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ADVANCED STRATEGIC WEAPON SYSTEM

BELL Aircraft CORPORATION

CONTRACT NO. AF33(616)-2419

REPORT NO. D143-945-018

29 APRIL 1955

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FOREWORD

During the period 1 April 1954 to 2 May 1955 the Bell Aircraft Corporation conducted a study program for the New Development Office, Bombardment Aircraft Branch, WADC, in accordance with USAF Contract No. AF33(616)-2419 RDO No. R441-47. The objective of this study was to investigate the possible design and development problems associated with flight in the speed and altitude regimes of the weapon system outlined in Bell Aircraft Report D143-945-010. The results of this study will provide the firm technical foundations necessary for planning future programs, funds, and facilities.

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

D143-945-012	Aerodynamics
D143-945-013	Structures
D143-945-014	Preliminary Global System Study
D143-945-015	Radar
D143-945-016	Navigation and Control
D143-945-017	Propulsion
D143-945-018	Final Summary Report

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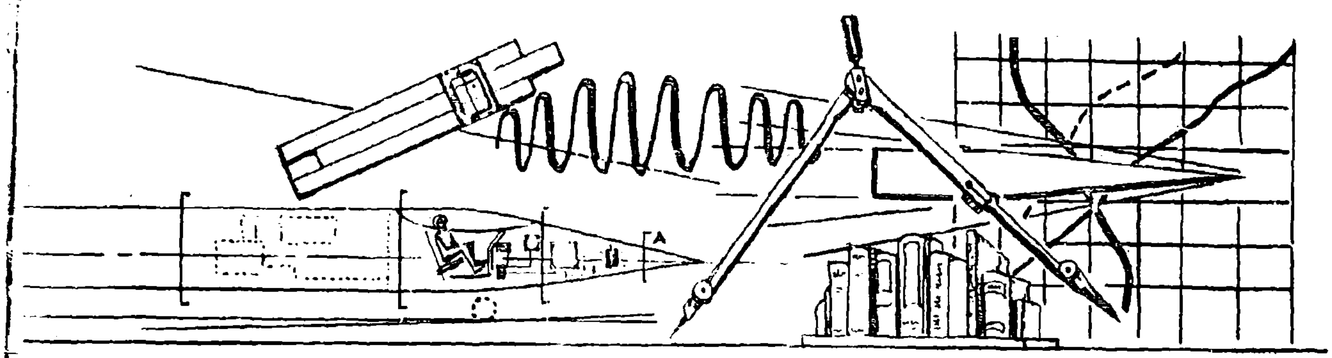
I ABSTRACT



The major design and development problems associated with flight in the altitude and speed regimes of the MX-2276 weapon system have been investigated to the extent that the present state-of-the-art permits. The environment necessary to maintain a crew in sufficient comfort for proper performance of the crew functions as defined in this report has been determined. Methods utilizing existing equipment with minor modifications are advanced to provide this required environment. A qualitative comparison of the manned system with an unmanned system shows the superiority of the former from an over-all weapon system viewpoint.

The aerodynamic parameters used in the initial report (Reference 1) were re-evaluated using data obtained during this study period. As a result, it now appears that the desired range can be obtained by means of a glide flight path initiated at lower altitudes than those previously estimated to be required. An analysis of the flight mechanics indicates that the direction of earth rotation has a major effect upon the weapon system range and must be considered in operational planning. A new evaluation of the heat fluxes and equilibrium skin temperatures encountered by the bomber, shows these values to be larger than previously estimated. Methods for the determination of these parameters for the nose and leading edge regions have been ascertained. Initial results indicate equilibrium temperatures as high as 5000°F may be encountered in these localized areas, if no cooling is provided. From a study of the use of transpiration cooling, it was determined that relatively large rates of air coolant expenditure are required to cool the first foot of the wing chord to 1600°F. The use of water as a coolant may result in lower coolant expenditure rates. The effects of shock boundary layer interaction upon various aerodynamic phenomena have been estimated for flow about a two-dimensional wedge. The effects consist chiefly of an increase in pressure, with a consequent increase in equilibrium skin temperature on the upper surface. The increase in pressure also provides an increase in cruise altitude. The over-all results remain to be evaluated.

Preliminary criteria and loads data for use in structural analysis of the vehicles have been established. A survey of structural, in-

sulating, and cooling materials was made and the results compiled and summarized. These data have been utilized in several typical wall configurations for use on the bomber. The most promising configuration consists of a light outside skin structure separated from the inside skin structure by a layer of insulation. The outside skin, which carries only airloads, is allowed to heat to equilibrium skin temperature while the primary inside structure is kept at the desired temperatures by insulation and cooling. Tests of fabrication, strength, thermal warpage, and thermal cooling have been made on this type of structure. The results indicate that it will be satisfactory; however, several development problems were indicated.

As a result of survey of propellant conbinations and rocket hardware, the use of oxygen and JP-4 in the first two stages and oxygen-fluorine and JP-4 in the third stage are recommended. The use of the advanced propellants in the third stage results in appreciable weight saving, requires development of a smaller engine only, and eliminates the problem of the toxic exhaust products at low altitude.

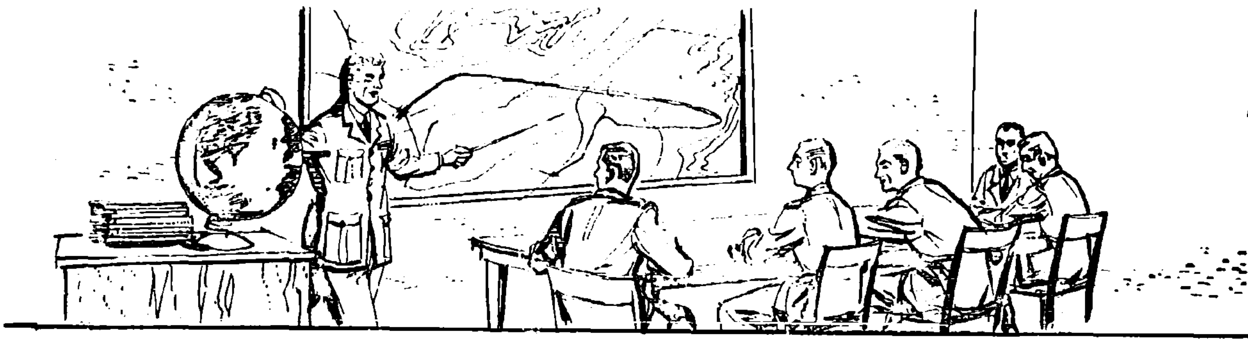
A radar-monitored inertial navigation system in the bomber will provide the necessary accuracy. A similar but less precise inertial guidance system in the bomb, used in conjunction with the bomber system, will provide an accuracy of approximately 1500 feet CPE in a 300-nautical-mile bomb flight. A K_u band side-looking simultaneous-lobing radar will provide the resolution required for the navigation system.

A design investigation is being sponsored by Bell Aircraft Corporation in conjunction with the study contract. Several interesting design features such as tandem staging and the use of circular bodies are presently being evaluated.

A global weapon can be obtained with increased take-off weight, and the advantages and disadvantages of several possible paths have been determined and summarized.

It is concluded that the initiation of design and fabrication of the vehicles comprising the MX-2276 weapon system is feasible at this time, provided certain research and development test programs are initiated in the very near future.

II INTRODUCTION



Evaluation of the territory of our present potential enemy shows a large dispersion of targets that must be destroyed in order to obtain a decisive victory. To reach all of these targets it is necessary to traverse from 2000 to 8000 miles of enemy territory. Improvements in modern defense systems consisting of tight radar warning nets, communications, intercept control systems, improved interceptors, and guided missiles will tend to make this requirement for deep penetration costly in terms of men and material. However, the effectiveness of such defenses can be reduced to a negligible level by a strategic system capable of operating at extremely high altitudes and high speeds.

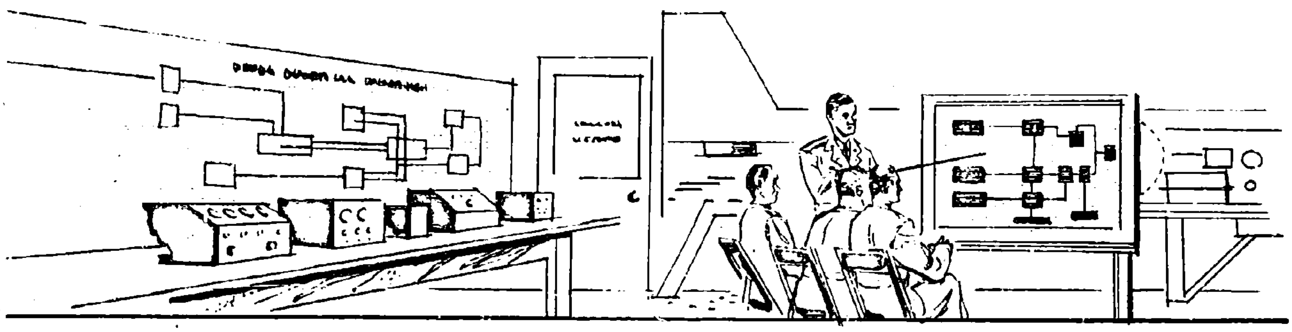
During recent years there has been a pronounced trend toward completely automatic missile systems for many purposes, including future strategic warfare. In order to fly at the speeds and altitudes required, and at the same time achieve the necessary accuracy, these weapon systems have become very complex. This increased complexity results in reduced reliability. Furthermore, many of the long-range guidance systems presently under development require knowledge of the exact geographic location of the target prior to launch in order to achieve their objectives. Since this information is not available for all targets, the required accuracy cannot be obtained even with the most complex guidance systems. Long-range missile systems are also inherently inflexible, i.e., after launching, the missile is, in general, committed to a specific target location and cannot deviate to alternate targets.

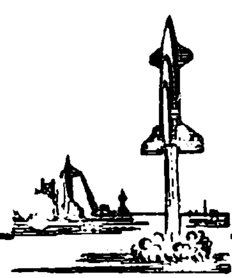
As a result of these considerations, Bell Aircraft Corporation proposed a rocket-boosted, manned, strategic weapon system which combines the best features of aircraft and missiles, to provide the high levels of speed and altitude for invulnerability, the accuracy required for precision bombing, and adaptability to reconnaissance functions. The initial concept was a three-stage vehicle consisting of a first stage, manned, recoverable booster airplane; a second stage expendable booster; a third stage, manned, rocket-boosted glide airplane. The last stage contained an air-to-surface inertially guided bomb which was launched as the third stage approached the target area. The philosophy behind this system stems from the fact that no mechanical means has yet been discovered or devised which can substitute for the mental powers of a human being. Many of the limitations of automatic flight previously described are eliminated by the inclusion of a human mind with its capability of observing a large variety of different types of data, utilizing these data to arrive at a decision and initiating a variety of actions to implement these decisions. This weapon system is visualized as one of the essential weapons in the composite group of strategic systems necessary for the successful prosecution of future wars. It is not anticipated that any single system will be capable of fulfilling all requirements of future strategic operations.

On 1 April 1954 a one-year study contract was initiated for the New Development Office, Bombardment Aircraft Branch, WADC, to investigate the possible design and development problems associated with flight in the altitude and speed regimes of this weapon system. The major effort under this contract was to be devoted to crew requirements and functions, and aerodynamic investigations including heat transfer, guidance and navigation, radar, structural problems, and improved propellants. This report summarizes the results of the studies performed under this contract. These studies are reported in detail in the six technical reports listed in the Foreword. A development program for this advanced strategic weapon system is described in Bell Aircraft Report D143-945-019.

In addition to the work performed under the above Air Force contract a Bell Aircraft Corporation-sponsored preliminary design and layout program was initiated. The initial results are also summarized herein.

III WEAPON SYSTEM DESCRIPTION





A. INITIAL CONFIGURATION

The configuration used for this study program is the configuration proposed in Reference 1. For convenience, a description of this configuration is presented here. The system is composed of (1) a Stage I, manned, rocket-powered booster airplane, (2) a Stage II, expendable, rocket-powered booster, (3) a Stage III, manned, rocket-boosted glide aircraft (bomb carrier-director), (4) an inertially guided bomb carrying a 2800-pound special warhead, (5) a pilot in Stage I to aid in recovery of the aircraft, (6) a pilot in Stage III to monitor the automatic navigation and control equipment, operate and monitor reconnaissance equipment, identify the target, correct the bomb guidance system, and evaluate damage to the target, (7) an automatic navigation and control system for Stages I and III, (8) mapping and photographic equipment for reconnaissance, (9) other components of equipment necessary to interrelate the guided bomb to the carrier-director, and (10) equipment to provide a satisfactory artificial environment to integrate the human being into the weapon system

Figure 1 is a three-view of this configuration. As shown, the carrier of this strategic weapon system is a composite rocket-powered airplane, composed of three stages, which takes off vertically, the first two stages serving as boosters for Stage III, the actual carrier-director.

Stage I (Figure 2), the main booster, is a manned aircraft of canard configuration, which supplies a total of 900,000 pounds of thrust at take-off from four 150,000-pound and four 75,000-pound regeneratively cooled rocket motors. The propellants, hydrazine-ammonia mixture as the fuel and liquid oxygen as the oxidizer, are delivered to the thrust chambers at approximately 1100 psia by means of turbine pumps driven by gas generators. Propellant storage is provided in Stage I to serve Stage II up to the time of separation. Automatic navigation and control which is provided in Stage I is monitored by the pilot who eventually takes over and lands the aircraft. Conventional tricycle landing gear and landing flaps are provided for this stage. Two 75,000-pound thrust chambers are gimbal-mounted and used to control the aircraft during ascent. Conventional aerodynamic surfaces are used for control of Stage I after burnout and separation.

Stage II (Figure 1) is an expendable booster which supplies a total thrust of 300,000 pounds, from four 50,000-pound and four 25,000-pound thrust, regeneratively cooled rocket motors using the same propellant combination as Stage I. Two 25,000-pound thrust chambers are gimbal-mounted and used for control during the burning period following separation of Stage I.

Stage III (Figure 3), the bomber, is a manned aircraft, of modified delta configuration that performs the strategic mission. Automatic navigation and control equipment, together with the necessary equipment to allow the pilot to monitor, supply corrections, or override the automatic control, are included in this stage. In addition, equipment for bomb guidance, reconnaissance, and artificial environment and emergency provisions for the pilot are included. Boost thrust is supplied by two fixed 25,000-pound thrust rocket motors. The propellants, hydrazine as the fuel and nitric acid as the oxidizer, are supplied to the thrust chambers by a gas generator-driven turbine pump. Conventional aerodynamic surfaces are incorporated for control, and a retractable tandem two-skid landing gear is provided for landing at the recovery site.

An inertially guided bomb, weighing approximately 4200 pounds and carrying a 2800-pound special warhead, is carried in the aft section of Stage III between the two rocket motors, and is ejected rearward.

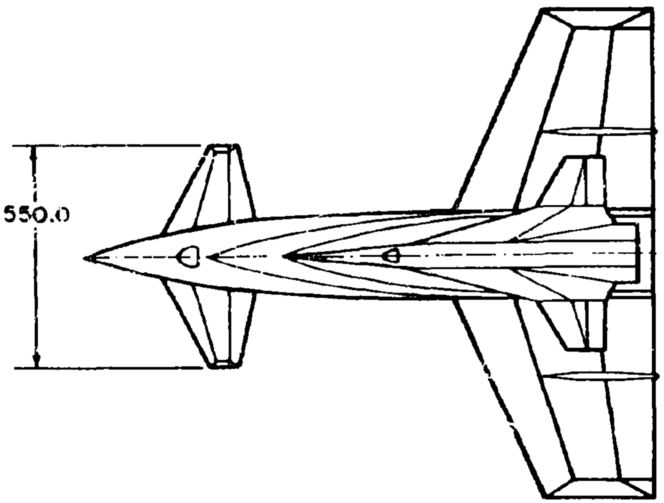
A summary of the weight estimation for this configuration is given in the following table:

Item	Weight, lb
Stage I	
Dry Weight	165,000
Propellants	486,000
Total	651,000
Stage II	
Dry Weight	30,000
Propellants	125,000
Total	155,000
Stage III	
Dry Weight	14,600
Propellants	26,200
Payload (bomb carrying 2800-pound warhead)	4,200
Total	45,000
Gross Weight (Bombing Mission)	851,000

A typical flight profile for a bombing mission is shown in Figure 4. Indicated on this figure are the velocities and altitudes attainable with this advanced strategic weapon.

DATA

TAKE-OFF GROSS WEIGHT	851,000 lb
TOTAL PROPELLANT	637,190 lb
DESIGN THRUST AT TAKE-OFF	1,150,000 lb
STAGE I	
WEIGHT SUMMARY	
GROSS WEIGHT	651,000 lb
STAGE II	
WEIGHT SUMMARY	
GROSS WEIGHT	155,000 lb
STAGE III	
WEIGHT SUMMARY	
GROSS WEIGHT WITH BOMB	45,000 lb



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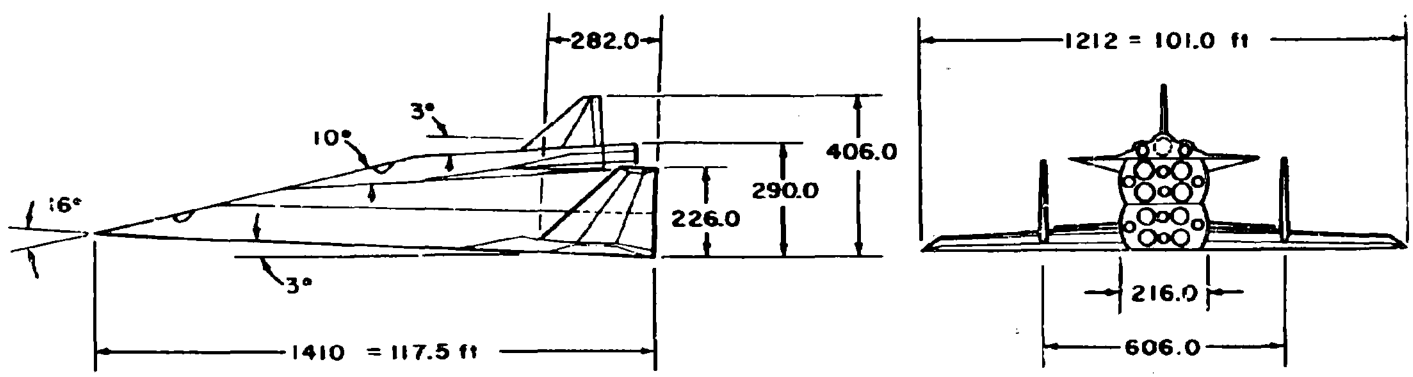


Figure 1. Initial Configuration of MX-2276

DATA

WEIGHT SUMMARY

GROSS WEIGHT		651,000 lb
PROPELLANTS	$N_2H_4 + NH_2$	262,500 lb
	O_2	223,500 lb
EMPTY WEIGHT		165,000 lb
POWER PLANT		30,000 lb
EQUIPMENT		5,000 lb
STRUCTURE		130,000 lb

SURFACES

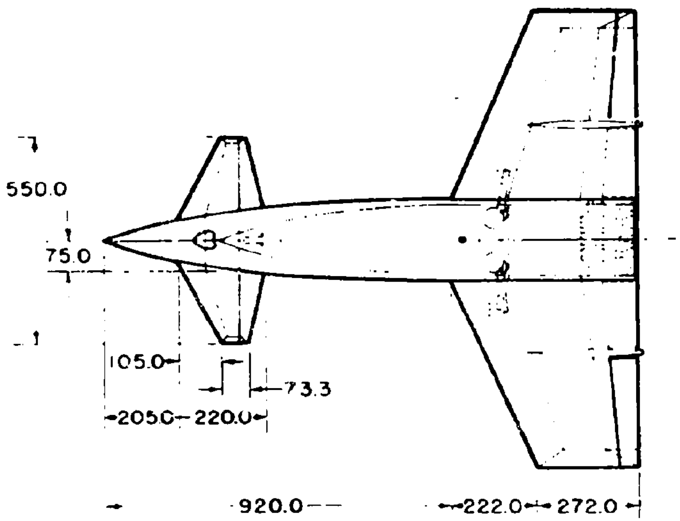
EXPOSED SURFACES		AREA - ft^2	A.R.	SECT	C_R/C_T
HORIZONTAL	AFT	2660	2.62	8%	2/1
HORIZONTAL	FWD	400	2.68	6%	3/1
VERTICAL	(EACH)	250	1.00	6%	3/1
WETTED AREAS					
FUSELAGE	TOP	900			
FUSELAGE	BOTTOM	1425			
FUSELAGE	SIDE	2890			
FUSELAGE	BASE	145			

ALIGHTING GEAR

MAIN WHEELS FOUR 52x16 TYPE VII
NOSE WHEELS TWO 46x14 TYPE VI

PROPULSION

FOUR 150,000-lb THRUST REGENERATIVELY COOLED ROCKET MOTORS
FOUR 75,000-lb THRUST REGENERATIVELY COOLED ROCKET MOTORS
FIVE 150,000 TURBINE PUMP UNITS
TWO 75,000 TURBINE PUMP UNITS



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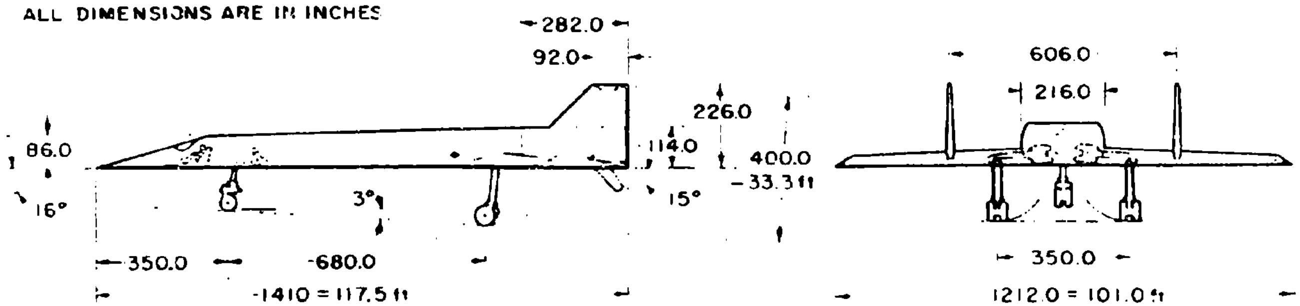
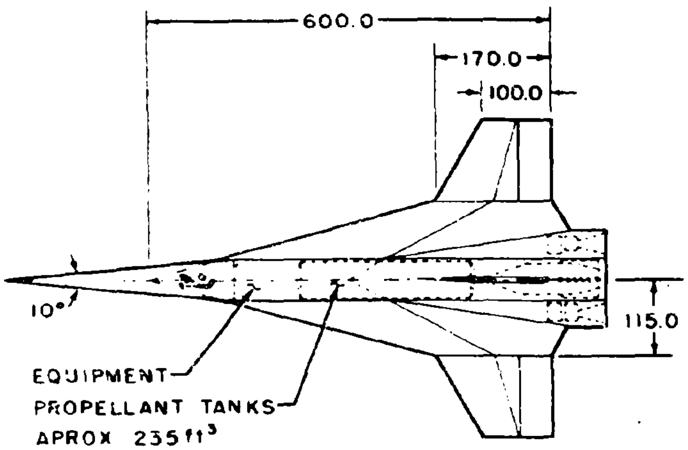


Figure 2. Initial Configuration of MX-2276 Stage I



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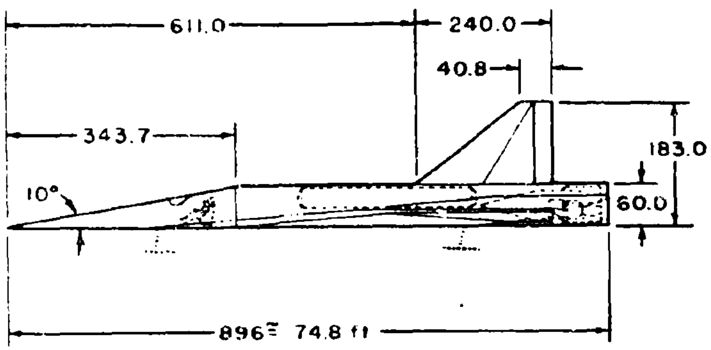


DATA

SURFACES

EXPOSED SURFACES	AREA - ft²	A.R.	SECT.	C_R/C_T
HORIZONTAL	622	1.90	4 %	
VERTICAL	104	2.0	5 %	5/1
BODY-NOSE BTM	71.4			
TOP	183			
CONSTANT SECTION BTM	196			
TOP	406			

WEIGHT SUMMARY

GROSS WEIGHT WITH BOMB	45,000 lb
EMPTY WEIGHT	14,600 lb
GLIDE BOMB	4,200 lb
EQUIPMENT	2,500 lb
N_2H_4	10,200 lb
WFNA	13,920 lb

PROPULSION

TWO 25,000-lb THRUST REGENERATIVELY COOLED, SWIVEL-MOUNTED ROCKETS; GAS GENERATOR, AND TURBINE PUMP

PROPELLANT STORAGE: 235 CUBIC FEET IN FUSELAGE, 80 CUBIC FEET IN WINGS

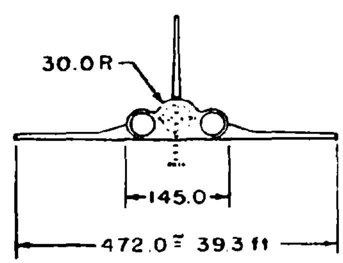


Figure 3. Initial Configuration of MX-2276 Stage III

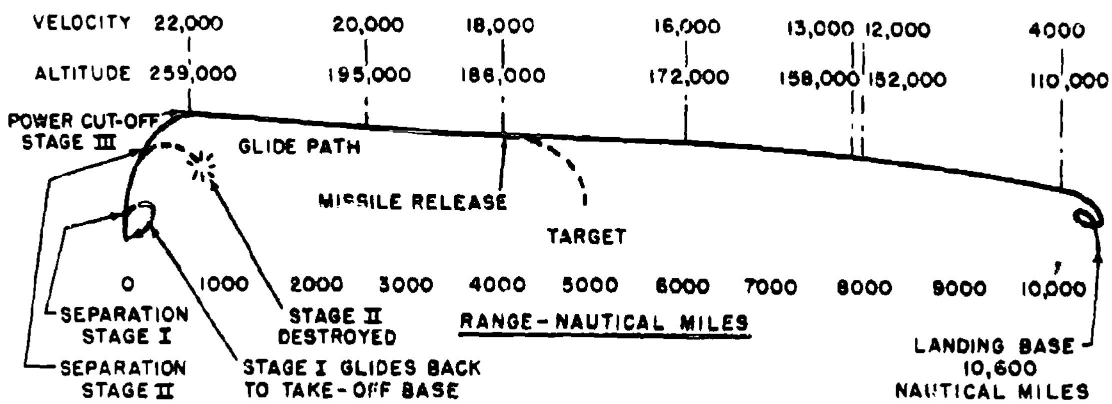


Figure 4. Typical Flight Profile

B. TYPICAL BOMBING MISSION

For purposes of illustration the mission has been divided into seven portions and each of these portions will be discussed separately.

1. Preparation for Take-Off

Prior to take-off the vehicle must be erected and fueled, the crew must enter and seal the enclosure, the inertial navigation system must be erected and started in operation, the desired flight path must be programmed in, together with checkpoint, bomb launch, target, and landing field location data.

2. Take-Off and Boost -

The vehicle is launched vertically. The crew position is such that a seated position will be assumed when the vehicle turns into a normal glide attitude. This arrangement is satisfactory since the boost accelerations thus occur normal to the chest, the direction of maximum tolerance to accelerations. During boost, using gimballed rocket motors, the vehicle is automatically programmed into the path which will yield the proper glide attitude at the end of boost. Both the first and second stages are ignited and burn during the first stage of boost. After approximately 112 seconds, Stage I burns out, is separated from the other two stages, and glides down to a landing. Stage II continues to burn for approximately 116 additional seconds at which time it is separated from Stage III and destroyed. Stage III power plants are then ignited and boost this stage to its maximum velocity and altitude. After burnout, Stage III glides along the desired flight path.

During the burning period of Stages I and II the vehicle is controlled by gimballed motors in these stages. Following the burnout of Stage II, the aerodynamic surfaces of Stage III provide the control forces.

3. Cruise

Throughout the cruise, the vehicle is in gliding flight at maximum lift-over-drag ratio. It is maintained on the desired flight path by means of the inertial navigation system which continuously computes the vehicle position and compares it with the desired position. The autopilot systems control the vehicle to provide minimum difference in these readings.

For reference purposes the crew is provided with a map, driven by the inertial system, which shows the position of the bomber with respect to the ground, as computed by the inertial system. Throughout the cruise phase the crew also observes the area traversed, using both the radar and the visual equipment. The radar provides a printed record of the region which the vehicle traverses. Since it is a side-looking radar any points of interest become visible on the radar presentation when they are abreast the bomber. Thus, the crew is able to locate a checkpoint with the radar after it has been recorded.

With the visual system, the crew will be able to look both ahead and behind for observation, when the weather permits. The presentation of both the radar and visual presentation will be provided in a manner such that the crew can locate checkpoints precisely. One method visualized for doing this is the flicker technique. The location as determined by the crew is utilized by the inertial system to obtain an error between the actual location and the computed location. This error is then used by the inertial system to correct the bomber flight path.

During cruise the bomber is flown completely by the autopilot using information from the inertial system. The crew is not required to navigate or fly the vehicle except in the case where it is necessary to override or ignore the inertial system.

4. Target Approach—Bomb Launch

As the bomber approaches the target area, the inertial system computes the range to go and compares this with the range at which the bomb is scheduled to be launched. When these ranges coincide, the bomb is automatically launched.

In this region, the location of checkpoints as determined by the crew becomes more important. This is true for two reasons: (1) the bomber has been flying longer and the errors increase with time, and (2) the actual target location may not be known precisely, although its location relative to checkpoints in the vicinity may be known with greater accuracy.

5. Bomb Correction

After the bomb is launched the carrier continues in gliding flight and passes the target, so that the target area can be observed with the radar. Thus the crew is able to obtain location of checkpoints, even after the bomb has been launched, and transmit any necessary correction to the bomb. The bomb is guided by its own inertial system which is erected from the inertial system of the bomber. The bomb dives into the coordinate location of the target. If the crew observes that these coordinates are no longer correct, the corrected coordinates are transmitted from the bomber to the bomb.

6. Postlaunch Cruise

After the bomb is launched the crew will obtain as much information concerning its actions as possible. Although it may not be possible to follow the bomb optically during its flight, certain factors will be known. These will include the fact that it was successfully launched, the exact time of detonation, and perhaps some measure of the yield. All of these are useful in assessing bomb damage. The remainder of the cruise is exactly as the prelaunch cruise.

7. Landing

It is planned that the landing operation will be performed by the crew. The total time of flight is so short that the conditions at the landing field will probably be known prior to take-off. Hence, the flights can be made when the weather is satisfactory. In order to eliminate weight by utilizing the full capabilities of the crew, only a minimum of equipment will be provided, such as GCA equipment. In addition to the regular equipment it may be desirable to provide ground radar, visual beacons, etc., to aid in locating the landing field.

C. RECONNAISSANCE

In addition to the requirement for bombing missions, long range reconnaissance requirements also exist. The invulnerability and reliability of the MX-2276 make this weapon especially suited to reconnaissance operations.

For reconnaissance missions various types of equipment such as radar, photographic, infrared, and visual could be included to obtain the desired information. Of these, only the radar technique possesses an all-weather capability and is, therefore, of prime importance. The basic elements of a radar system for reconnaissance use would be the same as those recommended for monitoring the navigation system. Extensive photographic equipment could be substituted for the bomb to provide high-resolution radar target analysis information.

IV RESULTS OF STUDY EFFORTS

A. MANNED WEAPON

1. General

This entire analysis has been directed toward the crew in Stage III since the conditions encountered by this crew are more severe and call for more unconventional measures than those in Stage I. The prime reason for the presence of a crew in the weapon system is to provide the system with the capabilities of fine discrimination, interpretation, judgement, and control which only a human can provide. These capabilities are the first to deteriorate if the human is required to function under some level of stress. Therefore, it is imperative to keep the human environment as comfortable as possible.

It is necessary to consider emergency conditions very early in the determination of crew provisions because these conditions may dictate the entire design. The magnitude of this effect will become apparent as this section developes. However, very briefly, the situation is as follows. If emergencies are considered, the crew must be clothed with special garments and a closed helmet. In this case, the clothing can be ventilated, the breathing air can be conducted directly into the helmet, the cabin can be pressurized with an inert gas, and there is no problem of ventilating the cabin, removing toxic gases, etc. The problems of pressurization and ventilation are relatively simple. The difficulty with this situation is that the pilot is enclosed in relatively heavy, clumsy clothing, gloves, etc. Under these conditions, comparatively minor irritations can become major distractions. In addition, the pilot is required to view everything through a visor which will reduce his vision to some extent. Since the most important functions of the crew are the interpretation of visual data from map, radar, periscope, and other instruments, this is a disadvantage.

If the emergency provisions are not required, the entire cabin can be pressurized and ventilated with a gas mixture satisfactory for respiration, the pilot can be clothed in comfortable garments of his own choice, and he can function in a very comfortable environment. The disadvantages of not providing for emergencies are of course obvious. However, in this type of weapon system, the probability of an emergency caused by enemy action is very remote due to the inherent invulnerability of the system. In addition, it appears that a very reliable pressurization and ventilation system can be designed which is independent of electronic equipment, auxiliary power supply, and other equipment which would serve to reduce reliability.

In this early study stage, it is not proposed to ignore the possibility of emergency. Therefore, the system discussed will include emergency provisions. Insofar as possible, the discussion has been divided into specific subject fields. However, because of the extreme interdependence of these fields, a complete separation is not possible.

2. Crew Functions

As previously explained, the advantages of a manned weapon system lie first in the capabilities of a human to discriminate, assimilate, and evaluate data of many different types and forms; second, in the ability to translate the results of this assimilation and evaluation and decide upon a course of action; and third, to initiate these actions which may be of a diverse nature. Such advantages are very difficult to evaluate quantitatively; therefore, in the following sections, a series of the occasions upon which the pilot will prove advantageous will be enumerated as a form of qualitative evaluation. The various phases of the flight will be discussed separately.

a. Navigation

Since the prime functions of the pilot are his duties in connection with the navigation of the bomber and guidance of the bomb, this phase will be discussed first. Throughout the flight the pilot will monitor the path of the bomber using either the radar or optical system to check the actual position of the vehicle with the position as shown on the map driven by the inertial system. The equipment for this function is provided primarily for use at the target area. However, if necessary, this equipment may also be used at various checkpoints for midcourse navigation. Prior to take-off the location of various checkpoints will be incorporated into the system. These will include checkpoints associated with the target as well as some midcourse checkpoints. It is anticipated that the midcourse points will merely be used as a reference. However, in the event of an error these can be utilized the same as those at the target. The target location may not be accurately known in an absolute set of coordinates, although its location relative to nearby checkpoints may be known with more accuracy. In addition, the target itself may not be visible either optically or with radar. Therefore, in order to have the greatest possible reliability, the terminal guidance phase must be provided with a capability for utilizing offset checkpoints.

In addition to these prime functions, the pilot performs many secondary functions. For example, as he approaches the checkpoint the pilot will select the best means of observation i.e., either radar or optical methods. If he uses radar he can adjust the gain, focus, contrast, scale, etc., in order to obtain the best possible presentation. Thus, the best possible data are available for use after the flight for reconnaissance or IBDA. If the system is obviously operating incorrectly, the pilot may elect to perform the navigation by pilotage or navigating himself.

b. Take-off

The take-off will be completely automatic; the function of the crew will be to observe the operation in general and take the proper actions in the event of an emergency. Examples of these are as follows:

(1) If the power plant stops operating prior to the correct burnout time, the crew can decide upon the proper action, depending upon the time of the stoppage. If necessary, the remaining stages can be jettisoned and Stage III returned to base.

(2) If the failure is in a navigation component the crew can decide upon the best alternative, i.e., circle and land, or continue the mission. This decision will depend to some extent upon the mission. A similar failure in an unmanned system results in loss of the vehicle and failure to accomplish the mission.

c. Cruise

During the cruise portion of the flight, the prime operation will be navigation. However, the crew will be provided with the means for flying the aircraft in the event of failure of all or a part of the navigating and autopilot equipment. This capability will not be limited to occasions of complete failure of the system. For example, if the roll, pitch, or yaw autopilots should malfunction, the crew could take over the function of the particular component and continue the flight with the remainder of the automatic system functioning properly.

At the beginning of cruise (as well as from time to time throughout this portion of the flight) the crew can observe the range, speed, and altitude to determine if the vehicle is following the proper flight pattern to achieve the necessary range. If it is not, a decision can be made as to the proper course of action. Such action may include:

(1) Determination and correction of trouble if possible.

(2) Deviation to an alternate, shorter mission.

(3) Deviation to an alternate closer landing field, jettisoning the bomb if necessary.

(4) If close enough, return to base.

During this portion of the flight the crew will also be able to observe both the terrain and air over which the bomber passes. This observation may provide various types of information.

(1) Points of interest such as missile launching sites, airfields, military installations of various types, city shapes, etc. The amount of material thus gained will depend upon the means of observation available, i. e., radar or visual, at the time of observation. The location of these points may be measured accurately by use of the aimpoint location mechanism or approximately by an estimation by the crew.

(2) If any type of defense against this weapon is developed the crew will be able to observe such defense. It will then be possible to take a limited amount of evasive action. An early report of such defense activity means changes in tactics, and possibly changes in the weapon system itself, can be initiated immediately.

An important function of the crew is to report defects in operation and equipment in order that these may be corrected. The very fact that these may be recognized and reported rather than result in mission failures and lost vehicles is an important advantage.

3. Crew Environment

a. Pressurization and Respiration

Although the pilot will be required to wear either a partial pressure suit or a full pressure suit as an emergency provision, it is not proposed that suit pressure should be the only type of pressurization provided. The present trend in suit development indicates that the suits will be stiff when inflated and only satisfactory for completion of missions in emergencies. It is proposed to pressurize the cabin to the necessary value, to leave the suit uninflated, and to have the suit inflate automatically if a loss of cabin pressure occurs.

The pressure requirements for the crew are dictated primarily by respiratory requirements, although they are also modified by aircraft weight considerations and the danger of both explosive decompression and aeroembolism. At sea level the partial pressure of oxygen is of the order of 3.5 psi. From a respiratory standpoint, if this partial pressure is maintained in the cabin, the respiratory system will function just as at sea level. Therefore, an atmosphere of 100% oxygen at 3.5 psi is satisfactory for respiration purposes. However, this pressure is so low that the possibility of aeroembolism becomes a major problem.

In order to prevent aeroembolism two choices are available. The pressure can be maintained at a higher value or the crew can prebreathe in an atmosphere of 100% oxygen at sea level for several hours prior to take-off. By this means the nitrogen dissolved in the body fluids and tissues (which cause the embolism) can be eliminated prior to take-off. The necessity for several hours prebreathing prior to flight imposes an operational limitation which is extremely undesirable. Therefore a cabin differential pressure requirement of 5 psi has been selected. Since the ambient pressure is less than 0.1 psi throughout the cruise, the differential pressure is essentially the absolute pressure.

