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011

COCKPIT DISPLAY REPORT

Contract AF 33(600)-37705

March 1959

ER 10390

THE UNITED STATES AIR FORCE

DYNA-SOAR

PROJECT WS464L

CONTRACT NO. AF 33 (600)-37705

APPROVED BY:

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THE UNITED STATES AIR FORCE

DYNA-SOAR

PROJECT WS 464 L

American Machine & Foundry Company

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SPACE FLIGHT

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U.S. AIR FORCE

FOREWORD

This report, prepared by Bell Aircraft Corporation, is submitted in compliance with Item 11 of Exhibit A of Contract AF 33(600)-37705.

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SUMMARY

The cockpit display system for Dyna-Soar I is designed to meet the following criteria: the pilot is provided with display information and control means to assume a wide range of control functions ranging from direct control during the manual flight mode to supplementary control functions during the automatic flight modes and to cope with emergencies in the event that vehicle malfunctions occur during either mode. The system also provides the first step in providing control and display equipment designed to enable the pilot to fulfill the maximum practicable role as the operator of the Dyna-Soar vehicle as a military weapon system.

The two factors which exert the most influence on pilot performance include the maintenance of a suitable cockpit environment and the provisions for effective control and display equipment. The nature of the cockpit environment and its control is the subject of Report ER 10379, entitled Crew Environment. Those factors concerned with the establishment of information requirements, task analysis, and control-display relationships are the subject of the Human Engineering Report, ER 10367.

The objectives of the cockpit display system are directed toward:

(1) Effective utilization of the human pilot in each flight control mode.

(2) Adequate and early warning of the onset and development of failure events, provisions for effective failure correction, and the initiation of effective emergency procedures if necessary.

(3) Suitable monitoring procedures for vehicle and subsystem performance during appropriate control modes to enable the pilot to devote the major share of his attention to functions of research, reconnaissance, or weapon delivery.

(4) The selection of display and control elements which will be operational and available by mid-1961.

Preliminary studies, aimed at satisfying these objectives, have resulted in establishing the functional control and display groups described in this report.

Considerable work has been done in examining promising developments in new control and display concepts. In the controls field, fingertip discrete-button controllers have been investigated. In displays, high-resolution radar has been investigated, as have new energy management presentations and solid state devices.

Further work in follow-on programs, in experimental simulation and other human engineering studies, will ensure that Dyna-Soar I will have the most advanced and effective cockpit displays possible within the limits of time and budgetary considerations.

II. MANNED FLIGHT OPERATIONS CONCEPT

The purpose of this section is to outline the concept of flight operation envisioned for manned Dyna-Soar flights. While the concept is applicable to short flights, it is described here in terms of a global flight from Cape Canaveral at Edwards Air Force Base. Figure II-1 shows the flight envelope during boost, with the nominal boost trajectory required for global flights. The nominal glide trajectory is given in Figure II-2.

The flight test program is planned so that Dyna-Soar will be proven a safe, stable, and controllable powered flying machine before a pilot is allowed to participate in the flight program. In determining the role of the pilot, it is important at this program planning juncture to appraise the pilot's role, keeping in mind the fact that many completely automatically controlled flights will have been successfully completed. If this were not true, the pilot could obviously not be added to the system.

Without detracting from one of the major goals of the Dyna-Soar program, namely that of evaluating the role of a man in a hypersonic boost-glide vehicle, one should logically ask what the pilot should do during the first few flights. The role that the pilot plays on the first few flights, at least, will be substantially influenced by what has happened on the previous unmanned flights.

Let us, therefore, estimate generally what may have happened on the unmanned flights. Even though the system is deemed safe enough for manned flight, flight failures may have been encountered in reaching that point. In the boost system, there may have been propulsion, guidance, and control failures. Significant deviation from the programmed trajectory may have been encountered. During glide, some failures may have been encountered in guidance, control, and propulsion systems. In the course of examining the cause of these failures, it will undoubtedly be found that, had a man been on board, the trouble might have been averted, or at least the severe consequences of the trouble might have been significantly reduced by timely action on the part of the pilot.

Thus, while it is a necessity that the machine have demonstrated a high degree of safety and reliability before man is phased into the system, it is reasonable to believe that the addition of the man will make the machine even more reliable and more flexible in its utilization. His functions, then, should be centered around this concept. With all systems operating properly, his job is fairly routine as far as flying is concerned. His real job is to monitor system operation, detect significant deviations, and take appropriate corrective action. The displays are the pilot's monitor detection tools and the controls are his corrective action tools. As flight experience is gained, it is of course envisioned that the pilot will find it possible to operate with greater flexibility and freedom as far as dependance upon the automatic control features are concerned.

A. FLIGHT OPERATIONS DURING BOOST

The following material serves to describe the method by which the boost operations will be conducted. First, a word concerning the importance of the boost phase. The events which occur during the brief five-minute boost phase commit the pilot for the remainder of the flight, a time which can extend to two hours. In terms of range, the events which occur during the first 500 nautical miles of flight determine what will happen during the remaining 20,000 nautical miles of flight. This great time and distance leverage serves to point out the critical nature of the boost phase and the necessity for developing a practical flight planning approach which will allow safe recovery should malfunction occur during boost.

1. The Zoned Boost Concept

At each point along the boost trajectory, the airplane possesses a maximum and minimum range capability and a maximum cross-range capability. Along the planned flight test range there exists a number of planned and emergency landing sites. Clearly, to handle recovery from booster malfunctions, these landing sites must be matched with boost trajectory segments. Therefore, the boost phase will be divided into a number of velocity zones, with a definite landing site within glide range of each zone. The range potential of the airplane changes rapidly during boost, increasing from zero to 20,000 nautical miles within 320 seconds. Because of the rapidity of change of destination, should a boost malfunction occur, it is logical to predetermine as many courses of pilot action as possible. Since the range and maneuver capabilities resulting from practically all combinations of airplane injection conditions can be calculated, it is logical to do so and zone the boost phase for compatibility with existing landing sites.

The cockpit instrumentation required to implement the zoned boost concept consists primarily of detection and indicated action displays. The pilot must know that boost has terminated, and he must know the destination associated with the particular velocity, altitude, and flight path orientation existing at termination. Also, by monitoring and by direct action, he must ensure that the proper airplane flight path program to that destination is placed in operation.

2. The Boost Corridor Concept

The previously mentioned zoned boost concept deals with predetermining courses of action which result from premature engine shutdown. The boost corridor concept deals with flight path deviations during boost for which the pilot would shutdown the engines, even though they are generating the proper thrust. Clearly, the criteria for voluntarily shutting down the engines in the face of flight path deviations would be that the pilot has more to gain by limiting the deviation than by allowing it to continue. Thus, such a corridor would be determined by factors such as booster and airplane structural limits, airplane recovery limits, and pilot g capability.

The boost corridor and associated zones are presented in Figure II-3 in terms of altitude and velocity. The line labeled nominal is the normal boost trajectory. The zone lines running across the nominal represent regions of altitude and velocity from which flight to preselected launching sites can be made. The maximum-minimum altitude-velocity lines represent the limits outside of which the preselected sites can not be reached. These limits will be established, taking into account all facets of system operation. While the zones have not been accurately established as yet, Zone 1 could be thought of as Edwards Air Force Base. If injection were to occur at the lower velocities of Zone 2, the destination might be Wake, Zone 3, Tilgano, Zone 4, Johannesburg, Zone 5, Ascension, Zone 6, Saint Lucia. Alternate flight path programs are stored in the inertial navigation system for each zone and corresponding destination. If additional zones are needed to cover adequately the boost trajectory from a malfunction standpoint they will be added.

The cockpit instrumentation required for this boost corridor concept consists of displays indicating that the corridor has been exceeded, and boost engine controls necessary for shutdown.

3. Critical Airplane Functions

While a launching obviously would not be conducted unless all airplane systems were functioning properly during the countdown, certain critical airplane systems could become inoperative during boost. Complete loss of airplane electrical power is a vivid example of such a critical function. Here again, the pilot would have more to gain by voluntarily shutting down boost than by allowing its continuation. Thus, the pilot requires displays which indicate the malfunctioning of those critical systems.

B. FLIGHT OPERATIONS DURING GLIDE

During glide, the flight control system (including pilot) is required to control the airplane along the proper flight path and to stabilize the short-term oscillations of the airplane. The latter function of attitude stabilization is a standard problem. The problem of controlling the airplane along the proper flight path, however, is quite a different problem than encountered in standard aircraft. At the end of boost, the airplane possesses a total energy (potential and kinetic) which must be managed carefully if the desired destination is to be reached. The energy management system must, of course, be capable of taking into account the many unknowns which are encountered during glide, such as velocity and altitude errors at end of boost, atmospheric density changes, errors in lift and drag prediction, and winds.

The flight control system has been designed to accomplish automatically both the energy management and the attitude stabilization functions. The systems are also designed so that the pilot can manually accomplish both functions. Thus, displays and controls are required so that the pilot can detect malfunctioning of the automatic equipment and can properly control the vehicle manually. The displays consist of graphical-type presentations showing attitude and velocity as a function of range to go. Included on these displays are the boundaries about the nominal values of velocity and altitude. As long as the aircraft is kept within these limits, the destination can be reached. If a limiting boundary is exceeded, information is also presented to enable the pilot to know immediately his alternate destination. Sample charts of the type to be displayed are given in Section V. The details on energy management system operations are given in the Flight Control System Report ER 10374. The approach leg and landing operations are covered in the Landing System Report ER 10380.

C. EMERGENCY PROCEDURES

The design concept which has been followed relative to the escape capsule is that the pilot would remain with the airplane as long as possible before resorting to the use of the escape capsule. An airplane separation rocket engine provides airplane recovery capability throughout all regions of boost except for a brief period following lift-off. From lift-off to approximately 4500 feet altitude, the capsule would be used directly if a serious malfunction occurs. While pilot judgement will be the ultimate factor in any decision to utilize the escape capsule, the proper malfunction warnings are displayed. In the lift-off region, however, absolute eject signals requiring little or no pilot interpretation should be given.

III. GENERAL DISPLAY ARRANGEMENT

Figure III-1 is a three-view drawing of the Dyna-Soar I airplane showing the geometry of the airplane and the location of the crew compartment (escape capsule) in the nose. Figure III-2 shows the cockpit area in more detail, i.e., the location of the seat and the position of the pilot relative to side windows, forward window, instrument panels, and consoles.

The crew station was designed to accommodate the anthropometric and mobility properties of 5-95 percentile pilots in a "shirtsleeves" condition, or 5-57 percentile pilots wearing a full pressure suit. Location of the displays and controls is such that the pilot can see the particular displays and reach the corresponding controls during the phase of flight in which they are applicable. Adequate supports and restraints are provided for the pilot over the wide range of accelerations which the vehicle attains.

Figure III-3 is a view of the instrument panel and consoles from the pilot's location showing the relative location of all controls and displays. This view is especially useful in illustrating the location of the side-stick controller, the landing engine throttle, the range control engine throttle, and the capsule ejection handles.

Figure III-4 is a detailed layout of the instrument panel and consoles and the accompanying Figure III-5 identifies each of the items shown. These figures will be used in conjunction with illustrations of each panel face to provide more detailed information in the following pages of the report.

One of the first steps in the development of suitable cockpit arrangement and in the selection of effective control and display equipment is the performance of an adequate pilot task and usage analysis. This analysis will provide most of the information defining the control/display requirements and their interrelationships as well as establish priorities on their demands on the pilot's time.

Time limitations did not permit the completion of a formal task analysis program in advance of the finalization of the present cockpit configuration. However, an informal, though somewhat rudimentary, task and usage analysis was performed concurrently with the development of the present cockpit configuration.

A formal and more detailed task analysis program has since been completed and although it remains to be brought up to date with the latest configuration changes, it will be used as a basis for evaluating the effectiveness of the present cockpit arrangement. The Human Engineering Report ER 10367 (and technical note C-2-59) includes the results of the complete task analysis program.

In general, the instrument panel has been designed such that the displays in the center are those most frequently used, and those used less frequently are located on either side. Similarly, the most frequently used controls are located near the normal position of the hands and those used less frequently are distributed in less premium positions.

A. BOOST DISPLAYS AND CONTROLS

During boost, the following displays are used: flight control group at the top center of the panel (Items 6, 7, 8, 9, and 10 of Figure III-5); the zoned booster corridor on the center panel (Item 18); the booster sequence panel and the engine status panel Items 19 and 20. Except for Items 19 and 20, all these displays are centrally located so that the pilot can see them when his head is restrained in a central position against the head rest. The sequence and status displays, which are chiefly light indicator types, are located to the right of center, but well within the pilot's field of view.

The controls for use during this period are located in the immediate vicinity of the pilot's hand; the side-stick controller is at the right hand, and the booster and engine control buttons are near the finger tips of the left hand.

B. FLIGHT CONTROL SYSTEM

During all phases of flight, the basic flight control group (Items 6 through 10 of Figure III-5) are of primary importance and have, therefore, been assigned the premium space on the panel, the top center approximately level with the pilot's eyes. In the center panel immediately below the array (Item 18), the boost corridor display is removed during glide and replaced by the graphic screen display for the control law in use at any given phase of glide. Items of secondary importance, such as the surface temperature indicator and trim indicators, are located along the sides of the center panel.

The primary control during glide is the side-stick controller, which is located so that it is in the pilot's hand when his arm is in a normal, comfortable position (Item 62).

The backup landing instrument group is located on the left-hand side; however, it is located vertically so that the pilot need only look horizontally from his window aperture to utilize the landing instruments.

C. GUIDANCE SYSTEM

The primary navigation system display is the map; hence, this is also located in a central position on the lower central panel (Item 23 of Figure III-5). Other items which do not require such frequent monitoring are arrayed on the sides; these include the cross-range error indicator and the time indicator. The navigation controls (Items 50 and 52 of Figure III-5) used relatively infrequently are located on the left-hand console and the pilot is required to extend his arm to reach them.

For landing, the top center panel (Item 18 of Figure III-5) is folded back providing an aperture which the pilot utilizes to look through the nose window. The commands from the automatic landing system are displayed on the attitude indicator (Item 8) during this phase of flight.

D. AIRPLANE PROPULSION AND POWER GENERATION SYSTEMS

The propulsion and power generating system displays are, in general, not primary control displays, but indications of the state system operation. As such, they are frequently in the form of warning or indicator lights which do not require premium positions in the center of the panel. The majority of these displays are located in Items 22, 29, 31, and 54 of Figure III-5.

The propulsion controls which may require use during boost (such as airplane separation engine switch, Item 51) or for flight control (such as the side-stick controller for reaction control engines) are located close to the pilot's hands. Those controls which are required to be used only intermittently such as for the range control engine are located more remotely.

E. FAILURE WARNING AND EMERGENCY

Since it is not possible to put all failure and emergency warning displays in a central position where they can be seen at all times, a panel containing these lights is placed on the right-hand side of the panel (Items 21 and 24) and selected indications on this display are used to actuate three master lights located in the top center of the panel (Items 1, 2, and 4). These are the fire warning light, the eject warning light, and the master caution warning light. These three lights are always visible to the pilot; they call his attention to existing trouble so that he can refer to the warning panel and other displays to decide on appropriate action. In general, failure and warning displays are located in a single area, i.e., lower right-hand side.

Controls for major emergencies, such as the capsule ejection handles and airplane separation buttons, are located in the immediate vicinity of the pilot's hands; the other controls, such as the fire extinguisher panel (Item 36), require extension of the pilot's arms.

IV. BOOST SYSTEM DISPLAYS AND CONTROLS

Because of the critical nature of the boost phase of flight, it is extremely important that the pilot be aware of the most important aspects of booster vehicle and airplane performance during this phase. Since the success or failure of the mission is essentially determined by the events that occur during this relatively short period of time, the ability of the pilot to monitor this phase and take corrective action when necessary, will greatly increase the probability that the mission can be completed successfully. Therefore, displays will be provided to present to the pilot the planned and actual sequence of events, planned and actual energy trajectory data, and marginal or out-of-tolerance conditions which occur during first and second stage booster operations. Also displayed are other critical cues, indicating normal or abnormal functioning of the vehicle, such as programmed and actual attitude, and g limits. Controls are also provided for manually backing up the booster sequence of operation and manually flying the vehicle, as well as controls for taking emergency action in event of a failure or marginal condition.

The seat and arm rests have been designed for properly restraining the pilot while under the influence of the expected accelerations. Since centrifuge tests indicate that a pilot will retain normal control over his wrists, fingers, and vision under the maximum peak acceleration expected during boost, the controls provided will be operable by wrist and finger-tip action.

Location of all boost displays to minimize eye scan and eliminate head motion was given careful consideration. During this phase, the pilot's attention must be given to these displays, plus the important flight control parameters and warning indicators. Therefore, all such displays are located close to eye level in the center or near central portion of the instrument panel. Thus, the pilot is able to readily observe all information of importance so that any action may be taken. The controls are specifically designed to be non-ambiguous and operable by the tactile and kinesthetic senses so that the pilot's attention need not be diverted from the important displays during the boost phase.

A. BOOST SEQUENCE PANEL

Location of the boost sequence panel is shown on Figure III-5, Item 19. This display is provided to allow the pilot to view both the planned and actual sequence of events as they take place during boost. Should any event fail to occur at the programmed time, the display indicates this, and, in turn, cues the pilot to operate a manual control causing the event to take place. The mission may then be continued as planned. (The controls provided are described in Section IV-C.)

1. Description

The boost sequence panel is presented in Figure IV-1. It consists of a series of lighted elements arranged in a vertical column which read from bottom to top in an order analogous with the normal sequence of events taking place. Next to each lighted element is a digital readout which displays programmed time from the previous event. The functions displayed are as follows:

1st Stage Fire

Lift-Off

1st Stage Cutoff

2nd Stage Fire

1st Stage Separate

2nd Stage Cutoff

2nd Stage Separate

The functions are not visible until each event actually takes place, at which time they are illuminated in green.

2. Input Signals

Input signals to the function lights are provided by the automatic sequencing system in the booster vehicle. When an event is initiated by the automatic sequencing system a signal is provided to the appropriate function light. This same signal is also applied to the next timer to start it running. The timers are electrically driven by the secondary power system of the airplane.

3. Usage

The timers are preset to the correct times on the ground. At a prearranged point in the countdown, an electrical signal starts the first timer. This is essentially a countdown indicator which indicates time in seconds until 1st stage fire. At zero time, the fire signal is given. This signal illuminates the "1st Stage Fire" light and at the same time starts the second timer running down to zero. At zero time on the second timer, the pilot will expect to see the "lift-off" light come on indicating that this event has been initiated. When the "lift-off" light comes on, the next timer is started which in turn starts its countdown. This sequence is continued through the boost phase. Should any timer show zero and the corresponding light fail to illuminate, indicating that the event failed to occur, the pilot may activate a manual control (described in Section IV-C) which will trigger the event, and, in turn, illuminate the event light and start the next timer. Thus, failure of an event to occur exactly on time will not result in accumulative errors in the subsequent timers. Should any event occur too soon, i.e., the light comes on before the adjacent timer indicates zero, the pilot will take no action (provided, of course, that the event did not occur prematurely enough to result in any hazard or malfunction). The next timer will then be started at the time the previous event actually occurred. In this manner, the pilot is provided with "should happen" and "did happen" information on the complete boost sequence.

