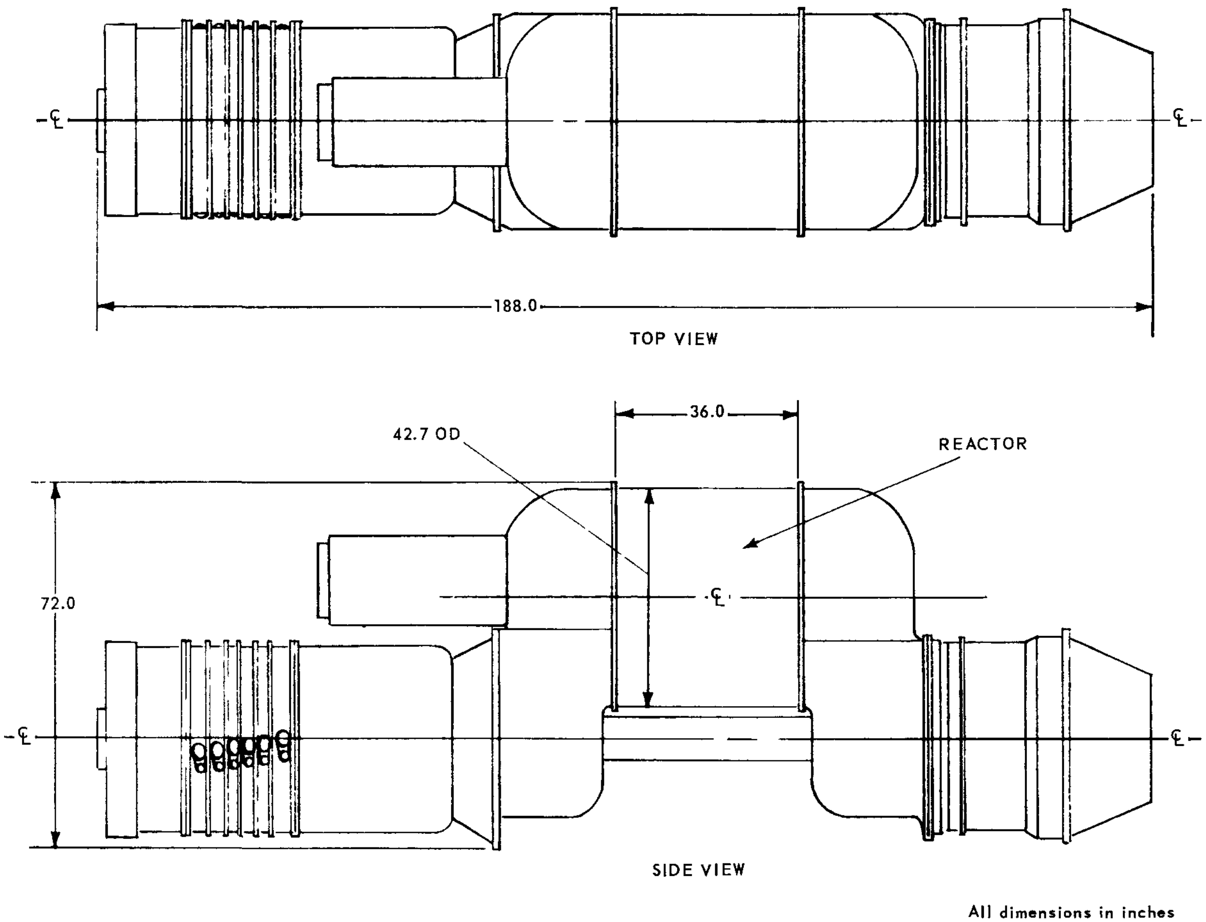

Fig. 4.10 – Offset ceramic reactor with modified J79X207 engine

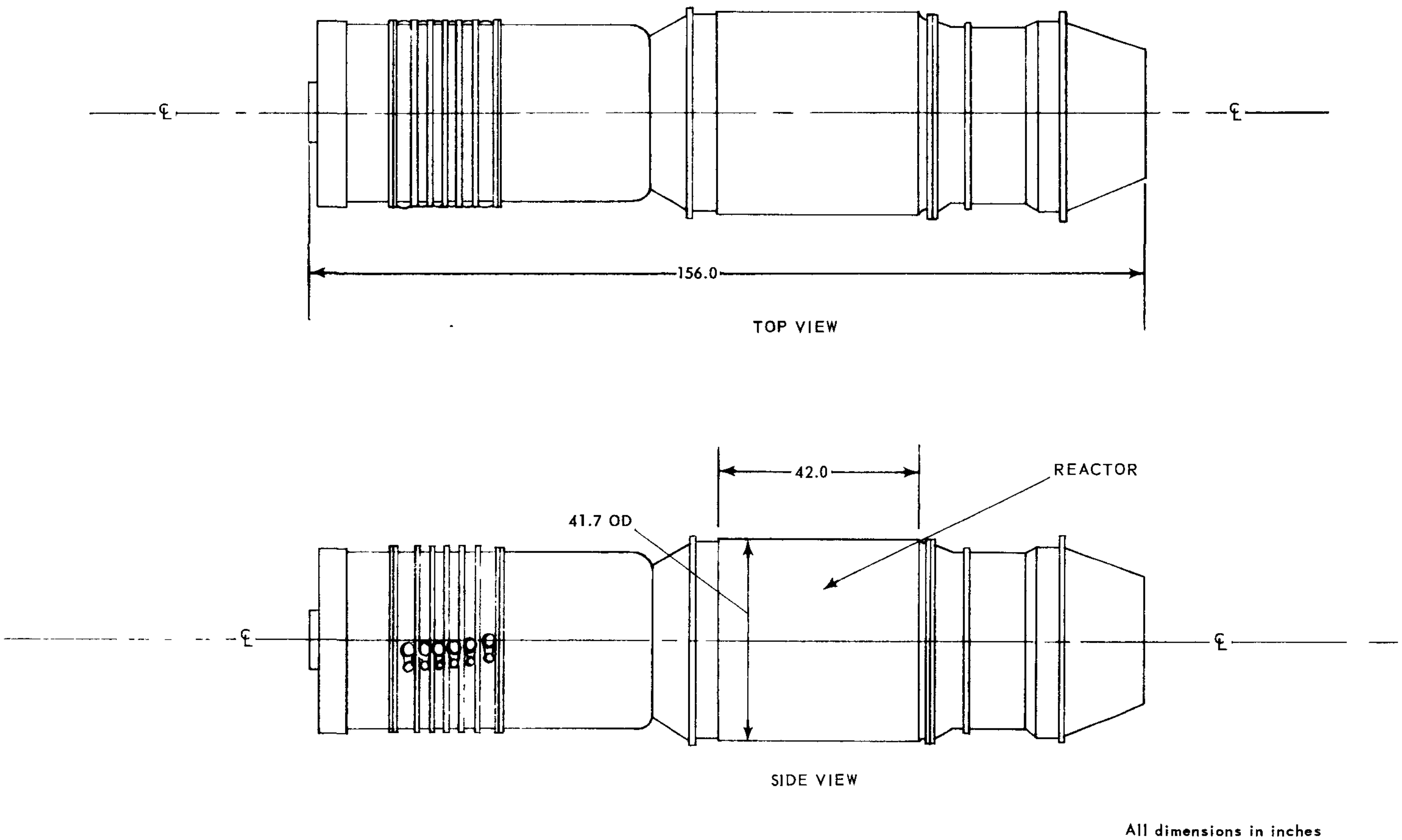

Fig. 4.11 – In-line hydrided zirconium-moderated reactor with modified J79X207 engine

4.2.2.4 Water-Moderated Core With Metallic Fuel Elements

The water-moderated core, while probably not an optimum design for turbojet applications, was considered because it represents a potential reduction in power-plant development time. The core is composed of 37 tubes arranged in a hexagonal pattern with radially varying spacing. Each of the tubes contains a fuel cartridge and is lined with a thin steel-jacketed insulation blanket. There are also 23 control rod guide tubes in the tank which locate the control rods in their 30-inch penetration through the core. The maximum distance across the corners of the lattice is 35.5 inches and the length of the active portion is 29.125 inches. The maximum distance is increased 9-1/4 inches with the inclusion of the beryllium radial reflector, plus approximately 2 inches for the aluminum-core-tank wall spacing and thickness, increasing the over-all core diameter to 46.75 inches. The tank contains the moderator water around the 37 aluminum tubes.

The fuel cartridges enclosed in the aluminum tubes are composed of 18 concentric-ring elements in tandem, each of which is composed of 14 to 16 concentric nichrome fuel rings. The total fuel inventory is 90 pounds of 93.5 percent enriched UO_2. The fuel elements are designed for an operational temperature of 1750°F under the aerodynamic loadings imposed by the specified flight conditions.

4.2.2.5 Solid-Moderated Core With Metallic Fuel Elements

The solid-moderated core uses concentric-ring nichrome fuel elements, basically similar to those of the water-moderated core. They are slightly smaller in over-all and hydraulic diameter and also slightly longer. The moderator consists of circular cells of unclad hydrided zirconium sections that hold the nichrome fuel elements. That the performance of this core is actually no better than the water-moderated core is shown in Table 4.8. However, its advantages lie in the following facts:

TABLE 4.8

CORE PERFORMANCE COMPARISON

		Moderator Material			
		Water	Hydrided Zirconium	Beryllia	Beryllia
Fuel element material		Nichrome	Nichrome	Beryllia	Beryllia
Power-plant arrangement		Offset	In-line	In-line	Offset
Power-plant weight, lb		11,000	10,500	9,200	9,000
Reactor weight, lb		4,310	5,870	4,400	3,230
Reactor O.D., in.		45	41.7	45.5	42.7
Reactor length, in.		41	42	36.7	36
Fuel loading, lb[a]		90	250 (est)	60	50
Turbine inlet temperature, °F		1,400	1,400	1,400-1,500	1,400-1,500
Power to air	SLS, mw	34	34	34-39	34-39
	30,000 ft, Mach 0.9, mw	17	17	17-20	17-20
Power overboard	SLS, mw	3.5	0	0	0
	30,000 ft, Mach 0.9, mw	1.7	0	0	0
Net thrust	SLS, lb	7,100	7,100	7,850-8,600	7,850-8,600
	30,000 ft, Mach 0.9, lb	3,350	3,350	3,500-3,950	3,500-3,950
Fuel burnup	30,000 ft, Mach 0.9, lb/hr x 10^3	1.8	1.8	1.8-2.0	1.8-2.0

[a]For a cold, clean reactor.

1. The solid-moderated power plant needs no external heat exchanger for moderator-heat dissipation.
2. The solid-moderated reactor has future growth potential for substantially higher turbine-inlet temperatures.

Because of the rather small size and the high poison content of the nichrome fuel elements and the engine shaft, the uranium loading is quite high. Detailed multi-group criticality calculations have not been performed on this core specifically, but estimates based on similar larger cores indicate that loadings on the order of 200 to 250 pounds will be necessary. From a practical-application standpoint, this may be excessive. Higher loadings than this may be prohibitive from a fuel-concentration standpoint.

As in the water-moderated system but unlike the all-ceramic system that follows, the nominal target power-plant weight of 10,000 pounds is about as low as practical (Table 4.8). A slight reduction in weight may be possible; however, this would be accomplished at the expense of a large increase in fuel inventory.

4.2.2.6 All-Ceramic Core

The all-ceramic core was the third type of reactor studied. The major material of this semi-homogeneously loaded core is beryllium oxide in the form of a cluster of hexagonal tubes. The tubes are loaded to various fuel concentrations with 93 percent enriched uranium in the form of uranium oxide. Yttrium oxide is used as a stabilizer. Each tube is loaded homogeneously; radial power flattening is accomplished by varying the concentration in different core regions. The final degree of power or temperature flattening is accomplished by a combination of regional variation in fuel loading and variations in the inside diameters of the tubes. The designed maximum surface temperature is intentionally held to a conservative level in the interest of future performance growth as well as to minimize certain development problems.

Although the maximum allowable surface temperature is 2750°F, the design surface temperature is less than 2100°F for an outlet air temperature of 1400°F and less than 2200°F for 1500°F outlet temperature. This low surface temperature is combined with a relatively high design competence factor. Even in the smaller version of this core in the offset power plant, as shown in Table 4.8, the power densities, and therefore thermal stresses, are below current design practice by as much as one order of magnitude.

For the in-line power-plant application, a 13-inch hole is provided through the center of the core for the 8-inch modified turbine shaft, two inches of reflector material, and a steel pressure shell. The core structure for positioning the fuel tubes is outside the actual core and within the shaft hole. The control system for the in-line configuration consists of a series of rotating radial segments which become neutron absorbers when rotated 180 degrees. This design is used since it may be possible to provide sufficient negative reactivity with standard control rods spaced around the periphery of the engine compressor because the core diameter is 33.5 inches and the compressor outside diameter is approximately 36 inches. The reactivity worth of the control system will have to be high to compensate for fuel burnup in this relatively lightly loaded core during its long operating time.

The weight and diameter, and the corresponding fuel inventories, of the ceramic in-line and offset systems represent cores of void volume fraction which appear to be well within the range of feasible and practical nuclear-design values. From a thermal and structural design standpoint, they are considerably but intelligently overdesigned. These cores probably represent the optimum combination of availability, development time, and performance capability. Although the development time for the ceramic reactor would probably be longer than for the water-moderated type, the performance potential is substantially higher.

4.2.2.7 Turbojet Engine

The turbojet engine studied with the reactors described above is the General Electric J79-X207,[13, 14] suitable modified as follows for an external heat supply:

1. Removal of tailpipe variable nozzle, reheat burners, and flame holders.
2. Installation of special short tailpipe.
3. Removal of primary burners and conical compressor shaft.
4. Installation of cylindrical compressor shaft and collector scrolls.
5. Installation of bearing shields.
6. Adjustment of blade schedules for optimum cycle performance.

4.2.2.8 Power-Plant Performance

Thrusts available at Mach 0.9 operation at various altitudes are shown in Figure 4.12. The lowest curves are for both the solid- and the water-moderated systems with metallic fuel elements. The performance of these two systems is about the same from the standpoint of pressure ratio for equal maximum-outlet-air temperatures of 1400°F. The advantage of the ceramic systems over the metallic systems is due to their higher pressure ratios.

4.3 NUCLEAR-POWERED ICBM

In 1954, a conceptual design was made for a two-stage ballistic missile of performance comparable to the data then available for the Atlas ICBM.[15] The missile (shown in Figure 4.13) was designed to propel a 3000-pound warhead 5500 nautical miles. The maximum velocity at power cut-off is 23,000 feet per second.

The first stage of the missile utilizes a nuclear power plant with liquid hydrogen as the working fluid. A conventional chemical rocket power plant powers the second stage. The gasoline and liquid oxygen propellants in the second stage provide radiation shielding for the bomb and guidance equipment.

The reactor is constructed primarily of graphite and uses enriched uranium carbide in tubular fuel elements. The hydrogen propellant flows parallel to the core axis. The reactor operates at a maximum surface temperature of 5000°F and an inlet pressure of 1000 psia. The core pressure drop is 350 psi and the hydrogen exit temperature is 4500°F. The operating time is three minutes at a power of 10,000 megawatts. The U-235 investment is approximately 100 pounds.

The specific impulse of the nuclear stage is estimated to be approximately three times as great as a comparable chemical rocket power plant using gasoline and liquid oxygen. This improvement is due to the low molecular weight of hydrogen compared to the products of chemical combustion. The take-off gross weight of the vehicle is approximately 200,000 pounds, compared to 440,000 pounds for the chemically powered Atlas which was designed for the same range and payload. The nuclear missile is larger in over-all length than the Atlas (187 versus 109 feet) and in maximum diameter (17 versus 12 feet). It also has a higher empty weight for structure and power plant (64,000 versus 22,593 pounds).