In order to set up a satisfactory pressurization program, emergency conditions must be considered early in the design. These consist primarily of two effects, the first of which is explosive decompression. Explosive decompression through a differential of 5 psi is the limit which it is presently believed a human can withstand without major injury. Differentials higher than this may have fatal results. Even with this limit, with expansion to the extremely low ambient pressures involved in this case, the decompression must be followed by almost immediate recompression. For this purpose the pressure suit is provided which automatically pressurizes in the event of loss of pressure. The pressure suits presently available, both the partial type and the full type, can only supply a breathing pressure of less than 5 psi. This fact gives rise to the second problem, i.e., aeroembolism in the event of emergency. The drop from 5 psi to suit pressure may very readily cause aeroembolism for most people. Therefore, it is this emergency condition which must be considered, rather than the normal cabin atmospheres. This requirement means that prebreathing may be required as a precaution against such an emergency. If the suit development program produces a suit which can provide a 5 psi differential pressure for emergencies, this requirement will not exist.

It has been established that a 5 psi pressure differential is satisfactory from a physiological standpoint. It is also more satisfactory than 3.5 psi for comfort considerations. A minimum oxygen partial pressure of 3.5 psi has been shown to be desirable, so therefore the remaining problem is the selection of the gas to be used for the additional 1.5 psi partial pressure. Thus far, the discussion has been concerned with the gas used for respiration. At this point it becomes necessary to consider the over-all system. As a provision for emergencies, it is necessary that the pilot wear a helmet which can be pressurized with a gas suitable for breathing. Since this provision must be made for emergencies, it can also be used for normal operations With this arrangement, the cabin can be pressurized with nitrogen or another inert gas to reduce any possibility of fire hazard, while the pilot can breathe 100% oxygen at 5 psi. The exhaled gases of the crew can be exhausted overboard, thus eliminating any problems of water vapor accumulation.

A more suitable arrangement, which is recommended, consists of a cabin environment with 3.5 psi oxygen and 1.5 psi helium. With this system the pilot will be able to breathe the cabin atmosphere. These breathing provisions are such that the crew is effectively prebreathing throughout the flight. Thus, if explosive decompression occurs, the dangers of aeroembolism occurring are reduced, even without prebreathing prior to flight.

The source of the cabin and helmet oxygen will be a standard liquid converter. With the volumes required, this will provide a lower installation weight than the use of high pressure gas bottles. The helium used to make up the 1.5 psi increment of the cabin pressure will be provided by a high pressure gas storage cylinder because of the extremely low temperatures required to maintain liquid helium (8° R). The amount of oxygen required for respiration is very small relative to that required for ventilation, and therefore the quantities of gases to be used will be discussed in the next section.

b. Ventilation

The problem of ventilation is very closely associated with both pressurization and respiration. However, ventilating inside the pilot's clothing rather than ventilating by moving and changing the entire cabin atmosphere will reduce the required supply appreciably. For example in recent tests men were able to wear a completely impermeable suit in a 90°F environment for 3 hours with very little stress buildup. In other tests in 130°F environment, men wearing underwear, a T-1 partial pressure suit, coveralls, and a ventilating suit which is presently under development, were comfortable with 6 cubic feet of air per minute supplied at 50° to 60°F. For the flight time of this weapon system, this would require a supply of approximately 13 pounds of ventilating air. The environment of this cabin will be of the order of 60° to 70°F (see the following section). Therefore, the required ventilating rates will be considerably less.

If, on the other hand, the emergency conditions are ignored and the pilot is to breathe the cabin atmosphere at all times and be ventilated by movement of the cabin atmosphere, a greater air supply will undoubtedly be necessary.

c. Temperature

The problem of cabin wall temperatures which was originally considered the most serious problem of the cabin design, is solved very well by the proposed structural design (Section IV-C.) The results of the preliminary heat protection tests indicate that the walls can be maintained at very nearly the boiling point of the coolant water. In the cabin area it is desirable that the coolant should be ventilated to ambient pressure rather than cabin pressure. The boiling point of water at ambient pressures (or even the local pressure on the aircraft) is of the order of 80°F maximum. With temperatures of this magnitude there is no problem from the standpoint of pilot environment.

The wall temperatures may thus be kept to tolerable values. If the atmosphere is supplied from liquid stores, the temperature of the entering gases may be kept as low as desired. In fact, they must be warmed prior to admission. Thus, both requirements for a tolerable thermal environment can be satisfied.

d. Acceleration

Accelerations may be divided into two categories; those occurring during boost and those occurring during flight. The boost accelerations are imposed normal to the chest of the crew, the direction in which humans possess the highest tolerance. The boost program required for the original configuration (Reference 1) is well within the limits of human tolerance. Human subjects have been subjected to very similar acceleration patterns in the centrifuge at WADC and suffered no ill effects other than some loss in motor proficiency while undergoing the accelerations. This loss was not noticeable when they were subjected a second time.

Accelerations during flight will be limited by the airframe strength. During cruise the acceleration will necessarily be low because of the low dynamic pressures. However, during the terminal and landing phases of the flight, high accelerations may be imposed. The crew will be required to wear anti-g suits. However, accelerations should be no greater than in conventional aircraft.

e. Emergencies

Some of the problems arising in the event of emergency have already been discussed, but the complete problem must be considered. In the event of loss of cabin pressure, the pilot is provided with a pressure suit and helmet. The suit is automatically inflated and the helmet pressure (which already exists) is modified to a value compatible with the suit pressure. An emergency pressure source is available for the helmet and suit if the ship's supply is lost. If the emergency merely consists of loss of pressure, the crew may continue the mission, abort, or choose an alternate mission. If the emergency is more serious, i.e., the vehicle is out of control and falling, the following sequence occurs: The crew remains in the airplane until it falls to an altitude and slows to a speed where ejection may be accomplished safely. Ejection seats are provided. These are automatically controlled such that ejection cannot be accomplished above the safe speeds and altitudes. Analysis shows that objects falling from the altitudes and speeds attained by the weapon system will have terminal velocities of approximately Mach 1 at altitudes where ejection may be safely accomplished. In approaching these velocities, however, high skin temperatures will be encountered. Thus it is necessary to provide heat protection throughout the descent.

A special capsule is not provided. The weight penalties involved in the provision of a capsule, the special disconnect fittings, special stabilization provisions, etc., impose higher penalties than the advantages warrant. Instead, the entire airplane will be provided with emergency equipment such that the crew can ride it down until safe ejection conditions exist. Since the airplane already has provision for temperature protection and stabilization, these provisions need not be duplicated. In addition, after burnout, the fuel and oxidizer tanks will be inerted. Similarly, the cockpit and instrument section can be pressurized with an inert atmosphere in an emergency, thus reducing the fire hazard appreciably.

The long range of this weapon system combined with the locations of the launching sites, targets, and landing sites, require operation over arctic, temperate, and tropical regions. Much of the operation will be over water. Therefore a global survival kit and exposure suit will be incorporated as part of the pilot's equipment.

With the original pickaback configuration, it is necessary that crew ejection in the bomber take place upward, in order that it can be accomplished during the boost period.

f. Clothing

Throughout the foregoing sections, various special types of clothing have been described as necessary equipment. Specifically, clothing capable of the following four functions are required:

(1) Pressurization

(2) Anti-g protection

(3) Ventilation

(4) Exposure protection

Several types of partial pressure suits are under development by the Air Force, and both the Air Force and the Navy have existing programs for the development of a full pressure suit. The latter is the type most desirable for this weapon system and the prospects of having a suitable, operational, full pressure suit by the time this weapon system is operational, appear very good. Various types of sealed helmets are also under development for use with these suits. Since the crew must operate with visors closed at all times and good vision is so important, the type of helmet recommended for this vehicle is one with a single-curvature visor to avoid distortion and using gas defogging rather than electrical defogging in order that the visor may be free of wires. Helmets of this type are under development and should be available within the desired time period.

Anti-g protection is presently provided either by special suits or is incorporated in the partial pressure suits. This feature can also be incorporated in the full pressure suits.

There is presently under development at WADC a ventilating suit which has proven satisfactory for the type of operation desired here. This suit fits over the existing partial pressure suits and, since the latter are of a porous construction, satisfactory ventilation is obtained. These suits are designed to be worn under an anti-exposure suit which has four air exit valves, one at the end of each extremity.

The full pressure suit program is aimed at providing all four of these features in a single suit. As previously stated, there is a good chance that this garment will be available within the operational date of this weapon system.

g. Visibility

Two different conditions must be considered: visibility inside the cabin and visibility of the ground from the airplane.

At the extreme altitude at which this system cruises, the sky will be relatively dark. This fact, combined with the fact that any windows will be covered by hatches during cruise, raises a requirement for adequate cockpit lighting. Experience from experimental high-altitude flights has been that deep contrast exists between objects in the cabin illuminated by sunlight through the windows, and those not illuminated. Therefore, an even artificial illumination will be more satisfactory than that provided by natural means. The optical, radar, and map presentation must also be visible inside the cabin. Therefore, it is necessary that the level of cabin illumination should be no higher than absolutely necessary.

Visibility of the terrain from the vehicle presents another major problem. Under static conditions, the normal eye is able to discern an object subtending an angle of one minute of arc (Reference 2). At an altitude of 260,000 feet this means that an object with a minimum dimension of 75 feet can be distinguished with the naked eye. Several factors affect this capability; it falls off with velocity and the direction of the velocity with respect to this dimension; it is a function of the angle of the path of sight along which the object is viewed; and it also can be varied by magnification. For the present application Figure 5 indicates the apparent terrain speed throughout the cruise phase of the flight. The maximum speed of 6.2 degrees per second corresponds approximately to that of a 65-knot airplane flying at 1000 feet. In Reference 3 it is shown that terrain speeds of this magnitude reduce the acuity very little and that speeds of the order of 30° to 40° per second are required before this acuity begins to degrade very markedly. If the eye is aided by magnification the increase in dimension of the objects being viewed would be directly proportional to the magnification, while the terrain speed would also increase directly in proportion. Thus assuming a magnification of 4 times, under static conditions, the 75-foot minimum dimension reduces to 19 feet, and if the acuity was halved by the increased terrain speed, the minimum dimension would be 38 feet. This minimum dimension is for a path of sight directly below the airplane and will increase as this path moves out from the vertical.

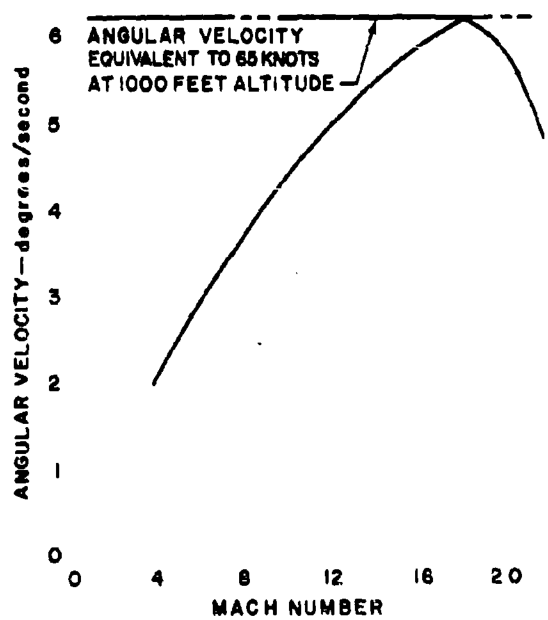


Figure 5. Equivalent Terrain Speed During Cruise of MX-2276

Thus far the discussion has been concerned with acuity; visibility is another phase of the same problem. If the object is large enough to be discerned, its visibility is determined by several other factors:

(1) General level of illumination.

(2) Contrast between brightness of object and background against which it is seen.

(3) Attenuation by atmosphere.

These factors cannot be controlled except in a general way. For example in the case of (1) the flights can be executed during daylight hours and the magnification will be of some aid.

It is proposed that a projection-type periscope with a field (or true) lens presentation should be provided for external vision. By means of a periscope the pilot can be provided with more vision than can be obtained with a canopy. In addition it is easier to cool the small lenses of the periscope than the large area of a canopy. Although this type of periscope cannot provide as wide a field of view as would be available with an ocular-type scope it has several outstanding advantages. The first and most important of these is that the presentation can be observed while the observer is wearing a helmet with the visor closed. This is not possible with an ocular-type periscope. A second advantage is that the head of the observer is located an appreciable distance (16-18 inches) from the field lens during observation. This is especially important during landing where it has been shown to be difficult for an operator to keep his head in position on the periscope without severe bumping. An additional advantage is that the pilot can look at the instrument more easily and probably have less eye adaptation time than if an ocular-type were used.

Although the field of view is limited with a periscope, the sighting head can be moved manually to scan the areas of interest. This head can also be moved manually to watch a single area of interest as the aircraft passes it. The normal position of the sighting head in cruise will be such as to provide optical coverage of the area shown by the radar. Two additional functions are desirable; it should be possible to look forward and down for landing, and also to look back and follow the bomb after launch. Figure 26 shows the vertical angle through which the head must be moved in order to see the bomb during descent. This figure shows that it is necessary to look horizontally behind the carrier at launch, and very nearly horizontally at the termination of the bomb flight in order to observe the detonation, and determine its relative location. Depending upon the location of the sighting head, this capability can be provided.

In addition to the visual presentation, it is planned to provide the radar and the map information through the same field lens. This will conserve cockpit space and make the use of the flicker or other matching technique more feasible.

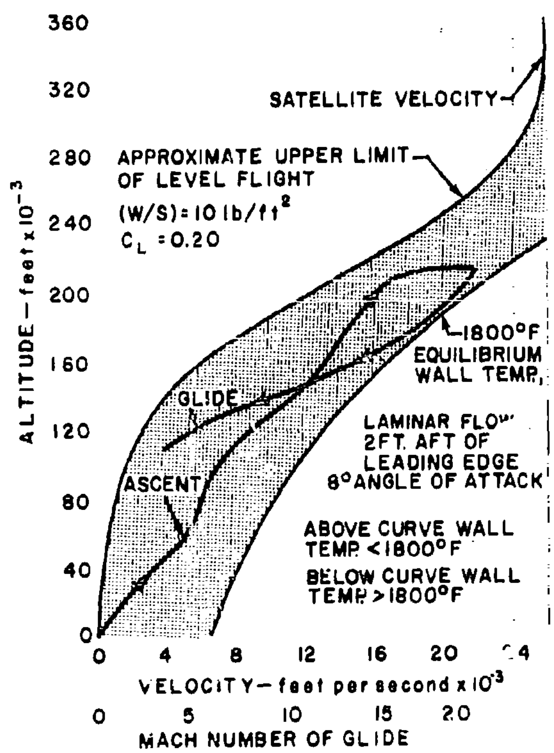


Figure 6. Aerodynamic Flight Limits

h. Cosmic Radiation

One of the many problems to be considered in Stage III flight is the question of probable health hazard from cosmic radiation. This problem can be divided into at least two major categories: (1) the physical data describing the intensities of cosmic radiation (mass and energy spectra and frequency of occurrence), and (2) the biological effects of such radiation.

The first category has been investigated to the extent that it is fairly well agreed, (Reference 4) that the maximum radiation within the atmosphere occurs at an altitude of 12 nautical miles, followed by a sharp decrease with a minimum at about 20 nautical miles and a gradual increase on out into space. At the initial cruise attitude of MX-2276 (approximately 40 nautical miles) the vehicle is at this minimum radiation level. As the vehicle glides down during cruise, it approaches the altitude of maximum radiation.

Heavy nuclei begin to appear at altitudes of 12 nautical miles and increase with altitude. The Stage III cruise therefore occurs in regions where the heavy nuclei are just beginning to become a problem but are still rather sparse.

Investigation of the second category has not resulted in such good agreement. The radiobiological effects of different portions of the radiation spectrum are decidedly different. The effects of the heavy nuclei are not yet well established since it has not yet been possible to duplicate them from terrestrial sources. As a consequence, the over-all problem is not very well defined; therefore, the solutions are not yet evident.

Several facts do appear which can be stated with some confidence. Effective protection from heavy nuclei by shielding does not seem feasible (Reference 5). Unless extremely heavy shielding is used it will not provide adequate protection and may even intensify the danger. The probability of a hit from the very heavy nuclei is quite low especially at the altitudes in question. Therefore, judging from the present state of the art, the probability of significant somatic radiation injury is very low. This is especially true for the very short flight times involved.

4. Comparison of Manned and Unmanned Weapon System

In this study contract, which was conducted without configurational investigations, it has not been possible to make a quantitative analysis comparing the manned and unmanned systems. Even if configurational data were available, many of the advantages and disadvantages are of an abstract nature and cannot readily be assigned a number value. A qualitative analysis is the only type which can be made for such values.

The disadvantages of a manned system are rather apparent and can be listed briefly as follows.

1. The weapon system is heavier since it must include the weight of the man, his ejection seat, and the various gear required for his use.

2. The man occupies a volume which could otherwise be reduced or eliminated.

3. The volume of the cabin occupied by the man must be pressurized and cooled. Pressurization requires increased structural weight, and additonal equipment is necessary for both pressurizing and cooling.

4. The presence of a human is required over enemy territory.

If the foregoing disadvantages are examined it becomes apparent that they are not completely eliminated by the use of an unmanned system.

1. The weight of the man and his gear is supplanted by additional equipment which is necessary to perform his functions.

2. The volume occupied by the man also cannot be completely eliminated since the replacement equipment will occupy some volume.

3. The volume which must be pressurized and cooled will be reduced but not eliminated, since it is still necessary to provide such a volume for certain types of gear. Thus, if the

equipment must be provided in any event, the additional requirements for the man show up chiefly as increased amounts of materials, such as stored fluids.

4. The danger to the man flying over enemy territory is reduced because the system vulnerability is low and the reliability will be high due to the presence of the man.

Whereas the foregoing discussion is concerned chiefly with weight penalties of individual vehicles, the advantages of the manned system lie in reduced missions and, hence, reduced numbers of vehicles required to achieve given results. To illustrate all of these advantages it is necessary to consider the development of the weapon system from the very beginning.

Early in the design of the system certain problems of guidance, communication, and specialized equipment are eliminated. In the flight test and development stages the advantages are more apparent. In these tests the pilot can observe and report on the functioning of many of the vehicle and equipment characteristics. In an automatic system such observation would require extensive instrumentation. More important is the ability of the pilot to cope with new phenomena which develop in the testing of such advanced systems. It is difficult to anticipate all such phenomena and to design and install special equipment for detecting and handling them prior to their occurrence. This type of advantage is typified by the experience gained during the record speed run in the X-1 airplane. Recoverability of the test vehicle is an extremely important feature provided with the manned system. The vehicles for weapon systems performing the missions anticipated will be large and expensive. Many hours and dollars are required to get such a vehicle ready for flight with all components operating correctly. If the flight is the order of one hour and the vehicle is expended at the end of the test, the test program becomes extremely expensive. If the vehicle is recovered, the entire system can be available shortly for further testing with substantially fewer laboratory and ground test hours.

Throughout the flight tests the following advantages accrue for the manned system.

1. Qualitative reports of various vehicle characteristics can be obtained from the pilot without special instrumentation. Examples of these would include yaw, pitch, roll, stability, handling, etc.

2. Telemetering can be largely eliminated, together with a large amount of the recording equipment otherwise required.

3. Elaborate flight programs for obtaining specific information can be followed by the pilot without extensive programming equipment and automatic inputs.

4. Test instrumentation can be made to operate better if the pilot is available to make adjustments during the test.

5. Through the elimination of such links of communication as telemetering and recorders, a large area of question is removed when the data indicate a malfunction, i.e., the question of whether the data is poor because of vehicle operation or data transmission is eliminated.

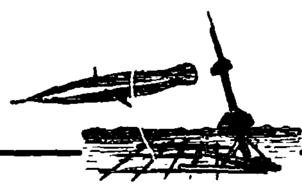
In the operational missions of the aircraft many of these advantages carry over, but additional advantages also become important. The first and probably most important in the case of the MX-2276, is the many capabilities which the presence of a human at the target area provides to the weapon system. These include the reduction in the required accuracy of the navigation system and the reduction in the preciseness with which the location of the target need be known. The exact coordinates of the target need not be known if its location relative to a radar or optical checkpoint is known. In addition to these primary navigational advantages, the crew will be able to report a large amount of information concerning the flight which would not otherwise be available. These data would include description of the vehicle operation in general, such as how the speed and altitude agreed with those programmed, how well the vehicle remained on the prescribed course, etc. Other information which the crew may obtain will include data obtained by observation, i.e., a type of reconnaissance information. Through visual and radar observation the crew may observe such items as military installations, city configurations, possible missile launching sites, aircraft runways, etc. In addition to this type of information any defense which the enemy may develop for use against the weapon system can be reported as soon as it is used. Such reports may result in immediate changes in tactics for subsequent missions or even modification of the weapon system.

Another major advantage is the information which can be obtained concerning the bomb. In the normal course of events the crew will know where the bomb is launched and whether or not the launch was successful. The bomb should be observable by optical means from several seconds after launch until detonation if sufficient magnification is provided. Thus its behavior during flight can be determined to some extent. In addition to observing the bomb, the crew will also utilize available checkpoint information to correct the bomb flight path. The location of the detonation will be observable. When this material is compared with that available from an unmanned vehicle of similar range the advantage is apparent.

The manned system has operational flexibility. Alternate targets can be attacked, a certain amount of correction is available if the target is not precisely where it is reported, and in the event of certain types of failure the bomber can be saved and also the warhead if desired. The presence of a man in the system improves the over-all reliability through his ability to perform the functions of many parts of the automatic systems involved.

The final advantage of including a man in the system consists of his ability to perform the landing operation. It is probable that an automatic or remotely operated system could be designed which would perform this function. However, the problem of bringing the vehicle close enough to the landing area so that ground control equipment could be used to bring it in and land it, would require additional complexity in the guidance system. The problem of performing the necessary maneuvers to reduce speed and altitude from the high residual values at the end of cruise would require the development and installation of complex automatic equipment.

It is concluded that from the viewpoint of an operational system the advantages justify the use of a manned system.



B. AERODYNAMICS

1. General

The basic intent of this effort has been to determine the areas in the field of aerodynamics which must be considered in the design and development of a very high-speed and high-altitude aircraft. In addition to ascertaining the extent of specific problems in these areas, the effort also included a review, improvement, and extension of the initial methods of analysis to provide the best indication of the characteristics of this weapon system consistent with the most advanced state-of-the-art. The aerodynamic studies are reported in detail in Reference 6.

The areas investigated during this study included atmospheric characteristics, glide performance parameters, flight mechanics, the general field of aerodynamic heating, stability and control, stage separation, and bomb trajectories.

In order to plan the study in these basic areas and hold the investigation to applicable conditions, it was necessary to define the flight conditions encountered by this weapon system. Figure 6 shows the altitude-velocity relationships for the boost-glide type of path as determined from the present study. Although this path may be modified by future optimization, it is adequate for the purpose intended.

Also shown in Figure 6 are the aerodynamic flight limits of control, lift, and heating. The control limit corresponds to a dynamic pressure of 10 pounds per square foot. This value of dynamic pressure has been suggested as the lower limit which still permits the use of aerodynamic-type controls. The lift limit is for a low static wing loading of 10 pounds per square foot and a hypersonic lift coefficient of 0.20. This is indicative of the upper altitude limit for level flight. The effect of centrifugal force resulting from flying a circular path about the center of the earth is included. It should be noted that the lift limit loses its significance as satellite velocity is approached because the effective gravity is becoming zero. The heating curve is for an 1800°F skin temperature two feet from the leading edge of the wing, and approximates the temperature problem with respect to the flight path. The shaded area shows the region of flight possible with respect to these limits.

In many of the aerodynamic investigations for the present study it was necessary to have a fairly specific configuration to evaluate. Since the study did not require development of better or optimized shapes, the configuration presented in the initial work was retained (Figure 1). The majority of the aerodynamic studies have been concerned with the hypersonic flight of the bomber i.e., the glide airplane. In the initial layout of the bomber, major consideration was given to obtaining good glide performance, since the system performance potentialities were of greatest interest at that time, and less attention was given to stability and control. The configuration may not represent the final configuration, but it was considered to be sufficiently realistic and typical of the class of vehicles in question for use in the present study. An additional advantage was that performance re-estimations could be directly compared with the initial estimates.

2. Glide Performance (Nonrotating Earth)

A complete re-evaluation of the lift and drag coefficients and the maximum L/D characteristics of the aircraft has been made. These characteristics have been evaluated at $4 < M < 20$ since the major portion of the range (approximately 97%) is attained between these Mach numbers.

Shock expansion theories were employed to predict the local surface pressures and flow conditions, except in the case of the nose where the concept of Newtonian flow was also used. The skin friction drag coefficients were determined from incompressible skin friction formulas modified for compressibility by reference temperature parameters. Boundary layer transition was assumed to occur at 2.8×10^6 local stream Reynolds number throughout. Altitude and angle of attack were found to have a large effect on skin friction in the higher altitude region.

The performance based on these methods is shown in Figure 7. The effect of the bomb is also indicated. Except for the lower altitude at the higher velocities, this performance is substantially the same as the original estimate.

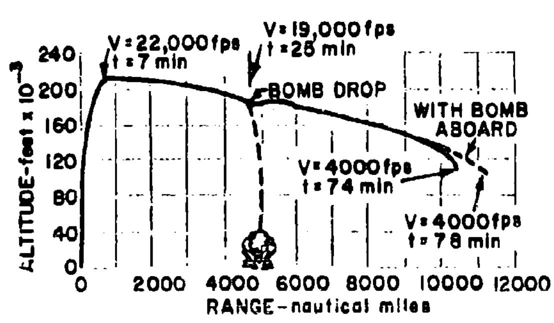


Figure 7. System Flight Path

The dynamic pressures of the glide flight of the bomber with and without payload are shown in Figure 8. Throughout the flight, these values indicate the feasibility of aerodynamic stability and control. Even at the highest altitudes, the dynamic pressures are of the same order as for present subsonic aircraft. The maximum indicated airspeed is 316 knots and the minimum is 146 knots.

The values of L/D_{max} and altitude for the present estimation and those originally estimated are compared in Figure 9. The L/D_{max} curves are very similar to those of the present analysis although giving a slightly lower L/D_{max}. As a result, the nonrotating earth glide range for the present calculation does not differ appreciably from the initial one. The equilibrium altitude is lower for the new calcu-

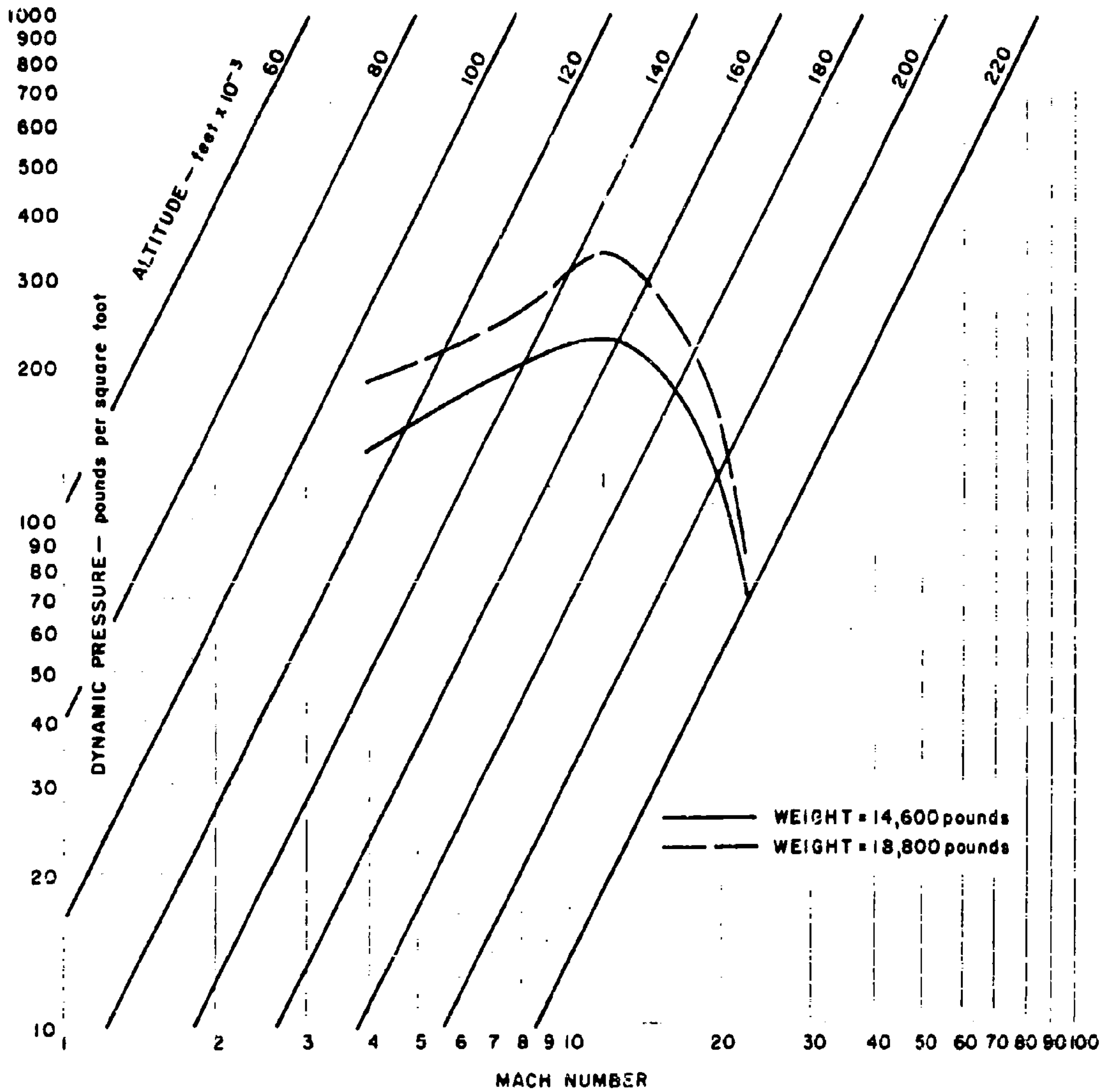


Figure 8. Free Stream Dynamic Pressure

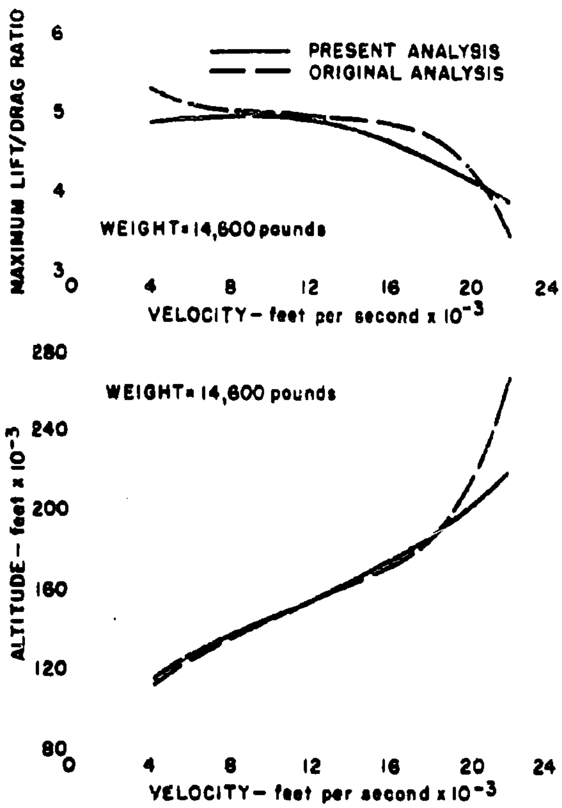


Figure 9. Comparison of Original and Present Flight Paths

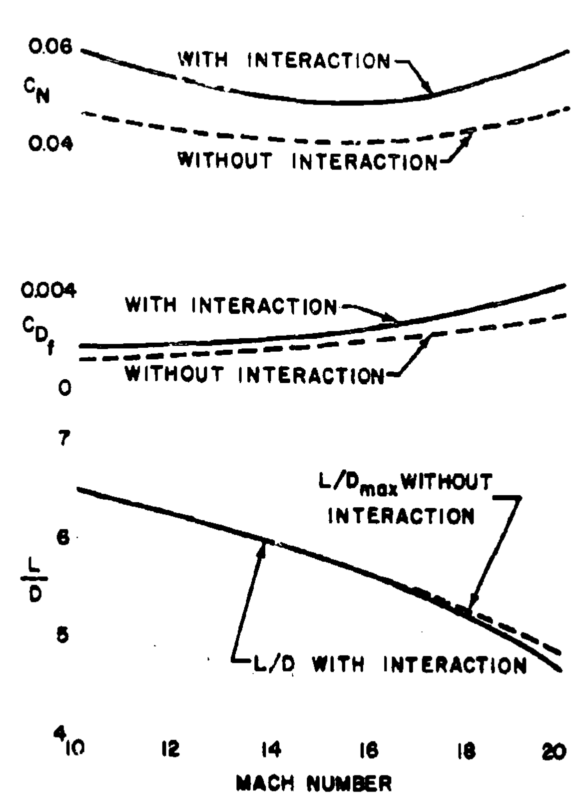


Figure 10. Effect of Shock-Wave-Boundary Layer Interaction on Wing Characteristics

lation at the higher speeds. The latter is partially due to the use of the new Rocket Panel* atmosphere.

A preliminary investigation into the effects of shock wave-boundary layer interaction on the wing L/D has been made. It was found that at the present equilibrium glide altitude there are appreciable effects on surface pressures and skin friction, but that the summation of these effects on L/D produces only small changes from the no-interaction L/D_{max} values. Figure 10 presents these results. It should be noted that the differences in pressure result in an increase in lift, which will, in turn, increase the equilibrium altitude somewhat. Since L/D_{max} is a function of altitude, differences in L/D_{max} may result. This effect of increased equilibrium altitude has not been included in the present study.

*The Upper Atmosphere Rocket Panel, Harvard College Observatory, Cambridge, Mass.

3. Flight Mechanics

a. Flight Path

(1) Effects of Earth Rotation on Glide Range

In order to become more familiar with the new terms in the linear equations of motion and to demonstrate the differences in glide trajectories typical of the MX-2276 when great circle courses in various directions about the rotating earth are taken, several glide trajectories have been calculated with the aid of IBM computing equipment. Since earth rotation effects can be illustrated using constant aerodynamic parameters over the velocity range, typical constant values of wing loading, lift-drag, and lift coefficient for maximum lift-drag ratio were assumed for most of the calculations. These values were taken as W/S = 32.0, L/D = 4 and C_L = 0.09.

For flight about the equator the calculation of the glide trajectory reduces to a two-dimensional problem, since for this case the Coriolis and centrifugal forces act in the vertical direction. For flight to the east the Coriolis force adds to centrifugal force, reducing the lift required for any given velocity and, hence, reducing the drag, thereby increasing the glide range. For flight to the west the opposite effect occurs and the glide range is reduced accordingly. The results of these calculations are presented in Figure 11 together with the glide range which is obtained from the assumed parameters when the rotation of the earth is neglected. For the given assumptions and for an initial velocity of 22,000 feet per second relative to the surface of the earth, a 25% increase in range results for flight about the equator to the east, and 15% reduction in range results for flight to the west as compared to the range calculated for a nonrotating earth.

Flight about the poles of a rotating earth results in a three dimensional problem since in this case components of the centrifugal and Coriolis forces act along both the normal and lateral axes of the vehicle. For this condition glide range may be calculated in several ways. First, the flight of the vehicle may be conducted so that the angle of roll is maintained at a zero value. For this case it is necessary to yaw the vehicle in order to provide the force required to maintain the great circle polar path. The disadvantage of this approach is that the vehicle will most likely be less efficient in generating aerodynamic forces in yaw than in generating lift forces; hence, some penalty will be paid in order to maintain a zero roll angle great circle path. The other alternative for such a path

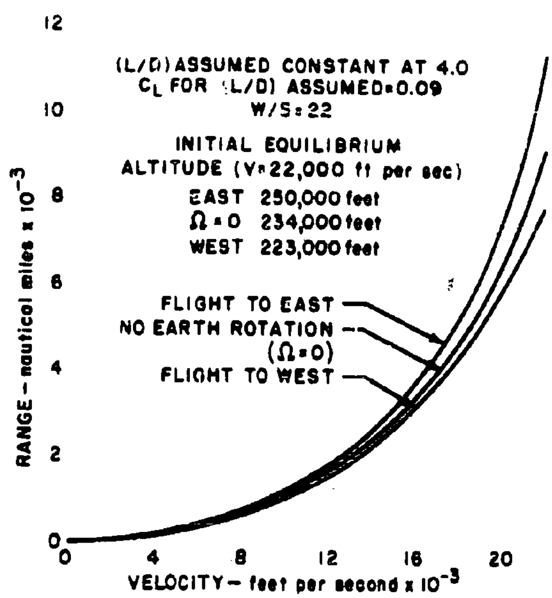


Figure 11. Effect of Earth Rotation on Glide Range. Equatorial Flight

is to utilize increased lift force to overcome the lateral components of the centrifugal and Coriolis forces. This procedure will require rolling the aircraft to some bank angle so that the vertical component of lift force will maintain the desired glide path while the horizontal component of lift force is employed to maintain the desired great circle path. Approximate calculations were made to determine these effects for the former case. The results, shown in Figure 12, indicate an appreciable effect on range but much less than the effects on equatorial flight. It is apparent that flight direction must be considered when the mission of the weapon system is planned.

(2) High Altitude Trajectories

In general, the heat transfer from the boundary layer to the adjacent aircraft surface decreases as the local airflow density is decreased. This effect suggests that the MX-2276 temperature problems might be alleviated to some extent using higher altitude flight paths than those originally proposed. There are sev-

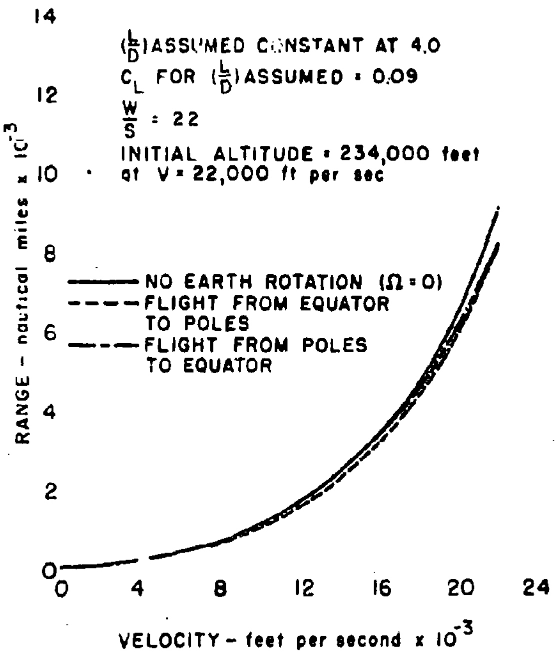


Figure 12. Effect of Earth Rotation on Glide Range. Flight on a Great Circle through the Poles. Zero Bank Angle

eral ways of achieving these higher altitude flight paths: (a) by decreasing the wing loading; (b) by increasing the lift coefficients, or (c) by flying a partial lifting path above the original equilibrium path. Since the wing loading of the initial configuration is already low (less than 25 pounds per square foot) the second and third methods were given the most consideration.