B. BOOST ENGINE STATUS PANEL

The boost engine status panel as shown on Figure III-5, Item 20, is located immediately below the boost sequence panel.

This display is provided to allow the pilot to visually observe the status of each engine during both the first and second stages of boost. Should a boost engine operate outside its tolerance limits, a warning light will be energized. The pilot may then operate a manual control (described in Section IV-C) to shutdown that engine before a hazardous condition develops. Stage malfunction lights are also provided to allow shutdown of complete stages.

1. Description

The boost engine status panel is illustrated in Figure IV-2. The panel consists of four lighted elements in a row depicting the four 1st Stage engines, one element above depicting the 2nd Stage engine, and two malfunction indicators. Each lighted element depicting an engine is divided and displays two engine parameters: turbine speed and thrust chamber pressure.

2. Input Signals

All indicators on the panel are supplied with signals from the booster vehicle "Propulsion Surveillance System."

3. Usage

It is necessary to monitor the status of the booster engines to provide the pilot with advance warning data on possible out-of-tolerance conditions developing within the booster vehicle. The two parameters of turbine speed and thrust chamber pressure were chosen for monitoring from the many parameters available since they are easily instrumented and are highly representative of normal and abnormal operation. Many more parameters could, of course, be displayed, but this would soon overburden the pilot and clutter the display to the extent that the data presented could not be intelligently used or integrated into the boost control task.

The engine status indicators are designed to operate as follows: When the engines are not running, the indicators are not lighted. As soon as the first stage is fired and engines are operating normally, the 1st Stage indicators are illuminated in green. At 1st Stage cutoff, the indicators are extinguished; at 2nd Stage fire the 2nd Stage indicator is illuminated in green. If at any time the 1st or 2nd Stage engines go outside the established limits of turbine speed or chamber pressure, the respective indicator segment glows in red. The pilot may then shutdown the marginal engine before a dangerous or hazardous condition occurs. A positive indication of normal engine operation by means of the green indicators (rather than merely "off" for normal and "on" in red for abnormal) is required for other reasons. If it is desired to shutdown an engine owing to conditions other than engine malfunctions, the pilot must be provided with a positive indication of the engine which was shutdown.

Thus, with all indicators glowing in green during normal operation, when an engine is shutdown, the respective indicator will be extinguished.

The stage malfunction indicators will be illuminated in red when the complete 1st or 2nd Stages must be shutdown and separated to prevent an impending disaster. When illuminated, the pilot may shutdown and separate the respective booster stages by means of the controls described in the following.

C. BOOST ENGINE CONTROLS

The location of the boost engine control and airplane separate panel is shown on Figure III-5, Item 51. The controls on this panel are provided to allow the pilot to back up manually the automatic boost staging sequence and manually perform emergency operations as may be required during the boost phases. Cues directing the pilot to perform the required operations are provided by the boost engine status panel and boost sequence panel described previously.

1. Description

The boost engine control and airplane separate panel is presented in Figure IV-3. The controls on this panel consist of eleven push-buttons located at the pilot's left finger-tips, all separated by raised barriers. The boost command mode button (Boost Comm. Mode) is integrally illuminated when depressed. The functions which these push-buttons control are as follows:

(1) Engine 1 Cutoff (First Stage)
(2) Engine 2 Cutoff (First Stage)
(3) Engine 3 Cutoff (First Stage)
(4) Engine 4 Cutoff (First Stage)
(5) First Stage Cutoff
(6) Second Stage Fire

(7) First Stage Separate

(8) Second Stage Cutoff

(9) Second Stage Separate

(10) Airplane Separate

(11) Boost Command Mode

2. Usage

The controls are designed to prevent inadvertent operation yet be highly accessible to the pilot and operable during maximum boost acceleration conditions by means of tactile sensing. The top four push-buttons, provided to cut off individual thrust chambers in the 1st Stage should a marginal condition develop, are used in conjunction with the engine status panel as described in Section IV-B. They may also be employed to shutdown individual engines should this be desired because of cutoff tolerance conditions as presented on the boost corridor screen display (described in Section IV-D). Manual backup control over the automatic boost staging sequence is provided by the push-buttons located in the second row plus the "Stage 2 Sep." button in the bottom row. These buttons are then associated with the lighted elements in the boost sequence panel display and are used in conjunction with that panel as described in Section IV-A.

The "Airplane Separate" push-button is used in conjunction with the 1st Stage malfunction or 2nd Stage malfunction lights located on the Engine Status Panel. Depressing this push-button will automatically shutdown whichever booster is operating and fire the aircraft separation engine to allow airplane escape.

The "Boost Comm. Mode" button is used to engage the three-axis side-stick controller to the booster automatic control system for manual flight control during boost and is more fully described in Section IV-E.

D. BOOST CORRIDOR GRAPHIC SCREEN

The Booster Corridor Graphic Screen is centrally located in the instrument panel at approximately the pilot's eye level as shown in Figure III-5, Item 18. The display has been assigned this location due to its relative importance and use frequency during the boost phase of flight. This same area is used during glide for the energy management glide displays described in later sections of this report. The screen is later folded out of the way leaving an aperture to the nose transparency for forward vision during approach and landing. By employing this section of the instrument panel in this manner, highly efficient usage of this valuable center section of the panel is accomplished, since only that information needed for a particular phase of flight is displayed. These data are then replaced by new graphic displays as required for other phases of flight.

Since the airplane's maximum kinetic and potential energy is firmly established at the end of the boost phase, once the required energy level to achieve global range has been imparted to the airplane, the pilot is virtually assured that (barring any serious malfunctions) he will be able to achieve his pre-planned destination. The importance of this display then lies in the fact that the energy of the vehicle is rapidly built up in a relatively short time during boost and during this critical period the airplane's maximum range will be firmly established. Therefore, information must be available to the pilot as to whether or not there is sufficient energy to continue with the mission as planned and, if not, what destinations are possible to achieve with the energy remaining or which will be remaining at cutoff.

The boost corridor display is provided to allow the pilot to view his progress along the programmed attitude-velocity trajectory and to supply him with data regarding the vehicle's potential ability to achieve the planned destination. The display also provides the pilot with the cues necessary for him to decide what destination should be selected when he has determined that the original destination no longer can be achieved.

1. Description

The boost corridor graphic screen will provide a display consisting of a plot of altitude versus velocity as shown in Figure IV-4. The nominal boost flight path of the vehicle is marked on the display as well as allowable limits off this nominal path within which the pilot may still make good his pre-planned destination. The display is divided into various zones which relate to the alternate destinations available. That is, each zone indicates a definite range capability of the vehicle. These zones then are matched to known bases along the flight path of similar range and preselected before take-off as landing areas. During boost the actual altitude and velocity curve of the vehicle is traced on the display by means of a moving spot which will leave a persistent trace for a short period of time.

The graphic display is generated by means of a slide projection system immediately behind a ground glass display screen. Thus, images projected on the rear face of the screen will be visible to the pilot from the front. As different displays for other phases of flight are required, new slides are inserted automatically from the slide magazine. Appendix C contains a description of the slide projection mechanism. Other means for the generation of this display have also been studied. One of these methods makes use of an electroluminescent screen to present the required data. This concept appears to be very promising and is further discussed in Appendix D.

The energy display panel is provided for use with the graphic display and is located as shown in Figure III-5, Item 63. This panel contains five integrally lighted push-buttons, a brightness control, and two trimming controls as shown in Figure IV-5. The push-buttons allow the pilot to select manually the graphic display desired Normally, these will be switched in automatically by the guidance system and the push-button associated with the display presented will be illuminated. The trimming controls are provided to allow initial adjustments to the moving trace on the display screen and need not be adjusted once this has been accomplished. (The panel is further described in Appendix C.)

2. Input Signals

The display is provided with both velocity and altitude signals from the inertial system of the vehicle. These signals drive the X and Y motors to position the moving spot. Signals are also provided from the inertial system to program the proper slides for viewing during the various phases of flight

3. Usage

The pilot observes the trace of the actual velocity-altitude information and compares it with the nominal flight path. If he finds that it tends to move outside the permissible bounds, he may try to bring it back by using the side-stick controller. (During boost, the three-axis stick may be connected directly to the booster control system for this purpose) If the pilot cannot maintain the trace within the required limits, he institutes shutdown and separation of the airplane. He then observes the zone number in which the trace is at shutdown By punching this number into the "destination" section of the computer programmer, he automatically directs the airplane to follow a programmed course to this new destination In the case of emergency or premature shutdown as may be required because of information received from the boost sequence panel or engine status panel, the pilot also observes the zone in which shutdown occurred and follows the same procedure described previously. If the trace remains within the permissible bounds throughout the boost phase, the pilot then knows that the airplane has sufficient energy at the end of boost to make good his original destination.

E. MANUAL BOOST FLIGHT CONTROL

In addition to the pilot's duties of monitoring the booster engines and providing backup to the staging sequence, he will be required to maintain the vehicle on its programmed flight path within pre-defined limits. Display references which the pilot uses for this purpose are provided by the boost corridor graphic screen described in the previous section, plus the attitude indicator/flight director which provides both programmed and actual attitude information. Other cues, such as high acceleration warnings and excess pitch angle warnings, are also provided.

Manual control during the boost phase will be accomplished by connecting the output of the three-axis side-stick controller into the booster flight control system in either of two possible modes of operation. The side-stick controller is located as shown in Figure III-5, Item 62, and can be operated by the pilot during all phases of boost. A complete description of this control is given in Section V-C. Initially, the stick will be disconnected from the booster flight control system to prevent the pilot from introducing inadvertent commands. Should the pilot desire to "take over" command to correct the booster control system, it will be necessary for him to depress the "Boost Comm. Mode" momentary push button located at his left finger tips and hold it down. This will connect the stick to the booster's attitude reference control loop and movement of the stick in any of the three axes will introduce commands into the booster control system. Thus, vernier corrections may be made by the pilot to maintain the vehicle on its programmed flight path. When the pilot releases the boost command mode push button, the stick will be disconnected from the booster control system.

A second and equally important reason for giving the pilot manual control during boost is to cover the contingency that the booster control system may experience some type of failure which would result in a radical departure from the programmed flight path. Since it is extremely probable that with such a failure the control stick will be useless if connected into the booster control system in the normal manner, a second or backup mode of manual control over the boost phase is provided. To control the vehicle in this mode, the pilot depresses and holds down a trigger switch mounted on the stick grip. This will connect the stick directly to the booster vehicle's rate loop; thus, most of the automatic flight control system in the booster will be bypassed. Release of this trigger switch will disconnect the control stick from the booster vehicle returning the pilot command to zero. Upon separation of the second stage booster a switch associated with the separation mechanism will revert the switch mounted on the control stick to its glide function of engaging the Backup Flight Mode in the airplane flight control system.

V. FLIGHT CONTROL SYSTEM DISPLAYS AND CONTROLS

The Flight Control System displays and controls consist of three major groups:

(1) Primary Flight Control Displays. - These displays provide the pilot with information regarding the flight situation of the airplane and boosters. Several of these displays also indicate the variation (error) between the existing flight situation and that commanded by the inertial navigation system or automatic landing system.

(2) Backup Flight Control Displays. - These displays serve the same function as the primary displays but are more basic in nature, simpler and more reliable. They are installed primarily to provide vital information should a primary display fail. The backup displays are of particular value in the low altitude, flight regime where conventional instrument mechanism can be applied.

(3) Flight Control System Controls. - This group of equipment consists of switches and controllers for selecting the flight

control system mode of operation and for manual operation of the system when changing or controlling the flight situation. Also included in this group are those displays which indicate the status of the flight control system.

The following features are of significant interest in the areas of flight control displays and controls.

(1) Where applicable the displays are driven from inputs received from either inertial navigation system.

(2) Graphic screen displays are utilized to present lift and velocity information to facilitate the control of energy dissipation during the glide.

(3) Expanded scale presentations are provided to permit close control of parameters which vary over extreme ranges.

(4) A three-axis attitude indicator/flight director is provided to display attitude information required for control throughout the flight including boost.

(5) "Quickening" is provided in the attitude indicator/flight director display to facilitate accurate control of attitude.

(6) A temperature display is provided to indicate temperature on the outer surface of the airplane for use as structural limit warnings.

(7) A group of backup displays are provided to assure the availability of instruments suitable for landing and to provide a second attitude indicator display suitable for use over the entire glide portion of flight.

(8) The pilot is provided with four selectable modes of control, each requiring different levels of pilot effort. These allow him to make a suitable selection consistent with the status of the inertial navigation and flight control systems.

(9) A three-axis, wrist-type, side-stick controller is provided to enable the pilot to introduce pitch, roll, and yaw commands into the booster and airplane flight control systems.

(10) Control surface position indicators are provided to enable the pilot to monitor the output of the flight control system.

A. PRIMARY FLIGHT CONTROL DISPLAYS

This group of displays is located on the panel in the position indicated in Figure III-5. The instruments are listed in the following. Items in parentheses are keyed to numbers on the diagram.

Attitude Indicator/Flight Director	(8)
Altitude, Descent Rate, and Lift Error Indication	(9)
Lift Control Graphic Screen Display	(18)
Velocity Control Graphic Screen Display	(18)
Altitude Control Graphic Screen Display	(18)
Velocity and Range Indicator	(7)
Acceleration Indicator	(10)
Surface Temperature Indicator	(6)

For maximum integration of the airplane subsystems, much of the data displayed by this equipment are derived from the inertial navigation system (INS). The pilot may drive the displays from either INS system by operating the INS selector switch on the Computer Programmer Panel (Item 52).

Because of the Dyna-Soar flight requirements, the instruments in the primary display are required to indicate small variations in conditions such as velocity and altitude. To accomplish this over the great range of velocities and altitudes encountered, several indicators and the graphic displays are constructed to permit expanded scale presentations. The indicators use servo driven moving tapes which display only a portion of the available instrument range at a time thus allowing a choice of scale factor compatible with the accuracy of control required. See Appendix B for a detailed description of moving tape instrument construction.

An expanded scale presentation is obtained from the graphic screen displays by projecting only a portion of the display on the screen at any time. See Appendix C for a detailed discussion of the graphic screen display construction.

1. Attitude Indicator/Flight Director

The Attitude Indicator/Flight Director has been assigned the prime position at the upper center of the display panel. Item 8 of Figure III-5 shows the exact location. The instrument face is shown in Figure V-1.

a. Description

This instrument is a three-axis astro-type attitude indicator similar to the Lear, Inc. Model 4060 three-axis remote attitude indicator modified in accordance with Lear Proposal No. 5089. The three-axis instrument was selected to prevent ambiguities in presentation during the early portion of the boost phase when the airplane indication will be oriented 90° from normal.

The instrument presents the following information.

(1) Airplane attitude in pitch, roll, and yaw by means of a 4-inch sphere free to rotate in all three planes.

(2) Attitude error (difference between commanded attitude and actual attitude) in pitch and roll. This information is presented by a pointer free to move vertically and to rotate. Vertical deflection from the centerline indicates a pitch attitude error and rotation indicates a roll attitude error. "Quickening" has been incorporated by subtracting, from the appropriate error signal, rate of change of pitch or roll attitude.

(3) Heading error information is presented by a vertical pointer moving horizontally across the meter face. "Quickening" has also been incorporated in this display by subtracting rate of change of yaw from the heading error input signal.

(4) Side slip information is displayed in a conventional manner by a ball at the bottom of the instrument. This ball will be magnetically restrained during early portions of the glide when the normal gravitational acceleration is practically cancelled by the centripetal acceleration.

(5) Angle of attack information is displayed by a vertically moving index on the left-hand side of the instrument.

(6) The master OFF flag is located in the center of the lower left-hand quadrant of the instrument.

(7) The OFF flag for the commanded pitch and roll indication is at the 0300 position.

(8) The OFF flag for the commanded yaw attitude is located at the top of the instrument.

(9) The OFF flag for angle-of-attack information is located at the 0900 position.

A multiposition switch is provided to lock out the heading error indication at the pilot's option and to free the side-slip indicator.

b. Inputs

The signals driving the three axes (pitch, roll, yaw) of the ball are supplied from the inertial platform.

The pitch attitude error signal is generated within the General Purpose Computer (part of the INS). "Quickening" (rate of change of pitch attitude) supplied from instrument rate gyros is subtracted from this signal before it is applied to the meter movement. The equations mechanized to provide the signal during the various flight phases are as follows:

(1) Phase I, lift control (injection altitude to 300,000 ft)

$$\theta_c = \theta_p$$

where θ_c = commanded pitch attitude

θ_p = programmed pitch attitude

$$I_s = \theta_c - \theta - K_1\dot{\theta}$$

where I_s = instrument signal

θ = actual pitch attitude

$\dot{\theta}$ = time rate of change of pitch attitude

K_1 = constant

(2) Phase II, velocity control (300,000 to 100,000 ft)

$$\theta_c = \theta_p + K_2R\ (V - V_p)$$

where K_2 = constant

R = range to go

V = actual velocity

V_p = programmed velocity

$I_s = \theta_c - \theta - K_1 \dot{\theta}$

(3) Phase III, altitude control (100,000 to 10,000 ft)

$\theta_c = \theta_p - K_3 (h - h_p)$

where K_3 = constant

h = actual altitude

h_p = programmed altitude

$I_s = \theta_c - \theta - K_1 \dot{\theta}$

(4) Final approach and touchdown, automatic landing system (ALS). - In this phase the error signals are generated by a ground-based computer and transferred to the aircraft via a data link. The reference attitude at engagement of the ALS is provided by the flight control system. Engagement of the ALS transfers the pitch error signal input to the instrument from the general purpose computer to the output of the data link receiver.

$\theta_c = \theta_K + \theta_g$

where θ_K = pitch attitude at engagement of the ALS

θ_g = ground generated command

therefore, the signal to the instrument is

$I_s = \theta_c - \theta - K_1 \dot{\theta}$

The roll attitude error is generated within the general purpose computer. "Quickening" signals, supplied by rate gyros, are subtracted from the error signal before it is applied to the meter movement.

During boost, the error signal indicates to the pilot how the vehicle must be rolled so that the airplane may be injected along the proper trajectory to the target. From the latter stages of boost until a range to go of 7000 nautical miles, the computer will command a wings level attitude. From 7000 nautical miles to 10,000 feet altitude, the computer will command an attitude as a function of cross-range error. From 10,000 feet altitude, roll commands will be obtained from the ALS through the data link in a manner similar to the pitch signals.

The heading error is obtained from the platform of the inertial navigation system. "Quickening" is added as for the other axes. The error signal will be proportional to the angular difference between the velocity vector and the heading of the aircraft. When the lift forces acting on the airplane become sufficiently great, the pilot may switch off the heading error indicator and release the ball in the side-slip indicator.

The angle of attack signal is generated within the inertial navigation system with provisions to derive the signal from the barometric system as desired.

The equation mechanized is

$$\alpha = \theta - \gamma$$

where α = angle of attack

θ = pitch attitude

γ = flight path angle

c. Usage

The instrument is employed by the pilot in a manner similar to a conventional attitude indicator in providing information relative to the instantaneous attitude of the airplane body axes. In addition, it will be used in a similar manner as a zero reader. The pilot always flies the command or error signal needles to the center of the display. The additional specific use of the instrument for each phase of the flight is as follows:

Boost Phase

The pilot will monitor the position of the three body axes and, if necessary, introduce steering signals to zero-out the command needles. The instrument provides him with backup information to assure that the proper pitch program is being followed. The boost control graphic screen will be the primary source of information during this phase. In addition, the instrument will indicate that the vehicle has rolled out on the proper heading.

