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RECONNAISSANCE AIRCRAFT WEAPON SYSTEM

SUMMARY REPORT

BELL Aircraft CORPORATION

CONTRACT NO. AF33(616)-2419
SUPPLEMENTAL AGREEMENT NO. 2

REPORT NO. D143-945-029

1 DECEMBER 1955

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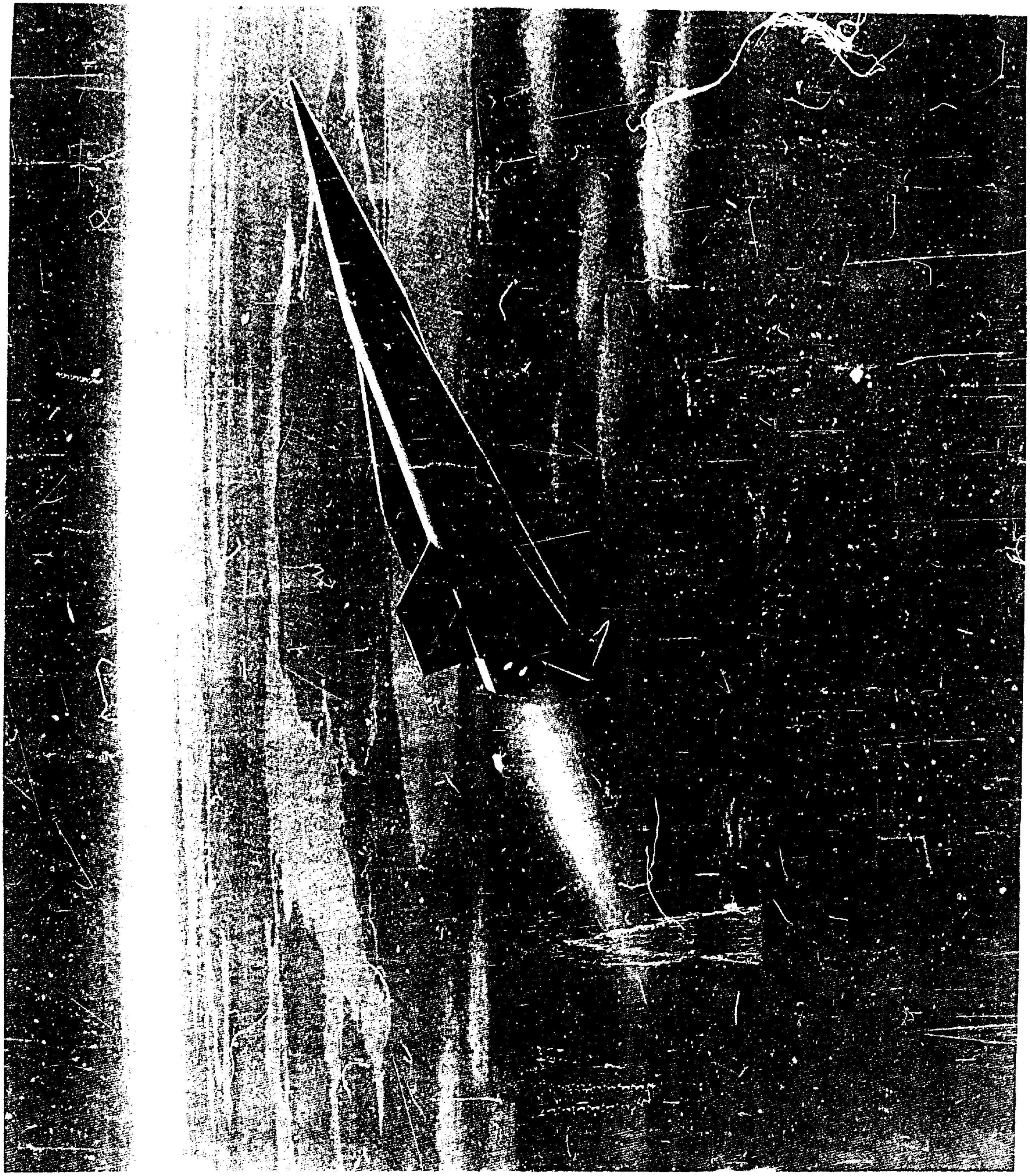

FOREWORD

During the period from 2 May 1955 to 1 December 1955, the Bell Aircraft Corporation conducted a study program for the Directorate of Systems Management, Deputy Commander for Weapon Systems, RDZ 1SB, Headquarters Air Research and Development Command under Supplemental Agreement 2 (56-284) of USAF contract AF33(616)2419 — RDO No. R 441-47. The primary objective of this study is to conduct analytical investigations and design studies of a weapon system which adapts the MX-2276 concept to satisfy or exceed the criteria presented by Development Requirements, System No. 118P. A secondary objective is to supplement and advance studies previously made under this contract.

The work accomplished during this program is reported in the following series of reports:

D143-945-024	Aerodynamics
D143-945-025	Structures
D143-945-026	Navigation and Control
D143-945-027	System Design
D143-945-028	Photographic Subsystem
D143-945-029	Summary Report

In addition to these reports, separate powerplant proposals are submitted by the Bell Aircraft Corporation (Report No. 02-945-118) and North American Aviation, Inc. (Report No. PC-177P).

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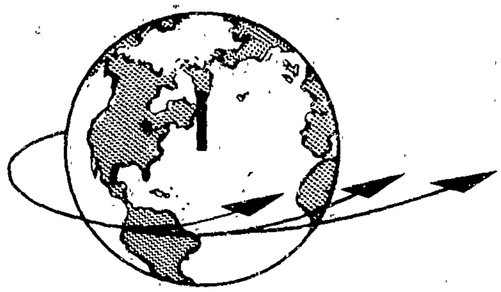
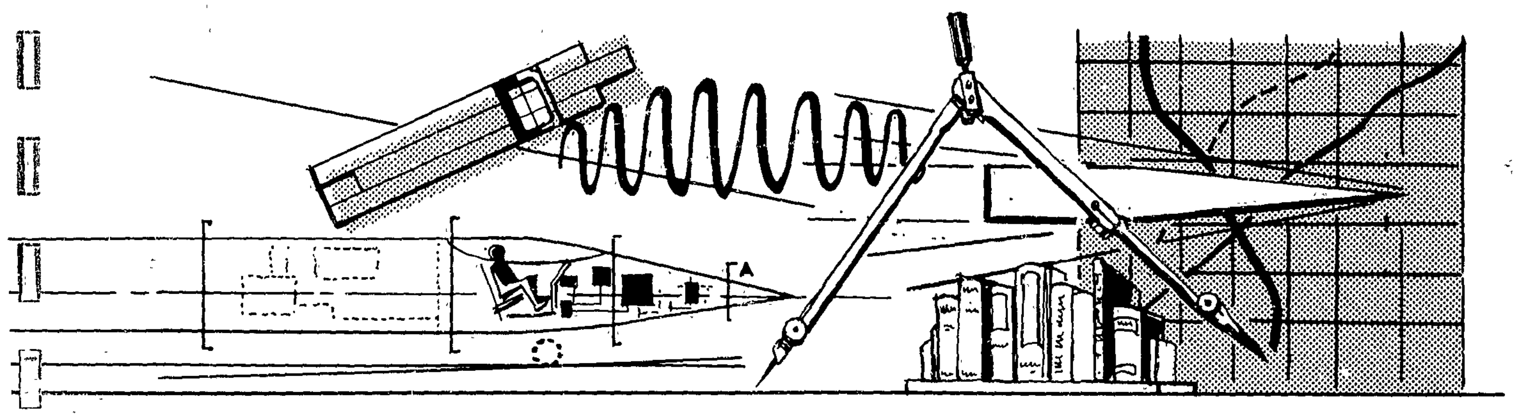


ABSTRACT



This report is concerned with a weapon system which adapts the rocket-boosted, glide airplane principle of the MX-2276 to a reconnaissance system possessing radar, ferret, and photographic capabilities. The configuration presented is a two-stage vehicle consisting of a booster, which is either expendable or recoverable by parachute, and a manned, rocket-powered glide airplane which carries the reconnaissance equipment and performs the mission.

On the basis of an analysis of the crew functions it was decided that a crew of two would be necessary. The first crew member — the pilot — monitors the rate of energy expenditure with range and flies the airplane in emergencies and during landing. The second member — the navigator — monitors the automatic navigation system, takes over the navigation in emergencies, and performs any duties in connection with the reconnaissance system.

The staging and weight distribution between stages is the result of an analysis conducted to obtain the lowest take-off weight for a system capable of attaining a cut-off velocity of 16,600 feet per second at 165,000 feet altitude. These conditions were selected, on the basis of previous work, as those necessary to attain the required range. The aerodynamic performance parameters for the booster-final stage combination were calculated in the subsonic and supersonic regimes. The parameters for the airplane only were calculated in the subsonic, supersonic, and hypersonic regimes. An investigation of shock wave-boundary layer interaction showed that the phenomenon was not important at M = 16. The ascent of the vehicle was computed for both vacuum conditions and for existing atmospheric and drag conditions. The results indicate that the vacuum trajectory is an excellent approximation of the actual trajectory. The glide performance of the airplane, after boost, flying at maximum lift-drag ratio was computed. The total range of the system, including boost, was calculated to be 4680 nautical miles, with a total flight time of 61 minutes. An analysis of the landing characteristics of the airplane shows it to be in a category with contemporary research and experimental aircraft. Preliminary stability and control calculations were made.

The magnitude of the aerodynamic heating over the vehicle was determined with special reference to the fuselage nose and surface leading edges. The heating, although high, was within the limits imposed by proposed structures. The penalties in range from turning the vehicle through various angles at various load factors were determined. It was found that better maneuverability was possible than had been expected. Analysis of the outside environment at the camera location showed the effects of shock waves and boundary layer on light ray deviation to be small. Preliminary results on the luminosity of this environment showed this condition to be small also.

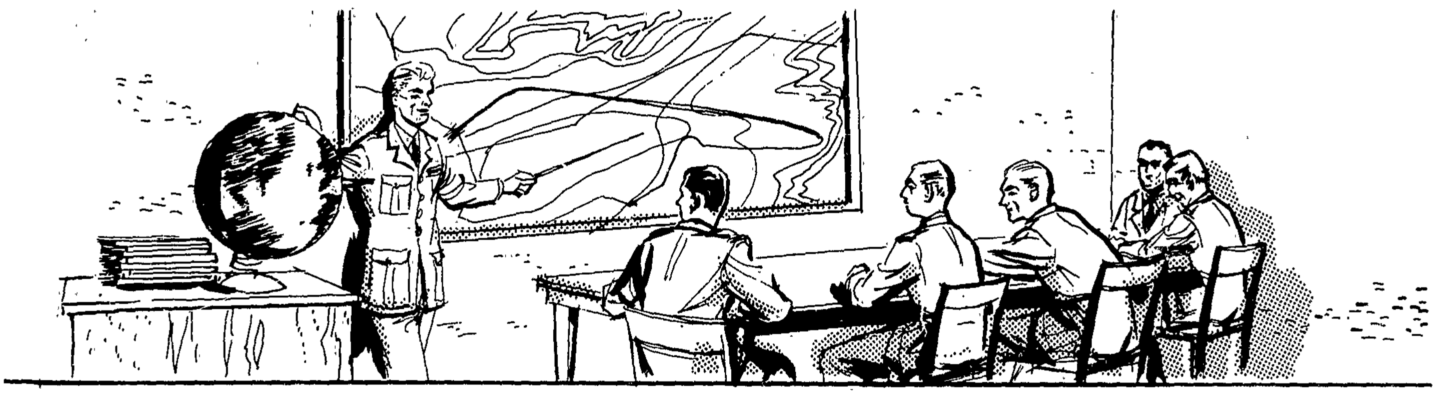
Preliminary design criteria and structural loads were determined. The fuselage structure is basically a ring-stiffened aluminum alloy shell with a conical nose. The aluminum shell is cooled by water circulating through passages in the aluminum skin. This skin is protected by a layer of insulation between the aluminum and the outer wall. The outer wall is a skin of Inconel X or Haynes Alloy No. 25 backed up by a corrugation of the same material. The wing and vertical tail surfaces are constructed of aluminum alloy trusses, acting as ribs and beams; and each structural member is cooled by water circulating through passages in the members themselves. Aluminum skin and insulation are not used. The external surface is formed by the skin and corrugation construction used on the fuselage.

The leading edge and nose areas are cooled by circulating molten lithium through these areas and permitting it to boil away. The skin and cooling passages are formed by a coated molybdenum alloy. The booster is protected by the same outer wall construction as the fuselage. However, no cooling or insulation other than an airspace is necessary.

Automatic navigation and control are provided by an all-inertial system. The stable platform which serves as a reference for this system also provides reference data for the reconnaissance information and is used as a reference for stabilizing this equipment.

Propellants for the booster are LOX-JP-4, while for the airplane LOX-Fluorine-JP-4 are recommended.

II INTRODUCTION



In 1951, Bell Aircraft Corporation conceived the idea for an advanced strategic weapon system which would be capable of performing both bombardment and reconnaissance missions. The weapon would fly at extremely high speeds at very high altitudes, thus providing an almost invulnerable system.

On 2 May 1955, Bell Aircraft completed a one-year study program under Air Force Contract 33(616)-2419. The purpose of the contract was to study the more important problems that might be encountered in the design and development of such a weapon system. The weapon system forming the basis of this study consisted of a manned hypersonic glider and liquid rocket booster units, which accelerated the airplane to a maximum speed of 22,000 feet per second at an altitude of 214,000 feet.

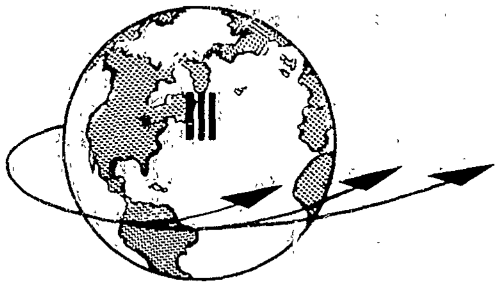
On 1 September 1955, Supplemental Agreement No. 2 (56-284) was added to contract AF33(616)-2419. The major effort, as outlined in the work statement, was to be devoted to analytical investigations and design studies of a weapon system which adapts the MX-2276 concept to satisfy or exceed the criteria presented by the Development Requirements, System No. 118P.

In addition, a limited effort was to be devoted to special studies to supplement and advance those previously made under this contract. The contract completion date was extended from 2 May 1955 to 1 December 1955 to permit this work to be performed.

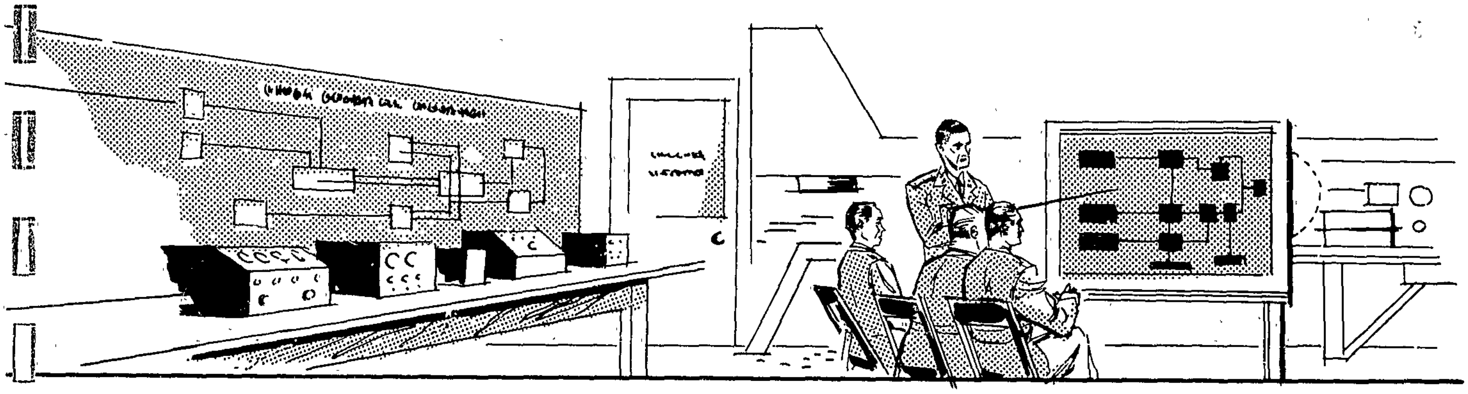
System No. 118P outlines the requirements for a piloted special reconnaissance weapon system for use in tactical and strategic reconnaissance operations. A preliminary examination of the MX-2276 approach, which showed the development of the final weapon system in logical progressive steps, indicated the possibility of adapting the boost-glide concept to satisfy this military requirement. The airplane and one booster stage of the MX-2276 system appeared to possess the required range and altitude capability.

Presented herein is a summary of the analytical investigations and preliminary design analysis of this special airplane weapon system. In addition to the work performed at Bell Aircraft, a study of the photographic reconnaissance equipment was performed by Chicago Aerial Industries, Inc. under purchase order from Bell. Also, the Rocketdyne Division of the North American Aviation Corp. submitted a proposal for the booster and airplane engine systems, and the Rocket Department of Bell Aircraft submitted a proposal for the airplane engine system.

In a study of this type, it is necessary to conduct many of the technical analyses simultaneously. In some cases the results of these technical studies become available for incorporation into the configuration at earlier dates than others. For this reason, many of the analyses described herein were conducted on different configurations. However, the final configuration described in this report incorporates the results of all these efforts. The particular configuration to which each analysis applies is described with that analysis.



III WEAPON SYSTEM CONCEPT



A. REQUIREMENTS

The military requirements for a piloted special reconnaissance weapon system for use in tactical and strategic reconnaissance operations as outlined in Development Requirements System No. 118P are listed below.

1. Operational Features

a. Capabilities of daylight photographic, high-order ferret, and high-resolution radar reconnaissance.

b. Maximum survival in the air by high flight altitudes and low detectability.

c. A high degree of weapon system reliability.

d. Capability of take-off, penetration, and recovery under poor weather conditions.

e. Operational availability in 1960-1965 time period.

2. Performance Objectives

a. Basic mission combat zone altitude of 100,000 feet, with 150,000 feet desired.

b. Basic mission radius of 1500 nautical miles, with 2000 nautical miles desired. An alternate permissible mission having a total range of 3000 nautical miles, with 4000 nautical miles desired. A total of 300 nautical miles at the beginning and end of the total distance may be considered to be outside the combat zone.

c. Penetration speed will be the maximum possible. However, altitude and radius are of greater importance.

3. Operational Capabilities

a. Photographic search installations capable of resolving surface dimension detail of 100 feet or smaller. Photographic detail targeting installations capable of resolving detail to the degree that objects approximately 20 feet on a side can be positively identified.

b. Ferret surveillance from 30 mc to 40,000 mc.

c. Radar surveillance using high-resolution techniques.

4. Reconnaissance Functions

It is desirable, but not required, that a single vehicle should be capable of conducting all reconnaissance functions on one mission.

B. SYSTEM PERFORMANCE

The very high speeds and altitudes of the MX-2276 rocket boost-glide concept make pos-

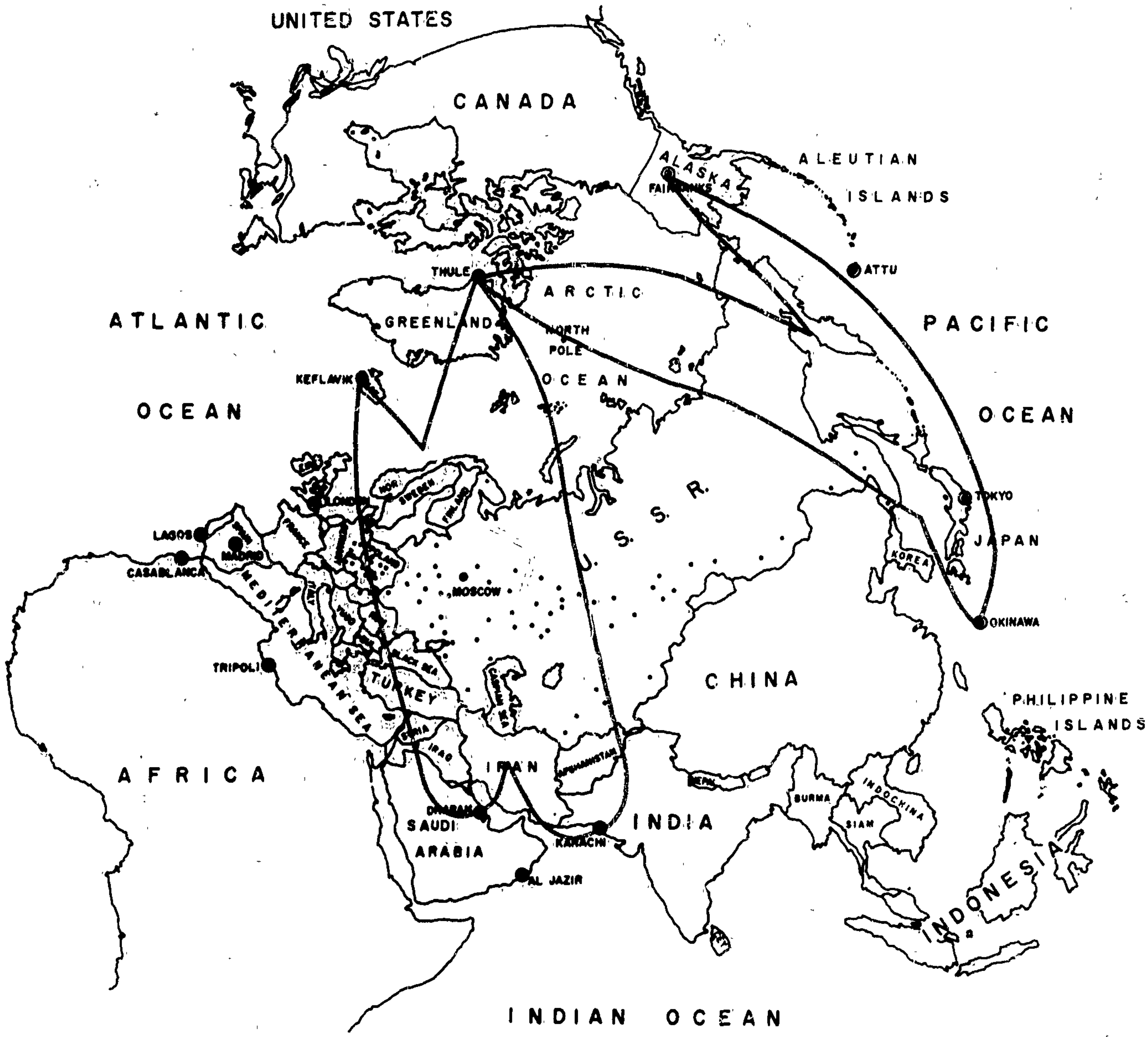


Figure 1. Coverage with Two-Stage 4650-Mile System

sible a weapon system capable of meeting or exceeding the operational and performance capabilities enumerated in the previous section. From previous work, it was estimated that a two-stage system in which the second stage achieved a velocity of 16,600 feet per second and an altitude of 165,000 feet at the end of launch, would attain the desired range. With the objective in mind, the design of such a configuration was undertaken. At the same time, studies of the other subsystems and performance functions were investigated.

One of the performance items studied (Reference 1) consisted of operational performance using various overseas bases. The aircraft performance used for this analysis consisted of a vehicle capable of 4650 nautical miles useable range from launch to landing. The area which can be covered with this weapon system

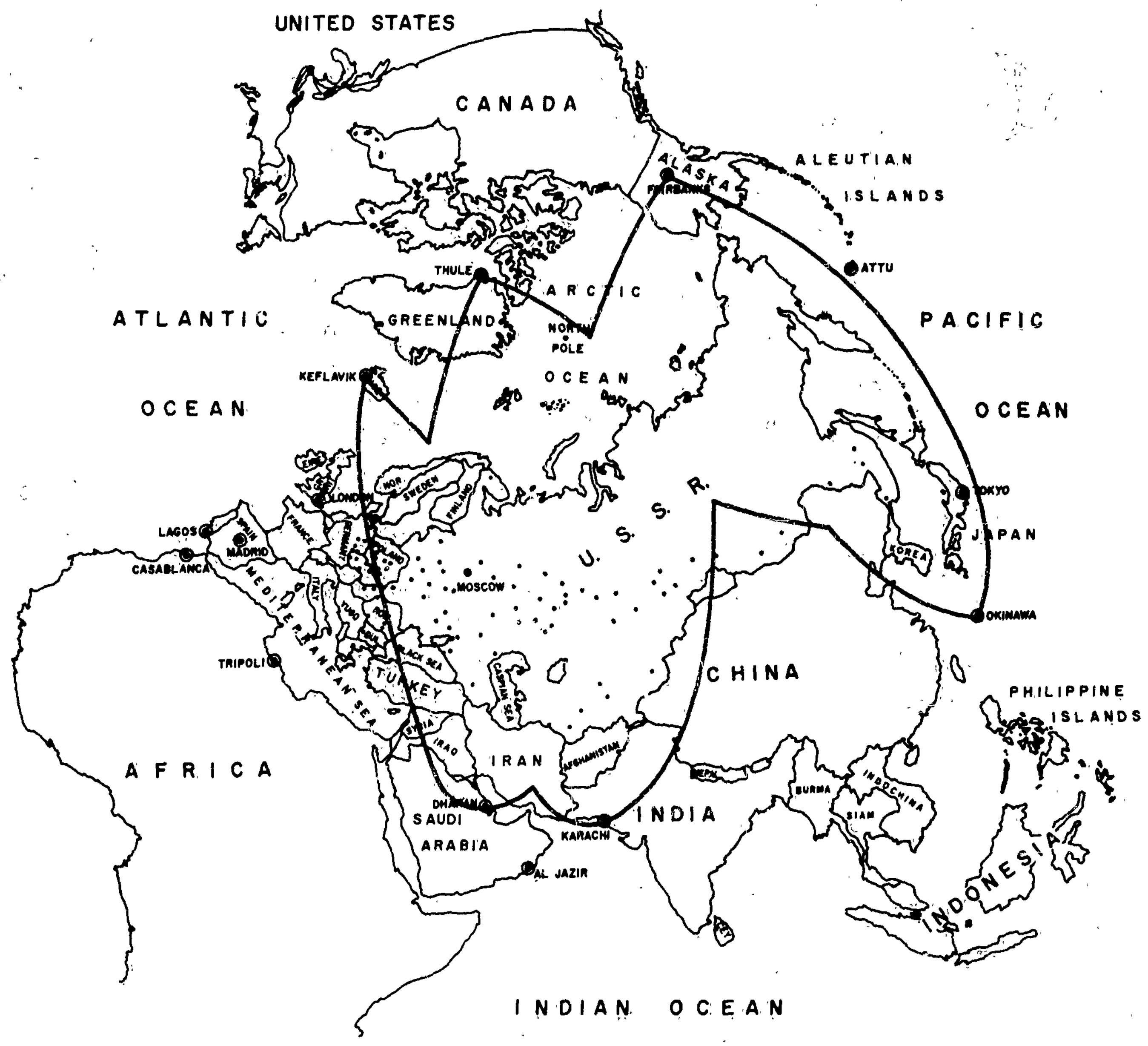
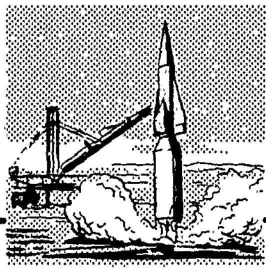


Figure 2. Coverage with a 5500-Mile System

by utilizing Keflavik, Thule, and Fairbanks as launching sites and Dharan, Karachi, Tokyo, and Okinawa as landing sites is shown in Figure 1. In order to obtain an idea of the necessary range to get 100% coverage of the USSR from these bases, Figure 2 was prepared for a system capable of 5500 nautical miles useful range. On the basis of these data, it is estimated that a vehicle capable of a useful range of approximately 5700 nautical miles will be necessary to obtain total coverage of the Soviet Union using North to South flights.



C. CONFIGURATION

1. General

The configuration selected for the reconnaissance weapon system is a two-stage rocket-powered vehicle. This configuration consists of (1) an unmanned rocket-powered booster, (2) a manned rocket-powered glide aircraft which carries the reconnaissance equipment and performs the mission, (3) a crew consisting of pilot and navigator in the airplane, (4) an automatic navigation and control system for the airplane, and (5) the reconnaissance equipment necessary to perform radar, photographic, and ferret reconnaissance.

Figure 3 is a three-view of the weapon system. The tandem arrangement was selected because it provides simpler separation, lower drag, and a more readily accomplished physical attachment of the airplane and booster than a parallel arrangement. In addition, the alignment of drag, mass, and thrust are superior. The configuration shown has substantially lower weight than predicted in earlier estimates made on the basis of the work done on the three-stage configuration. In addition to lower weight, more hardware is included in the recoverable airplane and less in the booster which may or may not be recoverable. Also, the center of gravity of the two-stage vehicle lies further forward, with resulting favorable effects with respect to aerodynamic stability and control. Stability and control during boost are provided by gimballed rocket motors in the booster.

2. Airplane

The airplane configuration (Figure 4) is considered to be a reasonable compromise between the major aerodynamic and structural considerations. In general, the configuration was obtained by incorporating the best hypersonic characteristics without losing sight of the fact that the vehicle must fly satisfactorily at all speeds and have acceptable landing characteristics.

The main feature of this configuration is the low aspect ratio, highly swept wing. The combination of sweepback and aspect ratio chosen was based on many considerations, including gust loads in ascent as well as the factors previously mentioned. This planform also affords a very favorable structural weight. The wing has a flat bottom, with a typical section consisting of a wedge to the 35% chord station at root (39% at tip) followed by parallel upper and lower surfaces. The trailing edge is full blunt. Thickness is a constant 4% over the span. The leading edge is rounded to ease the heating problem, the radius being 1/4 inch in a plane perpendicular to the leading edge.

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TAKE-OFF GROSS WEIGHT	201,050 lb
Total Propellants	166,350 lb
Design Thrust at Take-Off	300,000 lb
BOOSTER (GROSS WEIGHT)	142,450 lb
Propellants O_2 and JP-4	126,450 lb
RECONNAISSANCE AIRPLANE (GROSS WEIGHT)	58,600 lb
Propellants $F_2 + O_2$ and JP-4	39,900 lb

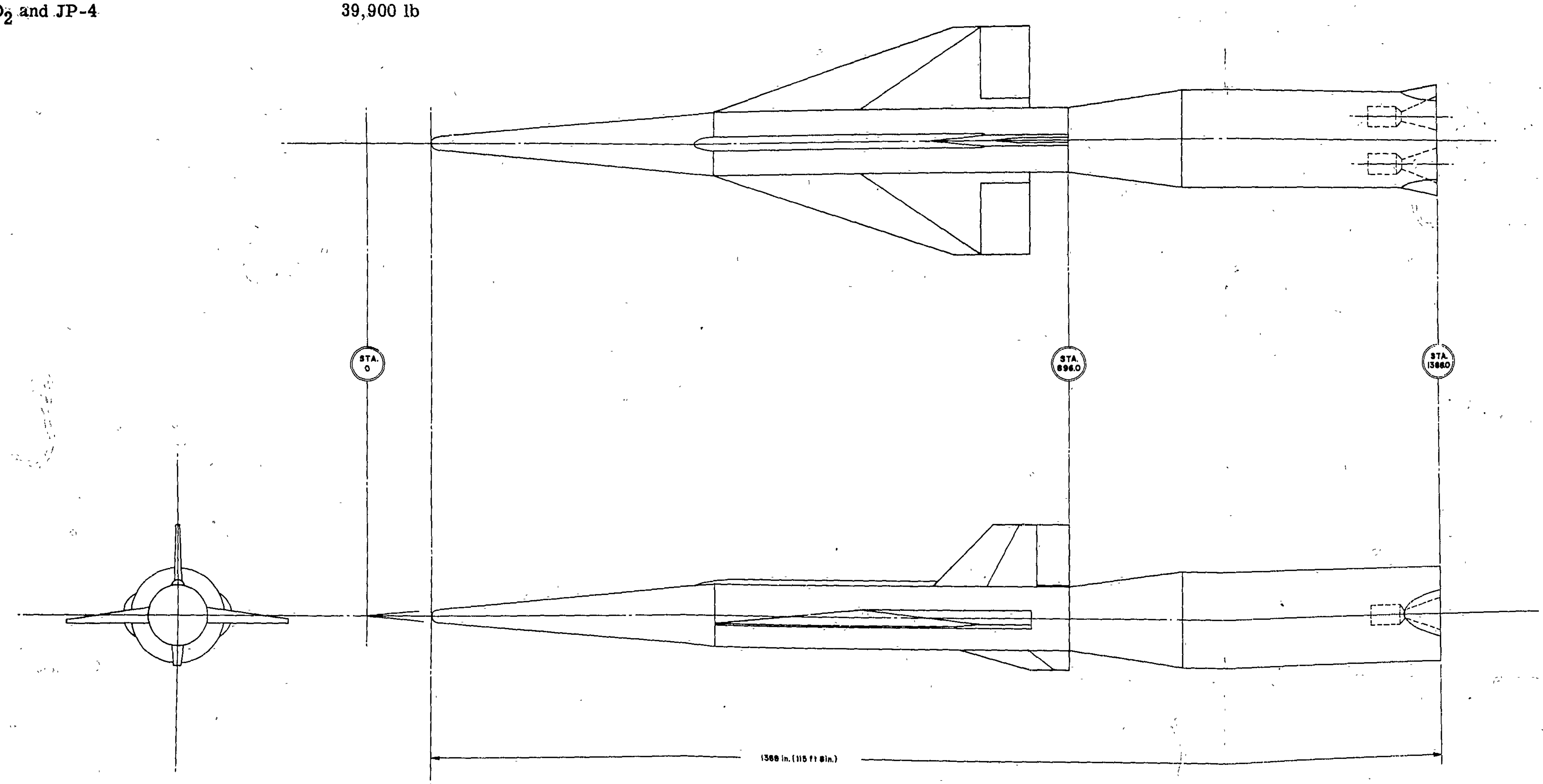

Figure 3. Three-View: MX-2276 Two-Stage Reconnaissance Vehicle

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DIMENSIONS

Wing Span (Over-All)	286.3 in.
Aircraft Length (Over-All)	896.0 in.
Aircraft Height (Over-All)	177.0 in.
Landing Gear Tread	89.6 in.