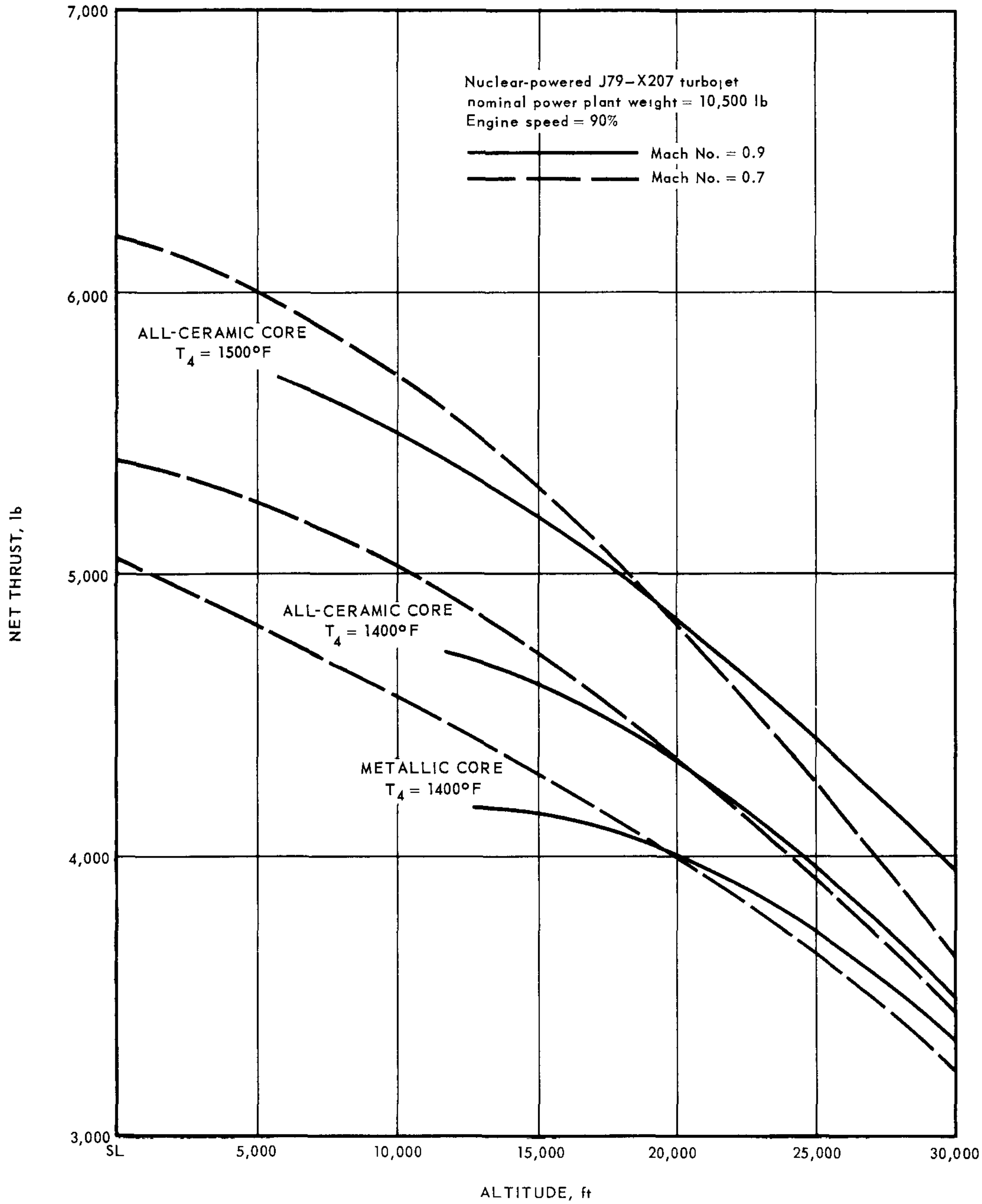

Fig. 4.12 – Variation of thrust with altitude

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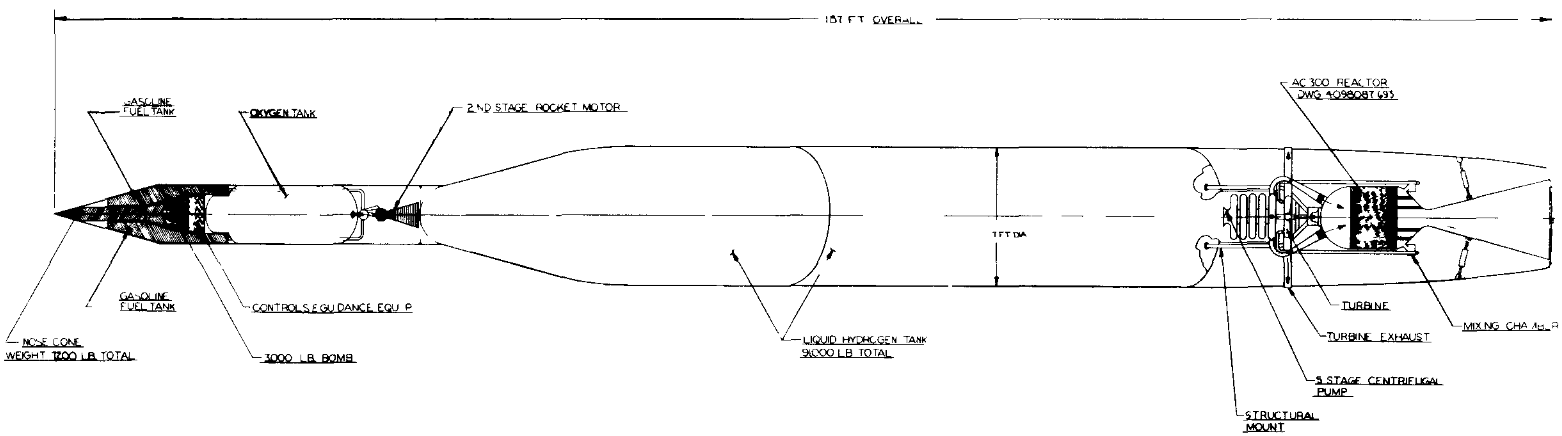

Fig. 4.13 – Nuclear-powered ballistic missile

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5. NUCLEAR ROCKETS

5.1 EARLY STUDIES

5.1.1 COMPARISON OF NUCLEAR AND CHEMICAL SYSTEMS

Two studies were made in 1956 and 1957, to obtain relative trends in power-plant designs for nuclear-rocket propulsion systems, and to compare the economics of nuclear systems and chemical systems.[1,2] The referenced reports summarize a preliminary feasibility study of single-stage nuclear-rocket propulsion systems using liquid hydrogen as the working fluid and capable of accelerating payloads of up to 90,000 pounds to burn-out velocities of approximately 23,500 feet per second. Single-stage nuclear rockets with payload capabilities of 5,000, 10,000, 60,000, and 90,000 pounds are compared with multi-stage chemical rockets designed for the same mission. The chemical systems use the following combinations of propellants: Lox-JP, H_2-O_2, H_2-F_2, and N_2H_4-F_2. (All but the Lox-JP are termed "exotic" propellants.) A comparison of the physical sizes of the four chemical systems and the nuclear system, all based on the 60,000-pound payload size and the same range capabilities, is given in Figures 5. 1 and 5. 2.

The following conclusions were obtained from the studies:

1. Existing nuclear-rocket propulsion technology appears to be restricted to future weapon systems either of extremely high payload capability or requiring very high specific impulses. The studies indicate that nuclear-rocket propulsion cannot compete economically with chemical-rocket propulsion in ICBM vehicles with payloads under 5000 pounds. For payloads of 40,000 to 90,000 pounds, nuclear rockets using liquid hydrogen appear to be economically competitive and permit a reduction in take-off gross weight by a factor of from two to six.
2. The use of exotic propellants enables chemical-rocket systems to compete economically with nuclear-rocket systems at higher payloads than standard propellants.
3. Exotic chemical systems offer altitude-specific impulse values of slightly more than 400 seconds. Nuclear-liquid hydrogen systems are expected to produce altitude specific impulses of more than 800 seconds at existing levels of heat-transfer-reactor technology.

5.1.2 GAS-FISSION ROCKET-PROPULSION SYSTEMS

Steady-flow (continuously discharging powdered U-235) gas-fission, single-stage rocket-propulsion systems were analyzed in 1956[3] to determine the consumption of U-235 fuel in producing 500,000 pounds thrust for 300 seconds. Three of the reactor systems studied used D_2O reflectors and three used graphite. Operating temperatures of from 5000°R to 10^{11}°R were used in the analysis.

The results of the study (Tables 5. 1 and 5. 2) indicate that the D_2O-reflected systems are superior to the graphite-reflected systems in the plasma temperature region from 5,000°R to 10^{6}°R, but neither system is even remotely feasible in this temperature range. For ex-

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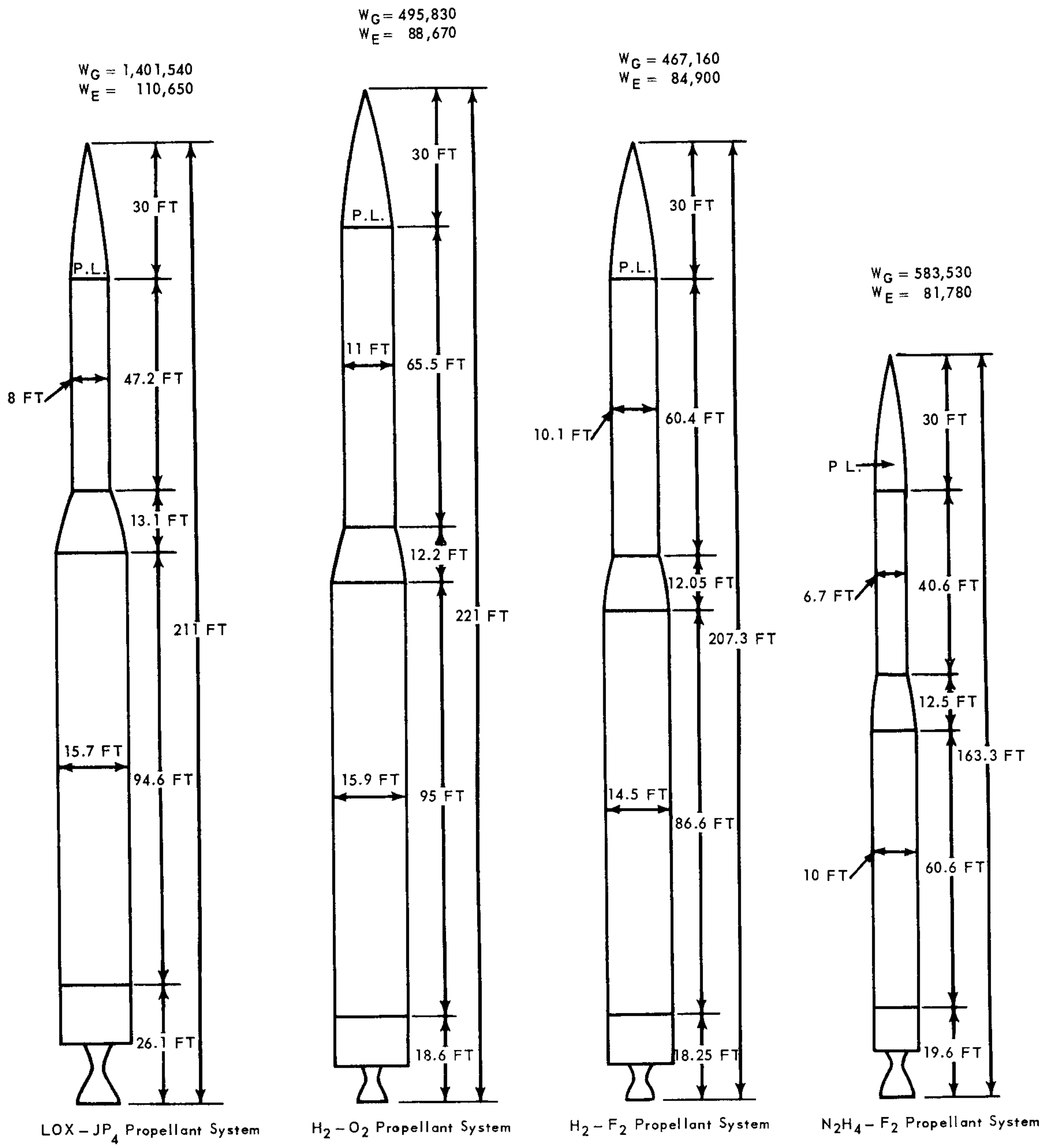

Fig. 5.1 – Advanced chemical systems

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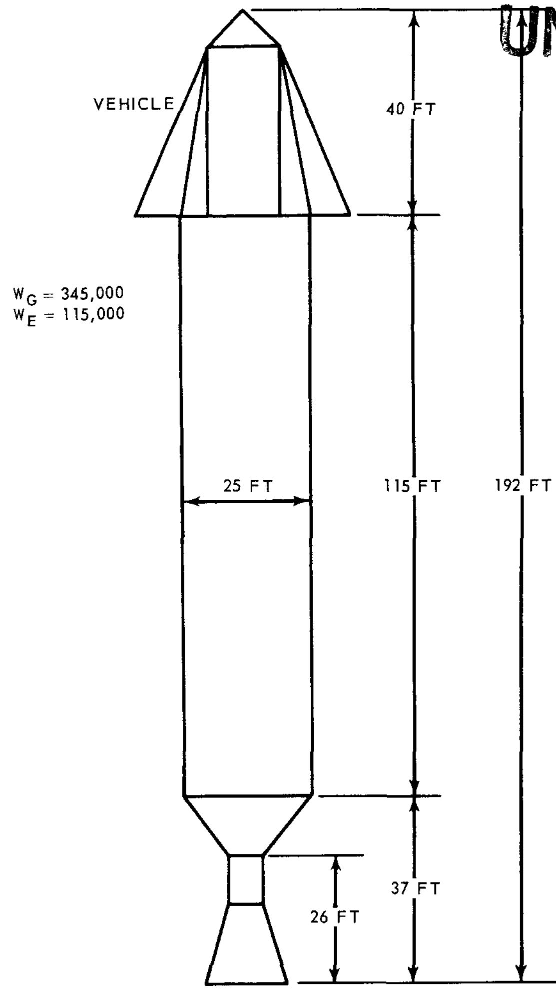

Fig. 5.2—Nuclear-H_2 system for 60,000-pound payload

TABLE 5.1

TOTAL U-235 CONSUMED IN D_2O-REFLECTED SYSTEMS AT VARIOUS TEMPERATURES

Temperature, ^oR	Total U-235 Consumed in Various Systems, lb: System 1 (V_c=19.4 ft^3)	System 2 (V_c=82.4 ft^3)	System 3 (V_c=1241 ft^3)
5×10^3	56.13×10^3	26.68×10^3	6.62×10^3
2×10^4	29.66×10^3	19.44×10^3	6.51×10^3
1×10^5	17.49×10^3	15.36×10^3	9.02×10^3
5×10^5	8.35×10^3	8.11×10^3	6.95×10^3
10^8	0.60×10^3	0.60×10^3	0.60×10^3
10^9	0.19×10^3	0.19×10^3	0.19×10^3
10^{10}	0.06×10^3	0.06×10^3	0.06×10^3
10^{11}	0.019×10^3	0.019×10^3	0.019×10^3

NOTE: V_c = reactor core void volume.

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TABLE 5.2

TOTAL U-235 CONSUMED IN GRAPHITE-REFLECTED SYSTEMS AT VARIOUS TEMPERATURES

Temperature, °R	Total U-235 Consumed in Various Systems, lb		
	System 1 V_c=82.3 ft^3	System 2 V_c=155.1 ft^3	System 3 V_c=1241 ft^3
5×10^3	106.40×10^3	76.91×10^3	27.93×10^3
2×10^4	38.28×10^3	34.05×10^3	20.03×10^3
1×10^5	18.59×10^3	18.10×10^3	15.52×10^3
5×10^5	8.46×10^3	8.42×10^3	8.13×10^3
10^8	0.60×10^3	0.60×10^3	0.60×10^3
10^9	0.19×10^3	0.19×10^3	0.19×10^3
10^{10}	0.06×10^3	0.06×10^3	0.06×10^3
10^{11}	0.019×10^3	0.019×10^3	0.019×10^3

NOTE: V_c = reactor core void volume.

ample, a D_2O system with a reactor void volume of 83 cubic feet requires 23,000 pounds of U-235 for the mission at a plasma temperature of 10,000°R, and 6,500 pounds at a temperature of 10^6°R. A graphite system of similar size requires 82,000 and 38,000 pounds of U-235 for the same mission while operating at plasma temperatures of 10,000° and 10^6°R respectively.

As the plasma temperature becomes higher and higher, the fuel consumption by the D_2O and the graphite systems approaches equality; for plasma temperatures of about 10^{10}°R, gas-fission propulsion systems begin to show first-order feasibility.

Systems designed to retain or recapture the majority of the fuel instead of continuously discharging it with the plasma, would be feasible at much lower temperatures.

If the technology of containment of ultra-high-temperature plasma or retention of some of the fissionable material becomes sufficiently developed, together with the capability to solve the allied problems of handling plasma in a propulsion system, it will then be possible to give serious consideration to the application of gas-fission propulsion systems to future weapons systems.

5.2 ROCKETS FOR SPACE MISSIONS

5.2.1 BACKGROUND

Some studies of nuclear rockets[4, 5, 6, 7, 8] were reported on by GE-ANPD in 1960. One of these reports, reference 7, outlines a method of analyzing the space-mission capabilities of rockets based on propellant and rocket physical characteristics, and is directed primarily at methods of evaluating nuclear rockets using hydrogen as the propellant. Another[8] is an analysis of the effects of engine weight, propellant flow rate, impulse times, and power levels for three rocket space missions: (1) interplanetary probe; (2) manned lunar scanning mission; and (3) large booster. In addition, information is given on flight time, aftercooling problems, and other operational considerations involving reactor-design requirements for these missions.

5.2.2 TRAJECTORY PROGRAM DEVELOPMENT

IBM 7090 Digital Computer Program No. 686 was developed for more thorough treatment of trajectory studies. The program was designed initially for use with multi-stage rocket

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vehicles. A spherical-central-force-body treatment is used. This body can be rotated at any velocity, and gravity can be varied to simulate the earth, moon, or other planets of the solar system as the central force field. Atmospheric drag is accounted for. Trajectory output data giving location, velocity, vehicle weight, etc., are printed out at specified intervals. The thrust vector can be varied with respect to the velocity vector. Thus, parametric trajectory data for multi-stage rocket vehicles can be generated from this program.

The program was also modified for the study of trajectories of maneuverable orbital vehicles by starting the vehicle in some pre-established orbit and applying a defined thrust program. In addition to the trajectory output data described above, the program also calculates the elements of the elliptical transfer path of the vehicle during specified increments of the thrust period. This permits parametric study of the effects of thrust direction and thrust level to achieve specified orbital changes.

5.3 NERVA-PHOEBUS STUDIES

In early 1961, intensive application studies of nuclear rockets were carried out by GE-ANPD in conjunction with the Company's Flight Propulsion Laboratory Department. The purpose was to determine the flight capability and potential problem areas of the proposed 1500-megawatt NERVA engine and the usefulness of growth versions of up to 5000 megawatts. The Saturn chemical S-I and S-II were used as the first and second stages and the nuclear engine for third-stage propulsion.

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6. NUCLEAR-POWERED HELICOPTERS

6.1 INTRODUCTION

A study of the feasibility of flying-crane helicopters using nuclear power plants was performed under a subcontract to the Aircraft Division of the Hughes Tool Company in May 1959.[1] The study was based on requirements established by GE-ANPD as follows:[2, 3]

1. The systems investigated were to be jet driven to eliminate the complexity of shaft drives.
2. The power plants were to be of the open-cycle type.
3. The systems were required to operate satisfactorily at 95^{o}F, 6000 feet altitude, and static-flight conditions.
4. Existing technology was to be used in all components and materials.

Power-plant cycle and weight data were supplied by GE-ANPD. Hughes provided the aircraft data and performed the system analysis.

6.2 POWER PLANTS

Power-plant cycle data, turbomachinery weights, and reactor-shield assembly weights were supplied in parametric form as functions of compressor pressure ratio and turbine-inlet temperature for three types of cycles:

1. Hot cycle, in which the rotor is driven by turbine discharge air from a conventional open-cycle turbojet.
2. Cold cycle, where the rotor is driven by discharge air from a fan operated by a conventional open-cycle turbojet.
3. Mixed cycle, a combination of cold and hot cycles, in which the rotor is driven by a mixture of fan air and turbine discharge, at pressures comparable to hot-cycle pressures and temperatures intermediate to those of the cold and hot cycles.