Phase I of Glide (injection altitude to 300,000 feet)

Through use of the pitch command needle, the pilot is assured that the airplane is following the proper pitch program. During this phase, the primary displays are the lift control and lift control graphic screen displays.

Phase II (300,000 to 100,000 feet)

The instrument during this phase will provide the pilot with the backup display necessary to follow the programmed flight path by keeping the pitch and roll command needles at their zero position. The primary displays during this phase are the velocity, velocity control, and graphic screen displays.

It should be noted that velocity information is combined with the attitude information signal to the command bars.

Phase III (100,000 to 10,000 feet)

Again, the instrument will provide the backup instrument necessary to following the programmed flight path. The pilot flies the command bars to the zero position during this phase. The primary displays are the altitude and descent rate indicators.

Landing Phase

During this phase, the instrument is the primary source (plus window) to accomplish the final approach and landing. Command information from the automatic landing system is displayed on the pitch and roll command needles, and the pilot flies to keep them centered.

2. Altitude, Descent Rate, and Lift Control Indicator

The altitude, descent rate, and lift indicator is located just to the right of the attitude indicator/flight director. Item 9 of Figure III-5 shows the exact location. The instrument is shown in Figure V-2. The indicator contains three displays: altitude, descent rate, and lift error. These displays were grouped together because, during Phase I of the energy management program (from injection altitude to 300,000 feet), the expenditure of energy is controlled by lift and is closely associated with altitude and rate of change of altitude.

In addition to the three basic displays, the instrument has a digital readout of the injection altitude. This readout serves as a ready reference to the pilot as to his initial glide altitude.

The three basic displays are discussed separately in the following.

a. Altitude

Description

Studies have indicated that an expanded scale will be required to permit adequate monitoring and/or pilot control of altitude during the flight. Therefore, altitude will be displayed by a servoed tape type of instrument similar to the Lear, Inc. Model 3905A Altimeter.

Consideration of the factors involved indicated that the scale factor of the tape should be 4000 ft/in. and that approximately 5 inches of tape should be visible at any one time. The scale factor indicated above is very nearly optimum for the entire flight regime. It may be necessary to introduce some further expansion of the scale below 10,000 feet to provide the pilot with an optimized instrument for the final approach and landing.

Actual altitude is indicated by a single bar across the face of the tape and the programmed altitude by a double moving bar. The tape and bar drive mechanisms are so integrated, that under normal flight conditions the single and double bars should form a single wide bar at the center of the scale.

Inputs. - The input of actual altitude, which drives the servo tape, is obtained in several ways. During boost, the input is geometric altitude from the inertial navigation system. During Phase I of the glide, the input is density altitude as computed from lift which is measured by the vertical accelerometer on the platform of the INS. During Phases II and III, the input is geometric altitude from the INS. During the latter stage of Phase III, the pilot may switch over to an input from the barometric system by operating the switch provided on the display.

The input of programmed altitude, which drives the double moving bar, is obtained from the INS. During boost and Phases II and III of glide, this input is a programmed geometric altitude. During Phase I, this input is a density altitude as computed from a programmed lift.

Usage. - During Phase I lift control, the pilot will utilize the lift control graphic screen display to ascertain that the airplane is being held within the proper altitude limits. In this connection, his altimeter will serve as a convenient and precise readout of altitude. However, the lift error indicator, described subsequently, will provide the most accurate indication of when forward or retro thrust is required for control. During

Phase II velocity control, the altimeter will be used for only general information purposes. During Phase III altitude control, the indicated and programmed altitude readouts will provide prime control information to the pilot.

b. Rate of Descent

Description. - Studies have indicated that an expanded scale will be required for this display. This instrument will be of the servoed tape type similar to the Lear, Inc. Model 3977A rate-of-climb and acceleration indicator. Consideration of the factors involved show that the tape scale factor of this instrument should be 1000 ft/min per inch of tape. Five inches of tape will be visible at any one time. The actual rate of descent is indicated by a single bar in front of the tape and the programmed rate of descent by a double moving bar. The tape and bar drive mechanisms are so integrated that under normal flight conditions the single and double bars should form a single wide bar at the center of the scale.

Inputs. - Normally, the rate of descent is generated within the inertial navigation system and the programmed rate of descent in the general purpose computer. Provisions have been incorporated in the display system to drive the instrument from a barometric system below approximately 60,000 feet. A switch is provided to transfer from the primary signal source to the barometric system at the option of the pilot.

Usage. - This display is used during glide Phase I, II, and III, in much the same manner as the altitude indicator previously described. During landing (Phase IV) the display will be used in the same manner as a standard rate of descent indicator.

c. Lift Error

Description. - This display has the same general appearance as the other two. However, it is not of the servoed tape type. The lift error is indicated by a single moving bar. When the lift error exceeds the tolerance about zero shown on the instrument, forward or retro thrust is applied. The "no thrust" tolerance bound has been chosen through simulator studies to minimize propellant consumption. The "no thrust" region is shown by the gap between the two pointers on the left side of the scale.

The upper pointer is labeled forward thrust and the lower retro thrust. The reason for this nomenclature is that should the indication of lift error fall outside of the indicated tolerable limits it will be necessary to apply forward or retro thrust (in the direction indicated) to maintain the lift error within the limits. This is the prime energy management indicator for glide Phase I flight.

Inputs. - The lift error signal will be generated in the Inertial Guidance System (General Purpose Computer). The equation that is mechanized to provide the signal is as follows:

$$\%\ \text{lift error} = (L/L_p - 1) + K(\dot{h}_p - \dot{h})$$

where L = actual lift

L_p = programmed lift

$\dot{h}_p$ = time rate of change of programmed altitude

$\dot{h}$ = time rate of change of geometric altitude

L = $mAz \pm KT$

where L = actual lift

m = mass

K = constant

T = thrust

A_z = output of vertical accelerometer

The signal for driving the lift allowable error limits is generated in the general purpose computer from the programmed lift parameter.

Usage. - During Phase I of the flight (injection to 300,000 feet), lift control is used for the management of energy dissipation. During this phase the pilot will use the instrument to monitor the operation of the automatic system; and should it fail to function properly the pilot can manually apply forward or retro thrust as indicated by the display. The application of thrust will also change the rate of descent and this display is adjacent to the lift error display for ready reference. The pilot will use this instrument in conjunction with the lift control graphic screen display during Phase I.

3. Lift Control Graphic Screen Display

The lift control graphic screen is centrally located in the instrument panel at approximately the pilot's eye level as is shown in Figure III-5, Item 18. This display will occupy, during Phase I of the glide, the same space that was occupied by the boost corridor graphic screen during the boost phase.

At the end of boost this display replaces the boost corridor graphic screen display. This is the primary energy management display for this phase and is, therefore, assigned central location directly below the attitude indicator/flight director described previously. Figure V-3 illustrates the concept of this display. The actual shape and details of the curves will be determined from simulation studies.

This airplane, unlike the powered aircraft, has all of the energy available to make good a destination in the form of kinetic and potentional energy. The importance of this display then lies in the fact that the pilot must know whether the remaining energy is sufficient to continue the preplanned mission, or what other destinations are possible. The lift control display is provided (for Phase I) to allow the pilot to visually observe his progress along the preprogrammed trajectory and to supply him with data regarding the vehicles potentional ability to achieve the preplanned mission.

a. Description

This graphic display is located in the same area as the boost corridor graphic display. The mechanism is described in Chapter IV and in greater detail in Appendix C.

In this case, the "Y" axis of the chart is positioned by lift data and the "X" axis with range-to-go. The lift information is displayed as a density altitude which is computed from an actual measurement of lift.

b. Input Signals

The display is provided both altitude and range-to-go signals from the Inertial Navigation System. These signals drive the "X" and "Y" motors to position the moving marker. Signals are also provided from the inertial system to program the proper slides for viewing during the various phases of flight.

c. Usage

The pilot observes the trace of actual altitude/range-to-go information and compares it with the nominal flight path. If he finds that it tends to move outside of the permissible bounds, he may try to bring it back by engaging either the Pilot Command or Piloted Flight Modes and introducing steering signals with the side stick controller.

If the pilot finds it impossible to maintain the airplane within the required limits, he selects an alternate destination and punches this number into the "destination" section of the computer programmer, thereby automatically directing the airplane to follow a programmed course to the new destination. This change in destination alters the range-to-go thus causing the marker to translate in the "X" direction relative to the projection of the slide. If the marker relocates within the permissible bounds of the chart, the pilot knows that the airplane has sufficient energy at this time to make good the new destination.

4. Velocity Control Graphic Screen Display

The location and general description of this display are identical to that indicated in the previous section.

In this case the ordinate is velocity and the absica range-to-go. Figures V-4 and V-5 are typical illustrations of the chart displayed. These charts are automatically interchanged to cover the entire velocity control range.

This display is the primary energy management display for Phase II of the glide (300,000 to 100,000 feet). The possibility of displaying a boundary indicating skin temperature limiting condition is under consideration.

5. Altitude Control Graphic Screen Display

This display replaces the velocity control display in Phase III of glide (100,000 feet to 10,000 feet). The parameters here are altitude as a function of range-to-go. Initially geometric altitude is displayed and this is replaced by barometric altitude when the pilot operates the appropriate selector switch after the pitot mast is extended.

The altitude control graphic screen display is stowed when the pilot desires to use the forward looking window. Range-to-go altitude, and altitude limits will then be obtained from the appropriate moving tape instruments.

6. Velocity and Range Indicator

The velocity, and range indicator is located just to the left of the attitude indicator/flight director. Item 7 of Figure III-5 shows the exact location. The instrument is shown in Figure V-6. The indicator contains three displays: coarse velocity, fine velocity, and range-to-go. These instruments were grouped together because, during Phase II of the energy management program (300,000 to 100,000 feet), the expenditure of energy is controlled by velocity and the range-to-go is of prime importance.

In addition to three basic displays, a digital readout of the injection velocity is provided; this readout serves as a ready reference to the pilot as to his initial glide velocity.

a. Coarse Velocity

Description. - This is a thermometer type indicator with a scale length of five inches covering the velocity range from zero to 26,000 ft/min. The programmed velocity is indicated by a double bar moving vertically along the left hand side of the scale.

Inputs. - Two signal sources are provided for the instrument. Normally, the horizontal velocity is generated within the Inertial Navigation System and the programmed velocity within the General Purpose Computer. Provisions have been incorporated in the display system to drive the instrument from a barometric system below 60,000 feet. A switch is provided on this display group to transfer from the primary source to the barometric system at the pilot's discretion.

Usage. - The pilot will use this display to determine the velocity during the boost phase when the velocity is changing very rapidly. At this time a coarse scale presents the information in a readily interpreted manner.

b. Fine Velocity

Description. - Studies have indicated that an expanded scale will be required to permit adequate monitoring and/or pilot control of velocity during the glide phase. Therefore, velocity will be displayed by a servoed tape type of instrument.

Consideration of the factors involved indicated that the scale factor of the tape should be 1000 ft/sec per inch of tape. This scale is optimized for the entire flight regime. It may be necessary to introduce some further expansion of the scale below 1000 ft/sec in order to provide the pilot with an optimized display for the final approach and landing.

The actual velocity is indicated by a single bar and the programmed velocity by a double moving bar. The tape and bar drive mechanisms are so integrated, that under normal flight conditions the single and double bars should form a single wide bar at the center of the scale.

Inputs - The inputs to this instrument are identical with those of the coarse velocity indicator described previously.

Usage - This display will be of prime importance during Phase II of the glide (from 300,000 feet to 100,000 feet) when the energy management program is dependent on the accurate control of velocity. It will be used in connection with the velocity control graphic screen display when an accurate reading of velocity is required by the pilot.

c. Range-To-Go

Description. - This display is also an expanded scale servoed tape type instrument to provide the readout accuracy required. A scale factor of 1000 nautical miles per inch will be used throughout the flight.

The range-to-go is indicated by a single bar and the maximum and minimum available ranges are indicated by two vertical moving indices.

Inputs. - The range-to-go (distance to target) is generated within the Inertial Navigation System The maximum and minimum available range indices are generated in the General Purpose Computer.

Usage. - The pilot will use the instrument to determine his actual range-to-go and should this value fall outside the available range, as shown by the indices, he may then choose an alternate destination. This choice will be made in conjunction with information obtained from the map display and the graphic screen display.

7. Acceleration Indicator

This instrument is located at the right-hand end of the line of primary flight instrument at the level of the pilot's eyes. Item 10 of Figure III-5 shows the exact location. Figure V-7 shows the instrument face details.

a. Description

The instrument displays the three orthogonal accelerations, longitudinal (fore and aft), normal (vertical) and lateral on respectively two vertical and one horizontal scales.

Indices are provided on the vertical and lateral acceleration scales to indicate the maximum permissible values of these accelerations. In addition to the visual indices, suitable electrical contacts are provided to illuminate the High G warning light on the warning panel. Separate indices have been provided on the vertical acceleration scale for the boost and glide phases.

Inputs. - The orthogonal accelerations of the vehicle are sensed by two sets of accelerometers. One set containing normal, lateral, and axial pickups is located near the cg of the airplane. The second set, containing only normal and lateral pickups is located at an appropriate point in the booster to obtain the proper information during boost. Automatic switching will be accomplished at separation.

Usage. - The instrument will be used by the pilot to monitor acceleration throughout the boost and glide phases of the flight. During the boost phase, should acceleration limits be exceeded as indicated by the instrument, the pilot will initiate the booster shutdown sequence.

For a similar indication during the glide phase the pilot will assume manual control and attempt to restore the airplane to a less severe flight condition.

8. Surface Temperature Indicator

This instrument (item 6, Figure III-5) indicates temperature of a leading edge position, a body station, and a lower surface point. It also displays predicted values (30 to 60 seconds hence) for the same points The instrument face is shown in Figure V-8.

a. Description

The instrument has a cathode ray tube display area approximately two inches high by three inches wide with two fixed horizontal indices marked (sustained flight maximum and momentary maximum). The cathode ray tube face is divided into three vertical areas; marked leading edge; nose; and lower surface. Above these areas the instructions pitch up, pitch up, pitch up, pitch down, are arranged in the same order. Focus and position controls as well as a press-to-test button are located at the left side of the panel face. Temperature is indicated by traces on the cathode ray tube. For any given area the lower end of the trace indicates present temperature while predicted temperature is indicated by the upper end. To avoid confusion, only predicted values which are greater than present values are shown. Thus steady state or decreasing temperature is indicated by a single dot. Only the temperature range from the sustained flight maximum to 500°F below this maximum, is shown. This provides sufficient scale expansion to perceive motion at the expected normal rate of increase.

b. Inputs

Thermocouples or other sensors located in the airplane outer surface will provide signals which will be amplified as required and fed to the cathode ray tube display circuitry. Predicted temperatures will be computed based on rate-of-change of present temperature.

c. Usage

The DS-I airplane, in its descent, passes through a region of high heating, down a "corridor" bounded at bottom by maximum permissible structural, and temperatures bounded at the top by maximum available lift. This corridor is made as wide as possible by the aerodynamic configuration and the use of new heat resistant materials. However, to maximize range and maneuvering capability, and ensure safety of the crew, monitoring of structural surface temperatures has been provided.

B. BACKUP FLIGHT CONTROL DISPLAY

The purpose of the backup flight control displays is to provide the pilot with a second set of basic instruments to assure a high degree of display reliability in case of failure in the primary flight control displays The following four instruments located in Figure III-5 comprise the backup displays.

Altimeter	(16, Figure III-5)
Airspeed Indicator	(14, Figure III-5)
Rate of Climb Indicator	(17, Figure III-5)
Attitude Indicator	(15, Figure III-5)

1. Altimeter, Airspeed Indicator, Rate of Climb Indicator

Three of the backup instruments, the Altimeter, Airspeed Indicator and the Rate of Climb Indicator are conventional flight control instruments and presently available

a. Description

The altimeter shown in Figure V-9 will be a three inch Kollsman Counter Altimeter, Type 2371-02 or equivalent with an operating range of 0 to 50,000 feet.

The Airspeed Indicator (Figure V-10) for this group will be a Pioneer Control (Bendix) Type 1426-1 or equivalent. It will display indicated air-speed over a range of 49 to 300 knots.

Rate of descent will be displayed on a MS28049-1 rate of climb indicator or equivalent. (Figure V-11) The Instrument has a three inch dial face and a logarithmic scale to facilitate readout of low rates of change. While the range of the instrument is for rates from 0 to 6000 ft-min, rates of descent in excess of this will not damage the instrument.

b. Inputs

Static pressure piped directly from the pitot tube is used by all three in addition to impact pressure required for the airspeed indicator. These pressures are made available below approximately 60,000 feet when the nose cap is blown off and the pitot tube extended.

c. Usage

The primary use of these three instruments will be to provide a simple, reliable backup for the primary displays for the approach and landing phase of flight.

2. Attitude Indicator

The fourth instrument of the backup group is a modified version of an existing instrument and is shown in Figure V-12.

a. Description

The attitude indicator is a two axis moving background attitude reference similar to the Lear, Inc. Model ARU-2/A. Both the pitch-roll and the yaw command bars have been removed to keep the instrument as simple as possible.

The instrument presents the following information:

(1) Airplane attitude in pitch and roll by means of a three inch sphere free to rotate in two planes.

(2) Angle of attack - angle of attack information is displayed by a vertically moving index on the left side of the instrument.

(3) The master "OFF" flag is located in the center of the lower left quadrant of the instrument.

(4) The "OFF" flag for angle of attack information is located at the 0900 position.

A multi-position switch is provided to lock out the sphere during the boost, and to free the side slip indicator.

b. Inputs

The signals driving the two axis (pitch, roll) of the ball are supplied from the inertial platform and are suitably resolved in the General Purpose Computer to convert the Geometric Axes of the inertial platform to the body axes of the airplane. This is the same input as is supplied to the flight director.

An output from the General Purpose Computer supplies the signal to the angle of attack section until the approach and landing phase. At this stage the angle of attack signal will be derived from air data rather than from the Inertial Navigation System.

c. Usage

From the time of boost separation the attitude indicator backs up the flight director for attitude and angle-of-attack information and may be used in place of the primary instrument in the event of its failure.

C. FLIGHT CONTROL SYSTEM CONTROLS

The Flight Control System Controls consist of the following three groups:

(1) Flight Control System Control Panel (25, Figure III-5)

(2) Side Stick Controller (62, Figure III-5)

(3) Surface Position Indicators (39, Figure III-5)

A brief description of the Airplane Flight Control System operation is as follows.

Four flight control modes and two channels of operation are provided for selection by the pilot. The modes require various degrees of pilot effort. In the Guidance Command Mode the pilot monitors the performance of the Inertial Navigation System/Flight Control System combination. In the Pilot Command Mode he must vary attitude but the control system will hold the attitude desired with the assistance of the Inertial Navigation System platform. In the Piloted Flight Mode the pilot must vary and maintain the attitude desired but the automatic control channel, required for operation in the higher modes, is retained to provide optimized feel and airplane dynamic performance. In the Backup Flight Mode the pilot must vary and maintain the attitude desired; this is accomplished through the backup channel, which provides the least complex means of achieving acceptable aircraft control. A more detailed description of the Airplane Flight Control System is given in Appendix A.