Figure 13 presents the effects of increasing the angle of attack and, hence, the lift coefficient, and thereby increasing the glide altitude for a given velocity. The effects of shock boundary layer interaction were not included and it was assumed that flow on the upper wing surface continues to expand with increasing attitude, although separation is quite probable. It should be noted that the lower surface temperature is not significantly relieved by increasing the altitude through increasing the angle of attack. This is because the local lower surface pressure and velocity remain nearly the same through angle of attack variations, since the lift force required to support the vehicle is essentially constant with altitude at a given velocity. However, the upper surface temperature and heating are considerably reduced.

Increasing the angle of attack from 8° (approximately the angle for maximum L/D) to 15° yields a relatively small decrease in the total heating. The reduction in range is shown by the curves presented in Figure 14 which are calculated for a constant 8°- and 15°-angle of attack.

The third method of achieving higher altitude flight, the partial lifting path, requires that the initial flight path angle be greater than that for a maximum L/D glide. Increases in the flight path angle can be easily attained by programming the ascent path to the

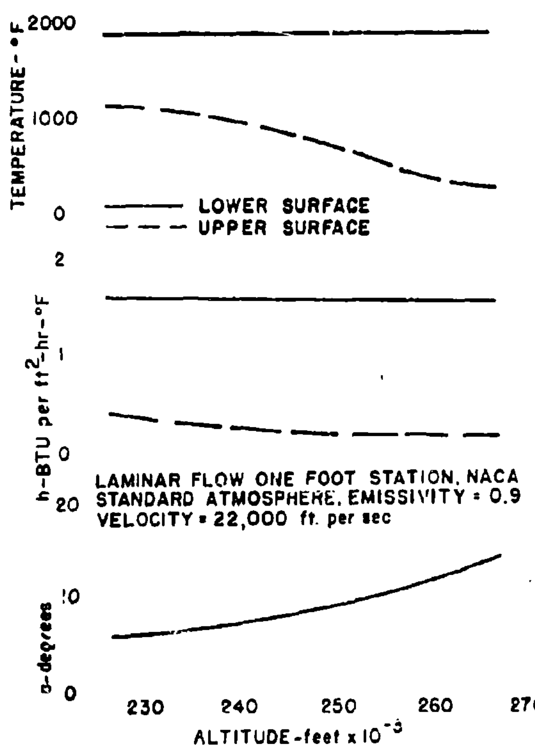


Figure 13. Effect of Altitude on Equilibrium Temperature with Constant Lift and Airspeed

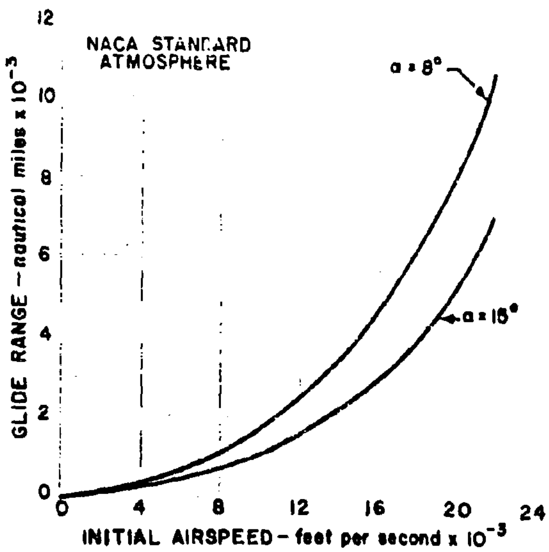


Figure 14. Effect of Angle of Attack on Glide Range

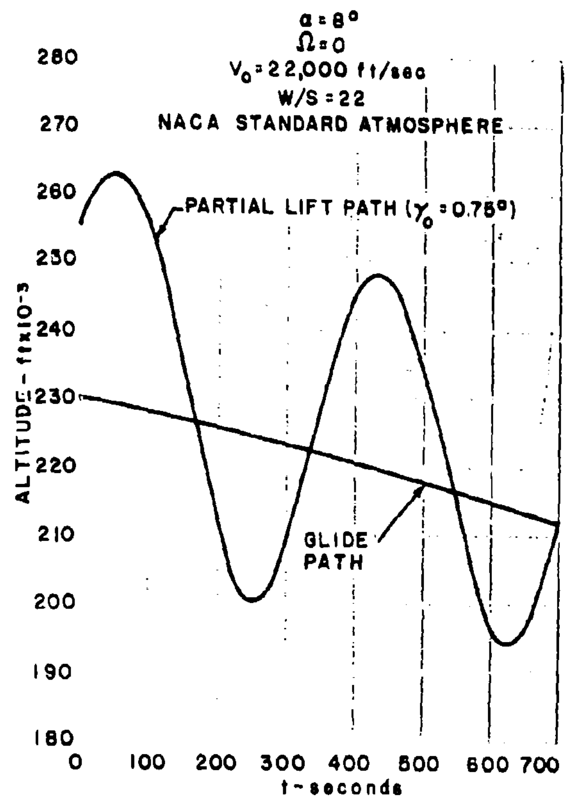


Figure 15. Comparison of Partial Lift and Glide Trajectories

desired final angle. Such a program will in itself result in a higher initial altitude. Thus, at first glance, the partial lift path appears advantageous. This advantage disappears, however, when the results of the trajectory calculations presented in Figure 15 are studied. For these calculations a nonrotating earth was assumed and the lift coefficient and lift-drag ratio for an angle of attack of 8° were chosen. The first 700 seconds of an equilibrium glide and the trajectory for a final ascent path angle (initial partial lift path angle) of 0.75° are shown. As would be expected, the initially inclined partial lift flight path degenerates very quickly into an oscillation about the equilibrium glide path and approaches the well known skip trajectory. In this respect, the partial lift path possesses the disadvantages of the skip path wherein high loads, temperatures, and heat fluxes are encountered at the bottom of the oscillation.

These methods of achieving higher altitude will not provide large reductions in temperature or heating effects. Another method which should be the subject of future investigations is discussed in Section IV-F.

b. Flight Path Heading Control

In order to accomplish a given mission, that is, to deliver the hypersonic vehicle between two designated points on the surface of the earth, it is necessary to arrive at a means by which the vehicle may be guided between the specified positions. One of the important effects to be included is the rotation of the earth, which, in effect, means that a vehicle which is guided to a specific point on the surface of the earth is being directed at a target which is moving in space. It may be most convenient to accomplish this navigation by conventional means, that is, for flight between two designated points to follow the connecting path of a great circle on the surface of the earth.

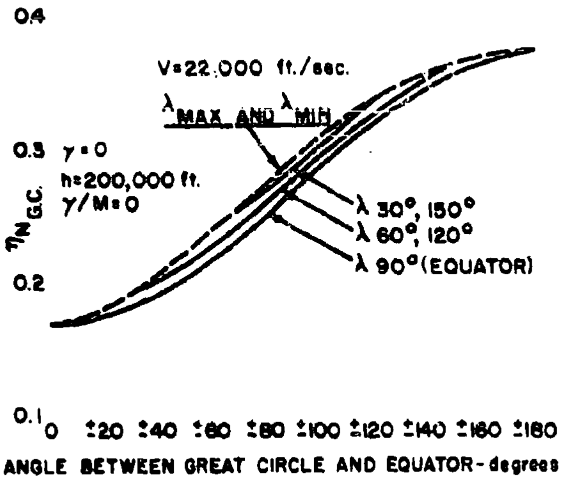


Figure 16. Normal Load Factor for Banked Great Circle Flight

In the discussion of flight mechanics it was indicated that flight about a great circle which is inclined with respect to the earth's equator results in components of the centrifugal and Coriolis forces that lie along the lateral axis of the vehicle. A possible method of countering these lateral forces lies in rolling the vehicle to a bank angle wherein the horizontal component of lift force balances the Coriolis force component. The feasibility of such a program depends upon the degree of roll angle and the amount of lift required, since it must be remembered that increasing lift results in increasing temperatures on the lifting surfaces, and excessive roll angles may have an adverse effect upon the navigation equipment. Calculations show that the maximum roll angle required to fly a great circle path anywhere on the earth is 22° at a velocity of 22,000 feet per second. Figure 16 shows the normal load factor required for banked great circle flight at the same speed and altitude.

Of equal interest is the maneuverability which can be obtained for various normal load factors. Figure 17 shows the rate of turn as a function of the ratio of normal load factor in a turn to normal load factor for banked great circle flight, for flight at the equator, and flight where the initial heading is toward the poles. From these results it is evident that a considerable increase in normal load factor (hence lift) is required to obtain even moderate turning rates.

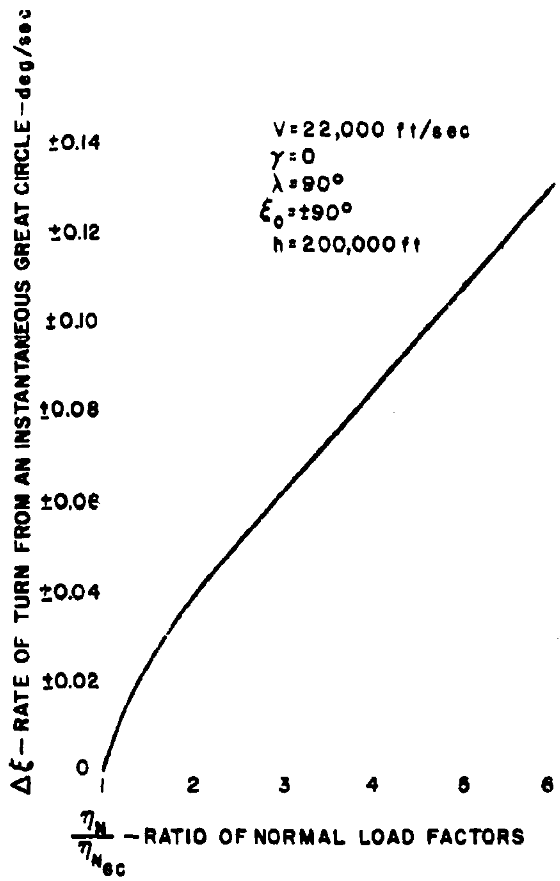


Figure 17. Rate of Turn from an Instantaneous Great Circle

4. Aerodynamic Heating

Aerodynamic heating of the structure at the hypersonic conditions necessary to achieve the desired performance poses a problem in this weapon system. In order to estimate the parameters involved, it has been necessary to extend present methods of analysis beyond the speeds and conditions at which any data exist to substantiate their accuracy. Because of the high speeds and relatively low decelerations in glide, i.e., long duration of flight, the bomber was the object of the most extensive analysis during this study. The airframe design under consideration (Section IV-C) utilizes a thin outer skin which will attain equilibrium conditions very quickly.

a. Heat Balance

Skin temperatures and heat transfer to aircraft surfaces are determined from a summation of heat flux both into and away from a surface. For this application the heat flux on the outside is so much greater than any which may occur on the inside, that this latter can often be neglected insofar as skin temperature calculations are concerned. The heat balance then consists of the fluxes which result from convection from the boundary layer to the skin, radiation from the surrounding environment to the skin, and radiation from the skin to the surrounding environment. The condition wherein heat flux into the surface is balanced by heat flux out (skin temperature constant) is referred to as the condition of equilibrium skin temperature.

b. Radiation

Several sources of radiation exist in the surrounding environment: the sun, the atmosphere, and the boundary layer. The magnitude of the two former sources is so small it has been neglected in this analysis. An investigation of the latter source was conducted during this study, and results indicate that it has important effects and should receive further study. However, sufficient results were not obtained to include this source of radiation in the heat transfer results described herein.

Radiation from the skin to the surrounding environment is highly influenced by emissivity of the skin. Since the absolute value of this factor was not known, a value of 0.9 was assumed and the effects of varying this value are illustrated.

c. Convection

The convective heat flux into the surface is governed by the compressible heat transfer coefficient. Various solutions for obtaining this coefficient exist and the one selected for use in this analysis together with the assumptions involved is the reference temperature method reported in detail in Reference 6.

d. Aerodynamic Heating Effects Aft of the Leading Edge

(1) General

The equilibrium temperatures for the one foot station both top and bottom of the wing during glide flight are given in Figure 18. These temperatures are based on the flight plan for the 18,800-pound weight condition (with bomb). Since the local stream Reynolds number is below 2.8×10^6 for these stations over the entire glide path, the results for laminar flow are shown. The temperature of the bottom surface is highest at burnout and decreases with time. On the other hand, the temperature of the upper wing reaches a peak during the glide about 30 minutes after power shut-off. The reasons for this are twofold: (a) the angle of attack decreases which means there is a smaller angle of expansion and (b) the effect of expansion on the local conditions is relatively lower per degree at lower Mach numbers.

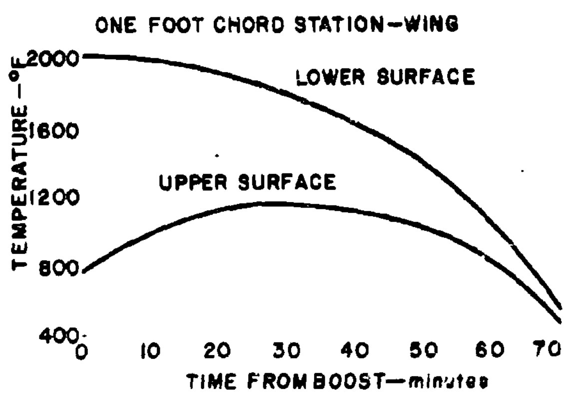


Figure 18. Equilibrium Temperature Time History

TABLE I. PARTICULAR FLIGHT CONDITIONS CHOSEN

Time at Temp.	Mach Number	Velocity (ft/sec)	Altitude (ft)	Angle of Attack (degrees)
10	20	20,640	198,000	7.7
27	16	17,200	172,000	6.8
52	10	10,680	140,000	6.4

A calculation was made to compare the temperature on the bottom of the wing at M = 16 on the 18,800-pound vehicle to the same point on the 14,600-pound vehicle (after bomb release). The temperature was reduced from 1825°F to 1755°F, which is a relatively small reduction compared to the 22 percent change in wing loading that occurred. This is due in large part to the fourth power radiation term in the heat balance equation. It also indicates that a large change in wing loading will be necessary to strongly influence the surface temperatures. If subsequent design shows a larger wing loading is necessary, the accompanying temperature rise would be small. This reasoning, of course, applies particularly to the case of radiation cooling.

In order to give a representative picture of the equilibrium temperatures on the vehicle as a whole over the flight path, profiles of equilibrium temperatures at three specific Mach numbers for the 18,800-pound configuration have been computed. The particular flight conditions that were chosen are given in Table I.

Figure 19 gives equilibrium temperature profiles on the bottom of the body and wing for the three flight conditions discussed. The point of transition is shown in this and subsequent figures as being at a local stream Reynolds number of 2.8×10^6. For the bottom the transition point is located at 40, 18, and 9 feet for Mach numbers 20, 16, and 10, respectively. If the transition were delayed to Re = 10×10^6, it would materially reduce the heating problem as shown by the extension of the laminar curves beyond transition. In particular it would move the transition point back so that the bottom of the body would be completely laminar at M = 20, laminar back to 64 feet at M = 16, and 32 feet at M = 10. The effect of the steep temperature gradient on the heat flux in the region of laminar flow has been examined and in general was found to be small.

The abrupt temperature rise shown at transition will not actually occur. Transition requires a finite length so that the increase

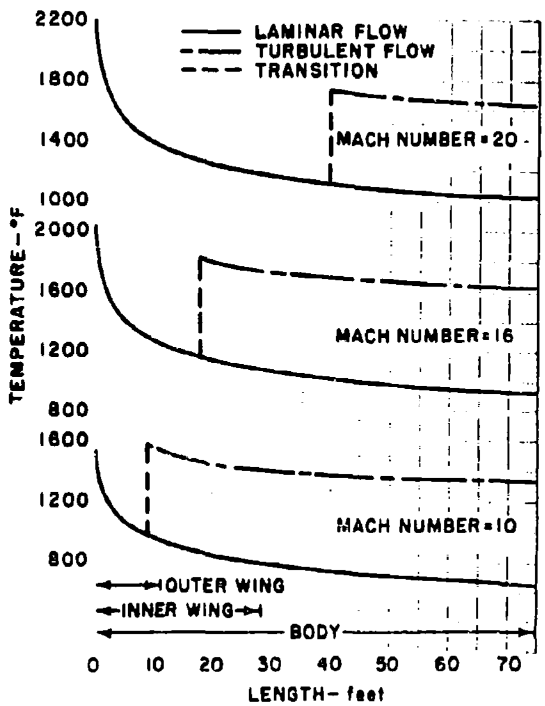


Figure 19. Equilibrium Temperature at Bottom of Body and Wing

should occur over an extended region. However, there is insufficient information available to define this region and its actual profile.

It should be noted that the maximum temperature experienced over the bottom of the body and wing is below 1800°F except for the first two feet. In the region of turbulence 1700°F is a representative average temperature at the higher speeds.

In order to illustrate the combined effects of varying body length, emissivity, and angle of attack, Figures 20 and 21 are presented for laminar and turbulent flow, respectively. One particular flight condition at Mach number 20 was chosen for this representation. The effect of body length has been noted previously. The importance of the coefficient of emmissivity of the aircraft surfaces is demonstrated. The selection of surface finish for the glide vehicle merits careful appraisal with respect to emissivity.

Angle of attack is a particular significant parameter showing a variation of approximately 100°F per degree angle of attack for laminar flow. For tubulent flow this variation is about 150°F. In light of this, it is evident that maneuvers requiring additional angles of attack and control deflection may increase the heating loads on the surfaces. It is apparent that pull-up and turns would be temperature-limited rather than "g"-limited, e.g., at the beginning of glide flight the aerodynamic lift provides one-third of the lifting force; therefore, in order to provide a one "g" maneuver, large angles of attack would be required.

(2) Effect of Shock-Boundary Layer Interaction on Equilibrium Temperature

The interaction of the shock wave and boundary layer has been demonstrated experimentally at hypersonic speeds. This interaction is greatest at the nose or leading edge, and decreases downstream. The magnitude of this interaction and its effect on equilibrium temperatures has been estimated from two dimensional analyses for the top and bottom of the wing aft of the six-inch station. On the bottom surface, the equilibrium temperature is increased a negligible amount (less than 40°F). On the upper surface the effect is much larger; the temperature increases from 1100°F to 1800°F at the six-inch station at Mach 20. At lower Mach numbers and greater distances aft, the shock-boundary layer interaction effect is reduced. Thus, within the limitations of the present flight path, it may be concluded that the effect of shock wave-boundary layer interaction must be included on the upper surface at the higher Mach numbers.

e. Leading Edge Heating

Temperatures and heat fluxes in the stagnation areas of leading edges and nose have been estimated for two conditions on the flight path presented in the performance section. These are shown in Table II. The estimations are based on an extension of the theory for a cylindrical leading edge normal to the flow and the theory for a hemispherical nose. The theories are for incompressible flow; it is

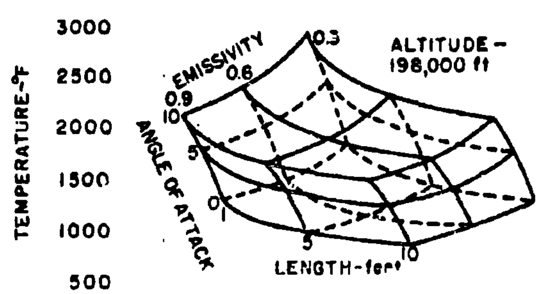


Figure 20. Equilibrium Temperature at Mach 20 for Laminar Flow

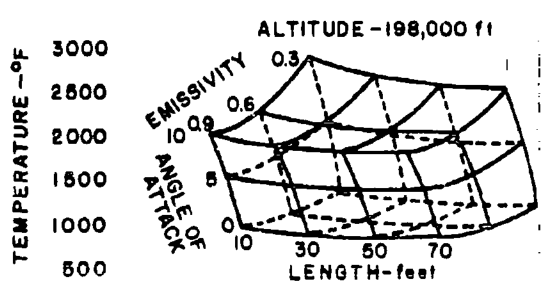


Figure 21. Equilibrium Temperature at Mach 20 for Turbulent Flow

TABLE II STAGNATION POINT CONDITIONS

Assumptions:

1) No Dissociation
2) Emissivity = 0.8

Model	Equilibrium Wall Temperatures — FLIGHT POINT: h = 214,000 ft, M = 21.9	h = 172,000 ft, M = 16
1/2-inch Sphere	6126°R	5828°R
2-inch Sphere	5237°R	5008°R
1/2-inch Cylinder	5927°R	5643°R
2-inch Cylinder	5064°R	4845°R

Heat Flux Through Surface: (BTU/sq ft-sec)

Model	h = 214,000 ft, M = 21.9		h = 172,000 ft, M = 16	
	Wall Temp. = 1500°R	Wall Temp. = 3000°R	Wall Temp. = 1500°R	Wall Temp. = 3000°R
1/2-inch Sphere	720.6	628.6	657.2	550.7
2-inch Sphere	359.3	298.8	327.6	259.8
1/2-inch Cylinder	621.4	538.0	566.7	470.8
2-inch Cylinder	309.7	253.4	282.3	219.8

assumed that they apply to the subsonic flows behind the normal shock waves at the leading edge and nose stagnation areas. Dissociation effects have not been included, though the air temperatures behind the shock are sufficient to produce some dissociation. At the present time the dissociation properties of air are not known well enough to predict quantitatively the effect of dissociation. However, it is thought that dissociation will not increase the temperatures shown.

The equilibrium temperatures indicate the necessity for either a very high temperature material or else cooling. It is of interest to note that the smaller radii shapes produce the higher temperatures. It should be remembered that the temperatures and heat fluxes shown are only for areas near the stagnation points. Theories have been developed which predict a reduction in heat transfer coefficient from the stagnation point to the 90° shoulder on spheres or cylinders. This result has been substantiated to some extent by tests.

The temperatures and heat fluxes in Table II are given for an unswept leading edge. In tests at $M \cong 7$ (unpublished) the NACA has found that the heat transfer to the front half of a cylinder is reduced by sweepback at a rate approximately equal to the cosine of the sweep angle. Since the relation between leading edge length and sweep is inversely proportional to the cosine of the sweep angle the total heat input is not reduced by sweep as is the local heating. Thus, it appears that, if the leading edge is to be cooled entirely by an internal coolant, sweepback in terms of necessary coolant is not a prime consideration. However, if significant radiation cooling is present, for a given surface temperature the total radiation will increase directly as the leading edge area increases with sweep resulting in a definite over-all gain from sweepback.

f. Transpiration Cooling

For the higher glide velocity conditions the temperatures for the first several feet of surface may require cooling. Transpiration cooling has been studied as a means of accomplishing this. In this method of cooling a coolant gas is passed through a porous outer skin into the boundary layer where it modifies the boundary layer flow profiles such that the heat transfer to the surface is reduced. The effectiveness of this method of cooling has been proven in low speed tests, but quantitative experimental information at hypersonic speeds is lacking.

A transpiration cooling theory has been developed in the present study and is discussed in detail in Reference 6. The theory applies to a laminar boundary layer which is most pertinent to the present case since transpiration cooling will most probably be confined to the areas near the leading edges where the flow is expected to be laminar. It is probable that the injection of the relatively small amounts of coolant into the boundary layer will not destabilize the laminar flow. In the strict sense air must be used as the coolant because the theory is based on homogenous boundary layer considerations for which the coolant and boundary layer flows must be of the same gas. It is believed that a dissimilar coolant can be handled with sufficient accuracy through a simple extension of the present theory, however.

To evaluate the merit of transpiration cooling, the quantities of air injection necessary to cool the first foot and the first 10 feet of the lower surface (though not the leading edge radius itself) have been estimated for the Stage III (18,000 pounds) glide conditions. The average coolant flow per square foot for these

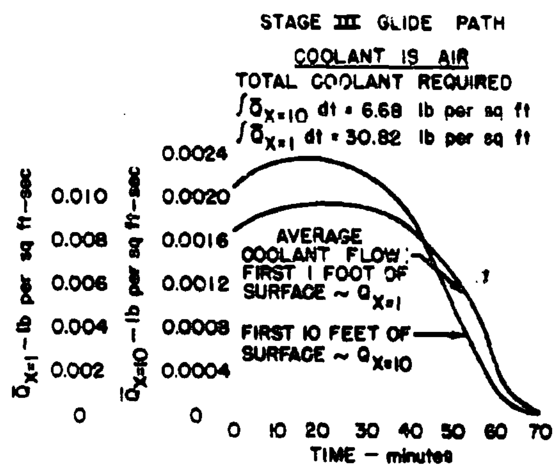


Figure 22. Average Coolant Flow vs Time for Transpiration Cooling of Wing Lower Surface to 1600°R

conditions is shown in Figure 22. The total quantities of coolant air necessary are found to be 6.68 and 30.82 pounds per square foot of surface cooled for the ten and one foot surface length, respectively. Thus, for example, to cool the first foot of the approximately 40-foot span of the third stage, 1230 pounds of coolant air would be required.

There is considerable promise of further reduction in the coolant rate from the value quoted through use of better coolants than air. Water may be much better because it adds a high heat of vaporization to the process. It is believed its use would at least halve the previously mentioned coolant requirement. It is concluded that transpiration cooling definitely merits further development. The effects of shock-boundary layer interaction and slip flow, which may both be strong near the leading edge, are not accounted for in the present theory and should be included in future studies.

5. Stability and Control

a. General

A hypersonic vehicle of the MX-2276 type must be controllable and have acceptable

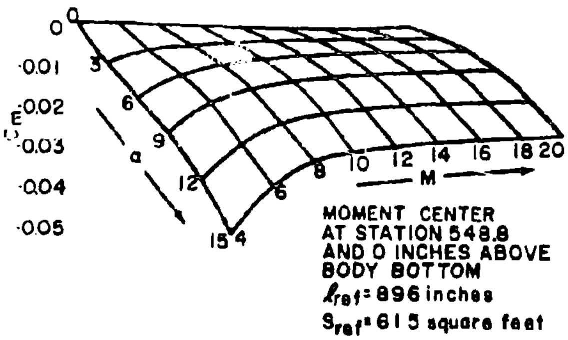


Figure 23. Pitching Moment Coefficient vs Mach Number and Angle of Attack

handling characteristics throughout the regime it encounters--from the ascent, with separation of various stages, to the peak of the hypersonic glide, during the glide at hypersonic and supersonic flight velocities and for the low-speed landing conditions. The stability and control characteristics of aircraft up to low supersonic speeds are presently understood to a reasonable degree. The design in this regime should encounter no fundamental lack in methods of analysis. In the hypersonic flight regime, however, new conditions are encountered which require a review of present methods. The equations which govern the motion of the vehicle include new terms which are significant due to the high velocities which are expected--this requires a complete reanalysis of the methods which are presently employed to investigate the dynamic stability. Free flight test vehicles are presently approaching the flight regimes of the MX-2276 and will be able to furnish empirical data. Approximate theoretical flow models may be employed to obtain theoretical estimates.

b. Static Stability

Some preliminary estimates of the aerodynamic static longitudinal stability of the third stage have been made. The pitching moment coefficient variation with Mach number and angle of attack is presented in Figure 23. It is of interest to note that the principal variations in moment coefficient occur in the region M = 4 to M = 8 and that above M = 8 only slight variations in moment coefficient occur with Mach number at a given angle of attack.

On the basis of these preliminary investigations, it appears that no undue difficulty will be encountered in obtaining static longitudinal stability in hypersonic flight with proper location of the center of gravity of the airframe. The problem of matching the requirements for stability at hypersonic velocities with those at lower flight speeds has not been considered as yet and will require further studies. In addition, the method used to obtain the moment characteristics of the airframe has neglected the effects of shock-boundary layer interaction on the distribution of forces and moments. A preliminary evaluation of interaction indicates that an appreciable effect may result and should receive further consideration.

c. Control Surfaces

The moment characteristics of several control surfaces have been studied briefly to determine the feasibility of using aerodynamic control in hypersonic flight. Several types of controls of equal surface area have been considered to determine the relative merits of each. These are: (1) a slab-type trailing edge (constant chordwise thickness) control shown on the present configuration, (2) a moveable tip control which is a portion of the outboard section of the present wing, and (3) a trailing edge wedge control formed by making a wedge of the outer wing from the 50% chord back, with upper and lower wedge angles equal to the wedge angle of the present wing.

The effectiveness of these control surfaces has been determined from inviscid two-dimensional shock or expansion theory. A comparison of the variation of pitching moment coefficient with control surface deflection is shown for all three types of controls at Mach number 20 in Figure 24.

A preliminary investigation indicates that increasing control deflection for trim results generally in a reduction in maximum lift-drag ratio. This will result in a reduction of the range calculated for zero control deflection

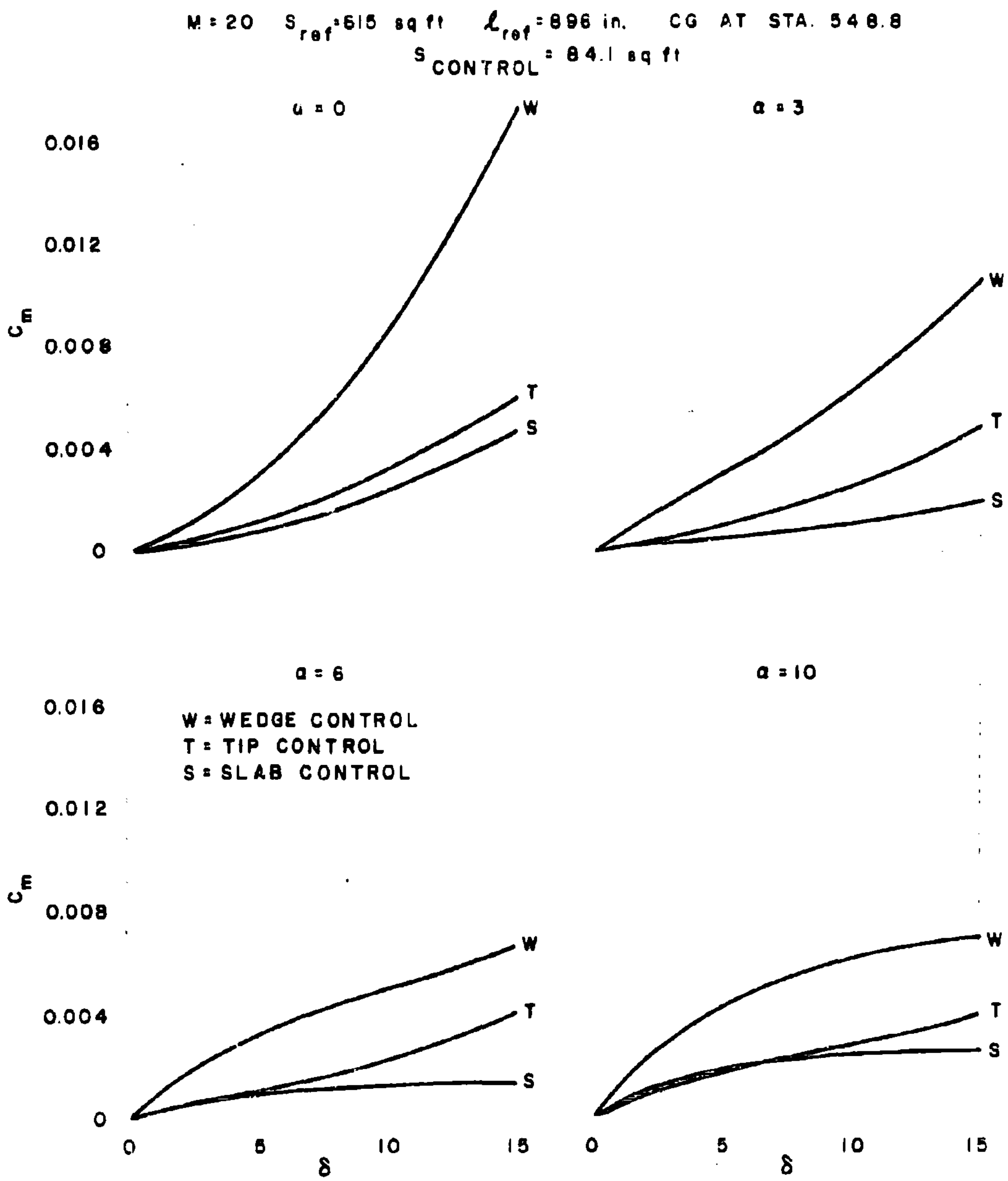


Figure 24. MX-2276 Stage III Control Surface Pitching Moment Coefficient

and, since the angle of attack for a given lift coefficient is somewhat higher with control deflections than with no control deflection, an increase in the heat transfer.

It is apparent therefore, that selection of a control surface will result from a number of compromises which would be evaluated in a design study. The present results indicate that sufficient control effectiveness is available at hypersonic velocities to make aerodynamic control feasible without large aerodynamic losses.

d. Shock Boundary Layer Interaction

The preliminary evaluation of static stability and control at hypersonic flight velocities has been made using aerodynamic parameters which were determined from inviscid fluid flow theory. As previously noted, the hypersonic flight path of the MX-2276 enters regions wherein the effects of fluid viscosity become increasingly important in determining the aerodynamic pressure forces which act on a moving body. Fluid flow theories which neglect these effects may be expected to give only approximate estimates of these forces. A preliminary theory which considers the case of the two-dimensional flat plate at an angle of attack in viscous flow including shock boundary layer interaction has been developed and is presented in Reference 6. With this theory, a sample calculation has been made for simple semiwedge airfoil section with a chord length equal to the mean aerodynamic chord of the MX-2276 outer wing, at M = 20, at an angle of attack of 8°, and an altitude of 200,000 feet.

For this two-dimensional analysis the effects of shock-boundary layer interaction increase the moment coefficient by 13 percent and the left by 16 percent. The effects on three-dimensional shapes such as bodies and wings are as yet undefined.

6. Separation

As shown in Figure 1, the initial configuration uses three stages of boost. The first and second stage boosters and the bomber are assembled adjacent to one another in a parallel arrangement. The first and second stage boosters separate and drop away as their fuel loads are expended and the final stage then accelerates to the initial glide conditions. The aerodynamics of these separations was considered in a qualitative manner but no definite conclusions can be drawn. It is recommended that during the preliminary design of the system, booster-vehicle combinations be put into aerodynamic test as soon as possible since this is the only way of evaluating such effects in a quantitative manner.

7. Tandem Staging

While in the original concept the stages are arranged in parallel, the feasibility of tandem staging should not be excluded from future design considerations. Some of the relative advantages of the two arrangements are as follows:

a. Handling - The parallel arrangement is apparently easier to erect, combine the stages, and service before launch.

b. Stability - When the final stage is winged, it is easier to make the parallel configuration aerodynamically stable. It is more difficult to arrange the thrust axes of the rocket motors of parallel stages to pass through the over-all center of gravity.

c. Control - Control motors would generally have longer moment arms, and thus be more effective, for the tandem case.

d. Performance - A tandem configuration will probably have less aerodynamic drag. The parallel arrangement allows simultaneous burning of motors from several stages and reduces over-all powerplant weight.

e. Aerodynamic Heating - The tandem stages can be arranged so that the final stage forms the nose of the complete configuration thus producing thicker boundary layers on the aft stages and generally less heating.

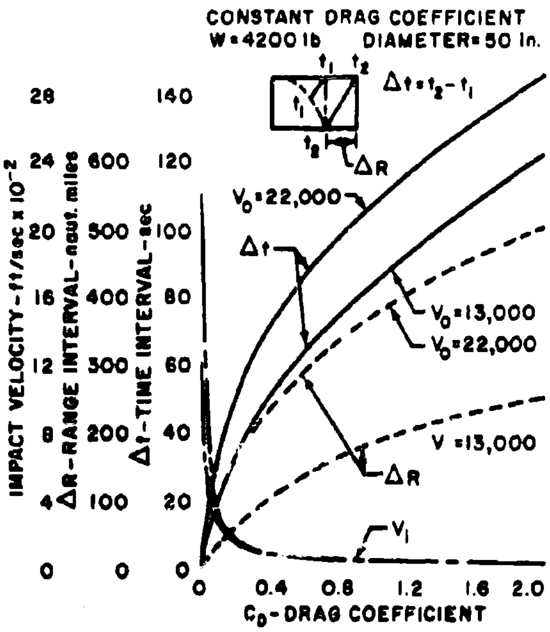


Figure 25. Bomb Zero Lift Trajectories

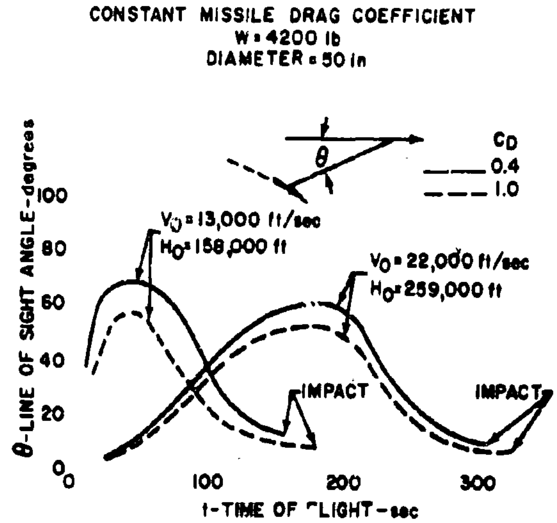


Figure 26. Line of Sight Angle from Carrier to Bomb

f. Separation - The parallel configuration is subject to high interaction forces, large local loads, and the possibility of collison of stages. Tandem separation is essentially instantaneous, there is no aerodynamic interaction on the boosted stage from the booster; however, experience has shown that even the tandem separation may give large angular accelerations to the boosted stage.

8. Bomb Trajectories

A preliminary analysis of the zero lift trajectory of the MX-2276 bomb has been made for two release conditions to illustrate the mechanics of the bomb drop. Trajectories were calculated for initial velocities of 22,000 and 13,000 feet per second, and corresponding initial altitudes of 259,000 and 158,000 feet, respectively. The effects of earth rotation were neglected. A range of drag coefficients for the bomb was used, and it was assumed that drag coefficient was constant over the range of flight velocity. Figure 25 presents the impact velocity, and the time and range interval between the time at which the carrier passes over the target and the time at which bomb impact occurs. It is apparent from these results that the design of the bomb will require a compromise between the desired impact velocity and the time and range interval. Figure 26 presents the line of sight angle from the carrier to the bomb as a function of time from the drop point for two assumed drag coefficients.

9. Test Facilities

As a part of this study program a preliminary survey was conducted of the test facilities, where the problems associated with the design and development of such a hypersonic vehicle could be investigated.