The three cycles are shown schematically in Figures 6. 1, 6. 2, and 6. 3. Cycle, performance, and weight data for the hot cycle are documented in references 4, 5, 6, and 7. These same data for the cold cycle are given in reference 8. The mixed cycle data are a composite of the hot and cold data. Details of component performance assumptions, materials, and limitations are contained in the preface of reference 1.

6.3 SYSTEM ANALYSIS

The basic criterion used in the system analysis was the ratio of payload to gross weight. To determine the various values of this ratio, hovering aerodynamics and propulsion requirements were first considered, using the parametric power-plant information, to plot

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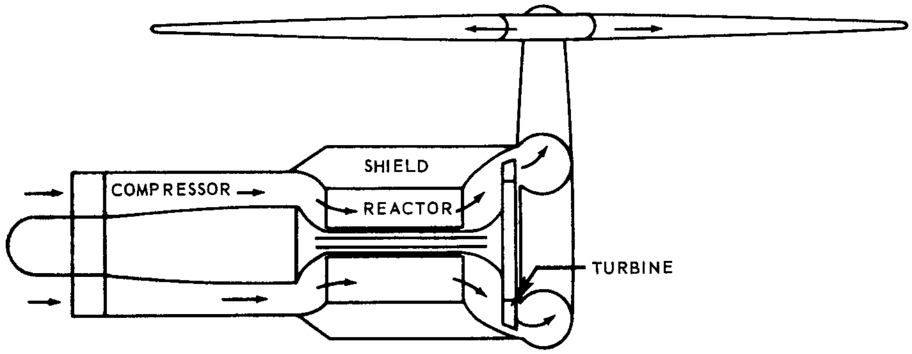

Fig. 6.1 – Hot-cycle nuclear helicopter system

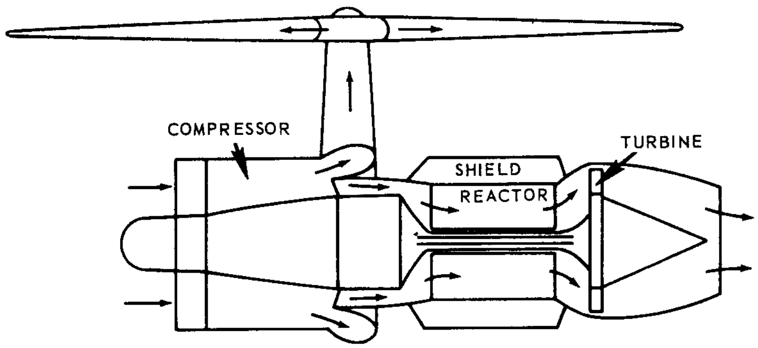

Fig. 6.2 – Cold-cycle nuclear helicopter system

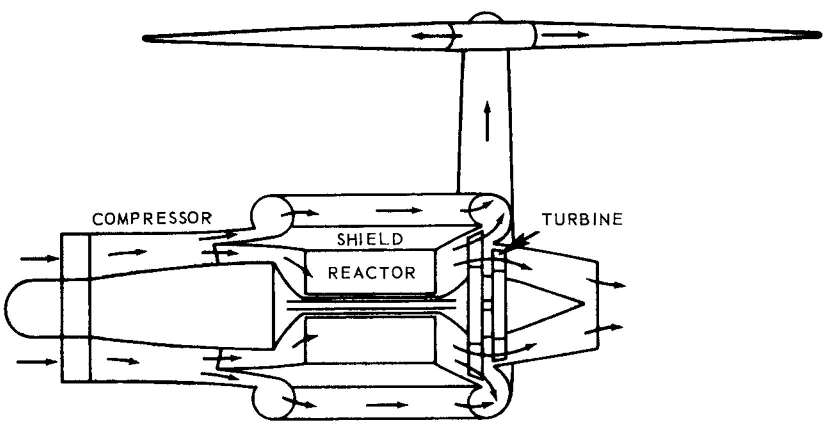

Fig. 6.3 – Mixed-cycle nuclear helicopter system

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other ratios: shield plus payload to gross weight and power plant plus payload to gross weight. Final plots of payload to gross weight were extracted from these for a range of gross weights and payloads as affected by variations in radiation level in both manned and unmanned vehicles. All of the assumptions, calculations, and system analyses of the nuclear helicopter are included in reference 1.

6.4 CONCLUSIONS

The following results were derived from the study:

1. For a nuclear helicopter of 100, 000 pounds gross weight, hovering at 6000 feet at 95°F, with a power-plant dose rate of 25 rem per hour at 50 feet:
   a. The maximum hot-cycle payload-to-gross-weight ratio was 0. 03.
   b. No cold-cycle payload-to-gross-weight ratio was found for any combination of power-plant or helicopter parameters.
   c. A payload-to-gross-weight ratio of 0. 115 was found for the mixed cycle. However, the power-plant weights used in this analysis were optimistic (they were obtained from simplifying assumptions rather than from analysis).
2. Assuming the mixed cycle was the best of the tip jet drives, the maximum payload-to-gross-weight ratio was 0. 12 at 114, 000 pounds gross weight. At 195, 000 pounds gross weight, the ratio became zero because the empty-weight-to-gross-weight ratio increased drastically above a gross weight of 100, 000 pounds.
3. Reducing the dose rate from 25 to 2. 5 rem per hour at 50 feet increased the shielding-weight-to-gross-weight ratio so much that even the mixed cycle would have no net payload.

Based on the foregoing results, helicopters powered by all-nuclear power plants using the direct-cycle reactor did not appear feasible at the time of the study for:

1. The hot-cycle system as currently being developed.
2. The hot-cycle system using advanced materials or insulated blades.
3. The cold-cycle system.

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7. HYDROFOIL PROPULSION

7.1 INTRODUCTION

A study of a closed-gas-cycle nuclear power plant to power a hydrofoil craft was completed in August 1958.[1] The only two requirements specified for the hydrofoil craft were a vessel displacement of 200 long tons and a maximum speed of 100 knots. No information was provided on hull design or mission, and no data have been released or are available for hydrodynamic analysis at the speed specified for the vessel. The draft, clearances, and propeller design used in the study are based on data presented in References 2, 3, and 4. The power requirements of the ship were determined from extrapolation of experimental data presented in References 5 and 6.

The draft and clearances selected for the study are shown in Figure 7.1. The hull was designed for five-foot submergence at rest and at least two-foot clearance above water at full speed. The propeller and foils were required to be submerged at least five feet. With a four-foot-diameter propeller, the draft at rest would be 16 feet or more.

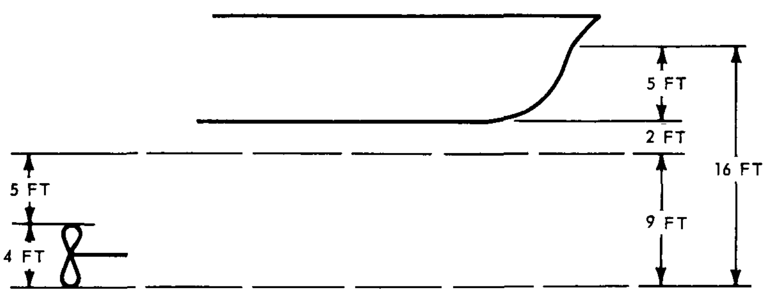

Fig. 7.1 — Draft and clearances for hydrofoil study

It was found that a three-bladed propeller, four feet in diameter, with a four-foot pitch and a rotational speed of 1000 revolutions per minute would satisfy the requirements.

The power plant could be mounted either in the hull or in submerged pods. It was predicted from the experimental data presented in Reference 6 that the shaft power required by the propeller is 26,000 horsepower. The lift-drag ratio of the vessel is 5.5. Using a lift coefficient (C_L) of 0.42, the foil area was calculated to be 36.3 square feet to plane at 100 knots. If the power plant is pod-mounted, calculations revealed that the power required to overcome the basic drag of the submerged power pod is 5,000 to 7,000 horsepower for a streamlined design equivalent to a cylinder 4 feet in diameter and 18 feet long with spherical heads.

The nuclear power plant proposed for this study is a simple closed-gas-turbine cycle, using neon as the working fluid. Neon was selected because it is inert, requires smaller rotating machinery than helium, and can be replaced by air at a lower temperature in an emergency. The major items of power-plant equipment are located in the following sequence along the power shaft:

1. Control actuators
2. Reactor
3. Turbine
4. Aftercooler, wrapped around the compressor drive shaft
5. Compressor
6. Power shaft with starting motor and jacking gear take-off
7. Speed reducer
8. Clutch or fluid coupling
9. Propeller shaft
10. Shaft seal at hull or pod penetration

A schematic diagram of the power-plant cycle with the design-point temperatures and pressures indicated is shown in Figure 7.2. The 1400-pound, twelve-stage compressor has a design pressure ratio of 5 and a design efficiency of 85 percent. Its maximum diameter is 24 inches. The efficiency of the 500-pound turbine is 88 percent, and the maximum diameter is 30 inches. The neon flows through the power plant at 200 pounds per second. Heat is rejected from the cycle in the aftercooler to sea water at 59°F (density = 64 pounds per cubic foot).

A reactor power of 65.16 megawatts is required to raise the temperature of the neon to 1940°F at the turbine inlet. The reactor uses yttrium - zirconium hydride for the moderator and niobium fuel elements. The reactor design is limited by nuclear considerations and fuel concentration rather than pressure drop. The maximum average fuel element temperature is 2300°F, the maximum average moderator temperature is 2000°F, the core diameter is 31.7 inches, the length is 32 inches, and the weight is 10,400 pounds.

The equipment designs and weights are based on aircraft design philosophy. The weights of the power-plant components on the power shaft are shown in Table 7.1.

The shield design is based on a continuous exposure to 3.6 millirem per hour, determined as follows:

1. 300 millirem per week = 1.8 millirem per hour.
2. Weekly operation at an average power level over 50 percent of full power is very unlikely.

Reactor shields were investigated for hull-installation and for power pods. In the power-pod application, Hevimet was used as the gamma shield and water as the neutron shield. The weight of the shield for a single reactor in a submerged pod was established at 26,800 pounds. Hevimet was also used as the gamma shield for the in-hull configuration, and lithium hydride was specified for the neutron shield. The total shield weight for each in-hull reactor is 82,000 pounds. These shield weights for both configurations are 20 percent lower than the calculated weights, on the assumption that a weight saving can be achieved by power shaping.

7.2 CONCLUSIONS

A summation of the power-plant component weights for both the in-hull and pod-mounted configurations is presented in Table 7.2.

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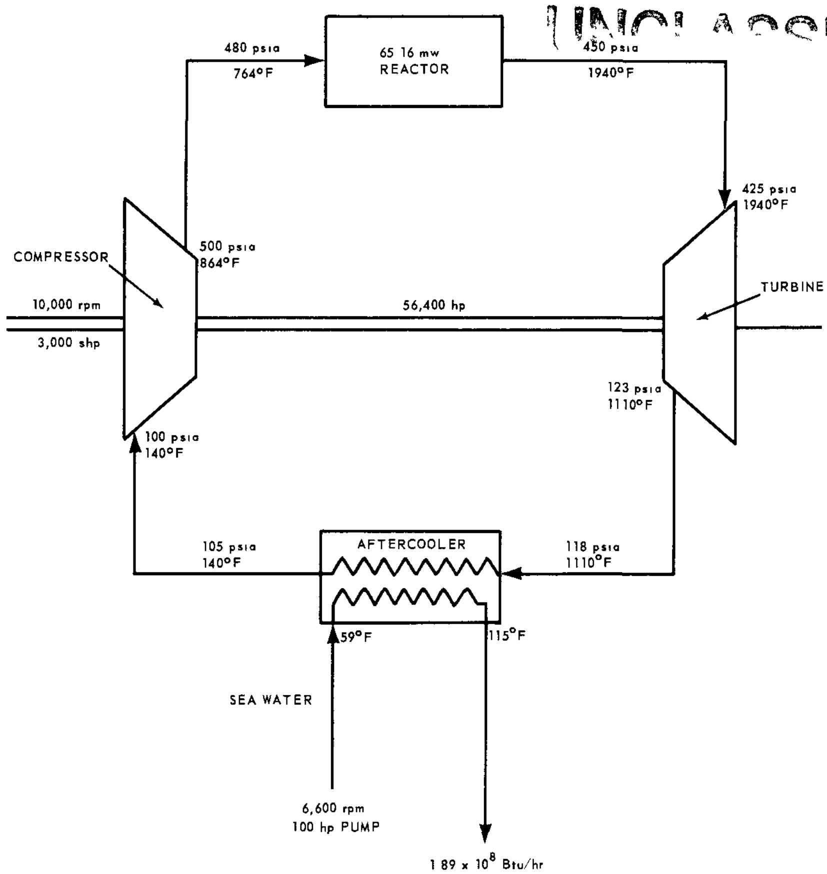

Fig. 7.2—Schematic of closed-neon-cycle hydrofoil power plant

TABLE 7.1

WEIGHTS OF POWER-PLANT COMPONENTS

Component	Weight, lb
Power cycle	
Reactor	10,400
Turbine	500
Compressor	1,400
Aftercooler	3,810
Total power cycle	16,110
Ducts (10% of power-cycle weight)	1,610
Tube system for power cycle	200
Reduction gear	2,500
Torque converter	1,500
Starting motor	500
Circulating pump	600
Miscellaneous controls, piping, wiring	1,300
Total system weight	24,320

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TABLE 7.2

WEIGHT SUMMARY OF POWER-PLANT COMPONENTS

	Weight, lb		
Component	In-Hull Configuration	Pod-Mounted Configuration	
Power shaft	24,300	24,300	24,300
Pod	-	3,400	3,400
Shield	82,000	26,800	26,800
Total weight per power plant	106,300	54,500	54,500
Number of power plants	2	3	4
Total power-plant weight	212,600	163,500	218,000
	- 3,000[a]		
	209,600		

[a]Weight saving anticipated by combining auxiliary equipment and because of the greater accessibility for maintenance which reduces the required component reliability for the in-hull design.

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8. ARMY PORTABLE NUCLEAR POWER PLANT

8.1 INTRODUCTION

A study was made in mid-1960 to determine the feasibility of applying HTRE No. 3 technology to an Army portable nuclear power plant.[1] The reactor and all other components of the long-life power plant were to be based entirely on current technology. Therefore, nichrome fuel elements and zirconium hydride moderator were used in the reactor design.

8.2 TURBOMACHINERY

Preliminary investigations of turbomachinery indicated that a larger engine than the T58 and T64 is desirable, both from the standpoint of the availability of developed industrial versions of engines, and cost per kilowatt of output. The rating of the T58 version would be slightly under 500 kilowatts and the T64 version slightly under 1000 kilowatts. The MS240 turbomachinery, a marine or industrial adaptation of the J79 engine, was selected.

The MS240 turbomachinery uses a CJ-805 gas generator and a power turbine. Minor modifications are necessary, such as a change in the compressor stator-angle schedule, and a 10 percent increase in power-turbine nozzle area. Compressor and turbine scrolls are substituted for the chemical combustion system, similar to the X39 version of the J47 engine described in APEX-904 of this Report. Standard reduction gears and electrical and lubricating equipment are used.

8.3 SCOPE OF THE STUDY

Since the maximum turbine-inlet temperature that can be maintained throughout long reactor life is uncertain, two such temperatures (1300°F and 1350°F) and two compressor-to-turbine pressure drop values (0.8 and 0.85) were used in the study. Electrical-power-output data were prepared for these values and are presented as functions of compressor-inlet temperature in Figure 8.1. The altitude was assumed to be 1000 feet.

8.4 GENERATOR SELECTION

In the choice of a generator size, the significance of the rating should be considered. It is common practice to rate generators in terms of kilovolt-amperes (KVA). The actual output in kilowatts is a function of the power factor. In the portable nuclear power-plant application, a power factor of 0.8 was considered to be typical. Thus, if a power output of 6000 kilowatts is required at a power factor of 0.8, a generator rating of 7500 KVA should be

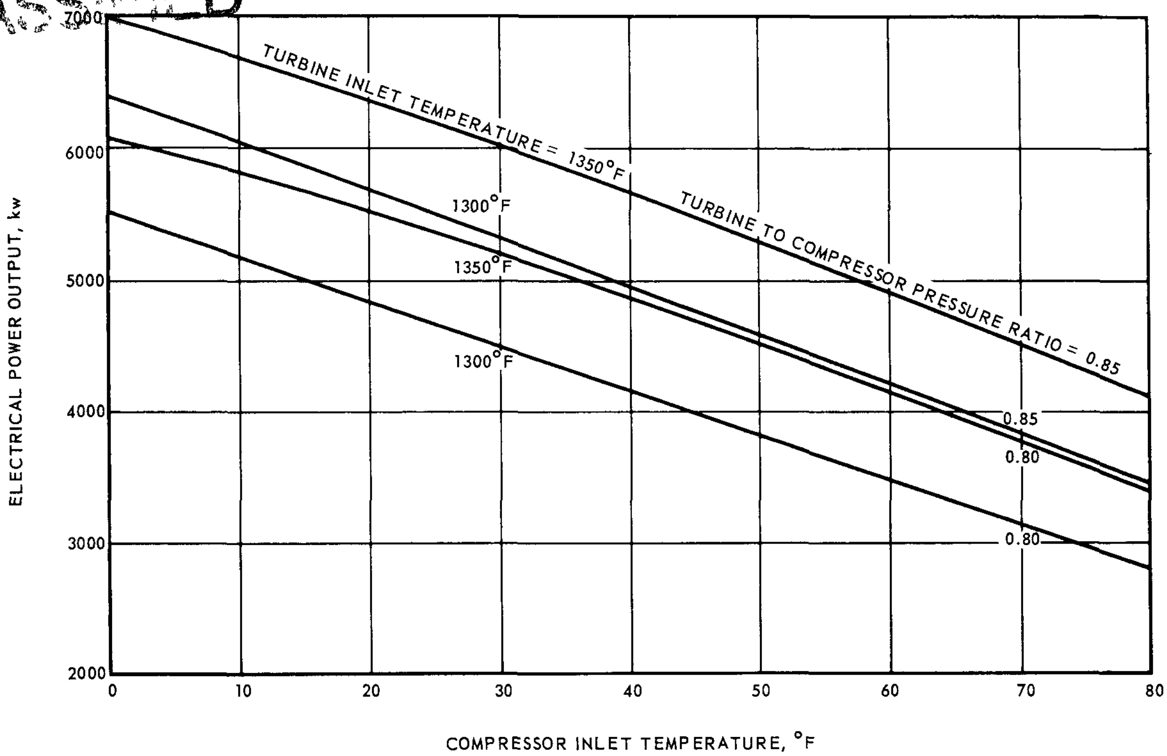

Fig. 8.1—Portable nuclear-power-plant performance

specified (such a generator actually delivers 7500 kilowatts at unity power factor). For a power plant designed to operate without derating over a compressor-inlet temperature range from 0° to 80°F, a 5000 KVA generator could be adequate. If low-temperature applications are important, the low-temperature power-output capability of the turbomachinery can readily be used to drive a 7500 KVA generator.