WEIGHTS

	Photo Reconn.	Ferret
Empty Weight	17,876	16,926
Useful Load	40,724	40,724
Design Gross	58,600	57,650
Landing Weight	17,433	16,483
Propellants F_2+O_2 and JP-4	39,900	39,900

CENTER OF GRAVITY

	Sta.	Sta.
Design Gross Weight	597.4	593.3
Extreme Forward	597.4	593.3
Extreme Aft	680.6	671.0

TAIL

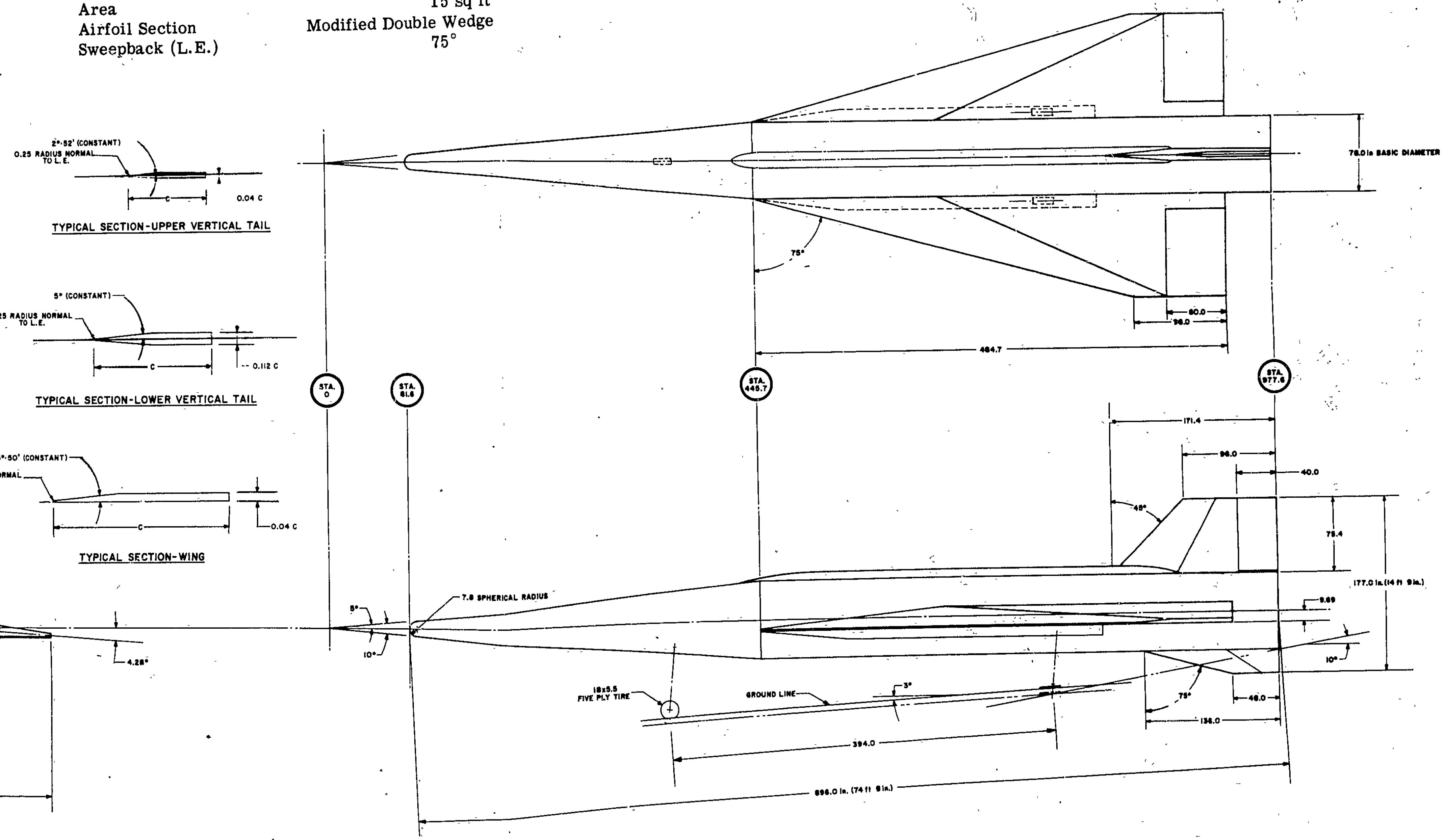
Fin and Rudder	
Area	70 sq ft
Airfoil Section	Modified Double Wedge
Sweepback (L.E.)	45°
Ventral Fin	
Area	15 sq ft
Airfoil Section	Modified Double Wedge
Sweepback (L.E.)	75°

WING

Area (Exposed)	420 sq ft
Airfoil Section	Modified Wedge
Incidence	0
Sweepback (L.E.)	75°
Aspect Ratio (Subsonic)	0.834
Cathedral	4.28°

POWER PLANT

230,000-lb Thrust Chambers A_e/A_t=25
Propellant F_2(70)+O_2(30) and JP-4
Gas Driven Turbine Pumps



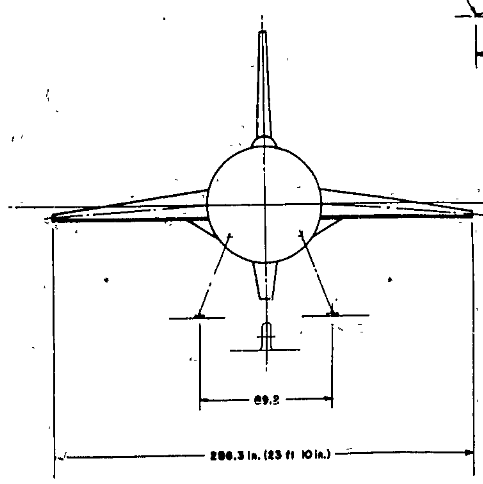


Figure 4. Three-View: MX-2276 Final Stage, Photo Reconnaissance Version

The fuselage is a simple cone-cylinder with the nose blunted sufficiently to ease the heating problem, but not enough to significantly increase the drag. The very slender nose cone provides low heat input as well as low drag. The symmetrical nose provides higher lift-drag ratios and reduced temperature over the body in general, as compared with a nose configuration drooped to provide a straight bottom line. The latter configuration provides reduced temperatures on the bottom of the nose, however, and some downward tilting may prove beneficial from an over-all standpoint.

The vertical tail consists of an upper vertical surface, which is adequate for stability and control at low speeds, and a lower vertical surface to augment directional stability at hypersonic speeds. The lower surface is jettisoned or folded prior to landing.

The interior arrangement of the aircraft with typical photographic reconnaissance equipment is shown in Figure 5. The camera array shown in this figure is for the search mission. For other types of reconnaissance missions, this equipment is replaced by the proper type for the mission involved.

This configuration accommodates a crew of two in a side-by-side seating arrangement which tends to simplify the instrumentation. Wherever possible, standard equipment is used. However, some specialized instrumentation will be necessary for the crew. The landing gear is of the tricycle type with a nose wheel forward and main skids aft. The space allowance for equipment is sufficient to accommodate a 4600-pound bomb with a 3000-pound special warhead. No analysis of the bomber system has been made in this study.

The basic fuselage structure is insulated and cooled. Aluminum is the primary structural material. This structure is of the double-walled type with a layer of insulation between the hot outer skin and the inner structure which is cooled by water. The primary structure of the wing and tail surfaces is a gridwork of aluminum alloy ribs and beams in which each member is water-cooled. The fuselage and the outer wall panels are used to form the aerodynamic contour. An entirely different type of cooling system is necessary for the leading edges and nose because of the much higher temperatures and heat fluxes encountered. A feasible design for a lithium cooling system capable of cooling this area has been developed.

The propulsion system is a 60,000-pound thrust liquid rocket engine. The present engine system and tank layout is for a propellant combination of liquid oxygen and liquid fluorine with a JP-type fuel.

3. Booster

The booster configuration is a very simple design consisting chiefly of propellant tanks and a rocket engine package. The tank ends are hemispherical with a nested arrangement between the LOX and fuel tanks in order to keep weight to a minimum. The stiffening effect of the internal pressure within this integral tank structure is substantial and results in a reduced over-all structural weight. However, the booster does not require internal pressurization in ground handling or storage, or when supporting the airplane at launch.

The aerodynamic heating of the booster is appreciably lower than for the airplane. However, some form of insulation is required if an aluminum primary structure is used. It was found that air-insulated aluminum provided a lower structural weight than an alternative uninsulated steel structure.

The propulsion system is a two-chamber 300,000-pound thrust engine now under development. No auxiliary power generation problem exists, since the propellant pumping system can provide ample shaft power throughout the entire booster flight, thus reducing electrical and hydraulic power generation to mainly installation problems.

The dimensions of the booster are such that they can be carried in railway freight cars, thus presenting minimum shipping problems. Air transport using a C-124 type aircraft and air ferry using a B-36 aircraft are also feasible for both the booster and the airplane.

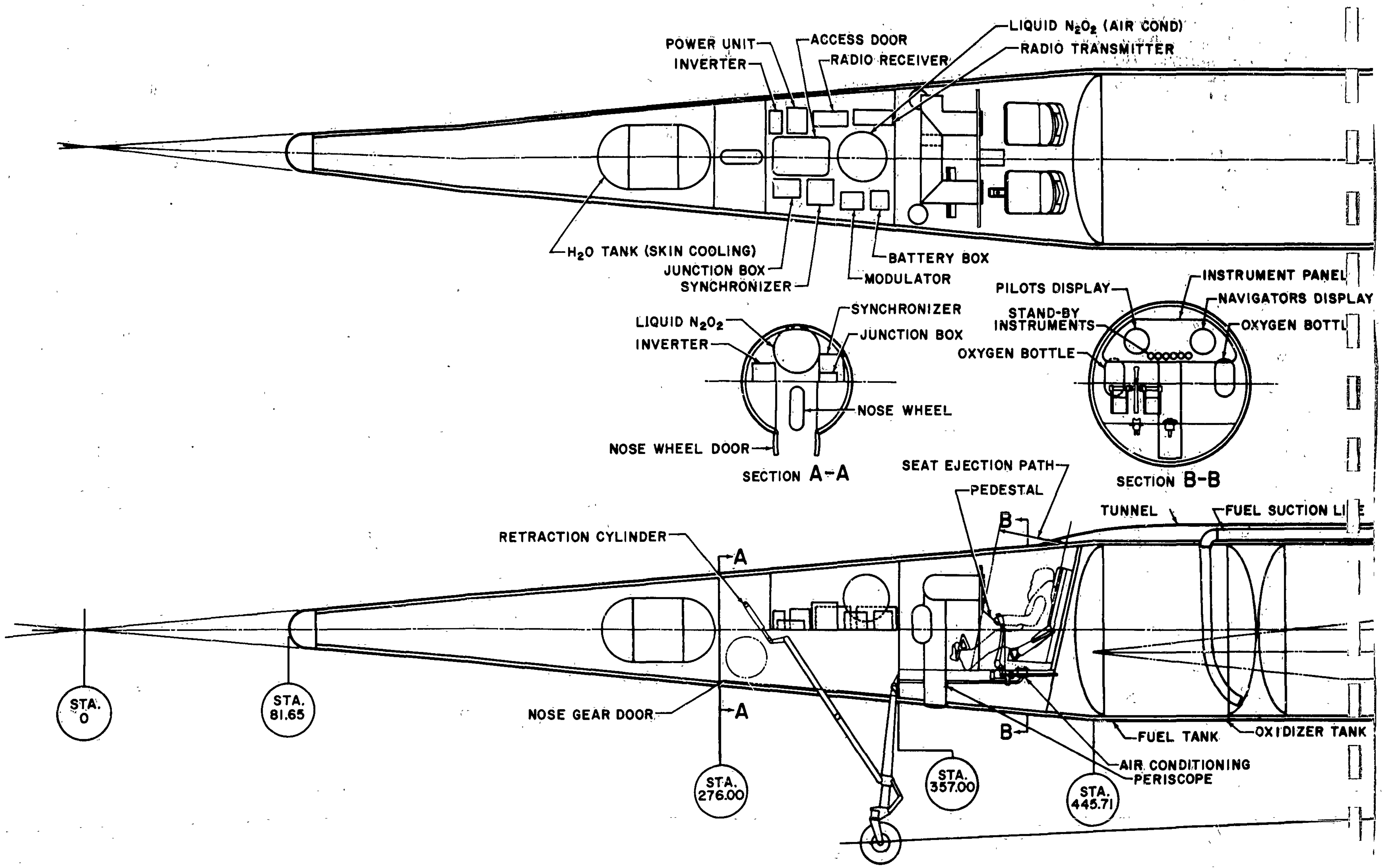


Figure 5. Inboard Profile: Two-Stage Reconnaissance Vehicle

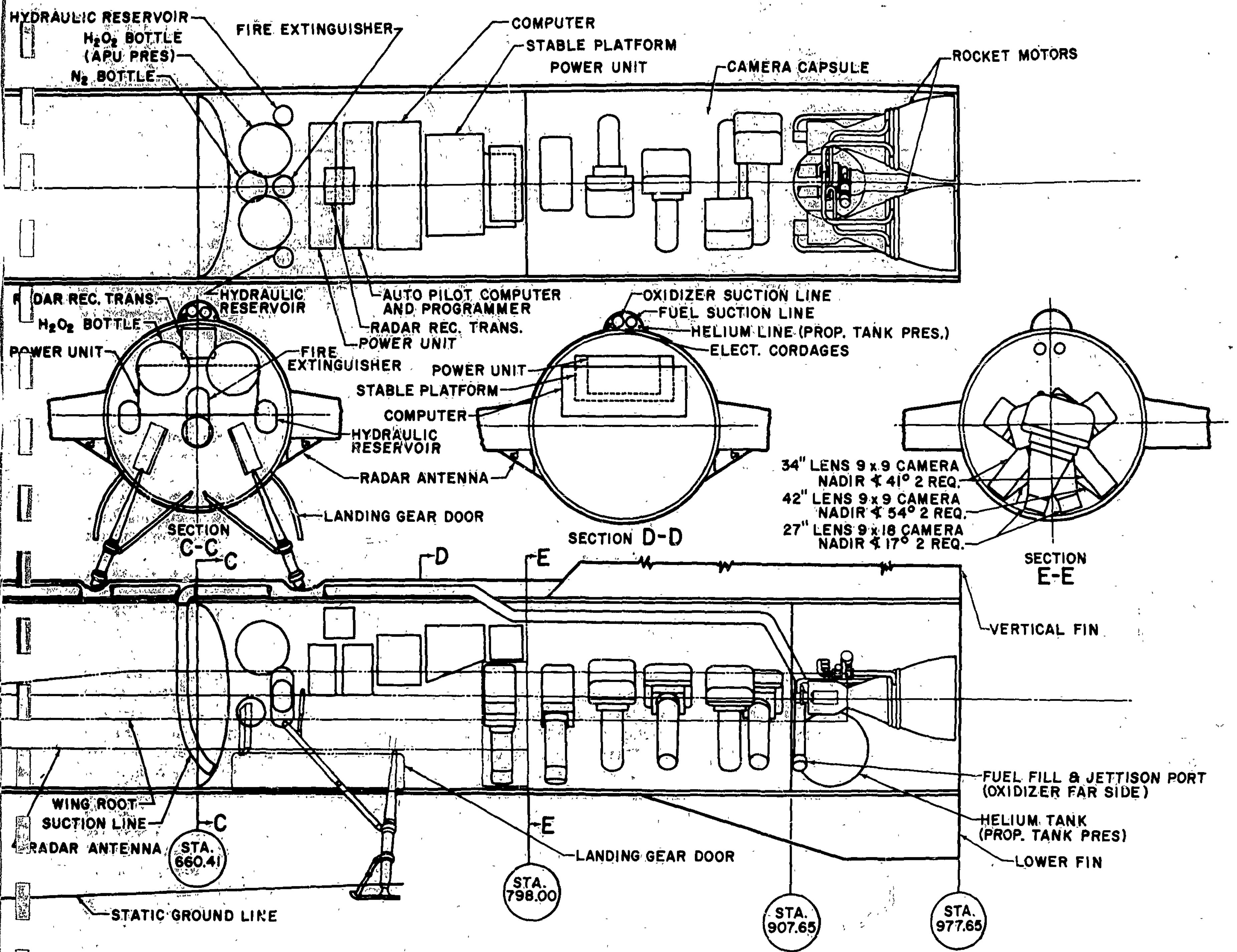
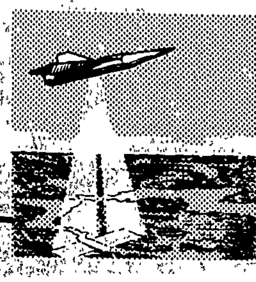


Figure 5. Inboard Profile: Two-Stage Reconnaissance Vehicle



D. TYPICAL RECONNAISSANCE MISSION

To illustrate the over-all concept of this weapon system, a typical mission has been divided into five portions each of which will be discussed separately.

1. Preparation for Take-Off

Prior to take-off, the vehicle must be placed in launching attitude and fueled, the inertial navigation system must be erected and aligned, and the desired flight path must be programmed into the navigation system. Along with flight path data must be programmed starting and stopping signals for the reconnaissance systems and their associated data storage elements. The crew will spend one to two hours prior to take-off pre-breathing oxygen. A portion or all of this pre-breathing may be accomplished during the preflight briefing and aircraft checkout.

2. Take-Off and Boost

The vehicle is launched vertically and automatically programmed into the path which will yield the proper glide attitude at the end of boost. The crew is seated such that the men attain a normal seated position when the vehicle turns into the proper glide attitude. During boost, control is maintaincd by gimballed rocket motors. The first stage burns for approximately 102 seconds. At the end of this time, the vehicle has attained a velocity of 5400 feet per second and an altitude of 65,000 feet. Stage I, the booster, is then released and the rocket motor in Stage II, the airplane, is ignited and burns for 226 seconds. Stage II attains a velocity of 16,600 feet per second an an altitude of 165,000 feet. At this time, the glide-cruise phase of the flight begins.

During the burning period of Stage I, the vehicle is controlled by gimballed motors. Following the burnout of Stage I, control is obtained by aerodynamic surfaces.

The Stage I booster is either expendable or may be recovered by parachute.

3. Cruise

Throughout the cruise, the vehicle is in gliding flight at maximum lift-drag ratio to achieve the maximum possible range. The desired flight path is maintained by means of the inertial navigation system which continuously computes the vehicle position and compares it with the prelaunch programmed position. Any errors are used to compute signals to the autopilot systems, which then control the vehicle so as to provide a minimum difference in these readings.

For reference, the navigator is provided with a map, driven by the inertial system, which shows the position of the airplane with respect to the ground as computed by the inertial system. Throughout the cruise phase, the navigator also observes the area traversed, using both the radar and the visual equipment. The radar presentation can be recorded automatically to provide a permanent record. Since the radar utilizes a side-looking array, any points of interest do not become visible on the radar presentation until they are abreast the airplane. With the visual system, the crew will be able to look ahead and behind, weather conditions permitting, of course. Both the radar and visual presentations will be placed on the same display surface as the map, with any of the three available at the selection of the crew.

The inertial navigation system is considered to have sufficient accuracy for reconnaissance purposes without any outside sources of correction. Hence, the radar and visual displays will not be used for map-matching purposes, to provide correction data for the inertial system. Instead, they will provide information to the crew for monitoring the weapon system operation and could be used as secondary navigation devices in the event of malfunction of the inertial unit.

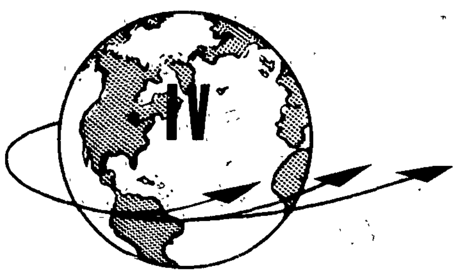
During cruise, the airplane is flown completely by the autopilot. The pilot will not have to fly the airplane except when the automatic system ceases to operate correctly. Similarly, the navigator will be required to navigate only when the inertial system malfunctions.

4. Reconnaissance

The reconnaissance portion of the flight actually occurs during cruise; however, for clarity it will be discussed separately. The vehicle has three separate reconnaissance functions, photographic, ferret, and radar. Insofar as possible, all three of these functions will be conducted automatically with the crew serving chiefly as a monitor. However, the navigator will be able to perform some adjustments of this equipment, if necessary. For all three types of reconnaissance, the airplane is maintained in a level attitude. For the photographic and radar missions, the flight paths will consist of a series of paths parallel to the great circle route between the take-off and landing sites. For the ferret mission, strict adherence to this type of path is not as essential. For all three systems, the reconnaissance data are obtained and stored automatically in a manner consistent with the ground handling equipment for the particular type of data being obtained.

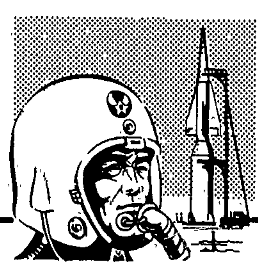
5. Landing

The landing will be accomplished by the pilot with the assistance of landing aids such as ground-controlled approach and radio beams. Insofar as possible, the landing aids will be of the type located outside the aircraft. The flight path will be laid out to terminate at approximately M = 4 and the pilot will take over at this point, locate the landing field, and bring the aircraft in. The landing will be unpowered. The stalling speed of the airplane is comparable to present-day high-speed aircraft.



RESULTS OF STUDY EFFORTS

A. CREW EFFECTIVENESS



1. General

This weapon system has been designed for completely automatic operation in its normal mode. However, a crew has been included to increase the system reliability and provide the weapon with capabilities which cannot be achieved by an automatic system. The normal function of the crew will be to monitor the operation of the automatic equipment. In order to perform this function the proper data must be made available to the crew in a form which they can understand. In order to utilize this information to improve the weapon system reliability, the crew must have available means for implementing any actions that are necessary as a result of decisions based on these data. The capabilities which the crew adds to the weapon system include judgment, decision, anticipation, and the ability to assimilate and translate into action an extremely wide variety of data.

In order that the crew can execute their duties at a high level of performance, it is necessary to provide them with the best conditions possible. Three major factors affecting their abilities are work load, environment, and time of concentration. The factor of work load can be controlled by providing enough members in the crew. The environment should be kept as comfortable as possible to reduce distraction. The third factor is optimized by keeping the time as short as possible. Fortunately, for this weapon system, the flight time is short enough to fall well within human tolerance limits.

Of the preceding three factors, the second has been discussed to a considerable degree in Reference 2, where it has been shown that the crew can be maintained in a relatively comfortable environment. The third factor is kept to a desirable value by the weapon system characteristics. This leaves only the first factor to be evaluated.

2. Crew Functions

Any analysis of the number of crew members necessary must begin with a determination of the amount of work to be performed. The performance of this weapon system may be measured by two basic criteria: the accuracy with which the vehicle traverses the desired ground path, and the efficiency with which the potential energy, provided by the high velocity, is transformed into range. To monitor the accuracy, the crew must compare the position as it actually is with the position as the automatic system says it is, and with the position as it should be. To monitor the efficiency with which the energy is utilized, the rate of dissipation of velocity and altitude with range must be compared with the ideal rates of dissipation.

The high speed of the vehicle is advantageous in that it reduces the over-all time of flight. This same speed requires very rapid decisions and actions on the part of the crew, since any deviations caused by the automatic system may result in extremely large errors in azimuth or range in a very short time. These errors may very easily be so great that adequate compensation cannot be made if they are not detected and corrected quickly enough. In addition to rapid decisions, this means that the crew must monitor the automatic system very closely in order to detect any troubles as soon as they begin.

The prime function of the crew, therefore, will be to monitor the two major phases of the automatic system operation, accuracy and efficiency. When any trouble develops the crew will have to decide on the proper procedure and take over whatever functions of the automatic equipment are necessary. In addition to these major functions the crew will also monitor reconnaissance equipment, observe activities and areas of interest on the ground over which the airplane passes, land the final stage, and perform other related activities.

3. One-Man vs. Two-Man Crew

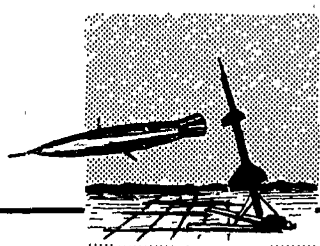
A relatively detailed evaluation of the crew work load compared with the increase in system weight caused by the addition of the crew is presented in Reference 1. This evaluation indicates that the crew work load is great enough to require the full-time concentration of two men. The work breaks down quite readily into two categories — for convenience these will be described as pilot and navigator. The evaluation also indicates that the second man is carried at the expense of a relatively small increase in weapon system weight. Since the advantages provided by the crew are obtained only if the crew can work at high performance levels, and the cost of the second man from a weight standpoint is relatively small, a crew of two has been selected for this weapon system.

a. Pilot

The duties of the pilot will include the duties usually performed by a pilot and a flight engineer. Since there is no powerplant operation during cruise, the flight engineering duties will not consist of monitoring the engine operation by means of SFC, RPM, etc. Instead, the important parameters will be velocity, altitude, angle of attack, lift-drag ratios, etc., as a function of range. Since these are parameters with which the pilot is familiar, it is logical that he should assume this portion of the work load. In addition to these flight engineering duties, the pilot will also be responsible for flying the airplane and utilizing whatever automatic equipment is available in the event of emergencies. These duties will include approach and landing after the mission is completed.

b. Navigator

The navigator's duties will be concerned primarily with monitoring the navigation by the automatic systems. He must maintain a continual awareness of the airplane's position at all times and be prepared to take over all or part of the navigation function at any time. His primary source of information will be the inertial reference unit. However, for emergency purposes, a more conventional group of instruments will also be available. In addition to navigation, this member will also be responsible for any duties necessary in conjunction with the reconnaissance equipment in order to obtain the best coverage consistent with conditions existing at the target area.



B. AERODYNAMICS

1. General

During this study, the aerodynamic effort has been concerned with design and analytical studies of a weapon system which adapts the MX-2276 concept to the requirements described in Section III of this report. In addition, investigations were made to supplement and advance those made in previous studies. Wherever possible, the results of present investigations supplementing previous work are integrated with the general configuration analyses and their effects are demonstrated. Since this study effort was limited in length and scope, a design optimization program could not be carried out. For this reason, the configuration shown is not final or optimum; however, it is a realistic configuration and typical of the aircraft needed to perform this mission.

The complete discussion on the aerodynamic studies is contained in Reference 3.

2. Mass Ratio and Staging Determination

From the reconnaissance weapon system requirements, and considering the nature of the flight paths possible with this vehicle, i.e., overflights, it was decided that a configuration with a cutoff velocity of 16,600 feet per second at an altitude of 165,000 feet would be representative of the initial glide conditions necessary to provide the range required. An investigation was then conducted to arrive at the lowest take-off weight of a system capable of attaining these conditions. The effects of varying the propellant loading of the stages, the initial thrust-to-weight ratio of each stage, and the flight path angle at the separation of the first stage were investigated. During the course of the calculations, it was decided to compare two different propellant combinations. One system used propellants consisting of LOX and JP-4 in both the initial and final stages. The other system substituted a combination of LOX-Fluorine and JP-4 in the final stage. The specific impulse of the initial stages was assumed to be 250 seconds. This value was increased to 290 seconds in the final stage using the LOX and JP-4 propellant combinations. These values represented an over-all specific impulse value which included the propellant required to drive the turbine pump. The final stage using the Lox-Fluorine and JP-4 was assumed to have a chamber specific impulse from its propellants of 349 seconds. However, in order to take into account the reduction in over-all impulse from that of the chamber value, the useful propellant loading of this final stage was assumed to be only 94% of its design propellant loading. This assumption was additionally conservative because the performance of this final stage, from a mass ratio standpoint, was calculated assuming the propellants used to drive the turbine pump were not thrown overboard.

After estimates of the gross weights of the final stage having specified useable propellant loadings were available, it was found that the final-stage vehicle having a useable LOX-Fluorine and JP-4 propellant loading of 0.65, with a chamber specifc impulse of 349 seconds, combined with an initial stage having a useable LOX and JP-4 propellant loading of 0.6, was representative of the lowest take-off weight attainable. In these calculations, it was assumed that 3% of the propellants were not useable because of trapping, etc.

In the selection of the initial thrust-to-weight ratio, it was found that the required maneuvering loads at low altitudes could be considerably reduced when an initial thrust-to-weight ratio of 1.45 was used. In this manner, it was hoped to reduce the total loading (gust load plus maneuver load) that would determine the structural weight of the system.

In Figure 6 is presented the time histories of the altitude and velocity of a two-stage system having the aforementioned propellant loadings and thrust-to-weight ratios. The resultant axial and normal accelerations to which the vehicle was subjected during its ascent are shown in Figure 7. The range of the vehicle was not estimated during its ascent as this was, during the preliminary stages of design, of secondary importance.

After the size of the configuration had been determined, additional information indicated that the over-all impulse for the final-stage LOX-Fluorine system would be 340 seconds. Using the assumption that 97% of the

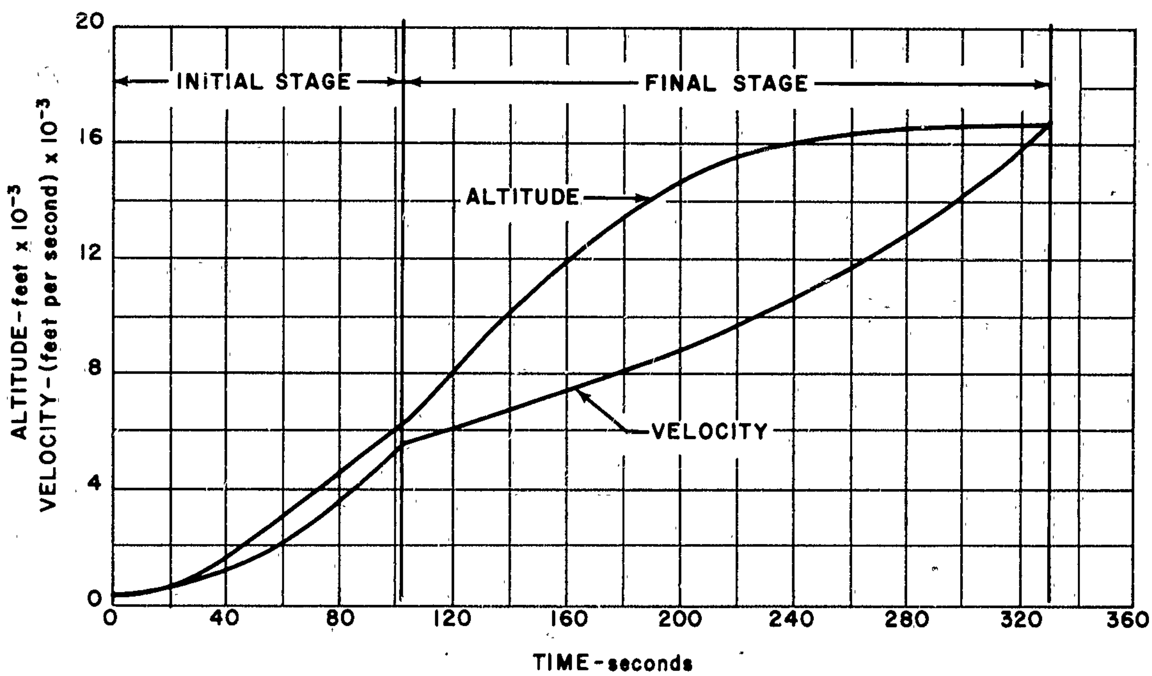


Figure 6. Time Histories of Altitude and Velocity during the Ascent

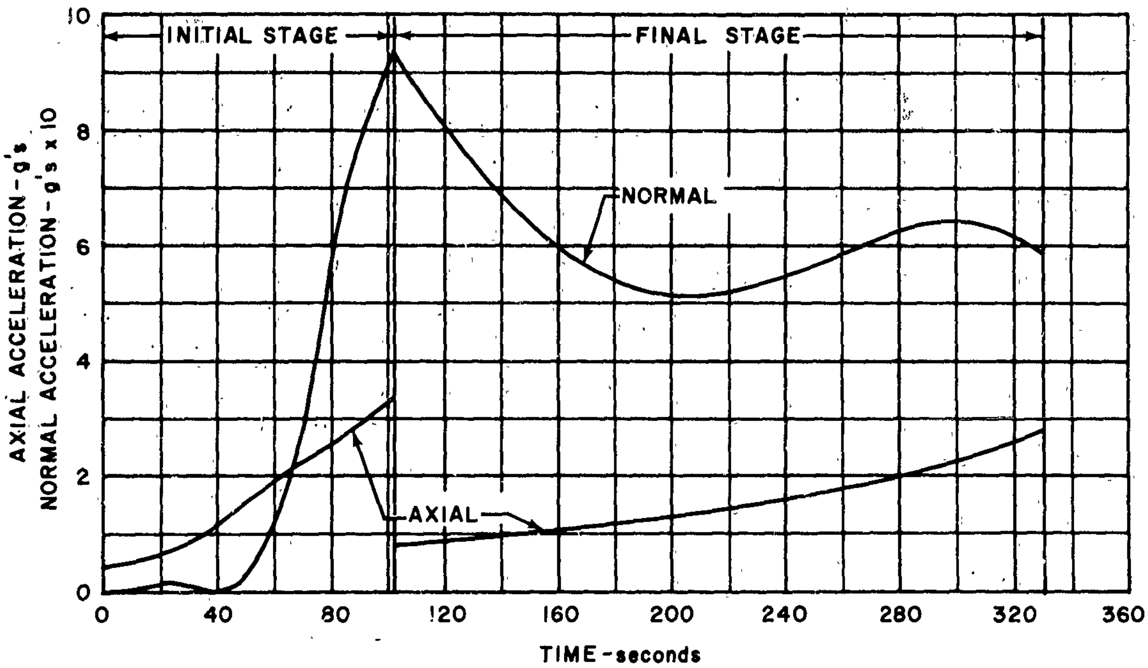


Figure 7. Time Histories of Axial and Normal Accelerations during the Ascent

propellants contained in the final stage were now available at an over-all specific impulse of 340 seconds (i.e., accounting for the propellants used by the gas generator), the velocity increment of the proposed final stage having the design propellant loading at 0.691 (i.e., 0.65/0.94) now had a useful propellant loading of 0.67 instead of the former value of 0.65. The increase in useful propellant loading, even though combined with the reduced specific thrust, resulted in a calculated velocity increment of the final stage of 11,500 feet per second instead of the required 11,200 feet per second. However, the design was not changed as it was felt the inclusion of aerodynamic drag would reduce this velocity increment of the final stage.

The flight path to be followed and the method of staging were selected for the following reasons:

a. The take-off weight (approximately 200, 000 pounds) was a minimum.

b. The required aerodynamic loading for maneuvering was low.

c. The gust loading conditions were reduced compared to other types of flight paths.

d. Axial accelerations were not excessive (maximum being approximately 3.6g).

e. Because of the conservatism in estimating the velocity increment in the final stage, the initial glide conditions might be achieved when the effects of aerodynamic drag were considered.

In the course of this work, paths with other end conditions were also calculated. Figure 8 shows the estimated weight of a two-stage system for various glide ranges. This curve should be accurate within approximately 5%.

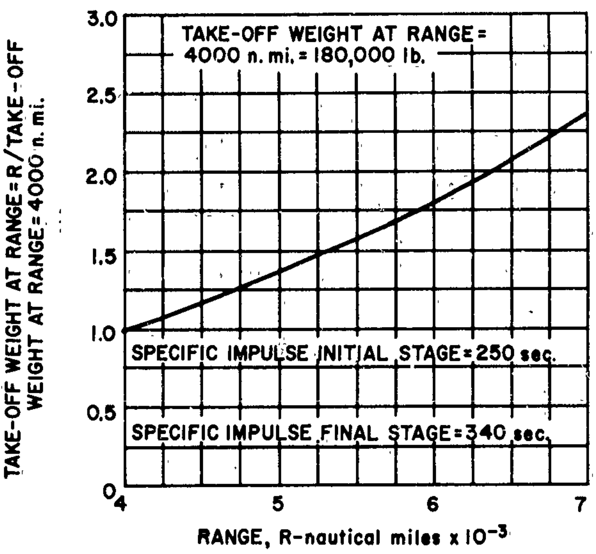


Figure 8. Estimated Take-Off Weight of a Two-Stage System for Various Glide Ranges

3. Aerodynamic Performance Parameters

The aerodynamic performance parameters have been calculated for the booster-final stage combination in the subsonic and supersonic speed regions, while the final stage parameters have been determined in the subsonic, supersonic, and hypersonic speed regions. The effects of wing-body interference were not included, since in the hypersonic speed region, where 90% or more of the range is obtained, the effects were found to be small.

In calculation of skin friction drag, a transition Reynolds No. of 5×10^6 was used in the present study on the basis of new data in the M=5 to M=10 region. These data indicate that this value may be conservatively low at higher Mach numbers. The effects of completely turbulent flow on the lower surfaces of the nose and body and the wing bottom have been included to illustrate one possible limit.

The L/D ratios for the glide portion of the final stage were obtained by dividing the summation of the lift coefficients by the summation of the power-off drag coefficients at various angles of attack, Mach numbers, and altitudes. The maximum L/D ratios at each altitude and Mach number are plotted in Figure 9 versus altitude for a constant Mach number. Also included in the figure are the lift coefficients and angles of attack at each maximum L/D. It can be seen that altitude has little effect on the maximum L/D, and that the level of the maximum L/D does not vary much with Mach number.

The maximum L/D ratios for the fully turbulent skin friction condition on the lower surfaces at each altitude and Mach number are plotted in Figure 10. The angles of attack and lift coefficients at the maximum L/D ratios are also included in this figure.

The effects of shock wave-boundary layer interaction on the pressure coefficient and skin friction drag coefficient have been investigated for the present wing at a Mach number of 16, at several altitudes, and at several angles of attack. A comparison of the wing lift and skin friction coefficients with and without interaction shows that the lift of the wing at glide angle of attack (approximately 7.0°) is increased 5% and the skin friction drag coefficient is increased 25%, but the effect on airplane maximum L/D is to lower it by only 1%. A summary of the L/D ratios and the maximum L/D ratios and associated parameters at various altitudes are shown in Figure 9. It appears from this figure that shock wave-boundary layer interaction in over-all effect is not important at this speed.