The availability of alternate modes and channels accomplishes the following:

(1) The pilot may fly the airplane in an optimum manner suitable to his desire or to a wide range of situations which may be encountered.

(2) The pilot is provided with two parallel paths of control significantly increasing the reliability of flight control.

1. Flight Control System Control Panel

The Flight Control System Control Panel contains the controls necessary to operate the Flight Control System and the displays required to monitor its operation. The items on this panel shown in Figure V-13 are as follows:

(1) Three push buttons with integral lighting. These buttons are used to engage the various modes of the automatic flight control system. These three push buttons are suitably interconnected electrically so that only one mode can be engaged at a time. Pushing a second button will disengage the previously selected mode and engage the new selection. The illumination of the integral lighting within a push button indicates that the mode is engaged.

The Pilot Command and Piloted Flight Modes may be engaged at anytime since these modes are always synchronized. Because of the nature of the Guidance Command Mode, synchronization is not automatic. It is required that the pilot maneuver the airplane so that the computed attitude command and the actual attitude of the airplane be in agreement if a transient is to be avoided. For a normal mission to take full advantage of the Inertial Navigation System the Guidance Command Mode will be engaged at all times.

(2) A jewel type light to indicate that the Pilot Command Mode is synchronized.

(3) A light to indicate that the Backup Flight Mode is engaged (the switch to engage this mode is on the grip of the side stick controller).

(4) A push button with integral lighting to engage the automatic landing system. This push button will be suitably interconnected electrically so that the automatic landing system can only be engaged when operating in the Guidance Command or Pilot Command Modes of the Automatic Flight Control System. In addition, operation of the push button will switch the command displays on the flight director from the INS to the automatic landing system. Illumination of the integral light indicates that the automatic landing system has been engaged.

(5) A toggle switch center-off-momentary-on in two directions to provide manual yaw (beep type) trim in the Piloted Flight and Backup Flight Modes.

(6) A three position toggle switch to control hydraulic power to the servo actuators. The switch positions are as follows.

(a) Position 1, Hydraulic System No. 1 providing hydraulic power to Section A of the dual servo actuators.

(b) Position 2, Hydraulic System No. 2 providing hydraulic power to Section B of the dual servo actuators.

(c) Position 3 (mid position), Hydraulic Systems No. 1 and No. 2 providing hydraulic power to the respective sections of the dual servo actuators.

(7) A jewel type warning light. Illumination of this light indicates that hydraulic system pressure No. 1 is low.

(8) A jewel type warning light to indicate that hydraulic pressure in system No. 2 is low. Should either of the hydraulic warning lights illuminate, the pilot may change the position of the hydraulic power switch from both, to the position corresponding to the remaining functioning system.

(9) Servo actuator malfunction light. Illumination of this light indicates that there is an operating discrepancy within one of the dual servo actuators. It further indicates, when the failure results in a transient, that the source of failure is in the actuator rather than in the automatic control channel. In addition, it will indicate those failures of a gradual nature that might very well be masked by trimming.

It is expected, particularly when the failure is accomplished by a disturbance to the airplane, that the pilot will immediately engage the Backup Flight Mode and attempt to regain control through the cockpit controllers. He should simultaneously change the position of the hydraulic power control system from both position to either system No. 1 or No. 2. This will cause the hydraulic pressure in the opposite system to decay at a slow controlled rate. As the pressure decays the pilot will be able to ease up on the controls if he is depressurizing the correct system. If not, the switch must be thrown to the opposite direction.

The light will be suitably interlocked with the pressures in the hydraulic systems so that if either systems pressure is low, the servo actuator malfunction light will be inoperative (this condition is indicated by the low hydraulic pressure warning lights).

2. Side Stick Controller

The provisions for manual control include a three axis wrist type side stick controller. This controller is used to activate the reaction controls, when required, as well as the aerodynamic surfaces and to provide a means of pilot control during the boost phase. The overall system design takes full advantage of the available aerodynamic control forces prior to activating the reaction controls so that in general the control surfaces will be at the extreme of their travel before the reaction controls cut in. A detailed discussion of the results of the analog computer studies on the reaction controls can be found in References 11 and 12. While inputs for attitude control of the booster and the airplane are normally from an Inertial Navigation System, the pilot has the option of taking over command and supplying the inputs through the controller. The pitch and roll trim button, the backup switch, and "g" buzzer are also mounted on the stick.

The suitability of a side stick controller for piloted flight with the pilot in a full pressure suit has been well established (see References 2 and 9). Furthermore, much work has been done relative to the addition of the third axis of control (yaw) to the stick and the elimination of the rudder pedals. This work clearly indicates that improved piloted flight performance can be achieved through the use of a three axis stick. Typical accounts of the work accomplished in this field can be found in References 3, 4, 5 and 6.

Additional work, in connection with the X-15 program, has shown that rudder pedals are not satisfactory for high performance vehicles under some flight conditions, especially in the absence of stability augmentation.

These studies have also shown that the use of rudder pedals becomes increasingly difficult if not impossible under conditions of very high normal accelerations. Typical examples of the work accomplished along this line can be found in References 8 and 10.

Heretofor, one of the basic reasons for not eliminating rudder pedals in present day aircraft has been the extreme difficulty and complexity involved in mechanizing a mechanical backup system using a three axis stick. The choice of an electrical backup flight control system for the Dyna-Soar airplane eliminates this problem.

Based on results of experimental work previously accomplished and on a consideration of requirements peculiar to the Dyna-Soar I airplane the decision to eliminate rudder pedals and incorporate a three axis side stick controller was made for the following reasons:

(1) Centrifuge studies have indicated that unsatisfactory pedal control was experienced when operated at accelerations greater than six g. Acceleration in excess of six g is expected during boost.

(2) Use of an electrical backup removes the design complication of integrating three axes in a single controller operating a mechanical linkage system.

(3) Accumulated experimental evidence has consistently indicated that a pilot can perform better with a three axis stick than with a two axis stick and rudder pedals.

(4) Simulation studies have shown that rudder pedals are not satisfactory for high performance vehicles under certain flight conditions without stability augmentation.

(5) Elimination of pedals saves weight and allows improved utilization of cockpit space.

There are numerous possible designs of a three axis side stick controller - one handed, two handed, wrist (where all pivot points are through the natural pivot points of the operator's wrist). The selection of a wrist type stick was made after careful consideration of the following factors:

(1) References 1,3 and 7 show that one hand operation is as good as two hand operation of a manual control. In addition it is highly desirable to have one of the pilot's hands free at all times to perform other necessary tasks.

(2) Certain other types, e.g., ball type, gimballed stick with sliding contacts, are very difficult to mechanize reliably even with an electrical backup system.

(3) Other designs were eliminated because of a loss of control/display compatability, induced by the fact that the axes of rotation of the stick were not orthagonal with the axes of rotation of the aircraft or of the display.

(4) A wrist type stick is fully operable with the pilot's arm restrained, i.e., strapped down; a very important consideration for manual control during the boost phase when accelerations are high.

(5) In line item 4 a wrist stick can be mass balanced in all axes. A breadboard model of a three axis wrist stick has been fabricated and is now being used in simulated flight studies. This controller is spring restrained requiring definite motion of displacement, roughly proportional to the force applied, to obtain an output. This design was selected rather than a pure force stick for the following reasons. The design of the Backup Mode

should be as simple as possible to provide maximum reliability. It was therefore established that there should be no electronics between the controller and the surface actuator.

It is not possible to obtain an output of sufficient magnitude for a pure force controller having negligible displacement without electronic amplification. Electronic amplification can be avoided however, if sufficient controller motion is provided to drive a simple potentiometer. Several physiological considerations such as available wrist torques, wrist rotational limits and tactual sensing of the hand are important considerations in the design of a wrist stick. In addition the mechanical weight and space required and the available motion sensing devices are also of paramount importance. Therefore, the final design is, in general, a compromise resulting from the consideration of all pertinent factors.

Figure V-14 is a plot of torque verses deflection of the wrist type stick. This was established after careful consideration of the human factor and electro-mechanical design criteria. The small center dead space provides a positive detenting action without introducting unduly large breakout forces and consequent overshoot when attempting to introduce small commands. This detenting action will prevent the inadvertant introduction of a command into one axis while the pilot is applying full deflection in another axis. The force gradient in this region is sufficient to center the stick in the absence of external forces and to provide the pilot with the tactical information that he is not commanding an output. At the end of the dead zone there is a small sharp rise or change in slope of torque required to produce additional motion. This change in slope provides a clue to the pilot that he is now introducing a signal. The torque gradient over the next 16 degrees is approximately constant and is the region where stick output is proportional to torque. At this point there is another sharp increase in the torque gradient. Additional motion of 1-1/2 degrees will, in the Backup Mode, close a switch activating the reaction controls. Further motion of one degree brings the stick against the fixed stop. This sharp change in torque gradient serves as a tactual clue to the pilot that he is activating the reaction controls.

Stick rotation in all axes has been limited to ±20 degrees so that combined motions, roll, pitch and yaw remain well within wrist capabilities. These motion limits require that suitable gearing be provided between actual stick movement and pick off rotation.

The transducers to convert stick motion to an electrical output are synchros for the automatic modes and ruggedized direct current potentiometers for the Backup Mode. The detenting action is provided mechanically.

The hand grip has been tipped forward 15 degrees for maximum operator's comfort and to assure full range of pitch deflection. It was felt that for roll, a vertical handle reference was more important than the small additional deflection force that could be gained by tipping the handle. This is particularly true as the stick load is well below operator capability and as is the required wrist motion.

A trigger switch has been added to the stick grip in order to connect the stick dirtctly to the booster flight control system when in the boost phase or to the servo actuators or reaction controls when in the glide landing phase. The position of the trigger will allow the pilot to go to direct control of the vehicle in a minimum of time at an indication of malfunction in the automatic channel.

Trim buttons for manual trim of the airplane pitch and roll when in the Piloted Flight or Backup Mode are located on the stick. These are not used in the other modes as automatic trim is provided.

A stick buzzer receiving its signal from an accelerometer will be used to warn the pilot when an excessive normal acceleration has been reached. The pilot has the option of exceeding this level if he feels the situation warrants it.

This wrist type three axis stick controller has been installed in the simulator, and pilot comments resulting from its operation in the simulation program will be used to guide the refinement of the design.

3. Surface Position Indicator

This group consists of two instruments to display elevon and rudder position and is shown in Figure V-15.

a. Description

Each instrument will have two needles to indicate the position of the left and right hand surface. As the elevons may move in either direction, each pointer on the elevon indicator will be identified with the corresponding surface. The rudder indicators will each move outward from zero only corresponding to the surface motion.

b. Input

The inputs to the surface position indicators are obtained for electrical position transducers which are integrally mounted in the servo actuators which drive the surfaces.

c. Usage

Under steady state conditions, the difference between the position of the right and left hand elevon positions will indicate the roll trim, and the average displacement of the two needles from the zero position will indicate the pitch trim. In a similar manner, the difference in the position of the two rudders will indicate the yaw trim.

Stick position in the Piloted Flight Mode corresponds to a commanded rate of change of airplane attitude. At or near a stall condition, the elevons will continue to move out without a corresponding change in airplane attitude.

Therefore, during a manual landing in the Piloted Flight Mode, the elevon position indicator can be used by the pilot as a warning that he is approaching a stall condition.

In addition, the surface position indicators will be used during the prelaunch checkout of the FCS and a double check that the surface positions are trimmed prior to take-off.

VI. GUIDANCE SYSTEM DISPLAYS AND CONTROLS

The guidance display and control group is designed to allow the pilot to monitor the operation of the guidance system, and in case of malfunction to take the proper corrective action which will allow him to reach a landing site safely.

A map display shows the pilot his present position and heading which he can compare with the nominal flight path and flight paths to alternate landing fields. Detail maps of landing fields are provided. A fine indication of cross-range error is provided separately.

The pilot monitors ground radar checks and may correct the inertial navigation system on the basis of these checks. He may also switch from one inertial system to the other if a malfunction occurs. Alternate destinations may be set into the system if the available energy does not permit attainment of the nominal destination.

For the landing phase indications showing operation of the automatic landing system are provided. Visibility is available through a window in front and two on the sides.

A. COMPUTER PROGRAMMER PANEL

1. Description

The computer programmer panel is located in position 52 of Figure III-5.

The face of the panel is shown in detail in Figure VI-1. The panel contains on the left three indicators giving position provided by ground radar or either inertial navigation system according to the light above the indicators. Two more columns of indicators show the difference between the indication on the left and each inertial system. A switch changes the navigation system operating the airplane.

Buttons to change destination are provided near the bottom of the panel. A light indicates if the desired destination cannot be reached. Destination in more comprehensive form is shown on the graphic screen displays.

2. Input Signals

During a radar position check, the operating inertial system transmits its memorized position to the ground. The ground radar computes the difference between radar and inertial position and transmits this back to the airplane. This error is displayed directly in the error column corresponding to the operating inertial system. It is also added to the memorized inertial system position, providing the radar position shown in the left column. The memorized position of the other inertial system is subtracted from this radar position to provide the other inertial system error. All of these computations take place in the general purpose computer of the navigation system.

In checking the discrepancy between the inertial systems, the information for the first column is obtained directly from the desired inertial system while the discrepancy is computed as the difference between the two system outputs.

The "NOT AVAILABLE" light operates by comparing range-to-go with maximum and minimum range attainable in the general purpose computer. Range-to-go is available from the DDA; maximum and minimum range is computed in the general purpose computer as a function of altitude and velocity.

3. Usage

a. To Insert Error Correction Data from Ground Radar

When the airplane passes over a ground radar station, a position check is initiated by the ground station. If the station neglects to initiate such a check, the pilot will request it via the voice data link.

The ground radar transmits a pulse to the airplane which causes the navigation system computers to memorize their respective position indications at the same time at which the radar makes its position fix. This timing pulse also turns on the "RADAR" light at the upper left of the panel. A few seconds later, the airplane's position is displayed in the three indicators at the left. At the same time, the differences between this radar position and the memorized position of each inertial system are displayed in the two columns labeled "INS #1 ERROR" and "INS #2 ERROR". If the pilot wishes to correct either system, he pushes the "INSERT" button at the top of the corresponding indicators. This will insert the correction, and change the indicators in this column to zero.

b. To Check Discrepancy between Inertial Systems

If at any time the pilot wishes to check the discrepancy between the two inertial systems, he pushes either the "INS-1" or the "INS-2" button above the left column of indicators. If, for instance, he pushes the "INS-1" button, the button lights up, and the left column indicates the airplane's position as indicated by the primary navigation system at the time the button was pushed. At the same time, the right column, "INS-2 ERROR", indicates the difference between the two systems that existed at the instant the button was pushed, while the "INS-1 ERROR" column indicates zero.

If, at any time "INS-1" or "INS-2" button is pushed, the airplane receives a time pulse from a ground radar station, the light on the "INS-1" or "INS-2" button goes off, the "RADAR" light goes on, and the sequence described in a. commences. In other words, operation a. overrides operation b.

c. To Change Navigation System Guiding the Airplane

If the errors of the navigation system currently connected to the flight control system are excessive, or if a malfunction is indicated, the pilot switches to the other system by selecting "INS-1 On" or "INS-2 On" located near the top of the panel.

d. To Change Destination

The eight buttons near the bottom of the panel correspond to eight preset destinations. Ordinarily, the button corresponding to a pre-determined destination will be lit. If energy conditions are such that the airplane is unable to make this destination, the sign "NOT AVAILABLE" will light up. The pilot will then choose the next alternate destination by pushing the corresponding button. This inserts the alternate destination into the "INS", and changes to range-to-go indication on all applicable displays. The button then lights up and the "NOT AVAILABLE" light goes out.

B. CROSS-RANGE ERROR INDICATOR

1. Description

The cross-course error indicator is located in position 13 on Figure III-5. It is shown in detail in Figure VI-2. The instrument indicates the cross-course error in miles on a scale moving behind a stationary pointer. A scale of 10 miles per inch is used.

2. Input Signals

The cross-course error is obtained as the difference between the cross-range from the y-axis channel of the inertial system and the programmed cross-range from the general purpose computer.

3. Usage

The cross-course error indicator gives the pilot a more accurate read-out of cross-range error than he gets from the map.

During the early portions of glide, the pilot does not correct the cross-course error primarily because the roundness of the earth will tend to cancel it half way around. In the second half of the glide, the cross-course error will be corrected automatically if the FCS is in the guidance command mode, and the pilot will monitor the operation of the system from this instrument; if the FCS is in another mode, the pilot will hold the cross-course error near zero by lateral maneuvers.

C. MAP DISPLAY AND CONTROL PANEL

1. Description of Instrument and Usage

The map display is located in position 23 of Figure III-5. As shown in detail in Figure VI-3, it consists of a screen nine inches high by twelve inches wide on which a terrain map is projected. Figure VI-4 illustrates the proposed display mechanism. The width of the map corresponds to 2000 n.m., equivalent to a scale of about 170 n.m. per inch. The height then corresponds to 1500 n.m. A small representation of the airplane is shown about an inch above the bottom edge of the display. When the selection switch on the Map Control Panel (located at position 50 on Figure III-5 and shown in detail in Figures VI-5 and VI-6) is placed on "MAP", the strip film showing outlines of continents and islands is driven in accordance with along-range position, the airplane indicator is driven sideways according to cross-range position. The desired flight path is indicated on the map as a black line, the actual horizontal flight path angle by a shadow cursor. Alternate bases and the flight paths to each base are also shown. Other features of the map may include positions of radar ground stations where position checks are expected, and positions of emergency bases where rescue facilities are available.

A shaded area showing the range of the escape capsule (if ejected) will also be provided. This allows the pilot in an emergency to time separation of the capsule so that impact occurs in the vicinity of a rescue facility.

The pilot is able to slew the map independently of motion of the airplane to view any portion of the flight path in the future or past by moving a slewing switch on the Map Control Panel. The map automatically returns to the present position when he pushes the "TRACK" button on the same panel.

Detail maps of landing areas are available on slides to be projected on the map display. Placing the selector on the Map Control Panel on "INDEX" projects an index of detail maps each identified by a three-digit number. The pilot inserts the appropriate number with the "FRAME SELECT" thumb wheels and sets the selector on "SLIDES". The proper map is then projected on the display. This map will be similar to a standard Air Force chart with special terrain identifying features. The map scale will be 1:250,000 or 3.43 n.m. per inch corresponding to an area of 27.4 n.m. by 37.68 n.m. The map is stationary; the airplane indicator moves across it corresponding to its actual position. A typical detail map is shown in Figure VI-7.

The areas covered by the detail maps are also shown and labeled by number on the terrain strip map. This enables the pilot to select the detail maps without the selector, and informs him when the airplane is in an area covered by a detail map.

In addition to detail maps, a number of slides are available containing check lists and other pilot aids which may be projected in a manner similar to that of projecting the detail maps.

A brightness control is included in the Map Control Panel.

2. Input Signals

The strip map is driven directly by the x-axis position output from the inertial system. The lateral motion of the airplane indicator is driven directly by the y-axis position output of the inertial system. The cursor showing escape capsule range will be driven by an output of the inertial system's general purpose computer which obtains this range as a function of velocity and altitude.