8.5 CONCLUSIONS

A possible schematic layout of the power plant is shown in Figure 8.2. This layout resembles very closely the "Red Truck" version of the MS240 power plant. Estimated component weights are tabulated in Table 8.1 for several generator ratings. Although at a 7500-KVA rating a 1200-rpm generator weighs considerably less than a 3600-rpm generator, this advantage is largely offset by the heavier gear required by the 1200-rpm generator.

The continuous operating life of the reactor was to be 10,000 hours. The life of the power plant may be limited by fretting, corrosion, erosion, and fatigue in the turbomachinery. Because the gas generator operates at about 90 percent of rated speed in this application, and the power turbine operates at less than 80 percent of rated speed, the thermal capability of the turbomachinery is in excess of 10,000 hours.

Subsequent work indicated that the compressor-to-turbine pressure ratio would be about 0.84 for a power plant in which the reactor was immediately adjacent to the turbine and 0.825 if there was a 15-foot duct between the reactor and the turbine. Since it may be desirable to perform inspections and minor maintenance of the turbomachinery, the wider separation between the reactor and the turbomachinery might be desirable. Therefore, the more conservative value, 0.825, was assumed. Under the assumptions of a turbine-

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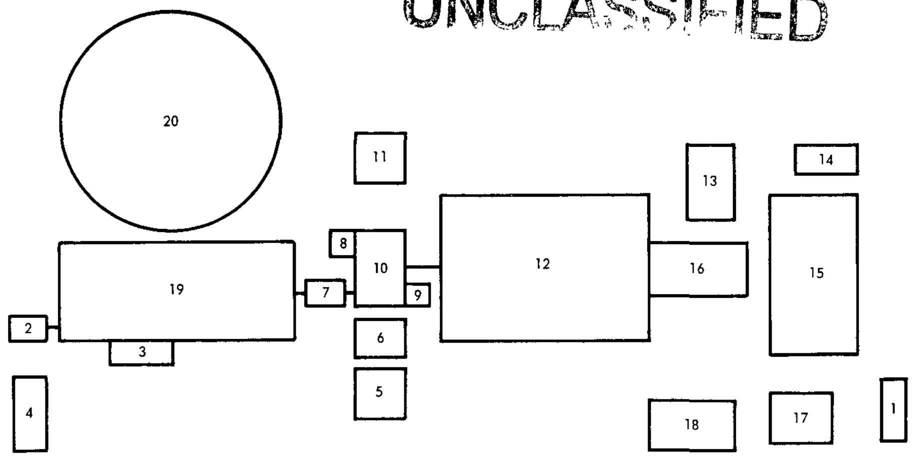

1 – Outlet panel
2 – Air turbine starter
3 – Turbomachinery oil tank
4 – Gas turbine air starter set
5 – Air-oil heat exchanger
6 – Main lube tank
7 – Flexible coupling
8 – Control alternator
9 – Lube pump
10 – Reduction gear set
11 – Electric blower for air-oil heat exchanger
12 – Generator
13 – Neutral ground equipment
14 – Output bushings
15 – Generator and exciter equipment
16 – Field exciter
17 – Auxiliary power equipment
18 – Power plant control panel
19 – Turbomachinery
20 – Reactor shield assembly

Fig. 8.2 – Portable nuclear-power-plant-schematic layout

TABLE 8.1

PORTABLE NUCLEAR POWER-PLANT WEIGHTS

Generator Rating, KVA	3,750	5,000	7,500	7,500	9,375
Generator Speed, rpm	3,600	3,600	1,200	3,600	3,600
Gas generator	2,980	2,980	2,980	2,980	2,980
Power turbine and bearing	1,588	1,588	1,588	1,588	1,588
Exhaust system	883	883	883	883	883
Scrolls and ducting	1,500	1,500	1,500	1,500	1,500
Reactor-shield assembly	120,000	120,000	120,000	120,000	120,000
Prime mover auxiliaries	1,640	1,640	1,640	1,640	1,640
Gas turbine starter set	550	550	550	550	550
Reduction gear and lubrication pump	4,500	8,000	17,000	8,000	10,000
Lubrication auxiliaries	2,040	2,040	2,040	2,040	2,040
Generator and exciter	32,000	39,000	41,820	53,900	66,000
Electrical auxiliaries	8,650	8,650	8,650	8,650	8,650
Control panel	500	500	500	500	500
Mounting base	10,000	10,000	10,000	10,000	10,000
Totals	186,831	197,331	209,151	212,231	226,331

NOTE: All weights are in pounds. This tabulation is based on a 31.8-inch core with complete shielding.

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inlet temperature of 1350°F, a pressure ratio of 0.825, and an altitude of 1000 feet, the estimated variation in power with ambient temperature is shown in Figure 8.3.

The size of the generator for this power plant depends on whether it is to be used in the tropics or the arctic. If a power factor of 0.8 is used, a 5000-KVA generator is probably suitable for the tropics. A 7500-KVA generator would match the power-plant capability down to an ambient temperature of 20°F, as shown in Table 8.2. If the additional 550 kw obtainable at 0°F is desired, then it is necessary to use a 9375-KVA generator.

The fuel inventory for this power plant has been estimated at 200 pounds of highly enriched U-235. During a 10,000-hour core life, about 35 pounds of burnup would occur for the maximum low-ambient-temperature rating of 6550 kilowatts at a 100-percent load factor. For a 3750-kilowatt output continuously for 10,000 hours, the burnup would be about 25 pounds, or from 12.3 to 17.5 percent. Actually, an 80-percent load factor is probably more realistic, in which case the burnup would vary from about 10 to 14 percent.

From these results, it appears that the direct-air-cycle system offers weight savings to the Army in portable power-plant applications.

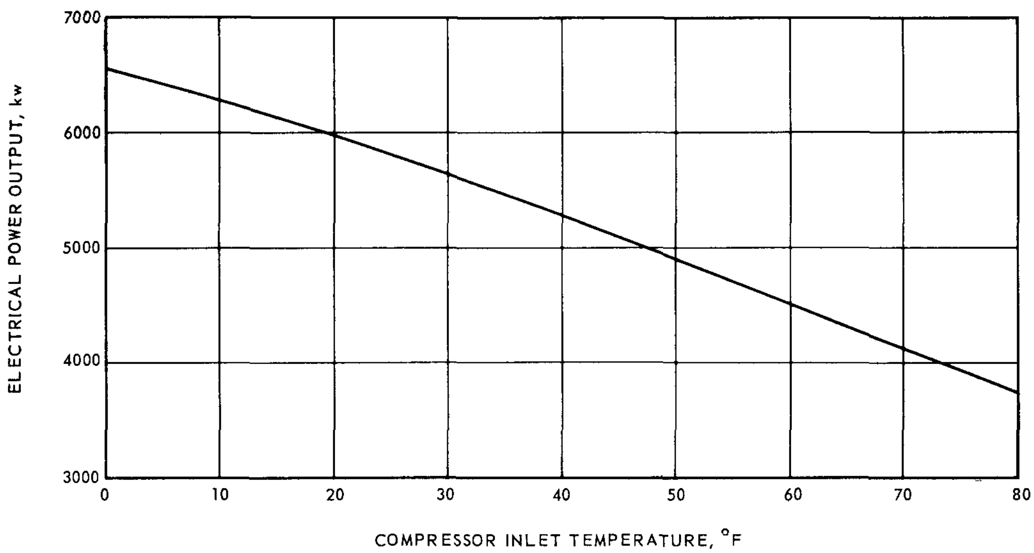

Fig. 8.3 – Portable-power-plant power as a function of ambient temperature

TABLE 8.2

GENERATOR SIZE

Ambient Temperature, °F	Generator Size, KVA	Power Produced at 0.8 Power Factor, kw
0	9,375	6,550
20	7,500	6,000
80	5,000	3,750

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1. Newlands, J. B., "Portable Nuclear Power Plant," GE-ANPD, DCL 60-6-150, June 27, 1960.

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9. CLOSED CYCLE HELIUM POWER PLANTS

An evaluation of a closed gas cycle nuclear power plant for the propulsion of aircraft was made in 1957 by the Flight Propulsion Laboratory Department for GE-ANPD.[1, 2] The evaluation was based on factors that were derived from an intensive thermodynamic and mechanical-design study of closed cycle helium compressorjet,[3] turbojet,[4] and turboprop[5] power plants. Probable advances in technology from 1960 to 1970 were factored into the evaluation. The discussion of helium systems was expanded to include closed cycle power plants in general.

Comparisons of performance were made between various types of closed cycle helium systems and also between closed and open cycle systems. The following conclusions were drawn from this study:

1. The closed cycle turbojet is superior to the closed cycle compressorjet until turbines capable of operating at $2800^{\circ}F$ can be designed.
2. For subsonic, low-altitude flight, the direct air cycle turbojet is superior to the closed cycle turbojet on the basis of qualitative factors.
3. At Mach 2. 5 and 60, 000 feet, the closed cycle turbojet is superior to the all-nuclear open cycle turbojet, provided additional air turbomachinery is added to maintain the helium flow at the design value.
4. At Mach 2. 5 and 60, 000 feet the chemically augmented open cycle turbojet is superior to the closed cycle turbojet.
5. The closed cycle nuclear turboprop is superior to the open cycle turboprop at Mach 0. 6 and 20, 000 feet, from the standpoint of the power-plant thrust-to-weight ratio. However, it retains all of the qualitative disadvantages of closed cycles.

Schematic diagrams of the systems used in the study are shown in Figures 9. 1, 9. 2, 9. 3, and 9. 4. An analysis of a closed cycle helium-to-air heat exchanger is reported in Reference 6. High-temperature turbomachinery and heat-exchanger design studies are reported in Reference 7.

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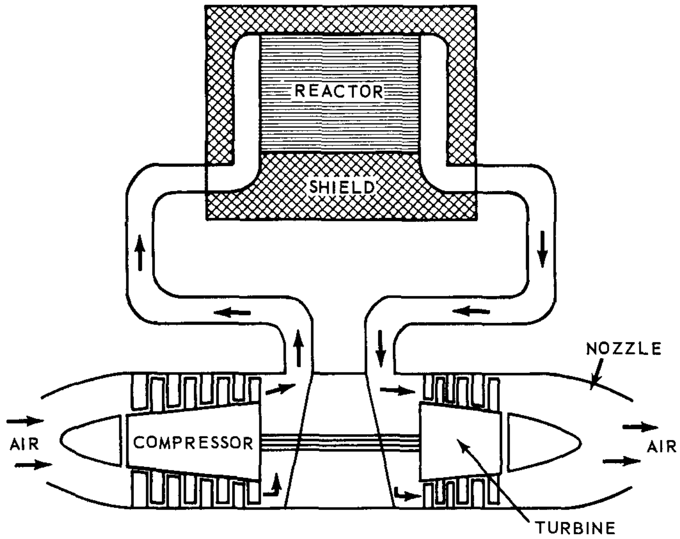

Fig. 9.1 – Schematic diagram of direct air cycle turbojet

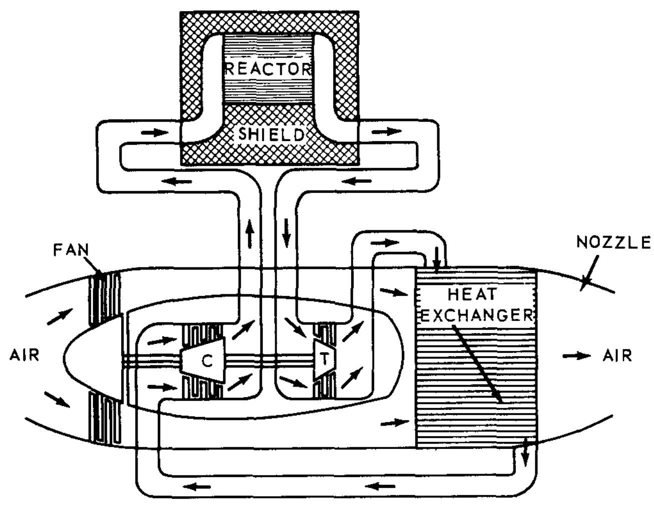

Fig. 9.2 – Schematic diagram of closed cycle compressor-jet

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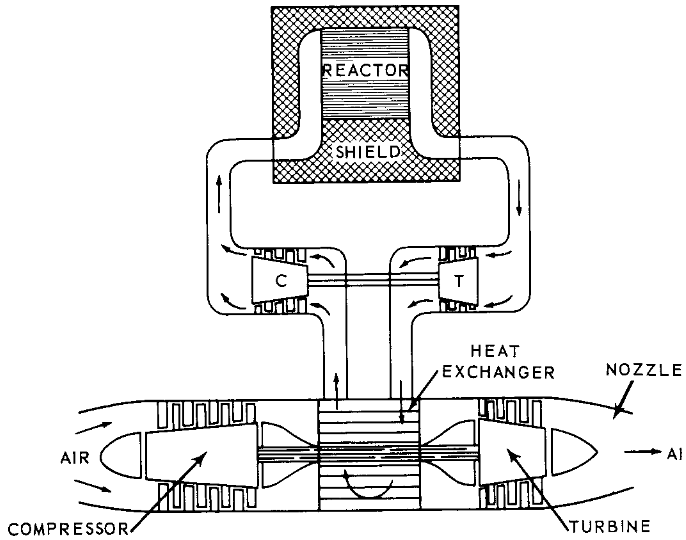

Fig. 9.3 — Schematic diagram of closed cycle turbojet

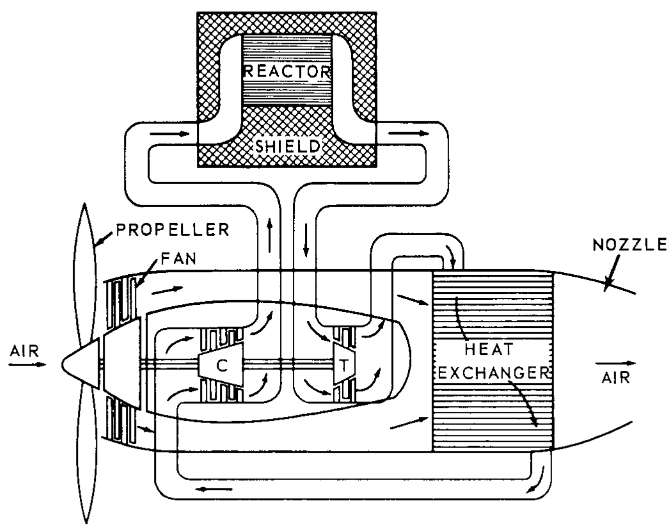

Fig. 9.4 — Schematic diagram of closed cycle turboprop

[redacted]

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9.1 REFERENCES

1. Comassar, S., "Closed Cycle Nuclear Power Plants for Aircraft Propulsion," GE-ANPD, APEX-434, June 23, 1958.
2. "Summary Report, Closed Cycle Nuclear Studies," GE-FPLD, R57AGT639, October 21, 1957.
3. Corliss, W. R., Nichols, H. E., and Wesling, G. C., "Closed Cycle Nuclear Compressor Jet," GE-FPLD, R57AGT597.
4. Corliss, W. R., Nichols, H. E., and Wesling, G. C., "Closed Cycle Nuclear Turbojet," GE-FPLD, R57AGT635.
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6. Nichols, H. E., "Closed Cycle Helium to Air Heat Exchangers," GE-FPLD, R57AGT-638.
7. Riedel, W. and Gisslen, J., "Closed Cycle High Temperature Turbomachinery and Heat Exchangers," GE-FPLD, R57AGT637.

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10. AIRSHIP PROPULSION

A preliminary investigation was performed by GE-ANPD in 1960 of a nuclear power plant for the propulsion of airships.[1] The study was based on the following design conditions furnished by the Goodyear Aircraft Corporation:

Reference altitude	5,000 feet
Shaft horsepower	6,000
Maximum flight speed	85 knots
Minimum reactor-crew separation	200 feet
Crew dose rate	0.0025 rem/hr

The general arrangement of a nuclear-powered turboprop power plant that could serve in this application is shown on Figure 10.1. In this arrangement, air is collected in scrolls from the compressor discharge of the engine and piped to a header at the bottom of the reactor-shield assembly. It is then ducted through individual risers to a plenum in the reactor-core inlet. Exit air from the core is ducted through pipes to a collector header and then to a scroll at the turbine inlet.

The reactor is similar to the HTRE No. 1 core configuration described in APEX-904 of this Report. An aluminum structural core is integrally attached to a stainless steel shield plug. This unit includes the reactor, reflector, control rods, control rod actuators, and nuclear sensors. A stainless steel cylindrical liner, into which the inlet and exit air passages are ducted, surrounds this unit. The liner, in turn, is welded into a spherical pressure shell that contains the lead gamma shield and liquid neutron shielding. The reactor moderator water circulates through the inlet and outlet pipes as shown in Figure 10.1. Circulation of the liquid shield is also necessary. A heat exchanger, to dissipate the heat generated in these two loops, is required. Not shown in the figure, the tube-fin heat exchanger measures about 10 square feet in frontal area and 1.5 feet in thickness, and incorporates a blower. The heat removal rate was assumed to be approximately 10 percent of reactor power.