The effects of slip flow in the boundary layer have not been taken into account at this time in the aerodynamic parameters, although the slip regime apparently is entered. The effects are expected to be small, but should reduce the skin friction coefficients, possibly adding some conservatism to the analysis.

Real gas variations from the ideal gas theory are small in the Mach number and angle of attack range encountered by most of the surfaces of this aircraft, and since the real gas calculations would entail considerable difficulty

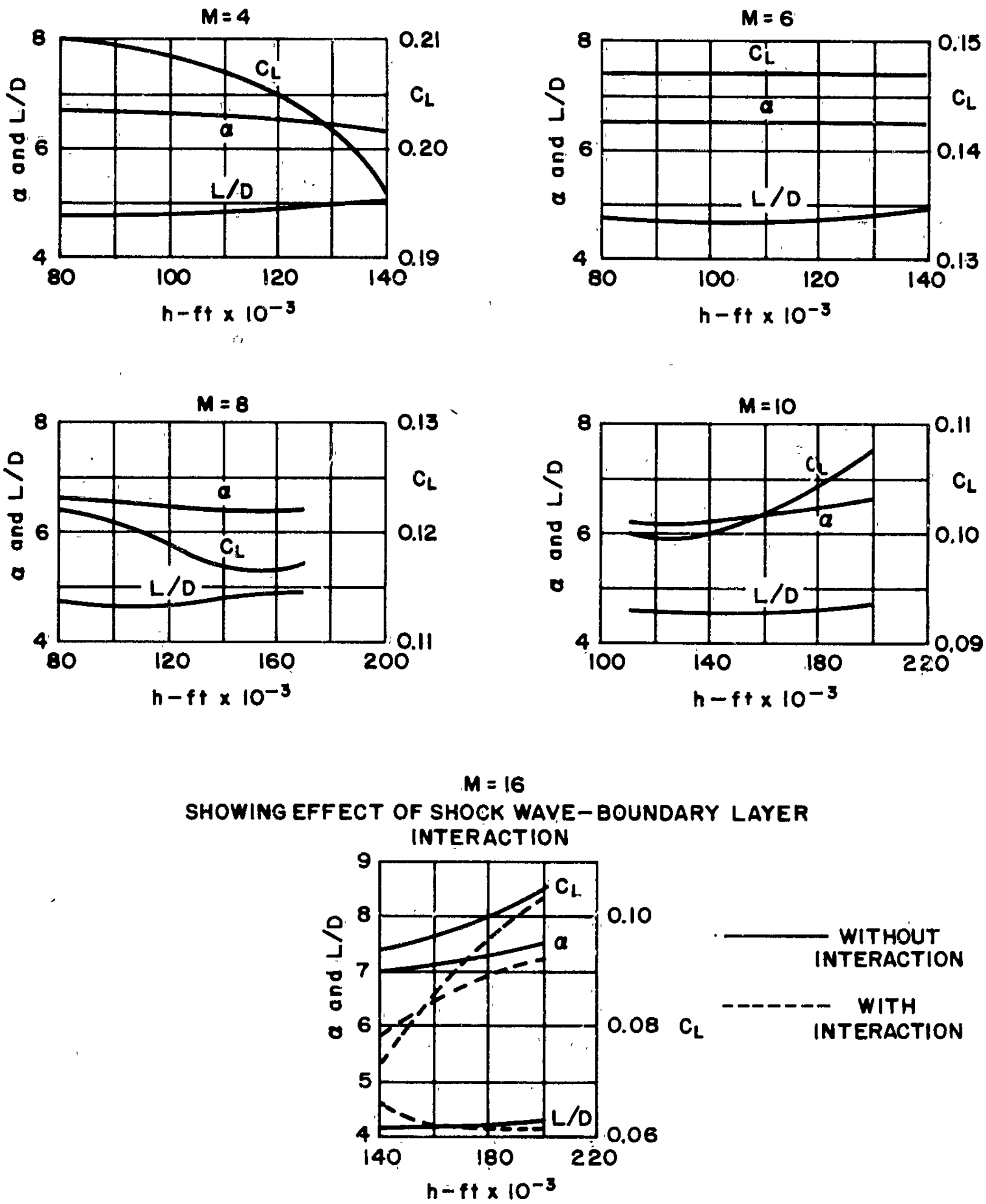


Figure 9. Summary of Maximum Lift/Drag Ratios with Corresponding Lift Coefficients and Angles of Attack at Constant Mach Numbers for Final Stage

TURBULENT FLOW – LOWER SURFACES

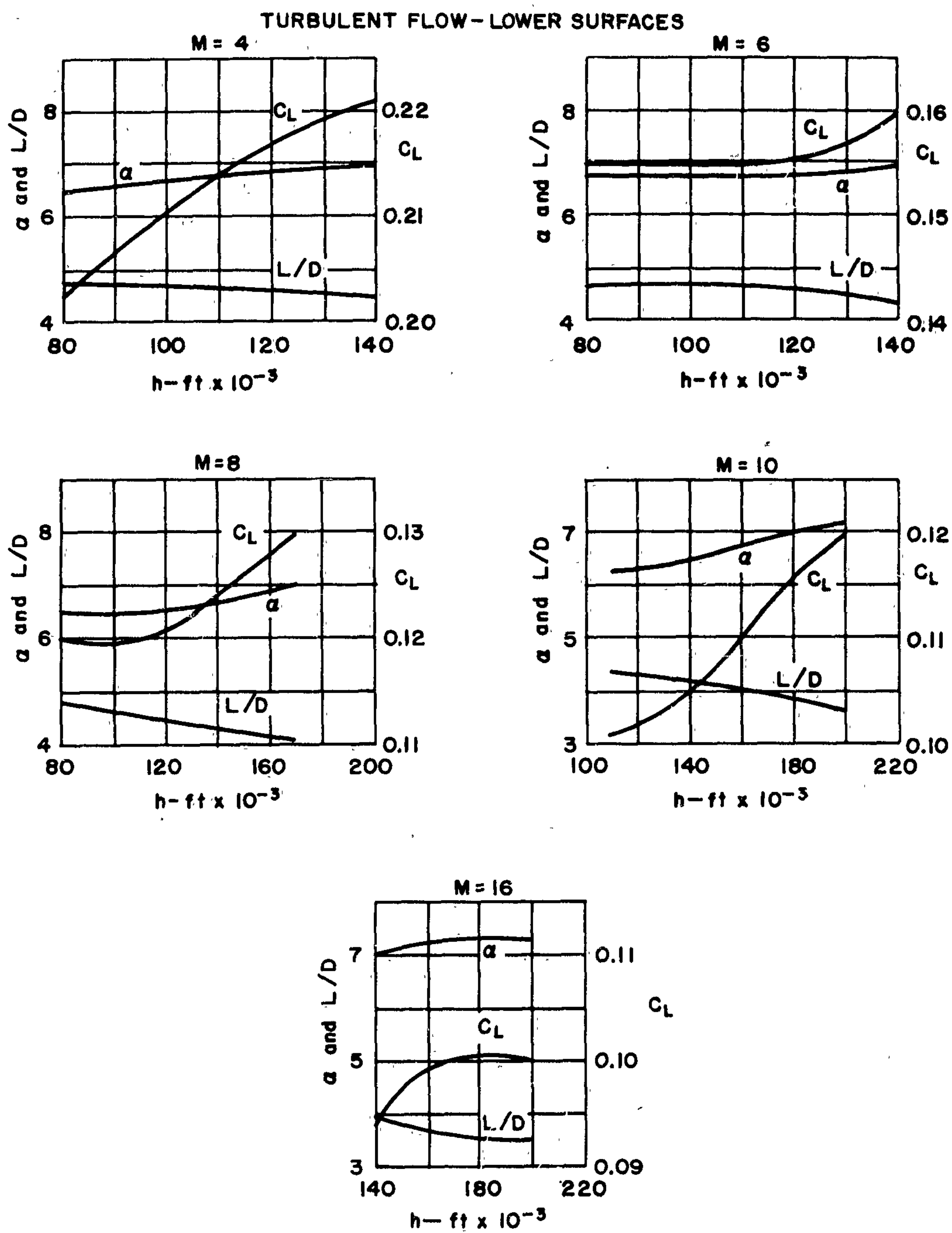


Figure 10. Summary of Maximum Lift/Drag Ratios with Corresponding Lift Coefficients and Angles of Attack at Constant Mach Numbers for Final Stage

and labor, adding little to the accuracy of the study, the real gas values were not employed in this analysis.

4. Ascent Performance

Figure 11 shows a comparison of the ascent path calculated for vacuum conditions as compared with a path calculated on the Bell Aircraft Performance Analyzer (BAPA) for existing atmospheric conditions. The Analyzer is an electromechanical analogue computer designed and built specifically to solve the differential equations of motion of a missile or airplane under trim flight.

In programming the ascent on BAPA, the vacuum ascent was used as a guide. The computer solution verified the original performance and provided a flight path angle (and hence altitude) versus velocity distribution. When all variables were included, a performance was finally obtained which achieved conditions of velocity and altitude from which the glide performance could be initiated. In this ascent, a velocity at the end of the powered phase of 16,600 feet per second at an altitude of 165,000 feet was attained. The flight path angle was 0° and the angle of attack was 3.2°. A ground range of 388 miles was covered in a total time of 328.3 seconds. As shown in Figure 11, although small performance differences exist, the overall agreement with the vacuum path is remarkable. The vacuum method can apparently be a highly advantageous means for obtaining preliminary performance estimates.

The study as conducted in no way establishes an optimum ascent. A more thorough investigation of ascent programming, including thrust variations, would be necessary to ascertain the maximum possible performance of the system.

5. Glide Performance (Nonrotating Earth)

In equilibrium glide flight, the maximum range is obtained when the aircraft is flown at that angle of attack for which the maximum lift/drag ratio at each combination of velocity and altitude is obtained. In other words, at any

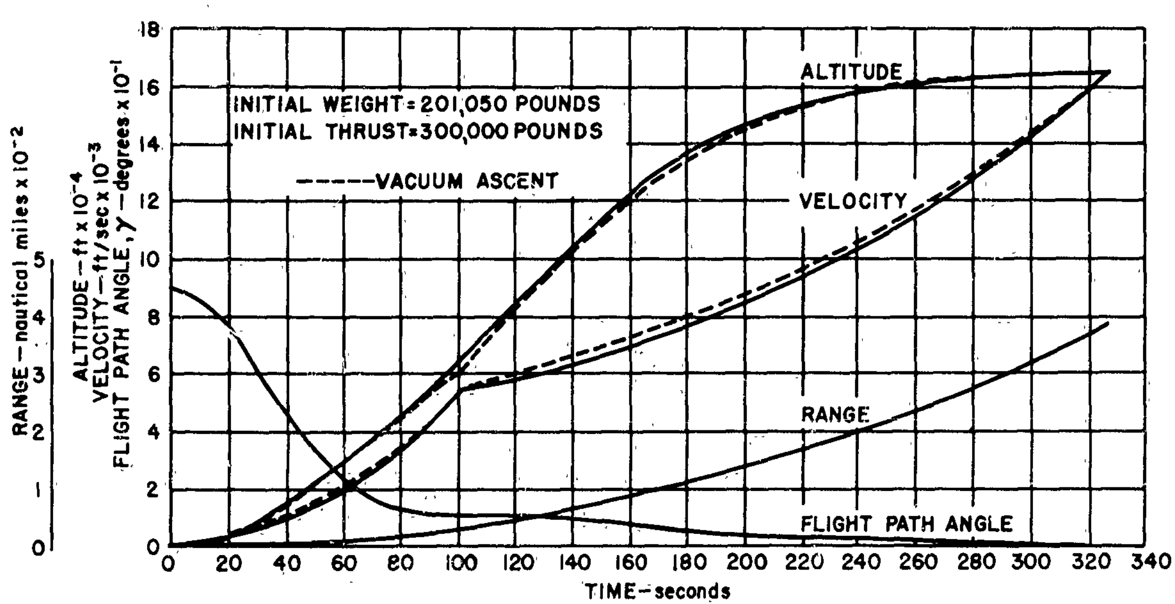


Figure 11. Ascent Time History (Nonrotating Earth)

given velocity of an unpowered aircraft, there is, at each altitude, a corresponding angle of attack at which the maximum L/D will be realized. There is, however, only one altitude and corresponding angle of attack for the given velocity at which the aerodynamic lift plus the centrifugal force due to the aircraft's circular motion about the earth will just equal the weight of the aircraft. The maximum L/D developed under these conditions will be that which must be obtained to produce the maximum rate of change of the range. If at each velocity these flight conditions are realized, the total range will be a maximum.

The equilibrium altitudes and corresponding maximum lift/drag ratios were determined by a method of graphical solution. The maximum range was computed by integrating these maximum L/D ratios with respect to the total energy available from the altitude and velocity, taking into account the effects of the centrifugal force and gravity variation with altitude on the weight. This calculation yielded a maximum glide range, to zero altitude and velocity, of 4,290 nautical miles for a weight of 18,870 pounds. Initial velocity and altitude conditions were 16,600 feet per second and 165,000 feet (initial Mach number of 15.25). The time history of the glide phase is presented in Figure 12. Equilibrium wall temperatures at the one-foot station on the upper and lower wing surface and the lift coefficient, maximum L/D, and angle of attack are also shown.

The maximum range as determined above was for a constant weight. Approximately 1,700 pounds of weight, however, can be discarded at some time during the glide (unuseable propellants, coolants, pressurizing gases, etc.). The manner in which these masses are jettisoned can improve the over-all performance. This evaluation also assumes a nonrotating earth. The effect of earth rotation on the flight will be advantageous if the flight is in an easterly direction as discussed in Reference 4.

As shown in Figure 9, the investigation of the effect of shock wave-boundary layer interaction at a Mach number of 16 indicated that no appreciable change would occur in either the maximum L/D ratio or the equilibrium altitude. For both quantities, the decrease with interaction was less than 1%. It was, therefore, not considered necessary to re-evaluate the maximum glide range for these lower values.

The total performance was determined by combining the boost and glide phases of the flight. The total range obtained from launch to landing was 4,678 nautical miles during a flight time of 60.2 minutes. The mission time history is presented in Figure 13.

An evaluation was made of the maximum glide range attainable assuming turbulent flow over the body bottom and the lower wing surface. These surfaces contributed about 75% of the total skin friction drag when transistion was assumed to occur at a Reynolds number of 5×10^6. These maximum L/D ratios were integrated as before with respect to the total available energy. For an initial velocity of 16,600 feet per second at the equilibrium altitude of 169,000 feet and the previous weight of 18,870 pounds, the range calculated to zero velocity and altitude was 3,925 nautical miles. This was 365 miles less than the range obtained with previous assumptions regarding laminar flow.

This examination of the effect of turbulent flow on the glide performance was made to give some idea of the accuracy of the range obtained under the assumption that transition occurs at a Reynold's number of 5×10^6. That assumption was probably the most significant and at the same time questionable one made for this entire study. It was not believed to be optimistic, but no published data were available to positively substantiate it. It was encouraging, therefore, to find that with essentially all turbulent flow the reduction in glide range was not too severe. The total mission range, if such flow were encountered, would still be approximately 4,310 nautical miles with 4,000 miles covered at altiudes above 100,000 feet.

The time history for this glide is presented in Figure 14.

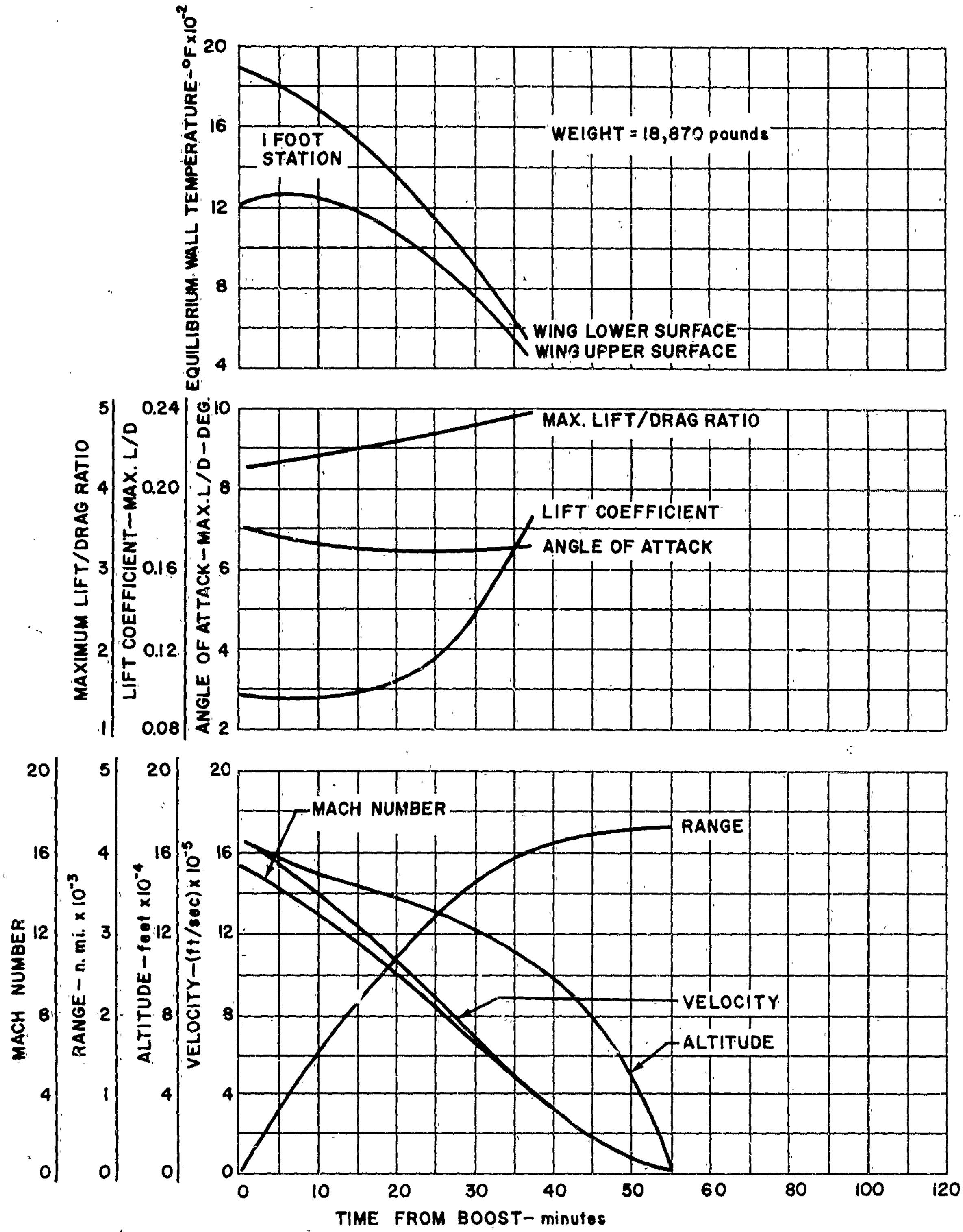


Figure 12. Glide Performance Time History (Nonrotating Earth)

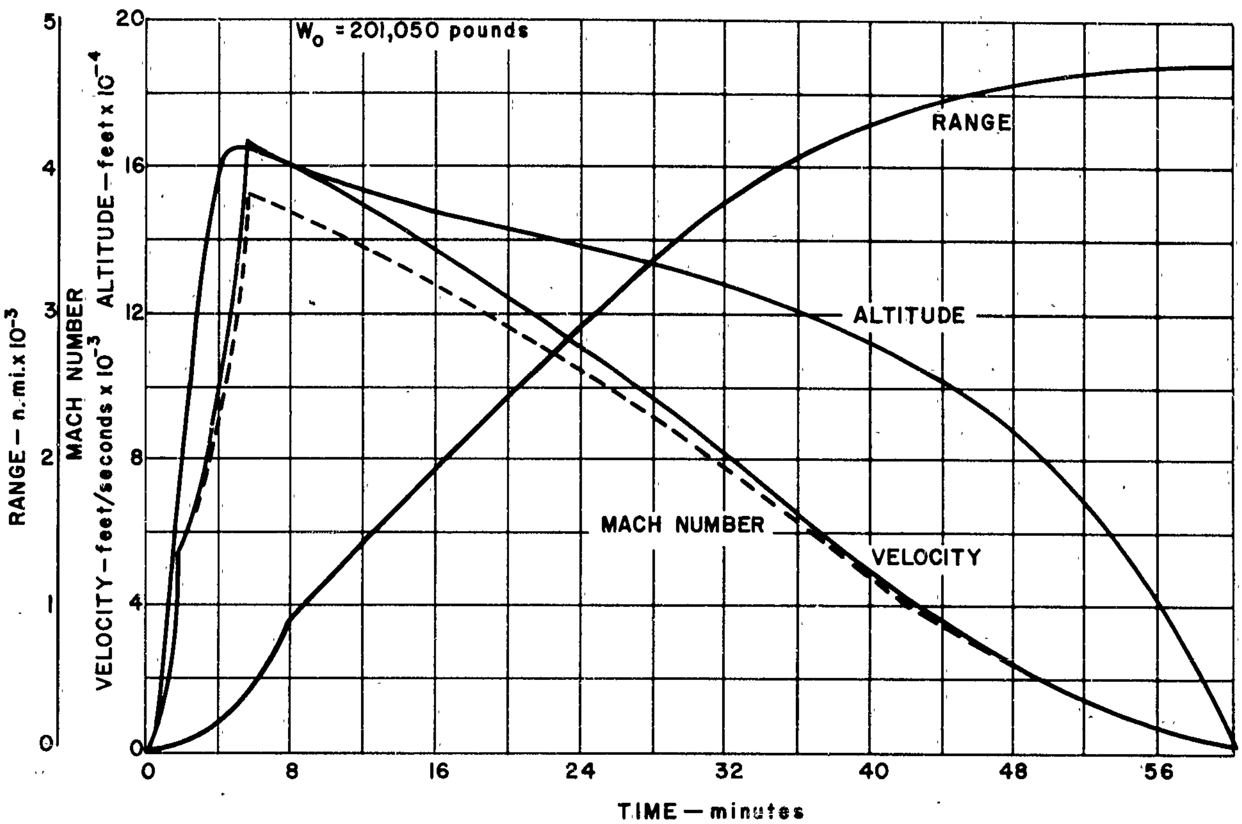


Figure 13. Mission Time History (Nonrotating Earth)

6. Turn Study (Nonrotating Earth)

In this study, coordinated turns during the glide flight of the MX-2276 were examined. The assumptions of the study were that turns would be coordinated and that flight would always be at the angle of attack for maximum L/D. In other words the aircraft would be rolled to achieve the turn but would maintain the angle of attack for the maximum L/D which could be realized at each combination of altitude and velocity during the glide.

The horizontal paths are plotted in Figure 15 for initial glide velocities of 16,600 feet per second. Points of constant velocity have been joined to show the positions during the descent. If the differences in altitude which exist along the constant velocity curves are disregarded, segments of the various constant acceleration curves may be combined to produce flight paths with any program of accelerations desired.

For negative lateral accelerations the curves should be extended symmetrically to the left of the down-range axis. The assumption that altitude variations with velocity and lateral accelerations can be disregarded results in some error, especially at high velocities. The magnitude of these errors is shown in Reference 3.

An examination of the temperature changes which would occur under the various turns was also made, and the results are presented in

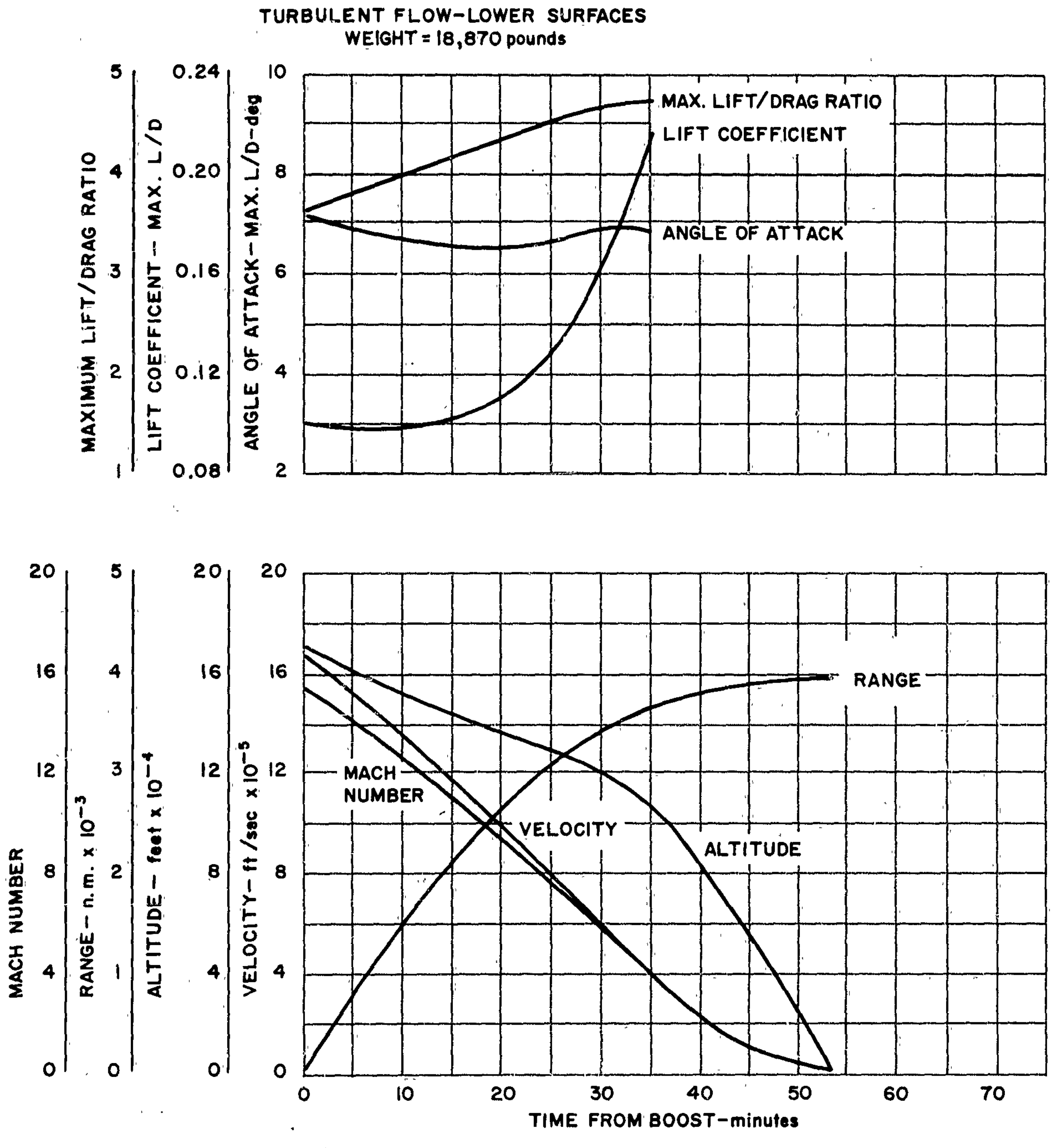


Figure 14. Glide Performance Time History (Nonrotating Earth)

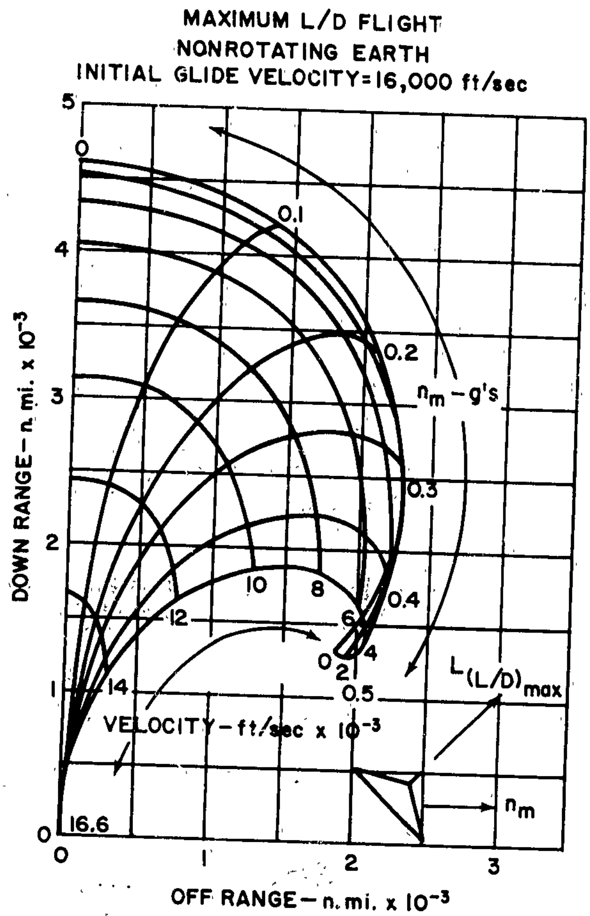


Figure 15. Turning Data

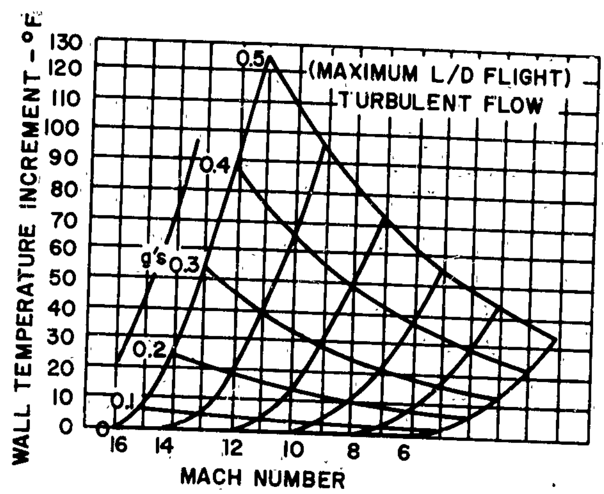


Figure 16. Equilibrium Wall Temperature Increment with Turning Rates

Figure 16. As expected, the increases are most significant at combinations of high speed and high acceleration.

7. Landing

With the use of delta or clipped delta wings to relieve some of the compressibility and heat problems at the higher end of the speed spectrum, other problems have come into being at the low speeds – such as unstable rolling oscillations, shifts in the static longitudinal stability, high effective dihedral, low damping in roll, static longitudinal instability at stall, and adverse yaw effects, to name the more significant troubles.

Measurements of approach and landing data have been made of seven high-speed research airplanes in Reference 5. Several landings were made in each airplane by pilots experienced in this type aircraft. The pilots knew that data were being taken, but no instructions or restrictions were given them concerning the landing maneuver. There was a wide variation between landings, but these were all described as being normal landings for the particular aircraft. From the data taken in these tests, it was apparent that each airplane had some undesirable characteristic at or near stall, and thus forced the pilots to land at speeds from 10% to 40% fasten than at $C_{L_{max}}$. In the case of the X-3 airplane, this characteristic meant that landing speeds of 210 to 220 knots and approach speeds of 285 to 296 knots were necessary, making it an extremely fast landing airplane in comparison to the more desirable landing speeds of the X-5 of 100 to 110 knots.

These aircraft had vertical velocities at contact on the average of two feet per second (the highest recorded was 4.6 feet per second for XF-92 airplane), although during approach the vertical velocities were 30 to 90 feet per second. Ground effect was generally a significant aid in the flare maneuver. It was noted that there was no significant difference between the vertical contact velocities of the XF-92A landing dead stock or idle power, or between the D-558-II glider and jet-powered airplanes.

The ground effect just mentioned was very pronounced for the low aspect ratios and short landing gear lengths. The pilots reported that the effect was noticeable on all the aircraft, but was very evident on the X-4 and the XF-92A, causing the landing of these two aircraft to be made very easily by maintaining a constant glide angle and utilizing the ground effect to reduce the vertical velocity to a low value near the ground.

The pilot's judgment of his altitude in the approach and flare of a landing maneuver, and the reaction time delay of pilot and airplane in response to a correction are the two most important factors in causing errors and variations in landing. In Reference 6, these factors have been dealt with and corrections to the idealized landing flare calculations were derived in order to define a safer and more realistic landing flare. The qualitative conclusion found in this reference is that, as the stalling speed of an airplane is increased, the effect of the height estimation factor becomes relatively more important than the time delay factor. Another conclusion reached as a result of several calculations in this reference was that landing maneuvers beyond the capabilities of the pilot would not be required even if the lift-to-drag ratio at the base of the flare was as low as 1.25 with accompanying high stalling and sinking speeds.

A landing analysis was made for the present design by iterating the equations of motion with respect to time and including pilot judgment and reaction considerations and the ground effect on the lift of the airplane near the ground.

The approach to the landing field was considered to be at maximum L/D, which corresponds to a gliding velocity of 183 knots, sinking speed of 57 feet per second, and a gliding angle of 6.6 degrees. The landing attitude of the aircraft was limited herein to 10 degrees, the ground angle dictated by the tail cone-landing gear geometry. The change from gliding attitude to landing attitude was made at the top of the flare, and the flare continued to ground contact, pulling enough additional load in the flare to touch down at a flight path angle of zero degrees and a load factor of one. It was found that the flare for the final stage should start at an altitude of 310 feet and a little over a mile away from the runway. These values include the ground effect on lift and the pilot considerations.

In Figure 17, the MX-2276 landing flare is compared with the landing flares, approach speeds, and landing speeds of the airplanes in Reference 5 which were discussed previously in this section. It is worth noting that the landing analysis of this aircraft falls well within the landing conditions of these present-day research airplanes and should not present any additional problems that are not now being encountered and successfully handled. The attitudes in the total maneuver are well removed from the maximum lift coefficient so that large changes in longitudinal trim and roll instability are not likely to be present. For comparison with the previous discussions, the landing maneuver lift coefficients were always less than 0.50. During the largest portion of the over-all landing maneuver, the approach glide, the C_L would be 0.36. Finally, the control system, at least as planned for the boost and hypersonic glide positions of flight, is to be automatic, the pilot only monitoring it. It is reasonable, then, to expect this system can at least be available to the pilot for automatic damping during the landing maneuver, which should further reduce the control problem there.