Facilities available, under development, and planned were included in this survey. A summary of these facilities and a general discussion of their usefulness and limitations is contained in Reference 6. Figure 27 is a bar

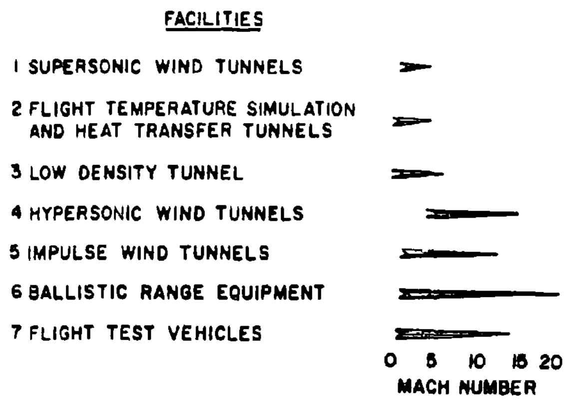


Figure 27. Approximate Mach Number Coverage at Present Test Facilities

graph showing the range of Mach numbers in which the various types of facilities will be useful. It can be generally stated that considerable progress has been made in the development of facilities for testing and evaluating the effects of very high-speed and high-altitude flight. The NACA has flown PARD models at a Mach number of 10 and are now devising means of increasing this speed. A hypersonic test vehicle (HTV) has been developed and flown to provide information in the Mach number range of 10 to 15. The summary of facilities indicates that at the present time there are 13 facilities capable of testing to a Mach number of 10 and 7 facilities which can test at Mach numbers from 10 to 20.

10. Applied Research

Applied Research is concerned with the existence and accuracy of methods for analyzing the force and heat loads to which the bomber will be subjected and with the major flow problems which need be solved in order to provide an adequate set of methods of analysis. One of the primary aims was to point out the "new" or "unconventional" phenomena of hypersonic flight which are not apparent or do not occur at ordinary supersonic speeds and to provide means for assessing their importance. This work is presented in detail in Reference 6.

a. Flow Region

One approach centered about a critical investigation of the foundations of the basic concepts of hypersonic flow theory in an effort to indicate the nature and types of flow patterns which could be expected to result from hypersonic velocities. It was desired to determine both the physical problems and the basic flow equations which could adequately and consistently describe these hypersonic flow problems. To this end, an analysis of the flow about a flat plate flying in the range of speeds and altitudes corresponding to the flight plan was made delineating the nature and the extent of the various flow regions. In attempting to build up an overall picture of the various flow regions, however, details of the flow about a plate for various Mach number-Reynolds number combinations are required, and since there are very few experiments in the high Mach number, low Reynolds number range of interest, these details must at present be supplied by theory. In particular, it was apparent that the boundary layer slip and shock-boundary layer interaction phenomena could be appreciable in parts of the Mach number-Reynolds number range of interest.

b. Shock Boundary Layer Interaction

A survey was made of the various shock-boundary layer interaction theories in order to compile and correlate the information on this phenomenon for use in building up a picture of the various flow regions and for predicting the pressure, shear, and heating parameters on a body in hypersonic flight. There were, however, several different theories predicting different results for some cases of shock-boundary layer interaction, while other cases of interest, e.g., the expansion side of a plate at angle of attack, had not been considered at all. It was necessary, therefore, to go into the shock interaction theory in some detail in order to evaluate these theories. As a result, some improvements were made on existing theory, (flat plate, zero angle) and new theory and numerical results were obtained for the cases of interaction on a flat plate at positive and negative angles of attack. The increases in pressure and skin friction coefficient due to shock interaction for all cases can be correlated in a general but simple and convenient form.

c. High Temperature Phenomena

Another line of approach followed in this study was to make probing investigations into the nature and magnitude of "new" effects arising from the high temperatures which would be realized in the boundary layer and behind strong shocks in hypersonic flight. To this end, studies were made of the emissivity of air (which governs radiative heat transfer), of the effect of dissociation of the air on convective heat transfer, and of such gas effects on shock flow relations. In these cases, determination of even the order of magnitude of an effect involved detailed investigations.

In considering whether or not the intensely hot air in the boundary layer radiates an appreciable amount of heat to the adjacent structure, the first step is to estimate the emissivity of air at temperatures of the order of 10,000°R and low densities. The estimates obtained from air analysis based on the quantum mechanical aspects of kinetic theory show that the order of magnitude of the emissivity of air at the temperatures under consideration is sufficiently high so that radiative heat transfer appears to be an important factor. It remains, however, to solve the flow equations in the boundary layer, including a radiative heat transfer term, in order to determine the exact way in which radiation will qualitatively and quantitatively affect the over-all heat transfer picture. A prerequisite to such a detailed study is a precise knowledge of the emissivity as a function of wave length, pressure, and temperature. A theory has been developed to compute this quantity; it was not, however, possible in this study to carry out the detailed numerical calculations.

Some brief thoughts and remarks on the calculation of the transport properties of dissociated gases are given and an investigation of the effects of assumed equilibrium dissociation of air on the boundary layer characteristics is reported. The results of the latter study show that skin friction and heat transfer are essentially unaffected by dissociation so long as both the stream and body temperatures are below dissociation values. It appears that a similar result holds in the stagnation region of a blunt-nosed body.

A recently completed program (at Bell Aircraft Corporation) to compute basic tables of flow parameters for both shock flow and isentropic flow, incorporating real gas effects up to dissociation temperatures, is discussed. Since the gas flow tables are basic to any numerical analysis of the flow, it was important to determine how the actual behavior of air at high temperatures differs from that described by the standard ideal gas tables, and thus the real gas flow tables were needed as a standard comparison. A numerical comparison at typical flow conditions of interest was made. It is of particular interest to the performance and viscous heating analysis that real gas effects on the flow adjacent to surfaces at reasonably low angles of attack, e.g., the Stage III lower surfaces, are small.

d. Transpiration Cooling

A survey and evaluation of the existing theoretical and experimental literature on the aerodynamic aspects of transpiration cooling was made seeking a basis for the calculation of coolant requirements. Practically all of the theoretical studies examined were restricted to supersonic flow at low Mach number, generally less than 3. Hence, it was deemed necessary to develop new solutions to the equations of the compressible laminar boundary layer including the effects of transpiration cooling for Mach numbers up to 20, and to carry out the calculations for the Mach number and altitude range of interest. As the end result, an approximate theoretical method was developed for computing the rate of mass flow injection of coolant required to keep a surface at a given (arbitrary) temperature under given initial free stream conditions.

The theory applies to a laminar boundary layer which is most pertinent to the present case since transpiration cooling will most probably be confined to the areas near the leading edges where the flow is expected to be laminar.

It is probable that the injection of the relatively small amounts of coolant into the boundary layer will not destabilize the laminar flow. In the strict sense, air must be used as the coolant because the theory is based on homogenous boundary layer considerations for which the coolant and boundary layer flows must be of the same gas; but it is believed that a dissimilar coolant can be handled with sufficient accuracy through a simple extension of the present theory. A set of exemplary design charts were calculated using air as the coolant.

e. Hypersonic Inviscid Flow Theory

The detailed investigation of shock-interaction theory led to a thorough study of hypersonic inviscid flow theory, since results of the latter have an important influence on the results of interaction theory based on the two-layer model. Furthermore, the so-called "Newtonian flow" approximation of inviscid hypersonic flow is an important practical method for determining pressure distributions on a body where viscous effects do not predominate, and, hence, the applicability and limits of this approximation were given consideration. Some contributions to an understanding of an improvement in accuracy of the approximate hypersonic inviscid theory were made.

f. Boundary Layer Transition

In any practical computation of friction drag or aerodynamic heating, the state of the boundary layer must first be assumed, i.e., a knowledge of the transition point is required. Unfortunately, the present state of reliable knowledge on this subject leaves much to be desired. The effect and the importance of the many variables which could effect transition and the mechanism of transition itself is not yet understood; hence, the assumptions of theory are incomplete and experiments are not fully controlled. The best that can be done at the present time is to assume a transition Reynolds number based on the trends exhibited by available wind tunnel and flight test data. In the original work and the present study a transition Reynolds number of 2.8×10^6 at all Mach numbers was assumed. This appears to have been conservatively low judging from the trends exhibited by the test data available, and from discussions with several experimenters during our visits to other research agencies.

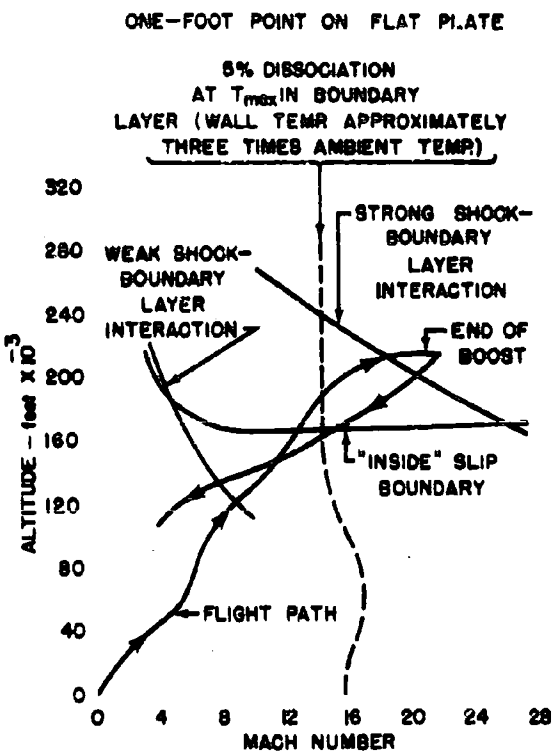
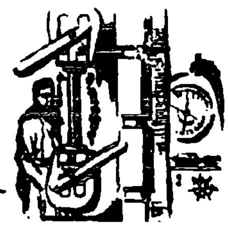


Figure 28. Flight Regions

Based on the preceding work, Figure 28 illustrates some of the flow phenomena which must be considered. The flight path is shown superimposed on a plot of boundary layer interaction and slip flow boundaries of fluid flow as they apply to a point one foot from the leading edge of a flat plate. In addition, hypersonic flight, through shock waves or viscous forces, produces air flow temperatures sufficiently high to cause deviations from normal air properties, i.e., real gas effects, and in certain areas also dissociation of the air. A curve showing where approximately 5 percent equilibrium dissociation could occur in the boundary layer is also shown.

C.  STRUCTURES

1. General

The problems of structural design for the MX-2276 weapon system are complicated by the high temperatures encountered during hypersonic flight. It was found during the analysis that the structure could be divided into two separate types:

a. Primary structure - This is the structure which carries the loads and comprises approximately 75 percent of the structural weight. It is subjected to moderate heat fluxes and, if no cooling is provided, maximum temperatures will not exceed 1700°F. Its weight is very critical.

b. Secondary structure - This is the lightly loaded structure for such areas as leading edges, fuselage nose, and control surfaces. In these areas localized equilibrium temperatures may go as high as 5000-6000°F and feasibility is the prime structural consideration.

In this section materials, structural configurations, insulating methods, and cooling methods have been evaluated for both types of structure. The results of preliminary tests of several cooling schemes are also included. Reference 7 reports the structural work in detail.

2. Criteria and Loads

The criteria and loads effort has been directed towards two goals:

(1) Exploration and definition by criteria of the new environment and flight forces to be encountered.

(2) Presentation of sufficient load conditions to provide an indication of configurations yielding minimum loads for use in future studies.

a. Structural Environment

The flight path originally estimated for this weapon system is shown in Figure 9. While the high altitude atmospheric properties have been established, knowledge of upper atmosphere disturbances is still limited. Gusts at moderate altitudes (0 - 50,000 feet) are well defined by current specifications. However, above these altitudes, they are not yet defined. Winds may produce the same effects as gusts when there are wind velocity and direction differences in strata through which the vehicle passes at high velocities. Using wind data and extending conditions already specified, criteria for high altitude gusts have been established. The gust velocities at higher altitudes are reduced by the ratio of $\sigma^{1/2}$ at the altitude under consideration to $\sigma^{1/2}$ at 35,000 feet. Such a procedure results in a 10-foot per second equivalent gust at 105,000 feet. Current investigations have not shown equivalent gusts or winds of greater magnitude at altitudes over 100,000 feet, so therefore the 10-foot per second gust velocity is retained to 260,000 feet. Using these criteria, methods of performing gust calculations have been adapted for use with this airplane.

b. Flight Loads

The effects of both centrifugal and Coriolis accelerations have been considered, methods for computing them developed, and the magnitudes computed. Criteria such as MIL-S-5700, MIL-S-8629 usually define the load factors to which an aircraft will be designed. This is done graphically by a "V-n" diagram for each design altitude. It is not possible to categorize the MX-2276 aircraft in this manner because of its advanced design. Therefore, a new diagram was evolved to replace the conventional V-n diagram usually associated with structural flight loads work. Based upon the sequential arrangement of load conditions the "t-n" diagram (time versus load factor) was chosen as the most descriptive of the load environment. This type of diagram, together with the definitive flight path of this weapon system, permits a comprehensive picture of the load environment to be presented on a single chart including the wide range of altitude and speeds encountered.

The principal parameters influencing load factors during powered flight are:

(1) Decreasing weight due to fuel consumption and stage separation.

(2) Increasing axial acceleration and velocity because of weight decrease with approximately constant thrust.

(3) Travel through the "gusty" portion of the atmosphere (h = 0 to 100,000 feet).

(4) Decreasing dynamic pressure over 100,000 feet and hence decreasing capability of realizing large aerodynamic loads.

(5) Low transverse flight path accelerations until the latter part of Stage III flight.

Preliminary estimates of the design load factors have been established and are presented in their proper relations in Figures 29 and 30 for powered flight and for glide flight.

Throughout the powered flight the varying weight causes large changes in axial accelerations. Figure 31 shows the rapid increase in axial acceleration as fuel is consumed, and the abrupt change at stage separation where a lesser thrust is initiated for the reduced size vehicle.

The probable magnitude of the transient dynamic maneuver load factors which can be tolerated while maintaining the flight path are listed below.

Aircraft Axis	Translational Load Factors	Angular Acceleration Radian/sec^2
Longitudinal	$0.1n_X$	±1.0
Lateral	±0.5	±1.2 $\ddot{\theta}$
Vertical	$0.2n_N$	±1.0

These factors must be small if the desired flight path is to be realized with the fuel available. The vertical factors are especially important in this respect and have been limited upon this basis. The lateral and longitudinal factors are not as critical and have been selected upon the basis of past experience.

c. Landing Conditions

Landing conditions, although of secondary interest in the study program, have been given some consideration. This consideration has centered around the initial configuration. Since stowage space and weight are prime considerations for the landing gear, the configuration limitations are considered first. The thin but deeply insulated wing precludes any gear or outrigger installations outside the fuselage. With space at a premium inside the fuselage and possible internal temperatures over 250°F, some form of flat retractable steel skid is most appropriate. Omission of any outriggers necessitates a side-by-side gear for stability. The aft center of gravity location and a wide rear body is suitable for such an arrangement.

Stability during a high-speed, nose-down landing requires a nose wheel rather than a skid. If weight and size of the nose gear are to be a minimum and the forward fuselage designed by flight loads only, the nose gear loads must be severely limited. This means a low attitude (high-speed) landing, a very efficient oleo strut, and a cg as close as possible to the aft or main gear. Rapid decelerations when the main skids touch down, cause hard nose gear impacts which can be reduced with the preceding precautions.

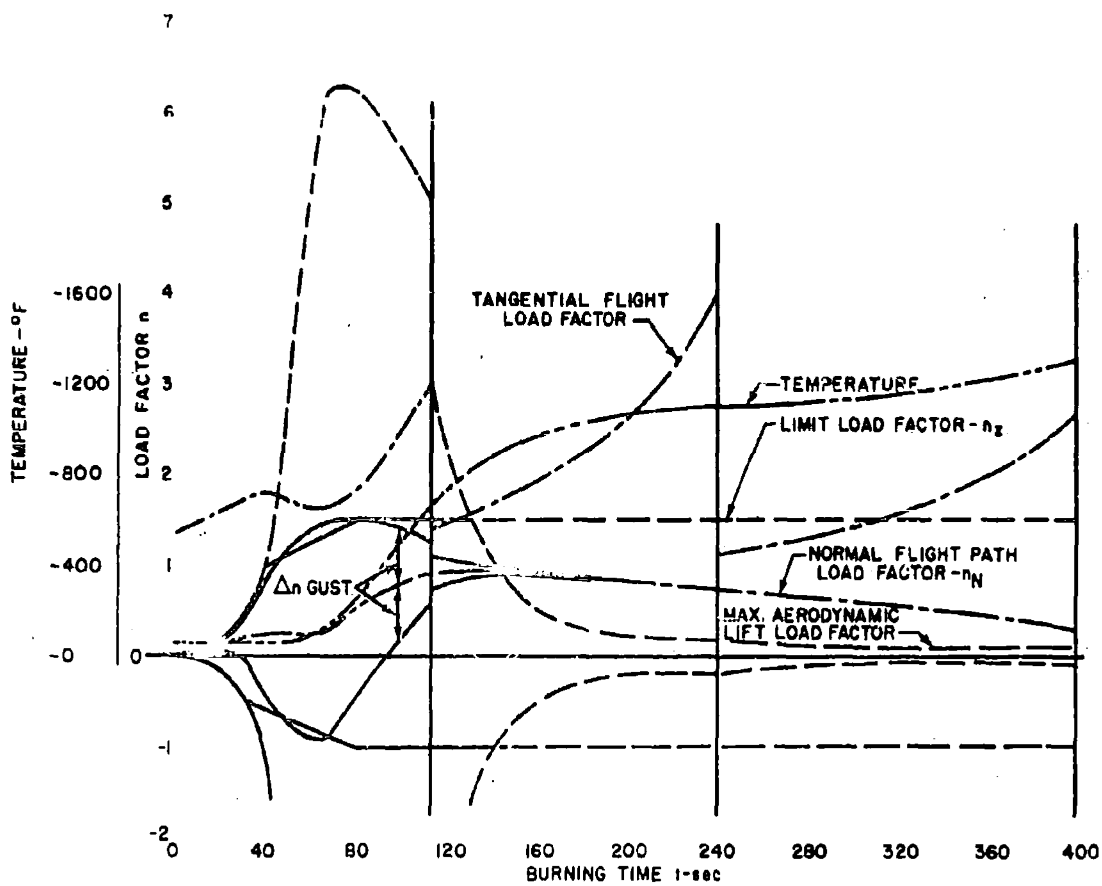


Figure 29. Typical T-n Diagram for Ascent

In addition to these limitations, a very important landing load parameter, the descent velocity, must be selected on the basis of wing area, landing speed, and lift efficiency of the wing. A limit descent velocity of the order of 8 feet per second as specified for heavy bombardment aircraft is appropriate. Wing design requirements for the long high-speed glide path necessitates a wing design that will afford a reasonably low wing loading of about 30 pounds per square foot. With this wing loading, it is reasonable to expect an 8-foot per second descent velocity can be realized, particularly for high-speed landings.

d. Ground Support

Ground handling equipment for the transport, erection, and assembly of the flight articles and for fueling and servicing the assembled flight vehicles, have been considered. The basic philosophy used requires that no ground handling condition should result in a structural weight increase in the flight article. In order to make the ground handling loads compatible with flight loads, transverse loads must be distributed over the airframe structure and axial reaction must be concentrated at the rocket engine gimbals.

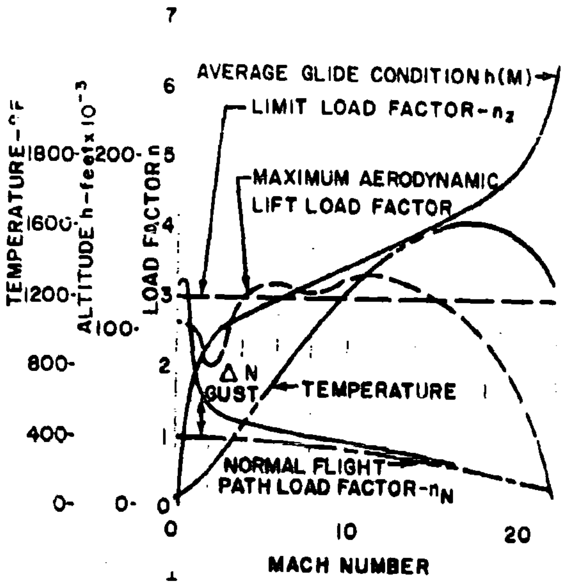


Figure 30. Typical M-n Diagram for Glide Flight Stage III

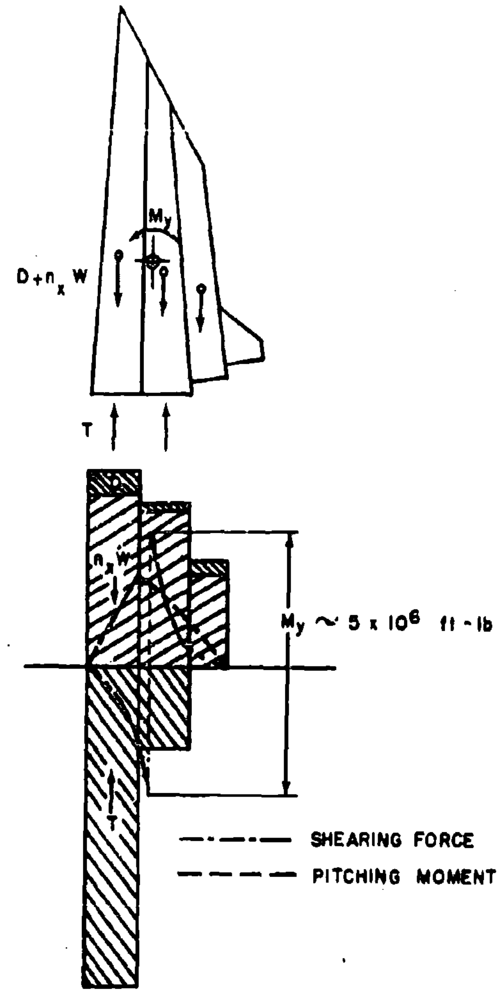


Figure 31. Typical Axial Acceleration Forces and Moments

Longitudinal loads pose the most critical ground handling condition for the assembled and erected vehicle. The ground reactions must support the empty or loaded aircraft either at the rocket gimbals or on some adjacent structure. In addition to extending around the rocket engines such a ground support must pull clear of the engines at take-off and withstand the full rocket blast for a matter of 10 seconds while under load. By adding weight in the form of built-in support structure extending to the outside of the body this high temperature condition can be relieved.

3. Structural Materials

Early in the study a survey of structural materials was made with the following objectives.

a. To select materials suitable for the various elements of the airframe, considering the type of construction under investigation and the associated temperature levels or heat fluxes.

b. To compare material efficiencies over a range of temperatures and under various loading conditions so that relative weights of elevated temperature structures could be investigated.

c. To gather detailed mechanical and physical properties of the materials selected for use, so that information is available for the design studies.

The materials were divided into two categories as previously mentioned. For the primary structure, equilibrium temperatures will not exceed 1700°F, which is approximately the upper limit of presently available metallic materials. Since any degree of insulation or cooling may be applied to the primary structure, its operating temperatures may be established at any value below 1700°F, whichever leads to minimum weights. Thus, it was necessary to consider, for the primary structure, all materials which will operate below 1700°F and this includes all available structural materials except the ceramics and cermets. The term ceramic is intended here to include not only true ceramics, which are basically oxides, but also the nitrides, carbides, etc.

For the secondary structure, such as leading edges, fuselage nose, and control surfaces, equilibrium temperatures in some localized areas go as high as 5000-6000°F. It is obvious that such areas must be cooled, but because flight times are long (70 to 80 minutes), it can be expected that "passive" structures, in which no attempt is made to protect it from the heat, would be lighter than a continuous supply of expendable coolant. Thus, for the design of secondary structures, all of the "newer" materials, the ceramics, graphite, and molybdenum were considered, and some thought was given to the potential of such materials as tungsten, beryllium, and even some of the precious metals, such as platinum.

The results of the survey are presented in Reference 7, where they have been discussed and are summarized in a series of tables. Detailed properties are presented for Inconel X and Haynes Alloy No. 25 which are the high-temperature alloys selected for potential use in the third stage airframe. Figure 32, which illustrates the tensile properties of Inconel X as a function of temperatures, is presented to show a typical variation of structural properties with temperature.

4. Heat Protection

Section IV-B-4 contains a brief discussion of the heat balance on the outer surface of the airframe. In that section it is stated that the heat transfer to the inside of the airframe may often be neglected insofar as calculation of equilibrium skin temperature is concerned. From a structural standpoint, however, this transfer is extremely important. As the skin temperature rises to the equilibrium value, the temperature of the entire airframe will continue to approach this equilibrium value unless insulation or cooling is provided to separate it from the heat source.

If cooling is provided, some or all of the heat entering the structure can be introduced by various means into the coolant. The coolant may then be expended overboard or circulated through a heat exchanger. If a heat exchanger is used, it can dissipate heat only by radiation since the hot boundary layer completely surrounds the airplane. In any case, the sum of radiation heat loss and the heat loss through expended coolant will equal the convective heat input. With an expendable coolant the weight of such coolant can be minimized by operating the surface at the temperature limit of the material so that the maximum heat is dissipated by radiation. This then becomes a compromise, or an optimization, between the quantity of coolant and the efficiency of the structure at high temperature.

Interposing a layer of low conductivity insulation material between the cooled structure and the hot boundary layer is a practical means of controlling the proportions of heat absorbed by the coolant and that dissipated by radiation. The selection of the proper insulation thickness introduces another variable into this optimization.

The optimization of the heat protection system may also include the condition where the heat taken into the structure is so small that it can be absorbed entirely by the heat capacity of the structural material. In the work which follows, such an arrangement is termed an "Insulated Structure" while the combination of insulation and coolant described previously is called an "Insulated and Cooled Structure".

The aerodynamic heating effects are relatively moderate over the major portion of the airframe and especially in the areas of primary structure, making it practical to consider heat

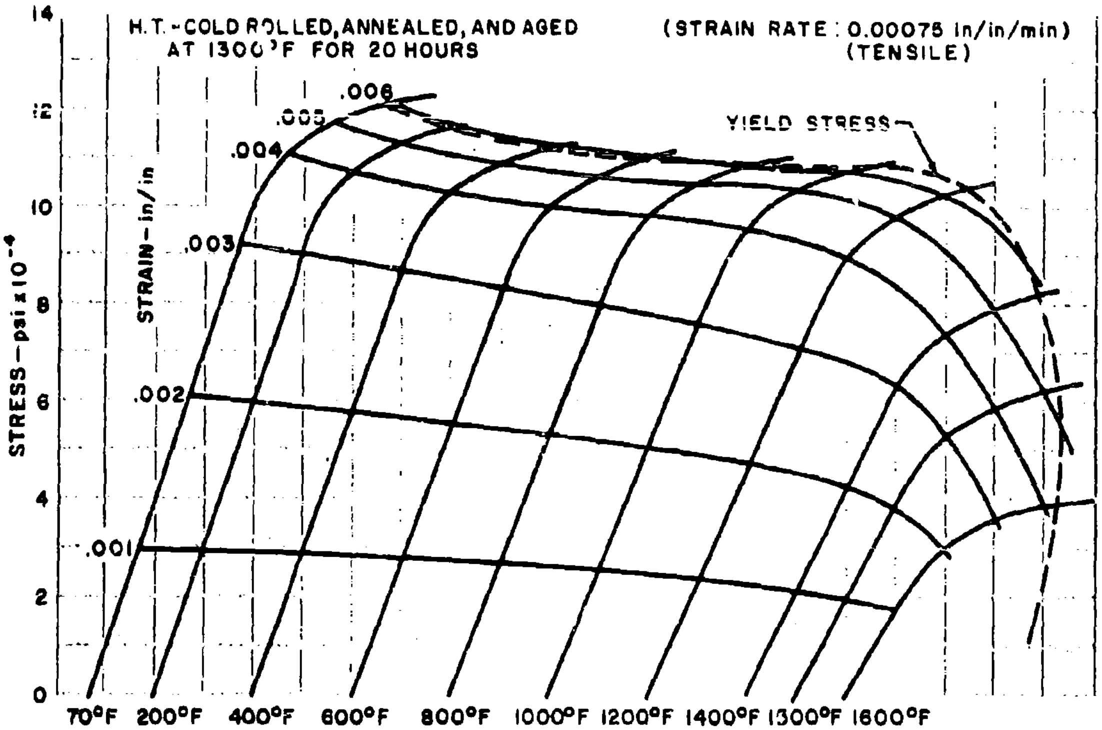


Figure 32. Stress-Strain Diagram for Inconel X at Elevated Temperatures

protection. Protection of the primary structure by insulation, by cooling, or by a combination thereof results in the conditions that full equilibrium temperature is never attained by the primary structure. Advantages that may be expected from the protection of the primary structure include:

(1) Reduction of structural temperature to the point where materials with a useful and reliable load carrying capacity can be used.

(2) Reduction of structural temperatures still further to allow the use of materials of higher strength-weight ratio.

(3) Elimination of thermal stresses so that the most efficient type of internal structure can be used.

A short summary of the materials, principles, and systems used to obtain these advantages, together with their application to the airframe of the bomber follows. Some of these systems have been developed to the stage of preliminary testing, and the results of these tests are included.

Heat protection systems for the high temperature secondary structure have been studied in a general manner with sufficient numerical support to indicate arrangements which deserve more detailed study.

a. Insulation and Cooling Materials

(1) Low Conductivity Insulation Materials

The densities, conductivities, useful temperature ranges, and other pertinent data for a number of insulating materials have been collected and presented in Table III. Vari-

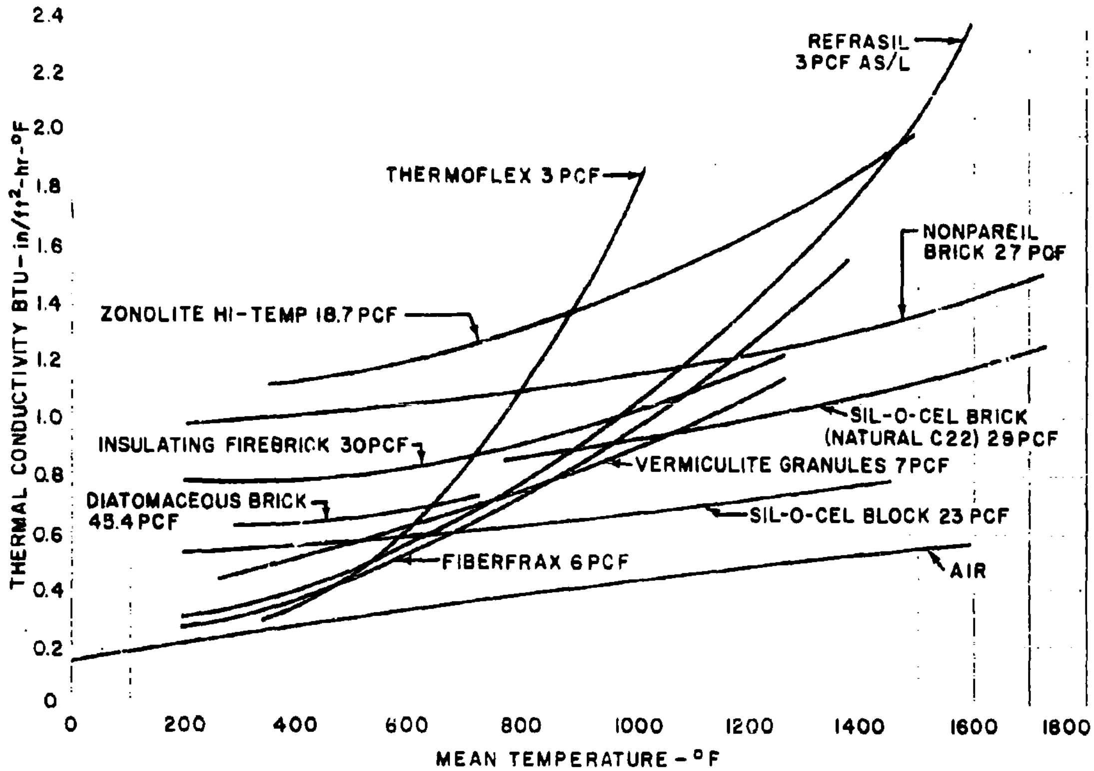


Figure 33. Thermal Conductivities of Insulators

ation of conductivity with temperature is presented in Figures 33 and 34, while Figure 35 is a plot of the products of conductivity and density, against mean temperature. This latter curve, therefore, indicates the weight of various materials for a required value of thermal resistance. Table III is concerned only with materials which show a high efficiency on the basis of weight.

It will be noted from table III that insulating materials fall into three broad groups with respect to the function of insulating structure:

(a) Relatively hard brick-type materials capable of forming the external surface of the aircraft, able to support aerodynamic pressures and to resist abrasion from the airstream.

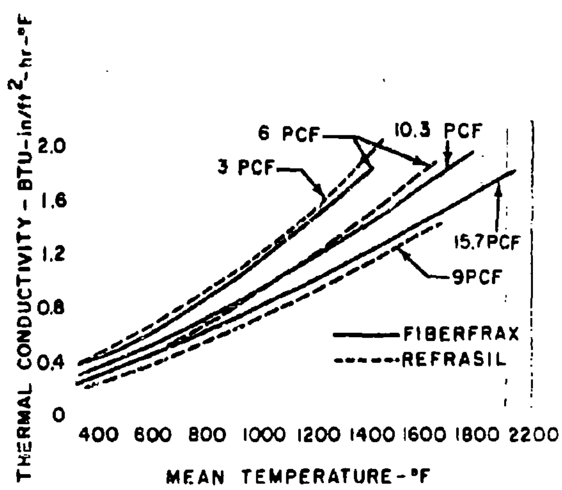


Figure 34. Thermal Conductivity of Fibrous Insulators

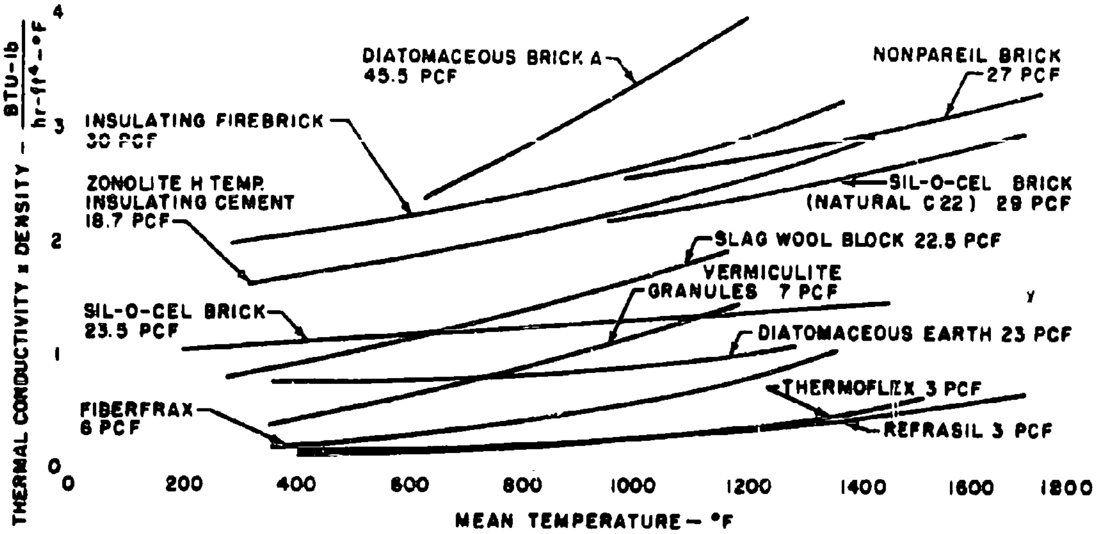


Figure 35. Relative Weight of Insulation

(b) Loose, fibrous or powdered materials which require an external cover to provide the aerodynamic contour and to carry pressure loads.

(c) Gases, chiefly air, which require an external cover to provide the aerodynamic contour and to carry pressure loads and also a means of minimizing heat transmission by radiation.

Ultimately, therefore, the comparison of efficiencies for insulating materials must include the weight of the protective outer wall, since this will differ greatly between the groups mentioned above.

(2) Material for Radiation Barriers

The important requirement of a material for radiation barriers is that it have a surface condition which is highly reflective to radiant energy in the infrared band. Generally, this requires a highly polished metallic surface, but, unfortunately, wide deviations from the optimum result from a slight deterioration of the surface. This deterioration is generally, though not exclusively, the result of oxidation, and it is usually greatly accelerated by the high-temperature conditions under which most of the radiation foils will necessarily operate.

The other requirement for radiation barrier material is availability in very small thicknesses, since this will govern the weight of the barrier assembly. Mechanical strength is not important except to the extent that the foils should support themselves over a reasonable distance. The foils will be separated, at intervals, by spacers which prevent the foils from sagging and touching. Since the spacers form a conduction path of relatively low resistance through the air space, it is clear that an optimum combination exists between the thickness of the foils and the number of spacers in which the maximum insulation value is achieved for each pound of weight.

Figure 36 shows emissivity values for a number of pure and alloyed materials, plotted against temperature. It can be seen that the low values of emissivity, and particularly the ability to retain these low values under high-temperature conditions, are best realized by certain of the precious metals. This conclusion led to the study of plating and cladding as a means of using extremely thin sheets of pre-

TABLE III. PROPERTIES OF INSULATING MATERIALS

Material	K at 1000°F	Density	Specific Heat	Diffusivity	Temperature Limit	Mechanical Strength	Availability	Remarks
	BTU-in./ft^2-hr°F	lb/ft^3	BTU/lb-°F	in.^{2}/sec	°F			
Refrasil (Fibrous)	1.2	3	0.20	0.006670	1800	None	Good	Optimum Weight
	0.93	6	0.20	0.002580	1800	None	Good	
	0.75	9	0.20	0.001390	1800	None	Good	
Fiberfrax (Fibrous)	1.18	6	0.20	0.003280	2300	None	Good	Does Not Melt to 3000°F
	0.94	10.3	0.20	0.001520	2300	None	Good	
Sil-O-Cell (Soft Insulating) (Fire Brick)	0.67	23.5	0.22	0.000434	1800	50 psi Comp. at 1600°F		Mechanical Strength only to support own weight. No resistance to abrasion
Beryllium Oxide (Hard Ceramic)	13.2	195	0.497	0.000454				
Alite AE'212	130	214	0.190	0.010700	2900	26,500 psi Tension to 2200°F	Good	A popular and readily available ceramic material with good structural properties.
Air	0.306							

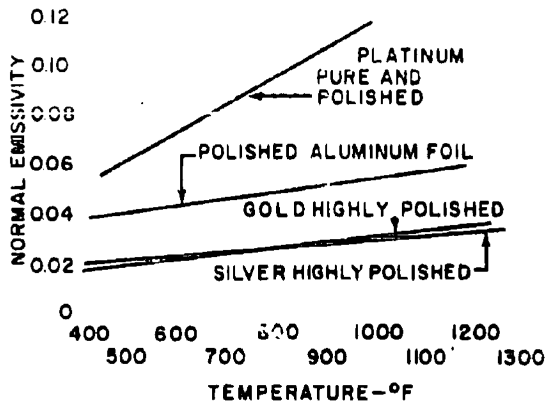


Figure 36. Emissivity of Metal Surfaces

cious metal. Tests of plated materials indicate that the plating was completely removed by oxidation and diffusion into the base metal when exposed to temperatures of 1600°F in air. Exposure of gold foil, 0.002-inch thick, to the same conditions produced no visible deterioration of the polished surface. From these tests it was concluded that only the pure metal foils should be considered.

Tests of silver, platinum, and gold foil were conducted. The silver tarnished to the extent that it was unsatisfactory. The platinum, although satisfactory, was eliminated because of its excessive cost compared to gold which was also satisfactory. Gold foil has therefore been selected for further consideration.