Calculations were performed on the amounts of shield required to meet the design conditions. The reactor is centered 2.5 inches aft of the vertical centerline of the shield. The thicknesses of the front and rear shields are shown in Table 10.1.

These thicknesses result in a spherical reactor-shield assembly 159 inches in diameter, weighing approximately 120,000 pounds (including the reactor). The weight of each engine assembly is approximately 1430 pounds (600 pounds for the engine, 600 pounds for the gear box, and 230 pounds for the propeller). The weight of the heat exchanger, including fan and piping, is about 2300 pounds. The hot ducting weighs about 11.5 pounds per foot, and the cold ducting (compressor discharge) about 3.5 pounds per foot. The estimated total weight of the installed power plant is approximately 130,000 pounds.

Because this was a preliminary study, further design studies could be expected to provide significant decreases in these weights. For example, jet fuel can be utilized for shielding in place of the water within the shield envelope. This substitution can be made on an approximately equal-volume basis, resulting in a weight saving of 15 to 20 percent. Using jet

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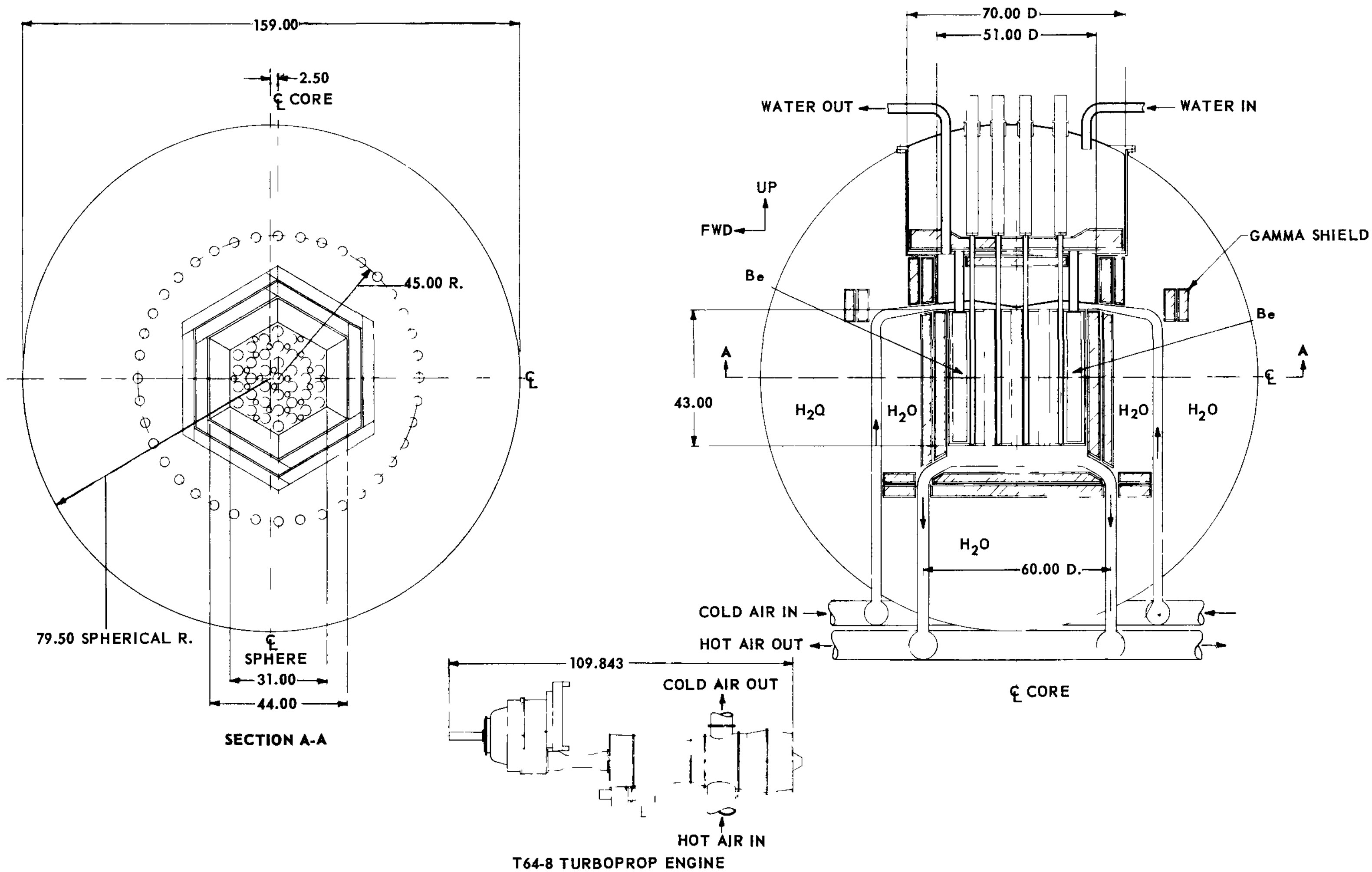

Fig. 10.1—Propulsion system proposal layout (629E288)

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TABLE 10.1

SHIELD THICKNESSES FOR AIRSHIP POWER PLANT

	Front Shield	Rear Shield
Water	46.5	49.0
Lead	9.0	3.5
Iron (shield tank and lead cladding)	3.5	1.5
Total thickness	59.0	54.0
Inlet-air annulus thickness	1.0	1.0
Total shield thickness	60.0	55.0

NOTE: All dimensions in inches.

fuel as a shadow shield beyond the shield envelope would be somewhat less efficient, depending on its location. Although no consideration was given to the shielding effect of the structure and helium in the airship, this, too, would tend to reduce the over-all weight of the system.

The power-plant study shows that the required performance can be obtained by using a core of the HTRE No. 1 type with six General Electric T64 engines. To match the design conditions of the HTRE No. 1 core, it is necessary to operate at a turbine-inlet temperature of 1400°F at a turbine-inlet to compressor-discharge pressure ratio of 0.84. A constant-area exhaust pipe with no discharge nozzle was assumed. At the rated operating point, the equivalent net jet horsepower is about 30 per engine. The estimated performance under these conditions is shown in Table 10.2.

TABLE 10.2

PERFORMANCE OF AIRSHIP POWER PLANT

Condition	Cold Day	Standard Day	Hot Day
Ambient temperature, °F	-14.7	41.2	83.4
Shaft horsepower, 1 engine	1613	1015	600
Shaft horsepower, 6 engines	9678	6090	3600
Reactor power with 6 engines, mw	28.25	21.2	16.25

The variation of shaft horsepower with ambient temperature is presented in Figure 10.2. It can be seen that the power output is adequate on standard or cold days. However, if full power is required on hot days, chemical augmentation is required. Interburners have been assumed for chemical take-off and landing.

A preliminary estimate of half-power operating conditions was made. For a total of 3000 shaft horsepower (500 per engine), the turbine-inlet temperature is about 1285°F, and the reactor heat release is about 14.9 megawatts, assuming that the turbine-to-compressor pressure ratio remains at 0.84.

A preliminary investigation was made of the possibility of using a General Electric MS240 engine in place of the T64. For the specified flight conditions on a standard day, this configuration can produce 6000 shaft horsepower at a turbine-inlet temperature of about 1315°F and a turbine-to-compressor ratio of about 0.84. One MS240 unit has a

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greater power capability than six T64 engines. However, flight trimming would be less advantageous in the single-engine configuration. Also, considerable redesign would be involved, whereas major modification would probably not be necessary with the six T64 engines.

In the course of the preliminary design phase various engines would have been investigated to determine the best "match" with requirements and with the reactor, as well as minimum costs. The estimated cost to conduct a research and development program on the power plant through flight test is given in Reference 1.

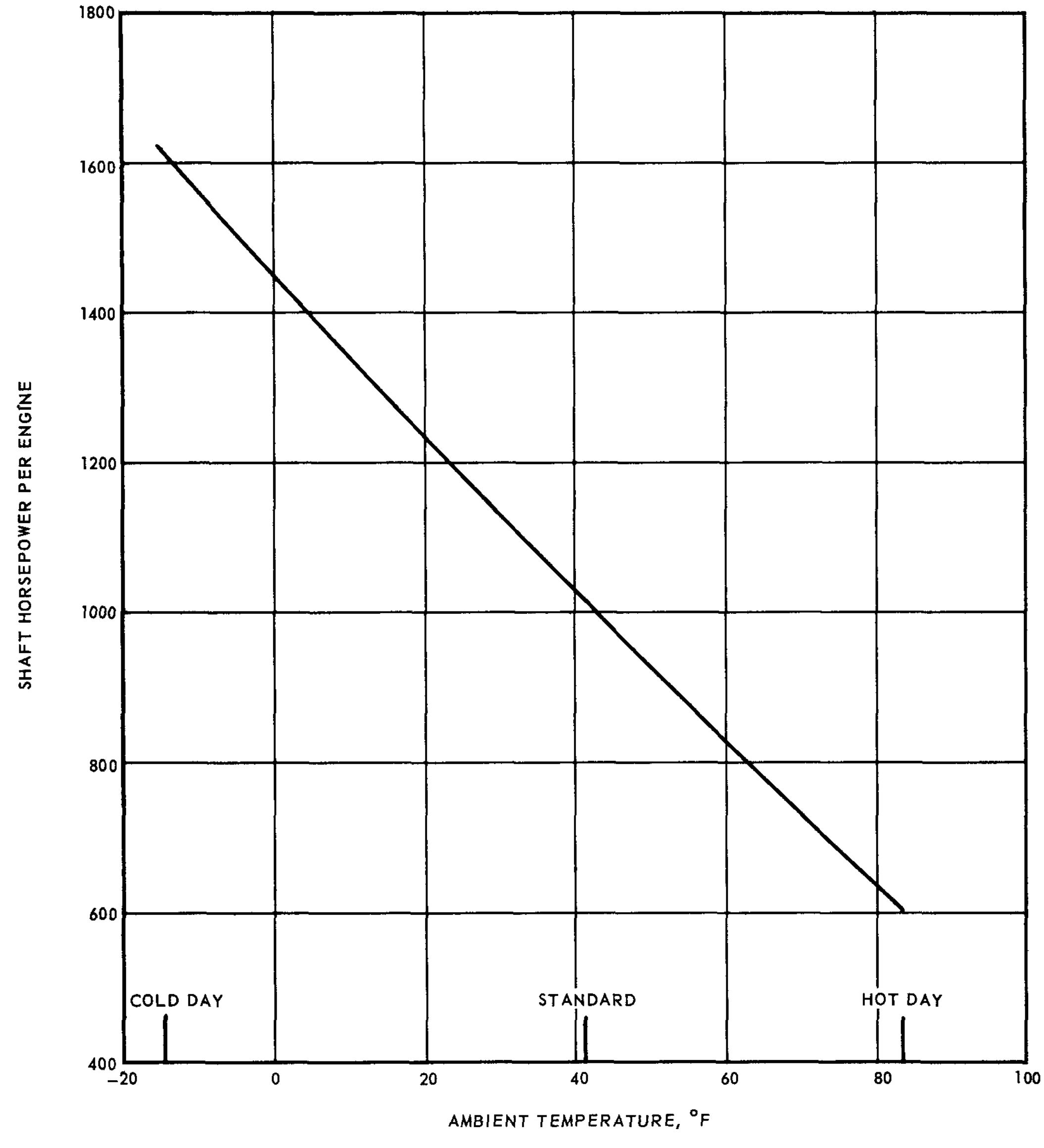

Fig. 10.2 – Shaft horsepower as a function of ambient temperature

10.1 REFERENCES

1. "Power Plant Application Study," GE-ANPD, XDCL 60-11-141, Nov. 25, 1960.

11. NUCLEAR POWER-PACKAGE APPLICATIONS

11.1 INTRODUCTION

Advanced design mobile nuclear power systems should have the following characteristics:

1. They must have sufficient reliability to permit extended operation without maintenance. This implies a minimum of design complexities and remote-handling requirements. Such reliability can be realized either in a single power package as is current practice, or with multiple power packages. The use of multiple power packages (multiengine aircraft have used them for years) increases the effective lifetime of the system rather than the lifetime of an individual power plant.
2. Mobile nuclear power systems should use geometry and environment as shielding to the maximum permissible extent. For example, it would be desirable to use sea water as a shield in marine applications.

The "package" concept is somewhat analogous to the use of JATO bottles, where, once the unit has been fired, the empty bottle is either thrown away or reprocessed if the economics of the system so justify. In order to determine the feasibility of this concept, the mechanical, nuclear, and other characteristics of the power package were investigated in some detail. These preliminary studies, based on achievable technology, have resulted in the completely integrated, closed-cycle, gas-turbine reactor combinations shown in Figures 11.1 and 11.2. These power plants are designated 601A and 601B. Their descriptions and designs are presented more fully in APEX-909, "Aircraft Nuclear Propulsion Systems Studies," of this Report.

11.2 601B POWER PLANT APPLICATIONS

The utilization of nuclear power plants for ship propulsion is indicated where long range at high speed or where sustained underwater travel is required. The 601 power package has application to these missions, particularly where low power-plant weight is a necessary or desirable feature. The low specific weight of the power package permits nuclear counterparts of conventional ships to be in the same size range, but with an increased payload capability.

The integrated nuclear power package can be used wherever there is sufficient water to provide a shield and heat sink. In general, 5 feet of water attenuates neutrons by a factor of almost 10^6 and gammas by a factor of nearly 10^2. The reflector and heat exchanger that are an integral part of the power package attenuate gammas by an additional factor of 10^3. These values indicate that when the power plant is used as an under-water propulsion device mounted external to the hull, a separation distance of about 5 feet is probably satisfactory for most applications.

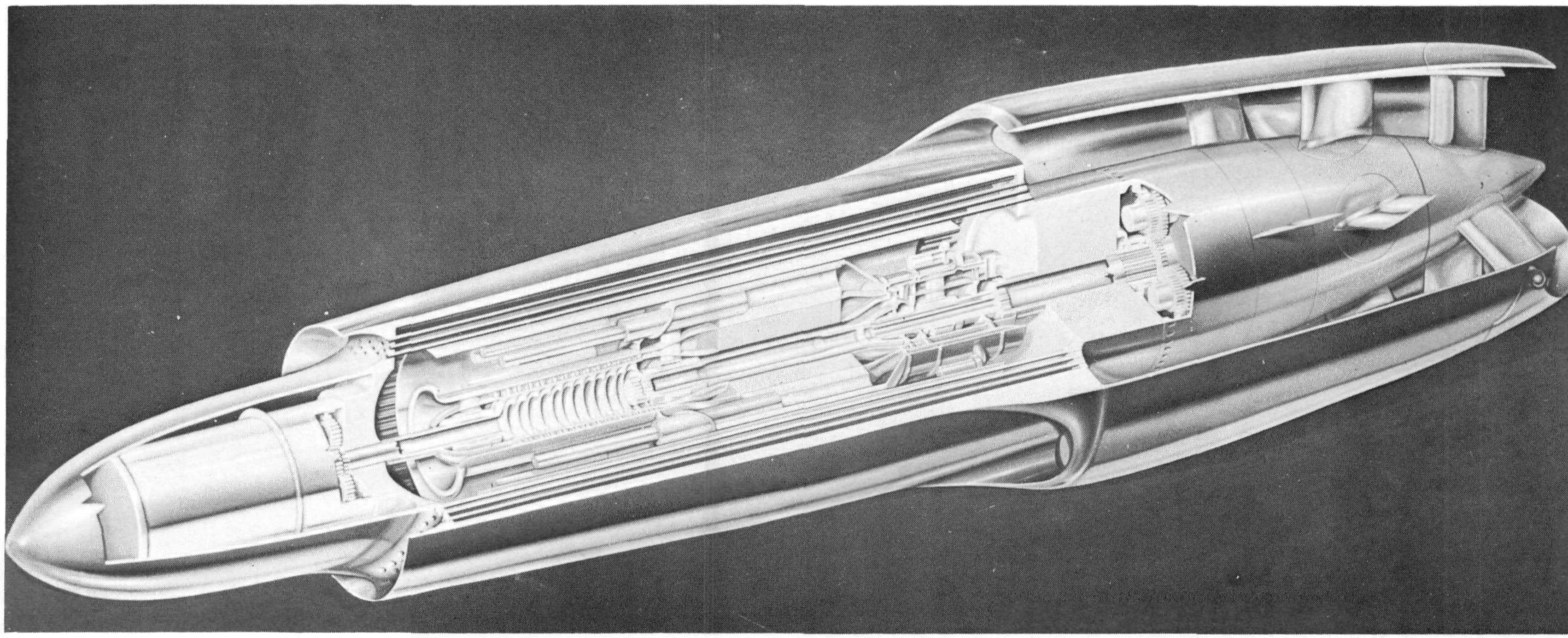

Fig. 11.1 – Artist's concept of 601A power package

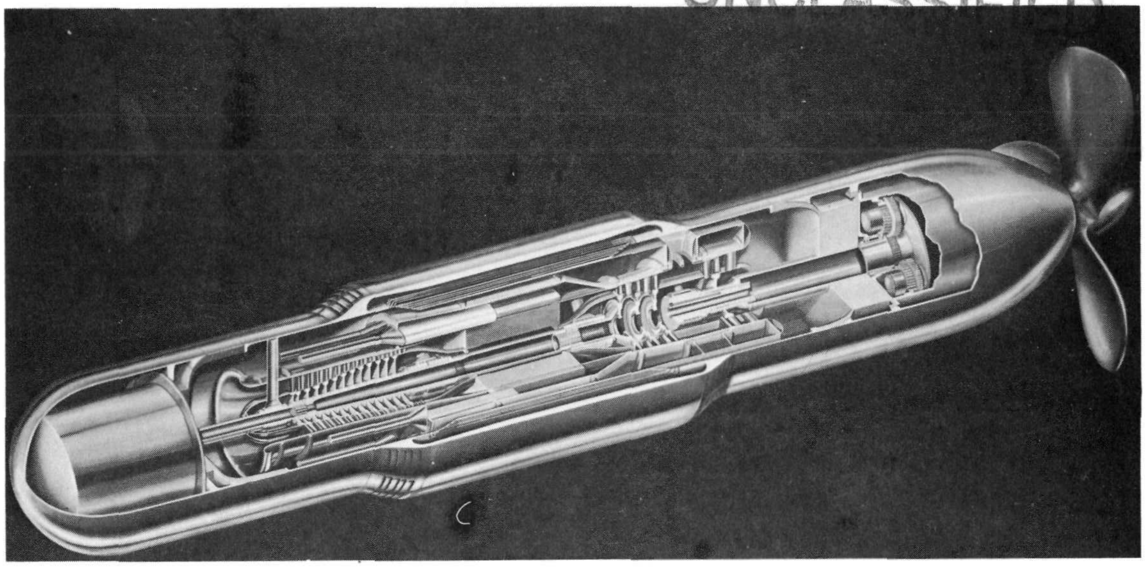

Fig. 11.2 – 601B power package

In the evaluations, the power packages were mounted on the ships by means of pylons attached to the hull. In installations on existing ships, this feature permits the space within the hull formerly occupied by the conventional propulsion machinery to be made available for other uses. An additional advantage of this type of power-plant installation is that after shut-down from a power run, or in the event of failure resulting in shut-down, the entire power package can readily be removed as a unit. A new unit can be installed, and the system can be operational during the time the original unit is undergoing repair or reprocessing. This significant feature of the package approach permits the maximum utilization of the vehicle.