8. Stability and Control

In the selection of the static stability criteria for which the final stage configuration was designed, the question of available control had to be considered. As the highly swept, low aspect ratio wing initially limited the size of the control surfaces for structural reasons to approximately 18% of the exposed wing area, it was necessary to design the aircraft for low static margins over its entire speed range. The aircraft was allowed to be statically unstable in pitch at lift coefficients below those required

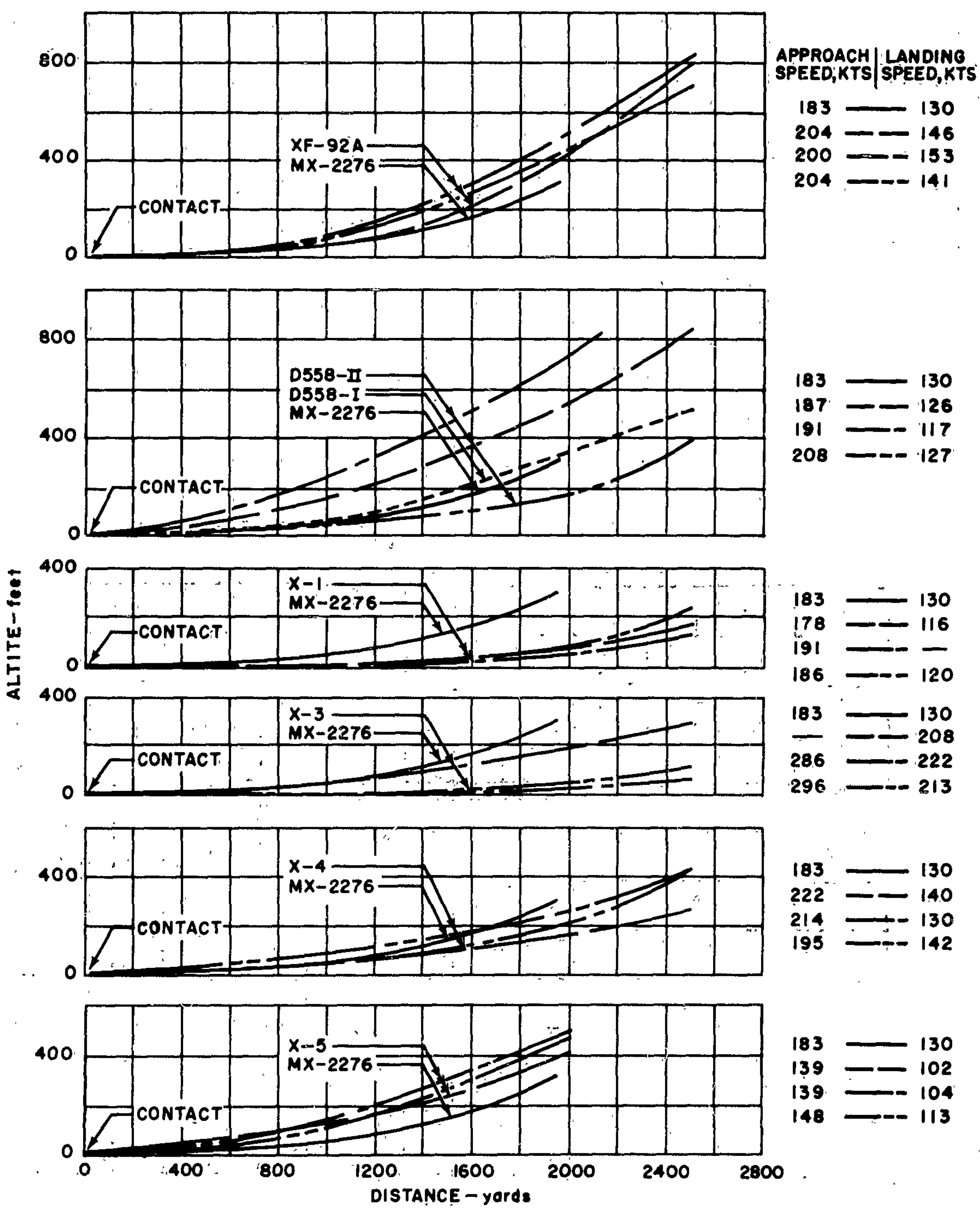


Figure 17. Comparison of Landing Flares

for flight at maximum lift-drag ratio. In this manner it also was possible to adjust the center of pressure travel such that down elevator deflections were required for trim at Mach numbers above M = 10.

The final stage was directionally stabilized at low speed by a conventional vertical fin on the top of the fuselage. At high Mach numbers, the addition of a lower vertical fin was necessary for directionally stabilizing the aircraft because the upper vertical fin lost effectiveness as the angle of attack was increased.

Because the combined configuration of booster and final stage are to be automatically controlled and stabilized, no static stability investigation was made at this time. However, the favorable weight distribution between the booster and final stage (approximately 2 to 1) placed the combined cg well forward relative to the rocket motors. This forward cg location of the combined two-stage configuration was beneficial in that it reduced the necessary thrust of the rocket motors for stabilization and control from that which previously had been experienced in the design of three-stage systems.

a. Static Longitudinal Stability of the Final Stage

In Figure 18 are shown the estimated aerodynamic center and cg locations as fractions of the m.a.c. (As the aerodynamic center location varied little from the values shown for a $C_L = 0.12$ for the higher values of C_L, its location was not shown in the interest of clarity of the figure.) The estimated location of the center of gravity of the configuration from burnout to landing is also shown on this plot. (It should be noted that before all the propellants were expended, the cg of the final stage was always ahead of the locations shown.) The lift coefficients for glide flight at maximum L/D were

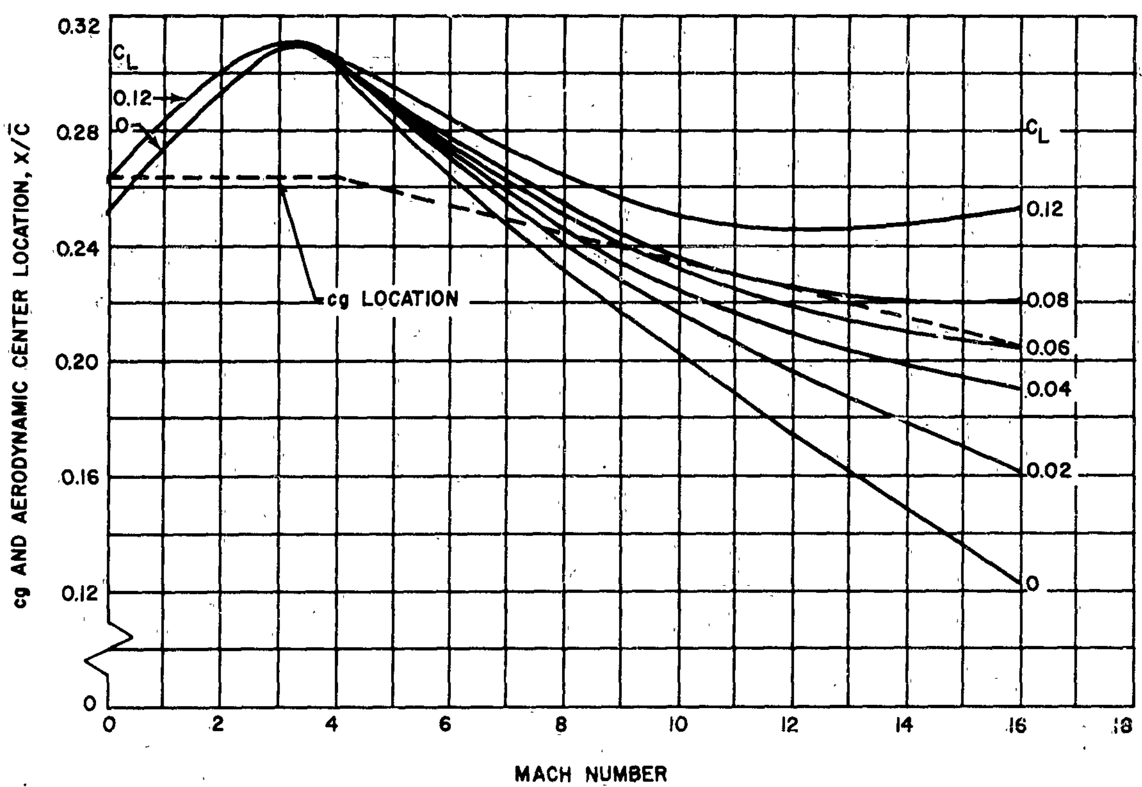


Figure 18. Estimated Center of Gravity and Aerodynamic Center Locations

such that a positive static margin existed at all flight conditions. During the supersonic portion of the glide flight, the aerodynamic center at flight lift coefficients was located approximately one to five percent of the m.a.c. behind the center of gravity. Although the static margins were acceptable during the supersonic glide, the rearward movement of the center of gravity was such that the static margin at low speeds was marginal. However, it was thought that further refinement of weight estimation and equipment placing could result in limiting this rearward cg shift to a more suitable amount.

b. Static Directional Stability of the Final Stage

The selections of the planform and placement of the upper vertical tail were made primarily on the basis of low speed considerations. There was no contribution to the low-speed directional stability from the lower vertical tail because it had to be dropped or folded to facilitate landing. The calculated low-speed yawing moment coefficients about an estimated cg location at station 681 are shown in Figure 19 for the anticipated range at angles of attack at low speeds.

The supersonic directional stability was checked at only M = 16.0, since previous experience had indicated the highest flight Mach number to be the most critical. The shadowing effect of the body upon the upper vertical surfaces at hypersonic glide angles of attack and small angles of yaw resulted in a reduced effectiveness of this surface. Therefore, a vertical surface was added on the lower side of the fuselage.

The estimated yawing moments at M = 16.0 for the final stage about a cg located at station 665 are also shown in Figure 19. As the angles of attack were not expected to go below 4° at any time in the glide flight, this proposed design should possess static directional stability over the entire flight range.

c. Control

An analysis of the elevator trim deflections indicated that elevator deflections of

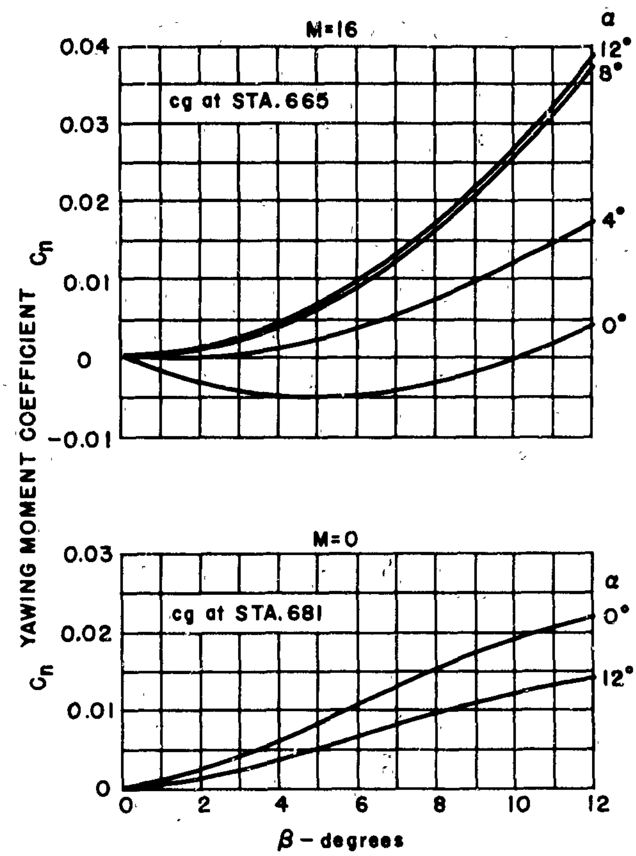


Figure 19. Estimated Yawing Moment Characteristics

10° trailing edge down and 15° trailing edge up would be sufficient for longitudinal control. The elevators are full blunt in order to retain their effectiveness in the transonic and hypersonic ranges. The elevator trim deflections were found to be so small that the effect on range should be very small also.

Directional control at low speeds is provided by a rudder on the upper vertical surface. The rudder becomes ineffective at hypersonic speeds owing to body shielding; however, turning at these speeds can be accomplished by banking the airplane. This arrangement also has the least effect on range.

Roll control is provided by deflecting the elevators independently in the manner of

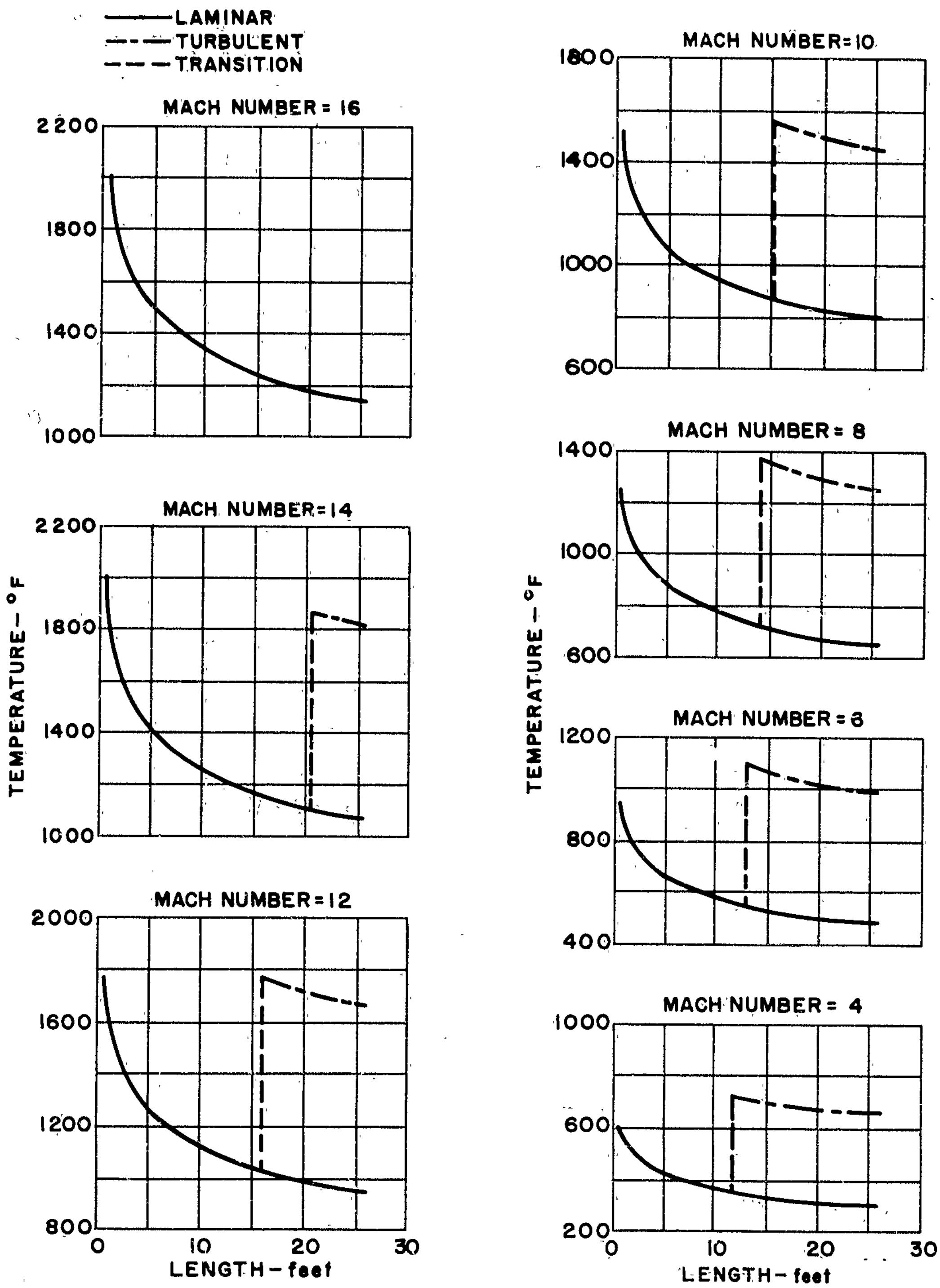


Figure 20. Equilibrium Temperature: Bottom of Wing

ailerons. The slenderness of the configuration is indicative of a low damping in-roll parameter. Therefore, this method of control should be adequate.

9. Aerodynamic Heating

The aerodynamic heating problems discussed in Reference 3 have been extended during this study period. A more extensive consideration of the equilibrium wall temperature was undertaken and the heat fluxes to the stagnation regions, i.e., nose of the vehicle and leading edges of the wings, have been investigated in much more detail.

a. Viscous Heating

Figure 20 shows the equilibrium temperature for the bottom of the wing. This region is more critical than the upper surface, because of the higher pressures prevailing. In Figure 20, the temperature distributions have been carried forward either to the one-half foot station or to 2000°F, whichever is lower. Wherever the equilibrium temperature was above 2000°F, the wall temperature was assumed to be 2000°F and the amount of coolant required was obtained, and will be discussed later. The laminar boundary layer equilibrium temperatures were calculated back to the trailing edge, for if the transition Reynolds numbers were as large as 10 or 20 million, the flow over the bottom would be laminar for this entire glide. The turbulent flow temperatures have been computed and also plotted on Figure 20, using the effective turbulent length at transition as was calculated for the skin friction. Because of the large increase in equilibrium temperature (about 750°F at high Mach numbers) it is evident that laminar flow must be maintained as long as possible and that its prediction must be made accurately if design economy is to be achieved. It also can be noted that the decrease of temperature with chord distance is much less rapid with turbulent than with laminar flow. Time histories of the equilibrium temperature in the glide have been presented in Figure 21 for the laminar flow condition at the one-foot and ten-foot stations of both the bottom and top of the wing. Also, the turbulent flow time history of the temperature on the bottom at the 25-foot chord station is presented in this figure.

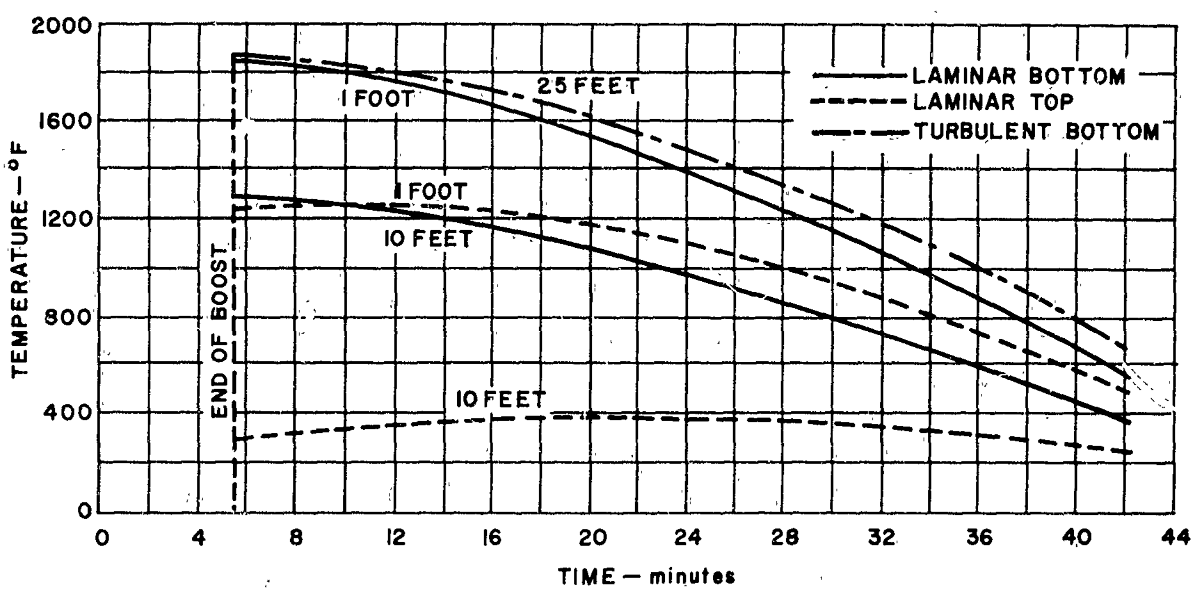


Figure 21. Equilibrium Wing Temperature Time History

The temperature profiles for the bottom of the body are presented in Figure 22. The viscous heating rates were calculated from the intersection of the hemisphere cone-frustrum aft to the cone-cylinder shoulder using laminar flow relations. The cooling required forward of the nose station, where the equilibrium temperatures are 2000°F, was also obtained. It also will be discussed later. The equilibrium temperatures for laminar flow are calculated back to the cone-cylinder shoulder because, if the transition Reynolds number were as large as 12 x 10^6, this region would be laminar for all Mach numbers above 4. The turbulent flow temperatures are also presented on Figure 22. Again the importance of delaying transition and its prediction are exemplified. It should be noted that there is a region on this bottom meridian which exceeds 2000°F for about 15 minutes if transition occurs at a Reynolds number of 5 x 10^6. This region would require an outside skin that could withstand this higher temperature, or the skin would require cooling. Alternately, the condition could be alleviated by modifying the configuration to reduce the angle of attack of the bottom meridian by drooping the nose or by going to an ogival shape.

Some comments as to the possibility of obtaining a transition Reynolds number sufficiently high to relieve this high turbulent boundary layer heating condition also are in order. A transition Reynolds number of 8.5 x 10^6 on the lower nose cone element at glide conditions greater than Mach number 10, instead of the average transition Reynolds number of 5 x 10^6 generally used throughout the present study, would do this; and, actually, the probability appears good. Present day research is showing

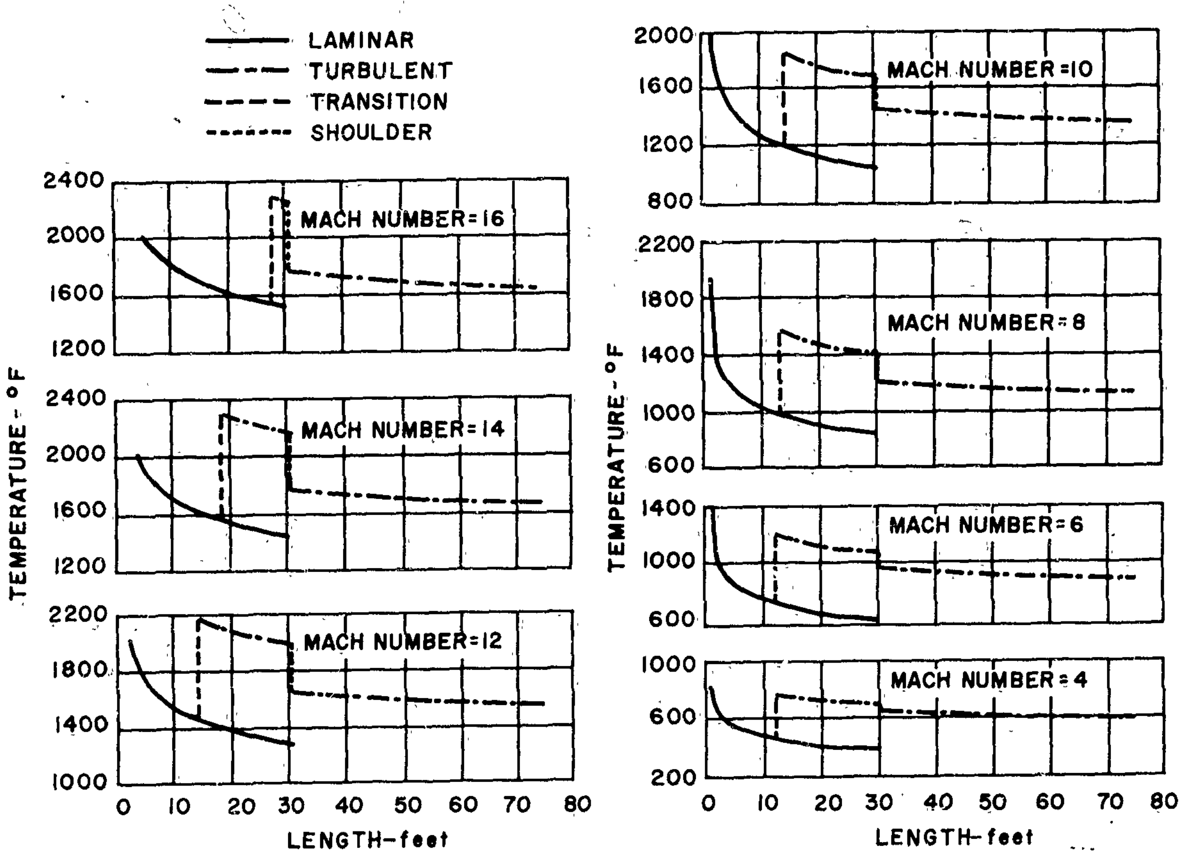


Figure 22. Equilibrium Temperature: Bottom of Body

that higher transition Reynolds numbers than this can be achieved at much lower Mach numbers on slender cone and body shapes, particularly if outer surface cooling is present. Cooling by radiation and/or that cooling needed to maintain structural integrity of the outer surface will be high; the recovery or adiabatic wall temperature-to-local stream temperature ratios at Mach numbers greater than 10 would be on the order of 10:1 or greater, but with radiation cooling the wall-to-local stream temperature ratios are about 3:1. This high degree of cooling should tend to stabilize the boundary layer. Further, it is expected that the noted trend of increasing transition Reynolds number with increasing Mach number will continue to these higher Mach numbers. The nose cone of the present configuration is ahead of the sharply swept wings so that interference or shock induced transition would not seem to be a factor.

Temperature distributions on other portions of the wing and body are included in Reference 4. However, they are all lower than the bottom surfaces; therefore, they have not been included here.

A study was made of the effect of shock-boundary layer interaction upon the bottom and top of the wing. The effect on the bottom surface is negligible, while on the top surface the effect is noticeable above Mach 6. The temperatures on the top are still well below those on the bottom, however. A similar interaction could be expected on the top of the body, but because of three-dimensional effects, it would be expected to be less severe.

b. Leading Edge Heating

The leading edge of the wing has been considered at various angles of sweep and for different diameters. Perpendicular to its planform line, the leading edge cross-section is assumed to be a constant diameter circular segment having shoulders tangent to the upper and lower wing surfaces; hence, in the stream direction, the leading edge cross section is a semiellipse whose semimajor axis is the radius of the normal-plane circular segment divided by the cosine of the sweep angle. This leading edge is followed by a 5.74° included angle, modified wedge wing. The major portion of the analysis was limited to a 75° sweep angle and to a one-half inch leading edge diameter.

The total heat flux to the region of the leading edge has been determined for the one-half inch diameter, 75° sweep by assuming that the average heat flux over the cylinder is 70% of that at the stagnation point and that the heating rate at the shoulder is 25% of that at the stagnation point. These total heat fluxes are presented in Figure 23 as net fluxes, i.e., the heat flux by convection minus the heat flux by radiation using an emissivity of 0.9. The heat flux to the wing forward of the chord station where the equilibrium temperature is 2000°F is also shown in Figure 23. The total heat flux per foot of span has been calculated to be 41,000 BTU. Thus, the total cooling requirement for the leading edges of the wings and tail was found to be 1,122,000 BTU, which could be realized by vaporizing about 1,122 pounds of water or 112 pounds of lithium. If the cooling process included transpiration of the coolant vapors into the local boundary layer, these figures could possibly be reduced in the order of 50%.

The nose of the body is a hemisphere of 0.65-foot radius followed by a 5° semivertex angle cone. The net amount of cooling which must be provided on the nose for the entire flight plan is shown in Figure 24. It has been assumed that the heat flux in the region of the nose is as large on the top and sides of the cone as it is on the bottom. This, of course, is a rather conservative estimate, but there are other factors which must be considered such as shock-boundary layer interaction and three-dimensional flow, real gas, radiation, and other effects which are not well understood, so that further refinements were not considered necessary at this time. The total net heat flux which must be handled during the flight is 371,000 BTU. The net heat flux to the hemisphere itself is 230,000 BTU with 141,000 BTU to the frustrum of the cone. This cooling requirement could be met with about 371 pounds of water or 37 pounds of a liquid metal such as lithium to cool the outside skin of the nose. If the cooling were done by transpiration, about half this much coolant would

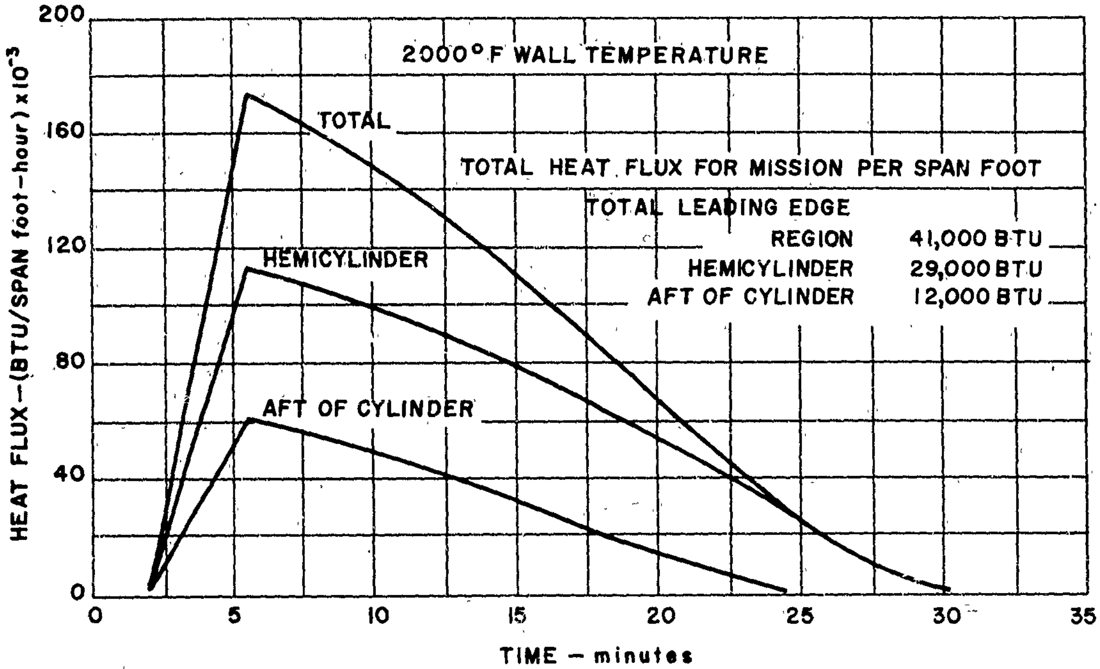


Figure 23. Total Leading Edge Heat Flux

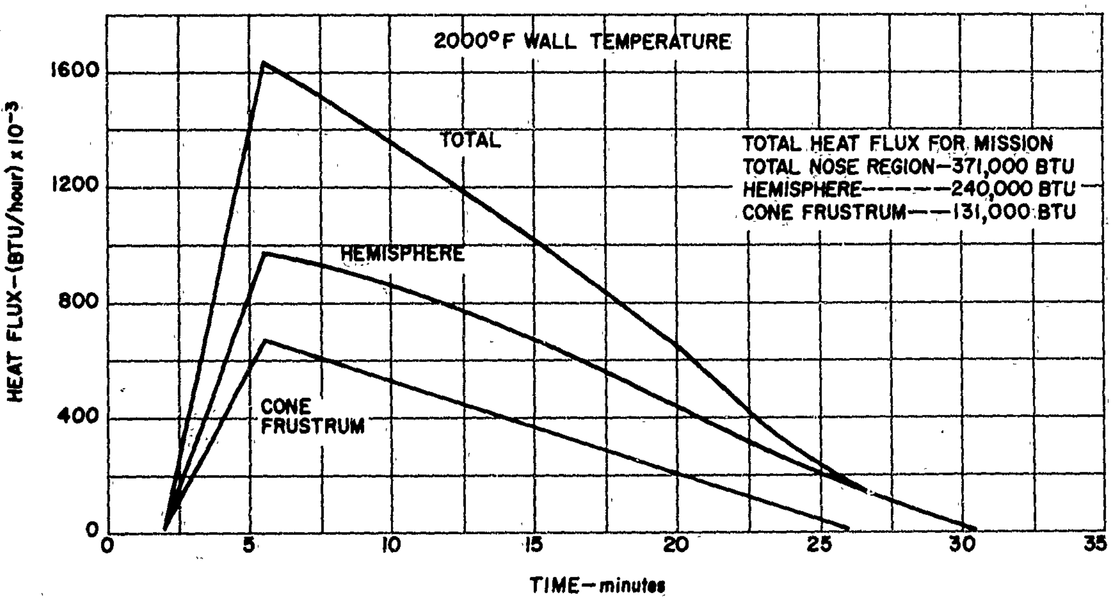


Figure 24. Total Nose Heat Flux

be required. The total heat flux in the region of the nose could also be reduced considerably by reducing the diameter of the nose sphere.

Generally, less effort was expended on the aerodynamic heating problem during the ascent, one reason being that the ascent is so short in time that the heating problem is not severe. The outside skin equilibrium temperatures of the glide aircraft will be less during the ascent than they are during the initial part of the glide. The ascent heating problems are most important to the design of the booster which performs its complete function during the ascent period.

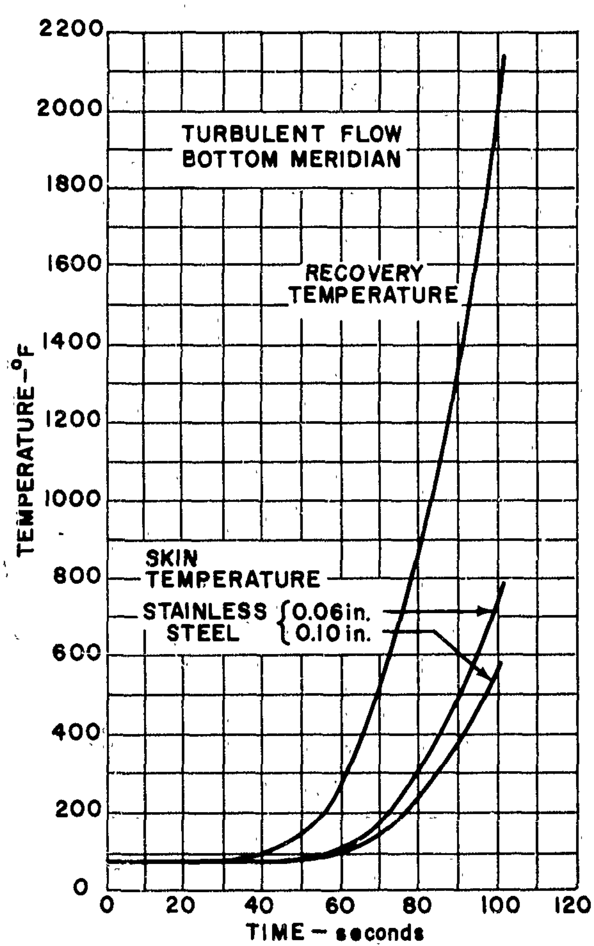


Figure 25. Transient Heating on Cone Frustum of Booster

The temperatures reached by a structural external skin of the expendable booster were of interest with respect to the feasibility of an uninsulated construction. A transient calculation was made for two thicknesses of stainless steel skin, 0.06 and 0.10 inch on the cone frustum of the booster. A time history of the skin temperature and recovery temperature is presented in Figure 25. It is of interest to note that virtually no temperature rise is experienced in the first minute, but after that the temperature rises very sharply by as much as 20°F per second. Although the temperature at end of boost is only about 800°F and 600°F, respectively, for the two thicknesses, the equilibrium temperature at this point is about 1,400°F, which indicates the significance of the thermal lag of the skin.

10. Photographic Environment

The photographic and visual effectiveness of a system operating at hypersonic speeds at altitudes over 100,000 feet will be governed by two principal factors: optical refraction caused by air density nonhomogeneity and radiation in the visual and/or photographic spectrum caused by boundary layer phenomena. The heating of camera components by high skin and boundary layer temperatures must also be considered.

Atmospheric density at the operational altitudes will cause less disturbance than has been experimentally detected at lower altitudes. At Mach No. 16 refraction due to boundary layer and turbulence density will cause approximately 0.5 second of arc deviation. A slowly varying or constant shock wedge deviation of approximately 12 seconds of arc will occur which can be corrected by adjusting sighting angles of individual cameras and locations. Figure 26 shows the shock patterns about the camera locations at M = 15.

The extent of luminous radiation in the boundary layer is dependent on more detailed information currently unavailable regarding the characteristics of air in the 3000° to 6000°K temperature region. It is known that radiation associated with recombination of dissociated molecules accounts for most of the energy transfer at high temperatures. This radiation

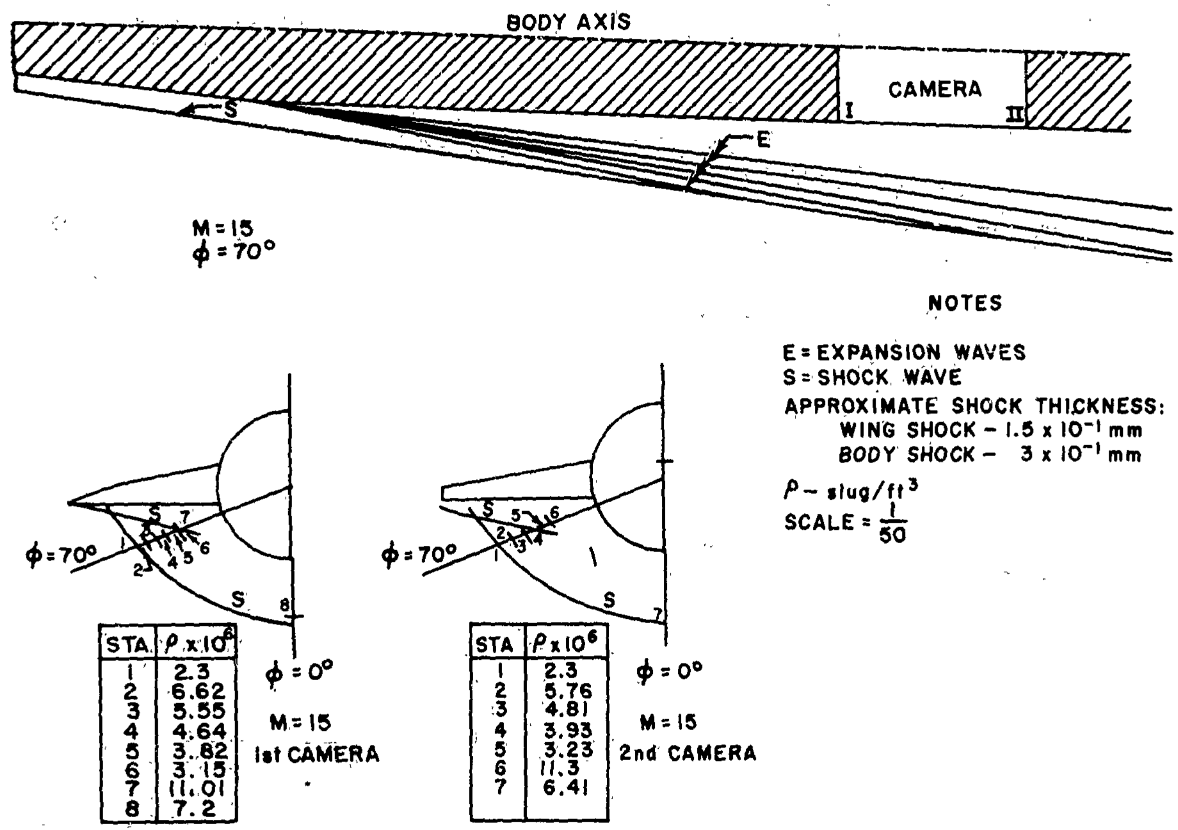


Figure 26. Shock Pattern in Neighborhood of Camera

is largely in the ultraviolet region. The electronic transitions accompanying the formation and breakup of the minor constituents of air at high temperatures produce visible and near infrared radiation, however. Measures can be taken for cooling the boundary layer ahead of the camera or visual ports once the magnitude of the radiation effect can be determined. Visible radiation in the vicinity of the bow shock will not disturb cameras or viewers located further back in the fuselage.