When a detail map is presented and the airplane is within its region, the airplane indicator is driven longitudinally and laterally by the x-axis and y-axis outputs of the inertial system, respectively.

D. TIME INDICATOR

1. Description

The time indicator is located at position 12 in Figure III-5 and is shown in detail in Figure VI-8. It contains two digital indicators, one showing Greenwich Mean Time in hours and minutes, the other showing time elapsed from launch in hours and minutes. Seconds of elapsed time are indicated by a conventional sweep hand. Time to go to destination is indicated by a vertical bar display. A "RESET" button for the elapsed time indicator and a "SET" control for the Greenwich Mean Time indicator are also included.

2. Input Signal

Time pulses to drive the two digital indicators are obtained from the digital computer of the inertial system. Time-to-go is computed by the general purpose computer as a function of range-to-go to destination. Elapsed time and time-to-go indicators are started by a ground signal at "1st Stage Fire".

3. Usage

The time indicator informs the pilot of the elapsed time, and of the remaining time of flight. A change in destination will result in a change in the indicated time to go. A "RESET" button resets the elapsed time indicator.

Greenwich Mean Time is set into the indicator by pushing in and turning the "SET" control knob. This will be accomplished prior to launch according to voice instruction via the data link.

E. AUTOMATIC LANDING SYSTEM INDICATOR AND CONTROL

1. Description

The Automatic Landing System ("ALS") Indicator and Control is located in the Flight Control System Panel described in Chapter V. It is a button which is used to engage the ALS, and which lights up when the ALS is engaged.

2. Input Signal

The signal indicating ALS engagement is obtained from the relay which is used to couple the ALS to the flight indicator and to the FCS, the latter only if it is in one of the two command modes.

3. Usage

At the end of the approach phase, about 10 n.m. from the runway, the airplane passes through the ALS gate. Normally, the pilot is in voice contact with the ground, and is told when he may engage the ALS by push- the ALS Control. If the pilot should be unable to do so and the FCS is in the guidance command mode, the system may be engaged from the ground. In either case, the button lights up, indicating that the ALS is engaged.

If the FCS is in either the guidance or the pilot command mode, the ALS is coupled directly into the FCS and the pilot only needs to monitor the flight director to make sure that the command bars are at zero. If the FCS is in the piloted flight on back-up mode, the ALS indicates the errors in the flight director and the pilot must fly the airplane to keep the command bars at zero.

F. LANDING VISIBILITY

Forward visibility during landing is provided by a single nose window, and side visibility is provided by windows located on each side of the cabin. The nose window is protected by a molybdenum nose cap during the high pressure portion of the flight. On the initial approach to the landing field at an altitude of 60,000 feet and a speed of M=1.5, the nose cap is jettisoned by the pilot's action in closing the nose cap ejection circuit.

The nose window is a transparent spherical segment having a mean diameter of approximately 21 inches. It is faired with the fuselage skin lines in the position indicated on Figure VI-9. Forward vision through this nose window is made possible by swinging a centrally located portion of the instrument panel to its stowed position behind the panel. This provides a 9 x 12 inch rectangular aperture in line with the nose window and the pilot's eyes which affords a pyramidal field of vision forward and down of 20 degrees in the vertical plane and 24 degrees in width. Horizontally, the vision angle is symmetrical about the centerline. In the vertical plane the mean visual angle is depressed 15 degrees to the ship's longitudinal axis.

The visible ground area is a function of altitude, flight path angle, and angle of attack. Figure VI-10 illustrates the nose window visibility during final approach for the power-off glide path. It can be seen that the runway is visible throughout the final approach and flareout, assuming a 10,000 feet runway with touchdown at 2500 feet from its edge.

The application of power during final approach sets up a higher angle of attack condition which elevates the field of vision. A partial power approach, which would be used in the event of a missed approach and go-around or a malfunction of the automatic landing system, results in a flat glide path at low altitude and an attitude that provides visual contact with the horizon and a field of veiw sufficient to land the airplane.

The side vision is currently based on round windows approximately 11-1/2 inches in diameter. The fuselage structure in the side area of the cabin permits a great deal of freedom in locating these windows. Their present location is based on providing forward lateral fields to assist in lateral (roll) and pitch orientation during the final landing and touchdown phase. Due to the structural design of this area, additional side vision can be provided at the expense of some increase in weight, if it later appears advisable.

VII. AIRPLANE PROPULSION AND POWER GENERATION SYSTEM DISPLAYS AND CONTROLS

The airplane propulsion and power generation systems require the operation of certain control functions by the pilot prior to take-off and the monitoring and actuation of controls during flight, particularly in case of equipment malfunctions.

Controls and displays are provided for the following elements of the airplane propulsion and power generating system.

The propulsion and power generation systems consist of the following items:

(1) Separation Rocket Engine. This engine which for a duration of 2 or 0.2 seconds depending upon the time of usage is employed to separate the airplane from the boosters in case of a booster malfunction. Because of its simple modes of operation, very little display and control information is required in the cockpit.

(2) Range Control Engines. Two retro rocket engines of five pounds thrust each are provided to reduce the range of the airplane when actuated. They will be

used after separation from the boosters. In addition, the exhaust of the two power-generating units is used to provide forward thrust when required for range control. Although controlled automatically, a throttle-type control handle on the left console can be used by the pilot for manual control.

(3) Reaction Control Engines. During the period in which the airplane is flying at extremely low dynamic pressures, reaction control rockets are provided for pitch, yaw, and roll control to supplement the aerodynamic controls. The pilot can control the reaction controls manually, if he desires, by deflecting the three-axis control stick to its limits.

(4) Landing Engine. A turbojet engine is to be used during the landing phase of flight to provide a go-around capability and also to provide some extension to the range of the airplane. The displays and controls for this engine have been held to a minimum and consist of conventional types.

(5) Power-Generating Units. Two power-generating units are provided for supplying electrical and hydraulic power to the airplane subsystems. Controls and indicators for these units are provided to indicate the station of the power units, the fuel, and the generator being driven by the power units.

A. RANGE CONTROL

1. Description

Controls for the range control system consists of a manual control handle located on the extreme left-hand console as shown in Figure III-5 and a two position "arm-off" toggle switch located on the Power Control Panel, Figure VII-1. This panel is Item 29 in Figure III-5. Operation of the range extension or range reduction system is located adjacent to the manual control handle and is indicated by two green lights on the Range Control Indicator Figure VII-2 located adjacent to the manual control handle and is identified as Item 54, Figure III-5.

The manual control handle is a three-position throttle type control with "neutral", "increase range" and "decrease range" positions. A two position switch mounted on the handle is used to select either the manual or automatic mode of operation.

2. Data Input

The signals to actuate the range control indicator lights are supplied from the position indicator of the PGU exhaust valve for range extension and a pressure indicator on the retro-thrust chamber for range reduction. When either of the range control devices are operating the appropriate indicator light is actuated.

3. Usage

Normal operation of the range control system is accomplished automatically but provisions for manual operation are made by use of the range control handle. When this handle is placed in the "increase range" position with the selector switch in "manual" position, the power generating unit exhaust nozzle valve is positioned to produce thrust by expelling the exhaust stream aft. When positioned in the "decrease range" position (selector switch in "manual" position) it actuates the propellant valve of the range reduction rockets. When the selector switch is in "automatic" position, the navigation system operates the range control system.

The range control "arm-off" switch when positioned to "arm", opens the propellant valve located in the feed line between the tank and range reduction rocket propellant valve, and also supplies power to the power unit exhaust nozzle control circuit. In the "off" position the range control system cannot be operated either manually or automatically. This switch is normally positioned to "arm" during launch pad count down.

The indicator lights provide a comparison of the system operation with the movement of the lift error indicator (Figure V-2). For instance, if the lift error indicator shows that range control thrust is required but the range control light is not on, (indicating thrust is not being provided) the pilot will actuate the range control engine manually.

B. REACTION CONTROLS

1. Description

The reaction control propulsion rockets may be actuated normally by the three axis side stick controller when it reaches its limit of travel. A complete description of the side stick controller is presented in Chapter V of this report. The controls consist of an "arm-off" toggle switch, peroxide tank "pressurization-vent" toggle switch, and a peroxide "Jettison" switch all located on the power control panel, Figure VII-1. A warning indication for excessive temperature of the peroxide monopropellant is provided on the Warning Indicating Panel, Figure VIII-3, as well as a warning indication for low propellant supply.

2. Data Input

The hydrogen peroxide temperature warning indicator is supplied by a signal from a temperature sensing device which senses the temperature of the monopropellant as it leaves the storage tank. A red light flashes on and off on the warning indicating panel when the temperature reaches 170° F and remains on continuously when the temperature reaches 180° F. The warning indicates "Hot H_2O_2-Jettison". The peroxide low indicator is signaled by a quantity flowmeter totalizer instrument sensing flow from the tank. This indicator is set to actuate when 1/3 of the propellants remain in the tank.

3. Usage

Operation of the side stick controller which actuates the reaction controls is described in the flight control section of this report

The "arm-off" toggle switch is provided to control the peroxide propellant valve located between the monopropellant tank and the main propellant valve on each reaction control thrust chamber. This switch is operated to arm the system during count down on the launch pad. During boost the system is locked out by the flight control system until airplane separation.

In the event of a malfunction in the reaction control system (propellant valve stuck open or stuck closed), the pilot can deactivate the system by turning the arm switch off.

The peroxide tank pressure switch is actuated to pressurize the tank during count down on the launch pad. The Jettison switch is used by the pilot to dump the peroxide supply in the event of an abbreviated flight or hot peroxide warning.

The low level peroxide warning indicator is purposely set high to warn the pilot of excessive usage of the propellant since both the range and reaction control peroxide propellants are stored in a common tank. He may then take action to conserve the remaining fuel.

When the "Hot H_2O_2-Jettison" warning light is flashing, the pilot knows an emergency condition is imminent. When the light remains on steadily, the pilot must then jettison the hydrogen peroxide propellants or they will explode.

C. POWER GENERATING SYSTEM

1. Description

The two power generating unit displays, "PGU No. 1" and "PGU No. 2" are located on the Power Control Panel, Figure VII-1. An

"Off-Run" toggle switch is provided for each PGU as well as individual green lights ("Run") to show that the units are operating.

Two displays and a warning light indicator are provided for the power generating unit propellant system, which utilizes hydrazine as a monopropellant.

(1) The Power Control Panel contains the "N_2H_4 Tank" display consisting of a "Pressure-Vent" toggle switch and a "Jet" (jettison) switch.

(2) The quantity of hydrazine remaining, "PGU Fuel Qty," is displayed on the "Helium and PGU Propellant Quantity Indicator," Figure VII-3. This indicator is Item 31 on Figure III-5. The "PGU Fuel" gage has a black moving element to indicate quantity of propellants remaining and a cross-hatched fixed indicator to show the minimum amount of propellant required for an emergency descent of the airplane from the top of the trajectory.

(3) The red warning light indicator, "PGU Fuel Low", is located on the Warning Indicating Panel, Figure VIII-3.

2. Data Input

The green "Run" indicators for "PGU No. 1" and "PGU No. 2" are operated by a signal from each power unit control box and indicate when the power unit is operating at normal speed.

The "PGU Fuel Qty" indicator is provided a signal from a flow-meter-totalizer system in the feed line to the power units. The "PGU Fuel Low" warning light receives its signal from this system when 10 percent of the propellant remains.

3. Usage

The "Off-Run" switches for each power unit controls the automatic start and run sequences of the power generating units. (The details of the operating sequence and circuitry of the power units is contained in Reference 13.) Both power units are started and operated on the pad prior to launch. Each power unit is protected with automatic shutdown controls in the event of a malfunction. The remaining operating power unit then automatically takes over the full hydraulic and electric loads of the airplane. The pilot may manually shutoff either power unit at will. The "PGU No. 1" and "PGU No. 2" indicators provide the pilot a means of determining the operational status of each power unit and when to switch the generators on the line.

The "Pressurize-Vent" switch control for the "N_2H_4 Tank" is actuated to pressurize the hydrazine tank before the power units are started. The jettison switch is operated by the pilot in the event of an aborted flight to reduce the landing weight of the airplane.

The hydrazine "PGU Fuel Qty" indicators provide the pilot with information on the propellant remaining, and allow him to determine the quantity of fuel jettisoned during an aborted flight.

The "PGU Fuel Low" warning indicator reminds the pilot of impending loss of power.

D. A-C GENERATOR

1. Description

A-C Generator control is provided as part of the Power Control Panel, Figure VII-1, and includes a switch and two indicating lights for each of the two generating systems.

a. "GEN" Switch

The generator on-off-reset three position toggle switch performs the following functions:

(1) Controls energizing of bus contactor to provide a-c electrical power to the main bus.

(2) Controls resetting of the protection and control system.

(3) Controls manual switching from the operating generating system to the standby generating system.

b. "LOAD" Indicating Lamp

The amber "LOAD" light indicates that the generating system with this lamp glowing is supplying power to the main bus.

c. "STANDBY" Indicating Lamp

The green "STANDBY" light indicates that the generating system with this lamp glowing is operating as a standby system.

2. Data Input

The data input is identical for each of the two a-c generating systems and is as specified in paragraphs a through c.

a. "GEN" Switch

The "GEN" switch connects the "ready signal" output from the Voltage Regulator and Supervisory Unit to the bus contactor coil. Normally closed accessory contacts of the bus contactor for the other generating system are also in series with this switch.

b. "STANDBY" Lamp

The "STANDBY" lamp is connected directly to the normally open side of the accessory contacts described in paragraph a.

c. "LOAD" Lamp

The "LOAD" lamp is connected in parallel with the bus contactor coil. The "ready signal" will be removed under conditions of generator overvoltage, undervoltage, feeder line fault or generator overheat.

3. Usage

Each "GEN" switch is normally placed in the "ON" position after the PGU "RUN" indicating light for that system is glowing. Normal operation is depicted by a glowing "LOAD" light for the operating system and a glowing "STANDBY" light for the standby system.

a. Automatic Transfer

Switching from a generating system that has malfunctioned to the standby system is automatic. Both the "LOAD" lamp and the "STANDBY" lamp for an inoperative system will be extinguished.

b. Manual Reset

The pilot may attempt to reset a system that has malfunctioned by placing the "GEN" switch from "ON" to "RESET" and then returning to the "ON" position. If the "STANDBY" light glows, the system has been reset and will automatically come on the line if a malfunction occurs in the other system.

c. Manual Transfer

Manual switching of generating systems for checkout purposes may be accomplished by placing the "GEN" switch for the operating system from "ON" to "OFF" and returning to the "ON" position.

E. LANDING ENGINE

1. Description

The landing engine system displays consist of a Landing Engine Control Panel as shown in Figure VII-4. This panel is Item 22 in

Figure III-5, and includes a fuel quantity indicator ("Fuel") graduated to read in percent of a full tank with a moving black strip for indication; an exhaust gas temperature indicator ("EGT") graduated to indicate hundreds of degrees Fahrenheit, with the dangerous temperature range marked by cross-hatching; and a tachometer ("RPM") graduated to read in percent of maximum RPM.

The throttle for the landing engine is located on the left console, Item 44 of Figure III-5. It has five distinct positions indicated to provide for proper operation of the engine and its intake system. As shown in Figure VII-5 "Engine Start" switch is located immediately to the right of the throttle.

2. Data Input

The fuel quantity indicator on the Landing Engine Control Panel senses the fuel remaining in the JP6 tank by a capacitance type of transducer. The engine "RPM" indicator is driven by the tachometer system located on the engine. The engine tailpipe temperature indicator "EGT" is driven by a thermocouple system located in the engine.

3. Usage

The various throttle positions for operation of the landing engine are described in step 1 through 5.

(1) "OFF". Position fully aft and all landing engine subsystems inoperative.

(2) "INTAKE EXTEND". In this position, the retractable engine air inlet is extended beneath the aft fuselage into the air stream.

(3) "FUEL ON". The fuel valve between the engine and the tank is opened permitting fuel to enter the engine.

(4) "START AND IDLE". The throttle is placed in this position which provides the correct amount of flow for engine starting and idle power. The start button may be pressed with the throttle in this position to energize the ignition circuit.

(5) "START AND IDLE through MAX. POWER". Continuous modulation of engine power is obtained by throttle movement through this range.

F. AIRPLANE SEPARATION ENGINE

1. Description

The airplane separation rocket engine is controlled by the "AIRPLANE SEPARATE" push button located on the left-hand pilot's arm rest control panel, Figure IV-1. This is Item 51 of Figure III-5 and is discussed in Chapter IV of this report.

2 Usage

The engine is actuated by the "AIRPLANE SEPARATE" switch which supplies a voltage signal to the engine control box. This engine is not operable during boost until an altitude of 4500 feet and velocity of 380 feet/seconds is achieved and the signal from the actuation button is locked out until then by a signal from the flight programmer. From the 4500 feet altitude to Stage I burnout, the Engine Control Box is timed for a two-second operation. From Stage I burnout to 25,400 feet/seconds, the Engine Control Box is timed for 0.2 second operation to conserve propellants required for subsequent PGU operation. At 25,400 feet/seconds, a signal from the navigation system locks out the separation engine since the propellants for this engine are already committed to the Reaction Control and Power Generating Systems for the return glide flight. Also, the thrust of the separation engine could place the airplane in orbit if fixed near the end of Stage II burnout.

VIII. FAILURE WARNING AND EMERGENCY DISPLAYS AND CONTROLS

The displays and controls described in this category have been functionally grouped for the purpose of this report. They include the central warning group, the warning indicator panel, the circuit breaker panel, the fire extinguisher panel, and the capsule escape system controls. These are located in various positions around the cockpit as dictated by their relative importance and usage requirements during the various phases of flight. Figure references in the following detailed descriptions will serve to locate these various items in the applicable illustrations.

A. CENTRAL WARNING INDICATOR GROUP

Fire warning, eject warning, and master caution lights are displayed directly in front of the pilot on the panel, for most rapid perception of their activation, in accordance with HIAD.

Fire warning light (Item 1, Figure III-5) comes on and flashes red if fire develops in any compartment where sensors are located; and the pilot then turns attention to the fire control panel described in subsequent paragraphs.

Eject warning light (Item 2, Figure III-5) glows red when activated from any of several signal sources within vehicle which denote imminent catastrophe requiring ejection to save life. This warning may also be lit by radio signals received from ground safety personnel, who may detect dangerous conditions unknown to the pilot, on the extensive ground-based instrumentation.

Master caution light (Item 4, Figure III-5) glows in steady amber color, when any light in the caution panel is illuminated, directing pilot's attention to that panel. This master caution light is an illuminated push button; the light should be extinguished by depressing it; and will then come on only if a second element on the caution panel is illuminated.

Figure VIII-1 shows the central warning indicator group in full scale.

B. WARNING INDICATOR PANELS

The warning indicator panels (Items 21 and 24, Figure III-5) are located in an important region of the front panel where they may be quickly and readily seen. They display self-explanatory warnings of most important and most frequently expected sources of difficulty, arising in vehicle structure, subsystems, or environment.

The warning and caution indicators are divided into two categories with those failures which will result in probable loss of the airplane grouped in the top panel. As shown in Figure VIII-2, these critical items glow red when illuminated. The caution indicators representing less critical items glow amber when illuminated. These occupy the lower panel, as shown on Figure VIII-3.