(3) Cooling Materials

Cooling materials will generally be selected on the basis of cooling capacity per unit weight, with secondary consideration being cooling capacity per unit volume, thermal conductivity, boiling point, etc. The heat capacity of a coolant will include that which produces a temperature rise (specific heat), and that which produces change of state (latent heat).

Materials having good heat capacity fall naturally into two groups in which the boiling temperature is the basis for division. Suitable materials with a low boiling point include hydrogen, helium, oxygen, nitrogen, and water, and with the exception of water, these materials rely upon temperature rise for heat absorption. Since these materials have boiling points well below 0°F they will normally be used in the gaseous form with corresponding large volumetric flows and small heat transfer coefficients. Water boils at a convenient temperature for structural use and has a large value of latent heat which can be readily used.

The other group of materials having large heat capacity are the light metals. Heat capacities of these materials are greater than that of water, but the boiling points which must be achieved to make full use of this capacity are very high. The metals have the added advantages of good conductivity, good heat transfer coefficients, and a large heat capacity per unit volume, but there is the problem of maintaining them in the liquid form.

Table IV presents a summary of the important properties of materials suitable for structural cooling, and Figures 37 and 38 show heat capacities on a comparative basis, plotted against the temperature at which the coolant leaves the hot surface. Since boiling temperatures are controlled by pressure, Figure 38 shows heat capacities over a range of pressures.

b. Insulation and Cooling Principles

(1) Use of Insulation

Insulation applied to airframe structures has three functions:

(a) Protection of structure from high temperatures.

(b) Reduction of heat flux into a cooling system.

(c) Protection of crew, equipment, etc.

Items (a) and (b) are discussed in this section and (c) in Section IV-A. In general,

TABLE IV. PROPERTIES OF COOLING FLUIDS

Material	Melting Point	Boiling Point		Latent Heat of Fusion BTU/lb	Latent Heat of Vaporization BTU/lb	Specific Heat of Liquid	Density of Liquid lb/in.^{3}	Thermal Conductivity of Liquid BTU-in./ft-^{2}hr-°F
		Temperature	Pressure					
	°F	°F	psi					
Lithium	367	1373	0.20	59.1	8380	1.22	0.0182	247
		2113	3.90					
		2403	15.00					
Aluminum	1220	2709	0.20	172	4070	0.26	0.086	640
		3351	3.90					
		3735	15.00					
Magnesium	1204	1296	0.20	160	2420	0.32	0.057	
		1773	3.90					
		2017	15.00					
Sodium	208	1018	0.20	49	1810	0.33	0.033	
		1386	3.90					600
		1621	15.00					
Water	32	101.7	1.00		1036.3	1.00	0.036	4.05
		162.2	5.00		1001.0			
		193.2	10.00		982.1			
		213.0	15.00		969.7			
		281.0	50.00		924.0			
		358.4	150.00		863.6			

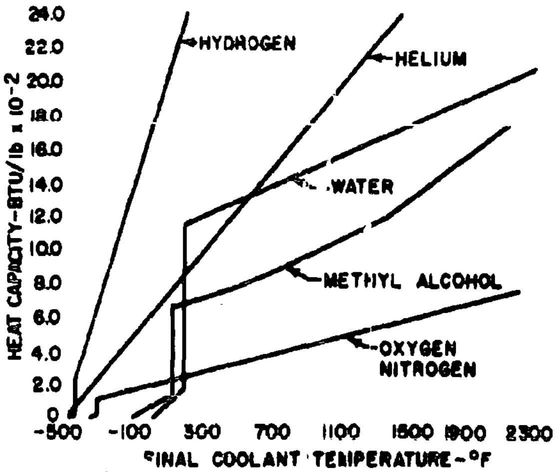


Figure 37. Heat Capacity of Cooling Fluids

satisfactory accomplishment of the first two functions, results in satisfaction of the third.

The heat transmitted between two points by conduction is proportional to the temperature difference between the two reference stations, while the heat transmitted by radiation is proportional to the difference of the fourth powers of the absolute temperatures. If the heat problem of the bomber is visualized as a cool structural skin, separated from an outer skin or covering which may be at an equilibrium temperature of 1700°F, it is apparent that the intervening space should be filled with material very opaque to radiation. Such materials, of course, are necessarily dense, and therefore heavy, and even if materials of low conductivity are used, such as diatomaceous brick, an appreciable amount of conduction is introduced. These considerations lead to the study of an "insulation" consisting of a series of parallel metal foils each with highly reflective surfaces and acting as a barrier to radiation. The air between these foils and the necessary spacing materials result in some conduction.

(2) Theoretical Expressions for Insulation Requirements

To evaluate analytically the amount of insulating material required to maintain a given structural temperature, after exposure to the high boundary layer temperatures for a specified time, two simple expressions have been developed based on a number of practical assumptions. These expressions correspond to the two extreme types of heat transmission:

(a) Pure conduction with constant conductivity.

(b) Pure radiation.

The assumptions on which this work is based are as follows:

(a) No heat capacity in the insulation.

(b) Infinite conductivity through the structural skin, applicable to thin shell structures.

(c) Constant external temperature at the outer layers of the insulation.

(d) Constant conductivity with temperature, low conduction materials.

(e) Constant emissivity with temperature, of all surfaces of radiation barriers.

(f) Neglect of conduction paths through the radiation foils due to spacing material, connections, etc.

Note that assumption (c) permits inclusion of the heat transfer by forced convection from the boundary layer to the outer covering, and also the radiation from the covering back into the boundary layer. This is normally difficult to include in a complete solution because of the fourth power radiation, but with insulation it can be assumed that heat conducted into the structure is small enough that the outer covering, or the outer layers of insulation, reach equilibrium temperatures instantaneously.

(3) Conduction and Radiation Combined

As noted previously, the use of radiation "barriers" will require consideration

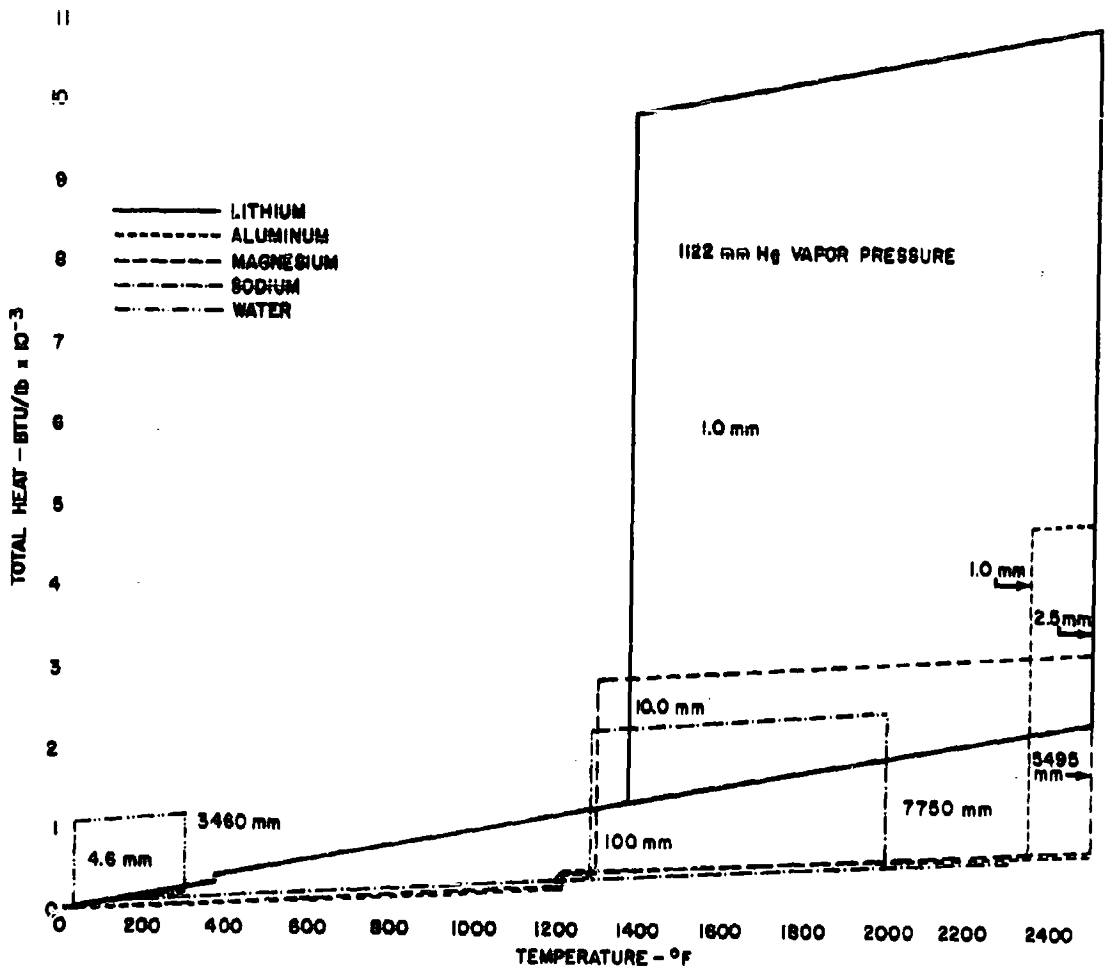


Figure 38. Total Heats of Metal Coolants

of the conductivity of the air between the foils. It has been determined that the conductivity of air is not substantially reduced by reduction in pressure until the molecular mean free path becomes of the same order as the dimensions of the air space. Thus the high altitudes at which the MX-2276 flies are not likely to produce any significant reductions of air conduction, so that this term must be included in the evaluation of radiation barriers.

(4) Cooling

The division of the structure into the load carrying primary type subjected to moderate heat fluxes and the lightly loaded secondary type, some of which are subjected to high heat fluxes (e.g., the leading edge), has the following significance from a cooling standpoint.

For the primary type structure, the equilibrium temperatures are within the range of structural materials. Thus, although it may not be desirable from a weight standpoint to use the high-density, high-temperature materials for the heavily loaded primary structure, it may be profitable to use such material in a light, protective, outer wall. One of the highly efficient fibrous insulators could then be used between the outer wall and the primary structure, to reduce heat flux into the cooling system.

For the secondary type of structure, equilibrium temperatures are so high that cooling must be used at the exposed surface, regardless of the weight involved. The problem in these regions is to study various methods of cooling so weight may be minimized.

(5) Cooling Combined with Insulation

The properties of coolants shown in Figures 37 and 38 serve to demonstrate the overwhelming superiority of water as a cooling medium for areas that can also be protected by insulation. The liquid metals require very high temperatures before a heat capacity can be realized that is comparable with that of water. Oxygen, nitrogen, and methyl alcohol are significantly inferior. Helium shows a weight superiority over water except in the most useful temperature range (200 to 700°F). The practical difficulties and the hazards of using hydrogen are obvious, and it has not been considered during the present study. The potential weight saving by the use of hydrogen is very significant, however.

In addition to its general convenience and efficiency, water has two other important advantages as a coolant for primary structure:

(a) The boiling temperature of water permits the use of aluminum structure with resulting high structural efficiencies.

(b) The use of the change of state of water from liquid to vapor as the principal source of heat absorption provides a guarantee of uniform temperatures. This is particularly important with a circulatory type of system, since it insures that coolant temperatures are constant regardless of the length of the circulation path or of the variation of flux along the path, until all the water is boiled.

A system of heat protection which uses insulation and cooling combined will obviously require a system for distributing coolant to the surface, and, if the efficient fibrous insulators are used, it will require an external protecting wall to carry aerodynamic forces. The weight of this outer wall will vary slightly depending on the thickness of insulation, since the height of the attachment structure will be affected. Similarly, the weight of cooling system will be affected by the quantity of coolant to be used.

For a first approximation it will be assumed that outer wall and cooling system weights are independent of variation in the proportions of insulation and cooling. Then the total weight can be optimized by minimizing the sum of insulation and coolant weights.

(6) Low Capacity Cooling Systems Using Water

In conjunction with the combined use of insulation and water cooling for the protection of primary structure, a number of schemes have been devised for supplying the required amount of coolant at all points on the structure.

(a) Circulation System

In this system the water is pumped through suitable passages in the structure, a small part is converted to steam, and the mixture of water and steam is taken back to a central separator so that the remaining water can be recirculated.

Advantages of the circulation system are: a guarantee of coolant, where required, and in the necessary quantity; the ability to deal with a wide tolerance in heat flux without failure of the system; no fundamental research required since the mechanics of water boiling have been adequately studied and much empirical information is available on the design of water cooling and boiling systems.

Disadvantages of the system are: weight and the undesirability of using a mechanical system.

(b) Distribution System

In this system water is fed to the areas of skin at the rate at which it is evaporated, no return lines are required. Two arrangements have been devised for this system. In the first, water is fed to many points on the inner surface of the structural shell, there being approximately one such point to each square foot of surface. The water is then absorbed and spread uniformly over the local area around each feed point, by a thin "wick" material covering the entire inner surface of the structure. The success of the system depends on the ability of the wick material to transport water by capillary action over the local area around a feed point; and also on the development of a satisfactory metering device to control the flow. Two wick materials have been considered during the present study: an all-wool felt, approximately 1/8 inch in thickness, and a woven fiberglass material of brand name "Refrasil".

The second arrangement uses a layer of material adjacent to the skin which contains the total quantity of water that is used during the flight. The material used is "Vermiculite", an exploded mica, which has the ability to hold an amount of water equal to between three and five times it own weight. Steam charging of the "Vermiculite" is visualized as a convenient means of filling it with the necessary quantity of water before take-off.

The advantages of the first system are the reduced coolant flow and the elimination of return lines. The second system has these same advantages plus a lighter distribution system and since the distribution occurs prior to take-off, the system is "fail safe".

The disadvantages of these systems are the additional weight of lines, ducts, manifold, and residual water; the fact that the water may not be diffused through the wicking properly; and the presence of water and steam throughout the airframe may prove unsatisfactory.

(c) Compound System

In this system the water sink is confined to a tank, and air is circulated through a closed system as a means of transporting heat from the structure to the water. The air is pumped through ducts covering the entire structural surface, and through a heat exchanger within the water tank. After being cooled by giving up heat to the water, the air would be recirculated.

This system has the advantage of not requiring water throughout the structure. The disadvantages are large volumes of air required with the attendant large ducts and pumps. This volume can be decreased through the use of compressors and turbines, at the expense of increased weight and complexity.

(7) High Capacity Cooling Systems

In the areas where equilibrium temperatures are too high for the use of insulation, cooling systems capable of handling much higher values of heat flux than the system just described will be required. This difference will be a factor of at least 100, so that it may be safely assumed that water-soaking devices will be totally inadequate. However, the possibility of developing very high temperatures allows consideration of coolants such as the liquid metals, in addition to water. Such a system, working at high temperatures, takes maximum advantage of heat dissipation by radiation, and at the same time these areas are generally lightly loaded secondary structure so that the use of high-temperature, high-density structural materials may be acceptable.

Cooling systems for areas of high heat flux divide naturally into those with and without an expendable coolant. If the coolant is not expendable, it must be circulated to areas of the airframe with high temperatures and then to those where the natural equilibrium temperatures are lower than the temperature limit of available materials. The temperatures of those areas can then be raised by the coolant, with a resulting dissipation of radiant energy. The study of cooling systems that function without an expendable coolant thus becomes a study of means of transporting heat from the areas of very high flux, to areas where heat can be lost by radiation.

Systems using an expendable coolant as a means of cooling reduce essentially to the choice of coolant on the basis of heat capacity, working temperatures, etc., and of means of distributing, circulating, and expending this coolant.

(a) High Capacity Systems - Expendable Coolant

Two methods are available for using an expendable material as a structural coolant: (a) transpiration cooling, and (b) convective cooling. Since transpiration cooling involves interaction between the coolant and the boundary layer, this is treated in Section IV-B. Figures 37 and 38 show the properties of various possible coolants and show the superiority of liquid metals as expendable coolants. Estimated coolant weights for liquid metal expendable systems apply to the leading edge of the bomber using aluminum, magnesium, and sodium as coolants. Aluminum has the highest latent heat of vaporization but requires operating temperatures higher than those at which any known structural material is useful. Magnesium has half the heat of vaporization but operates at temperatures within the useful range of existing high-density superalloys. Sodium has a smaller latent heat, but is useful at even lower temperatures. Water has a cooling capacity of about one-half that of sodium, but has the obvious advantage of convenience and safety.

(b) High Capacity Cooling System - Nonexpendable

Two methods are also available for use of this type of system. In the first, coolant may be pumped through the system absorbing heat at the hot area and losing heat on the cooling area. In the second a high-temperature refrigeration cycle is necessary. With

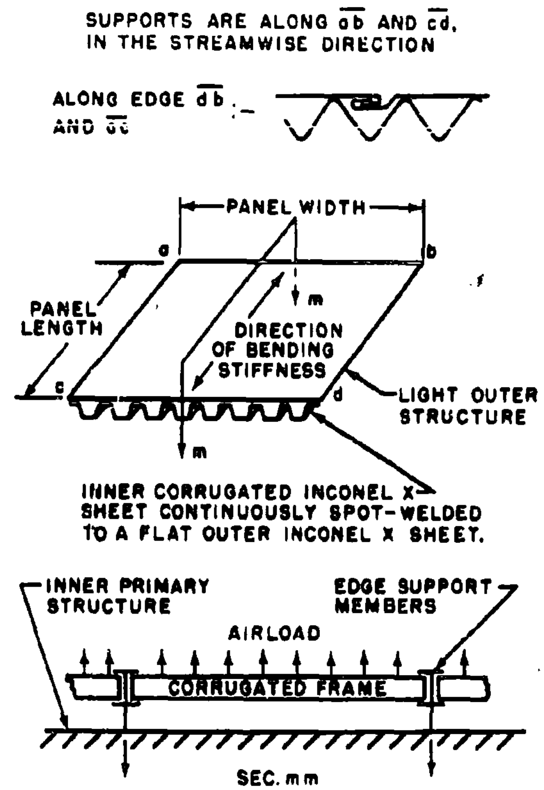


Figure 39. Structural Configuration Principles

this system, the area being cooled can be operated at a lower temperature level than the area from which heat is being dissipated. However, since materials are setting the limits, this is a dubious advantage to offset the additional weight and complexity.

5. Structural Configurations

The preceding discussion of the problem of protecting primary structure indicated promising results for systems which utilize external insulation either with or without cooling. Preliminary analyses indicated that a light outer wall structure to provide aerodynamic contour and carry airloads, combined with an inner primary structure to carry structural loads was the most satisfactory solution. This principle is illustrated in Figure 39 which also shows the

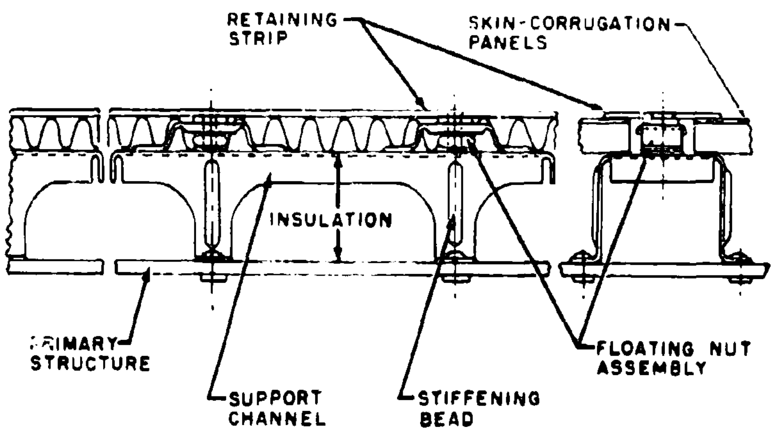


Figure 40. Outer Wall Attachment Structure

mechanism of airload transmission to the primary structure via the supporting edge members while at the same time retaining the capability of expanding thermally in all directions. This floating characteristic avoids transfer of thermal stresses to the supports and hence to the primary structure. Inconel X was chosen as the external skin and corrugation as previously explained.

Two designs utilizing these principles have been devised. In the first of these (Figure 40) the nuts supporting the outer retaining strip are arranged to float after the screw has been tightened, so that differential expansion between the retaining strip and the support channel is accommodated. Both the support channel and the retaining strip are divided into approximately 4-inch lengths to allow for thermal expansion, but free expansion of the support channel is somewhat restricted by the attachment to the aluminum primary structure. Thermal stresses and deformations induced in the support channel by this restraint have been estimated and found acceptable. Heat transmission into the primary structure has been minimized by arranging for edge contact only at the base of the floating nut, and by reducing to a minimum the width of the conduction path through the support channels. This latter also minimizes the weight. Total weight of this attachment structure, based on 8-inch wide panels, is equivalent to 0.70 pound per square foot of surface.

Figure 41 shows a second edge attachment development in which the retaining strip and the support channels have been combined to provide, with reduced thicknesses, the necessary bending strength to carry loads from the corrugations. Figure 41 also shows how the weight and heat conduction have been minimized in the attachment by removal of material, and strength and stiffness maintained by maximum use of flanges, beads, etc.

The attachment of Figure 41 has the advantage over that of Figure 40 in that it is appreciably lighter in weight, being only 0.31 pound per square foot of surface for an 8-inch wide skin panel. This advantage is offset, however, by the fact that individual skin panels are not readily removable for repair or maintenance, and also by the fact that the proposed attachment is more susceptible to thermal stresses. This latter condition arises because of the additional continuity of material and, hence, additional restraints in the scheme of Figure 40 compared with that of Figure 41.

6. Testing

The type of construction which was analytically found to be of minimum weight for the

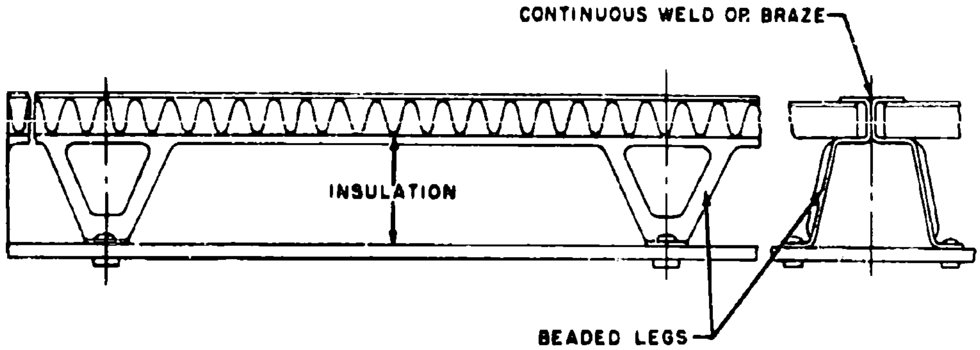


Figure 41. Outer Wall Attachment Structure

primary wing structure of Stage III (Figure 40) was subjected to a series of tests. The test program included tests of fabrication, strength, rigidity, thermal warpage, and cooling.

a. Fabrication

The fabricating tests indicate that satisfactory methods are available for handling all problems connected with fabrication. These include forming, joggling, and welding of panels, all to very close tolerances. These methods are available even though much of the stock is of foil gauge.

b. Strength and Rigidity

Bending and simulated air loading tests were conducted on outer wall specimens at room temperatures. These tests showed that stresses of 88,500 psi under positive conditions and 136,000 psi under negative conditions were required to fail the specimen. In the air loading tests, a positive average distributed loading of 14 psi was successfully withstood, and failure occurred at 17.9 psi average distributed negative loading.

c. Thermal Warpage Tests

A wall panel was heated from a heat source such that a uniformity within ±50° at 1500°F was achieved on the test panel. A maximum deflection at the center of the six-inch span of 0.037 inch occurred at a time of 80 seconds. The temperature program applied consisted of heating at a rate of 14°F per second from 70°F to 1470°F in 100 seconds, holding 1470° for 30 seconds, and decreasing at a rate of 5°F per second to 224°F at a total elapsed time of 375 seconds.

d. Thermal Cooling Tests

A series of tests were conducted to demonstrate the feasibility of the proposed wick-type cooling system. These tests were designed to determine:

(1) If the wick which covers the inner surface of the aluminum alloy wall can distribute the coolant under the prescribed heat flux and not develop dry areas next to the skin or in local patches.

(2) The amount of coolant required to cool the aluminum skin at the boiling temperature of water.

The basic specimen consisted of 0.125-inch x 8.00-inch x 10.88 aluminum inner wall and two 4.17-inch x 8.37-inch sections of outer wall, the necessary supports, nut plates, retainer strips, screws, and locating pins to duplicate the proposed means of attaching the outer wall to the inner wall.

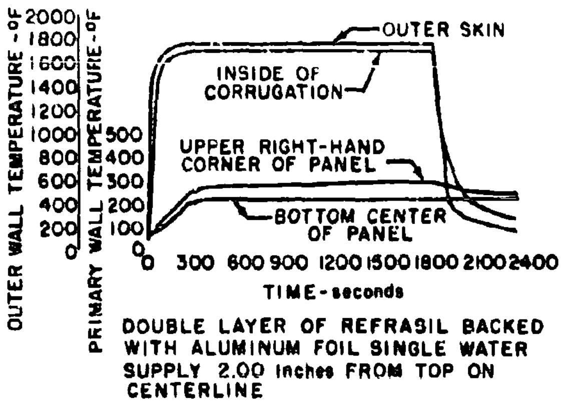


Figure 42. Thermal Cooling Test: Double Layer Refrasil

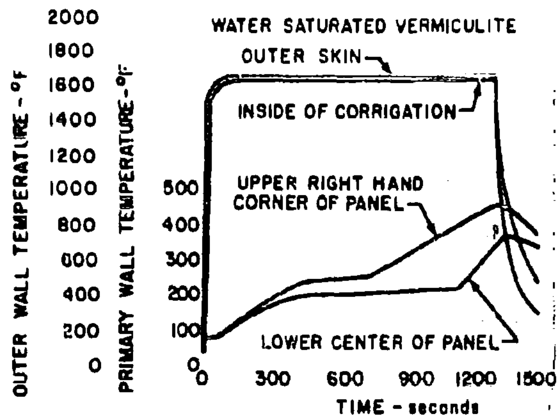
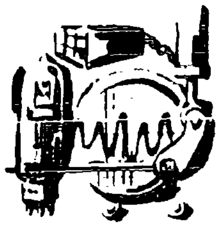


Figure 43. Thermal Cooling Test: Saturated Vermiculite

The most successful cooling system utilized a double layer of refrasil backed with a perforated aluminum foil sheet. During the tests water was supplied at a rate of approximately 17 cc per minute through a single supply point located 2.00 inches from the top of the panel. Figure 42 shows the test results. It is significant that the sections where a supply of water was continuous were maintained at the boiling point of water.

Another test of interest was that of the cooling system wherein the water required for cooling was stored in 0.26 pound of 15 pounds per cubic foot density vermiculite which was held in contact with the inner wall by a box made of 0.0045 diameter 100 mesh stainless steel screen. In this vertically mounted test panel a water ratio of 1.54 was obtained as compared with 3.83 obtained in a horizontal panel. Figure 43 shows the temperatures attained during the tests.

Comparison of the time elapsed before the sharp temperature rise at the various thermocouple locations shows the marked effect of gravity on the distribution of the water over the panel. It is apparent from these results that the portion of the vermiculite adjacent to the inner wall tended to dry out more rapidly than water could be drawn from the remaining vermiculite.



D. NAVIGATION AND CONTROL

1. General

The tremendous speed, short time of flight, and longe range of Stage III create many navigational problems which are peculiar to this type of weapon system. In addition, the navigation and control system must be capable of operation for either bombing or reconnaissance missions. Reference 8 is a detailed report of the study of the navigation and control system.

Considering the general requirements, the following four classes of navigation systems have been investigated for possible application for the MX-2276 weapon:

a. All-inertial navigation

b. Doppler-aided inertial navigation

c. Position-aided inertial navigation

(1) Loran type

(2) Radar beacon

(3) Navarho

(4) ATRAN map-matching

(5) Star Tracker

d. All-electronic positional navigation with the same position fix possibilities as mentioned above.

These classes of navigation were evaluated considering the following aspects in addition to the basic accuracy requirements.

a. Weight of airborne equipment (plus cost and power consumption).

b. Number of ground stations and their complexity (plus operating personnel, maintenance cost, etc.).

c. Present state of the art (and estimate of development in 10 years).

d. Flexibility of use and capability of independent action by pilot. (The factors shown in parentheses were secondary considerations only.)

As a result of these studies, the primary bomber navigation system selected is all-inertial. The radar and optical systems are used by the pilot as sources of navigation information for use at his discretion in diverting to an alternate target or landing site, correction of mapping errors, or other action requiring judgment and decision. They also serve as a means of obtaining reconnaissance data. For the reconnaissance mission, extreme accuracy is required of the navigation systems. In order to illustrate the accuracy which this system can attain for this purpose, a preliminary error analysis to determine instrumentation requirements has been made for a total CPE of 4000 feet at an 8000-nautical mile range. This range and accuracy is considered to be the maximum capability. A relaxation of the accuracy requirements would eliminate the need for some of the advanced instrumentation techniques recommended for attaining this accuracy.

During a bombing mission, the bomber's navigation and control system must navigate the bomber to the desired release point, provide the proper initial conditions to the bomb, and release the bomb at the correct time. After release, by means of radar or optical fixes on the target or aimpoints, the crew measures and transmits position corrections to the bomb via a radio link. On the basis of this correction to the bomb guidance system, the accuracy requirements of the bomber navigation system are less stringent, for a bombing mission alone, and the instrumentation can be less precise.

The bomb navigation system should guide the bomb to the target over a range in the order of 300 nautical miles and detonate the warhead with a 1500-foot CPE at the desired target.

2. Bomber Navigation and Control System

For clarity in discussion, the system can be considered in three sections. An inertial reference system determines the position of the bomber with respect to a set of reference coordinates. The navigation system determines the flight path the bomber is to follow and generates the required control signals. Finally, the control system exerts the proper forces on the bomber to cause the desired maneuvering and to obtain satisfactory stability.

Figure 44 is a block diagram showing the major components of the navigation and control system. The inertial reference system is represented by the platform, accelerometers, and position computer. A flight path computer and flight programmer fulfill the requirements of the navigation system in supplying the proper roll, pitch, and yaw control signals to the bomber autopilot.

a. Inertial Reference System

The heart of the inertial reference system is the accelerometers measuring along a set of reference coordinates. This coordinate system is established by the orientation of a multi-axis, gyro-stabilized platform. A position computer double integrates the outputs of the accelerometers to obtain the desired position information. In addition to this function, the position computer generates the signals necessary for keeping the platform aligned with the reference coordinates and the signals necessary to convert the output indications of the accelerometers from an inertial to an earth-fixed reference. These signals are the correction terms for centripetal and Coriolis accelerations.

(1) Reference Coordinates

One of the simplest systems to instrument is the conventional latitude and longitude coordinate system where the measurements are performed in the horizontal North and East directions. However, this system is limited to a nonpolar operation due to the excessive azimuth torquing rates (infinity at a pole), necessary to keep the system North aligned.

Since the operational area of the MX-2276 extends over the polar regions, a transverse polar coordinate system has been selected, utilizing poles which lie outside the operational area of the bomber. Although the required instrumentation is more complex than for the conventional polar coordinate system (North-East system), navigation is permitted over vast areas of the globe.

The transverse system is based in its orientation on a great circle between the take-off and landing points. Since it is desirable for simplicity to measure horizontal accelerations in two mutually perpendicular coordinates, the system is based on transverse longitude and latitude.

The equator of this system is taken along the great circle connecting the take-off and landing points. This avoids the possibility of approaching the transverse poles even though the target may conceivably be off this great circle by an appreciable distance. As can be seen from Figure 45, this system is similar to a conventional latitude-longitude system with the equator displaced. The vertical coordinate is established by the direction of the instantaneous local vertical.

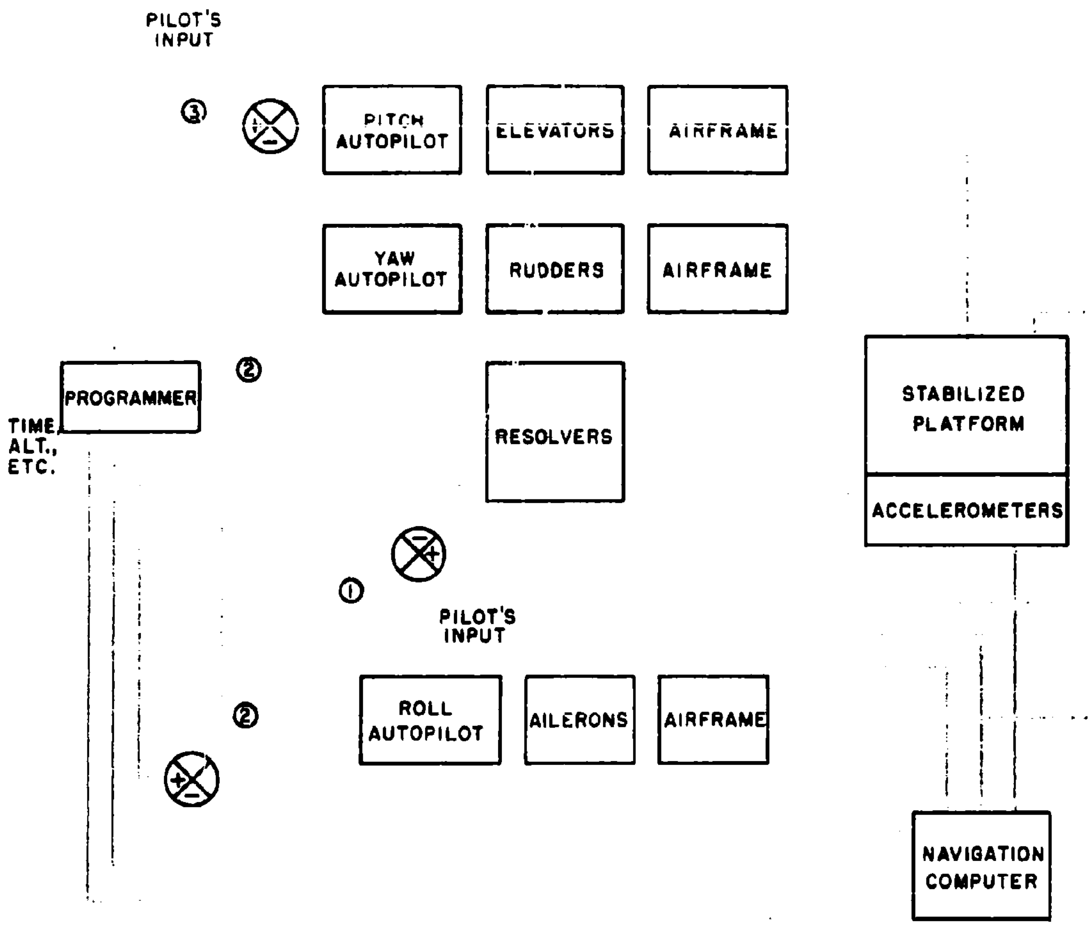


Figure 44. Bomber Navigation and Control System

(2) Stabilized Platform

The stabilized platform maintains the measuring axis of three accelerometers along the direction of the transverse longitude and latitude coordinates and the local vertical. The supervision of the platform to establish this reference coordinate system can be achieved by several different methods. Two of these methods seem appropriate to this project according to the present state of inertial techniques.

The first of these methods, which involves the use of a star-tracker, has special merits for very long times of flight since there is no steady state drift involved. However, the instrumentation required is complex, and this complexity combined with the short time of flight makes the use of this system unwarranted.

The second method which is recommended is a conventional gyro-stabilized platform which uses very accurate gyros. From the many possible platform gimbal config-

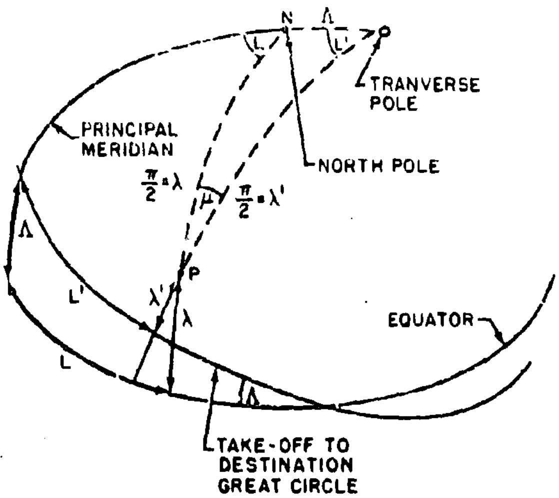


Figure 45. Relationship of the Transverse to a Conventional Latitude and Longitude Coordinate System

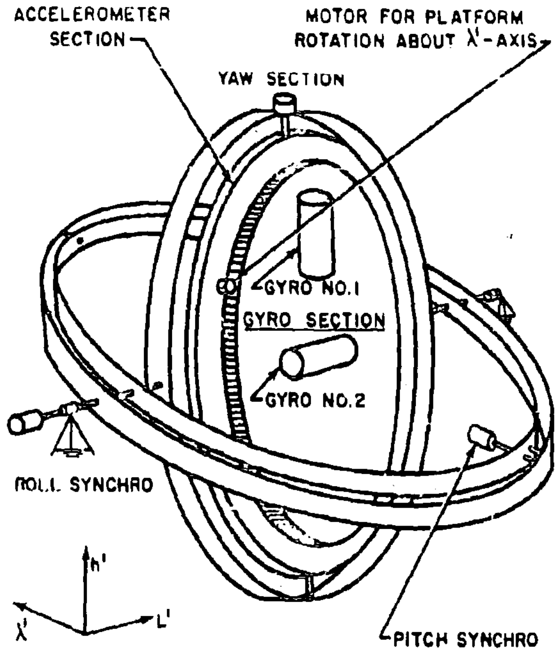


Figure 46. Platform Design for Bomber Navigation System

urations, the platform shown in Figure 46 was selected since it easily accommodates both the vertical launch and a large roll angle for maneuvering. It is gimballed as a yaw-roll-pitch platform with the addition of an outer roll gimbal. The inner element of the platform is separated into the accelerometer section and the gyro section which are rotatable relative to each other.

To maintain the platform horizontal and aligned with the reference coordinate system, it is necessary to rotate the platform to compensate for earth's rotation and curvature.

(3) Position Computer

The position computer determines the instantaneous position of the bomber in terms of altitude and transverse longitude and latitude. In addition, the position computer supplies the various signals for torquing the gyros and rotating the accelerometer platform section relative to the gyro section in order to keep the stabilized platform slaved to the reference coordinate system.

The accurate determination of altitude by pure inertial means presents certain problems. To circumvent these problems a radar altimeter will be combined with the inertial instrumentation. Corrections must also be made for the change in radius of curvature of the geoid as a function of latitude, direction, and altitude. In order to keep the accelerometers properly aligned with the axes of the reference coordinate system, a combination of Schuler tuning and compensation for earth's rotation is utilized.

(4) Accuracy Considerations

The overall CPE of the proposed inertial navigation system is determined by the accuracy of the components of the system.

The following assumed accuracies of these components are values that will be attainable within the time period of this program.

Accelerometers	0.5×10^{-5} of full range
Integrators	2×10^{-5} of full range
Gyros: Random Drift	0.01 deg per hour
Torque Accuracy	0.01 deg per hour (torquing rates less than 100 deg per hour)

The preceding value for gyro drift does not reflect improvements possible with the special gyro operating technique discussed later in this section.