The uncooled skin in the vicinity of the cameras' location is expected to reach 1500°F temperature. Transpiration or film cooling before the camera location as well as interior temperature control will maintain acceptable temperatures for equipment.

11. Applied Research

The various applied research activities – surveying and evaluating the literature on hypersonic flow theory and experiment basic to the development of methods for analyzing the force and heat loads on a hypersonic aircraft, indicating "new" or "unconventional" phenomena which are not apparent or do not occur at ordinary supersonic speeds, and contributing, where possible, to the understanding and improvement of hypersonic theory – were continued during the period covered by the present report, though at a low level of effort dictated by the limited scope of the study. The current status of these efforts is reported along with some remarks on other topics of current emphasis.

Theoretical investigations of the efficiency of transpiration cooling in reducing the heat transfer from a laminar compressible boundary layer at a hypersonic Mach number — (a) with air as a transpirant, and including the effect of shock boundary layer interaction, and (b) with helium, for constant external pressure — were initiated and are reported.

The use of the classical Kirchhoff's law for engineering calculations of thermal radiation is based on certain assumptions which require careful examination as to their applicability to problems of boundary layer radiation. One conclusion is that rough estimates of radiant energy transfer from the boundary layer in very high hypersonic flows given previously should be recalculated on the basis of a nonequilibrium theory which takes into consideration the dynamic nature of the recombination processes of the dissociated and ionized air components.

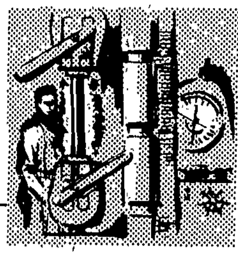
In the previous MX-2276 report (Reference 2), a critical study and an extension of bow shock wave-boundary layer interaction theory was carried out. Experiments in helium at velocities of approximately M = 12 showed, however, considerable discrepancy with theoretical predictions, whereas, at lower Mach numbers, theory and experiment are in good agreement. A recent interpretation of these experimental results suggests that the discrepancy may be attributed to the pressure distribution generated by the curved bow shock in high Mach number flow about a blunt body, due to the creation of a large entropy gradient and, hence, large vorticity at the shock. To check this effect more precisely for air, a numerical characteristics solution of the hypersonic inviscid flow about a flat plate with a circular leading edge was carried out. The results obtained indicate that the blunt nose effects on the inviscid flow are significantly large (of the same order as the shock-boundary layer interaction effects) and must be taken into account.

Some research effort was directed towards two particular aspects of the three-dimensional boundary layer flow problem which are basic to the determination of local rate of heat transfer, skin friction, and pressure distribution for the general type of configuration of interest. These are the flow over a yawed (sharp) wedge and the flow about a slightly rounded leading edge, particularly in the neighborhood of the stagnation region.

In the yawed wedge study, the applicability and limitations of the "strip theory" are indicated, and the effect of shock-boundary layer interaction on a swept wing is pointed out.

Low-speed theory confirmed by recent experiments has indicated the value of rounding off the leading edge and sweeping the leading edge to reduce the local heat transfer rate. Approximate quantitative relations for these effects have been advanced. However, the validity of these relations under the flight conditions associated with the operating range of the MX-2276 needs further justification. Such a justification is given, in part, in a critical review of the theory where the validity of the basic equations employed is considered and the role of the assumed viscosity-temperature law is pointed out.

The Aerodynamics Section has also sponsored a set of experiments (now under way in the Princeton University hypersonic tunnel) to gain some insight into the flow phenomena over a plate with deflected flaps at hypersonc Mach number (M = 13). No results are available at the publication date of this report.

C.  STRUCTURES

1. General

The structural portion of this study has been divided into two general categories: structural design of the specific vehicle as adapted to the requirements listed in Section III, and supplementary work to advance certain studies made under the previous contract. The structural design will be considered first and the supplementary work later. The structural work is discussed in detail in Reference 7.

2. Design Criteria

With the specifically programmed flight plan, and the inherent variation of operating conditions as a function of time, much of the flight-condition criteria defined in MIL-S-5700 becomes inapplicable. Also, the critical thermal conditions must be correlated with loading conditions to define basic design cases for use in insulation, cooling, and stress analyses. Therefore, limit flight-conditions are formulated in a manner which emphasizes time dependence. Condition assumptions are applicable to all probable configuration variations of such a two-stage vehicle, and load factors and associated weights become secondary, dependent parameters to be presented in preliminary loads analyses. The criteria used accordingly differ fundamentally from those defined in MIL-S-5700 in that load factor is not an initial assumption, but rather is a product of the assumed limit condition.

a. Flight Load Environment

The fundamental flight load environment during the ascent consists of an acceleration along the flight path, and a combination of gravity, centrifugal, and lift forces to maintain the flight path. All of the foregoing forces are programmed to occur at specific times, and in a sense, are quite similar to the 1g level-flight loads experienced by conventional aircraft during the majority of their operations. At any time throughout this planned trajectory, however, there may also be: (1) transient loads due to gusts, and/or (2) transient loads due to stabilization system corrections necessary to maintain the trajectory. For this particular preliminary analysis, the long-range basic mission is considered to provide all the design conditions, and other missions are assumed to be programmed in such a way that no more severe conditions are encountered.

Design gust conditions for aircraft are specified in MIL-S-5700, wherein increment loads resulting from specified gust velocity inputs are superimposed upon the 1g level-flight loads. These conditions were not felt to be directly applicable to this system because of the high cruise altitudes. Therefore a study was performed to obtain a conservative extension of the MIL specification criteria. On the other hand, since severe-weather operation of the system is considered unnecessary, the basic sea-level design gust is reduced from 55 feet per second to 40 feet per second for this study.

Transient loads due to stabilization system corrections are strictly a function of the characteristics of the servo system and the load capabilities of the control mechanisms. In this vehicle, stabilization corrections are accomplished by tilting thrust chamber axes, which in turn produce pitch and/or yaw angles of attack. Accordingly, in the design of the stabilizing systems there is imposed the requirement that the maximum permissible transient variations of angles of attack in both pitch and yaw must not exceed ±2°. This criterion is considered reasonable from a stabilization system design standpoint, and provides the basis for structural analysis of transient correction loads.

From the point in the flight trajectory where the speed has reduced down to Mach 4.0, the pilot takes over manual operation of the airplane. This speed may be considered a "V_{h_e}" (level flight design high speed) at the planned altitude at which this occurs. Below this altitude, the speed for maximum L/D may likewise be considered to be the desired V_{h_e}. On this basis, it is logical to specify that the airplane be operated as a Class II Reconnaissance Airplane (per Table I of MIL-S-5700), wherein the maximum permissible pitch maneuver is 2.25g. Accordingly, the conventional aircraft design criteria specified in the MIL specification become applicable during the unprogrammed glide conditions, and the operating instructions to the pilot are derived from these.

b. Design Flight Conditions

In view of the limited applicability of the design conditions contained in MIL-S-5700, special primary design condition combinations were developed for structural analysis. The premise governing selection of loading combinations is: "Each maximum severity loading is considered in combination with programmed-trajectory loads, but never in combination with another maximum severity loading." During the manually-controlled portion of the glide, the conditions of MIL-S-5700 become applicable in design of the airplane.

c. Thermal Criteria

In the case of a single programmed flight trajectory such as is used in this study, the temperatures and heat inputs at all parts of the airplane and booster can be calculated at all times in the flight. These calculated thermal conditions have been analyzed in combination with the calculated load conditions at each particular time that a critical load and/or thermal condition exists. Since each time has a singular load and thermal condition in the planned trajectory, with only momentary transient load inputs from gusts and path-deviation corrections, no arbitrary criteria for load-temperature condition combinations need be formulated. The structure must merely withstand the loads plus transients, and temperatures, encountered in the programmed trajectory.

A new criteria element is introduced by the phenomenon of "creep". Stress allowables at elevated temperatures for short-time single-load applications can be obtained by merely consulting stress vs temperature data for each material. In sustained or often-repeated applications of load at elevated temperature, creep results, causing appreciable permanent deformations or occurrence of buckling failure at lower stress levels. It is therefore necessary to define the "life" of the airframe, and for initial analysis, some amount of permissible permanent deformation. For this study, the life of the airplane is taken as 100 flights, and the booster is used in only one flight. For creep-set analysis, the permissible permanent set is taken as 0.5%, and by analysis it must be shown that this does not result in deformations which adversely affect required performance. Conservative time-durations of limit loads at critical temperatures in each flight must be assumed, prior to multiplying by one hundred, to get total load-at-temperature times for use in obtaining allowable stress values from creep curves.

d. Landing Load Criteria

The landing sinking speed criteria of MIL-S-5703 is applicable to this vehicle. Using these criteria with appropriate system weights the landing loads can be calculated.

e. Propellant Tank Conditions

The rocket-engine propellant tanks are an integral part of the airframe structure in both the booster and the airplane. These tanks are subjected to: (1) gas pressures to prevent pump cavitation, (2) flight-acceleration fluid pressures, and (3) airframe carry-through loads. Realistic combinations of these, along with the aspect of repeated-load cycling, have been used for criteria. The conditions are compatible with the requirements of MIL-S-5700 series specifications, as well as those for rocket engine systems contained in MIL-E-5149.

f. Safety Factors

Since the vehicle is manned throughout both boost and glide flight, the conventional aircraft safety factors of 1.0 for yield and 1.5 for ultimate are employed. Normally, no other primary safety factors are needed for design except: (1) safety factors for pressurized equipment, and (2) a margin of safety in speed between limit design speed and the speed at which flutter or divergence occurs. The factors for (1) are discussed in the preceding subsection; the factor for (2) is the value 1.15 used in conventional aircraft design practice.

Where thermal aspects are introduced in design, it might be argued that additional factors of safety be used. Such factors might be used to cover inaccuracies in boundary-layer temperature analyses, heat transfer coefficient, stress allowables at elevated temperatures, creep-allowable data, and thermal stress analyses. Each of these inaccuracies has a rather large error-tolerance range — in some respects similar to that evident in fatigue-allowable data. It is premature at this time to establish collectively a single safety factor, or even to establish individual values for each of these considerations. Accordingly, no attempt is made to define a "thermal safety-factor", and the analyst is obligated to use appropriately conservative methods of analysis and test data in performing the stress analysis.

3. Structural Description

The philosophy of structural design for this vehicle was established in Reference 8 after a comprehensive study of all methods of construction that might be applicable. In that reference, it was shown that the lightest construction for the primary structure could be achieved by using conventional aluminum alloy structures protected by a layer of insulation and with a water cooling system within the structure to absorb the heat passing through the insulation. To complete this construction a light, heat-resistant outer wall is necessary to protect the insulation from aerodynamic forces and to form the aerodynamic contours (Figure 27).

In addition to its light weight, this construction has other advantages which arise prin-

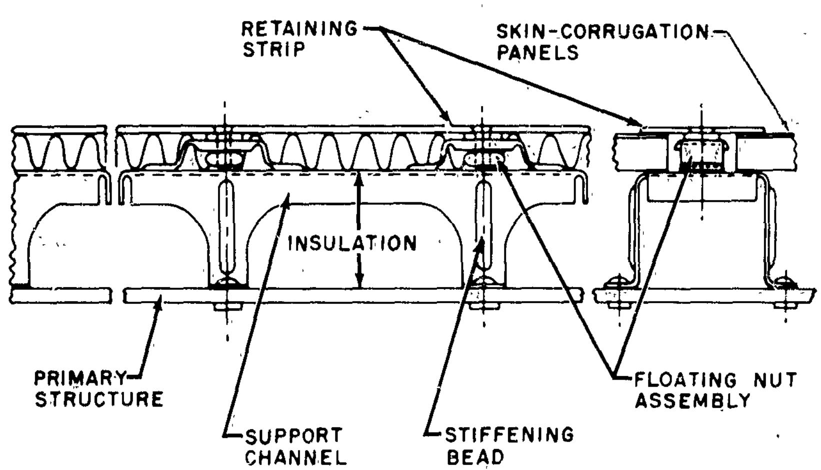


Figure 27. Outer Wall Attachment Structure

cipally from the use of conventional structure and material, and the freedom from thermal stress effects as a result of the cooling system. It is evident, however, that the outer wall presents a complex design problem, since it must have many characteristics such as strength to resist aerodynamic pressures; stiffness to prevent panel flutter; freedom to expand under temperature relative to the aluminum structure; resistance to oxidation, abrasion, high temperatures, and small thermal stresses within its own structural elements during transient flight conditions; and, of course, light weight.

The problem outlined previously was investigated thoroughly, and it was shown that suitable high temperature materials, such as Inconel X and Haynes alloy 25, are available for the construction of an outer wall. It was also shown that the remaining problems could be handled by dividing the wall into small panels, each consisting of a thin skin backed by a corrugated sheet of the same thickness, and each supported in a manner giving complete freedom of expansion.

In the present study, the structural configuration previously discussed, i.e., a water-cooled aluminum structure, protected in turn by a layer of fibrous insulation and a heat-resistant, thermal-stress-free outer wall, has been applied successfully to the primary structure with two modifications. The fixed portions of the aerodynamic surfaces of the proposed vehicle are of very low aspect ratio, and it has been found that an efficient aluminum shell structure cannot be constructed for the very low loadings. Accordingly, these surfaces are constructed of aluminum alloy trusses, acting as ribs and beams, and each structural member is water cooled. No aluminum skin is provided, neither is insulation used, but the external surface is formed by the heat-resistant, free-floating panels described previously.

The other modification in the use of a double-walled construction occurs in the booster where the short flight time makes water cooling unnecessary, and where sufficient insulation is provided by an air space of 0.20 inch between the aluminum shell and the outer wall.

For areas of large heat flux, such as surface leading edges and the fuselage nose, the proposed structure for the present vehicle consists of a skin of coated molybdenum alloy with an internal skin of the same material arranged to form cooling passages. Ribs and other stiffening members, also of coated molybdenum alloy, are attached by welding, and the whole is attached to the wing primary structure in a manner which permits relative expansion. The area is then cooled by circulating molten lithium under pressure; the lithium being finally expanded through a valve into a low-pressure area where part of it boils and the vapor is expended.

In the proposed system, therefore, the heat entering the high flux areas is absorbed by the latent heat of vaporization of lithium, a metal of large heat capacity. Systems using transpiration cooling and using constructions of materials able to resist the high equilibrium temperatures have also been considered, and are promising. The system expected to have the least development time has been selected for this proposal.

a. Fuselage Primary Structure

The fuselage of Stage II is a ring-stiffened aluminum alloy cylindrical shell with a conical nose. It has an over-all length of approximately 70 feet and a basic diameter of 6 feet. The complete surface of this shell is protected from the effects of aerodynamic heating by a cooling system and an outer wall of heat-resistant material. A structural layout of the fuselage is shown in Reference 1.

From the nose back to the cabin, the shell consists of 2024-ST skin material stiffened by light, channel section rings of the same material spaced 11 inches apart. These rings also provide attachment points for the protecting outer wall. Additional skin stiffening is provided by the cooling passages, which are formed within the aluminum walls by constructing the wall of two sheets, each having depressions forming one half of the cooling passages.

Behind the nose section, the pressurized cabin is an unstiffened shell of welded construction, fabricated from 6061-T4 aluminum alloy material, which is artificially aged to the T6 condition after welding. The forward pressure bulkhead is hemispherical in shape, while the aft bulkhead is flat, because of space limitations, and is stiffened by vertical channel section stiffeners. At an 11-inch spacing, the cylindrical shell has extra thickness in the form of circumferential rings to act as attachments for the outer wall.

Behind the cabin, and constructed in the same manner, are the integral fuel and oxidizer tanks, respectively. The fuel tank is approximately 80 inches long, while the oxidizer tank is 150 inches long, and both have elliptical ends of 3-to-1 ratio. As with the cabin walls, the tank walls are extruded so that additional circumferential thickness can be added locally, to provide support for the outer wall. To avoid welded lap joints carrying tension loads, the knuckle is also formed from an extrusion which is then butt-welded to the tank shell, the end, and the skirt.

From the rear of the oxidizer tank to the end of the Stage II fuselage, the shell is aluminum alloy, stabilized by close spaced, "Z" section rings. At 20-inch spacing much heavier rings are used to carry the wing beam reactions. Because of the low aspect ratio of the wings, and the multiplicity of beams in its construction, it is possible to dispense with the conventional carry-through structure by carrying wing bending moments into the fuselage rings. This arrangement permits the structural simplicity of a mid-wing design, while retaining a clear space within the fuselage for equipment and power plant. The most aft beam of the wing also carries the control surface, so that its supporting ring within the fuselage is of box section.

Immediately behind the oxidizer tank, two large cut-outs are provided in the lower part of the fuselage to permit retraction of the main landing gear. These are symmetrically disposed so that a central "keel" is formed between them along the bottom of the fuselage. Heavy edge members bound these cut-outs and carry the axial loads from the interrupted skin. At the rear end of the cut-outs the fuselage frame is again heavy, and its lower portion is deepened into a solid bulkhead and carries the main gear fittings. A second heavy frame is provided, farther forward, to resist vertical loads from the landing gear drag struts, and a fitting running longitudinally transfers drag loads to the skin.

Behind the landing gear cut-outs, the fuselage is structurally complete, but the lower portion is readily removable for equipment servicing. Fittings are provided to effect continuity of load-carrying frames, and skin attachments ensure skin effectiveness in resisting fuselage bending.

At the rear of the fuselage, a six-point attachment is provided for the booster, and local longerons are provided at each fitting to diffuse load into the fuselage shell. The tubular motor mount is also attached to the same fittings. It projects forward into the fuselage, thereby surrounding the motor. In this manner, the mount tubes are placed in tension under thrust loads rather than the more usual compression.

A skin-stringer construction was also considered for the fuselage and, in fact, gives a smaller weight when comparisons are made of the basic structural material. It was considered, however, that no weight advantage would be shown for the added complexity, if full account were taken of the extra stringer clips, frame cut-outs, reduced effective depth of frames, and extra material for the diffusion of stringer loads into the skin in the tank regions. The ring-stiffened shell was therefore adopted.

b. Lifting Surface Primary Structure

In Reference 8, a wing construction was proposed similar to that described previously for the fuselage. It could still be applied satisfactorily to the present wing, but the wing geometry now permits a much lighter arrangement.

The very low aspect ratio combined with the high taper, as presently proposed for the wing planform, gives very low axial loadings in the skin, so that any shell-type construction limited by minimum practical sizes becomes structurally inefficient. The present proposal for the wing structure, therefore, consists of a grid work of spanwise beams and chordwise ribs, with no structural skin. The wing surface is formed by small light panels of heat-resistant material which are supported by the ribs in such a manner that free expansion due to temperature is permitted. These panels and their attachements are identical with those used on the fuselage to cover the insulation. (A structural layout of this wing is shown in Reference 1.)

Because the aluminum alloy ribs and beams will intercept radiation from the hot skin panels, they are provided with cooling passages through which water is circulated. The quantity of heat absorbed is minimized by using an open truss construction, rather than a plate web, for the ribs and beams, and tubular section truss members to form the coolant passages. The "T" section caps are also provided with coolant passages. A one-inch thickness of insulation is applied over the caps of the ribs and webs to minimize direct conduction from the surface panels.

Calculations show a total weight saving of 1080 pounds for the complete wing. The flap area is excluded, since the flap is a shell-type construction. This large difference in primary structure weight justifies additional explanation. It results from the fact that conventional skin-stringer type of construction is inefficient at very low loadings because of the limitations of practical thickness and dimensions, and also because of the condition that no skin buckling should occur below limit loads. This latter requirement is imposed so that the buckles do not produce fracture or crippling of the coolant passages within the skin.

The flap, or control surface, is a conventional aluminum alloy shell structure, but is split by a horizontal plane along the centerline to form two separate torque boxes. These are hinged, at individual ribs, at the upper and lower skin lines, respectively, and are deflected in parallel. The advantage of this arrangement is that no gap is produced at the surface when the control is deflected. It is also possible to deflect the upper and lower portions of the surface individually to provide stability, although such a procedure is not considered necessary at present. The outer surface of the flap is protected from the effects of heating by an outer wall and a cooling system similar to that of the fuselage.

The vertical surface, together with its control, and protection against aerodynamic heating, is similar in construction to the wing.

4. Heat Protection

a. Second-Stage Primary Structure

The primary structure for the second stage is of approximately conventional construction and utilizes aluminum alloy material. Two systems are used for protecting this structure from the effects of aerodynamic heating; one system is applied to the shell-type structures of the fuselage and the control surfaces, and the other to the open-grid structures of the wing and vertical tail. Both systems employ water cooling to absorb the heat entering the primary structure.

(1) Shell-Type (Fuselage)

In the shell-type structures, the cooling passages are formed by pressing the necessary shapes into a thin sheet of aluminum which is then bonded to the outer face of the skin. Alternatively, the skins may be made by the patented Roll-Bond process in which two sheets of material are imprinted with the cooling system circuit and then bonded by mechanical pressure. The bonding is effective only where the imprint of the required passage does not exist, so that by the use of compressed air the unbonded areas can be expanded into cooling ducts. In most cases, the passages are of sufficient length and continuity that the added material is fully effective in resisting primary loads, and some stiffening of the skin is generally realized. For heat fluxes such as will be entering the water cooling system (approximately 1000 BTU per square foot-hour) a passage length of 15 feet can be used, while still retaining only nominal pumping requirements.

To minimize the coolant weight, it is necessary that the latent heat of evaporation be utilized for heat absorption, so that provision is made for boiling of the water as it flows through the system. Because of the large volumetric change between liquid and vapor, the possibility exists that the passage walls may become dry for a sufficient time to permit excessive temperature rise. Accordingly, the flow rates are arranged so that only 10% by weight is evaporated at each passage through the system, and the flow velocities at the end of a 15-foot passage remain low.

After leaving the structure, the steam-water mixture is passed through a separator from which the steam is exhausted overboard, and the remaining water is returned to the tank for recirculation. A small centrifugal pump driven by a steam turbine is used for water circulation, the steam being provided by a small area of the cooling system which is arranged to operate at high pressure.

Around the outside of the aluminum shell, a one-inch thickness of fibrous insulation is applied to reduce the heat flux entering the cooling system to a maximum value of approximately 1000 BTU per square foot-second (Figure 27). To protect the insulation from aerodynamic forces, and also to form the aerodynamic contours, an outer wall of heat-resistant material is provided around the insulation. Since this outer wall experiences high temperatures, provision is made for its expansion relative to the aluminum primary structure by dividing it into small elements of outer skin to which is welded a corrugated sheet of the same thickness. The panel is supported away from the aluminum skin, to provide space for the insulation, by a supporting channel one-inch high and running, in a direction normal to the corrugations, down each edge of the outer wall panel. To accommodate thermal deformations between the supporting channel and the aluminum structure, the channel is divided into short lengths, each length being attached to the aluminum skin by four legs. This arrangement not only minimizes thermal stresses within the supporting channel but also provides only small conduction paths through the insulation. To minimize weight and retain strength, the supporting channels are provided with stiffening flanges.

To hold the outer wall panels into the supporting channels, external rotating strips are provided running down the outside of each edge of adjacent outer wall panels. These retaining strips are also broken into short lengths to permit free expansion. To insure this freedom, each strip is attached to the supporting channel by two screws and two inverted-type floating anchor nuts. These nuts are designed to float in order to accommodate expansion of the retaining strips. They are also arranged to further minimize conduction of heat.

With this arrangement, the outer wall panels are free to expand in the direction parallel to the corrugations by sliding further under the retaining strips. In the direction normal to the corrugations, the outer wall panels are continuous for a considerable length, and thermal expansion in the corrugated sheet is accommodated by a slight closing of the corrugation. Thermal expansions in the outer skin of the outer wall are accepted by separating this skin into elements only four inches wide. And in order to prevent the parallel free edges of the thin skin from being lifted by aerodynamic forces, the edges of adjacent panels are interconnected by a "Pittsburg" joint. This joint is a standard sheet metal joint, but in this application it is made with considerable clearance so that adjacent panels are free to move relatively to permit thermal expansion.

Where the temperatures are 1600°F or less, the outer wall material is Inconel X, while Haynes alloy No. 25 is used for temperature levels between 1600 and 1800°F. On the wing surface, all areas over which equilibrium temperatures exceed 1800°F have the outer surface directly cooled, and an outer wall is not used. A similar situation exists at the fuselage nose. The remaining area is the lower surface of the fuselage between the 14- and 30-foot stations. For this region the outer wall panels are constructed of a molybdenum alloy containing 0.45% titanium. An oxidation-resistant coating is applied to all surfaces of the molybdenum.

b. Open-Type (Surfaces)

In the open-type structures of the wing and vertical tail, the structural cooling is performed by routing the cooling passages along the structural members of the trusses. Generally each truss, such as a rib or beam, has one cooling circuit along each cap, and one through each of the truss shear members in succession. The shear members are circular in cross-section, and the structural size requirements are such that the tubes can be economically used as the cooling passages. The cooling system is identical in operation with the system described for the shell structures.

In order to form an external surface over the grid-work of ribs and beams, floating panels are used, which are again identical with the outer wall described for the shell structures, since the same requirements exist of freedom for thermal expansion and ability to carry aerodynamic loads. Since, with the open-type structures, large air spaces exist between the inner face of the floating panels and the cooled aluminum structure, nothing is gained by the general use of insulation. Hence, it is restricted to areas over the rib caps where the supporting channels for the surface panels are attached.

c. Secondary Structure

For the purposes of the present discussion, the term secondary structure is applied to those areas of the airframe where extreme heating conditions are experienced. These are generally also the areas where structural loads are small. Using this definition, the important areas of secondary structure are the leading edges of the surfaces, the fuselage nose, and, possibly, the control surfaces.

Three methods have been considered for the construction of the leading edges and the fuselage nose. These methods include transpiration cooling with water, passive structures constructed of materials able to withstand the high equilibrium temperatures, and internal cooling using a light metal as a coolant. Of these three methods, the last is presently considered the most feasible and requiring the least development time. It is, therefore, proposed for use in the two-stage vehicle.

The proposed leading edge of the horizontal surface is divided into four sections in each wing, each section being 8.5 feet long and having a chordwise dimension, measured normal to the swept leading edge, of six inches. Each section is mounted so that it is free to expand lengthwise, relative to the primary structure of the wing.

The structure (Figure 28) consists of a leading edge tube, 0.50 inch in diameter and having a 0.025-inch thick wall, to which upper and lower skins of the same thickness material are welded. An internal skin, top and bottom, forms spanwise passages 0.10-inch deep and covering the skin area behind the leading edge tube. Double vertical bulkheads at the ends of each section are arranged to form manifolds connecting the supply and return lines to the surface passages and the leading tube, and internal divisions in the manifold permit the leading tube to be supplied at much higher pressure than the skin area behind. Other bulkheads retain the leading edge shape and form the points of attachment with the wing primary structure. The leading edge structure is fabricated from a molybdenum alloy containing 0.45% titanium, and all surfaces not in contact with the coolant are protected with a ceramic coating. Welded construction is used throughout.

To provide the required cooling, liquid lithium is pumped through the passages at a rate sufficient to keep the maximum molybdenum temperature below 2,000°F. Sufficient pressure is contained within the system to ensure that the lithium does not boil at 2,000°F, in order to prevent vapor from drying the walls and producing burnout under the high flux conditions.

To retain the safety feature inherent in having all passages full of liquid and yet to use

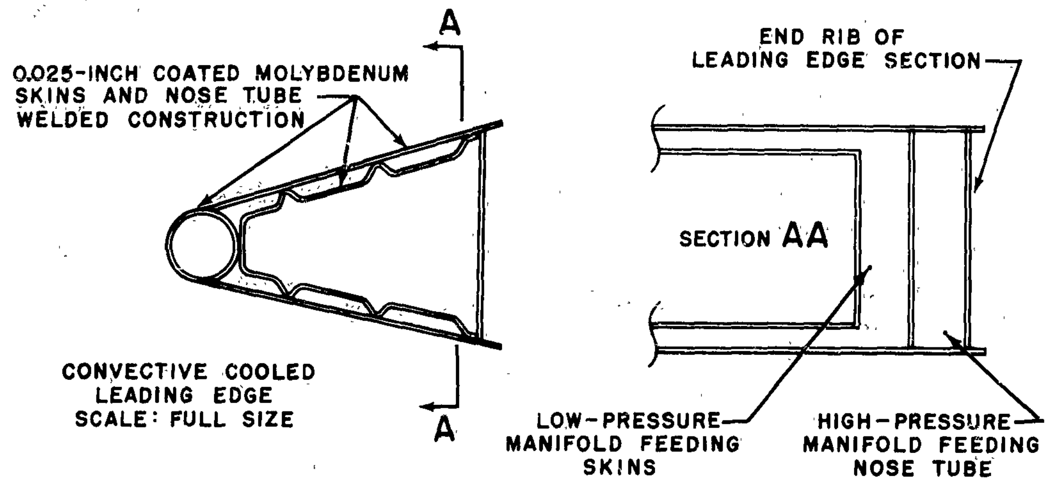


Figure 28. Leading Edge Designs

the latent heat of the lithium as the heat absorbing means, the liquid is forced through an expansion valve, after passing through the leading edge structure, into a low pressure area within the lithium reservoir. As the coolant expands, its temperature drops to the boiling point at the lower pressure, and the excess heat vaporizes part of the lithium, which vapor is then expended. The remaining liquid is pressurized and recirculated, using an electromagnetic pump to avoid the problems of bearings and seals in contact with the liquid metal. Two pumps are required to provide high pressure for the leading edge tube and low pressure for the areas behind.

The fuselage nose and the vertical surface leading edges are constructed in a manner similar to the wing leading edges, and the same type of cooling system is used. Calculations of temperatures experienced on the control surfaces, when the latter are in the deflected position, have not been made. The present proposal is to protect these areas in the same manner as the fuselage primary structure. If the equilibrium temperatures are eventually shown to be less than 2,000°F, it remains only to select the appropriate material for the outer wall panels. If temperatures exceed 2,000°F, then either the panels can be made from a higher temperature material such as a ceramic or cermet, or a heat sink can be provided which takes advantage of the short time during which the surfaces will be deflected. This may be accomplished by using double-skin panels fitted with a good heat absorbing material such as sodium or magnesium. Such materials would absorb the excessive heat input during flap deflection and would then be cooled by radiation and convection when the flap returned to neutral.

d. Booster

The booster has sufficient inherent strength that it does not require internal pressurization during handling and storage; also, stabilizing internal pressure is not required during missile erection while the booster is supporting the full gross weight of Stage II and with any combination of propellant load existing in the booster. The approach used in the design of the booster at this time is considered conservative in both weight and in fabricational methods; moreover, the construction has a weight saving potential which may be realized through more detailed analysis, by the use of higher strength-to-weight ratio materials, and by using more advanced fabricational methods.

Because of the particular flight path of the vehicle during the boost phase, wherein the vehicle assumes almost a horizontal attitude at the end of boost at an altitude of 65,000 feet, the vehicle experiences appreciable aerodynamic heating. This consideration has been an important parameter in the determination of structural arrangement and materials of construction.

The complete booster is made of 6061-T6 aluminum alloy, machine butt-fushion welded, and is an unstiffened, pure monocoque shell. It is covered by an Inconel X outer wall, similar to the fuselage of Stage II, except that no fibrous insulation is needed, since the air space provides sufficient insulation. In this manner, the temperature rise in the aluminum structure at the end of boost is held to a low enough value so that essentially room temperature material allowable strengths can be realized. This has other advantages; namely, thermal stresses which reduce structural strength are minimized, the temperature rise in the LOX propellant due to aerodynamic heating is reduced to an insignificant value, and LOX boil-off prior to launch is reduced.

The LOX and fuel tanks are separate vessels and are joined mechanically during assembly of the booster. In effect, therefore, each tank can be fabricated and acceptance tested individually in the field prior to launch. Both tanks have spherical ends. The upper end of the fuel tank is concave inward (reversed) to accept the lower end of the LOX tank.

Consideration has been given to the design and analysis of uninsulated booster construction employing stainless steel and Inconel X materials having ultimate tensile strengths of 180,000 psi at room temperature. A full monocoque construction having no stiffening elements would weigh more than the design previously proposed. A design using optimum stiffening would compare weightwise only if the serious thermal stresses which would arise are neglected.

Potential weight savings can be realized in the insulated aluminum alloy design by using 24S-T material, by tapering skin gages by chem-milling, and by stiffening the shell in an optimum manner; furthermore, the upper fuel tank end which is convex to pressure load can be reduced appreciably in weight by the development of a resistance-welded aluminum alloy sandwich.

5. Supplementary Studies

a. Secondary Structure — High Heat Flux Areas

(1) General

High flux areas of the airframe are defined, for the purpose of the present study, as those areas where heat input due to aerodynamic heating is sufficient to produce equilibrium temperatures exceeding the useful range of the accepted structural materials.

For the two-stage vehicle of this report, areas of high heat flux included the first six inches of all leading edges (measured in the stream-wise direction), approximately five feet of length at the fuselage nose, and an area on the bottom of the fuselage at the junction of the cone and cylinder. Other high flux areas may be expected to occur locally on the fuselage where the surface leading edges intersect, and possibly on the control surfaces when deflected.

During the present study period, the approach to the structural design of high flux areas has crystallized into a parallel study of three methods:

(a) Convective cooling using an expendable coolant

(b) Transpiration cooling

(c) Temperature sustaining structure.

Convective cooling with an expendable coolant offers the most immediate promise for a structure required to deal with the peak fluxes experienced on the present project. It also produces a system having a practical weight. Transpiration cooling may or may not result in a lighter system; the state of the art is not sufficiently advanced to give a positive answer to this question, and until it is, the study of transpiration cooling will be pursued, since it shows considerable promise.

Structures for high flux areas, constructed of materials able to withstand the extreme equilibrium temperatures, will undoubtedly result in the lightest weight solution to the problem, and will also be free of mechanical complexities. Their success, however, depends on materials research; because of the very high temperatures involved, this method of construction seems to require the longest development time. The present effort in this direction has, therefore, been concerned with following recent developments in very high temperature materials, and noting their applicability to the MX-2276 requirements.

(2) Convective Cooling

The brief study of convective cooling reported in Reference 8 showed the superiority of using an expendable coolant, with lithium giving a significantly lighter weight than all others because of its large value of latent heat. Lithium was not seriously considered, however, since the information available at that time indicated that it was extremely corrosive with all known materials. In a more recent release of information (Reference 9), it is shown that wrought molybdenum sheet material is resistant to the corrosive action of liquid lithium at 1,650°F for long-time service. No numerical data are published and no indication is given of the effects of higher temperatures or of lithium vapor. Presumably, however, the data are intended for industrial use so that the expression "long-time service" implies many hundreds, or, possibly, thousands of hours. For the present use, a life of only 100 hours is required. It is, therefore, assumed that the operating temperature can be raised to 2,000°F.