The red lighted warning group contains the following indicators:

Hydraulic System Out
Cockpit Leak
Pitch G High
Excess Pitch

Electrical System Out

Hot H_2O_2 — Jettison

Yaw G High

Of these the Hydraulic System Out, Electrical System Out, and the Cockpit Leak warning lights are paralleled with the Eject light in the key warning group. Failure of any one of these items would result in a decision to use the escape capsule.

The Pitch G High, Yaw G High and Excess Pitch lights are paralleled with the Separate Airplane light as these are most critical during the boost phase but do not necessarily indicate loss of the airplane.

H_2O_2 Hot-Jettison, indicates chemical decomposition of the hydrogen-peroxide which is equivalent to a fire in a fuel tank. This requires that the pilot actuate the H_2O_2 dump button located on the console just forward of his left hand.

The caution indicator group is comprised of the following:

PGU Fuel Low	Jet Fuel Low
H_2O_2 Low	Equipment Compartment Overheat
PGU 1 Off	PGU 2 Off
Hydraulic Pressure 1 Off	Hydraulic Pressure 2 Off
Gen. 1 Off	Gen. 2 Off
Struct. Temp High	FCS - Off
Pri. INS Off	Sec. INS Off
Control Surface Servo Actuator Malf.	ALS Off
Cockpit O_2 Low	Liq. O_2N_2 Low
Cockpit CO_2 Hi	Airplane Escape Now Possible
Struct. Coolant Water Lo	Struct. Coolant Water Pressure Off
Warning Lite Test	Equip. Comp. Pressure Low

The, Warning Lite Test, is a push to test switch which illuminates all warning and caution indicators on the panel. It is not self-illuminated.

All lights on the panel are of the dual lamp type required for increased reliability.

A third set of warning indicators displays warning and malfunction data from the booster engines. These displays are located directly above the warning indicator panel. They are described in Chapter IV-B as part of the boost system displays.

C. CIRCUIT BREAKER PANEL

This conventional circuit breaker ensemble is located on the right console (Item 67, Figure III-5), readily within the pilot's reach. The circuit restoration elements shown on it are self-explanatory and were arrived at after a careful analysis of expected fault, protection, and restoration required for all major subsystems within the vehicle. The panel is shown in Figure VIII-4. The breakers pop out when open. The circuit may be restored by depressing the breaker after the trouble is fixed.

D. FIRE EXTINGUISHER CONTROL PANEL

The fire extinguisher control panel is located in the right-hand vertical instrument panel (Item 36, Figure III-5). It contains four conventional fire pull handles and four push buttons arranged as shown in the detail illustration, Figure VIII-5. Each handle marked fire pull is an illuminated control which is lighted by fire detection devices in any of the four regions indicated; engine compartment, equipment compartment, left wing, and right wing. These are compartmented areas containing combustibles such as fuels, hydraulic fluid, etc.

In operation, a fire in any one of the four areas will cause the associated Fire Pull handle to be illuminated. The pilot then arms the system by pulling the handle, and releases the extinguishing agent by depressing the appropriate push button.

E. CAPSULE ESCAPE SYSTEM CONTROLS AND OPERATION

Two of these are provided, one on each armrest (Item 45, Figure III-5). During boost, escape, or any high-G maneuver, the pilot, while otherwise restrained, may grip these handles to restrain his arms. A thumb push button is located on the top of each grip; this arms the ejection system, and frees the handle for further movement. Rotation of the handle toward the seat centerline then initiates the ejection sequence which includes booster cutoff if ejection occurs during boost phase. Each handle is capable independently of arming and initiating ejection.

Handles are located on the adjustable armrest, within hand reach for 5 - 95 percentile body depressions. When these handles are not required, they may be depressed into concealment in the armrests; from here they may be released by pressing push buttons on the armrest.

IX. MISCELLANEOUS DISPLAYS AND CONTROLS

The various displays and controls which are not primarily a part of any of the major instrument groups described in the foregoing chapters are considered in this section. They include:

- Cockpit Environment Group
- Nose Cap Separation Handle
- Landing Gear Group
- Communication Panel
- Telemetering-Recording Panel
- Research Areas
- Panel and Console Lighting Controls
- Capsule Parachute Control
- Hatch Release Control

The location of the above items may be determined by figure references included in the detailed descriptions in subsequent sections.

A. COCKPIT ENVIRONMENT GROUP

Two panels are included to provide information on the means of control over cockpit environment. The Cockpit Environment Indicator, (Item 27, Figure III-5) and the Environment Control Panel (Item 28,

Figure III-5) are located adjacent to each other in a prominent position on the left-hand vertical panel.

1. Cockpit Environment Indicator

The cockpit environment indicator panel shown in Figure IX-1 is intended to provide information at a glance that the cockpit environment system is operating satisfactorily, and that the pilot is not exposed to any important stresses deriving from deviations in thermal or gaseous exchange control.

There are many important parameters of cockpit environment which must be kept within narrow range by automatic or manual control, if undue physiological stress is to be avoided, and performance capabilities of mind preserved. From this group of parameters, the elements as follows were selected to serve as cues of important deviations in condition on quantitative scales on this instrument.

(1) Oxygen concentration

(2) Liquid O_2 - N_2 remaining quantity

(3) Emergency O_2 remaining quantity

(4) CO_2 concentration

(5) Cockpit pressure

(6) Cockpit air temperature

A moving vertical bar is used to indicate change in each variable. Scales and references are arranged so that under normal conditions, a quick glance will show that satisfactory conditions prevail

These indicators derive data directly from sensors provided for the display.

(1) O_2 concentration may be sensed by a polarographic type pO_2 sensor, such that developed by USAF School Aviation Medicine.

(2) Liquid O_2 - N_2 remaining quantity can be measured by integrating flowmeter or a capacitance-type gage in liquid O_2 - N_2 converter.

(3) Emergency O_2 remaining quantity can be sensed by a pressure pickup at the O_2 bottle, where pressure will be proportional to quantity remaining.

(4) CO_2 concentration can be picked up by a rubber membrane and glass pH electrode assembly sensor, of a type under development at Ohio State University.

(5) Cockpit pressure is sensed by any of several simple electro-mechanical pressure pickups.

(6) Cockpit air temperature is easily measured by thermistor elements.

2. Cockpit Environment Control

All controls directly related to cockpit air temperature and cockpit pressure are contained in the cockpit environmental control panel shown on Figure IX-2.

The upper half of the panel carries the temperature controls consisting of a three-position air conditioner control switch marked, automatic, off, manual, and a rotating knob control to increase or decrease temperature. The latter controls function when the three-position switch is in the automatic as well as when in the manual position.

The lower portion of the panel contains the cockpit pressure controls. Three toggle switches are provided: a pressure off-on switch, a liquid gas supply transfer switch, (normal to emergency), and a cockpit pressure dump switch. The latter switch carries the legend, "Do Not Use Above 27,000 Feet", and is guarded to prevent inadvertent operation.

B. NOSE CAP SEPARATION HANDLE

The nose cap eject handle (Item 35, Figure III-5) permits ejection of the nose cover to expose transparency, just prior to approach and landing. The handle is pulled once to fire a ballistic device which will jettison the nose cap.

C. LANDING GEAR GROUP

Two small panels containing all landing gear controls and indicators are located for easy access near the pilot's left hand (Items 33 and 32 in Figure III-5). The landing gear control panel shown in Figure IX-3 contains a "Landing Gear Down" actuator switch with interlock which prevents use at high speeds or altitudes. A separate emergency landing gear switch, also shown, is used for backup and indicates a cartridge-actuated power drive to put landing gear down. The panel also contains a switch to silence the warning horn.

The landing gear indicator panel shown in Figure IX-4 contains three lights which glow when nose wheel and skids have been properly dropped into the "Down" position. These lights are arranged in a triangular pattern representative of the triangle described by the nose wheel and two tail skid positions. The light at the vertex of the triangle indicates nose wheel position, and the right and left lights at the base of the triangle indicate right and left skid position.

D. COMMUNICATION PANEL

A compact communication control panel (Item 56, Figure III-5) is located on the left-hand console, just aft of the computer programmer. It provides four push buttons for receiver-mode selection, as well as a volume control wheel and a four-way selector switch. An On/Off switch is also provided. Arrangement of detail functions of these various controls are shown in Figure IX-5. The pilot has a choice of selecting either receiver No. 1, receiver No. 2, or UHF. Normally, UHF will not be selected unless both receiver No. 1 or receiver No. 2 are not working. The volume control will control the sound level to the pilot's headphones from whichever receiver is in use. For transmitting, the pilot normally uses the "Voice-Video" channel. Should this transmitter fail to operate, the pilot can select the "Tele-Trans" channel and thereby transmit voice through the telemetry transmitter. The "UHF VOICE TRANS" control is provided for emergency use. This is a momentary switch and transmits voice through the rescue radio channel. The "UHF CW TRANS" control is provided as a backup to turn on CW for the rescue radios in the event that it does not turn on automatically when the capsule impacts on ground or water.

In use, the communication panel will allow the pilot to select the receivers and transmitters he desires to use for ground communication, as well as to adjust the sound level of the received signal.

E. TELEMETERING-RECORDING PANEL

This panel is located in the right-hand vertical panel (Item 37, Figure III-5). It contains the following control and indicators, as shown in Figure IX-6.

(1) Telemetering On/Off switch for activating this transmitter and auxiliaries during any period of the flight.

(2) Camera On/Off switch which energizes two small cockpit cameras to view the instrument panel over the shoulder of the pilot.

(3) Recorder On/Off switch which energizes the magnetic tape recorder carried aboard. A "Record" light shows that recording is in progress; and a "Runout Light" indicates that no more magnetic tape remains available for recording.

F. RESEARCH AREAS

A large area of the right-hand vertical panel has been reserved for experimental system displays (Item 34, Figure III-5). Most of the research displays presently being considered will require careful monitoring during test flights to properly evaluate the equipment. For this reason, the research space is located as close to center as is practical.

Panels considered for possible installation in the area are:

Radar Display and Controls
Physical Reconnaissance Displays
Recording Equipment

G. PANEL AND CONSOLE LIGHTING CONTROLS

A preliminary analysis has yielded some of the following design conclusions which are currently planned for installation in the Dyna-Soar I airplane.

The center instrument panel requires self-illuminated instruments, rather than flood lighting, which would wash out the rear projection displays.

The side wings of the vertical panel and the two consoles will be lighted by spot or flood lights mounted above and behind the pilot, projecting a fan-shaped beam. Such lighting supplies simplification of installation if used in place of the self-illuminated instruments.

Construction of the Dyna-Soar I mockup was started well before the preliminary lighting analysis was made. Side panels and consoles already constructed for integral lighting were not altered to agree with the above because of schedule limitations.

An additional spotlight will be available for illuminating check lists and maps and research equipment as well as serve for center panel illumination in event of integral lighting system failure.

White light is satisfactory at levels on the panels and console variable from 30 feet lamberts down. Incandescent sources will be used rather than fluorescent and other discharge or electroluminescent sources.

Light control panels (Items 65 and 66, Figure III-5) are provided which use combined On-Off switches and continuous brightness controls for independent control of instrument panel lights, console lights, and cockpit general illumination. Details are shown in Figures IX-7 and IX-8. Adjustment may be desired by the pilot for personal preference, and for ease of reading instruments, as cockpit light input from the outside varies, through forward and side windows, during later phases of flight.

H. CAPSULE PARACHUTE CONTROLS

"Capsule parachute override" handle (Item 64, Figure III-5) is shown on Figure IX-9 and is used as manual backup, to deploy the capsule recovery parachute, if this step fails in the automatic ejection sequence. Not shown is another handle "Capsule parachute release" on the aft bulkhead of the cockpit, used to manually disconnect the parachute from the capsule, after land or water impact.

The "glide escape" handle is used to escape with the capsule from the airplane using the four ballistic thrusters instead of the escape capsule rocket thrusters, so as not to go into orbit.

I. HATCH RELEASE CONTROL

The hatch release handle (Item 26, Figure III-5), can be used for normal or emergency exits from the cockpit, when desired.

TABLE X-1

DISPLAY-CONTROL INFORMATION CHANNELS

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
1. Boost System		
Boost Sequence Panel	19	
Countdown Start Signal		From Ground Control (Umbilical)
1st Stage Fire Signal		From Ground Control (Umbilical)
Lift Off Signal		From Ground Control (Umbilical)
1st Stage Cutoff Signal		From Booster Auto. Sequencer
2nd Stage Fire Signal		From Booster Auto. Sequencer
1st Stage Separation Signal		From Booster Auto. Sequencer
2nd Stage Cutoff Signal		From Booster Auto. Sequencer
2nd Stage Separate Signal		From Booster Auto. Sequencer
Boost Engine Status Panel	20	
Stage 1 Engine 1 Turbine RPM		From Booster Prop. Surv. System
Stage 1 Engine 2 Turbine RPM		From Booster Prop. Surv. System
Stage 1 Engine 3 Turbine RPM		From Booster Prop. Surv. System
Stage 1 Engine 4 Turbine RPM		From Booster Prop. Surv. System
Stage 1 Engine 1 Chamb. Press.		From Booster Prop. Surv. System
Stage 1 Engine 2 Chamb. Press.		From Booster Prop. Surv. System
Stage 1 Engine 3 Chamb. Press.		From Booster Prop. Surv. System
Stage 1 Engine 4 Chamb. Press.		From Booster Prop. Surv. System
Stage 2 Turbine RPM		From Booster Prop. Surv. System
Stage 2 Chamber Pressure		From Booster Prop. Surv. System
Stage 1 Malfunction		From Booster Prop. Surv. System
Stage 2 Malfunction		From Booster Prop. Surv. System
Boost Engine Control Panel	51	
Engine 1 Cutoff Signal		To Stage 1 Engine 1
Engine 2 Cutoff Signal		To Stage 1 Engine 2
Engine 3 Cutoff Signal		To Stage 1 Engine 3
Engine 4 Cutoff Signal		To Stage 1 Engine 4
Stage 1 Cutoff Signal		To Booster Stage 1
Stage 2 Fire Signal		To Booster Stage 2
Stage 1 Separate Signal		To Stage 1 Separation Mechanism
Stage 2 Cutoff Signal		To Booster Stage 2
Stage 2 Separate Signal		To Stage 2 Separation Mechanism
Boost Command Mode Signal		To Booster Control System
Airplane Separate Signal		To Stage 1 & 2 Cutoff Controls and Airplane Separation Engine

TABLE X-1 (Cont)

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
Boost Corridor Graphic Screen	18	
Altitude, Actual		From INS #1/INS #2
Velocity, Actual		From INS #1/INS #2
Screen Switching Signal		From INS #1/INS #2
Graphic Screen Control Panel	63	
Illumination Brightness		To Graphic Screen
Boost Screen Select Signal		To Graphic Screen
Lift Control Screen Select Sig.		To Graphic Screen
Velocity Cont. Screen Sel. Sig.		To Graphic Screen
Altitude Cont. Screen Sel. Sig.		To Graphic Screen
Display Stow Signal		To Graphic Screen
Airplane Pos. Vert. Adj.		To Graphic Screen
Airplane Pos. Horiz. Adj.		To Graphic Screen
Boost Screen Switching Signal		From INS #1/INS #2
Lift Cont. Screen Switch. Sig.		From INS #1/INS #2
Velocity Cont. Screen Sw. Sig.		From INS #1/INS #2
2. Flight Control System		
Attitude Indicator/Flight Director	8	
Attitude Pitch		From INS #1/INS #2
Attitude Roll		From INS #1/INS #2
Attitude Yaw		From INS #1/INS #2
Attitude Error Pitch		From INS #1/INS #2/ALS
Attitude Error Roll		From INS #1/INS #2/ALS
Attitude Error Yaw		From INS #1/INS #2
Pitch Rate		From Instrument Rate Gyro Package
Roll Rate		From Instrument Rate Gyro Package
Yaw Rate		From Instrument Rate Gyro Package
Side Slip		From Airplane Motion
Angle of Attack		From INS #1/INS #2/Air Data
Off Flags		From Electrical Power
Altitude, Descent Rate, and Lift Ind.	9	
Geometric Altitude		From INS #1/INS #2
Pressure Altitude		From Air Data
Descent Rate		From INS #1/INS #2/Air Data
Lift, Error		From INS #1/INS #2

TABLE X-1(Cont)

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
Lift Error High Limit		From INS #1/INS #2
Lift Error Low Limit		From INS #1/INS #2
Injection Altitude		From INS #1/INS #2
Velocity and Range Indicator	7	
Velocity, Actual Course		From INS #1/INS #2/Air Data
Velocity, Actual Fine		From INS #1/INS #2/Air Data
Velocity, Programmed		From INS #1/INS #2
Range-To-Go, Actual		From INS #1/INS #2
Maximum Available Range		From INS #1/INS #2
Minimum Available Range		From INS #1/INS #2
Injection Velocity		From INS #1/INS #2
Acceleration Indicator	10	
Forward-Aft G's		From Airplane/Booster Motion
Up-Down G's		From Airplane/Booster Motion
Left-Right G's		From Airplane/Booster Motion
Up-Down G Limit Boost		To Warn Panel G Limit Light
Up-Down G Limit Glide		To Warn Panel G Limit Light
Left-Right G Limit Boost		To Warn Panel G Limit Light
Left-Right G Limit Glide		To Warn Panel G Limit Light
Surface Temperature Indicator	6	
Max. Lead Edge Temp, Actual		From Thermocouple Leading Edge
Max. Lead Edge Temp. Predicted		From Temp. Electronics Package
Max. Nose Temperature Actual		From Thermocouple - Nose
Max. Nose Temperature Predicted		From Temp. Electronics Package
Max. Lower Surface Temp. Actual		From Thermocouple - Lower Surface
Max. Lower Surface Temp. Predicted		From Temp. Electronics Package
Focus Control		To Temperature Electronics Package
Position Control		To Temperature Electronics Package
Lift Control Graphic Screen	18	
Range-To-Go, Actual		From INS #1/INS #2
Altitude, Actual		From INS #1/INS #2
Screen Switching Signal		From INS #1/INS #2

TABLE X-1 (Cont)

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
Velocity Control Graphic Screen	18	
Velocity, Actual		From INS #1/INS #2
Range-To-Go, Actual		From INS #1/INS #2
Screen Switching Signal		From INS #1/INS #2
Attitude Indicator Backup and Landing	15	
Pitch Attitude		From INS #1/INS #2
Roll Attitude		From INS #1/INS #2
Angle of Attack		From INS #1/INS #2/Air Data
Off Flag		From Electrical Power
Altitude Indicator Landing	16	
Pressure Altitude		From Air Data
Pressure Altitude Set Cont.		(Altimeter Mechanical Link)
Air Speed Indicator, Landing	14	
Airspeed		From Air Data
Rate of Climb Indicator	17	
Rate of Climb		From Air Data
Flight Cont.Sys.Cont.Panel	25	
Engage GCM Signal		To Flight Control System (FCS)
Engage PCM Signal		To FCS
Engage PFM Signal		To FCS
Engage ALS Signal		To Auto Landing System (ALS)
GCM Engaged Signal		From FCS
PCM Engaged Signal		From FCS
PFM Engaged Signal		From FCS
ALS Engaged Signal		From ALS
Backup Mode Engaged Signal		From FCS
PCM Synchronized Signal		From FCS
Hyd. System #1 Select		To Hydraulic System #1
Hyd. System #2 Select		To Hydraulic System #2
Hyd. System #1 Pressure		From Hydraulic System #1
Hyd. System #2 Pressure		From Hydraulic System #2
Servo Actuator Malf. Signal		From FCS
Yaw Trim Control		To FCS