Initial ground alignment of the platform will be made to the following accuracies:

Leveling	4 seconds of arc
Azimuth Alignment	10 seconds of arc

It is believed that a leveling period of one-half hour will be sufficient to attain these values.

In performing a preliminary error analysis of the navigation system, the following assumptions have been made concerning an MX-2276 mission.

Maximum Navigation Range	10,400 nautical miles
Average Velocity	4,000 nautical miles in 25 minutes
Maximum Cross Range	1,000 nautical miles
Maximum Transverse Latitude	17°
Maximum Altitude	50 nautical miles
Maximum Longitudinal Velocity	22,000 feet per second
Maximum Lateral Velocity	4,000 feet per second
Maximum Vertical Velocity	1,000 feet per second
Maximum Longitudinal Acceleration	5g
Maximum Normal Acceleration	0.5g
Maximum Vertical Acceleration	1g

Based on the foregoing assumptions, the individual standard errors have been computed and listed in Table V.

These standard errors have been evaluated for various flight times and the CPE determined. The results of this error analysis are shown in Figure 47.

It should be noted that in the evaluation of errors the high cruising velocity of the bomber was considered. The centripetal acceleration due to this speed reduces the effective vertical acceleration to about 0.25g.

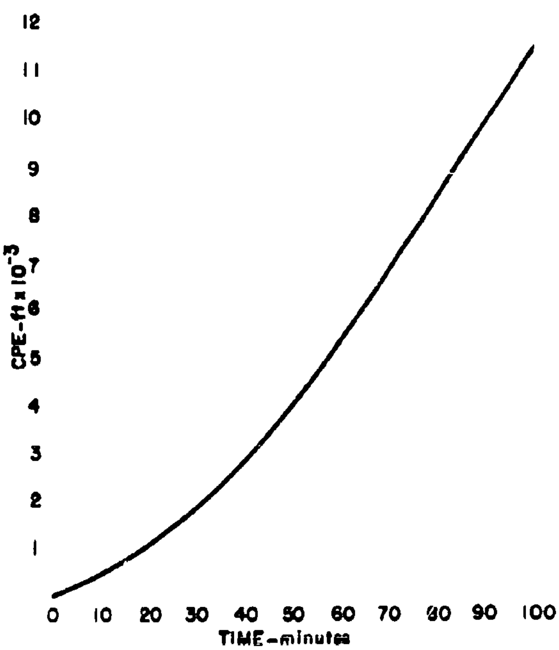


Figure 47. CPE as a Function of Flight Time and Range

TABLE V

INSTRUMENTATION ERRORS

	Range	Accuracy	Standard Error
A. Transverse Longitude Channel			
1. Accelerometer	±5g	0.5×10^{-5} of full range	2.5×10^{-5} g
2. Velocity Integrator	±22,000 ft/sec	2×10^{-5} of full range	0.44 ft/sec
3. Range Integrator	±10,400 n.mi	2×10^{-5} of full range	1265 ft
4. Schuler Loop Integrator	±180 degrees	2×10^{-5} of full range	13 sec - arc
5. Positioning of Accelerometer Section			2 sec - arc
6. Gyro Drift			0.01 deg/hr
7. Initial Platform Leveling			4 sec - arc
B. Transverse Latitude Channel			
1. Accelerometer	±0.5g	0.5×10^{-5} of full range	0.25×10^{-5} g
2. Velocity Integrator	±4000 ft/sec	2×10^{-5} of full range	0.08 ft/sec
3. Range Integrator	±1000 n.mi	2×10^{-5} of full range	121 ft
4. Gyro Drift			0.01 deg/hr
5. Gyro Torquer			0.01 deg/hr
6. Initial Platform Leveling			4 sec - arc
7. Initial Azimuth Alignment			10 sec - arc

Gravitational anomalies must also be considered for the ground leveling of the platform since the gravitational radial anomalies in the United States may approach 0.2 cm per sec^2 and deflections of a plumb bob up to 30 seconds of arc have been observed. Because of the height of the cruise portion of the flight path, it can be expected that local anomalies along the flight path of the bomber practically disappear.

The accuracy of the computation of the correction terms for the latitude and longitude channels (radius of curvature, Coriolis and centripetal accelerations) is selected so as to be compatible with the basic instrumentation.

Altitude is determined to 300 feet and vertical velocity to 0.1 foot per second through the use of the previously described altitude channel. This allows a computation of the radius of curvature correction with a 500-foot error and an error of 10^{-4} of the total corrections for centripetal and Coriolis accelerations. In the worst case (transverse longitude channel) this would correspond to an additional 5×10^{-6} g standard accelerometer error and a 0.5 foot per second standard velocity error.

The errors involved in the computation of the gravitational acceleration using a constant value for the radius of the earth is well within the allowable vertical error.

(5) Instrumentation

The instrumentation of the MX-2276 bomber 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. In addition, certain auxiliary computing elements are required to generate the necessary correction terms. The required computations can be performed by either analog or digital techniques; a third method, that may be termed "pulsed-analog", shows promise in that it avoids the complexity of arithmetic approximations while retaining the advantages of "counting." A preliminary analysis indicates that a combination of pulsed analog and digital techniques would provide the best system.

(a) Gyros

Several gyros are under development for various programs that could conceivably satisfy the accuracy requirements. However, these are of rather large size and weight so that the platform becomes extremely heavy. A different solution becomes mandatory because of the low permissible weight in the bomber.

One solution utilizes a slow and continuous rotation of the gyro housing about the gyro spin axis. By this process, most of the gyro drift torques are rotated so that their effect in terms of drift angle becomes negligible. Such a scheme allows the use of small gyros.

(b) Accelerometer and Integrators

The high accuracy requirement rules out the use of conventional direct reading accelerometers and necessitates a servo-constrained instrument. This is a null-type instrument whereby the reaction torque, due to the acceleration of an unbalanced mass, is nulled by an electromagnetic torque. The current in the torque coil is then proportional to acceleration.

b. Navigation System

Input information to the navigation system is obtained from the position computer which indicates the location of the bomber from map data and from flight operations as stored in the flight programmer. The system is automatic but may be corrected at the discretion of the pilot by information derived from the radar or optical equipment.

The basic problem of navigation can be considered in two parts: flight between two aimpoints, and aimpoint departure. The take-off and landing sites as well as the target are considered as aimpoints in this discussion. In the general case the desired flight path would consist of crossing several aimpoints for a reconnaissance mission or in passing sufficiently close to prescribed check points (aimpoints) to obtain radar or optical fixes while performing a bombing mission.

(1) Flight Between Aimpoints

The most direct flight path between two points on the earth is along a great circle defined in earth-fixed coordinates. While following such a path the bomber is subject to Coriolis forces which the bomber has to counteract. This requires an appreciable roll angle during the entire time of flight, resulting in a probable loss in range.

To avoid these forces the bomber may be navigated along a great circle course in a space-fixed coordinate system. The disadvantage of this form of navigation is the existence of large position errors with respect to the aimpoint, since it is impossible to predict the exact course of the bomber over the earth. These variations from the expected flight path are caused by deviations in the velocity and altitude of the bomber from predicted values.

The existence of large position errors requires either a switch to another form of navigation or a rapid turn in the vicinity of the aimpoint to obtain the required accuracy in passing over the aimpoint.

A method of navigation that would combine the advantages of the previously discussed systems would program transverse latitude as a function of transverse longitude as indicated by the position computer. The programmed latitude is the latitude at which the bomber should be in order to insure arrival over the aimpoint. This is a space-fixed great circle path if the actual values of altitude and velocity correspond to predicted values. If this assumption is in error, the bomber still follows the programmed path on the earth, but this path is no longer a space-fixed great circle. The bomber is then subject to Coriolis forces that are a function of the difference between actual and predicted values of altitude and velocity. The important consideration is that the bomber is navigated to arrive over the aimpoint.

(2) Aimpoint Departure

After leaving an aimpoint the bomber, in the general case, enters a turn to reach a new heading angle before navigating to the next aimpoint. The flight path, particularly the heading angle at which the bomber approaches an aimpoint, and the location of the next aimpoint are carefully selected in order to restrict the required turn angle to a small value.

(3) Flight Programmer

The bomber navigation system contains a flight programmer which is capable of automatically initiating the various functions required along the flight path. This programmer consists of a storage element and several function generators. The storage element permits setting in all pertinent information for the operation of the bomber. The function generators produce the various programs as functions of position, altitude, speed, acceleration, time, etc.

The following list, though not entirely complete, points out the most significant programs and commands which have to be generated during the bomber flight.

(a) Pitch attitude

(b) Stage separation signals

(c) Aimpoint location

(d) Alternate aimpoint location

(e) Automatic leveling for bomb platform

(f) Automatic bomb release

(g) Changing autopilot parameters

(h) Automatic checkout procedures

c. Control System

In addition to its normal function of providing stabilization of the bomber, the control system performs the lateral maneuvering and pitch control required to fly the desired flight path. Provision is made for the pilot to override the automatic control system and to provide manual control.

If a navigation system malfunction occurs, the pilot can assume control and navigate using conventional techniques. Similarly, in case of a failure of just the flight path computer, the pilot may navigate the bomber along the programmed path by monitoring the bomber's computed position on a map display.

3. Proposed Bomb Guidance System

The bombing mode consists of allowing the bomb to fall freely until it reaches a Mach number low enough that aerodynamic control may be initiated without producing excessive heating. Since the bomb must be prevented from rolling during its entire trajectory, a means of roll control other than aerodynamic should be provided.

A basic inertial system which can be corrected from the bomber by means of a radio link appears to be most suitable for guiding the bomb to the target after it is released from the bomber. The bomber operator can obtain accurate information on the target location by either radar or optical means as the bomber approaches the target area. These data can then be used to compute error signals for transmission to the bomb. This correction feature is of extreme value considering the accuracy required and the lack of map data on target locations.

a. Coordinate System

The choice of coordinates for this system is very much dictated by the bomber navigation instrumentation which delivers velocity and position information in a transverse polar coordinate system. The system selected consists of using the same basic velocity and position information as developed in the bomber navigation system. The lack of any transformation in the coordinate systems between bomber and bomb should result in a higher over-all accuracy without increasing the complexity of the total arrangement. Corrections in the target location required during the bomb flight can be accomplished very easily by varying the target coordinates.

b. Stabilized Platform

As in the inertial reference system of the bomber, the three accelerometers are mounted on a multi-axis platform which stabilizes their measuring axes along the axes of the desired reference coordinate system.

c. Transfer of Initial Conditions

For the proper operation of the bomb guidance system it is necessary to introduce the instantaneously correct initial conditions from the bomber navigation system. This allows a proper alignment of the stabilized platform in the bomb so that the bomb system accelerometers read the same accelerations as those on the bomber platform. The velocity and position outputs of the bomb system are also adjusted to the indications of the bomber navigation system. The position information from the bomber consists of the instantaneous position of the bomber less the coordinates of the target. The bomb guidance indication then develops in terms of range-to-go to the target.

d. Position Computer

The bomb position computer basically double integrates accelerations to determine position. It has the added requirement of providing platform rotation signals and Coriolis and centripetal acceleration corrections. Since the reference coordinates of the bomb and bomber systems are identical, the correction terms required to be computed by the bomb position computer are identical to those for the bomber.

From preliminary investigations it can be deduced that the random portion of the gyro drift should not exceed about 0.1° per hour, accelerometer errors should be in the order of 5×10^{-4}g, and the integrator accuracy should amount to less than 1 part in 5,000. As mentioned previously the small range and shorter flight time of the bomb allows a less accurate instrumentation of the corrections than was required for the bomber navigation system. The over-all error of such an instrumentation over a 300-nautical mile range, including any effects of these errors on the alignment performance, is estimated to be in the order of 1500 feet CPE, exclusive of map errors.

e. Flight Path Computer

The "range-to-go" information in terms of transverse longitude, latitude, and altitude increments in itself is not sufficient to direct the bomb to the desired detonation point. A flight path computer is necessary to derive from this information the necessary signals from the proper yaw and pitch control.

4. Radar and Optical Correction of Navigation System

a. General

Correction of both the velocity and position outputs of the bomber navigation system have been considered. It appears that the radar and optical data cannot be obtained with sufficient accuracy to warrant velocity corrections. However, they can be obtained accurately enough to warrant position corrections.

One possible method corrects only the range-to-go information computed in the bomber without correcting the bomber navigation system.

The method of position corrections can be seen from Figure 48 which assumes, for clarity, perfect position fixes. (L_O', λ_o') are the map coordinates of the aimpoint while (L_1', λ_1') are the true coordinates. (L', λ') is the location of the bomber as indicated by the navigation system and (L_2', λ_2') the true bomber location. The radar or optical equipment measures P_A, the relative distance between the true aimpoint and the true bomber location. However, in computing the true bomber location on the basis of position fixes, the map coordinates of the aimpoint are used, thus locating the bomber at (L_3', λ_3'). The distance, P_A, is displayed to the pilot.

An outline map of the terrain over which the bomber flies is also displayed to the pilot and is driven by the navigation longitude and latitude indications. The bomber position on this may may be represented by a pair of cursors which move over the map face, or by positioning the map with respect to the display such that the center of the display is maintained at the bomber position. On the map, (L' - L_O') represents the distance from the indicated position of the bomber to the predicted aimpoint along the L' axis, and ($\lambda' - \lambda_o'$) represents the corresponding distance along the λ' axis. These distances are also indicated to the pilot.

Position corrections are then made by the pilot by adjusting the map display to make the distance (L' - L_O') and ($\lambda' - \lambda_o'$) corresponding to the respective distances on the radar or optical displays.

b. Map Display

The map display is driven by the indicated longitude and latitude of bomber in such manner that the center of the display is the bomber location. Since the bomber location is also the center of the radar and optical displays, this map may then be superimposed over either the optical or radar display which have

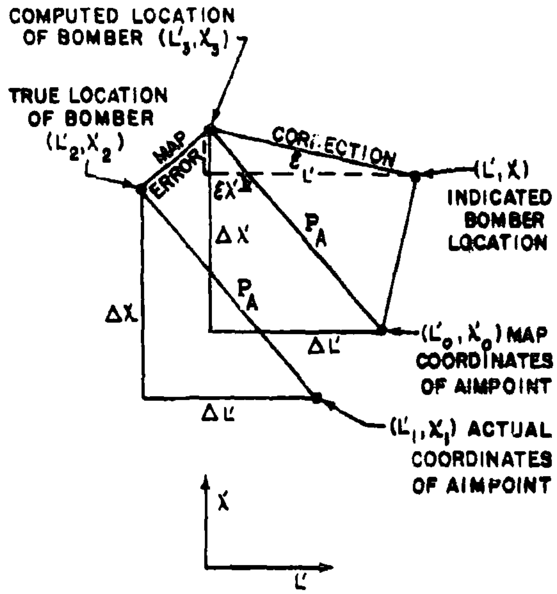


Figure 48. Position Correction of Navigation System

the same scale. The pilot is then readily able to see the difference in the location of an aimpoint on the map as compared to the radar or optical display.

Figure 49a shows a sketch of this map display. The map may be a reversed negative with aimpoint and major land features outlined on it with respect to the programmed flight path which is along the centerline of the map. The map is on a long strip and scaled equal to the scale of the radar and optical displays. Movement of the map is lengthwise through the display as the bomber travels along its programmed path. The programmed path, in general, deviates a certain distance from the L' axis of the transverse coordinate system, but because both the take-off point and the landing point are on this axis, these deviations are small. Therefore, the map may be driven directly with the computed transverse longitude and scaled as a function of the angle χ, the angle between the programmed path and the L' axis. This eliminates the possibility of errors arising from a computer used to con-

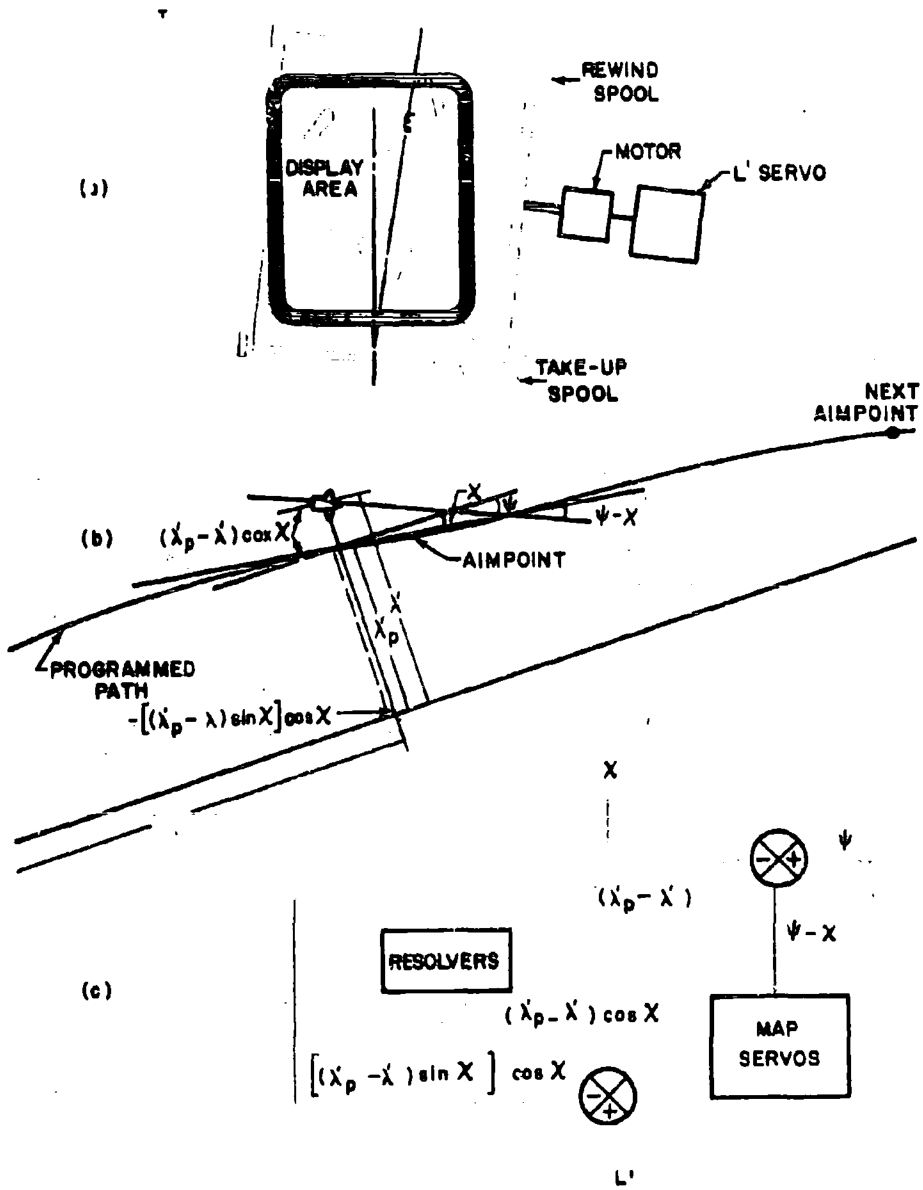


Figure 49. Map Data Presentation

vert distances along the L' axis to distances along the flight path.

Since the bomber, in general, does not exactly follow the programmed path, these deviations must be introduced into the map display. As shown in Figure 49b, the bomber location at an indicated longitude L' differs from the programmed path by $(\lambda_p' - \lambda')$ where λ_p' is the programmed latitude.

The map computer, Figure 49c, resolves this error into components normal and tangent to the programmed flight path. The normal component, $(\lambda_p' - \lambda') \cos\chi$ is used to drive the map normal to its centerline. Similarly the tangential component $(\lambda_p' - \lambda') \sin\chi$ represents an error in the location of the bomber along the programmed path as presented on the map. Since the map is driven with the L' indication, the term $(\lambda_p' - \lambda') \sin\chi$ is resolved into a component along the L' axis. This term is used to correct the L' indication driving the L' map servo.

It is also necessary to rotate the map about the center of the display by the angle between the bomber axis and the tangent to the programmed path. This angle is $(\psi - \chi)$ where ψ is the horizontal angle between the bomber attitude and the L' axis.

5. Radar

a. Recommended System

In Reference 1 in which the early work on the weapon system was reported, a K_u-band side-looking radar system was proposed. A technique of simultaneous lobing was specified to provide resolution improvement. Radar data were to be displayed on a facsimile printer or a rapid film developing and viewing device. As a result of the work accomplished during this study contract in which only the original configuration proposed for the airframe was considered, the most promise still lies in the simultaneous lobing, side-looking radar (Table VI and Figure 50), very similar to the one proposed at the start of this program. The power has been increased to improve ground painting and permit the observation of smaller target areas. Reference 9 contains a detailed description of the radar study.

The technique for simultaneous lobing, as incorporated in this radar, requires that the antenna array, looking to either side, be divided into two halves. Each half is end-fed and the combined pattern will depend on the relative phasing between the two halves. If the two halves are fed in phase, the combined pattern is as shown in the upper pattern of Figure 50b. If they are fed out-of-phase, the combined pattern is as shown immediately below (the center of the pattern will be displaced a few degrees from perpendicular to achieve a reasonably broad band impedance match). Both of these patterns are obtained simultaneously by connecting the two halves of the antenna to the "side" arms of a "Magic Tee" or other hybrid. The other two arms of the hybrid are respectively called "sum" and difference" arms in the block diagram, Figure 50d.

The remainder of the radar set consists of a modulator, magnetron and power divider, a local oscillator referenced to the magnetron frequency, 4 mixers and IF strips (a sum and a difference strip for each side) and 4 detectors. The "sum" detectors are ordinary linear detectors; the "difference" detectors are balanced phase detectors. The output of the phase detector is (approximately) the magnitude of the product of the two inputs of polarity determined by the relative phases. (See the bottom pattern shown in Figure 50b.) The detector outputs and synchronizer pulses go to the display.

The display which is proposed for this radar (Figure 51) is a flying-spot cathode-ray tube whose image is projected onto a moving photosensitive film which is developed in 2 seconds or less for viewing by the pilot. The image of the radar map would be combined optically with the previously prepared strip map to facilitate position fixes. Provision for comparison of the map with the actual ground as viewed by a periscope should also be included.

The details of this display are as follows. The spot of the high resolution cathode-ray tube is deflected by a range sweep starting

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

Frequency	16 kmc	
Peak Power	100 kw each side	
Pulse Repetition Frequency	1200 pps	
Pulse Width	0.4 microsecond	
Receiver Bandwidth	3.0 mc	
Antenna Length	22 ft	
Antenna Width	6 in.	
Ground Coverage	10 to 50 nautical miles, each side	
	Near Range	Far Range
Spot Size (from 200,000 ft)	500 ft x 600 ft	1000 ft x 250 ft
NOTES: See Figure 50c. The underlined figures represent the resolution in azimuth. They are subject to a further improvement of a factor of 3, achieved by indicator display technique (spot positioning).		
Resolution of Indicator	5000 spots for 100 n. miles	
Film Requirement for 10,000 n. miles	7 in. x 50 ft	
Minimum Discernible Spot Target	40 square meters	
Weather Penetration	Light rainfall	
Weight Estimate		
Antennas	300 lb	
Radar and Indicator (CRT only)	250 lb	
Size Estimate		
Antennas (2)	24 ft x 1.5 ft x 1 ft each	
Radar and Indicator	6 cu. ft	

with the altitude return and intensity modulated by the output of the "sum" receiver. The resulting line of spots is focused upon the moving sensitized film or paper whose speed is scaled from the aircraft speed given by the navigation computer. Separate tubes are used for right and left radars and the images are combined optically. The size of the projected spot on the film is about one-third of the corresponding beamwidth of the antenna. The position of this spot of illumination within the area on the film which represents the ground spot size is determined by the output of the "difference" receiver of the monopulse radar. Thus, if there are a number of strong reflectors within the illuminated ground spot, the position of the displayed spots on the film will, when averaged over a large number of pulses, show the targets approximately in their true locations. The ground intercept of the beam will be slightly hyperbolic

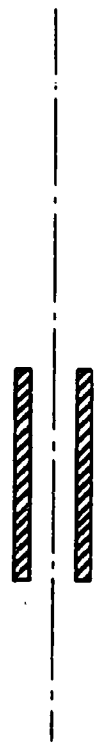


Figure 50a. Proposed Simultaneous-Lobing Radar: Possible Location c Antennas

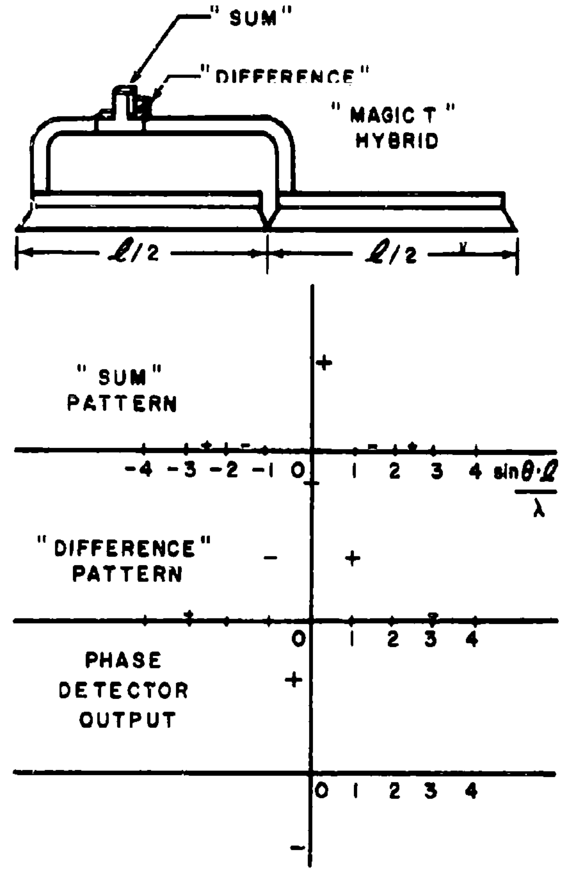


Figure 50b. Proposed Simultaneous-Lobing Radar: Antenna and Patterns

due to the deviation of the antenna axis from horizontal. The cathode-ray tube sweep would be slightly curved to compensate.

Since the antenna of this radar is not stabilized, and it is desired to present the operator with information in ground coordinates, some stabilization of the display is required. In order that the resolution of the recorded image will be limited only by the cathode-ray tube, the recording film must be about 7 inches wide (assuming 20 lines per millimeter optical resolution) and about 50 feet of film are required to record the 10,000-mile flight path.

b. Bomb Command System

Bell Aircraft Corporation has considerable experience in the design of command systems for the Rascal missile and has been working under contract to the Bureau of Aeronautics on improvements to render the system even more reliable and more difficult to jam. With this backlog of experience it has been relatively simple to specify the parameters of a missile control system which would meet the MX-2276 requirements. These parameters are shown in Table VII. The exact choice of frequency for this system will depend upon the configuration of both the bomber and the missile locations. A calculation of several possible frequencies shows that the indicated power will suffice for any of the possible frequencies.

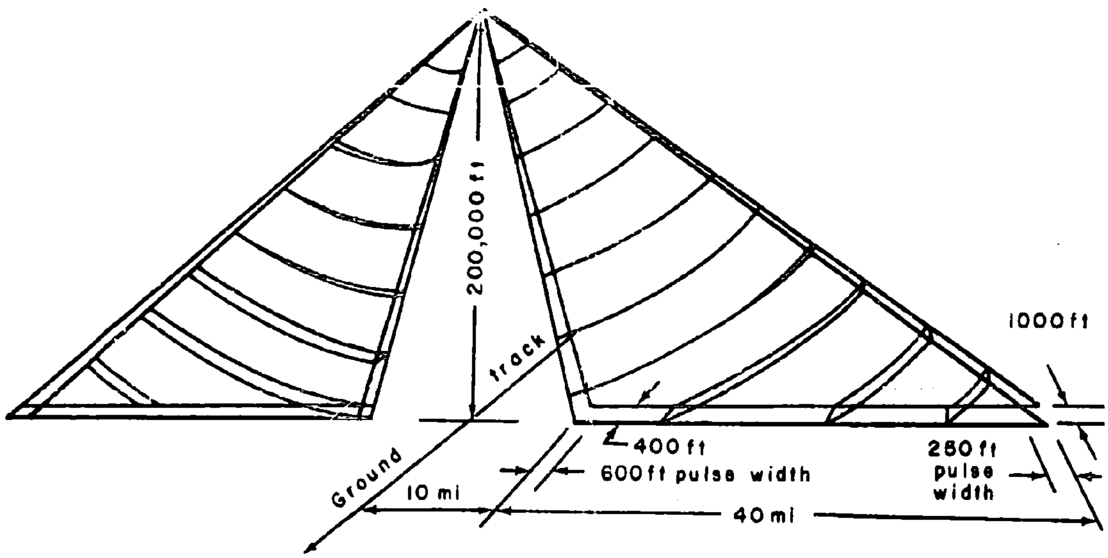


Figure 50c. Proposed Simultaneous-Lobing Radar: Ground Coverage

c. Associated Radar Problems

During the past year theoretical, and in some cases experimental, evidence has been obtained concerning the following problems which are common to all radar systems for this weapon. In each case satisfactory solutions have been obtained for application to the radar system proposed.

In addition, methods of resolution improvement have been studied intensively, especially those involving the techniques of coherent radar, in order to reduce the requirement for a large antenna array. It was found that because of the very high speed and altitude, none of the techniques studied are practical. The systems studied include CW radar, the Sherwin system, Redap, and Douser.

(1) Breakdown

The atmosphere in the region immediately surrounding the bomber is of a very low density (the pressure may be as low as 0.1 mm Hg). It is known that high-power radar energy will cause ionization of the oxygen and other molecules, provided the density of power flow is sufficiently great. An attempt has been made to determine from the theoretical data the limiting power level for antennas of various sizes operated at various frequencies in the range of altitudes at which the bomber is to fly. The primary source of experimental data on this problem is the work at the Research Laboratory of Electronics, Massachusetts Institute of Technology by Herlin and Brown in 1948. Utilizing these data it was determined that the radar systems proposed operate with powers that are less than the calculated breakdown power by a factor of four or greater, even for continuous operation. For pulsed operation at these altitudes, the power required for breakdown is expected to be considerably greater.

It appears that sufficient data are available on the breakdown problem to proceed with the design of a radar set provided safety factors of 2 to 5 times are used.

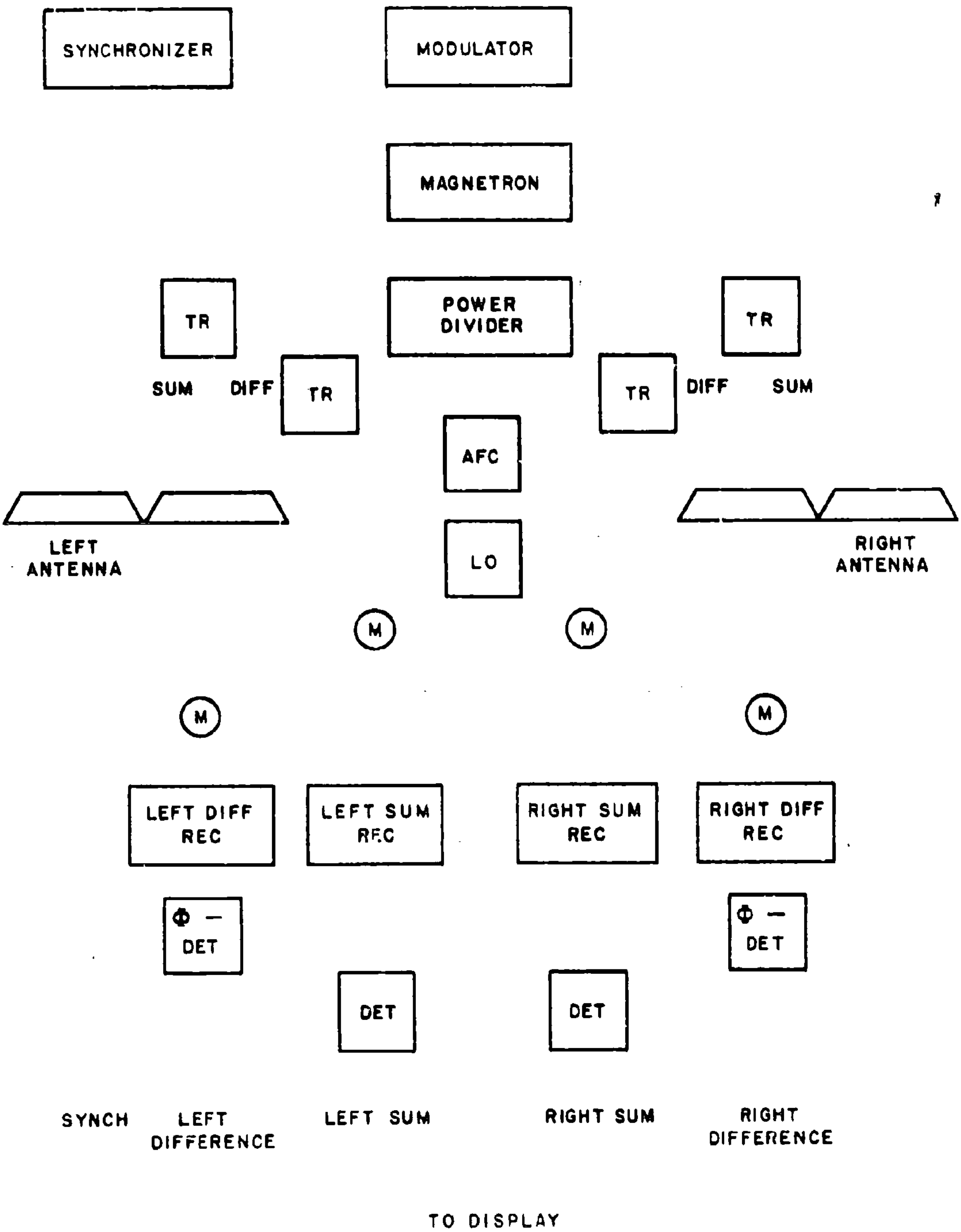


Figure 50d. Proposed Simultaneous-Lobing Radar: Block Diagram

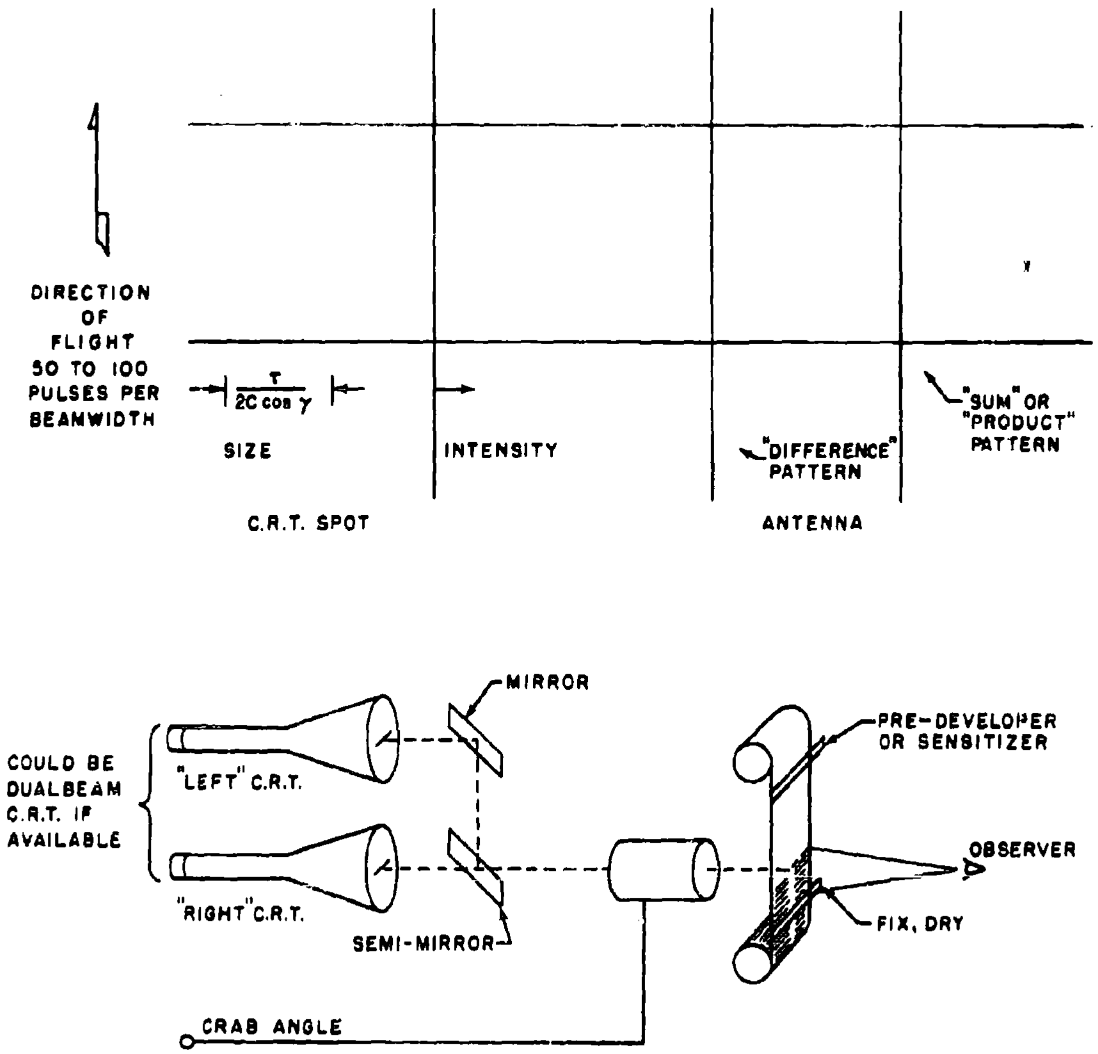


Figure 51. Display for Side-Looking Radar (Simultaneous Double-Pulse Lobing)

(2) Radome Problems

A survey of the ceramic and other radome materials disclosed one material which shows high promise for this application. This material is fused silica which is capable of withstanding the extreme temperatures, meets the thermal shock and structural strength requirements, and has acceptable electrical properties. Since some of the properties of fused silica far exceed those required, such materials as Fosterites, Steatites, Wollastonites, Aluminas, AF-156, and Vycor should be considered as possible alternatives. To evaluate their suitability, equipment for measuring electrical and mechanical properties at temperatures up to 1000° C at frequencies up to 36 kilomegacycles will be required.

TABLE VII. TENTATIVE SPECIFICATIONS
MISSILE COMMAND LINK

Frequency	1000 to 10,000 megacycles
Peak Power	5 kw
Channel Bandwidth	500 bits per second
R.F. Bandwidth	2 mc
Maximum Range for +20 db signal-to-noise ratio	240 n. miles (free space)
Carrier	
Weight Estimate	50 lb
Size Estimate	1.5 cu. ft (not including antenna)
Missile	
Weight Estimate	25 lb
Size Estimate	0.5 cu. ft

(3) The Effect of Atmospheric Bending on Radar System Accuracy

There are two types of bending which influence radar applications to this weapon: first, the air in the immediate vicinity of the bomber forms a weak but significant prism which will deflect radar (and optical) waves to a certain extent and will cause disturbances like blurring; second, the variable density and natural ionization of the air between the bomber and the ground will have the effect of bending and attenuating the radar waves. A calculation of the amount of bending which would be experienced for various angles of propagation was formulated theoretically. These show that for the angles within 60° of the vertical the angular disturbance due to the thermal ionization of the boundary layer will be completely negligible for frequencies above 1000 megacycles, while angular disturbances due to density variations are not important at frequencies below 36,000 megacycles. This applies both to the steady-state deflection and to random deflections due to turbulence.