An examination of possible applications of the 601B power package indicated that certain installations show promise of immediate advantages. These are:

1. Escorts and submarines, especially those of long range (compared to conventional ships), small size (compared to current nuclear vessels), and adequate military payload.
2. Hydrofoils, particularly those with long range and increased payload compared to their fossil-fueled counterparts.
3. Commercial vessels; the advantages of this power plant in weight and compactness also indicated the possibility of cost savings in the field of commercial vessels, although additional application studies in this area were still required.

The weights of the total propulsion system must be considered.

Power-Package Weights - In keeping with current naval practice, an emergency propulsion system must be provided in addition to the primary system. In this study, this is estimated to be 50 tons* for a surface ship and 110 tons for a submarine (the greater weight for the submarine is due to the batteries for emergency propulsion).

For nuclear surface ships, this results in total propulsion system weights of

30, 000 horsepower	185 tons
60, 000 horsepower	240 tons
75, 000 horsepower	270 tons

*Except where otherwise noted, "tons" refer to long tons.

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For nuclear submarines, the power plant weights are

15, 000 horsepower	215 tons
30, 000 horsepower	245 tons

11. 2. 1 SURFACE ESCORTS (ANTISUBMARINE AND ANTIAIRCRAFT)

A typical installation of the 601B power plant on an escort ship is shown in Figure 11. 3.

The following sections describe the performance and capabilities of three types of surface ships powered by the 601B power plant.

11. 2. 1. 1 DDG

4, 500 tons full load displacement
70, 000 shp 35 knots
1, 300 tons military payload

The installation of five of the 601B power plants in place of the existing propulsion system results in a propulsion system weight of 270 tons for 75, 000 shaft horsepower. The conventional installation requires about 2000 tons of propulsion machinery plus fuel. Exchanging the conventional propulsion system and fuel for a 601B nuclear system could approximately double the military payload to 3000 tons. Or, the size of the ship could be reduced, with a proportionate reduction in propulsion power, and still achieve an increase in the military payload. Although such an exchange would require consideration of stability and balance, there is no doubt that payload capability can be increased and size reduced.

11. 2. 1. 2 DD (Fletcher Class)

3, 000 tons full load displacement
60, 000 shp 35 knots
700 tons military payload (estimated)

The installation of four 601B power plants results in a propulsion system weight of 240 tons compared to the estimated 1600 tons of conventional propulsion machinery plus fuel. The military payload should increase from 700 to somewhat more than 1300 tons. This compares very favorably with the payloads of the conventionally powered DLG-6 (1300 tons) and DLG-16 (1800 tons), which are considerably larger ships.

11. 2. 1. 3 DE (Dealy Class)

2, 000 tons full load displacement
20, 000 shp 25 knots
500 tons military payload (estimated)

The installation of two 601B power plants results in a propulsion system weight of 185 tons for 30, 000 shaft horsepower with a speed of 28 knots. The propulsion machinery and fuel of the original installation is estimated to weigh 800 tons. By converting to the 601B nuclear system, the military payload should increase by 600 tons, approximately doubling the present payload. The increase in horsepower with a corresponding increase in speed is descriptive only. No analysis of loads has been made to verify the feasibility of this addition to this particular class of hull.

The foregoing examples are meant to be descriptive only, and do not imply that detailed treatment of the design was made. They do, however, indicate the increase in payload and decrease in size that can be achieved as propulsion weight decreases.

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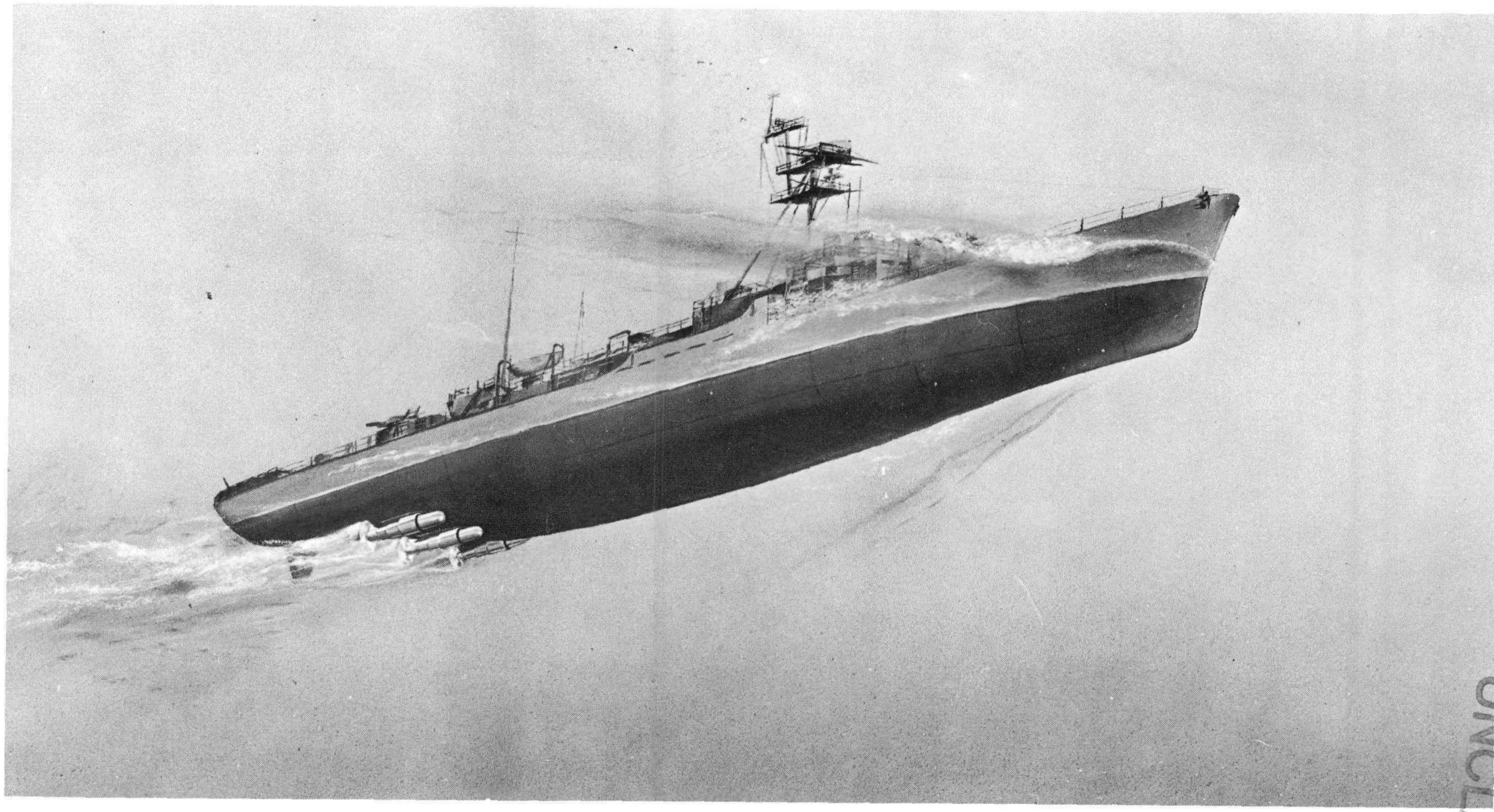

Fig. 11.3—Installation of 601B power package on destroyer escort

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11.2.2 SUBMARINES

Serious consideration has been given to small submarines in recent years since there is some correlation between size and cost. In deep-diving submarines, compactness is especially desirable to lessen the structural problems in the high-pressure environment. In examining a nuclear attack submarine in some detail, the advantages of decreased power-plant size become apparent. The nuclear submarine SSN 589 (Scorpion) was studied.

Taking into consideration the additional 110 tons required for emergency propulsion, the 601B propulsion-system weight for the SSN 589 would be 215 tons for 15,000 horsepower. This is a reduction of 550 tons or about 20 percent of ship weight without water and lead. This weight decrease would produce other reductions (e.g., hull structure), so that a design might be as small as 2000 to 2500 tons. This reduction in size may actually be more significant than reduction in propulsion system weight.

Of further interest and significance, since most of the weight of the 601B propulsion system is in emergency power and controls, the power could be doubled without incurring a large penalty; e.g., 30,000 horsepower can be generated by a system weighing 245 tons, only 30 tons more than the 15,000-horsepower system. A 2000-ton attack submarine with this power would have a speed of approximately 38 knots.

No attempt was made to assay the cost reductions made possible by the reductions in size in the foregoing studies.

A typical submarine installation of the 601B power plant is shown in Figure 11.4.

11.2.3 HYDROFOIL VEHICLES

Hydrofoil vehicles were built as early as World War II, but little application was found for them until the late 1950's. One of the largest, currently in the developmental and experimental stage by Grumman for the Maritime Administration, has about 80 tons displacement. The Navy's PCH-1, being developed by Boeing, will be a little larger.

There is considerable promise in the capability of hydrofoil vehicles to travel rapidly on the surface of the ocean. The major limitation is the power plant. While gas turbines can provide the required speed within the weight requirements, they do not have the necessary endurance because of the fuel requirements. Careful design is required to achieve a useful payload-range combination. Two sample hydrofoil vehicle designs, for example, have the characteristics shown in Table 11.1.

TABLE 11.1

PERFORMANCE OF TWO HYDROFOIL DESIGNS

All-up weight, short tons	200	1,000[a]
Shaft horsepower	16,000	80,000
Foil-borne operation at 60 knots, hr	12	12
Machinery plus fuel, short tons	90	414
Payload, short tons	25	84

[a]In the present state of hydrofoil technology, 500 tons is considered to be the maximum size. The 1000-ton boat is used as an illustration of the possibility of boats of this size.

[redacted]

Fig. 11.4—Installation of 601B power package on a submarine

If the specific weight of the 601B nuclear power plant for this application is assumed to be 5 pounds per horsepower, and the resulting weight savings is converted to payload, the payload increases to 75 tons and 298 tons respectively for these two designs. In addition, the nuclear propulsion provides an extended range capability.

The power-package concept in nuclear propulsion is particularly adaptable to the hydrofoil application. Because the power plant can be submerged, it does not have to be supported out of the water. Although this increases the drag, it is countered locally by the thrust so that the additional load is not transmitted over the strut.

11. 2. 4 COMMERCIAL SHIPS

Specific studies of installations of the 601B power package in merchant ships were not made. It appeared, however, that the savings in weight and space made possible by this compact power plant would also provide economic benefits in this area. Full utilization of this system can best be achieved in combination with innovations in ship design and operating procedures.

11.3 601A POWER PLANT APPLICATIONS

The 601A power package represents an effective miniaturization of a closed-cycle, gas-turbine nuclear power plant. The low specific weight per shaft horsepower, combined with its capability for generating electrical power, makes possible the production of an array of undersea weapons.

Many applications of the 601A power plants are possible. In the configuration studied, it can be used as an integrated power package wherever there is sufficient water to serve as a shield and a heat sink. Adding additional shielding and a modified waste heat exchanger permits utilization on land.

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12. AEROSPACE NUCLEAR PROPULSION SYSTEM

A study was made in 1960-61 of the aerospace potential of direct cycle nuclear propulsion,[1] and a system conceived to transit from the Earth's surface to Mars, and return. Nuclear propulsion is necessary for such a mission because it is the best available means of supplying tremendous quantities of power in relatively small packages. The system described in this section includes recoverability of all vehicles, use of the same reactors for several parts of a system, and a great reduction in the difficulties of atmospheric reentry. In addition, this system employs the only practical means of atmospheric travel on oxygen-lacking planets, nuclear propulsion.

The system uses three types of nuclear power plants: aircraft turbojet engines, ramjets, and rockets. Air and hydrogen are used in various parts of the system at different times.

Except for minor differences, the function of the direct-cycle heat transfer reactor are the same for the atmospheric turbojet during take-off, the ramjet for boost, and the space rocket power plant for escape and interplanetary transit. These requirements include:

1. Long life (high burn-up) capabilities.
2. High-temperature reactor operation.
3. Large void volumes and surface areas for heat transfer to the cooling and air hydrogen propellant.
4. Tolerable radiation without excessive shield weight.

The major difference between air-breathing and hydrogen-breathing reactors is the capability of the fuel element materials. In the air-breathing reactor, the materials must be able to resist oxidation, whereas in the hydrogen-breathing reactor the materials must resist reduction (reaction with hydrogen). There are certain high-temperature ceramic-type materials that may be developed to satisfy both of these requirements.

These materials would be used in fast-spectrum reactors. Their application in thermal or intermediate reactors would require excessive fuel inventories to yield criticality, since temperature-resistant materials are characterized by high cross sections for absorption of thermal-energy neutrons. Unlike thermal reactors, fast-spectrum reactors would not be as adversely affected when fueled with these materials and would be able to contain the large quantities of fuel dictated by high burn-up requirements. Further, because they use no moderator, fast-spectrum reactors yield an added propulsion advantage through increased volumetric efficiency.

If materials for the direct cycle reactor are developed to enable high-temperature operation and the transfer of heat to either air or hydrogen, the reactor can be used as the heat source for transitional flight vehicles. These vehicles would use nuclear turbojets from ground take-off to altitudes and flight speeds where the gradual transition from turbojet to ramjet configuration takes place. After the ramjet propels the vehicle to still higher altitudes and flight speeds, the final transformation from ramjet to rocket propulsion takes place at the outer fringes of the atmosphere. The reactor-

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heated, hydrogen-propelled rocket then transports the heavy cargo of hydrogen into orbit. Several such trips would be made to build up an orbiting fuel station with an adequate supply of hydrogen so that, on the last trip, a departure can be made from the Earth orbit into high-velocity, interplanetary flight. The same reactor or group of reactors may be used throughout the flight spectrum.

The immediate advantage of this system over the chemical-rocket system is that staging is unnecessary. The atmosphere is used to assist rather than hinder propulsion during transit to orbit, conserving the valuable hydrogen propellant for use in space. The variable configurations of the vehicle depend upon aerodynamic developments, but they are consistent with present efforts aimed at obtaining higher atmospheric flight speeds.

Turbojets today are capable of development to flight speeds of Mach 3.0 (such as the J93 engine for the B-70 airplane); nuclear-powered ramjets for speeds from Mach 3.0 to Mach 6.0 are also realistic. At Mach 6.0, the inlet air is at nearly $3000^{o}F$ due to ram heating and, for at least a brief time, severe high-temperature conditions would be imposed on the reactors. However, if the typical requirements (for nuclear aircraft) of long fuel element life expectancy and extreme containment of fission products are partially relaxed, and if sufficient effort is devoted to materials development, the required reactor temperature of approximately $3500^{o}F$ can be achieved to supply the energy for useful thrust at Mach 6.0. This would be a worthwhile speed since Mach 6.0 is nearly one-fourth of orbital velocity. Also, to achieve flight at approximately 100,000 feet altitude and Mach 6.0 requires a sizable fraction of the propellant of a chemical-rocket system under the same conditions. In typical chemical rockets for example, as much as 60 percent of the total energy is consumed in passing through the atmosphere, even though chemical rockets are accelerated to nearly vertical velocity vectors at 100,000 feet.

Additional advantages of this system further increase the incentive to develop this all-nuclear combination. By reversing the take-off sequence of transitions from turbojet to ramjet and then to rocket flight, atmospheric re-entry is no longer an extremely difficult problem. The procedure for re-entry and landing would be:

1. A rocket retrothrust with reactor-heated hydrogen in a nose-aft flight attitude to descend from a parking orbit and to enter the atmosphere at reduced velocities.
2. A single end-over maneuver for nose-forward flight orientation and an opening of ramjet inlets to permit nuclear-powered ramjet flight.
3. A transition at a lower altitude and flight speed to nuclear-powered turbojet flight, and a controlled landing at any desired location on the Earth. No other known space propulsion system proposed at that time offered the versatility of re-entry without excessive aerodynamic heating together with controlled atmospheric flight thereafter with a virtually unlimited range.

The assembled system is illustrated in Figure 12.1. It consists of a liquid hydrogen payload compartment, the turbo-ram-rocket vehicle, and recoverable, unmanned, nuclear turbo-ram boosters. The primary turbo-ram-rocket vehicle, with the payload compartment and boosters removed, is shown in Figure 12.2. It contains a translating inlet spike, a shielded crew compartment, a separate shadow shield, an integral nuclear turbojet engine, and the aft reactors that are used for both ramjet and rocket modes of operation.