TABLE X-1(Cont)

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
Side Stick Controller	62	
Engage Backup Mode		To FCS/Booster Control System
Pitch Trim Control		To FCS
Roll Trim Control		To FCS
Pitch Steering Signal		To FCS/Booster Control System
Roll Steering Signal		To FCS/Booster Control System
Yaw Steering Signal		To FCS/Booster Control System
Pitch Backup Signal		To FCS/Booster Control System
Roll Backup Signal		To FCS/Booster Control System
Yaw Backup Signal		To FCS/Booster Control System
Pitch Reaction Signal (Backup)		To Pitch Reaction Jets
Roll Reaction Signal (Backup)		To Roll Reaction Jets
Yaw Reaction Signal (Backup)		To Yaw Reaction Jets
Buzzer High "G" Signal		From FCS
Surface Position Indicators	38, 39	
R.H. Elevon Position		From FCS
L.H. Elevon Position		From FCS
R.H. Rudder Position		From FCS
L.H. Rudder Position		From FCS
3. Guidance System		
Computer Programmer Panel	52	
INS #1 On Signal		To INS #1 Switching Unit
INS #2 On Signal		To INS #2 Switching Unit
INS #1 Display On Signal		To Computer Prog. Digital Display
INS #2 Display On Signal		To Computer Prog. Digital Display
Radar Indication		From Data Link
On Course Digital Ind. Sig.		From Data Link/INS #1/INS #2
Cross Course Digital Ind. Sig.		From Data Link/INS #1/INS #2
Altitude Digital Ind. Sig.		From Data Link/INS #1/INS #2
INS #1 Error Insert Control		To INS #1
INS #1 On Course Error Dig. Ind.		From Data Link/INS #2
INS #1 Cross Course Error Dig. Ind.		From Data Link/INS #2
INS #1 Altitude Error Dig. Ind.		From Data Link/INS #2
INS #2 Error Insert Cont. Sig.		To INS #2
INS #2 On Course Error Dig. Ind.		From Data Link/INS #1

TABLE X-1 (Cont)

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
INS #2 Cross Course Error Dig. Ind.		From Data Link/INS #1
INS #2 Altitude Error Dig. Ind.		From Data Link/INS #1
Destination Insert #1 Signal		To INS #1/INS #2
Destination Insert #2 Signal		To INS #1/INS #2
Destination Insert #3 Signal		To INS #1/INS #2
Destination Insert #4 Signal		To INS #1/INS #2
Destination Insert #5 Signal		To INS #1/INS #2
Destination Insert #6 Signal		To INS #1/INS #2
Destination Insert #7 Signal		To INS #1/INS #2
Destination Insert #8 Signal		To INS #1/INS #2
Destination Not Avail. Ind.		From INS #1/INS #2
Cross-Range Error Indicator	13	
Cross-Range Error Signal		From INS #1/INS #2
Map Display	23	
On Course Position (X-Axis) Sig.		From INS #1/INS #2
Cross Course Pos. (Y-Axis) Sig.		From INS #1/INS #2
Bearing Angle Signal		From INS #1/INS #2
(Angle Between Velocity Vector and Ref. Great Circle)		
Escape Capsule Range Signal		From INS #1/INS #2
Map Display Control Panel	50	
Track Control Signal		To Map Display
Index Control Signal		To Map Display
Slides Control Signal		To Map Display
Map Control Signal		To Map Display
Slew Control Signal		To Map Display
Brightness Control Signal		To Map Display
Map Display Frame Select Panel	50	
Frame Select Digital		From Map Display (Frame Sel. Cont.)
Frame Select Control		To Map Display
Time Indicator	12	
Elapsed Time/Time To Go Initial Start		From Ground Control (Umbilical)

TABLE X-1 (Cont)

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
Elapsed Time		Time Pulses From INS #1/INS #2
Greenwhich Mean Time		Time Pulses From INS#1/INS #2
Time To Go		From INS #1/INS #2
4. Airplane Propulsion and Power		
Generating System Power Cont. Panel	29	
PGU No. 1 Off 1 Run Signal		To PGU No. 1 Controls
PGU No. 1 Run Signal		From PGU No. 1 Controls
PGU No. 2 Off/Run Signal		To PGU No. 2 Controls
PGU No. 2 Run Signal		From PGU No. 2 Controls
Generator #1 On/Off/Reset Sig.		To Generator #1 Controls
Generator #1 On Load Signal		From Generator #1 Controls
Generator #1 Standby Signal		From Generator #1 Controls
Generator #2 On/Off/Reset Sig.		To Generator #1 Controls
Generator #2 On Load Signal		From Generator #2 Controls
Generator #2 Standby Signal		From Generator #2 Controls
Reaction Control Arm-Off Sig.		To Reaction Jets
Range Control Arm-Off Sig.		To Range (Velocity) Engine
PGU Arm-Off Signal		To PGU Arm Valve
H_2O_2 Press/Vent Signal		To H_2O_2 Tank Controls
H_2O_2 Jettison Signal		To H_2O_2 Tank Controls
N_2H_4 Press/Vent Signal		To N_2H_4 Tank Controls
N_2H_4 Jettison Signal		To N_2H_4 Tank Controls
Helium and PGU Quantity Ind.	31	
Helium Main Pressure Sig.		From Helium Tank
Helium Regulator Press. Sig.		From Helium Pressure Regulator
PGU Fuel Quantity Signal		From N_2H_4 Tank
Landing Engine Indicator	22	
Fuel Quantity Signal		From JP6 Fuel Tank
Engine Gas Temperature Sig.		From Landing Engine
RPM Tachometer Signal		From Landing Engine
Jet Engine Throttle	44/53	
Fuel Off/On Signal		To Landing Engine Fuel Controls
Start & Idle to Max. Speed Sig.		To Landing Engine Fuel Controls
Intake Extend Signal		To Landing Engine Air Scoop Cont.
In Air Start Signal		To Landing Engine Spark Ignition

TABLE X-1 (Cont)

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
Range-Velocity Control	54/55	
Range-Velocity Inc. Sig.		To PGU Exhaust Diverter Controls
Range-Velocity Dec. Sig.		To Range (Velocity) Engine
5. Failure Warning & Emergency System		
Central Warning Indicators	1/2/4	
Fire Signal		From Fire Detectors
Master Caution Signal		From Caution Panel
Eject Signal		From Failure Analysis CKT or Data Link
Warning Lights Panel	21	
Hydraulic System-Off Sig.		From Associated System Sensor
Electrical System-Off Sig.		From Associated System Sensor
H_2O_2 Hot Signal		From Associated System Sensor
Cockpit Leakage Signal		From Associated System Sensor
Excess Pitch Signal		From Attitude Ind.Flight Director
Pitch G-High Signal		From Accelerometer
Yaw G-High Signal		From Accelerometer
Caution Lights Panel	24	
PGU #1 - Off Signal		From Associated System Sensor
PGU #2 - Off Signal		From Associated System Sensor
Gen. #1 - Off Signal		From Associated System Sensor
Gen. #2 - Off Signal		From Associated System Sensor
Hydraulic Press. #1 Off Sig.		From Associated System Sensor
Hydraulic Press. #2 Off Sig.		From Associated System Sensor
Primary INS - Off Signal		From Associated System Sensor
Secondary INS - Off Signal		From Associated System Sensor
PGU Fuel - Low Signal		From Associated System Sensor
H_2O_2 - Low Signal		From Associated System Sensor
Jet Fuel - Low Signal		From Associated System Sensor
Struct. Temp. - High Sig.		From Associated System Sensor
Struct. Coolant Water - Low Sig.		From Associated System Sensor
Struct. Coolant Water Press.-Off Sig.		From Associated System Sensor
Liquid O_2-N_2-Low Signal		From Associated System Sensor
Cockpit O_2 P.P. -Low Signal		From Associated System Sensor
Cockpit CO_2 P.P.-High Signal		From Associated System Sensor
Equipment Compartment Overheat Sig.		From Associated System Sensor

TABLE X-1 (Cont)

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
Equip. Compartment Press. -Low Sig.		From Associated System Sensor
Flight Control System -Off Sig.		From Associated System Sensor
Control Surface Servo Act-Malf Sig.		From Associated System Sensor
Automatic Landing System Off Sig.		From Associated System Sensor
Airplane Escape Now Possible Sig.		From INS #1/INS #2
Warning Light Test Signal		To All Warning and Caution Lights
Circuit Breaker Panel	67	
Primary INS A.C. Power		To Assoc. CKT Breaker Protected Equip.
Primary INS D.C. Power		To Assoc. CKT Breaker Protected Equip.
Secondary INS A.C. Power		To Assoc. CKT Breaker Protected Equip.
Secondary INS D.C. Power		To Assoc. CKT Breaker Protected Equip.
Communication Power		To Assoc. CKT Breaker Protected Equip.
Telemetering Power		To Assoc. CKT Breaker Protected Equip.
FCS A.C. No. 1 Power		To Assoc. CKT Breaker Protected Equip.
FCS A.C. No. 2 Power		To Assoc. CKT Breaker Protected Equip.
FCS D.C. Power		To Assoc. CKT Breaker Protected Equip.
Recorder Power		To Assoc. CKT Breaker Protected Equip.
Camera Power		To Assoc. CKT Breaker Protected Equip.
Radar Power		To Assoc. CKT Breaker Protected Equip.
Map Display Power		To Assoc. CKT Breaker Protected Equip.
Velocity Indicator Power		To Assoc. CKT Breaker Protected Equip.
Attitude Indicator Power		To Assoc. CKT Breaker Protected Equip.
Surface Temp. Indicator Power		To Assoc. CKT Breaker Protected Equip.

TABLE X-1 (Cont)

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
PGU No. 1 Power		To Assoc. CKT Breaker Protected Equip.
PGU No. 2 Power		To Assoc. CKT Breaker Protected Equip.
D.C. Power Supply #1 Output		To Assoc. CKT Breaker Protected Equip.
D.C. Power Supply #2 Output		To Assoc. CKT Breaker Protected Equip.
Air Conditioning Equip. Power		To Assoc. CKT Breaker Protected Equip.
Fire Extinguishers Power		To Assoc. CKT Breaker Protected Equip.
Engine Ignition Power		To Assoc. CKT Breaker Protected Equip.
Engine Scoop Power		To Assoc. CKT Breaker Protected Equip.
Instrument Lights Power		To Assoc. CKT Breaker Protected Equip.
Console Lights Power		To Assoc. CKT Breaker Protected Equip.
Cockpit Lights Power		To Assoc. CKT Breaker Protected Equip.
Warning Lights Power		To Assoc. CKT Breaker Protected Equip.
Landing Gear Down Power		To Assoc. CKT Breaker Protected Equip.
Fire Extinguisher Panel	36	
Fire Detector Engine Compt. Sig.		From Fire Detector
Fire Ext. Arm Engine Compt. Sig.		To Fire Extinguisher
Fire Ext. Discharge Eng.Com. Sig.		To Fire Extinguisher
Fire Detector Equip. Compt. Sig.		From Fire Detector
Fire Ext. Arm Equip.Compt. Sig.		To Fire Extinguisher
Fire Ext. Disc. Equip. Compt.Sig.		To Fire Extinguisher
Fire Detector L.H. Wing Sig.		From Fire Detector
Fire Ext. Arm L.H. Wing Sig.		To Fire Extinguisher
Fire Ext. Disc. L.H. Wing Sig.		To Fire Extinguisher
Fire Detector R.H. Wing Sig.		From Fire Detector
Fire Ext. Arm R.H. Wing Sig.		To Fire Extinguisher
Fire Ext. Disc. R.H. Wing Sig.		To Fire Extinguisher

TABLE X-1 (Cont)

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
Dual Escape System Actuate Controls	45	
Escape System Actuate Signals		To Sept.Mach./Capsule Escape Rockets
6. Miscellaneous Systems		
Cockpit Environment Indicator	27	
Oxygen Concentration Signal		From Oxygen Sensor
Carbon Dioxide Conc. Signal		From Carbon Dioxide Sensor
Cockpit Air Temperature Sig.		From Temperature Sensor
Cockpit Pressure Signal		From Cockpit Pressure Sensor
Liquid O_2-N_2 Remaining Sig.		From Liquid O_2-N_2 Tank
Emergency O_2 Remaining Sig.		From Emergency O_2 Tank
Cockpit Environment Control Panel	28	
Cockpit Temp.Auto/Off/Man. Sig.		To Air Conditioning Equipment
Temperature Adjust		To Air Conditioning Equipment
Cockpit Pressure O_2-N_2 Norm/Emerg.		To Air Conditioning Equipment
Cockpit Pressure On/Off		To Air Conditioning Equipment
Cockpit Pressure Dump		To Pressure Dump Valve
Nose Cap Separation Handle	35	
Nose Cap Separation Signal		To Nose Cap Jettison Mechanism
Landing Gear Control	33	
Landing Gear Down Signal		To Hyd. Lock Pin Removal Mechanism
Landing Gear Emerg.Down Sig.		To Ballistic Lock Pin Removal Mech.
Warning Horn Silence Switch		To Horn
Landing Gear Indicator	32	
Nose Wheel Down Signal		From Nose Wheel Limit Circuit
Left Skid Down Signal		From Left Skid Limit Circuit
Right Skid Down Signal		From Right Skid Limit Circuit
Communication Panel	56	
Power On/Off Control Signal		To Communication Equipment
Voice-Video Trans. Signal		To Voice-Video Transmitter
Telem. Trans. Signal		To Telem. Trans.

TABLE X-1 (Cont)

Instrument and System	Instrument No. (Figure III-5)	Display Input or Control Output
UHF-CW Trans. Signal		To UHF Transmitter
UHF Voice Trans. Signal		To UHF Transmitter
Volume Control Signal		To #1, #2, and UHF Receivers
Headphone Select Rec. #1 Sig.		To Receiver #1
Headphone Select Rec. #2 Sig.		To Receiver #2
Headphone Select UHF Rec. Sig.		To UHF Receiver
Telemetering Panel	37	
Telemetering On/Off Signal		To Telemetering Trans.
Recorder On/Off Signal		To Recorder
Record Indicate Signal		From Recorder
Run Out Indicate Signal		From Recorder
Camera On/Off Signal		To Cameras
Cockpit/Warning Light Cont. Panel	6	
Cockpit Lights Off/Dim/Bright Sig.		To Cockpit Lights
Warning Lights Dim/Bright Sig.		To Warning Lights
Flight Inst. Light Cont. Panel		
Flight Inst. Off/Dim/Bright Sig.		To Instrument Lights
Console Off/Dim/Bright Sig.		To Console Lights
Capsule Parachute Override Handle (Mechanical)	64	
Capsule Parachute Release Handle (Mechanical)	Not Shown	
Glide Escape Handle	57	
Glide Escape Signal		To Ballistic Thrusters
Hatch Release Handle (Mechanical)	26	

XI. REFERENCES

1. Brown, J. S,, Slater-Hammel, A. T. and Bilodeau, E. A. Characteristics of discrete movements in the horizontal plane when executed with one and with two hands. ONR, Contract N-5-ori-57 (Project 2, Report 5) Aug. 1948. (One hand faster than two, and just as accurate).

2. Craig, D. R. and Ellson, D. G. A comparison of two-handed and several one-handed control techniques and tracking tasks. WADC Memo Report MCREXD-694-2L, July 1948.
(A side-controller is better than a center controller even when operated with the non-preferred hand).

3. Fitts, P. M. Psychological research on equipment design, U. S. Government Printing Office, 1947.
(One hand is as good as two; one hand is better than rudder pedal operation).

4. Grether, W. F. Study of several design factors influencing pilot efficiency in the operation of controls. USAF, AMC, WADC Memo. Report TSE AA-694-9, November 1946.
(Rudder pedals are not as good as manual control).

XI. REFERENCES (Cont)

5. Grether, W. F. Direction of controls in relation to indicator movement in one-dimensional tracking. WADC Memo. Report TSE AA-694-4G, Oct. 1947.
(Reversal of displays leads to a significant loss of efficiency only in rudder pedals, not in stick).

6. Henschke, U. K. and Mauch, H. A. A study of the design for a three-dimensional hand control for aircraft. WADC Memo. Report TSE AA-696-110, May 1947.
(Rudder is worse than a three-dimensional manual control. Even after 15 days of practice the difference was eight times the standard deviation).

7. Johnson, L. V. and Lauer, A. R. A study of the effect of induced manual handicaps on automatic performance in relation to reaction time. J. Appl. Psychol. 1937, 21, 85-93.
(One hand is almost as good as two in tracking).

8. Stillwell, W. H. and Drake, H. NACA Research Memo H58618, Sept. 1958, (CONFIDENTIAL).

9. Woodling, et al. Simulation study of a high-performance aircraft, including the effect on pilot control of high accelerations during exit and re-entry flight. NACA RML 58E08a, July 1958 (CONFIDENTIAL).