The propagation through the remainder of the air, including the slightly ionized "D" layer, will not cause deviations detrimental to the radar system. As a result of the study on beam bending it was concluded that negligible distortion of the radar picture is to be expected.

(4) The Effects of Cloud and Rain

It is of considerable importance in the tactical applications of the MX-2276 that operations be independent of weather conditions on the ground. Radar provides this all-weather operation while optical methods are greatly handicapped by clouds. However, in order to provide a radar set having the best possible resolution and using the minimum power, it is necessary to go to a very short wavelength, which will result in some sensitivity of the radar system to weather effects. These effects have been extensively studied in many meteorological studies.

It is concluded from an application of these results to the MX-2276 problems that a suitable radar for this weapon system will be less affected by rain than navigation and reconnaissance systems presently in use, because of the high altitude, high depression angles, and high resolution.

(5) Large Antenna Arrays

Elementary considerations of power required and desired resolution dictate that the largest possible antenna structure be used.

All of the theoretical work and many of the practical techniques which have been developed in the past year on large-aperture airborne antennas are directly applicable to this antenna design problem. In addition, the techniques of surface-mounted radiators which have been developed at Ohio State may prove to be of value.

The largest antenna so far developed gives approximately 1/2° bandwidth at X band. The antennas considered above have slightly smaller beamwidths and are longer and perhaps will be heavier unless a higher frequency is used or the aircraft's structure is utilized to a very large extent.

(6) Jamming and Countermeasures

During this year's study, little effort has been devoted to consideration of jamming and electronic countermeasures because it felt that these properties of a radar system are very sensitive functions of the state of the art reached by the enemy and therefore cannot be predicted with enough reliability to warrant any firm conclusions. It is most likely that techniques for active jamming (noise, CW, and pulse) will be most fully developed in the 10,000-megacycle band because it has been the standard airborne radar frequency for almost 10 years and will probably continue to be so for some time. At higher frequencies (K_u and K_a bands) jamming will be less apt to be well developed and well organized. This is, of course, a matter of time, and if it were known or even suspected that such a frequency was to be used, the jamming equipment could be developed.

Techniques of passive jamming — radar camouflage and smokes — have been studied in this country and presumably elsewhere. The navigation is not strongly affected by radar camouflage because its inertial reference system makes large numbers of aim-points unnecessary. However, target identification could be confused by suitable techniques. The way to improve this situation is to increase the resolution, which is also desirable from other considerations. Against ionizing smokes, the higher frequencies have a definite advantage in that the required electron concentration for total reflection increases linearly with the radar frequency. The use of such techniques has only been postulated, however.

d. Alternative Radars

Since many radar systems were investigated prior to the selection of the system recommended, it is appropriate to list those which are most interesting and may later prove to be useful for this weapon system.

(1) Side-Looking Double-Pulse Lobing Radar, Table VIII

The lobing technique incorporated in this radar consists of transmitting two short pulses, a few microseconds apart, from an antenna which is a traveling wave array.

(2) Simultaneous-Lobing "Canted" Radar — Table IX

This radar may have application either as a "filler" radar to observe the area immediately under Stage III which the side-looking radar does not cover, or possibly as an alternative to the side-looking radar. Weight estimates are given for both applications.

(3) Frequency Scanned Radar — Table X

This radar provides a view of the area ahead of Stage III at the expense of increased antenna weight and display complexity. Although it uses advanced electronic techniques, it is felt to be completely practical within the MX-2276 developmental time schedule.

(4) Mechanically Scanned Radar of the Rascal Type, Table XI

The principle problem with this radar is the radome. Although special solutions are possible it is not as suitable for this application.

TABLE VIII. SIDE-LOOKING RADAR
(DOUBLE-PULSE LOBING)

Frequency	10 kmc	
Peak Power	120 kw each side	
Pulse Repetition Frequency	1200 pps	
Pulse Width	2 x 0.4 microsecond with 2-microsecond separation	
Receiver Bandwidth	3.0 mc	
Frequency Shift for Lobing	12 mc	
Antenna Length	22 ft	
Antenna Width	7 in	
Ground Coverage	10 to 50 n. miles lateral, each side	
Spot Size * (Considering resolution improvement of °2 from beam multiplication, attitude 200,000 ft)	Near Range 550 ft x 600 ft	Far Range 1000 ft x 250 ft
Resolution of Indicator	5,000 spots for 100 n. miles	
Film Requirement for 10,000 n. miles	7 in. x 50 ft	
Minimum Discernible Spot Target	22 square meters	
Weather Penetration	No thunderstorms.	
Weight Estimate		
Antennas	300 lb	
Radar and Indicator (CRT only)	250 lb	
Size Estimate		
Antennas (2)	24 ft x 1.5 ft x 1 ft each	
Radar and Indicator	6 cu. ft	

* NOTES: The underlined figures represent the resolution in azimuth. They are subject to a further improvement of a factor 3, achieved by indicator display technique (spot-positioning).

TABLE IX. CANTED RADAR
(FIXED CANTED ANTENNAS, SCANNING BY FORWARD MOTION, DUAL PULSE LOBING)

Frequency	16 kmc
Peak Power	100 kw each side
Pulse Repetition Frequency	1200 pps (Same as Side looking system)
Pulse Width	0.4 microsecond with 2-microsecond separation (Same as Side-looking system)
Receiver Bandwidth	3 mc
Frequency Shift for Lobing	12 mc
Antenna Length	12.5 ft
Antenna Width	8.5 in.
Cant of Antennas	45°
Squint of Beam	70° from forward end of antenna axis
Ground Coverage	0 to 11 n. miles each side (17 to 28 n. miles forward)
Spot Size *	Near Range: 290,000 ft^2; Far Range: 440,000 ft^2 (Distorted spot shape)
Resolution of Indicator) Film Requirement for 10,000 n. miles)	Indicator is integrated into Side-looking radar indicator
Minimum Discernible Spot Target	26 square meters
Weather Penetration	Heavy rain marginal.
Weight Estimate	
Antennas and RF System	250 lb
Indicator (CRT only)	50 lb Added to indicator of Side-looking radar
Size Estimate	
Antennas (2 required)	15 ft x 1.5 ft x 1 ft each
Radar and Indicator	3 cu. ft Added to Side-looking system
* See notes of Table VIII	

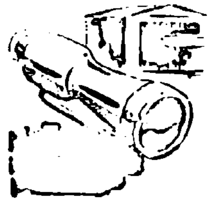
TABLE X. FREQUENCY SCANNED RADAR
(CONVENTIONAL PULSE RADAR)

Frequency	10 kmc	
Peak Power	80 kw	
Pulse Repetition Frequency	2500 to 4000 pps, depending on altitude	
Pulse Width	0.4 microsecond	
Receiver Bandwidth	3.0 mc	
Antenna Length	22 ft mechanical	
Labyrinth Length	110 ft	
Antenna Width	7 in.	
Scanning Frequency	2.0 scans per second	
Frequency Range for Scanning	±5% (9500 to 10500 mc)	
Maximum Scanned Angle	30° left and right	
Scans per Target	22	
Ground Coverage	60 miles wide, centered on ground track 15 to 50 miles ahead of bomber	
Spot Size (from 200,000 ft)		
	Range	Azimuth
On ground track, Near	600 ft	800 ft
On ground track, Far	300 ft	1600 ft
Edge, Near	400 ft	1200 ft
Edge, Far	300 ft	2000 ft
Minimum Discernible Target Radar Area	36 square meters	
Weather Penetration	No thunderstorms	
Resolution of Indicator	2000 spots square	
Film Requirement for 10,000 n. miles	70 mm x 50 ft	
Weight Estimate		
Antenna	500 lb	
Radar and Indicator (CRT only)	450 lb	
Size Estimate		
Antenna	24 ft x 2.5 ft x 1 ft	
Radar and Indicator	6 cu. ft	

TABLE XI. MECHANICALLY-SCANNED RADAR
(CONVENTIONAL PULSE RADAR, ROTATING DUAL PILLBOX ANTENNAS)

Frequency	36 kmc	
Peak Power *	300 kw *	
Pulse Repetition Frequency *	2500 to 4000 pps, * depending on altitude	
Pulse Width *	2 microseconds *	
Receiver Bandwidth	1.5 mc (to allow for Doppler shift)	
Antenna Length	6 ft	
Antenna Width	2 4 in. (each half)	
Rotation Rate (back-to-back antenna)	90° per second	
Time between scans	2 seconds	
Scans per target (at 20,000 ft/sec ground speed)	5	
Ground Coverage (from 200,000 ft)		
Range	18 miles to 50 miles	
Angle	70° left to 70° right	
Spot Size		
	Range	Azimuth
Near	2000 ft	1000 ft
Far	1200 ft	2000 ft
Minimum Discernible Target Radar Area	150 square meters	
Weather Penetration	Heavy clouds - no rain	
Resolution of Indicator	500 spots square (offset ppi)	
Film Requirement for 10,000 n. miles	35 mm x 10 ft	
Weight Estimate		
Antenna	300 lb	
Radar and Indicator (CRT only)	300 lb	
Size Estimate		
Antenna	7 ft diameter x 1 foot deep	
Radar Indicator	5 cu. ft	

* NOTE: These values are unreasonably high. They are given for comparative purposes only. They are required if ground painting must be seen as assumed in the preceding radar designs. Reduction of power possible only at the expense of size of minimum discernible spot target.



E. PROPULSION

1. General

The problems of selecting a propellant combination for this weapon system are somewhat similar to those encountered in design studies of other long-range vehicles. Therefore, previous studies have been reviewed with the following differences in mind:

a. The particular requirements of this weapon system differ somewhat from those of previous studies.

b. Additional propellant data and experience have become available since these studies were made.

c. The development time available may rule out certain propellants which require extensive preliminary investigation prior to the establishment of a preliminary design.

As a result of this review, it was found that liquid oxygen and JP-4, which were selected for long-range guided missiles now under development, were most feasible technically but would result in an undesirably large weapon. The study was then reduced to a search for a propellant combination having a higher specific impulse than liquid oxygen and JP-4 and which could be operationally available within the desired development time. The material discussed in this section is reported in detail in Reference 10.

2. Propellants Considered

Using liquid oxygen-JP-4 as a base line for comparison, the entire field of propellants was surveyed for those propellant combinations which offered promise of better performance. The various propellant combinations were examined on the basis of performance, combustion chamber temperature, regenerative cooling possibilities, potential availability and cost, toxicity, handling experience, and storage stability. The propellant combinations were first compared on the basis of specific impulse. Theoretical shifting equilibrium values of specific impulse for a chamber pressure of 300 psia and a nozzle exit pressure of 14.7 psia are shown in Table XII. Figures No. 52, 53, and 54 show the theoretical shifting equilibrium values of specific impulse chamber pressure with nozzle exit pressures of 14.7, 10.6, and 1.47 psia. The theoretical values of specific impulse were calculated for the various expansion ratios from the data in the table.

Additional increase in specific impulse resulting from reduced dissociation effects at the higher chamber pressures was neglected. Calculations show that an increase in specific impulse obtainable with an increase in combustion chamber pressure is almost entirely caused by the increased expansion ratio through the nozzle. Various physical properties of the propellants considered in this evaluation are briefly summarized in Table XIII. RFNA is included although its performance, even with hydrazine, is lower than desired. This was done because some consideration was given to having a more storageable oxidizer than liquid oxygen. The following propellants in various combinations were selected for further study:

Oxidizers

Liquid Oxygen
Liquid Ozone
Liquid Fluorine

TABLE XII. THEORETICAL SPECIFIC IMPULSE (SHIFTING EQUILIBRIUM) OF VARIOUS PROPELLANT COMBINATIONS

Propellant Combination	r*	I_{sp}**	P_B	I_d	T_c °K	T_c °F
F_2 & N_2H_4	2.2	314	1.32	414	4340	7352
F_2 & NH_3	3.0	311	1.17	364	4236	7160
F_2 & 60% N_2H_4 + 40% NH_3	3.5	310	1.29	400	4300	7280
70% F_2 + 30% O_2 & JP-4	3.8	298	1.26	376	4340	7352
O_2 & N_2H_4	0.83	272	1.01	275	3248	5386
O_2 & N_2H_2 $(CH_3)_2$	1.1	268	0.95	256	3100	5120
O_2 & 60% N_2H_4 + 40% NH_3	1.06	266	0.97	256	3150	5200
80% O_2 + 20% O_3 & JP-4	2.2	270	1.02	275		
O_2 & JP-4	2.4	262	1.01	265	3410	5678
O_2 & NH_3	1.25	262	0.86	225		
RFNA (Type III) & N_2H_4	1.25	246	1.26	272	2920	4796
RFNA (Type III) & N_2H_2 $(CH_3)_2$	2.7	242	1.24	263	3100	5120
RFNA (Type III) & JP-4	4.6	231	1.26	291	3080	5080

* Approximate mixture ratio for maximum specific impulse

** Chamber pressure 300 psia, nozzle exit pressure 14.7 psia, shifting equilibrium

Both ozone and fluorine can also be used as mixtures with liquid oxygen.

Fuels

JP-4
Dimethyl Hydrazine
Ammonia
Hydrazine
Boron Compounds

3. Oxidizers

a. Liquid Oxygen

This is the most widely known rocket oxidizer and offers good performance with many fuels in those applications where its limited storageability can be tolerated. Complete information necessary for the design of a rocket engine exists, and much experience in produc-

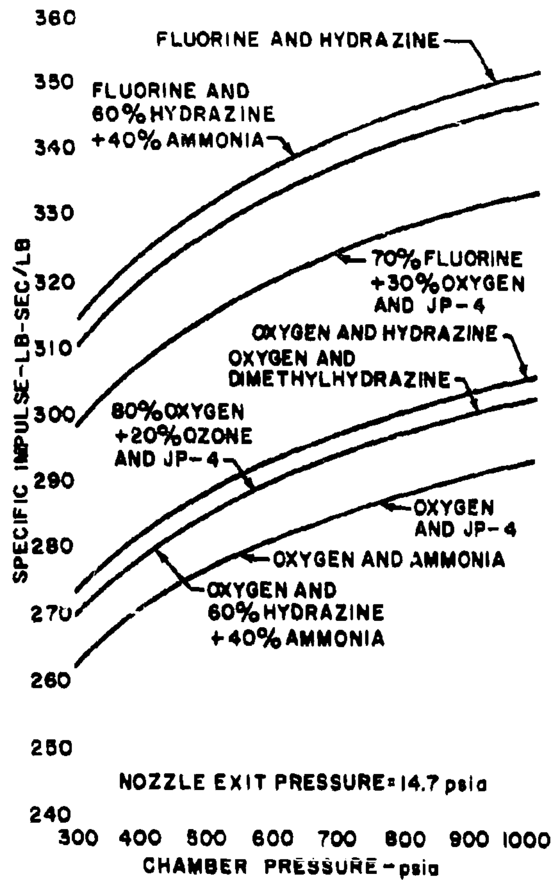


Figure 52. Theoretical Performance of Several Rocket Propellants (Expanded to 14.7 psi)

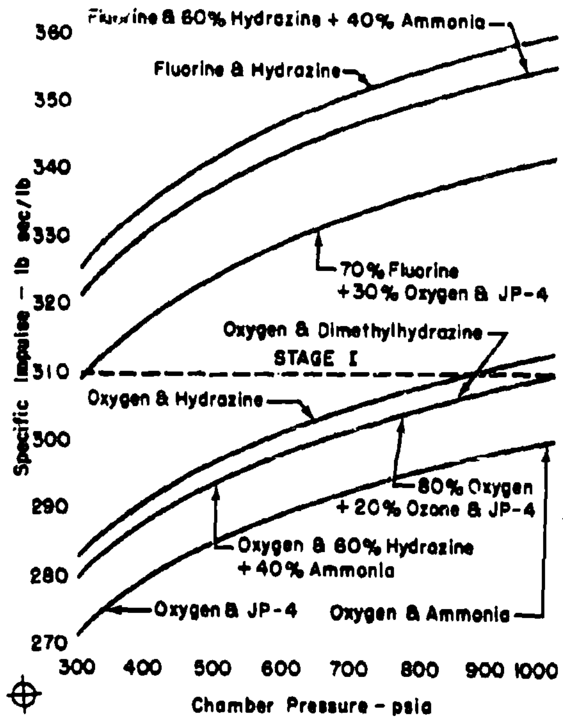


Figure 53. Theorctical Performance of Several Rocket Propellants (Expanded to 10.6 psi)

tion, handling, and testing has been obtained. It must be remembered that liquid oxygen is not a good coolant, and any fuel selected for use with it must be capable of cooling the thrust chambers of the engine. This places some limitation on propellant combinations which may be considered when liquid oxygen is used for the oxidizer.

b. Ozone

Ozone by itself is unstable and cannot be handled practically as an oxidizer. The addition of liquid ozone to oxygen has been investigated as a means for achieving a higher performance than for liquid oxygen alone. A concentration of 42.4 percent ozone in liquid oxygen, which is considered to be the upper limit for safe handling, provides an increase in performance of about 4 percent. The disadvantages of this oxidizer are (1) a serious problem of rapid decomposition in concentrations over this amount, (2) poor stability under local heating conditions, and (3) there is no production of ozone in other than laboratory quantities. Consequently, ozone is not considered seriously at this time either alone or mixed with oxygen.

c. Liquid Fluorine

Liquid fluorine is the highest performance oxidizer available for use with fuels con-

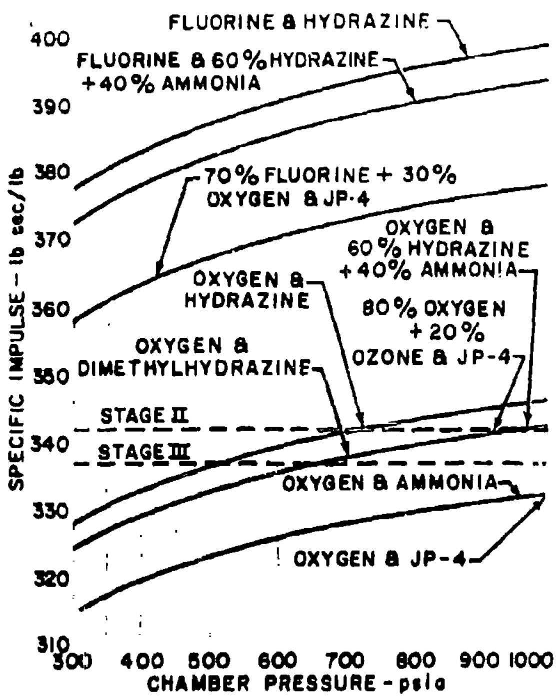


Figure 54. Theoretical Performance of Several Rocket Propellants (Expanded to 1.47 psi)

taining no carbon, such as hydrazine and ammonia. These fuels are not good coolants, particularly for use with fluorine where combustion temperatures are higher than with oxygen. With hydrocarbon fuels, fluorine alone offers a smaller performance advantage over liquid oxygen because the fluorine reacts only with the hydrogen, leaving the carbon to burn with oxygen. Therefore, with a hydrocarbon fuel such as JP-4 the optimum oxidizer appears to be a mixture of liquid oxygen and liquid fluorine. The variation in performance with increasing amounts of fluorine is shown in Figure 55. It can be seen that an increase in specific impulse of over 16 percent can be obtained by the addition of fluorine.

Because of this increase in performance over that obtainable with oxygen alone, a further study was made of the use of fluorine and oxygen with JP-4. Considerable experimental work has been done with fluorine-oxygen mixtures and JP-4 by both North American Aviation and NACA. Most of the problems relating to its use have been, or are being, investigated. There does not appear to be any serious factor which would prevent the use of fluorine in a rocket oxidizer, although the handling and operating procedures are more complex.

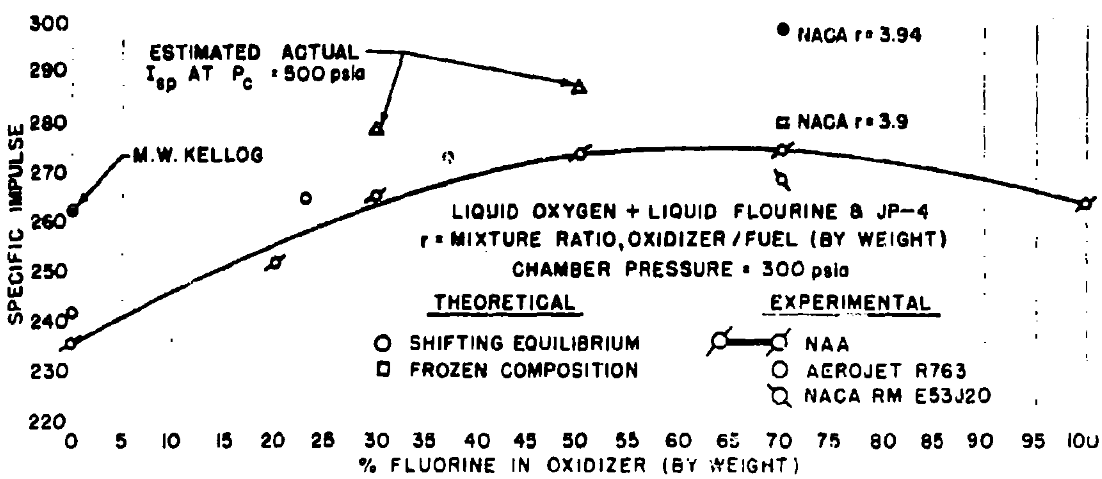


Figure 55. Specific Impulse vs Percentage of Fluorine by Weight in an Oxidizer for a Propellant Consisting of Oxygen plus Fluorine and JP-4

TABLE XIII. PHYSICAL PROPERTY DATA FOR PROPELLANTS UNDER CONSIDERATION

Propellant	FP °F	BP °F	Sp. Gr.	C_p Btu/lb-°F	P_c psia	T_c °F	Cost $/lb Present	Cost $/lb Future
F_2	-360	-306	1.54	0.29	808	-200	20.	1.00
O_2	-362	-297	1.14	0.39	731	-182	0.02	0.02
O_3	-418	-170	1.46		802	10	0.10	0.05
RFNA	- 65	140	1.58	0.42			0.10	0.10
N_2H_4	35	236	1.01	0.73	2130	717	2.50	1.00
NH_3	-108	- 28	0.66	1.07	1645	270	0.03	0.03
JP-4	- 76		0.78	0.48	310-510	575-710	0.015	0.015
$N_2H_2(CH_3)_2$	- 71	145	0.79	0.65	882	482	4.50	1.00

FP = Freezing Point

BP = Boiling Point

C_p = Specific Heat

The problem areas associated with the use of fluorine are discussed subsequently.

(1) Logistics

Fluorspar is presently on the critical materials list, chiefly because of the limited facilities available for reduction of the ore. A preliminary investigation indicates that sufficient fluorine for use only in the third stage could be obtained with a nominal expansion of existing facilities.

(2) Toxicity and Handling

The General Chemical Division of the Allied Chemical and Dye Corporation has a contract to develop the technology for producing and handling liquid fluorine. They have developed a trailer for transporting liquid fluorine and another capable of storing the liquid a maximum of 27 days without any servicing or loss. They have defined procedures for test operations, including the use of scrubbing chambers for decontaminating exhaust gases. The use of fluorine in only the third stage would eliminate the problem of releasing large quantities of toxic combustion gases during launch.

(3) Design Problems

Most of the design problems associated with the use of fluorine are related to the problems of material compatibility, resistance to corrosion, and maintenance of physical properties at extreme temperatures. Many of these problems are common to other rocket engine designs and many materials developed for them are suitable for fluorine use. Aluminum and stainless steels can be used with fluorine. The use of plastics for seals, etc., presents a problem since teflon and Kel-F are not satisfactory. Soft metal gaskets of aluminum and copper can be used; however, Monel is most suitable for reusable tanks, although stainless steel or aluminum can be used for certain types of service if properly designed. Thrust chamber components including injectors, have been made of aluminum and have proven satisfactory. Pump seals are under development, and several promising materials have been reported by North American Aviation. Regenerative cooling of a thrust chamber using fluorine-oxygen and JP-4 has not yet been demonstrated, but heat rejection rates measured by North American indicate that it is possible.

(4) Performance Attainable

The improvement in performance which is obtained by the addition of fluorine has been shown in Figure 56. These data represent the results of a large number of tests performed by more than one group and appear to be fairly well established. Theoretical calculations indicate that a 70 percent fluorine and 30 percent oxygen mixture will produce the highest specific impulse. The experimental data show that while the maximum impulse is obtained with the same mixture, the decrease in specific impulse with decreasing amounts of fluorine is not as great as the theoretical curve would suggest. Since such a small decrease in performance occurs, the 50 percent mixture has been selected for use in this study. This mixture has a lower combustion temperature and should therefore result in a more easily cooled thrust chamber. The optimum mixture ratio of oxidizer to fuel is lower with the 50 percent mixture which will also help to relieve the cooling problem.

4. Fuels

a. JP-4

Of the fuels proposed JP-4 is the most common, and the required information for its direct application to rocket design is readily available. It is, of course, already available in quantity production and has the lowest cost of any of the fuels considered. Although, as has been stated, a combination of oxygen and JP-4 does not present the desired high performance, in combination with liquid fluorine the performance is quite acceptable.

b. Pure Hydrocarbons

Substitution of a pure hydrocarbon for JP-4, or blending JP-4 and a pure hydrocarbon could improve the specific impulse. However, a significant improvement is obtained only by going to the light hydrocarbons which must be kept refrigerated or pressurized. It is doubtful if this complication is justified by the slight increase in performance obtainable. The coolant problem with such a system would be more difficult than with jet fuel.

c. The Hydrazine Fuels

There are three fuels available which can be used with liquid oxygen in the MX-2276. These are hydrazine, unsymmetrical dimethyl hydrazine, and the 60 percent hydrazine-40 percent ammonia mixture. The performance of these fuels at a chamber pressure of 300 psia expanding to one atmosphere is listed in Table XIV.

The performance of the hydrazine and unsymmetrical dimethyl hydrazine with liquid oxygen is greater than that of the hydrazine-ammonia mixture, the fuel suggested in the initial proposal. The bulk density and density impulse of the hydrazine-oxygen combination is substantially greater than the remaining two combinations, while that of the unsymmetrical dimethyl hydrazine-oxygen is approximately one percent greater than the proposed MX-2276 propellant. In addition, the performance and mixture ratio of the hydrazine-ammonia mixture and liquid oxygen are slightly different than previously used. The difference in performance of these fuels with liquid oxygen is not sufficient to base a selection on this parameter alone.

Since the MX-2276 thrust chambers must be regeneratively cooled, one of the propellants must be a satisfactory coolant. In the section on oxidizers, it is shown that oxygen is not satisfactory. Therefore, the fuel must be usable as a primary regenerative coolant. The coolant properties and related characteristics of the fuels as well as properties required for handling and logistics must therefore be considered.

TABLE XIV. PERFORMANCE OF HYDRAZINE FUELS*

Propellant	r	I_{sp}	P_B	Id = $I_{sp} \times P_B$
O_2 & N_2H_4	0.83	272	1.065	290
O_2 & $N_2H_2(CH_3)_2$	1.10	268	0.955	256
O_2 & N_2H_4 - NH_3 (60% N_2H_4)	1.06	266	0.951	253

(r = oxidizer/fuel by weight; P_B = bulk specific gravity of propellant)
*at a chamber pressure of 300 psia expanding to one atmosphere.

(1) Hydrazine

The major drawbacks to the use of hydrazine are its relatively high freezing point, its toxicity, its poor thermal stability, and its tendency toward accelerated decomposition in the presence of common materials such as mild steel. On the favorable side, hydrazine has a high density, a high specific heat, a low vapor pressure, and good storage stability. The results of experimental work are too limited in the range of operating conditions to justify the designation of hydrazine as a satisfactory regenerative coolant.

(2) Unsymmetrical Dimethyl Hydrazine

Unsymmetrical dimethyl hydrazine has a lower density, a slightly lower specific heat, and a higher vapor pressure than hydrazine. Its toxicity has not been fully determined, but is probably somewhat less than hydrazine. Its storage stability, under conditions of limited contact with air, is very good. It can be stored safely and without fear of deterioration or freezing over a wide range of temperature. Its thermal stability is much better than hydrazine. Its vapors are not explosive. On the basis of its superior physiochemical properties, unsymmetrical dimethyl hydrazine is probably more suitable for regenerative cooling than hydrazine. However, there is no experimental data on regenerative cooling with unsymmetrical dimethyl hydrazine to verify this.

(3) Hydrazine.Ammonia Mixture

The addition of ammonia to hydrazine results in a fuel with greater potentialities as a regenerative coolant than hydrazine.

Experimental data on regenerative cooling with the hydrazine-ammonia mixture are completely lacking although some tests have been carried out with ammonia. Tests at JPL show that regenerative cooling with ammonia can be accomplished. Since not all the tests were successful, a problem area exists even here with pure ammonia.

On the basis of the preceding information, no choice as yet can be made between the three hydrazine-type fuels. A program should be initiated to provide the basis for a logical choice between the three fuels. The heat transfer characteristics should be investigated in an apparatus designed for this purpose. Subsequently, thrust chamber firings should be made. Without substantial evidence such as obtainable from actual firings, any choice between the three fuels must be considered conjectural.

d. Boron Fuels

Consideration has been given to using fuels being developed under Project ZIP. Various boron compounds have been proposed from time to time for use as rocket fuels but have never found wide acceptance because of their cost, availability, and physical properties when compared with more common fuels. It is believed that at least a year of experimentation with these fuels in actual rocket engines, or at least thrust chambers, will be necessary before sufficient data are obtained to permit consideration of these fuels for MX-2276.

5. Propellant Combinations Reserved for Final Consideration

The preceding section discussed the various factors which were considered in the selection of a propellant combination for MX-2276. The combinations chosen are:

a. Liquid Oxygen and JP-4

b. Liquid Oxygen plus Liquid Fluorine and JP-4

The primary advantage of adding fluorine to the oxidizer is to reduce the size, weight, and, hence, the cost of the resulting weapon. Since it is actually the cost which is most important, a study was made to determine in some approximate manner the savings in cost which might be achieved by the addition of fluorine to the oxidizer.

A comparison based on the weight of the vehicle has been prepared for the several arrangements that are possible with these propellants. It is of interest that the cost of the propellants selected does not greatly influence the over-all cost for a significant number of missions. This is true because, although the propellants represent a large portion of the gross weight for any combination, their cost is much less than the airframe cost. As a consequence, the use of fluorine as an oxidizer results in the lowest over-all expense because it results in the smallest airframe. For the purpose of presenting a cost comparison the following values have been assigned:

Stage I and III (Manned Vehicle)	\$50 per pound
Second Stage (Unmanned)	\$35 per pound
Liquid Fluorine	\$ 2 per pound
Liquid Oxygen and JP-4	\$.02 per pound

The airframe values shown are arbitrarily chosen for the purpose of presenting a comparative evaluation and do not necessarily represent an estimated cost.

The possible arrangements in the order of their desirability from the standpoint of the size of the vehicle and the cost of the program are:

1. All stages: Liquid Oxygen-Fluorine and JP-4

2. Third stage: Liquid Oxygen-Fluorine and JP-4

   Second and First stages: Liquid Oxygen and JP-4

3. All stages: Liquid Oxygen-JP-4

Since the propellant combination, liquid oxygen and ammonia plus hydrazine, was presented in the first proposal, it has been retained in the figures accompanying this report for the purposes of comparison with the data included in the previous reports.

Further studies and more recent information on large rocket engines have resulted in reduction in the estimated weights of the power plants for MX-2276 of approximately 25 percent over that shown in the initial reports. This reduction is reflected in the accompanying figures.

For the purpose of permitting comparisons on a percentage basis, Figure 56 has been prepared. Since the oxygen and JP-4 bomber is the most expensive, it has been assigned the value of 100 percent. The cost of the other arrangements can consequently be determined as a percentage of the maximum probable cost for a desired number of missions.

The cost differential between the arrangement using liquid oxygen and JP-4 in all three stages and that wherein fluorine and oxygen and JP-4 are used in the third stage with oxygen and JP-4 in stages one and two is about \$135,000 per mission. This is relatively insignificant. However, if extended over twenty

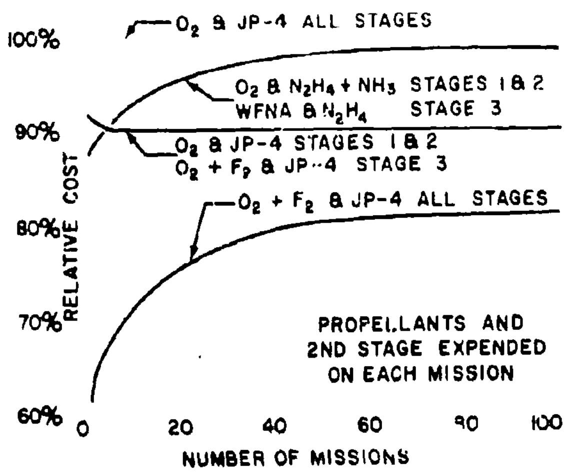


Figure 56. Relative Cost of Various Bomber Arrangements for a Number of Missions

missions and added to the difference in cost recoverable hardware (stage one and three) the savings become significant.

Savings:

20 Missions at $135,000	$ 2,700,000
Stage I and III Initial Cost Differential	733,000
Total Savings for One Bomber at 20 Missions	$ 3,433,000

Since it is logical to assume that more than one bomber would be operational, this figure may be multiplied by the number of bombers. If twenty bombers are assumed, the savings are 20 x $3,433,000 or $68,660,000. It is apparent that the cost of the fluorine program would be amortized over a short period.

If oxygen and fluorine are used in all three stages the decrease in cost of bombers and missions is even more impressive.

F. GLOBAL WEAPON CONCEPT

1. General

The material in this section is reported in detail in Reference 10.

By increasing the speed of Stage III from high hypersonic to circular velocity certain advantages are obtained which make such a measure worthy of consideration. These advantages include:

a. At or near circular velocity the centrifugal effect can be utilized to obtain lift instead of aerodynamic forces. By this means the drag is appreciably reduced and one or more circumnavigations of the globe become possible with the use of very little additional energy.

b. The foregoing advantage makes it possible that the take-off and landing can both be accomplished within the continental United States and independence from foreign bases is achieved.

c. Descent can be made at maximum lift coefficient rather than maximum lift-over-drag ratio as required for greater aerodynamic range. It may be possible by this means to somewhat reduce the aerodynamic heating.

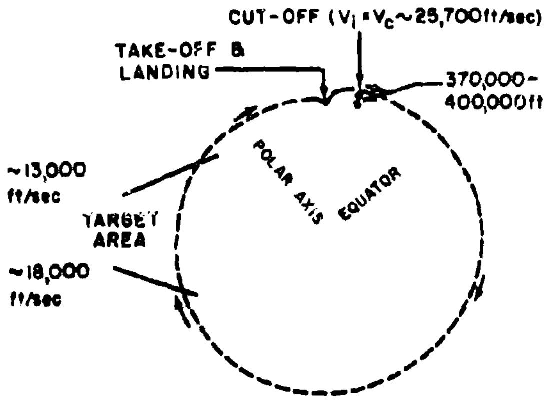


Figure 57. Path Type I: Horizontal Delivery at Circular Velocity; One Circumnavigation; Spiral of Descent; not Necessarily at L/D_{max}

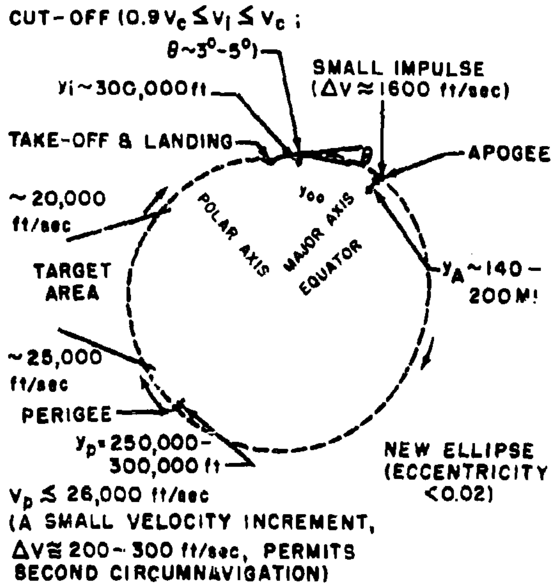


Figure 58. Path Type II: Low Eccentricity Ellipse with Perigee Near Target Area

2. Flight Paths

In order to learn more about the requirements for global flight a preliminary investigation of a manned global weapon system capable of one or more circumnavigations of the globe was made. Four types of flight paths were considered. A general description of each of these paths follows, along with some factors which must be considered when evaluating their relative merits. A preliminary weight estimate is also included.

a. Path I (Spiral) Figure 57

The vehicle is delivered horizontally at an altitude of about 400,000 feet at a local circular velocity of 25,700 feet per second. The body does not fully escape from the atmosphere and therefore loses speed throughout the cruise. As a result, the descent follows a spiral path. When entering the denser atmosphere, the aerodynamic portion of the flight is made at maximum lift coefficient rather than at maximum lift-over-drag ratio. The take-off is in the direction opposite to the shortest great circle route to the target area. In this manner the vehicle appears over the target area at a velocity and altitude comparable to that of MX-2276. If more than one revolution is desired, the initial altitude must be increased, probably to 500,000-600,000 feet.

b. Path II (Elliptic) Figure 58

The vehicle is delivered not horizontally, but at a small trajectory angle at near-circular or circular velocity. The resulting free-flight path is an ellipse whose apogee lies at an altitude between 700,000 and 1,000,000 feet (110-160 nautical miles) but whose perigee would lie inside the earth if the vehicle were permitted to follow this path beyond the apogee. Therefore, a small impulse (burst of power) is needed so that the vehicle enters a new ellipse whose perigee lies at an altitude of about 300,000 feet. The orientation of the ellipse is such that the perigee lies over the target area. Since most of the elliptic path is outside the atmosphere, the velocity loss between cutoff point and perigee is small and a second circumnavigation can be effected with comparatively little additional energy.