The recoverable booster vehicle is shown in Figure 12.3. Each boom contains a translating inlet spike, an integral nuclear pressure turbojet engine, and a separate reactor and nozzle. Liquid hydrogen is carried in the center fuselage and provides fuel for tur-

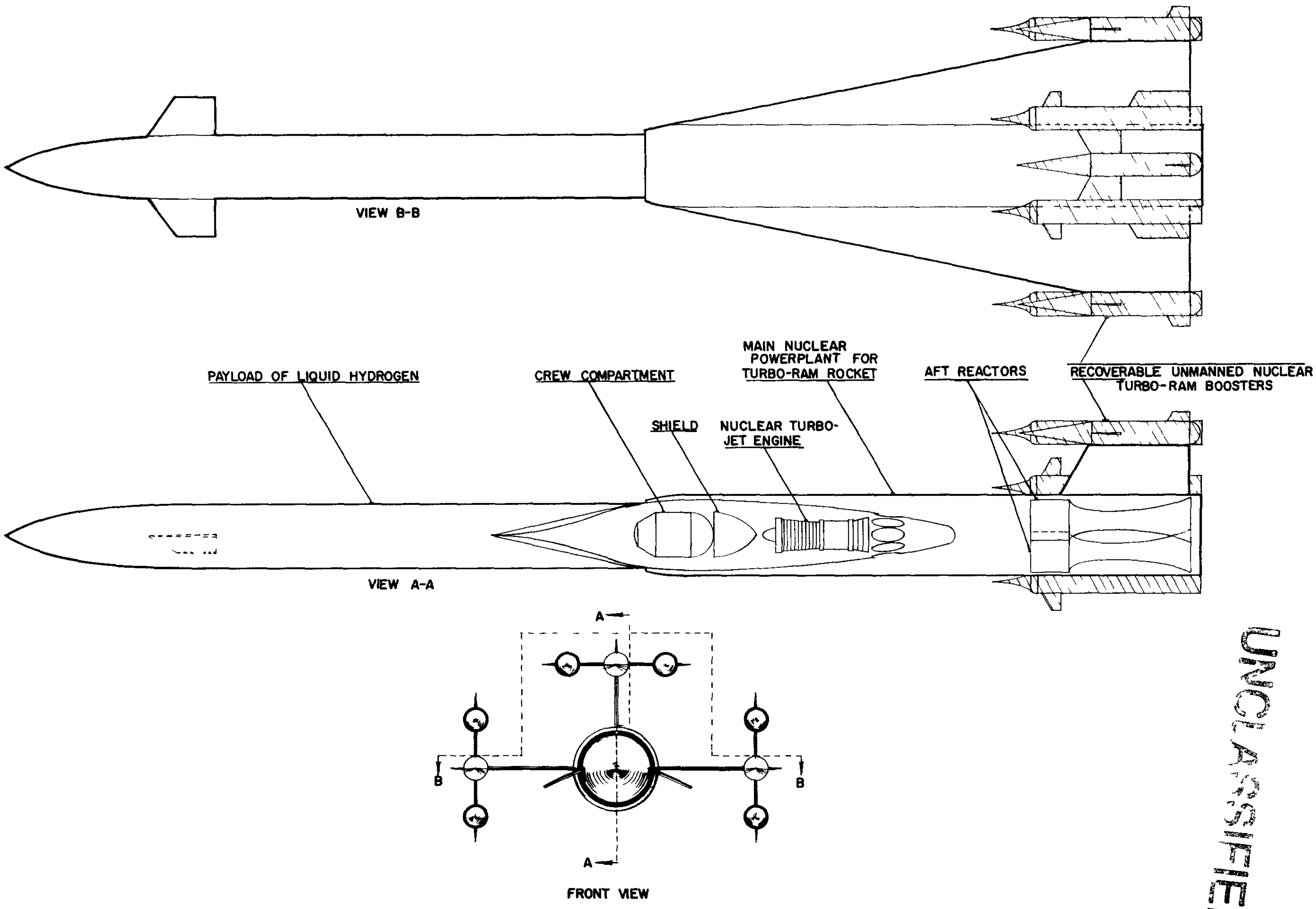

Fig. 12.1 – Nuclear-powered turbo-ram-rocket

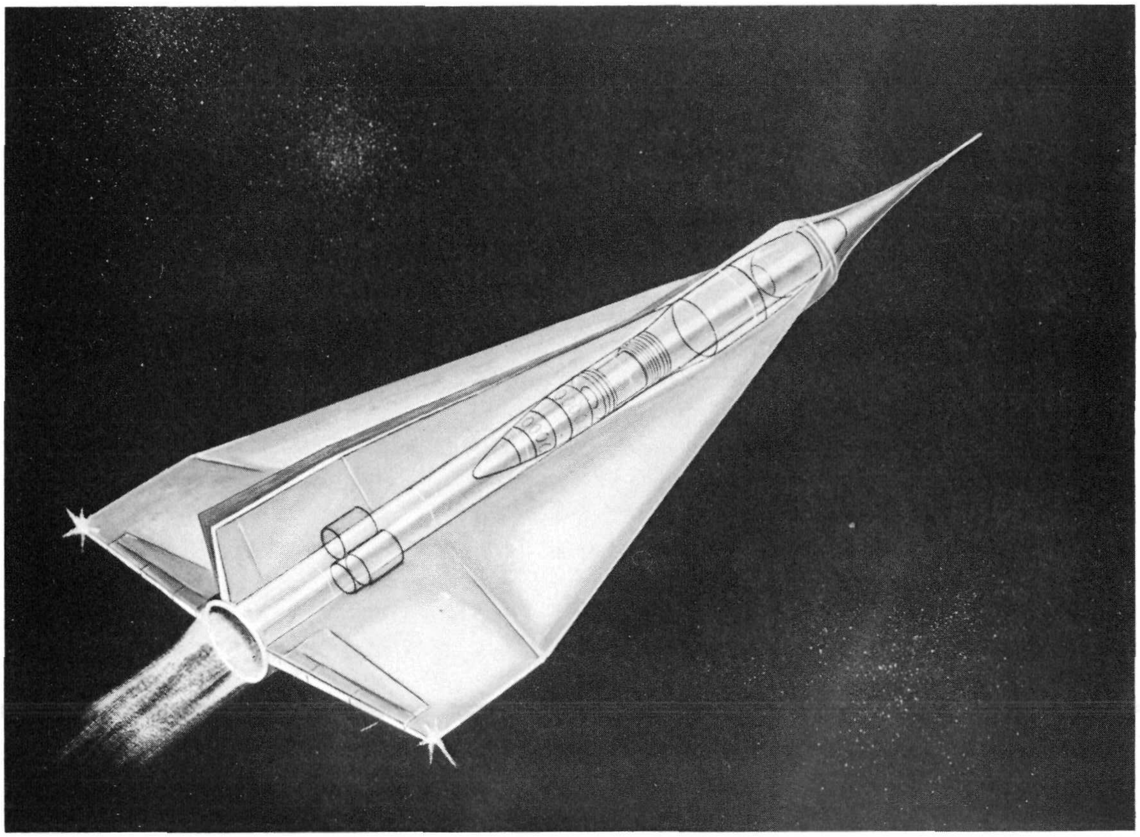

Fig. 12.2 – Main nuclear power plant for turbo-ram-rocket

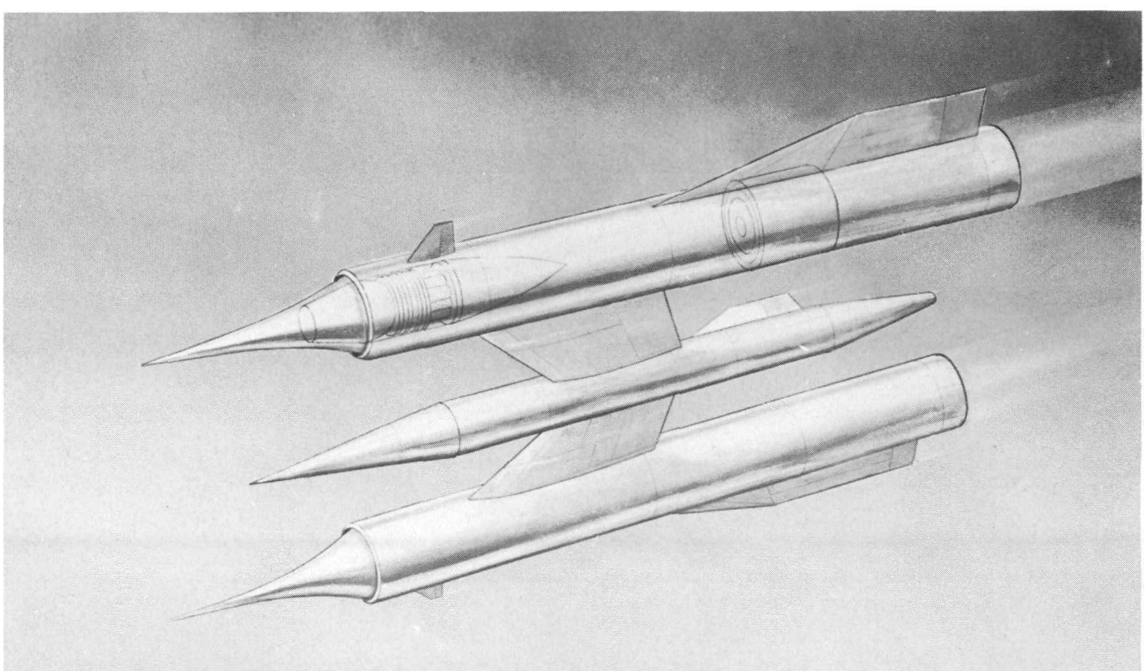

Fig. 12.3 – Recoverable unmanned nuclear-powered turbo-ram booster

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bojet operation. At approximately Mach 3.0, the pressure engines are cut out and inducted air bypasses them to provide nuclear ramjet flight. The boosters are unshielded and remotely controlled.

An artist's concept of the complete system shown shortly after take-off is shown in Figure 12.4. Three booster vehicles are used. Air induction for the primary vehicle is at the juncture of the payload compartment. The flight plan from Earth take-off to approximately Mach 6.0 and an altitude of 100,000 feet is shown in Figure 12.5. At Mach 6.0 and an altitude of 100,000 feet, the spike closes the air inlet and rocket propulsion commences.

In the initial phase of the operation, reactor-heated hydrogen is mixed with ram air to yield high specific impulses from the exothermic chemical reaction. This operation is similar to the air-scooping used in the LACE (liquid air cycle engine) chemical aerospace plane system, except the air is not collected nor liquified. The stored liquid hydrogen payload serves as a heat sink for cooling the power plant and vehicle structure. Because of the low air density above 100,000 feet, the flight is scheduled to accelerate to approximately Mach 10.0 while climbing to only about 125,000 feet. Hence, the kinetic energy of the system increases because the inducted high-temperature and high-pressure ram air readily combines with the reactor-heated hydrogen, but the reactor temperature requirements do not exceed 3500°F.

In this concept, thrust augmentation increases with chemical reaction at a greater rate than if the reactor temperature were increased to the 4500° to 5000°F limits used in the all-hydrogen ROVER nuclear rocket systems. At approximately Mach 10.0, the total weight of the system is reduced sufficiently to reach orbit with a considerable margin from the specific impulse of 710 seconds obtained from the 3500°F reactor.

The liquid hydrogen payload is deposited in orbit to build up a fuel reserve for the remainder of the mission. Upon reaching orbit, the vehicle separates from the payload and makes its return to Earth, as shown in Figure 12.6. Removal of reactor afterheat is not necessary and therefore propellant is not used except during separation.

The assembly of fuel stations in orbit is shown in Figure 12.7. The rendezvous of the 5th flight is indicated. The final assembly of the orbit payload prior to interplanetary flight is illustrated in Figure 12.8. Although 12 payloads are indicated, additional flights can be made to obtain any amount of required propellant payload. On each of the last two flights to orbit, a single ramjet booster vehicle is retained for later use. Also, the next-to-last flight (Figure 12.8) is made with the crew of two in a capsule carried at the front end in the liquid hydrogen compartment and not as indicated in Figure 12.1. The liquid hydrogen in the payload compartment provides additional shielding between the crew and the reactors. The last flight carries the remaining two crew members. A completely assembled interplanetary flight system is shown in Figure 12.9.

The two principal ingredients required for reliable, high-speed interplanetary flight, an abundance of liquid hydrogen propellant and several reactors for energy, are incorporated in this system. As many as 21 reactors may be contained in the system shown in Figure 12.9. Additional tiers of liquid hydrogen payload compartments may be added.

Because this system starts from orbit and has an abundance of energy and propellant, a relatively fast transient can be made to Mars. The transient time depends on the size of the orbiting fuel station; two to three weeks may be sufficient to reach a Mars orbit. The system uses acceleration and deceleration throughout most of the interplanetary flight; very little coasting is programmed.

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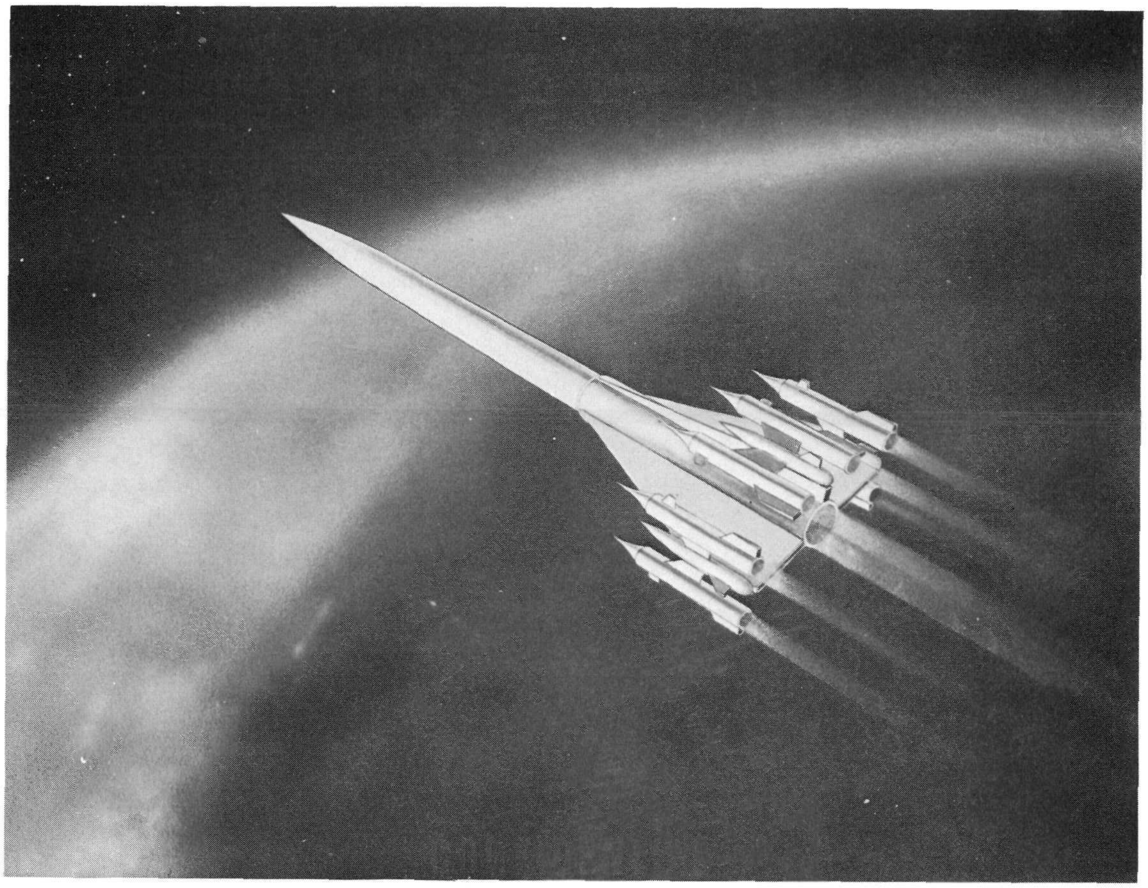