Molybdenum has many additional advantages, provided that its resistance to lithium can ultimately be substantiated experimentally. It is the only metallic material possessing useful mechanical strength at 2,000°F, thus allowing skin temperatures of this magnitude. It has a high thermal conductivity, and, hence, has small temperature gradients when used as a structural wall. This characteristic plus its small coefficient of expansion, tends to make the thermal stresses small. Its chief disadvantage is lack of resistance to oxidation at temperatures above 1,200°F. Considerable research on coatings to prevent this oxidation is currently underway, especially for applications to turbine blades and afterburners. During this period, no particular attempt was made to study molybdenum fabrication, but indications are that research in this field is also underway.

Lithium also has other desirable characteristics apart from its high latent heat of vaporization. These include high specific heat and high conductivity. The undesirable characteristics are its extreme corrosiveness, the ease with which it burns when exposed to air, and its high melting point (367°F). This last characteristic will require that it be heated for initial distribution through the cooling system and also that the pump and storage area will have to be heated at the initiation of circulation. For descent and landing, during which time the lithium would cool and solidify, it is recommended that the system be kept filled with an inert gas. This gas will fill any spaces left by the solidifying coolant and provide safe melting of the coolant again after landing.

The principle of the cooling system is to pass liquid lithium through suitable passages in the leading edge where heat is absorbed from the structure by evaporation of some or all of the coolant. The resulting vapor is then separated from the liquid and jettisoned. Unfortunately, the conversion of liquid to vapor is accompanied by a volumetric change of such magnitude that even the conversion of a small fraction of the total flow to a vapor may result in significant lengths of the cooling passages being devoid of liquid. Because of the large heat fluxes, the thin structural walls, and the poor heat transfer characteristics of gas, such a situation is intolerable and may result in wall burnout.

To overcome this difficulty and maintain all passages full of liquid, it is proposed to absorb heat by a temperature rise in the circulating liquid. The flow rates are adjusted so that a temperature of 2,000°F is not exceeded. The entire internal circulating system is also pressurized to 2 psi so that boiling of the lithium does not occur at 2,000°F. The lithium reservoir, however, to which the coolant is returned after circulation, is maintained at some conveniently low pressure such as 10 mm. Evidently, in order to operate a system at 2 psi. and yet pass the liquid into a low-pressure tank, the coolant flow must be throttled through a suitable valve. Since the boiling temperature at 10 mm pressure is 1,650°F, while the temperature of the entering liquid is 2,000°F, some of the liquid will be evaporated while the remainder will be cooled to 1,650°F. The metal vapor is jettisoned into the atmosphere and the liquid is recirculated by a pump which again raises its pressure to 2 psi. In this manner, advantage is taken of the latent heat of vaporization, but the safety of liquid cooling is maintained.

A typical section through the lithium-cooled leading edge is shown in Figure 28. The section is taken in a direction normal to the leading edge line and represents a 6-inch chordwise dimension in the stream direction, so that, as previously explained, all areas where equilibrium temperatures exceed 2,000°F are included.

Preliminary calculations were made to obtain an idea of the magnitudes of the equipment and flow required for convective cooling. These calculations indicate that both high- and low-pressure electromagnetic pumps should be used. The high-pressure pump would be required to develop 90 psi and pump approximately 55.4 gpm through the leading edge tube. The low-pressure pump would be required to pump 185 gpm and develop a pressure of 9.2 psi. This pump would service the remainder of the leading edge. A total pump weight of 180 pounds has been estimated.

(3) Transpiration Cooling

Transpiration or "sweat" cooling is a system, suitable for cooling the outer shell of an airframe, in which the cooling fluid is forced through an outer shell designed for that purpose and made of a porous material. If the coolant is a gas such as air, then cooling is obtained by the heat absorbed due to temperature rise in the gas as it passes through the wall thickness. In addition, a reduction in heat transfer coefficient, and, hence, in the heat transferred into the surface, is produced by the film of cool gas built up between the boundary layer and the surface. If a liquid coolant such as water is used, additional cooling is obtained by using the latent heat of the liquid. Again the coolant issues as a gas and forms a protective film over the surface. Since air is not readily available for the present application, only the use of water as a coolant will be considered. The high latent heat of lithium suggests its use also as a transpired coolant; but since this will involve many difficulties in addition to those experienced in transpiration cooling with water, it is considered at present to be a subject for future study. Furthermore, the use of cooling materials of high heat capacity for transpiration cooling is not entirely beneficial, since the resulting smaller mass flow of gas may give smaller reductions of heat transfer coefficient.

The important advantage of transpiration cooling, particularly with a liquid, is that a reduction of heat input is achieved in addition to the full use of the heat capacity of the coolant. This advantage may be expected to result in smaller coolant weights as compared with those of a corresponding convectively cooled system. A further advantage, of considerable importance where very large heat fluxes are present, is that the cooling is performed within the wall so that the heat is not conducted through the wall thickness. At large flux values, even a thin wall of good conductivity material may offer sufficient resistance to heat flow so that excessive temperatures are reached on the outer surface while the inner surface is convectively cooled.

A comparison between transpiration and other forms of cooling indicates that the heat flux to be absorbed if transpiration cooling is used is approximately one-half of that to be absorbed by a convectively cooled system. The calculations for these results assume that, for the transpiration system, the coolant is water, and that the mass flow of steam is related to the latent heat of water and the final heat flux. Because of present limitations in knowledge, many other assumptions are also involved in the aerodynamic aspects of the heat flux calculations.

The factor of one-half between heating rates for the transpiration and convective systems is overshadowed by the ratio of ten between the heat capacities of the presently proposed coolants—water (transpiration) and lithium (convective). Nevertheless, it is desirable to continue the study of both systems, since much more work is required on both before fully reliable weight comparisons can be made; also, it may ultimately be possible to use lithium for transpiration cooling.

From the structural point of view, therefore, the major difficulties with transpiration cooling are the production of a suitable porous material, and the metering problem already described. Two types of porous wall material are presently being studied and, in some cases, produced. In the powdered metal types, sheet material is formed by sintering compacts of metal powders, the porosity being controlled by the grain size of the powder and the density produced in the compacting process.

The second type of porous wall material is formed by taking a woven wire cloth and rolling it to close the spaces between wires in order to obtain the desired degree of porosity. Such a material is made by winding wire around a mandrel, overlapping the windings in two directions, and then sintering to fuse all wire cross-over points. The material is then taken from the mandrel, slit, and rolled out as a flat sheet. Further rolling may be performed to achieve the required porosity. This latter process is particularly flexible, since the porosity may be controlled by size and spacing of wire, the number of layers, and the degree of rolling after sintering.

A typical section through a transpiration-cooled leading edge design is shown in Figure 29. It will be seen that the outer porous wall is supported by spanwise fins on an internal skin. In this manner, chordwise divisions are made so that different flow rates can be applied to each element as required by the heating conditions. Since the internal, finned wall is at water temperature, it may be attached directly to an aluminum structure with no provisions for expansion. The finned wall itself is made of stainless steel, since the fins may be hot, and it permits spot-welding of the porous material for attachment. Sheet steel ribs are welded to the finned wall. For convenience of metering, the cooling passages are divided into spanwise sections 24 inches long by suitable barriers. Each section is fed by an internal manifold formed by two closely spaced ribs. This manifold is again divided into three areas feeding individually the top surface of the

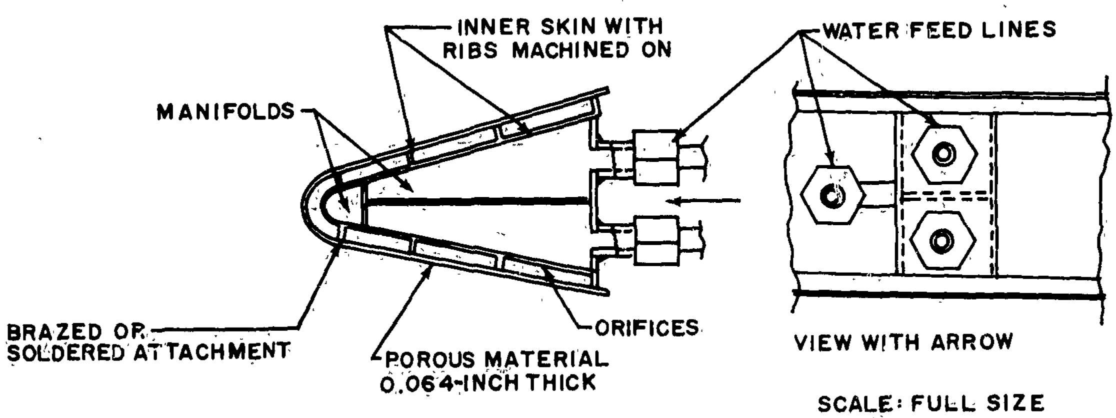


Figure 29. Transpiration-Cooled Leading Edge

quirements. In order to be sure of the capabilities of such a system, several aided inertial systems were also examined. These included:

a. Doppler inertial

b. Stellar inertial

c. Radar or optically monitored inertial

The Doppler inertial and stellar inertial systems were eliminated for various reasons, the most important being the unknown performance of the Doppler equipment at these extreme speeds and altitudes and the shock wave-boundary layer problems associated with the use of a stellar system at hypersonic velocities. The use of radar or optical monitoring was eliminated on the basis that it would not increase the accuracy enough to be worth the additional equipment. However, both radar and optical equipment will be provided for use by the navigator, but no provisions are made for using this information to correct the inertial system.

The pure inertial system selected is fully automatic. However, a secondary navigation system of reduced accuracy is provided to allow the pilot to navigate in case of emergencies involving failures of the primary system. This secondary system is chiefly to insure that the pilot can fly to a base and that the vehicle can be recovered.

The inertial system derives, for navigational purposes, much of the information which is also necessary for the reconnaissance equipment. For example, to attain the inherent resolution of the reconnaissance equipment, velocity and altitude information is required for image motion compensation in the photographic system, and attitude angular information is necessary for data stabilization of the radar presentation and photographs. For proper interpretation of the data, the outputs of the reference unit are simultaneously recorded and correlated to the outputs of the reconnaissance equipment. The inertial platform, which is the basic element of the navigational system, provides a highly accurate vertical reference for the reconnaissance equipment.

2. Inertial Reference System

a. System Description

The proposed inertial reference system will be essentially the instrumentation described in Reference 11. A multiaxis gyrostabilized platform establishes the orientation of a set of orthogonal reference coordinates and also provides a vertical reference and a means for determining the attitude angles of the aircraft needed for stabilizing the aircraft and for data stabilization in the reconnaissance system. The position computer double integrates the outputs of accelerometers, measuring along the axes of the reference coordinates, to obtain present position. The position computer also generates the signals necessary for the maintenance of the desired platform orientation and the correction terms to convert the output indications of the accelerometers from an inertial to an earth-fixed reference.

(1) Reference Coordinates

The set of reference coordinates selected for a two-stage reconnaissance version are identical to those for the three-stage version. This is an earth-fixed transverse spherical system, which can be considered as a modified latitude-longitude system based on a transverse equator defined as the great circle connecting the take-off and landing sites.

The relationships between the location of a point in the transverse system to its location in terms of conventional latitude and longitude can be derived from Figure 30. Λ is defined as the angle between the earth's equatorial plane and the plane of the great circle defining the transverse equator. The principal meridian is defined as the meridian passing through the northern most point of the transverse equator. L', longitude in the transverse system, is measured in an easterly direction from the principal meridian and λ', transverse latitude, is measured in a northerly direction from the transverse equator.

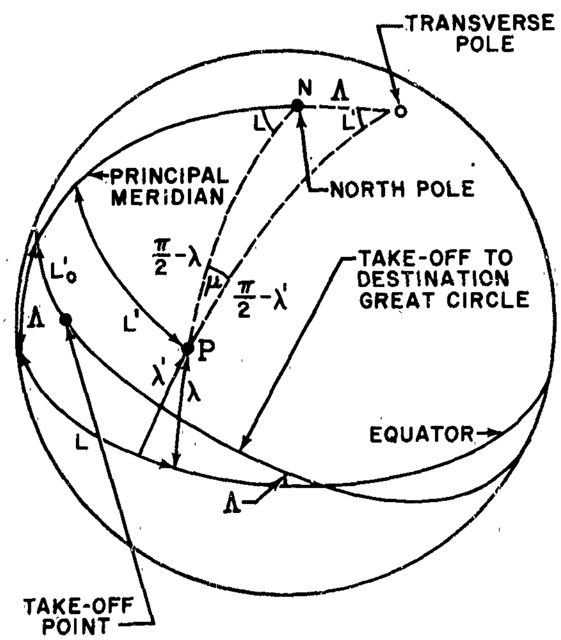


Figure 30. Relationship of the Transverse System to a Conventional Latitude and Longitude Coordinate System

(2) Stabilized Platform

The inner element of the stabilized platform maintains the measuring axes of the three accelerometers along the direction of the transverse latitude and longitude coordinates and the local vertical. Supervision of the platform is achieved through the use of accurate gyroscopes.

There are two state-of-the-art philosophies concerning the mechanics of the supervision. The first method maintains the gyroscope axes continuously aligned to the reference coordinates by torquing the gyros. In the second method, no torquing is involved and the gyros establish a reference fixed in space. The second method requires additional gimbals in order to maintain the accelerometers along the reference coordinate axes. To provide the proper alignment in the first method, it is necessary to torque the gyros at a rate dependent on the earth's rotational rate and the velocity of the vehicle. Thus, the advantage of less gimballing for a system using gyro torquing techniques tends to be balanced by the requirements of high gyro torquing rates as the speed of the vehicle increases.

The flight profile is such that the major component of velocity lies along the great circle connecting the take-off and landing sites (the equator of the transverse coordinate system). The velocities in the transverse latitude direction are lower. Therefore, the stabilized platform combines the features of the torqued and the untorqued gyro methods. This is accomplished by a mechanical rotation between the element containing the accelerometers and the element containing the gyros. The required torquing rates about the other two axes are resolved into the coordinates of the gyro torques.

Initially, at take-off, the platform and gyros are aligned to the local vertical. The gimbal geometry selected has an outer gimbal whose axis is coincident with the aircraft pitch axis and an inner gimbal whose axis is coincident with the aircraft roll axis when it is in level flight. This configuration allows unlimited freedom about the aircraft pitch axis. Although gimbal lock occurs at roll angles of 90°, operational studies have indicated that high roll angles will not be required. This platform configuration is shown in Figure 31.

This particular gimbal configuration requires an acceleration on the outer gimbal due to angular accelerations about the aircraft vertical and longitudinal axis when the aircraft is at a pitch attitude angle. However, the required torques should be sufficiently small so as not to constitute a problem. The platform stabilization system acts to null the gyro pick-off indications by controlling the platform torques.

b. Position Computer

The purpose of the position computer is to act on the outputs of the accelerometers aligned along each of the reference coordinate axes in such a way as to provide a present position indication of the aircraft. An auxiliary

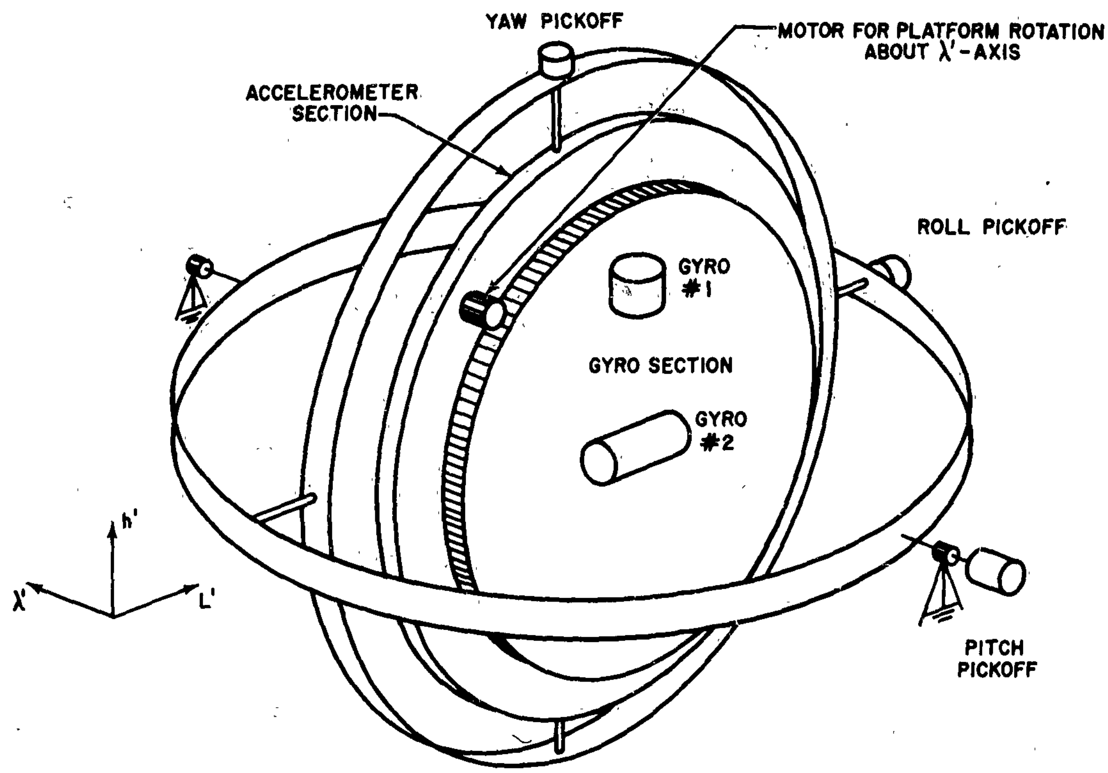


Figure 31. Stable Platform for MX-2276 Navigation System

function of the position computer is the computation of the signals for torquing the gyros and for positioning the accelerometer platform section that are required to maintain the desired reference coordinate system.

To determine the present position of the aircraft, the time derivatives of the velocity in all three directions are double integrated. Keeping the accelerometers aligned to a reference coordinate system that is fixed with respect to the earth causes them to be rotated with respect to inertial space. As a result, the accelerometers will indicate components of centripetal and Coriolis acceleration in addition to the desired time derivatives of velocity. Therefore, before integrating, correction terms must be added to the accelerometer indications to obtain the desired time derivatives of velocity.

Since the instrumentation of the vertical channel has some special problems, and its contribution to the indication of present horizontal position is in the computation of correction terms, it will be presented separately.

The long range and high altitudes of the vehicle's flight path require correction of the radius of curvature of the geoid as a function of latitude, direction, and altitude. In addition, a combination of Schuler tuning and compensation for earth's rotation is utilized to keep the accelerometers properly aligned with the axes of the reference coordinate system. Figure 32 is a block diagram of the instrumentation necessary for proper orientation of the position computer.

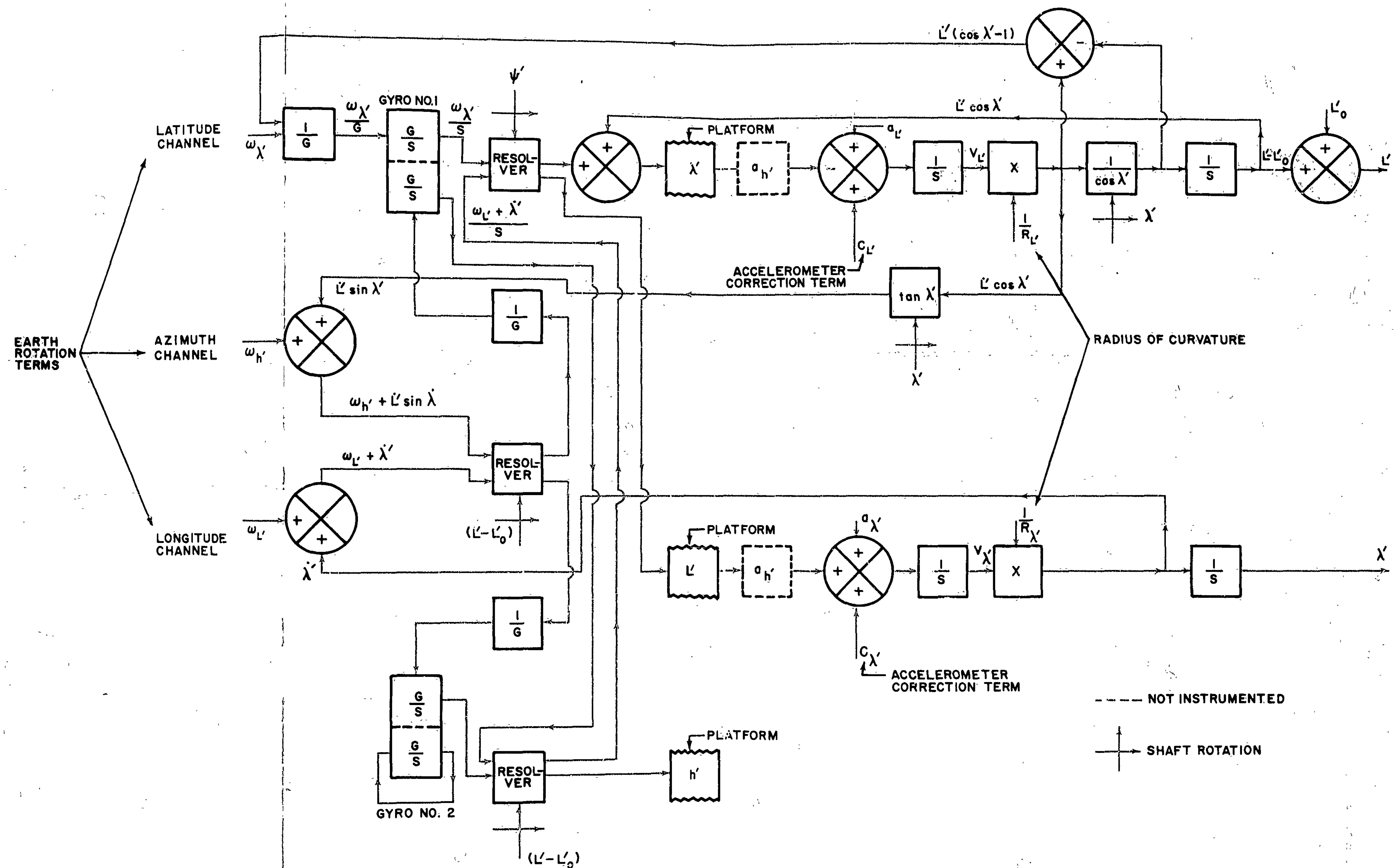


Figure 32. Functional Diagram of Navigation System

c. Accuracy Considerations

For this preliminary analysis, all errors are assumed to be standard and non-varying in time. The errors are subdivided as to those that appear as constant errors in gyro drift, acceleration, velocity, etc., and those that appear as calibration constants. In deriving the expressions for position errors, only first order effects have been considered. Cross-coupling through the vertical acceleration is assumed average and constant over the flight path. A study has shown that this approximation is valid within about two percent.

The effects of various constant errors for different average vertical accelerations were determined. From these analyses, it is apparent that a given acceleration and velocity error contributes a larger error in indicated position as the vehicle speed increases. The opposite is true for the effects of gyro drift and torquing.

Based on certain flight path assumptions, instrumentation ranges and accuracies were determined that would provide a 10,000-foot CEP at the end of a 4,500-nautical mile flight of 2,500 seconds duration. The individual standard errors have been computed and listed in Table I and the indicated position errors at the end of flight in Table II. Figure 33 shows the inertial system error in transverse longitude and latitude as a function of range. From this curve, it can be observed that if two bases can be accurately located with respect to each other, and if two flights are made, one in each direction, the largest error in indicated position is in the order of 3,000 feet. This is because only the first half of each flight is considered for location purposes.

The error in locating any point in enemy territory, then, is the intersection of the curves denoting the error in indicated position of the two flights combined with the error in the reconnaissance equipment in locating that point with respect to the aircraft and the error in the relative location of the bases. If it is possible to locate two bases within 500 feet, the reconnaissance data errors should be small enough to make these data useful for charting purposes.

TABLE I. INSTRUMENTATION PERFORMANCE SPECIFICATION

	Range		Standard Error	
	Transverse Longitude	Transverse Latitude	Random	Calibration
Acceleromater	±5g	±1.5g	10^{-5} of full range	3×10^{-4}
Velocity Integrator	±17,000 ft/sec	±60000 ft/sec	2×10^{-4} of full range	3×10^{-4}
Range Integrator	±5000 n. mi.	+500 n. mi.	10^{-4} of full range	3×10^{-4}
Gyro Drift and Torquing Torquer Calibration			0.03°/hr	10^{-4}
Computations Schuler Loop Feedback Gain Radius of Curvature of Earth Accelerometer Correction Terms	3.5 ft/sec^2	7 ft/sec^2	2×10^{-4} of maximum	10^{-4} 3×10^{-4}
Gyro Torquing Terms	23.5° /hr	75° /hr	2×10^{-4} of maximum	
Initial Platform Leveling			To accuracy of accelerometer	
Initial Azimuth Alignment			10 seconds of arc	

TABLE II. COMPUTATION OF CEP

t = 2500 seconds R = 4400 nautical miles		
Accelerometer		
Random Error	2475	743
Calibration Error	126	702
Velocity Integrator		
Random Error	985	347
Calibration Error	126	702
Range Integrator		
Random Error	2880	304
Calibration	126	1199
Horizontal Gyro		
Random Drift & Torquing Errors	6750	6750
Torquer Calibration Errors	--	965
Azimuth Gyro		
Random Drift & Torquing Errors	--	831
Computations		
Schuler Loop Feedback Gain	2631	965
Radius of Curvature	126	702
Accelerometer Correction Terms	1070	2140
Gyro Torquing Terms	1060	3380
Initial Platform Leveling	2040	614
Initial Azimuth Alignment	--	1300
R.M.S. Error	8625 ft	8360 ft
CEP 9990 ft		

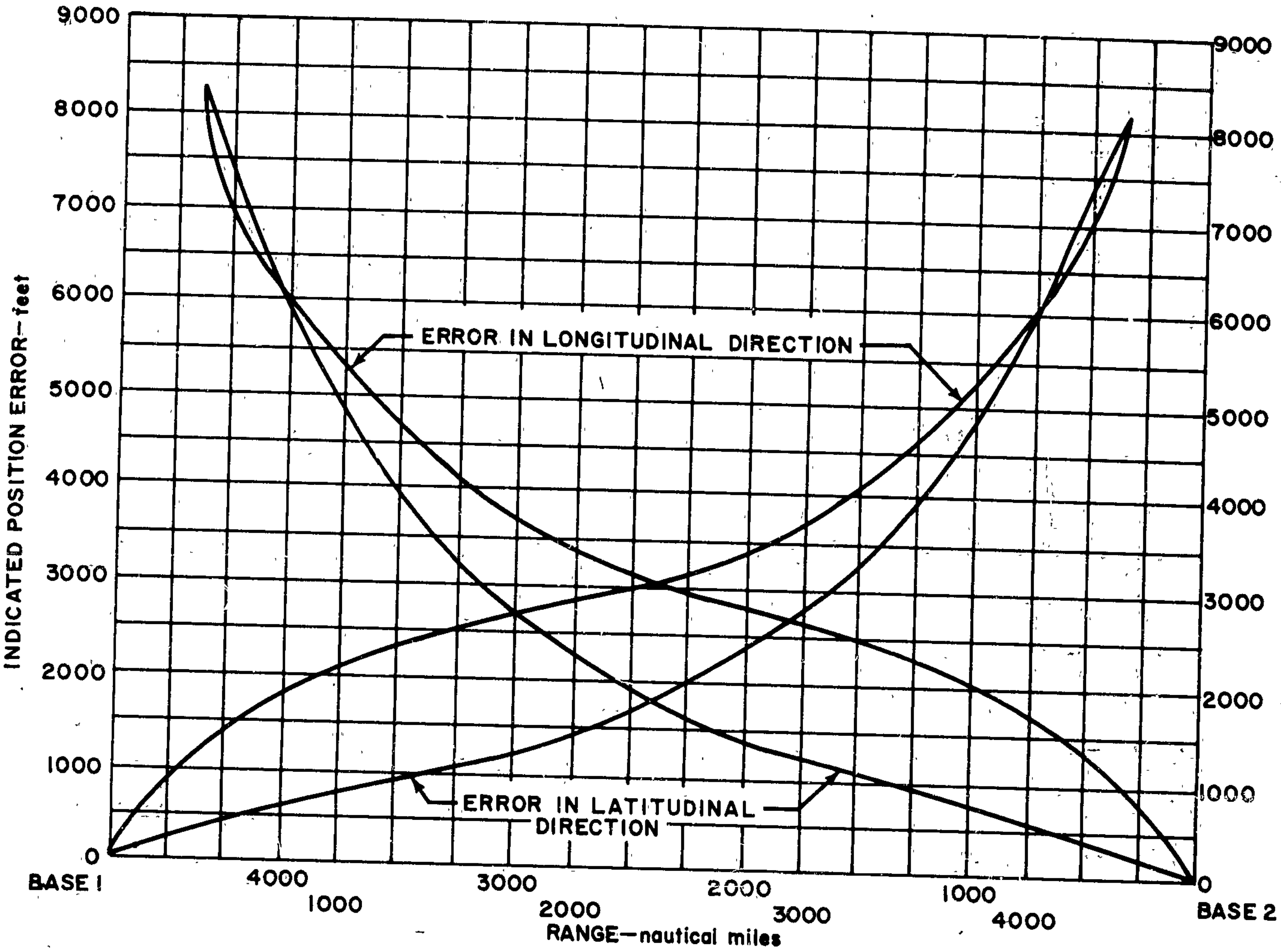


Figure 33. Indicated Position Error as a Function of Distance Traveled

Since the stable platform is also used as an attitude angle reference for the purpose of data stabilization of the reconnaissance information, Figure 34 is presented to show the errors in platform orientation as a function of time.

d. Instrumentation

The instrumentation of the MX-2276 navigation system will contain the basic elements of an inertial reference system. Gyros supervise the stabilization of the platform, accelerometers act as the primary sensing elements, and integrators obtain velocity and position information. Auxiliary computing elements are necessary to generate the required accelerometer corrections, the platform torquing signals, and the radii of curvature. In addition, the high speed and altitude of the MX-2276 vehicle requires an accurate instrumentation of vertical velocity and altitude for the purpose of performing the auxiliary computations.

As is true with any weapon system program, the instrumentation requirements previously specified are probably unique for the MX-2276 application and no components are presently available or under development that

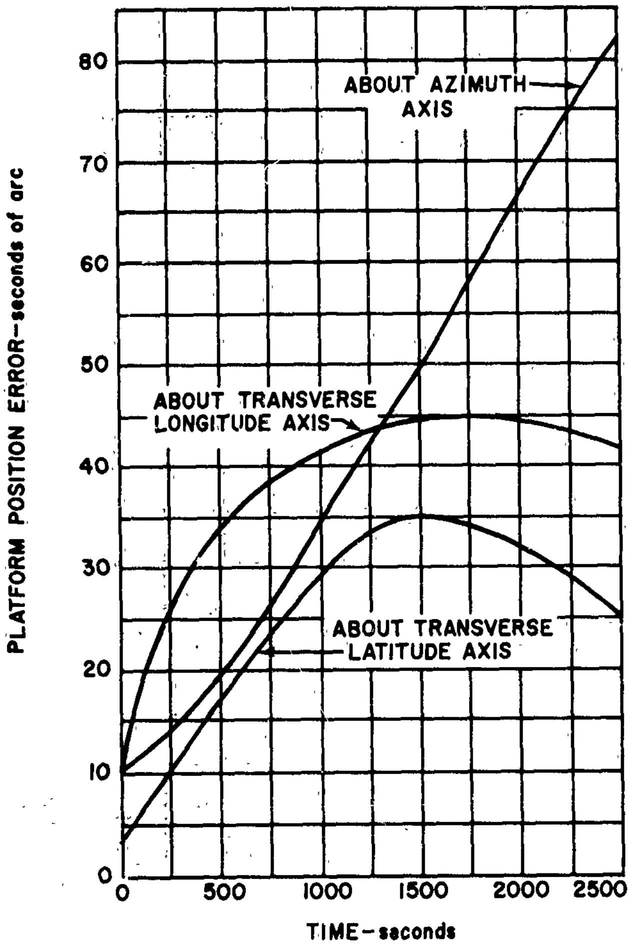


Figure 34. Error in Platform Position

would satisfy these particular requirements. However, the instrumentation requirements are not of a nature to necessitate breakthroughs in the state of the art in the development of inertial components, but rather they represent only a realignment and extension in range and accuracy of components existing or under development for present, definite, weapon systems applications.

(1) Vertical Channel Instrumentation

Vertical information is required in two places in the instrumentation. Vertical velocity is used in the computation of the accelerometer correction terms, and the altitude is utilized for determining the radii of curvature. An additional requirement, to be described later, is the computation of the image motion compensation signal for the photographic systems.

To obtain the accuracy specified in the preceding error analysis for the computation of the accelerometer correction terms, vertical velocity should be known to an accuracy of about 1 foot per second. The effect of errors in the computation of the radii of curvature has been shown to be relatively small; therefore, errors in altitude from a position indication aspect are not too important. However, image motion compensation requires a knowledge within 2,000 feet of altitude above terrain.

The high altitude of the MX-2276 vehicle precludes the use of the method of sensing static pressure to determine altitude and vertical rate to these accuracies. Therefore, to maintain a completely nonemanating system, an inertial instrumentation of the vertical channel would be required. Figure 35 shows a possible instrumentation.

The error in indicated altitude as a function due to standard errors in vertical acceleration, velocity, and position is shown in Figure 36, and the error in indicated velocity in Figure 37.

This instrumentation, while probably giving sufficient accuracy for the position computer computations, would still have to be more accurate, as seen from Figure 36, to provide the necessary image motion compensation signals towards the end of the flight. An additional consideration is that image motion compensation requires height above the terrain, rather than absolute altitude as measured by an inertial instrumentation.