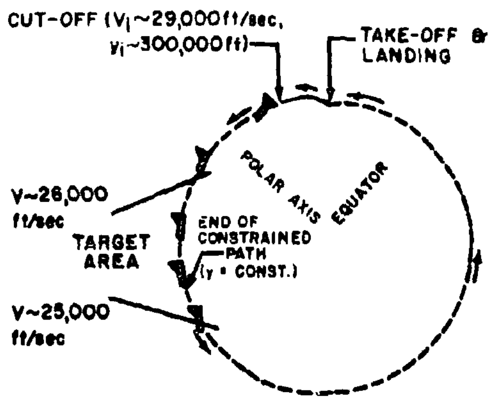


Figure 59. Path Type III: Constrained Path

c. Path III (Constrained) Figure 59

The vehicle is delivered horizontally at a cut-off speed which exceeds the local circular velocity. The excess, referred to as elliptical excess, sends the vehicle into an elliptic orbit unless it is forced, by negative lift, to stay at the initial altitude and to follow a circular path. In order to accomplish this, the vehicle must initially fly at a negative angle of attack, and the initial altitude must be low enough to permit action of a sufficiently strong aerodynamic force. The angle of attack returns to zero as the speed is reduced to circular velocity. Thereafter, the path is similar to the spiral (Path I).

d. Path IV (Sustained)

Again the vehicle is delivered horizontally, and the cut-off speed is sustained at constant altitude by a small sustainer rocket motor overcoming the comparatively low drag. So far, circular velocity has been assumed as cut-off speed; the effect of a lower velocity should be investigated. If circular velocity is used, the cruising altitude will have to be of the order of 450,000 feet to reduce the drag sufficiently, and, hence, the thrust and propellant consumption per revolution, also.

3. Discussion of Flight Paths

Table XV has been prepared in order to assist in the evaluation of the various flight paths.

The energy requirement, based on the number of revolutions assumed, is given in terms of the ideal velocity. Assuming that the total energy required to send the given rest mass, m, over the respective flight path, is expressed in terms of equivalent kinetic energy, then the ideal velocity is defined by

$1/2\ mv^2_{ideal}$ = kinetic energy of rest mass + potential energy of rest mass + energy lost due to gravitational pull during powered flight + energy lost due to drag + energy lost due to steering.

Therefore, v_{ideal} represents the velocity which the vehicle could obtain under the ideal loss-free conditions of propulsion in gravity-free vacuum. This is the velocity for which the over-all mass ratio of the vehicle must be laid out. Table XV shows that the energy requirements are about the same for all paths, with the exception of the constrained path.

For ignition and power plant operation, the number of individual propulsion periods is of importance. It is desirable to keep this number at a minimum for reasons of efficiency and reliability of the ignition process. In this respect, the elliptic path is less favorable, because it requires a second propulsion period at the apogee.

The characteristic flight path conditions indicate that extreme altitude is required for the elliptic path and extreme speed for the constrained path.

The conditions over the target area are most favorable for the spiral path, yielding lowest altitude and speed without, however, rendering the vehicle vulnerable to enemy defense. The elliptic and constrained paths show near circular velocity over the target area at an altitude of about 300,000 feet. The highest altitude over the target area is obtained for the sustained path.

TABLE XV. FLIGHT PATH CHARACTERISTICS FOR GLOBAL WEAPON SYSTEM

Subject Field	Parameter	Path			
		Spiral	Elliptic	Constrained	Sustained
Energy	Number of revolutions	1	1	1	≥ 2
Requirement	Ideal velocity (1000 ft/sec)	29-30	30	32-33	30
Ignition and Power Plant Operation	Propulsion Periods	1	2	1	Continuous
	Initial Velocity (1000 ft/sec)	25.7	25-25.7	28-29	25.7
Characteristic	Max. Velocity (1000 ft/sec)	25.7	26-26.5	28-29	25.7
Flight Conditions	Max. Altitude (1000 ft)	400-450	700-1000	290-300	450
Conditions over	Velocity (1000 ft/sec)	18-16	26	26-25	25.7
Target Area	Altitude (1000 ft)	200-180	300	300-290	450
Load Conditions	Normal load during coasting or cruise (g)	0-1	0-1	-0.2 -1	0-1
Stability and Control	Control Systems	Dual	Dual	Aerodynamic	Dual

The load conditions normal to the instantaneous flight path direction which result from the flight path configuration proper (i.e., excluding load conditions during powered flight or due to maneuvering) are small in all cases and never anywhere near the values obtained for the skip path. The negative sign in the case of the sustained path indicates that, during flight at greater than circular velocity, the apparent weight vector points away from the center of the earth.

With respect to stability and control requirements, the number of control systems needed is indicative of the weight and relative complexity, as well as reliability to be expected. Obviously, a vehicle operating inside and outside the atmosphere must have two types of control - aerodynamic control within the atmosphere, and jet control for vacuum flight. This dual control is required for all paths except the constrained path.

4. Weight

A preliminary study of the take-off weight of the global system was conducted to evaluate the energy requirements of these flight paths in terms of vehicle weight. Since this study was of a preliminary nature, several simplifying assumptions were made as follows:

a. Payload for Stage III - 4200 lb

b. (1) $\frac{\text{Actual Velocity}}{\text{Ideal Velocity}}$ Stage I = 0.75

(2) $\frac{\text{Actual Velocity}}{\text{Ideal Velocity}}$ Stage II = 0.94

(3) $\frac{\text{Actual Velocity}}{\text{Ideal Velocity}}$ Stage III = 0.98

c. Propellants - (50% F_2 + 50% O_2) + JP-4

(1) Specific impulse Stage I - 316 sec

(2) Specific impulse Stage II - 361 sec

(3) Specific impulse Stage III - 361 sec

d. $\frac{\text{Dry Weight}}{\text{Take-off Weight}}$ Stage I = 0.20

$\frac{\text{Dry Weight}}{\text{Take-off Weight}}$ Stage II = 0.15

$\frac{\text{Dry Weight}}{\text{Take-off Weight}}$ Stage III = variable

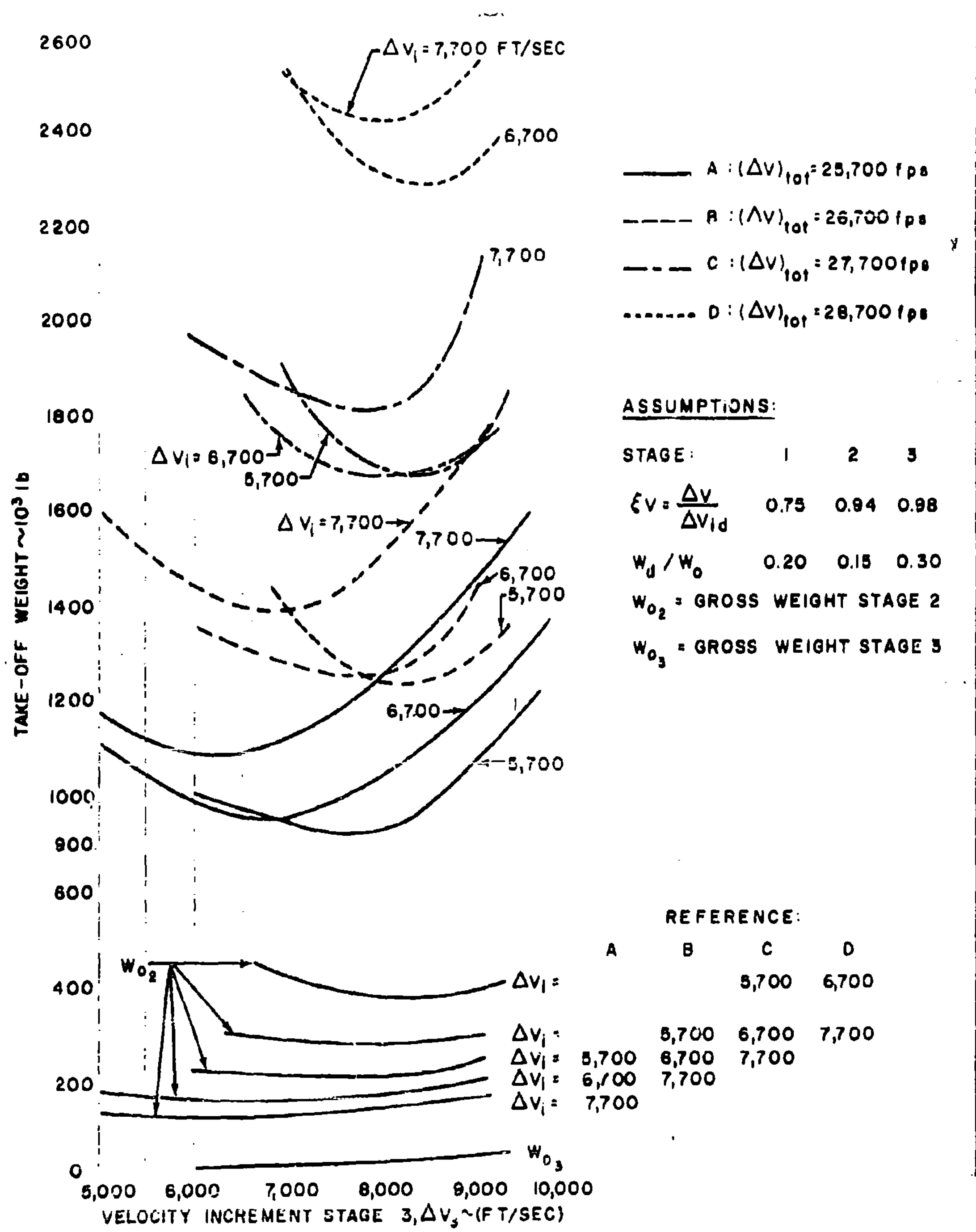


Figure 60. Example of the Effect of Energy Distribution Among the Stages on Take-Off Weight

The investigation consisted of varying the velocity increments of the second and third stage for a given first stage velocity increment and repeating this for three different first-stage velocity increments. By this means three curves were obtained for a given cut-off velocity and four cut-off velocities were investigated. Figure 60 shows the results of this investigation for a ratio of dry weight to take-off weight for Stage III of 0.30. This figure also shows the associated weights of the second and third stage as a function of the third stage velocity increment. The location of the minima indicate a trend toward lower take-off weights with decreasing first stage velocity increment, down to about 5700 feet per second. From the viewpoint of economy it is significant that the weight of the second stage increases with decreasing first stage velocity increment and, hence, more hardware must be thrown away at the lower take-off weight. This trend is generally true with multistage rocket vehicles, i.e., the more of the hardware that is made expendable, the lower will be the take-off weight.

From this graph and additional studies at different dry weight to gross weight ratios of the third stage, the weight curves in Figure 61 have been obtained showing four bands of overall take-off weights for four different values of dry weight to take-off weight for the third stage, plotted as a function of the actual velocity of the third stage. The limits of each band represent the values found for the "upper minimum" and "lower minimum" for each set of three different stage velocity distributions such as shown in Figure 59.

If a dry weight to gross weight ratio of 0.3 for the third stage is accepted as a reasonable approximation it can be seen that for the spiral path a take-off weight of the global weapon system of approximately 1,000,000 pounds can be expected. Similar weights can be anticipated for the elliptic path as well as

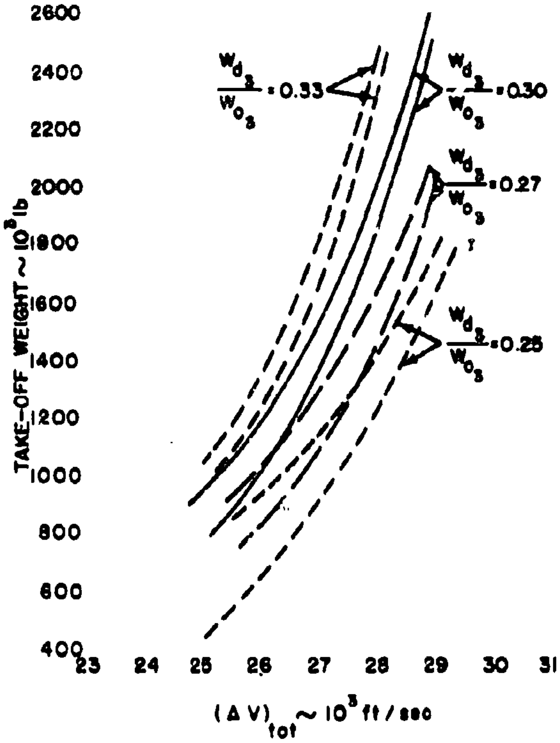
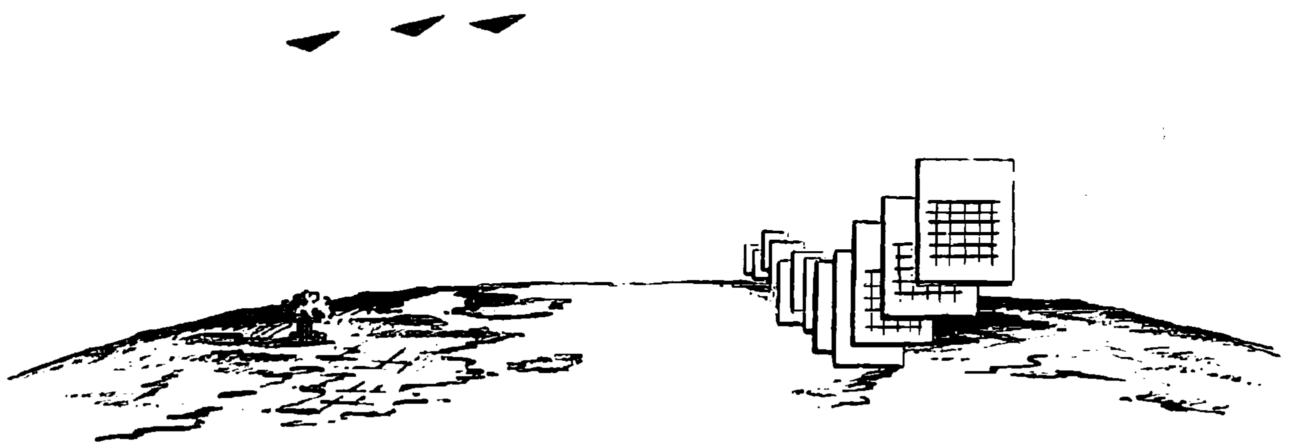


Figure 61. Take-Off Weight as a Function of Cut-Off Velocity

the sustained path for operation at 450,000 feet altitude and circular velocity. The constrained path however requires weights of 2.3 to 2.4 million pounds.

If the results of the weight study are considered with the flight mechanics considerations, it appears that global systems of about one-million pounds take-off weight are feasible and that the spiral or sustained paths are the most attractive unless there is a restriction on the maximum desirable altitude.

V SYSTEM TRENDS



This section contains a resumé of the work performed in a configuration study sponsored by the Bell Aircraft Corporation. The study was intended to augment the Air Force-sponsored study program, which did not provide for design work, by applying current results to new configuration studies. Because of the state-of-the-art, the configurations which have evolved are considered to be a next step rather than a final design.

The method of approach used in the study was first to design the bomb, then the bomber or final stage, and finally the booster stages, considering both as expendable and recoverable (glider) first stages. Since each in order is the payload of the next stage, this is the logical sequence to follow.

A. BOMB

Each additional pound in the bomb increases the take-off weight by more than 30 pounds, and the size of the bomb influences both size and weight of the bomber.

Consequently, minimum size and weight are desired for the bomb, yet it not only must have satisfactory stability and controllability characteristics, but for a short period it may be exposed to higher temperatures than the bomber. The high temperature peak, which is considerably higher than immediate postlaunch temperatures, is reached 80 to 120 seconds after launch for launch speeds of 13,000 and 18,000 feet per second, respectively. The temperature then drops abruptly and after about 30 seconds ceases to be a problem.

It should be noted that if large air brakes can be provided, the temperature peak can be reduced substantially. However, the various problems presented by the brakes seem more difficult to solve at this time than the problems brought on by the high temperatures.

These temperature considerations, plus stowage aboard the bomber, make the use of conventional fins or aerodynamic surfaces appear undesirable. In place of fins it seems more practical to use a self-stable body, which is controlled by flap-type surfaces located at the rear of the body. Small hydrogen peroxide rockets would be used for roll control. This self-stable body has a conical-ogival contour with the center of gravity being made to lie ahead of the center of pressure by suitable placement of the warhead and other large masses. Since the body is stable, the flaps are in the lee of the body when extended. Preliminary indications are that the necessary flap deflection angles can be made sufficiently small so that deflecting the flap will not produce a temperature problem on the flap itself. But the temperature on the windward side of the body does increase, of course, with angle of attack. It may prove desirable, therefore, to restrict control to the portion of flight just beyond the peak temperature region in order to minimize temperature problems. Even so, using a 2g turn, the bomb can be steered about 5 nautical miles to either side of the initial bomb aim-point in the last 40 seconds.

Preliminary structural studies show that water cooling is probably the most satisfactory heat protection method for the bulk of the body, with the exception of the nose, for which a completely satisfactory solution has not yet been evolved.

B. BOMBER

Once the size and weight of the bomb were determined, the weight of all the equipment, including items such as the navigation system, search radar, communication transmitters and receivers, cabin furnishings, etc., was estimated and design work on the bomber was begun.

In these studies, as differentiated from earlier configuration work, the total structural weight was estimated for each specific configuration, based on actual geometric characteristics of the wing and body. In making these estimates, the latest data from MX-2276 structural and heat insulation studies data were used.

The design objectives set for the bomber are largely the same as before. Propellants and bomb are located to minimize center of gravity travel on expending the propellants and bomb. The configuration must be satisfactorily stable and controllable throughout the entire flight regime. In addition, landing characteristics must be satisfactory.

The main differences from previous designs are shown in the wing-tip-mounted vertical surfaces and the high-wing-body arrangement. These changes are considered to improve directional stability substantially over the more conventional body-mounted upper vertical tail and low-wing-body arrangement.

In addition to the requirement for small center of gravity travel on expending the bomb to keep trim drag to a low value, it appears that rearward ejection of the bomb from the body base will be the most practical procedure considering both the temperature problems encountered with open bomb bays, etc., and the moments and separation problem associated with downward ejection of the bomb.

C. BOOSTER STAGES

The design work on booster stages has been concerned so far with two types of two-stage vehicles; one type is similar to the previously shown MX-2276 boosters; that is, the first stage is winged and is landed in a conventional manner, while the second stage is expended; the other booster arrangement expends both stages. The main advantages of all-expendable booster arrangements are low first cost and greater adaptability to multistage configurations which exhibit simpler separation problems. The recoverable first stage offers promise of lower over-all cost provided sufficient re-use is possible. These factors remain to be evaluated in a systems analysis.

The use of round bodies which permit integral tank-body construction has been emphasized in the present study, since this procedure tends to lead to minimum stage weight and consequently, take-off weight. With integral construction, the boosters are mainly propellant tanks with the rocket power plant housed in a rear fairing. For the all-expendable booster configuration, the first stage may be divided into two sections and placed on either side of the second stage; the bomber and second stage are then arranged in tandem. The use of two bodies for the first stage makes it possible for all the booster sections to be no more than 10 feet in diameter and less than 60 feet in length. These sections can be shipped in standard railroad freight cars.

The winged first stage can employ only one body, and with practical length-diameter ratios becomes quite long. Hence, a three-stage configuration utilizing a tandem arrangement of the bomber and second stage, which is carried pick-a-back on the first stage, may be the most suitable, when minimization of over-all length is emphasized. A favorable outcome is that the wing of the first stage provides an aerodynamic stabilizing moment.

As mentioned before, the second stage has been considered expendable. The reasons for this assemption are as follows: it is much smaller than stage one and thus more likely to be justified as being expendable; the research and development cost of a manned second stage vehicle will be high and probably difficult to justify; and, finally, the larger over-all size and weight of the three-stage vehicle, with all stages recoverable, would make logistic, maintenance, and ground handling problems more difficult.

It is in order to point out problems which arise when the booster stages are manned and recoverable. First of all, wings must be provided. For the second stage, which reaches very high Mach numbers, the wing would be similar in construction to the bomber wing. Cabin furnishings, control, communication systems, and auxiliary power sources would have to be provided. An extendable landing gear is required, and the structure must be able to withstand landing loads. Purging or inerting systems for the tanks may also be required. In addition, both stages are so large that unpowered landings may be hazardous. Consequently it may be considered necessary to provide auxiliary propellant tanks and rocket motors for landing.

D. FUTURE TRENDS AND STUDIES

In keeping with the objective of re-using as much hardware as possible, the present study has considered bomber configurations which do not discard the propellant tanks and rocket power plants on burnout. However, recent studies have emphasized the pyramiding characteristics of dead weight in the bomber, and it now appears desirable to examine in detail the over-all results obtained when the bomber propellant tank and power plants are expended. For example, to house the tanks and power plants in the fuselage means that the fuselage is larger and heavier, and has more drag than one that does not contain these items. The wing also is larger and heavier because it must support more fuselage weight as well as the weight of the tanks and power plants.

The over-all weight that could be saved in the bomber when the tanks and power plants are expended will be reflected more than 30-fold in take-off weight, and thus may prove to be an economical thing to do.

Another promising idea is the use of more than two boost stages. This is particularly applicable for all-expendable booster configurations for two reasons: first, booster arrangement is more flexible, and second, it is more important to minimize the weight of expended hardware.

Other items which should be examined thoroughly in future studies are the use of large wing sweep and perhaps a drooped wing leading edge and a drooped body nose to alleviate temperature and heat flow problems. Preliminary indications are that favorable results may be obtained by employing these measures.

VI CONCLUSIONS AND RECOMMENDATIONS

A. CONCLUSIONS

The results of the past year of study have, in general, verified the initial concept of this weapon system. No new major problem areas have been revealed and progress has been achieved in most of the areas of investigation required to support a program for the design and development of this weapon system.

The Bell Aircraft Corporation sponsored preliminary configuration design work has shown the importance of a two-pronged attack for this program, the first being configuration design and layout, and the second being evaluation of the experimental and analytical investigation of the phenomena associated with the flight conditions of this weapon system in order to support or modify the analysis of the system characteristics and performance.

The following conclusions can be listed as a result of the work performed during the past year under the terms of AF 33(616)-2419.

1. Crew

a. The functions required of the crew of this weapon system are well within the abilities of a human.

b. The environment necessary to maintain the crew of the bomber in sufficient comfort to perform the functions required can be provided by the methods outlined.

2. Aerodynamics

a. For glide performance the range and altitude are substantially the same as initially predicted except that the maximum altitude is reduced from 259,000 feet to 214,000 feet at a velocity of 22,000 feet per second. With the bomb aboard for the entire flight, the range is slightly greater than for a typical mission where the bomb is dropped.

b. The inclusion of shock wave-boundary layer interaction on performance shows both lift and drag are increased with a negligible effect on the lift-drag ratio (hence, glide performance). However, the effect of the additional lift which will permit higher altitudes to be attained has not been included in the present glide performance.

c. The effect of earth rotation on glide performance indicates large effects on range for equatorial flight. Flight in the easterly direction would increase the range by 2200 miles while in the westerly direction, it would be decreased by 1200 miles.

d. Attempts to reduce the viscous heating by programming the flight path to higher altitudes using higher lift coefficients than for L/D_{max} or by utilizing a partial lifting path were not profitable.

e. Temperatures on the upper and lower surfaces of the wing and body for glide flight conditions show that the lower surface temperatures are higher indicating values of 1800°F and lower from the 2-foot point aft. Leading edge temperatures are higher and indicate the need for cooling possibly by transpiration.

f. Shock wave-boundary layer interaction has a negligible effect on the lower surface temperatures. However, it does increase the upper surface temperatures by as much as 700°F near the leading edge (neglecting the effect of increased altitude due to increased lift). This increase falls off rapidly as distance from the leading edge increases. Resulting upper surface temperatures are still less than the lower surface values.

g. From a preliminary analysis, satisfactory stability and control can be obtained up to a Mach number of 20. Shock wave-boundary layer interaction effects on lift and moment values must be included along with effects of earth rotation.

h. Free flight as well as ground test facilities are available for investigation of the very high-speed and high-altitude phenomena of this weapon system.

i. Equilibrium dissociation of the air in the boundary layer should not affect the skin friction and heat transfer values appreciably as long as the local free stream and wall temperatures are below the dissociation value. Such a condition is the practical case.

j. Radiation of heat from the hot boundary layer air to the skin appears to be an important quantity which should be included in the boundary layer and heat balance equations. A theory to compute the emissivity of the hot air for such analyses has been developed.

k. A theoretical method has been developed for predicting the transpiration coolant requirements for hypersonic flight conditions using air as the coolant.

3. Structures

a. A survey of materials indicates that, for the outer panel, suitable materials are available for equilibrium temperatures up to 1800°F. Above 1800°F material fabrication techniques must be developed, and above 2400°F material development is required.

b. The study of cooling and insulation shows that the optimum structural design for the primary structure from a weight standpoint consists of a heat protection method combining insulation and cooling. The primary structural areas comprise a major part of the structural weight.

c. The heat protection consists of double-wall construction wherein the outer wall is constructed of heat-resistant material and is separated from the inner wall by insulation. A coolant system is contained inside the inner wall to maintain the proper structural temperature.

d. Tests of a sample of the structural configuration on an elemental basis for thermal warpage, strength, thermal conductivity, and coolant requirements verified the analytical estimates except for the amount of coolant required. This single discrepancy is believed to be due to the improvised test technique employed.

e. The portions of the bomber airframe which were not investigated in detail (such as leading edge and nose) and will require design, development, and test represent only a few percent of the total surface area.

4. Navigation and Control

a. An all-inertial navigation system for the bomber will provide an accuracy of 4,000 feet CPE for a range of 8,000 nautical miles. Component and instrumentation accuracies required are realizable in the development time period.

b. A similar inertial system for guiding the bomb will provide an accuracy of 1500 feet CPE for a range of 300 nautical miles.

c. Practical methods can be devised for comparing the inertial information with the predicted path and the radar or optical data on check points to monitor the midcourse navigation phase and provide the information for terminal guidance.

d. A K_u band radar using the technique of simultaneous lobing and incorporating a side-looking linear antenna array will provide a ground spot size resolution of 250 to 1,000 feet from an altitude of 200,000 feet, depending upon the range.

e. Generally speaking, the radar system can be designed and developed using the present state-of-the-radar-art with no more than routine problems or mechanical aspects of design.

f. Fused silica is a suitable radome material for this high temperature application. Other materials such as Fosterites, Steatites, Wollastonites, Aluminas, and Vycor are also satisfactory.

5. Propulsion

a. Current research and development on large rocket engines of the 120,000-pound thrust class using O_2 and JP-4 as propellants can be used for the propulsion units of the boosters. These engines are already being tested.

b. For the bomber, the higher performance propellant combination of $O_2 + F_2$ and JP-4 is recommended because of the reduction in over-all size and weight of the vehicle.

c. A propulsion unit using this improved propellant could be developed within the time scale of this program provided the engine development was initiated immediately.

d. Some expansion of fluorine manufacturing facilities would be required for the prosecution of this program.

6. Global Weapon System

A preliminary study of the global weapon system indicates that such a weapon is sufficiently feasible and advantageous to warrant further study.

B. RECOMMENDATIONS

The recommendations for future work are detailed in the various applicable technical reports (References 6-11). A summary of the most important recommendations is presented in Table XVI.

TABLE XVI. RECOMMENDATIONS FOR FUTURE WORK (Sheet 1 of 4)

Type	Item	Description	Suggested Facility
1. Crew protection	Full pressure suit	Expedite development of full pressure suit providing pressurization, respiration, anti-g, ventilation, and anti-exposure features.	Aero-Medical Laboratory Wright Air Development Center
2. Crew navigation presentation	Integrated presentation	Design of integrated radar, visual, and inertial presentation.	Bell Aircraft Corporation, Aero-Medical Lab., WADC, and optical equipment vendor
3. Atmosphere	Atmospheric characteristics at high altitudes (up to 80 n. mi.)	Determine pressure, temperature, gust magnitude and frequency, cosmic radiation.	High Altitude Research Facilities
4. Performance	1. Skin friction	Improve methods of prediction to include boundary layer radiation, transpiration cooling, wall temperature effects, transition, and three dimensional effects.	Analytical at Bell Aircraft Corp. Experimental at CIT hypersonic wind tunnel NACA PARD or Hypersonic Test Vehicle Flight Test
	2. Pressure forces	Improve methods of prediction including three dimensional effects.	Analytical at Bell Aircraft Corp. NACA Hypersonic Tunnel Ballistic Facilities
	3. Shock boundary layer interaction	Determine effects upon items 1 and 2 in more detail.	Analytical at Bell Aircraft Corp. Princeton Hypersonic Tunnel CIT Hypersonic Tunnel
5. Flight Mechanics	Maneuverability	Determine maneuverability limitations and correlate with navigation system.	Bell Aircraft Corporation

TABLE XVI. RECOMMENDATIONS FOR FUTURE WORK (Sheet 2 of 4)

Type	Item	Description	Suggested Facility
6. Aerodynamic Heating	1. Viscous	Investigate effects of additional viscous terms on transition, boundary layer, wall temperature, and three dimensional effects.	Analytical work at Bell Aircraft Corp. Various hypersonic wind tunnels
	2. Leading edge condition	Expand present investigation to include effects of sweepback and leading edge profile.	Analytical work at Bell Aircraft Corp. NACA Hypersonic Tunnels Polytechnic Institute of Brooklyn Aeronautical Laboratory Hypersonic Tunnel (PIBAL)
	3. Transpiration cooling	Estimate coolant requirements including effects of angle of attack and surface temperature. Include coolants other than air.	Analytical work at Bell Aircraft Corp. PIBAL Hypersonic Tunnel PIBAL Hydrogen-Oxygen Tunnel
7. Stability and Control	Equations and parameters	Establish methods for predicting dynamic stability parameters for hypersonic viscous flow.	Bell Aircraft Corporation
8. Criteria and Loads	Design factors	Establish design factors with respect to thrust, pressure, creep.	Bell Aircraft Corporation or Subcontractor
9 Cooling of primary structure	1. Coolant requirements	Determine amount of coolant required considering proper temperature distribution and time.	Bell Aircraft Corporation University of Florida
	2. Coolant distribution	Design and test coolant distribution systems including piping, metering, values, feed spacing and wick geometry.	System analysis and development at Bell Aircraft Corporation. Subcontract coolant feed system.

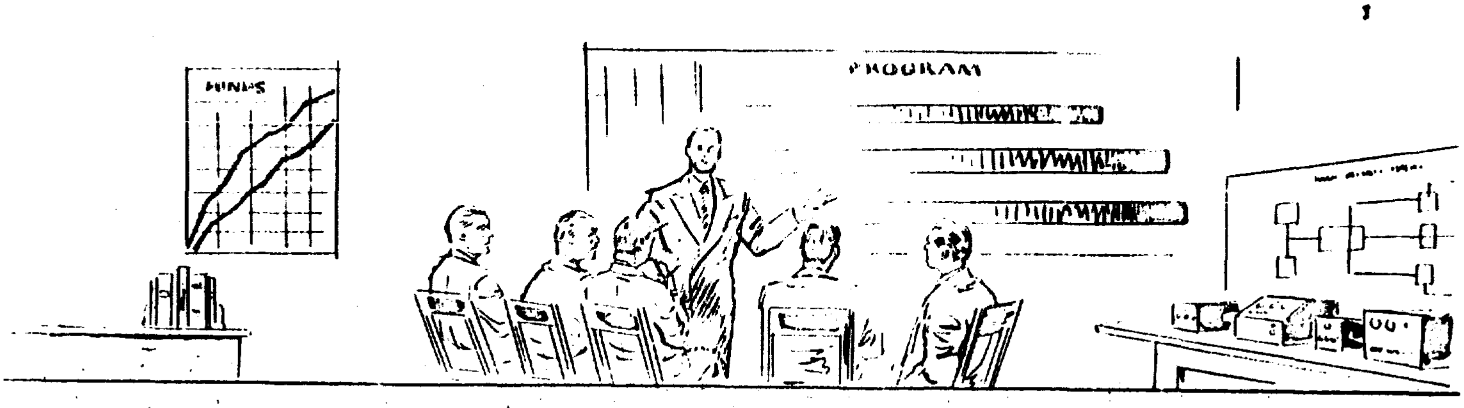
TABLE XVI. RECOMMENDATIONS FOR FUTURE WORK (Sheet 3 of 4)

Type	Item	Description	Suggested Facility
10. Double wall construction	1. Super alloy outer wall panels	Continue development of metallic outer wall panels for operation at high temperatures.	Metal fabrication at Bell Aircraft Corporation. All ceramic work by vendors. Physical and mechanical tests by NACA, vendors, or universities.
	2. Ceramic coated metallic outer wall panels	Develop and test ceramic coated foil stock of super alloy under specific environmental conditions.	
	3. All ceramic sandwich outer wall panel	Develop and test all ceramic panel.	
	4. Edge attachments for panels	Continue development and test of edge attachments to achieve most efficient design.	
11. Areas of high heat flux	1. Cooling systems and coolant requirements	Investigate metallic cooling systems designed under Atomic Energy Commission. Select two for development.	Analytical work at Bell Aircraft Corp.
	2. Metallic coolant distribution systems	Design systems for leading edge including pumping machinery, duct size, and coolant flow. Test typical elements.	Design of full scale system by subcontractor. Tests at Polytechnic Institute of Brooklyn Aeronautical Laboratory (PIBAL)
	3. Transpiration cooling	Tests of typical elements.	Analytical work at Bell Aircraft Corporation. Additional analytical work and tests at PIBAL

TABLE XVI. RECOMMENDATIONS FOR FUTURE WORK (Sheet 4 of 4)

Type	Item	Description	Suggested Facility
12. Inertial System	1. Gyro installation	Develop method for rotating gyro about gyro spin axis to reduce gyro drift.	Bell Aircraft Corporation
	2. Computing elements	Develop "pulsed analogue" computing techniques.	
13. Radar	1. Radome	Test existing materials for properties under conditions encountered by the bomber.	
	2. Antenna	Investigate incorporation of antenna arrays into bomber structure.	
14. Propulsion	1. Propellants	Optimize performance of fluorine-oxygen and JP-4 combinations with respect to chamber pressure, mixture ratio, etc.	Bell Aircraft Corporation or North American Aviation, Inc.
	2. Engine development	Develop oxygen-fluorine and JP-4 unit including thrust chamber, gas generator, and pump of capacity necessary for bomber.	
15. Global Weapon System	Over-all weapon	Investigate feasibility of such a weapon from a configuration standpoint.	Bell Aircraft Corporation

VII DEVELOPMENT PROGRAM



A Program Plan for the design and development of this advanced strategic weapon system has been submitted as a special report (Reference 12). The report describes in detail the program approach and the schedule, and also presents work statements for the areas of investigation during the first year of the program. Cost estimates have been submitted under separate cover.

A. APPROACH

The Bell Aircraft Corporation approach to this manned advanced strategic system is designed to provide an operational weapon at the earliest possible time using technology and equipment consistent with the most advanced state-of-the-art. The first year of the development program should provide sufficient information to initiate a Phase I program. This first year would include the preliminary design of configurations for the weapon system and the supporting technical analysis. This effort would be supplemented by analytical and experimental research which, although conducted primarily by agencies and organizations other than Bell Aircraft, should be monitored, evaluated, and incorporated by Bell Aircraft into the preliminary design work. A study of the weapon capability and value from a military standpoint would be included.

The program plan report recommends that the development program begin in September 1955. This will allow time for the evaluation of the past study results as well as provide the necessary lead time for the negotiation of the first year of effort on this program. It further recommends that AF Contract 33(616)-2419 be extended from 2 May 1955 to 1 September 1955. This four-month period not only permits further study of the more important problem areas but also provides a continuing effort which can be accelerated in an efficient manner to the greater level of effort beginning in September 1955.

B. PROGRAMS

Two proposals for the first year of the program have been submitted as Program I and Program II. Program I which entails a lower level of effort during the first year would increase the total system development by approximately one more year than would be required with Program II.

1. Program I

The effort during the course of this one-year program is the minimum that will provide the data necessary to evaluate the weapon system from a military standpoint and to outline the system to an extent which will permit the initiation of a Phase I program.

The three primary areas of investigation are preliminary design and system analysis, military requirements, and applied research. This program, although including a small experimental program to investigate phenomena associated with the very high speed and altitude of this weapon, relies heavily upon the analytical and experimental investigations which are being sponsored by other government contracts. Such investigations include the following broad categories:

1. Airframe heating as affected by factors such as dissociation, shock wave-boundary layer interaction, and boundary layer properties.

2. Heat protection including insulation, cooling, and radiation methods for the airframe and its contents.

3. Inertial navigation system and component developments.

4. Large-thrust liquid rocket engine development as well as small-thrust, higher performance units using an advanced propellant combination.

5. Radar system and antenna developments for high resolution.

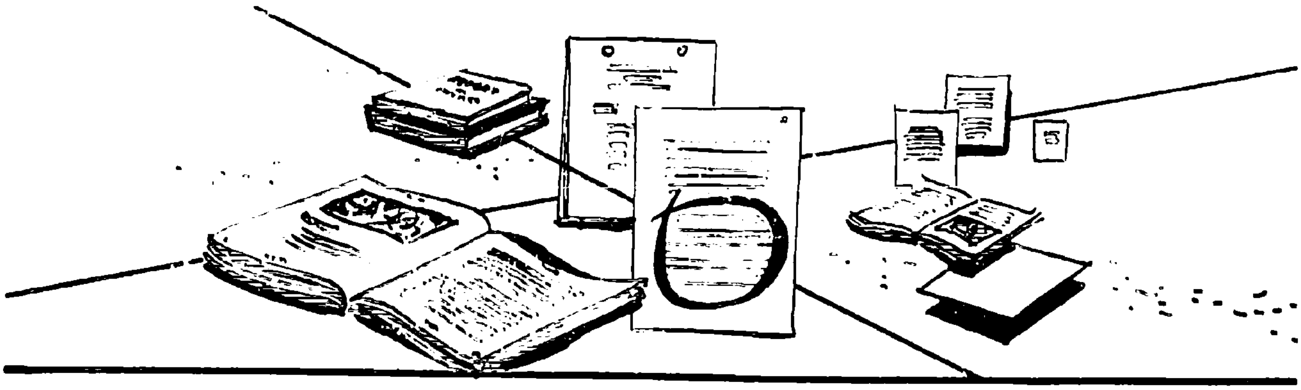
6. Stability and control variations at hypersonic speeds and very high altitudes.

2. Program II

This program includes all the effort required during the first year in order to develop the weapon system in the shortest possible time. The areas of investigation are similar to those for Program I. The experimental investigations and preliminary design work are expanded and the development and test of system prototypes would be initiated. This program would utilize subcontractors to a great extent not only to assist in the development of specific items but also to conduct parallel investigations in certain areas to insure the best solution in the shortest time.

Both Program I and Program II would include sufficient preliminary design work to permit the start of a Phase I program in September 1956 on the second and third stages of the final three-stage system.

VIII REFERENCES



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DEPARTMENT OF THE AIR FORCE
HEADQUARTERS 88TH AIR BASE WING (AFMC)
WRIGHT-PATTERSON AIR FORCE BASE OHIO

17 March 2006

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Defense Technical Information Center
Attn: Ms. Kelly Akers (DTIC-R)
8725 John J. Kingman Rd, Suite 0944
Ft Belvoir VA 22060-6218

Dear Ms. Akers,

This concerns Technical Report AD073760, MX-2276 Advanced Strategic Weapon System, 29 April 1955. This technical report, previously Unclassified/Limited Distribution, is now releasable to the public. The attached AFMC Form 559 verifies that it was reviewed by release authorities at Air Force Research Lab Air Vehicles Directorate (AFRL/VA) and determined to be fully releasable to the public.

Please call me at (937) 522-3091 if you have any questions.

Sincerely

Lynn Kane

Lynn Kane
Freedom of Information Act Analyst
Management Services Branch
Base Information Management Division

Attachment
AFMC Form 559, RUSH – Freedom of Information Act