Fig. 12.4 – Nuclear-powered turbo-ram-rocket

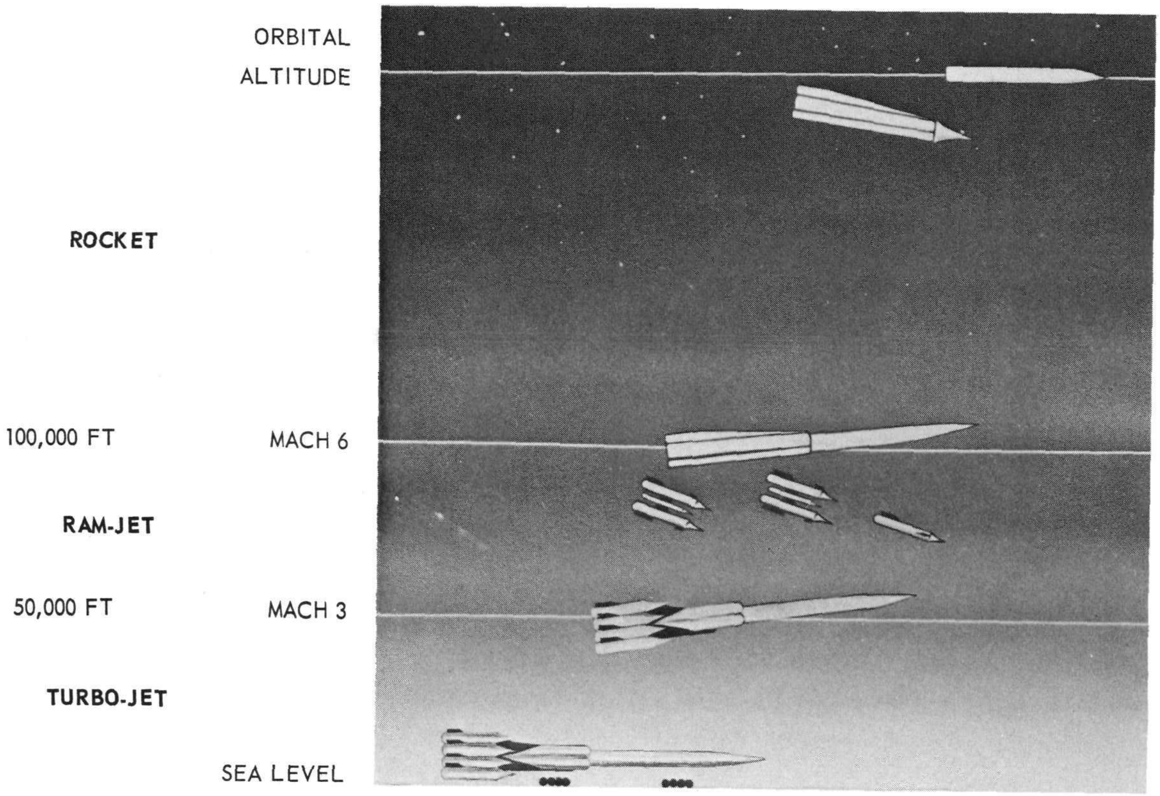

Fig. 12.5 – Flight plan for earth-orbiting fuel station

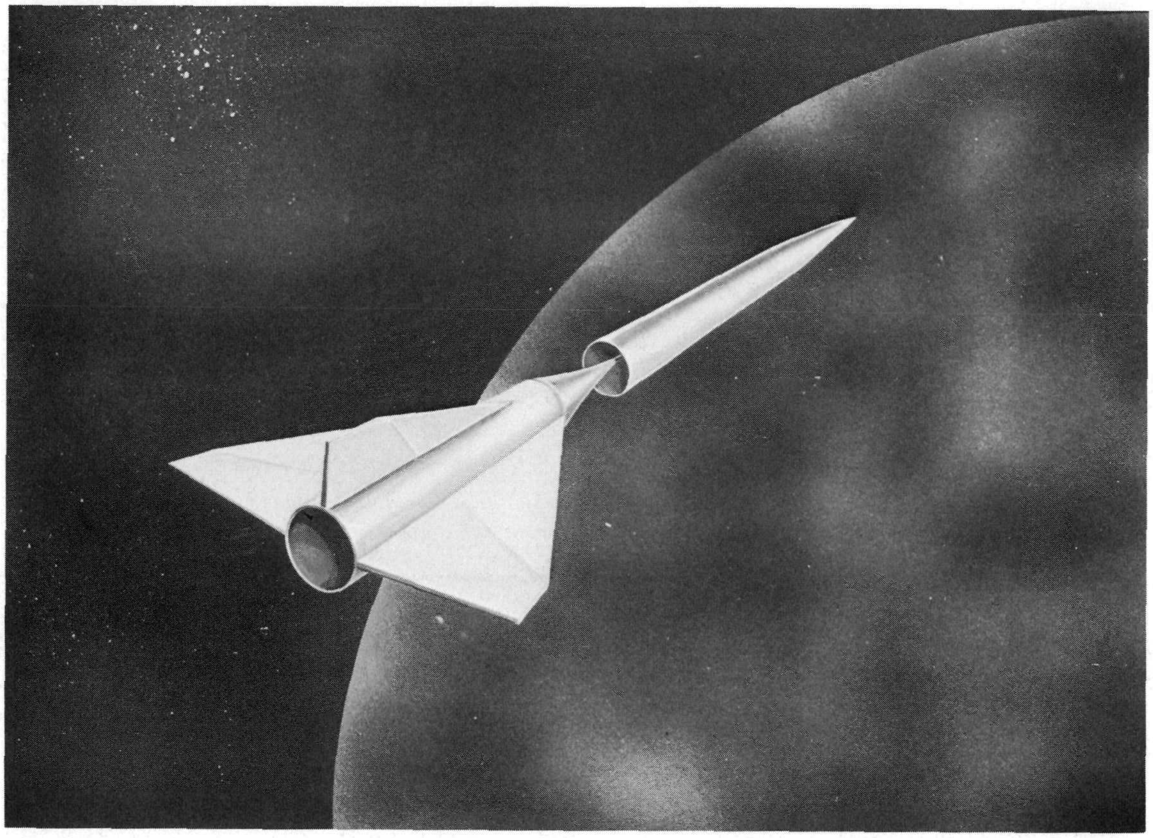

Fig. 12.6 – Separation of liquid hydrogen payload in orbit

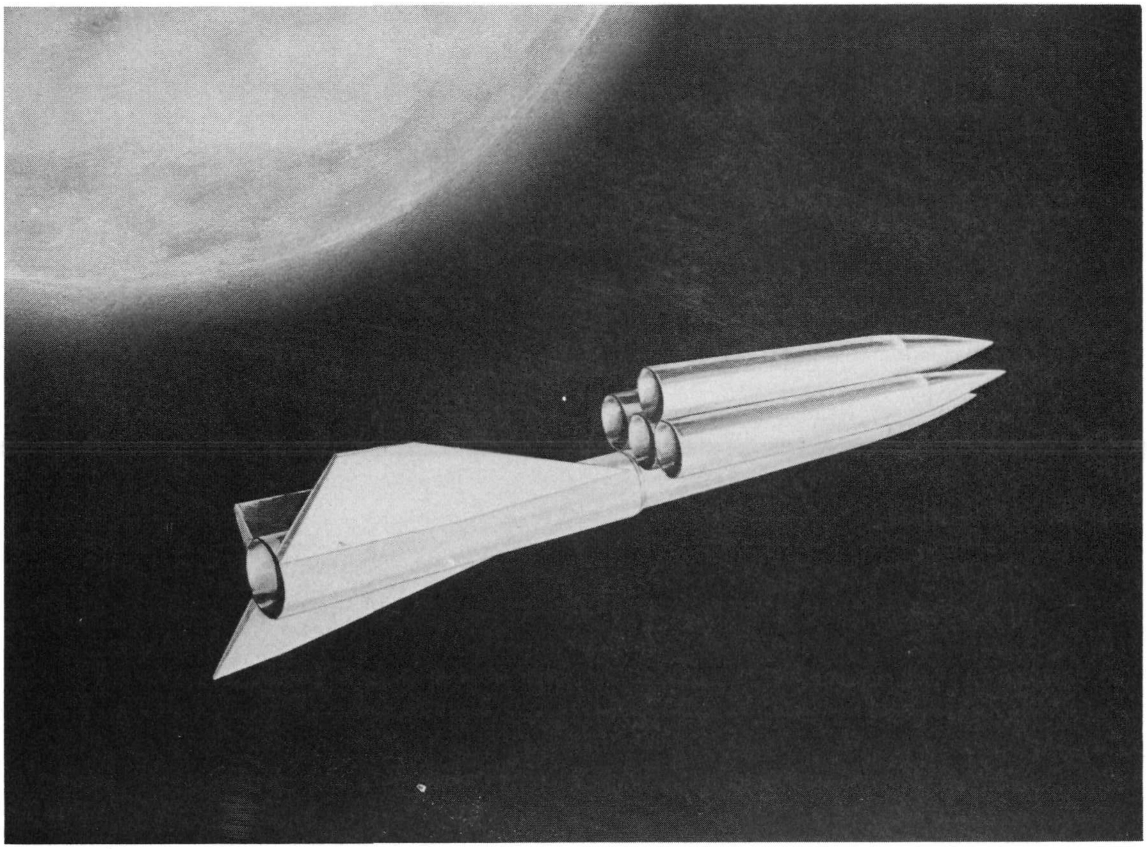

Fig. 12.7 – Assembly of earth-orbiting fuel station

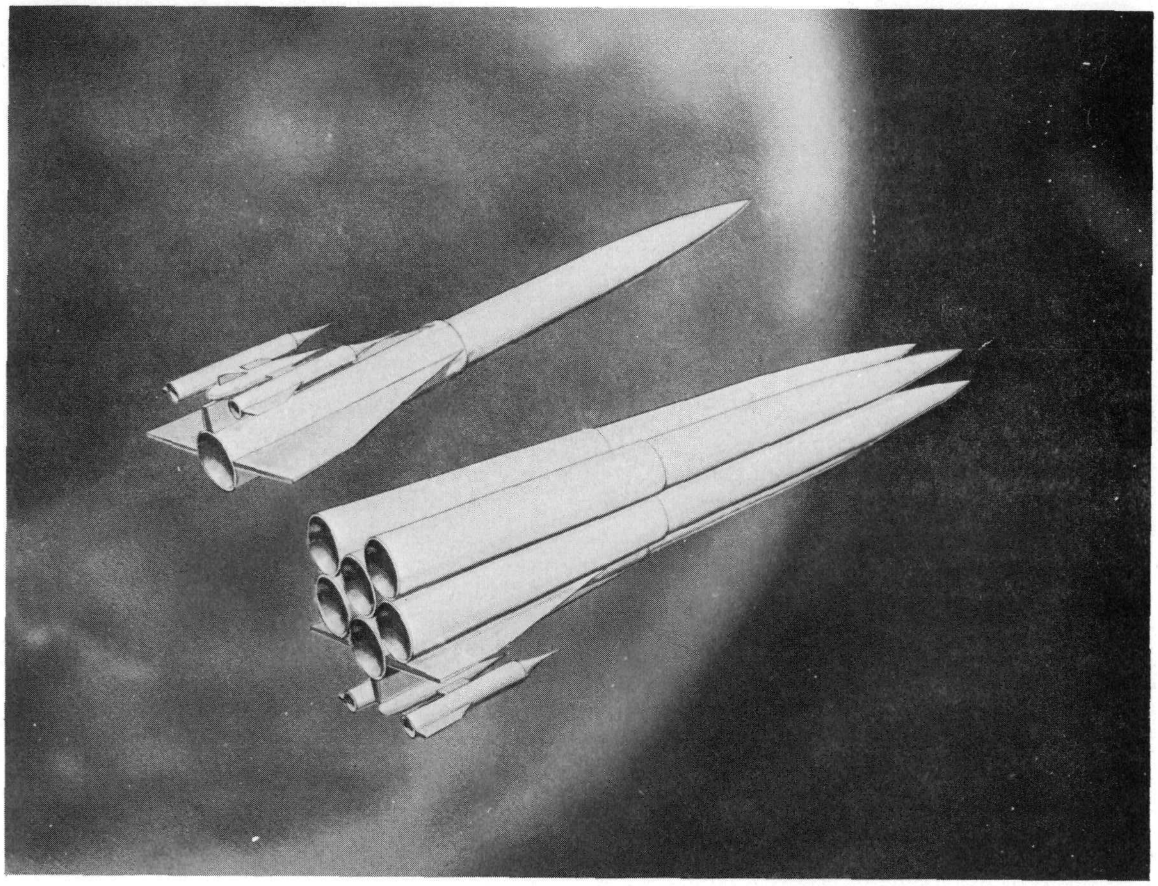

Fig. 12.8 – Approaching last-trip assembly prior to interplanetary flight.

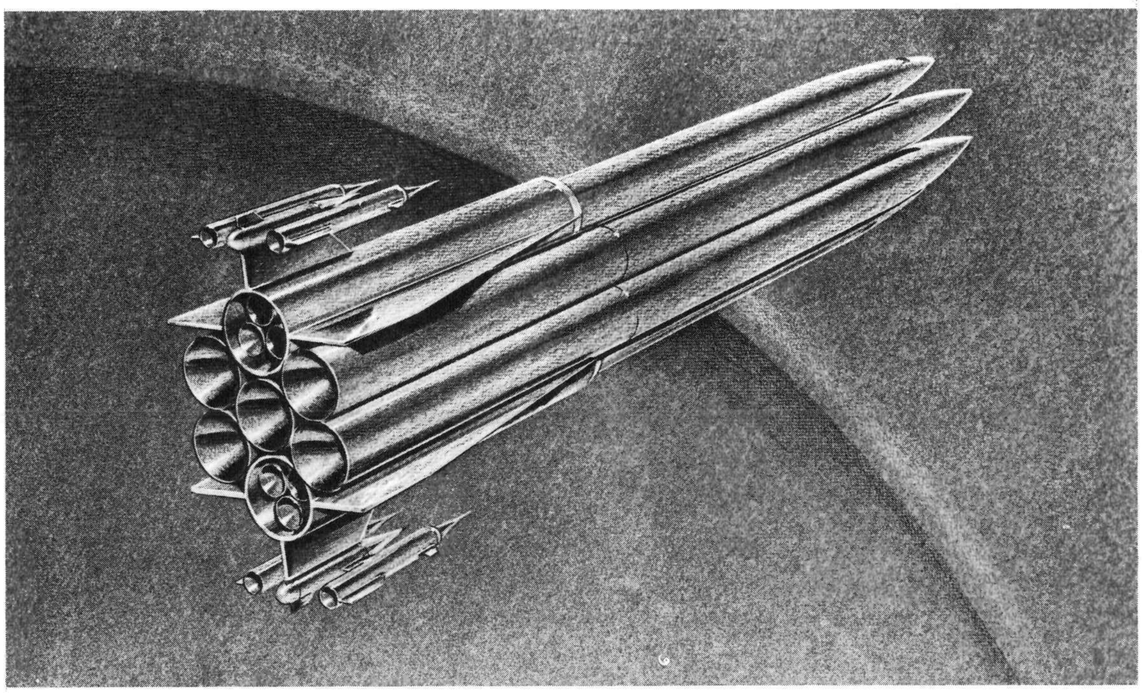

Fig. 12.9 – Final assembly for interplanetary flight

A preliminary study was made, exploring the possibilities for locomotion on Mars. After arrival in a Martian orbit, it should be possible to approach the terrain and move about. The most desirable method of transport would be atmospheric flight. Based on numerous spectrographic data and gravitic analysis, it is believed that the Martian atmosphere does not contain appreciable quantities of oxygen. Hence, propulsive power could not be obtained from chemical reactions using atmospheric oxygen. Although the combustion engine is taken for granted on Earth, it could not be used on Mars. Except for muscle power, only two means of propulsion are currently conceivable: chemical rocket propulsion and nuclear propulsion, neither of which requires oxygen derived from the atmosphere.

For atmospheric flight on Mars, nuclear propulsion has even greater advantages over chemical rocket propulsion than it has on Earth, since the propellant for the chemical rocket would have to be transported all the way from Earth, thus consuming transport power and imposing staggering payload limitations.

The Martian atmospheres should sustain nuclear turbojet or nuclear ramjet propulsion and the range, endurance and number of flights would be no less limited than in similar flights on Earth. Actually, the absence of oxygen in the Martian atmosphere could well extend the performance and life of nuclear power plants, since oxidation deterioration would not be a problem. Further, an aerodynamic flight vehicle designed for operation on Earth would perform better on Mars, since its weight on Mars would be only 39 percent of its Earth weight. The lesser density of the Martian atmosphere relative to the terrestrial atmosphere would reduce this advantage somewhat, since it is estimated that the atmospheric density at the Martian surface is equivalent to a terrestrial density at an altitude of 50, 000 feet. The effect of this disadvantage would be small.

The relatively high density above 100, 000 feet on Mars would be an advantage for an atmosphere-breathing nuclear-powered ramjet. Such a vehicle is shown in Figure 12. 10.

Early Martian flights would probably be used for television reconnaissance. Later flights could contain manned entry vehicles whose return-systems would be designed similarly to the Earth system described in this section.

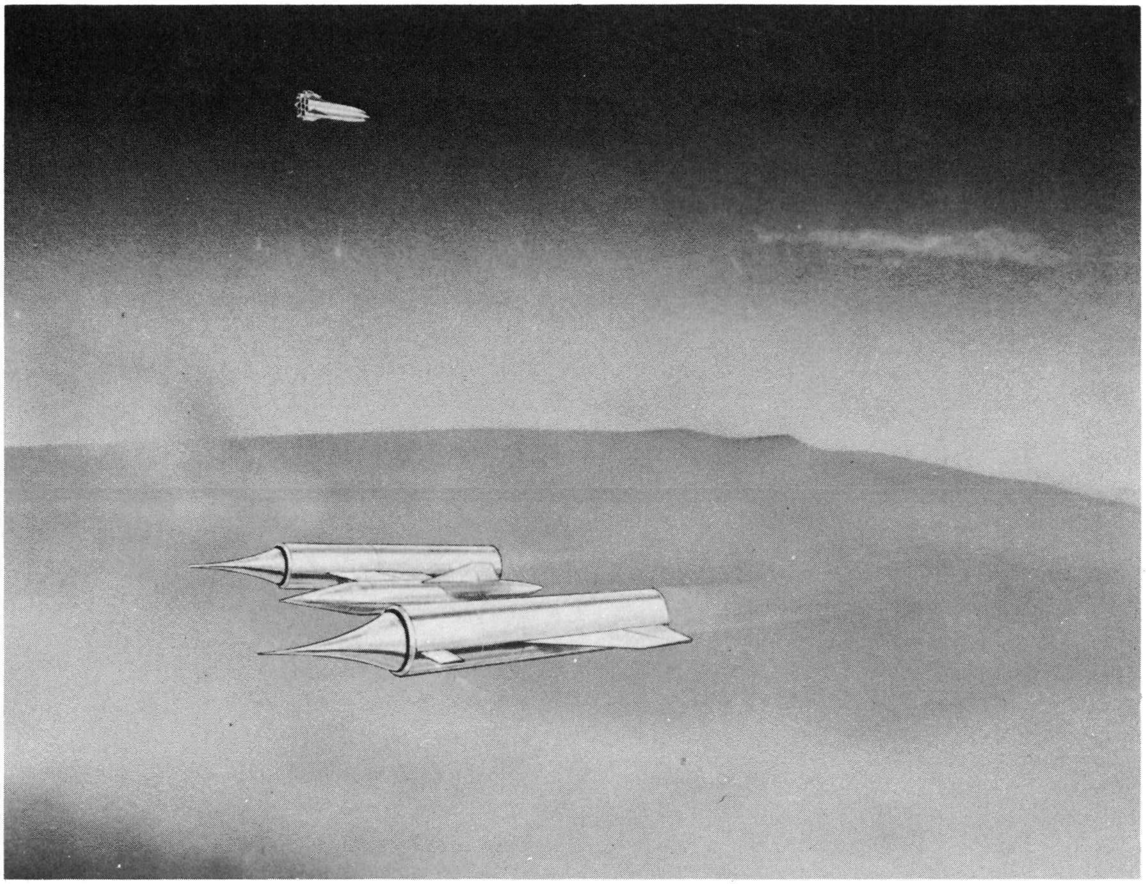

Fig. 12.10 – Nuclear flight in Martian atmosphere

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12.1 REFERENCES

1. Szekely, T., "A-136 Nuclear Turbo-Ramjet Power Plant - An Invention Disclosure," GE-ANPD, APEX-727, April 10, 1961.

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