The analysis of the vertical instrumentation indicates that a completely nonemanating instrumentation in this channel would present some complex computations, and that the determination of the variation of gravitational acceleration as a function of altitude, latitude, longitude, and terrain height may be of too

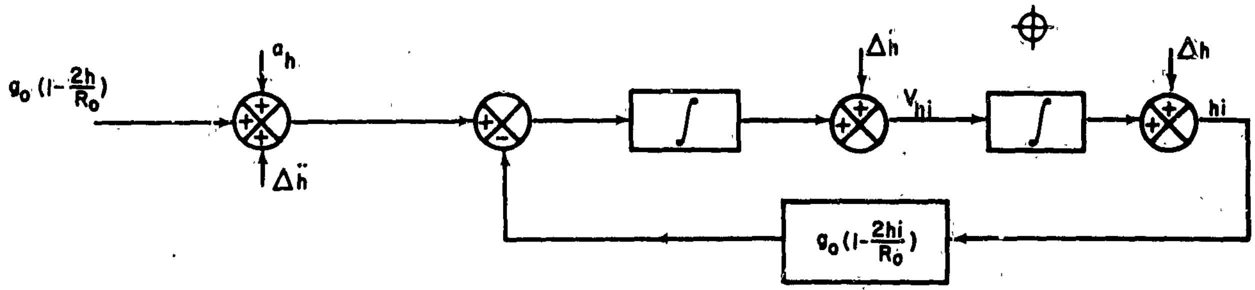


Figure 35. Inertial Vertical Instrumentation

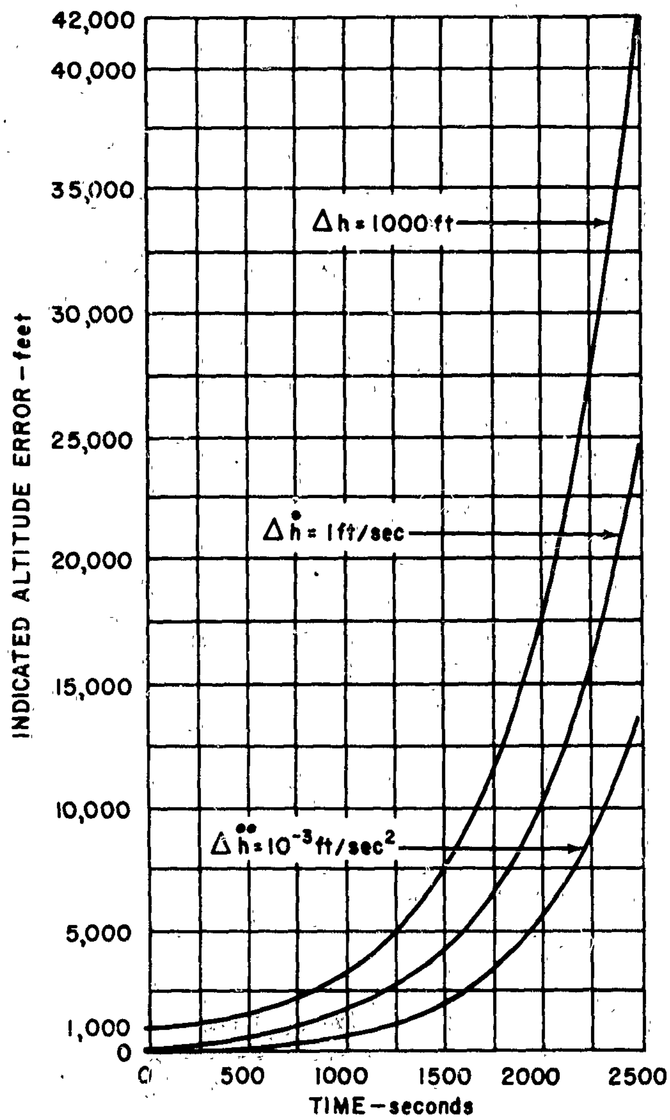


Figure 36. Error in Indicated Altitude for an Inertial Vertical Instrumentation

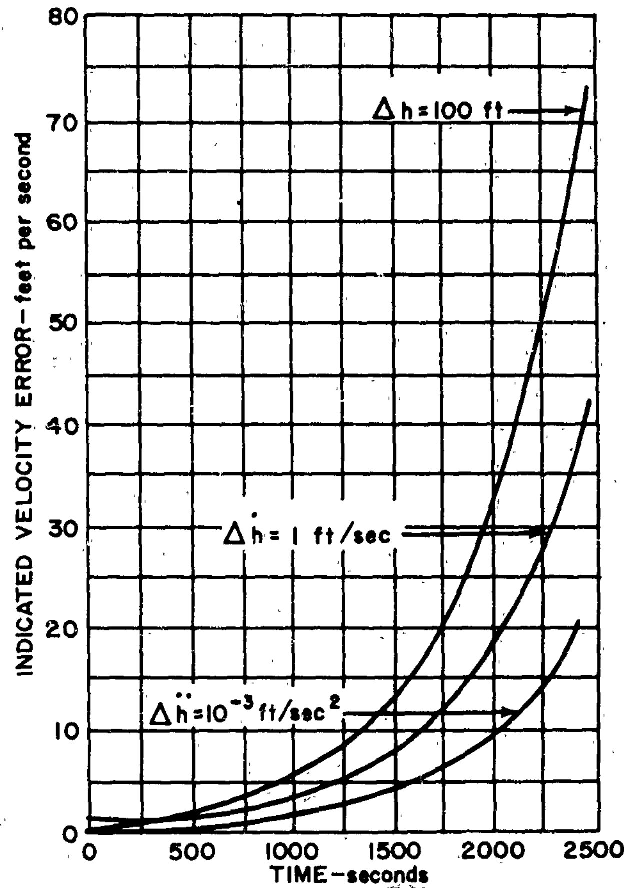


Figure 37. Error in Indicated Vertical Velocity for an Inertial Vertical Instrumentation

questionable accuracy to warrant its use. Therefore, it is felt that a radar altimeter is necessary to provide the basic altitude information. The output of the radar altimeter will be compared with the inertial indication and the difference used to correct the vertical accelerometer output. The inertial instrumentation filters the radar power spectrum thus tending to increase the instantaneous accuracy of the combined system to a considerable extent. It also acts as an extrapolator providing vertical information during radar cut-off periods.

As can be seen in Figure 37, a 1000-foot uncorrected terrain height would require a radar correction approximately every 250 seconds to provide the necessary vertical velocity accuracy. To eliminate the necessity for the knowledge of, and correction for, terrain height, it is proposed that the radar monitoring be continuous.

(2) Gyros

For the time of flight of the MX-2276 mission, the error analysis indicates that the gyro errors are most significant. There are several gyros under development for various programs that have bench accuracies that could satisfy the MX-2276 requirements. However, these gyros are of rather large size and weight. The use of smaller gyros with special operating techniques would constitute a considerable saving in weight and space.

Both the Navan principle of North American Aviation, whereby the spin direction of the gyros is periodically reversed, and the Bell Aircraft method of rotating the gyro housing about the spin axis have shown laboratory performances of better than 0.01 degree per hour. The successive acceleration and deceleration of the gyro rotors in the Navan system results in some deterioration of the individual gyro performance and also limits the frequency of reversing. Six, rather than three, single-degree-of-freedom gyros are required for this method. On the other hand, using two-degree-of-freedom gyros, rotated about the spin axis, does not require any additional gyros and also allows a higher frequency of rotation. Since a rotation about or reversal of the spin vector tends to average the gyro drift torques during each period of rotation, higher frequencies give better averaging properties.

All known gyros existing or under development, whether large or small and regardless of special operating techniques, have unbalances of sufficient magnitude to make their use alone under the MX-2276 acceleration profile extremely marginal. One solution to this situation would be to improve the balance characteristics of the gyros – which could prove to be a problem of considerable magnitude. The problem of balance shifts and nonisoelastic effects occurring during acceleration would also have to be improved. Therefore, since the latter effects would have to be reduced in any case, the effect of an unbalance could readily be eliminated by supplying a correction torque proportional to acceleration.

For a gyro operating in the conventional mode, unbalances and accelerations both along and normal to the spin axis contribute to the total unbalance torques. It then becomes necessary to compute the effect of each acceleration separately to avoid ambiguities in the compensation. The use of the rotating gyro housing principle, which is proposed for the MX-2276 application, necessitates a correction only for accelerations normal to the spin axis.

The nature of the operational flight path is such that appreciable accelerations occur along each horizontal reference axis, and the large changes in velocity cause corresponding changes in the vertical acceleration. Therefore, there does not appear to be an optimum orientation of the gyro spin axis to minimize the effects of unbalances under accelerations.

Further improvements in accuracy and reduction in size of the proposed navigation system could be accomplished with improvement in the performance of gyro torquers. The existing state of the art of gyro torquers precludes the selection of a gyro orientation that is earth fixed, since this requires a rotation of the platform about the transverse latitude axis due to the high velocity of the MX-2276 vehicle. A further restriction, although not critical from an operational aspect of the MX-2276 system, is the 18-degree limitation on launching with respect to the take-off-to-landing site great circle, and the 500-nautical mile deviation from the great circle. This is necessary to limit torquing rates about the transverse longitude axis to values (75° per hour) that can be performed by present gyro torquing techniques to the required accuracy.

(3) Accelerometers

Since acceleration information is believed necessary to compute correction signals to compensate for the effect of gyro mass unbalances, accelerometers, rather than velocity or distance meters, have been selected as the basic sensing elements of the MX-2276 navigation system.

The high accuracy requirement rules outs the use of conventional direct reading instruments. A servo-constrained, null-type accelerometer, however, is capable of providing the necessary accuracy. These units are based on the principle of a displacement of an unbalanced mass under acceleration being measured by a pick-off. This displacement signal, after being amplified and compensated, supplies a current in a torque caused by the acceleration of the unbalanced mass. The current in the torque coil is then proportional to the acceleration.

Several high-performance accelerometers of this type are presently in an advanced stage of development by various contractors. These instruments have random errors in the region of 2×10^{-5} to 10^{-4} of the full range. Despite a lack of satisfactory test procedures and equipment preventing the true evaluation of these instruments at inputs larger than one g, it is thought that these accuracies would also hold true for larger inputs. In fact, the instruments presently under development for various weapon systems are for ranges considerably in excess of one g.

With the accuracies of the instruments presently being developed, it does not appear that the MX-2276 requirement presents any development problem.

(4) Computing Elements

In addition to the velocity and range integrators, auxiliary computations are necessary for the generation of earth's rate, accelerometer correction, and radii of curvature terms, as well as for resolution of the gyro torquing and pick-off signals into the proper coordinates.

As previously mentioned, the outputs of the inertial reference unit are used in two places: first, in the navigation computer to generate the proper flight path; and second, the outputs of the inertial reference unit are recorded simultaneously with the reconnaissance data for the purpose of correlation. The navigation computer inputs may be either in a digital or an analog form. However, to record accurately the necessary data, this information should be in a digital form. To obtain the necessary accuracy, the accelerometers should have an analog output. Therefore, the position computer must be a hybrid instrumentation with analog inputs and digital outputs for recording. Somewhere in the instrumentation, a conversion from an analog to a digital form must be performed, and it is proposed to make this conversion at the angular velocity level.

This method of instrumentation reflects the latest philosophy of inertial system instrumentation. However, the accuracy requirements are such that a completely digital instrumentation utilizing time-sharing techniques may show a considerable saving in size and weight.

e. Leveling and Alignment

In discussing leveling and alignment, it is necessary to consider again the reference coordinate system. This system is based on the earth being an ellipsoid and, therefore, the transverse latitude and longitude axes are in a plane parallel to the tangent plane of the ellipsoid with the vertical axis normal to the tangent plane. The vertical axis corresponds to the normal direction of the plumb bob. Since the accelerometers are used to level the platform, the local gravitational anomalies at the launching site must be considered during the ground leveling of the platform. The initial azimuth alignment consists of aligning the transverse longitude axis of the reference coordinate system in the direction of the great circle connecting the launching and landing sites. The leveling loops for both the transverse latitude and longitude channels are essentially identical. If the platform is not level, the accelerometer has an output which is integrated by the velocity integrator. This output is fed back through the proper gyro torquer until the indicated velocity is zero. This condition will exist only when the output of the accelerometer plus the compensation, for the local gravitational anomalies is zero. The platform is thus leveled to the reference coordinate system within the accuracy of the accelerometer. The correction term for gravitational anomalies is different for each horizontal axis owing to its directional properties. Therefore, before the system is finally leveled, it is necessary to accomplish azimuth alignment.

The possibility of automatic alignment was investigated. It was found that extremely accurate gyro performance would be required; therefore, an optical system is proposed for final alignment.

The foregoing description applies to the normal method of ground leveling and alignment of the platform. This method of leveling can be modified in such a way that automatic setting of the proportionality constant of the gyro acceleration correction term can also be accomplished.

3. Navigation Computer

The purpose of the navigation computer is to generate the necessary signals to cause the airplane to follow two basic types of flight paths required for a reconnaissance system. Search or area coverage usually involves a series of parallel flights, while for a detailed target analysis the aircraft is navigated to a known area with an accuracy within the coverage capabilities of the reconnaissance equipment. The inputs to the navigation computer are obtained from the inertial reference unit. The outputs are used by the control system to cause the aircraft to follow the desired flight path.

a. Area Search Coverage

The area search coverage will consist of a series of flights parallel to the great circle connecting the take-off and landing sites. Ideally, these can be considered as parallels of transverse latitude. The easiest method of generating these flight paths is to program the aircraft to fly at a constant transverse latitude. In order to follow this path, it is necessary to counteract the the Coriolis and centripetal forces acting normal to the flight path. It is expected that the airplane will have to be rolled to provide the necessary side force. Figure 38 shows the roll angles necessary to counteract these forces as a function of velocity for various flight paths parallel to the great circle between the take-off and landing sites.

The amount of roll angle allowable during flight is restricted by the operating characteristics of the radar and photographic reconnaissance equipment as well as by a consideration of the aircraft performance capabilities. If the reconnaissance equipment is unstabilized, the existence of this roll angle would mean a decrease in coverage. However, since this roll angle is predictable, its effect could be partially compensated for in planning the missions. Another effect of these roll angles (i.e., causing distortion in the presentations) can be eliminated on the ground through rectification using the roll angles measured during flight.

To reach the desired parallel path, the aircraft takes off and follows a path at a heading up to 18 degrees from the take-off to landing great circle, at least until the end of the fuel-burning period. The flight programmer then institutes a turn until the aircraft is along the desired parallel of latitude. After this turn, the flight path velocity will be lower than the cut-off value. The resulting roll angles necessary to overcome the Coriolis and centripetal forces then appear to be sufficiently small so as not to cause any exceptional trouble of an aerodynamic nature, or for the proper operation of the reconnaissance equipment.

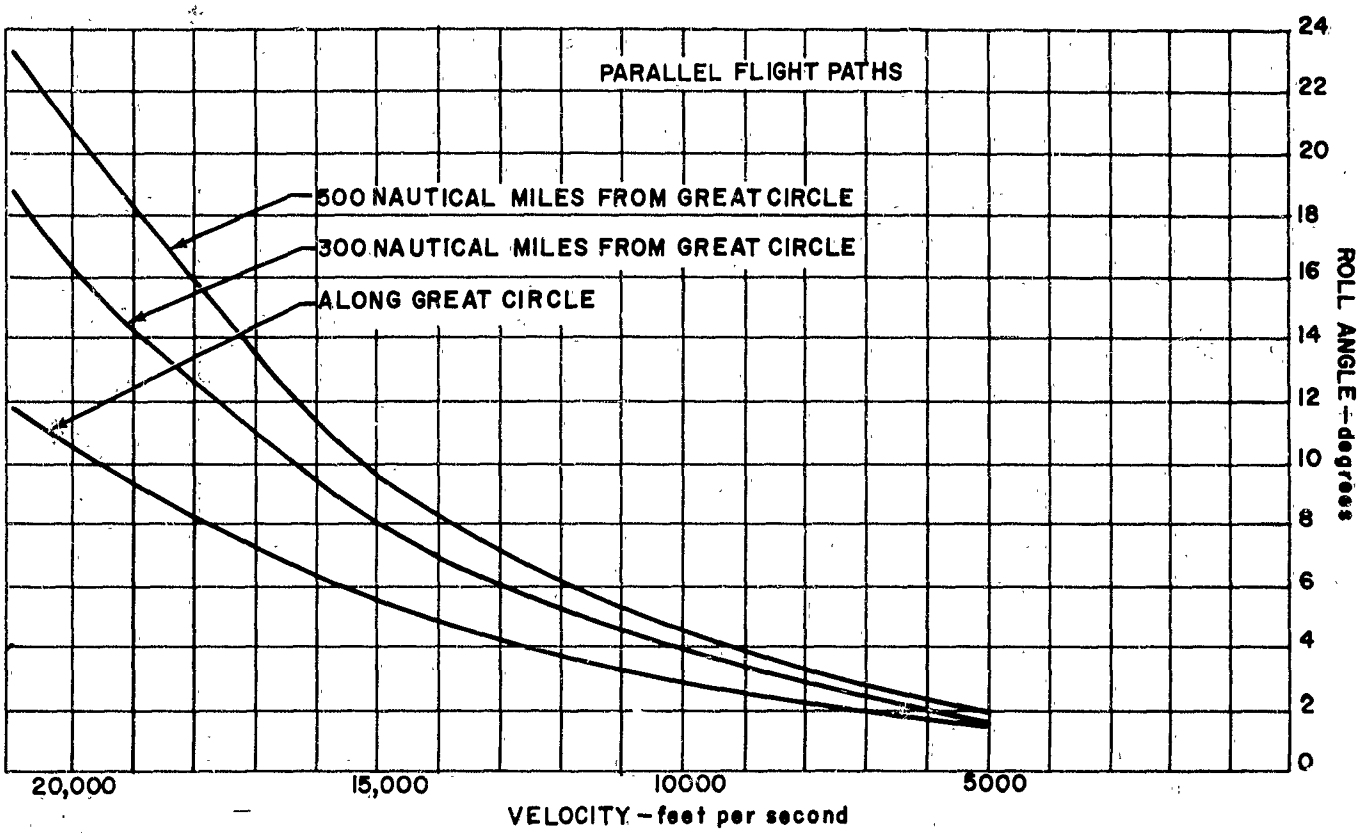


Figure 38. Roll Angle Necessary to Compensate for Coriolis Forces

b. Detailed Target Coverage

For the detailed target analysis mission, a navigational system is used that is based on line-of-sight angle to the target in terms of differential transverse latitude and longitude. This method closely approximates the great circle computation at low transverse latitudes, and especially so as the target is approached.

This flight path is also fixed with respect to the earth, so that a roll angle will be required to counteract the Coriolis and centripetal forces that exist. However, the forces for this type of flight will be less than those existing for a search mission.

Since the basic inertial reference unit is provided with an accuracy sufficient to allow navigation directly to a desired location within the coverage limits of the reconnaissance equipment, no intermediate checkpoints are required for navigational purposes.

c. Flight Programmer

The navigation system contains a flight programmer which is capable of automatically initiating the various functions and programs along the flight path. This programmer consists of a storage element and several function generators. The storage element is essentially a memory device containing all the pertinent information for the operation of the reconnaissance system. This information includes the pitch or longitudinal program for the flight path, the booster separation signal, the various locations for which detailed reconnaissance is required, operation control of the reconnaissance equipment, and other information. The function generators produce the various programs as functions of position, altitude, speed, acceleration, time, etc. Since most of these functions are basically derived from the inertial reference unit outputs, the function generators are merely follow-up devices working on these outputs. The follow-up devices in turn operate cams or read out of the storage element.

4. Secondary Navigation System

Although the primary navigation system is fully automatic, one of the main requisities of a reconnaissance system is the return to a friendly base of any information gathered. This, together with considerations of flight crew safety, necessitates the provision of a secondary navigation system to enable the pilot or navigator to satisfactorily navigate the vehicle to a landing base in case of a failure in the primary system.

The following four methods have been considered for secondary navigation:

a. A radio frequency system, such as Consol or Navarho.

b. Position-fix taking, using either optics or a search radar.

c. A secondary, less accurate inertial system.

d. Aerodynamic measurements, plus a compass.

The latter two methods appear to have the best possibilities; however, the final selection was not made.

5. Control System

The control system has the twofold purpose of providing the necessary moments for the stabilization of the vehicle and the forces to cause the vehicle to follow the desired flight path. The basic system will be automatic. However, provisions for the pilot to override and assume manual control will be included. A detailed analysis of the control system requires more information about the aerodynamic parameters than are available at this time. However, certain general comments can be made concerning the control system.

a. Ascent Stabilization

A preliminary analysis indicates that a two-axis gimballing of each of the booster combustion chambers about the vertical and transverse axes of the aircraft will provide satisfactory control. Yaw control is obtained by driving the two chambers together about the vertical axis. About the transverse axis the chambers are driven together for pitch control and differentially for roll control. It is planned to use the same chamber drives for both roll and pitch control. Therefore, limiters are necessary to prevent a loss of control in one direction due to a large error signal in the other direction.

This analysis indicates that in pitch, a 100-foot per second gust considered as a step input, would require an 8-degree per second maximum angular velocity of the combustion chamber. Rate gyros would be required to provide the required damping for compensation. When the bending characteristics of the airplane-booster configuration are considered, the location of these gyros near the end of the booster presents more satisfactory stability characteristics.

b. Glide Stabilization

Stabilization during glide will be provided by aerodynamic surfaces. The prime stability consideration during this phase is the limitation of the angular rates to prevent distortion in the outputs of the reconnaissance equipment. A preliminary analysis of this phase indicates that aerodynamic damping will have little effect; therefore, system damping will be provided by a rate gyro in this case also. In order to obtain an idea of the magnitude of the numbers involved, the maximum angular rates were evaluated for a gust velocity of 100 feet per second at 100,00C feet. These rates were found to be rather high. The rates can be reduced by a decrease in the natural frequency of the airplane or an increase in the natural frequency of the system.

c. Lateral Control

Lateral maneuvering of the aircraft could be accomplished either by the rudder or by rolling the aircraft. However, the relative ineffectiveness of aerodynamic surfaces at the speeds and altitudes at which the MX-2276 operates might possibly preclude the use of a rudder to compensate for large Coriolis forces or where high turn rates are required. Therefore, lateral maneuvering can be considered as being accomplished by rolling the aircraft with the rudder being used only to provide stabilization and for coordination during a programmed turn.

To perform a turn, the command signal is applied to the roll autopilot, thus causing an aircraft roll. The resolution rotation of the lift vector then generates the necessary side force. During the maneuvering, the rudder coordinates the turn by nulling the lateral acceleration in the plane of symmetry of the aircraft. During the period when only lateral control is necessary to insure the following of the desired flight path, the lateral errors are still sent to the roll autopilot, but the rudder is not used for coordination.

Since the aircraft roll system will be used to counteract side forces due to uncompensated Coriolis and centripetal forces, roll control during the glide will be rate control rather than angular control. During the ascent and landing where the effect of roll angles becomes critical, roll angular control with roll rate stabilization will be provided.

d. Pitch Control

The exact details of the longitudinal control program have not as yet been established. However, the following general comments are applicable.

The instrumentation of the inertial reference system is such that sufficient acceleration and angular information exists to compute normal acceleration. This would allow the programming of an acceleration-controlled or a zero-lift ascent trajectory.

The attitude angular data would also permit the programming of pitch angle as a function of time for both the ascent or glide phases of flight.

If an aerodynamic pressure sensing system is provided for secondary navigational purposes, this would also permit a somewhat less accurate means of longitudinal control.

An important consideration in selecting the optimum longitudinal control during the ascent is the angular limitations of the swiveling combustion chambers. Since the chambers are driven differentially for roll and pitch, suitable provision must be made for roll and pitch stabilization requirements as well as for pitch control.

e. Pilot Control

As has been previously indicated, an important advantage of a manned reconnaissance vehicle is the presence of a pilot who can navigate the aircraft to the landing base in the event of malfunctions in the automatic equipment. To provide the pilot with the necessary equipment to control the aircraft, a set of conventional controls and an integrated display of pertinent data for navigation and aircraft control are included as described in Reference 1. A map display, driven by the outputs of the navigation system, is also provided.

The manual and automatic systems are integrated so that the pilot overrides only the control or command input to the control system. The artificial damping, provided by the rate gryos, remains in operation to compensate for the relatively low aerodynamic damping characteristics of the airplane. In the manual mode of operation, this is equivalent to a power control system. Artificial feel, if required, can also be provided.

6. Integration With Reconnaissance System

The areas wherein the navigation system and the reconnaissance systems are correlated can be broadly grouped into three categories: stabilization, data correlation, and auxiliary computations and programming.

a. Stabilization

Stabilization, perhaps, is the most critical item affecting the proper operation of aerial reconnaissance systems. It is important with all three types of reconnaissance. However, it is most critical for the photographic installation. Motion of the reconnaissance equipment with respect to the ground during the period pictures are being taken results in a blurred image. An angular displacement of the equipment from a plane parallel to the ground also results in a displaced image. Providing the angles are known, the effects of displacement can be removed on the ground during interpretation of the data. However, the effect of motion during the time the reconnaissance data are being obtained cannot be removed at a later time and, therefore, the equipment must be stabilized against this motion. The results of studies reported in Reference 12 indicate that the allowable rates are in the order of one to four milliradians per second.

Two methods of stabilization are available — stabilizing the entire airplane and fixing the camera to the airframe, or mounting the cameras within a frame and stabilizing the frame. A detailed analysis of both methods is recommended. However, it appears that stabilization of the aircraft to very nearly the required limits will be possible. Since an aircraft stabilization system is required in any event, this system is preferred.

b. Data Correlation

To obtain the required intelligence from the reconnaissance data, it is necessary to have information concerning the location, altitude, and velocity of the aircraft as well as the time at the instant each individual piece of reconnaissance data is obtained. The correlation is accomplished by simultaneously recording the velocity and position outputs of the position computer, the angular indications of the platform, altitude, and time at the instant the reconnaissance data are taken.

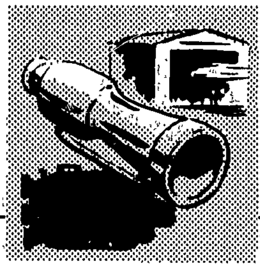
Two methods of correlating the information are under development. One consists of a central recording system with time as the correlating factor. With this method, only the time is recorded on the reconnaissance data. The other method consists of recording all information on the reconnaissance data. The latter system is preferred from a data reduction standpoint. Either system can be used in this configuration.

c. Auxiliary Computations and Programming

About the only computation, other than providing data for correlation and angular information for stabilization, that the navigation system is required to perform for the reconnaissance systems, is the computation of velocity/altitude (V/H) for image motion compensation of the cameras. This requires a resolution of the indicated velocity in the transverse latitude directions into the resultant velocity. The division by altitude is readily accomplished with the addition of a potentiometer on the output shaft of the altitude integrator. As was mentioned previously, to compute V/H within the allowable error, a radar altimeter is necessary to monitor the inertial vertical instrumentation.

Programming the control of the reconnaissance equipment will be accomplished by the flight programmer. This programmer will have the capability of generating functions or control signals as a function of time or position.

E. PROPULSION



1. General

Both stages of the proposed vehicle will be powered by liquid propellant rocket engines. The airplane will require 60,000 pounds thrust and the booster 300,000 pounds.

The selection of the power plants has been based on the policy of using existing developed units except where the cost or the performance of the over-all weapon system would be seriously penalized by so doing. The 300,000-pound engine requirement for the booster can take advantage of the double 150,000-pound LOX-JP-4 unit now being developed. The 60,000-pound engine for the airplane, however, will require a new engine using new very high energy propellants. The details of the propulsion system are presented in Reference 1.

2. Propellants

The LOX-JP-4 propellant combination has been established as the most efficient combination of propellants presently in use. The use of liquefied fluorine in the oxidizer provides a significant performance increase over that of the LOX-JP-4 combination alone. The development of the LOX-Fluorine-JP-4 propellants will be of value to many weapons and therefore cannot be charged to this weapon system alone. The logistics problem may be considered similarly. Since the exhaust products of this combination are toxic, it is considered advisable to use this propellant in the airplane only, where the exhaust products will be released at extreme altitudes. For the booster, the LOX-JP-4 combination is recommended.

The fuel used for all stages of the present proposal can logically be JP-4. It is the cheapest and the most plentiful fuel in the present Air Force supply program. There are other fuels such as ammonia and unsymmetrical dimethyl hydrazine which offer slight gains in performance over JP-4, but the gain may not be worth the increased complications, development cost, and more severe logistic problems.

The use of liquefied hydrogen offers further performance growth potential both with Lox for the take-off stage vehicle and with fluorine in other stages. The H_2 and O_2 combination has poor density characteristics which require large propellant tanks, but this may not be as critical as generally supposed. F_2 and H_2 can be used at the very high oxidizer to fuel ratios of 9-to-1 or even 16-to-1 to get reasonable density. Hydrogen, of course, is one of the most difficult liquefied gases to handle, but interest in its use is increasing rapidly because of its very high heat energy per pound. Hence, methods and equipment for handling and storage are expected to become available in a comparatively few years. Therefore, it is recommended that further consideration should be given to the use of fluorine and hydrogen as a possible later development of this program.

3. Airplane System

a. Engines

The rocket engines for the airplane have been studied by both North American Aviation Corporation, Rocketdyne Division (Reference 14) and the Bell Aircraft Corporation Rocket Section (Reference 15). These studies are included as a part of this investigation. It is expected that an over-all specific impulse of 334 seconds will be obtained at 55,000 feet

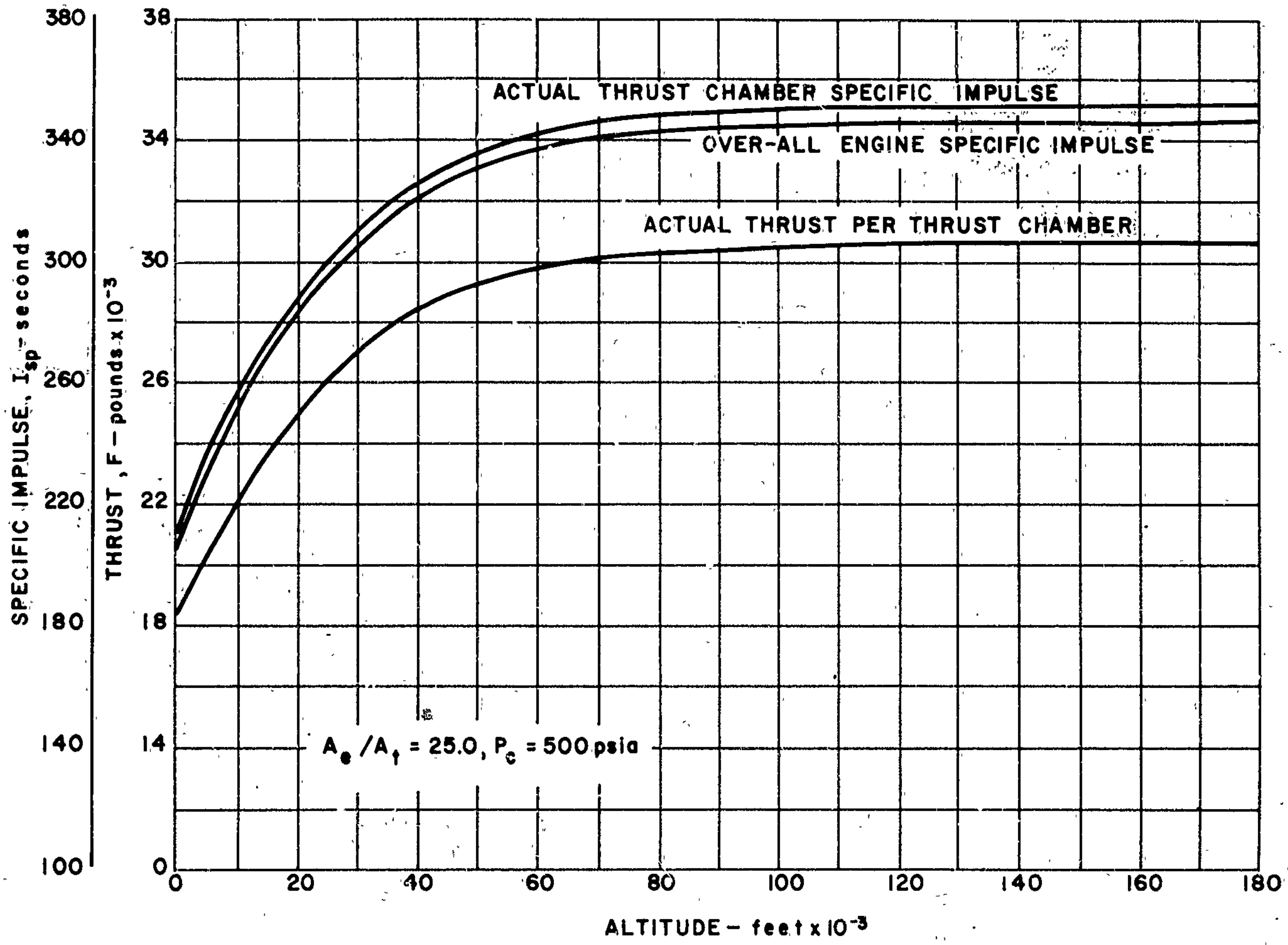


Figure 39. Variation of Performance with Altitude

altitude and 346 seconds at 155,000 feet altitude as shown in Figure 39.

b. Tanks

The propellant tanks are designed to be integral with the airframe structure. This design takes full advantage of the excess strength of a cylindrical pressure vessel that is not required for internal stresses and also takes advantage of the stiffness of the pressurized tanks during the critical phases of powered operation. The tanks are approximately 75 inches in diameter and have elliptical heads. Pilot safety in tank design is assured by adequate margins of safety against pressure stresses and by relief valves to prevent overpressurization. In addition, the tanks are designed to take all flight and landing loads without dependence on the tank pressure being maintained for the duration of the flight.

For the safety of the pilot in the event the engines do not start or an early malfunction shutdown occurs, the remaining propellants in the tanks must be jettisoned to reduce the landing weight to the normal design conditions. These jettison valves can be operated by a completely independent system from a cockpit control. Expulsion can be accomplished by the normal tank pressurization system, or an emergency pressure source such as the cockpit air supply system can be used.

The same jettison valve can be used on the ground prior to take-off to quickly unload the oxidizer in the event of a cancelled flight or some accident such as a propellant leak. For this purpose, the oxidizer loading vehicle would stand by with the connection to the jettison valve maintained until after tank pressurization just before actual take-off. If the oxidizer is jettisoned, the ground vehicle can safely contain the oxidizer and thus prevent contamination of the area.

For added safety to the pilots in the combat zone and during landing, the oxidizer tank and system will be completely emptied and purged immediately after power plant shutdown. The combination of natural evaporation of the small amount of remaining oxidizer and a helium purge will effectly eliminate any hazardous fumes. Extra capacity is included in the helium storage system for this purpose.

c. Pressurization

A chilled helium storage system has been selected for this system because of compatability, minimum complications, and weight. The system consists of a chilled high-pressure spherical storage tank, a start valve, two heat exchangers in the turbine pump exhaust duct, a first-stage regulator and two second-stage regulators, and a purging system. The helium is loaded into the storage sphere which has a safety valve to prevent overpressurization. The sphere is cooled by an oxidizer jacket in order to maintain low gas temperature. The pressure decay in the sphere is limited during operation, thus providing a sufficient drop across the second-stage regulator at shutdown to assure accuracy in controlling tank pressures.

In order to prevent cold gases from entering the first-stage regulator during operation, a first-stage heat exchanger is required to preheat the helium gas. This is done because of difficulty encountered in properly seating the regulator valve with extremely cold gases. The size of this heat exchanger is small because it operates on the high-pressure side of the regulator. The second-stage heat exchanger will operate on the low-pressure side of the first-stage regulator and will increase the temperature of the helium gas still further. Both heat exchangers will be of the one-pass, cross-flow type.

d. Auxiliary Power Units

For the proposed flight plan, auxiliary power cannot be obtained from the rocket engine turbine or from a ram air turbine for the entire duration of the flight. Similarly, a system that uses the energy available from aerodynamic heating cannot be used for the take-off or the final stages of the flight. To reduce complications and to improve reliability, it is very desirable to have a completely contained auxiliary power unit that does not require switching and is not dependent on other parts of the weapon system. For these reasons, the effort during this study was confined to an examination of a monopropellant turbine drive or a reciprocating drive which would be self-contained and independent systems.

The turbine drive unit is the simplest, and has the lightest dry weight, but the weight of propellant consumed is higher. This type of unit is now being developed and used, so that no general development difficulty is expected. However, the development of a specific size for this man-carrying which application will represent a considerable expenditure of time and effort.

The reciprocating type of unit is an outgrowth of conventional gasoline- and air-powered units which have been used for years for ground and moderate altitude airborne applications. However, for the extreme altitudes involved on this project, the combustion must be independent of an ambient air supply. The use of a monopropellant or bipropellant in a reciprocating engine has been successfully demonstrated by several organizations, and the basic development problems are reasonably well understood. It is expected that the development of a reciprocating engine for this specific project would be possible in the time available if an early start is made.

The reciprocating engine is able to use the propellants more efficiently than this size turbine, so that its original weight disadvantage is overcome for long durations. For the time established for this study, the total weight of the turbine system is greater. The difference is not significant for the present duration. However, in view of the fact that the growth factors of this weapons system would extend the range and the duration of auxiliary power plant operation, it seems advisable to start a detailed investigation of both types of units.

4. Booster System

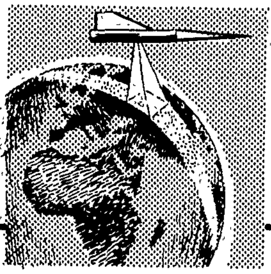
The propulsion system for the first stage of this weapon system has not required much detailed consideration during this study program because it is not considered a problem area. The motor has been selected as the 300,000-pound thrust assembly that has been developed for another project. Adequate information about this motor has not been available, but its specific impulse is expected to be 250 seconds and its weight about 3400 pounds, with a safety system.

The increased development required of this engine for a piloted application is indicated by the different specification requirements for unmanned and manned use. Proof of endurance and reliability for aircraft rocket engines requires at least 35 full-duration firings compared to 9 firings of missile engines. Proof of reliability and safety for the aircraft rocket engine is based on the performance of at least 100 starting and stopping tests, including proof of safety for all conditions of single malfunction occurrences during the entire duration of rocket engine operation, as compared to 10 tests for a remotely launched missile rocket engines. The development of the safety system and the additional testing required is expected to take at least two years of intensive work.

The propellant tanks for the booster will comprise most of the missile structure. They will be cylindrical with hemispherical heads and of approximately 120 inches diameter. The tank pressure is determined by the design requirements of the turbine pump. Recent information on these requirements has not been available, so a 35 psi pressure has been used for structural analysis purposes. The stiffening effects of internal pressure have been considered in determining the structure flight conditions when the engines are operating. However, to insure the safety of the pilot in the event of a loss in pressure just before take-off, the tank structure is made strong enough to take the static weight of the fully loaded vehicle without the benefit of pressurization.

A mixture ratio control system will be required for this stage as well as the final stage to reduce the amount of propellant mismatch. It is hoped that the system under development for these propellants and this system can be used with very little change; but the accuracy and weight of the systems being developed is not known.

The pressurization system for this stage will probably also be chilled helium since the design conditions are very similar to that of final stage. Here again, the lack of knowledge of the pressures required and other details prevents an accurate calculation of the size and weight of the pressurization system.



F. RECONNAISSANCE SYSTEM

The weapon system is required to provide three types of reconnaissance: (a) photographic coverage for search capable of resolving a surface dimension of 100 feet or smaller, and detail photographic coverage capable of identifying objects 20 feet on a side; (b) ferret surveillance from 30 mc to 40,000 mc; and (c) radar surveillance using high-resolution techniques. Equipment providing these capabilities has been included in the airplane; however, not all the equipment is carried on a single mission. The radar is carried on all missions for navigational purposes, and can be used for reconnaissance merely by recording the presentation. The ferret, photographic search, and photographic detail equipment must each be carried separately. The nature of this equipment and the number of flights required for each type of reconnaissance make this arrangement most practical. Each of the different types of equipment will be discussed separately.

1. Photographic System

a. General

The photo reconnaissance installation consists of the cameras, mount, windows, photographic control components, and data recording systems. Two types of installation will be used -- one each for the search and detail missions. The two camera assemblies (or arrays) will be interchangeably mounted in the stabilized mount. (The photographic system is discussed in detail in Reference 12.)

The search array is made up of six cameras. Two 27-inch effective focal length (EFL) cameras with 9 x 18-inch magazines are in a split vertical configuration with the 18-inch sides perpendicular to the line of flight. The axes of these cameras are at 17° to the aircraft nadir. Two 34-inch EFL cameras and two 42-inch EFL cameras each with 9 x 9-inch magazines are mounted with axes at 41° and 54°, respectively, on both sides of the nadir.

The detail array consists of three 72-inch EFL cameras with 18 x 18-inch magazines. One camera is mounted vertically and two are inclined at 13° on either side of the nadir. An alternate strip camera array using a 9 x 18-inch format may also be a possibility.

Both arrays are mounted in gimbal frames for primary stabilization in roll. The frames pivot on fore and aft brackets. The cameras are adjusted within their mounts for the angle of attack. Photographic windows of 12 inches maximum clear aperture are located to provide interchangeability of the two arrays. The windows for the 13° cameras are common to the two installations. A vertical window may replace those not used for the detail array. Lamination or air spacing, and the use of fused quartz or Vicor-covered glasses will minimize high-temperature deterioration of optical quality.

The photographic control components consist of velocity-altitude ratio (V/H) computation, exposure control, cycling of cameras, and pilot's manual operation equipment. V/H data will be provided by the aircraft navigation system and fine correction will be programmed from the flight profile. If necessary, automatic exposure control using transistorized amplifiers will be an integral part of each camera. Each camera operates on an autocycle basis. Image motion compensation and film transport techniques will be discussed below in more detail. Manual control will be minimized, consisting of starting cameras under some conditions, and such other functions as can best be done by the crew.

The data recording system proposed is the system in which all data regarding photographic conditions will be coded between frames for each negative. The data are automatically decoded and printed during ground processing.

All equipment discussed is of new but simple construction, with improvements based on past and current equipment characteristics.

b. Photographic Coverage

The search installation covers a continuous strip 90 to 70 nautical miles wide by 3000 nautical miles long with 60% overlap of adjacent exposures in line of flight. The detail installation covers a continuous strip 19 to 14.5 nautical miles wide by approximately 2000 nautical miles long with 60% overlap in line of flight. The detail mission might also be broken into sixty-one 18 x 36-nautical mile areas, for example.

c. Detection and Recognition

At a 150,000-foot altitude the search photography will permit detection of approximately 25-foot objects and recognition of 65-foot objects at a contrast level of 0.075 in the region below the aircraft. At the extreme distance, 40 nautical miles to either side, 70-foot objects will be detectable and 140-foot objects recognizable at a contrast level of 0.05. The larger numbers in the latter case may be conservative, based on the random probable orientation of the objects with relation to the scale parameters for high oblique photography.

The detail photography will permit detection of 9-foot objects and recognition of 25-foot objects at a contrast level of 0.075 anywhere within the area covered.

d. Film

Estimations are based on the use of Aero Super XX film in all cameras, with the exception of the four oblique cameras of the search array. A new Eastman emulsion with favorable contrast and latitude characteristics may be used in these two locations because of the more severe haze and attenuation conditions encountered with oblique photography. A thin base film of the new Cronar type will be used to decrease weight and allow the use of small-core film spools.

It must be emphasized that very recent advances in the photo emulsion industry show a strong possibility of greatly increasing photographic resolution and definition. The use of these materials in this system can significantly increase its capability from the 10 line per millimeter average resolution used in these calculations.

e. Cameras

The focal lengths chosen are based on an analysis of scale necessary for the resolution of ground detail and coverage. In the search array the focal length steps of 27 inches, 34 inches and 42 inches provide uniformity in scale for all cameras such that sidelapping photographs will match in the superimposed areas. All lenses will be modifications of the best presently existing lens designs. The search lenses will be f/8 with deep yellow filters for increasing contrast through haze. The 72-inch lenses will be of telephoto construction for space reasons, and will require the most effort of the four types in design modification. Their speed will be f/11 with minus blue filters. As currently conceived, three straight cameras using this lens with an 18 x 18 format are required for surveillance coverage.

Laboratory performance of all the lenses will be 30 lines per millimeter area weighted average resolution with no less than 20 lines per millimeter in the corners, when using Aero Super XX film.

Focal plane shutters will be used in all cameras. Exposures required will be in the region of 0.0025 second for both search and detail.

All magazines will be of the autocycling type. In all but the 27-inch 9 x 18 cameras the film will be in continuous motion at rates dictated* by image motion compensation values. Following exposure, shuttles in each magazine will move the next frame into position for the next exposure. Each magazine will be driven by its own variable frequency synchronous motor. Cycling rates will nominally be one cycle each 1.67 seconds for the search array and 1 per second for the detail coverage.

Ground speed data will be supplied by the aircraft navigation system for automatic correction of image motion compensation (IMC) and cycling rates during flight. The oblique cameras in the search array will incorporate variable IMC to accommodate relative image motion through the area covered.

The 18 x 18 magazines will have supply and take-up spools beside and below the focal plane in order to minimize the height of the cameras. Film capacities of the magazine are: 750 feet for each 9 x 9, 1500 feet for each 9 x 18, and 1350 feet for each 18 x 18 magazine. Thus a total of 6000 feet of 9 1/2-inch film can be carried on search missions and 4050 feet of 18 1/2-inch film on detail coverage missions.

f. Environmental Conditions

Pressurization to minimize focus shift due to high-altitude air density will be incorporated in the cameras. The inside camera pressure will be used by the vacuum platten for film flattening during exposure — an exhaust valve to outside pressure being opened in the platten as part of the autocycle operation. Sufficient humidity to prevent film brittleness will be included in the camera atmosphere.

g. Stabilization

Stabilization of roll, pitch, and yaw conditions to within one to four milliradians per second will be provided if necessary. A thorough study of aircraft characteristics will dictate development of stabilization equipment and the extent to which auxiliary equipment is necessary.

2. Ferret System

a. General

The objective of airborne ferret reconnaissance is to monitor all usable frequencies whose propagation characteristics make their detection by ground-based receivers unlikely. Under normal conditions, the lower limit is determined by the maximum usable frequency of the ionosphere; the upper limit is set by the highest frequency expected to be in use at the operational time of the equipment. Present knowledge places these limits at approximately 30 mc and 40 kmc, respectively. Information desirable from ferret reconnaissance is:

(1) Frequency of the transmission

(2) Location of the transmitter

(3) Type of modulation

(4) Modulation characteristics

(5) Polarization

(6) Antenna pattern and scan rate

These parameters are not necessarily of equal importance; their relative values are determined by the type of mission.

b. Recommended Ferret System

Two general types of ferret installation are recommended for this vehicle (Reference 1): a system with high probability of intercept will obtain the radar order of battle, while a system capable of detailed analysis will furnish technical intelligence. Common to both installations will be digital data recording equipment. Tables III and IV show the preliminary specifications for both systems.

It is not anticipated that a reconnaissance mission will be limited specifically to the use of one or the other of these types. Development of a complete ferret system should follow a building block concept so that the reconnaissance emphasis can be rapidly shifted as the need arises. The different building blocks, or subsystems, can be somewhat independent; the

TABLE III. PRELIMINARY SPECIFICATIONS OF PROPOSED FERRET RECONNAISSANCE SYSTEM TO OBTAIN RADAR ORDER OF BATTLE

Frequency Coverage		30 mc to 40 kmc
Frequency Accuracy		±10%
Directional Coverage		150 degrees on either side of the aircraft
Directional Accuracy		±1 degree
Modulation		Pulse only
Pulse Analysis		
Pulse Width	Range	0.1 to 10 microseconds
	Accuracy	±0.1 microsecond
Pulse Rate	Range	50 to 25,000
	Accuracy	±3 percent

TABLE IV. TENTATIVE SPECIFICATIONS OF PROPOSED FERRET RECONNAISSANCE SYSTEM TO OBTAIN TECHNICAL INTELLIGENCE

Frequency Coverage	30 mc to 40 kmc
Frequency Accuracy	±1.0 percent
Directional Coverage	150 degrees on either side of aircraft
Directional Accuracy	Unimportant
Polarization	Determine the type
Modulation	Determine whether pulse or CW
Modulation Characteristics	
A. Pulse Width	±0.1 microsecond for pulse width greater than 0.2 microsecond ±0.02 microsecond for pulse width less than 0.2 microsecond
B. Interval between pulses	±0.1 percent
C. Frequency-Modulated CW	Determine frequency deviation and modulating wave form
D. Amplitude-Modulated CW	Determine modulating waveform
Antenna Pattern and Scan Rate	Determine duration of illumination within ±10% and relative amplitude of signal within 10% of peak value

requirement for similarity is that all outputs be compatible with the common recording unit. Of course, the total capacity of any recording device is fixed, but in this way it can be used for information which is of maximum value at the time. Probability of intercept, frequency coverage, the number of parameters analyzed, and accuracy of analysis can be emphasized or de-emphasized as the situation requires. The total amount of information obtained will remain approximately constant.

It may be that detailed modulation analysis can be most economically performed using a manned, rather than a completely automatic system. An operator has the ability to recognize and discard transmissions previously analyzed. He can concentrate his attention upon new and unusual intercepts. Provisions are therefore made for inclusion of a presentation unit whereby an operator may select and record significant signals.

Figure 40 is a tentative block diagram of an over-all system.

c. Radome

Heat resistance problems similar to those existing for the radar antenna radome also exist for the ferret radome. The relatively wide beam ferret antennas, and the relatively high-power received signals allow the transmission characteristics for ferreting to be considerably less severe, however. While low loss tangent and minimum beam bending are always desirable, they can be compromised to

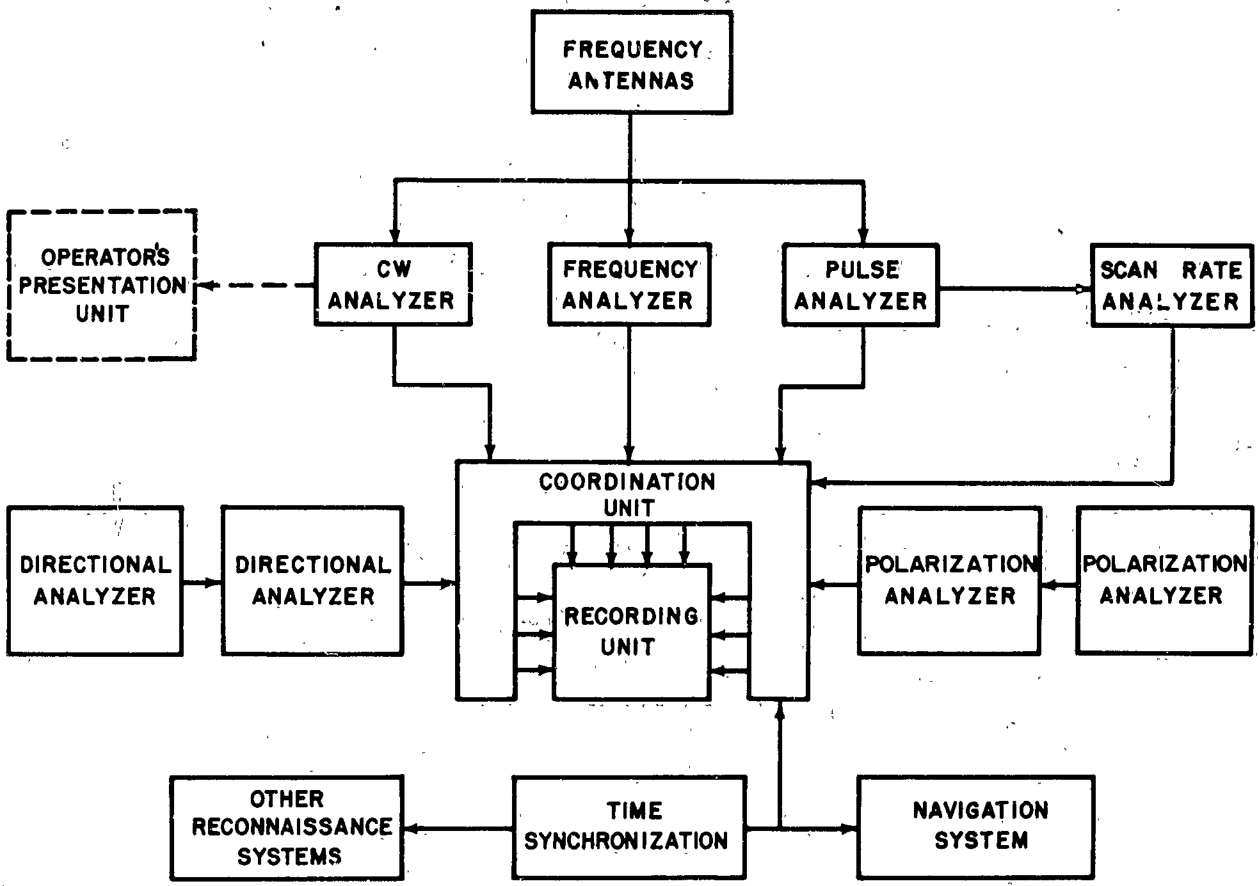


Figure 40. Proposed Ferret Reconnaissance System

a certain extent for heat resistance, light weight, and good structural characteristics.

The suggested ferret installation requires a section of the fuselage to be transparent to the radio-frequency portion of the spectrum. Since ferreting antennas are quite broadband, the use of a quarter-wavelength or half-wavelength radome is unlikely. A single radome which is electrically thin to all wavelengths in the 30 mc to 40 kmc band would be ideal; however, the extreme thinness of such a radome would be structurally unsatisfactory. A suggested solution is to use a radome thin to the longer wavelengths and to furnish small windows for the high frequency antennas. Experimental investigation is recommended to determine optimum window thickness from both a structural and an electrical point of view. Both very thin windows, and windows of half-wavelength thickness at the center of the frequency band should be investigated.

3. Radar

A K_u-band side-looking radar has been selected for the reconnaissance system because the high antenna gain and excellent resolution available from this type of equipment best satisfy the radar requirements for this weapon system. The radar chosen for the strategic system (Reference 13) is applicable to the reconnaissance system as well. Other radar types considered were deemed unacceptable, primarily because of the high speed and altitude of both the strategic and the reconnaissance systems. Table V summarizes the proposed radar which is discussed in detail in Reference 1. However, certain features concerning the use of this equipment require further explanation.

TABLE V. TENTATIVE SPECIFICATIONS OF A PROPOSED SIDE-LOOKING RADAR (FIXED ANTENNAS, SIMULTANEOUS LOBING)

Frequency		16 kmc	
Peak Power		120 kw (each side)	
Pulse Repetition Frequency		1200 pps	
Pulse Width		0.4 microsecond	
Receiver Bandwidth		3.0 mc	
Antenna Length		22 feet	
Antenna Width		8 inches	
Ground Coverage		10 to 50 miles each side	
Pulse Packet Size		Near Range	Far Range
	Azimuth	500 ft	1000 ft
	Range	535 ft	225 ft
Indicator Resolution		2000 spots for 50 miles	
Weight Estimate			
Antennas		320 pounds	
Radar		250 pounds	
Size Estimates			
Antennas		22 ft x 1 ft x 8 in. each	
Radar		6 cubic feet	
Power Requirements		2500 watts	

a. Data Presentation

In order to make the radar presentation an accurate representation of the terrain scanned, provisions must be made to compensate for aircraft pitch, roll, and yaw. Three methods of doing this are available:

(1) The antenna can be space-stabilized so that the radiation pattern covers a constant range on the ground and is, at all times, perpendicular to both the ground and the aircraft's ground velocity vector.

(2) The display can be distorted to compensate for all changes in aircraft attitude; this is known as data stabilization.

(3) A combination of the previous two methods can be used.

For this airplane, the second method (2) has been selected for two reasons:

(a) The side-looking antenna is very large and cumbersome; hence, antenna stabilization would require a considerable increase in weight and space. In addition, the difficulty of holding dimensional tolerances would be increased.

(b) Attitude changes in the airplane are sufficiently small that changes in radiation intensity levels at the ground will be negligible.

Figure 41 shows the antenna radiation pattern for level flight and Figure 42 shows the pattern when the aircraft has yaw, pitch, and roll deviations from the ground velocity vector. The displacement between the two patterns is the correction which must be computed in the data stabilization computer. It is suggested that this computer should be an analog device in which the beam intercept is computed in a rectangular coordinate system. The computed ground trace is then modified in accordance with aircraft attitude information from the navigation system before it is presented on the cathode ray tube.

In addition to the photographic process previously considered for the display, another process incorporating a phosphor belt appears to have considerable possibilities. This process also requires additional development.

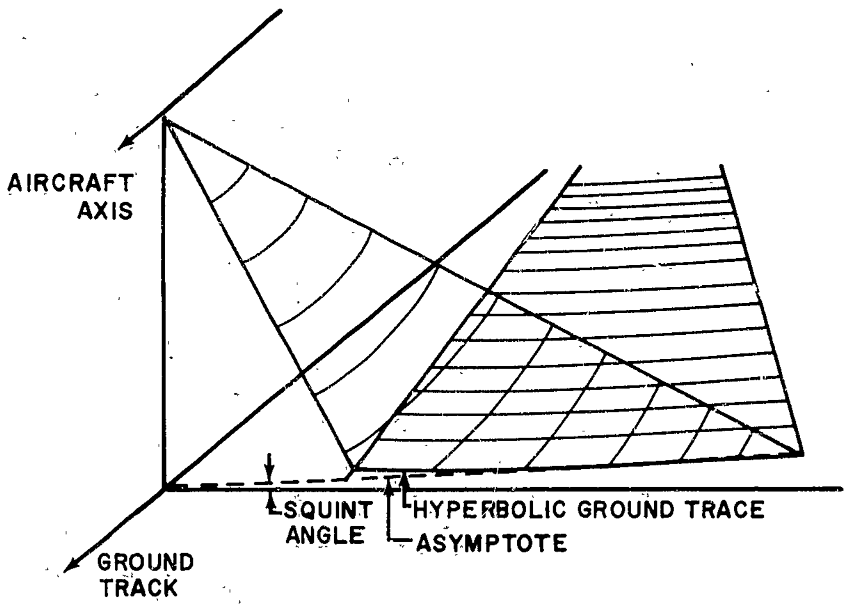


Figure 41. Ground Intersection of Radiation Beam with Airplane in Level Flight

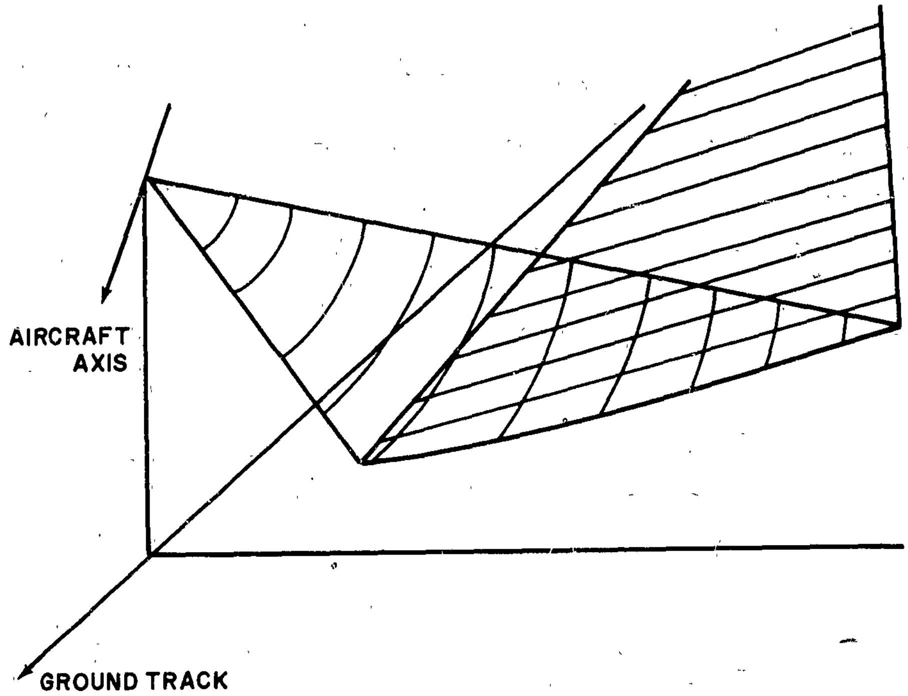


Figure 42. Ground Intersection with Yaw, Pitch, and Roll

b. Accuracy

Table VI shows the estimated CEP of a position fix using the assumptions listed in Reference 1. The magnitude of these errors is relatively large with respect to the estimated errors of the inertial navigation system (Reference 10). Therefore, it was decided not to use the radar information to correct the inertial system. Instead, the radar is used for reference, reconnaissance, and secondary navigation. Thus the inertial system provides a nonemanating guidance system. However, the radar is available for emergency navigation.

c. Altitude Measurements

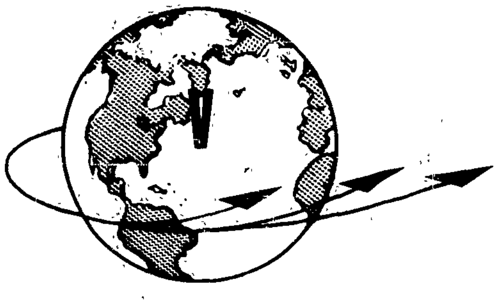
Analysis indicates that an altitude indication of the accuracy obtainable from a radar altimeter will be necessary for use in the navigation and photographic systems. When the search radar is operating, this information can be obtained from a vertical lobe of the side-looking radar. If the search radar is not operating, a portion of the magnetron output can be switched from the side antennas to a horn pointed toward the earth. The leading edge of the return energy is then tracked as in a normal operation and the altitude computed. If the radar system is not in the aircraft, a radar altimeter can be installed to supply this information.

d. Natural Ionization Effects

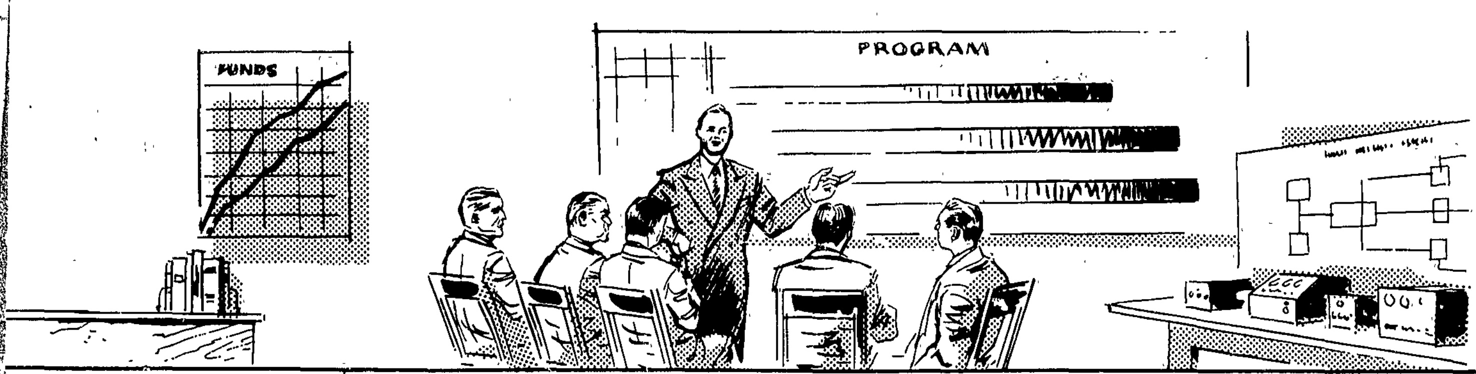
It has been shown (Reference 10), that the bending and absorption effects of the slightly ionized D layer below the aircraft are negligible. Additional analysis has shown that ionization of meteor trails will also have a small effect. This is the result of the fact that the narrow antenna beamwidth illuminates such a small area of the meteor layer. Another natural phenomena, aurora borealis, is expected to have negligible effect at the frequencies and altitude utilized by this system.

TABLE VI. RADAR FIX ERRORS
(Assumed Target of 100 Feet Square)

A. Parallel to Ground Velocity		
Spot Size (2000 spots per 40 miles)		120 ft
Altitude Computer		224 ft
Pulse Width		149 ft
Hyperbolic Sweep (0.1% of maximum range)		304 ft
Target Size		100 ft
Data Stabilization		300 ft
Matching Accuracy		200 ft
	Over-all	565 ft
B. Perpendicular to Ground Velocity		
Spot Size		120 ft
Target Size		100 ft
Beamwidth (1/3 of total to half-power points)		222 ft
Random Fluctuations in Film Drive		200 ft
Data Stabilization		240 ft
Matching Accuracy		200 ft
Squint		200 ft
	Over-all	530 ft
Total CEP		775 ft



V DEVELOPMENT PROGRAM



A. GENERAL

The program for the design, development, and flight test of this piloted special reconnaissance weapon system is shown in Figure 43. The objective of the program is to initiate the flight test of a prototype weapon system at the earliest possible time. The reconnaissance capability of the system would include high-order photographic, medium-order photographic, or ferret on separate flights. High-resolution radar coverage is provided on all flights to be used as required by weather conditions or specific mission objectives.

The development program provides for a one-year preliminary design period followed by a 1 1/2-year Phase I. Phase I includes the fabrication of a mock-up and the preparation of a detail specification for the weapon system. It should be noted that the development and test of the major subsystems such as inertial navigation unit; autopilot systems; photographic, ferret, and radar reconnaissance equipment; power plant; and auxiliary power supply unit would be initiated at the completion of the preliminary design phase. The investigation of specialized ground support and operational equipment required for the system would likewise begin.

The accelerated schedule would require the initiation of detail design of the airplane and the booster prior to completion of the Phase I program.

A flight test airplane would be completed 4 1/2 years after start of the contract. This airplane would use a B-36 as an airlaunch platform to evaluate the low-speed glide and landing characteristics. Initial airplane power plant tests could also be accomplished and reconconaissance equipment functional tests and performance evaluations made. Initial flight crew training could also be conducted in this manner.

The first flight of the airplane-booster combination would take place six years after program authorization.

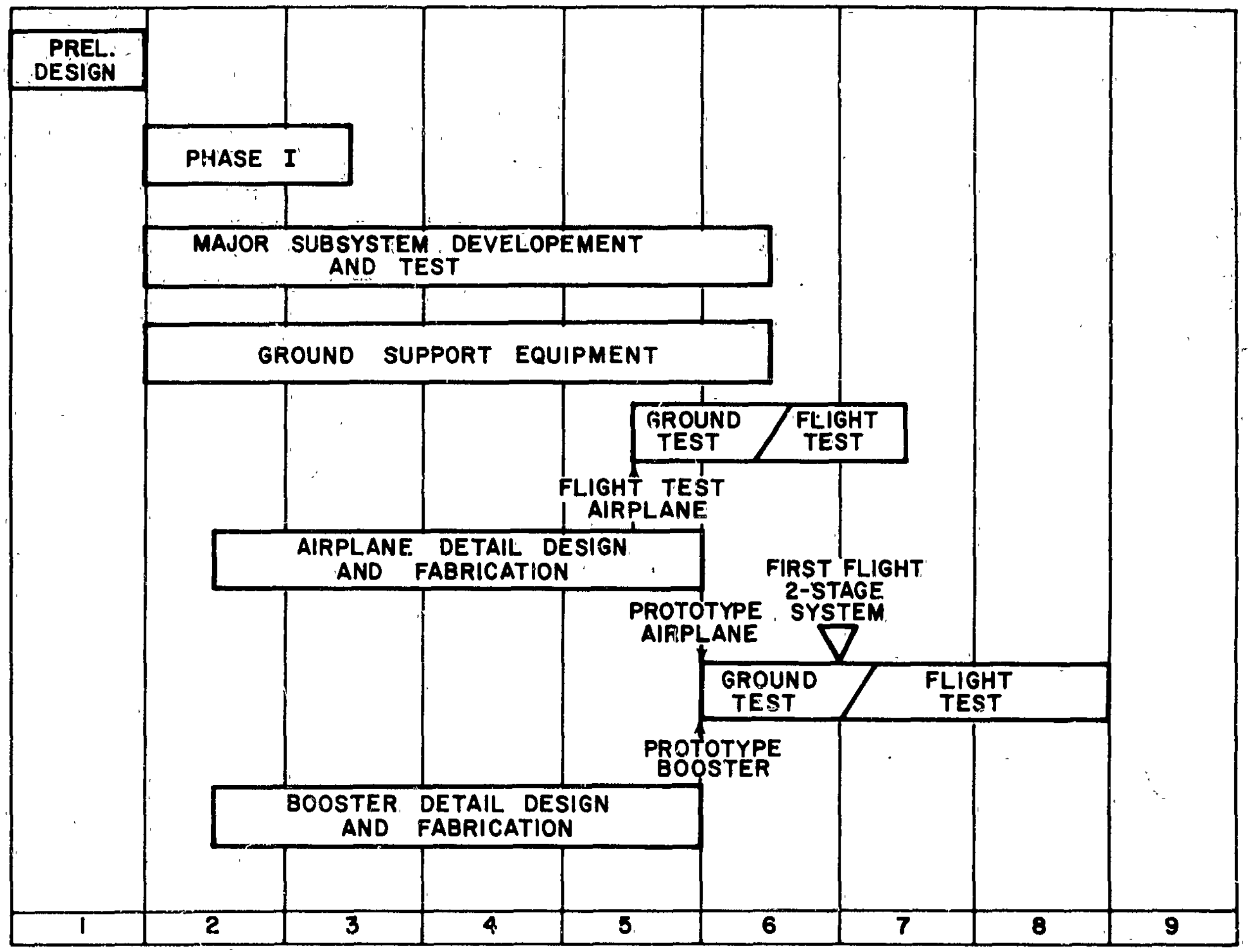


Figure 43. Reconnaissance Aircraft Weapon System Development Program

B. FIRST YEAR - PRELIMINARY DESIGN PHASE

The intent of the first year of the program is to obtain experimental test data in the fields of aerodynamics, structures, and human factors in order to establish the practicability of certain design approaches. In addition, the military requirements can be clarified with respect to range and special reconnaissance equipment objectives. Generally, the work would include the following three areas:

1. PRELIMINARY AIRFRAME DESIGN AND LAYOUT

This area should include airplane configuration considerations, booster arrangement, and structural layout of major airframe items. Major subsystem requirements would be refined, and preliminary equipment installation layout would be accomplished.

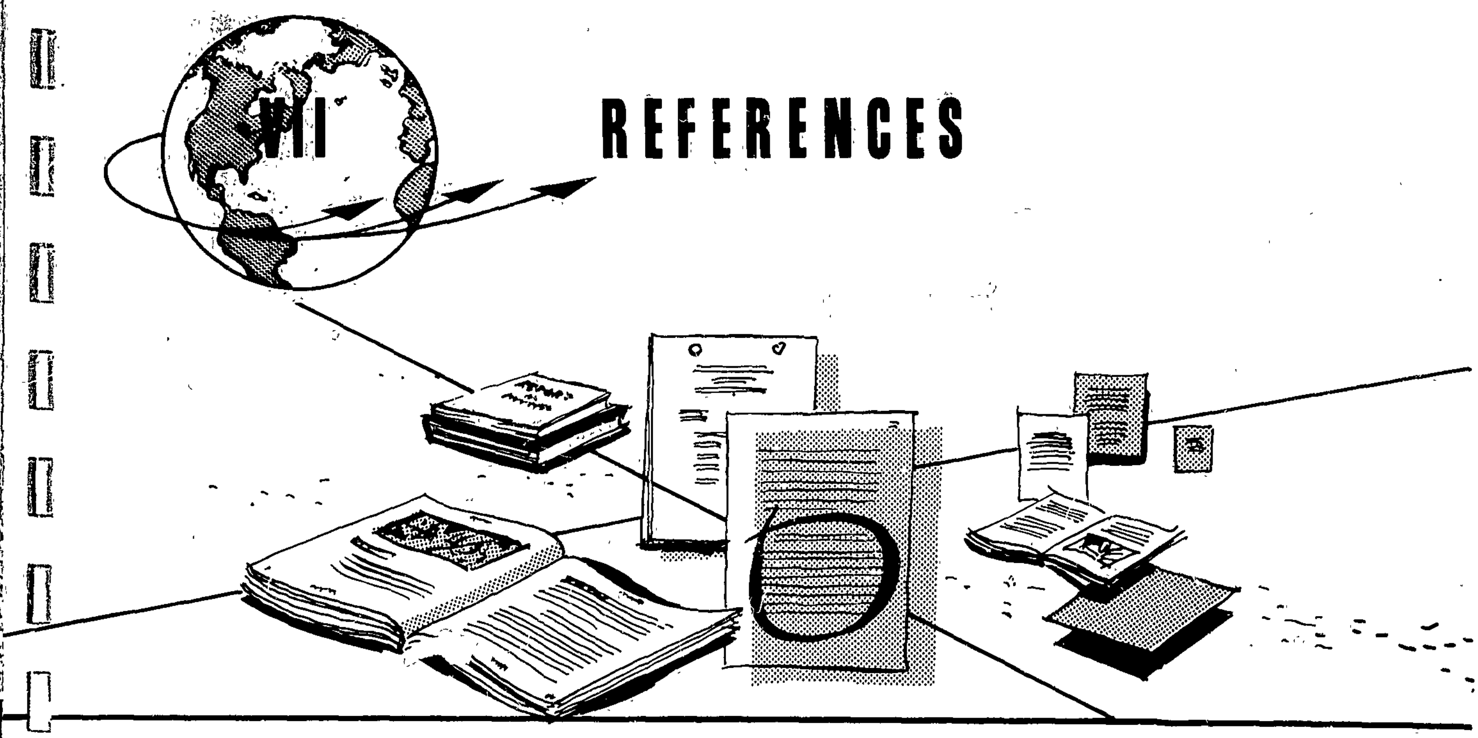
2. WEAPON SYSTEM ANALYSES

A definitive analysis will be made of weapon system ground-to-ground range requirements considering operational bases, target coverage, and intelligence data needed. Logistic and ground support aspects, crew workload, and effectiveness analyses and emergency precedures would also be included.

3. EXPERIMENTAL PROGRAMS

The specific aim of the experimental program during this period is to obtain test data to verify important system parameters such as aerodynamic heating values, control characteristics, and structural heat protection. Work concerning human factors would be included, using simulation to determine crew capabilities and the required instrumentation and display systems.

VII REFERENCES



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