10. X-15 Progress Report, July 1958 NACA Research Airplane Committee. (SECRET).

11. Flight Control Report E. R. 10374 Section VIII C. 5, Reaction controls Analog Computer Investigation.

12. Bell Aircraft Report No. 7021-0253-013 Dyna Soar-I Simulation Experiments.

13. Secondary Power System, Report No. ER 10832.

XII. ILLUSTRATIONS

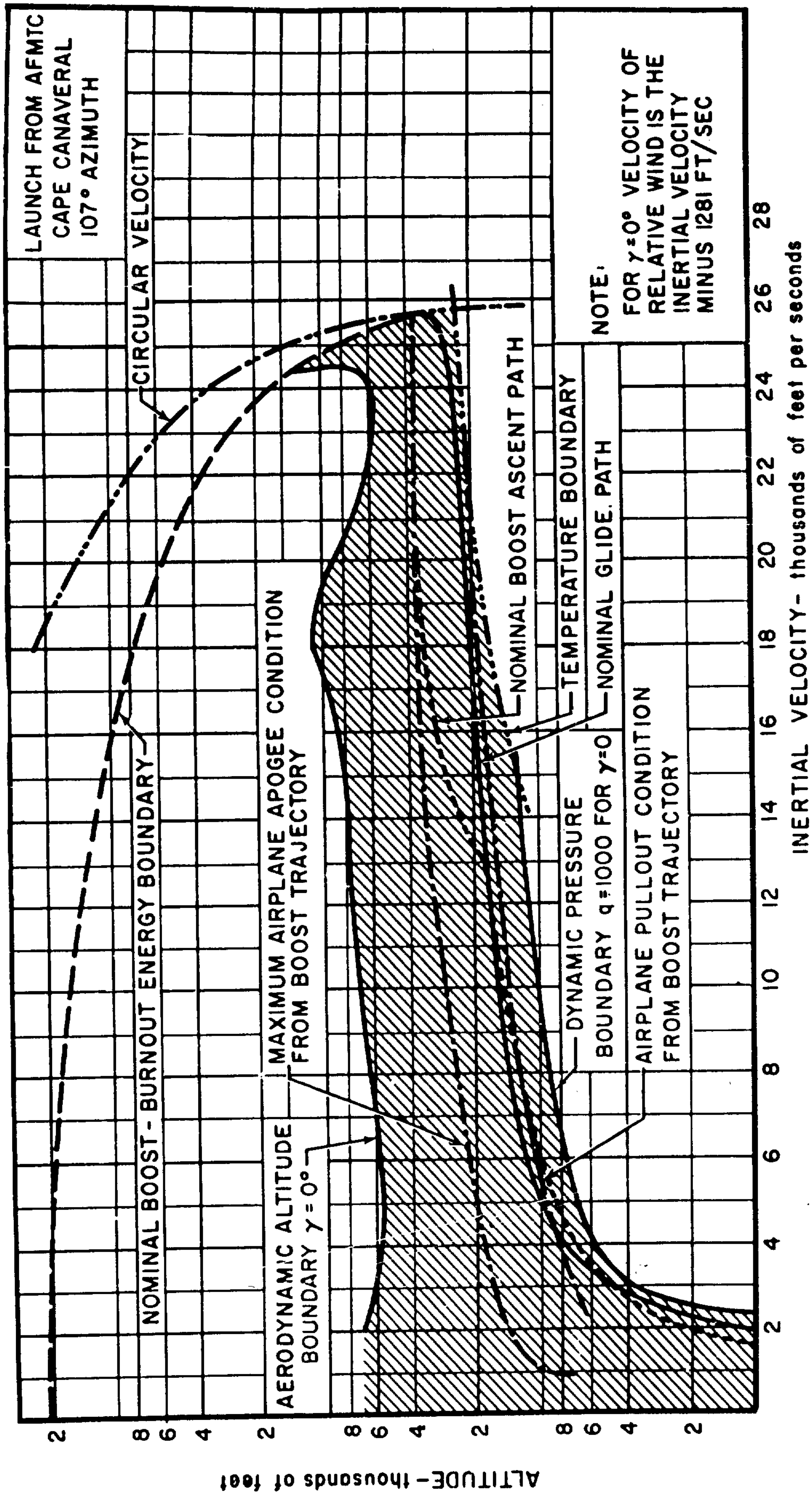


Figure II-1. Boost Flight Envelope

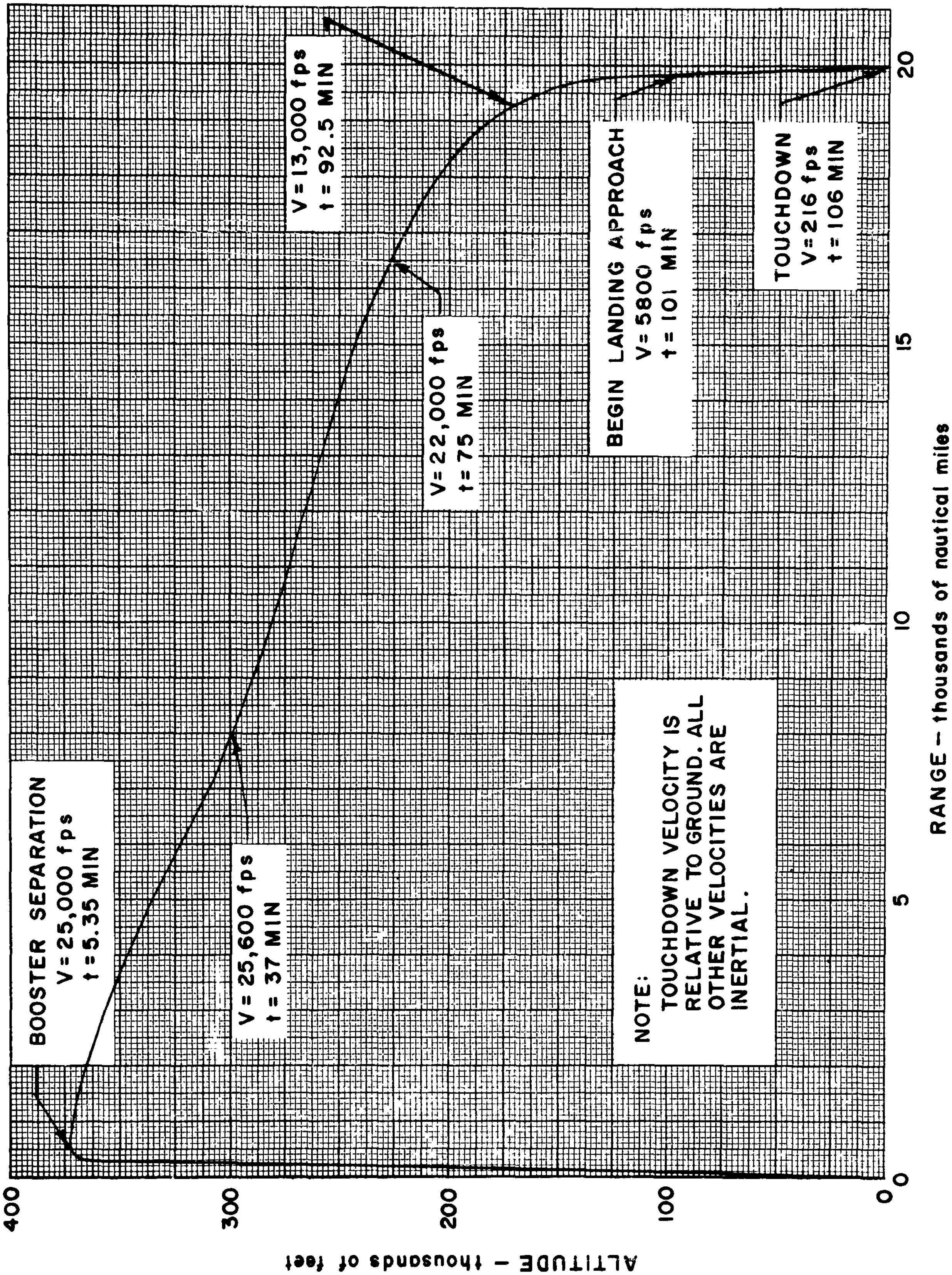


Figure II-2. Nominal Glide Path

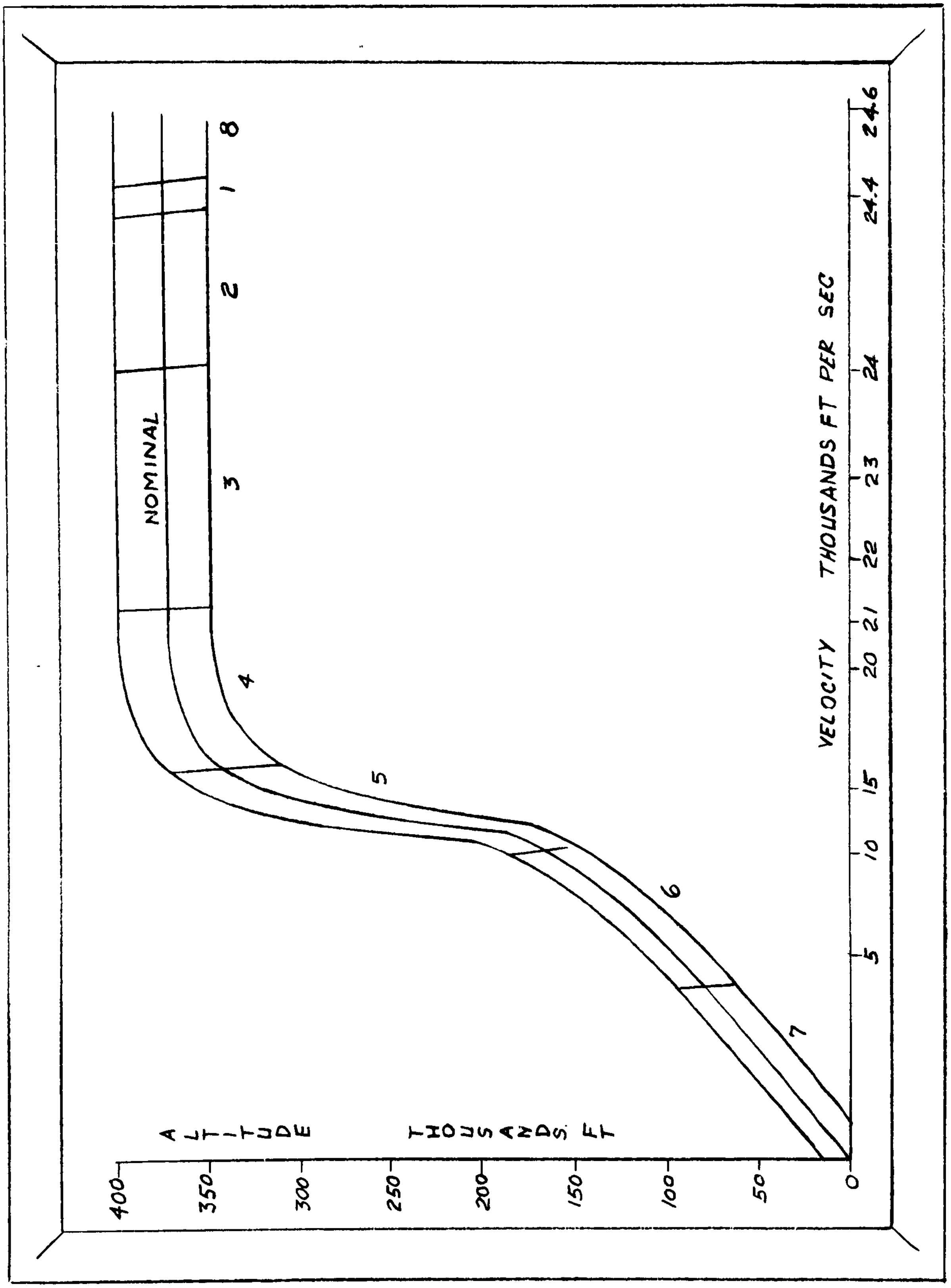
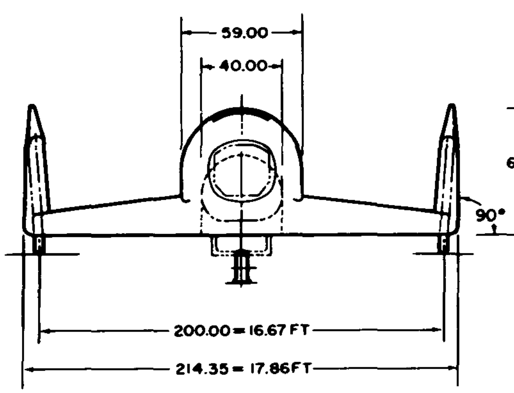


Figure II-3. Boost Zones

GENERAL DATA

AIRCRAFT DATA:		
WING SPAN (NOMINAL)	17.6 FT	
AIRCRAFT LENGTH	26.8 FT	
AIRCRAFT HEIGHT	7.4 FT	
WING:		
AREA-REF	265 SQ FT	
AIRFOIL SECTION	MODIFIED WEDGE	
SWEEPBACK AT L.E.	70°	
SWEEPBACK AT T.E.	25°	
VERTICAL FIN:		
AREA-BOTH FINS (INC. RUDDER) (THEOR.)	47.4 SQ FT	
AIRFOIL SECTION	MODIFIED WEDGE	
SWEEPBACK AT L.E.	50°	
SWEEPBACK AT T.E.	0°	
ELEVON:		
AREA-BOTH (AFT OF HINGE LINE)	30.3 SQ FT	
MOTION	UP 45° DOWN 15°	
RUDDER:		
AREA-BOTH (AFT OF HINGE LINE)	27.6 SQ FT	
MOTION	32° OUTBOARD ONLY	
WEIGHTS AND CGS:		
	CG STA	WT-LB
GROSS WT ON PAD	200.3	9000
SEPARATION WT	199.3	8923
LANDING WT	201.3	7838
EMPTY WT	202.7	7080
USEFUL LOAD	191.7	1920



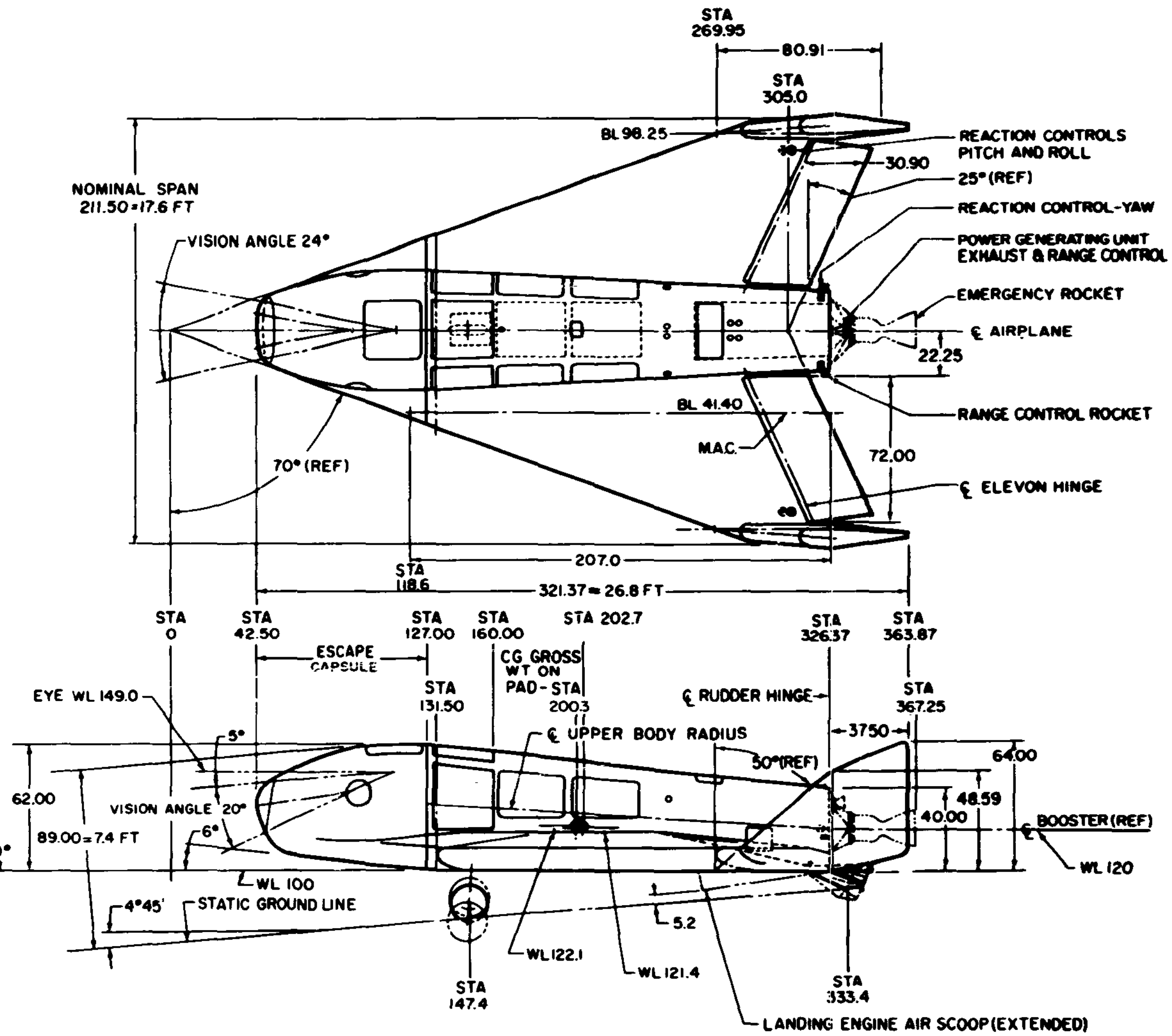


Figure III-1. Dyna-Soar I General Arrangement

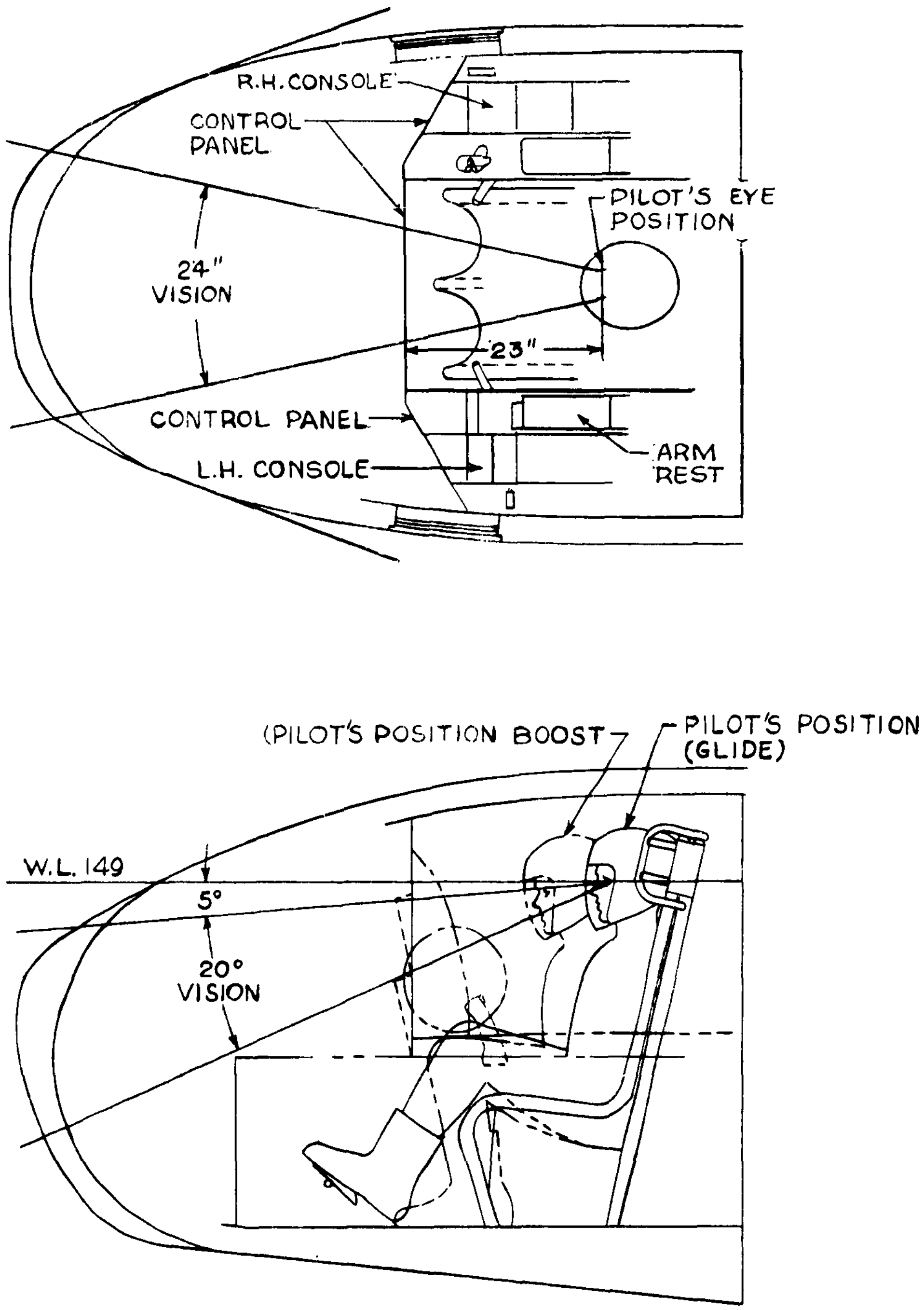
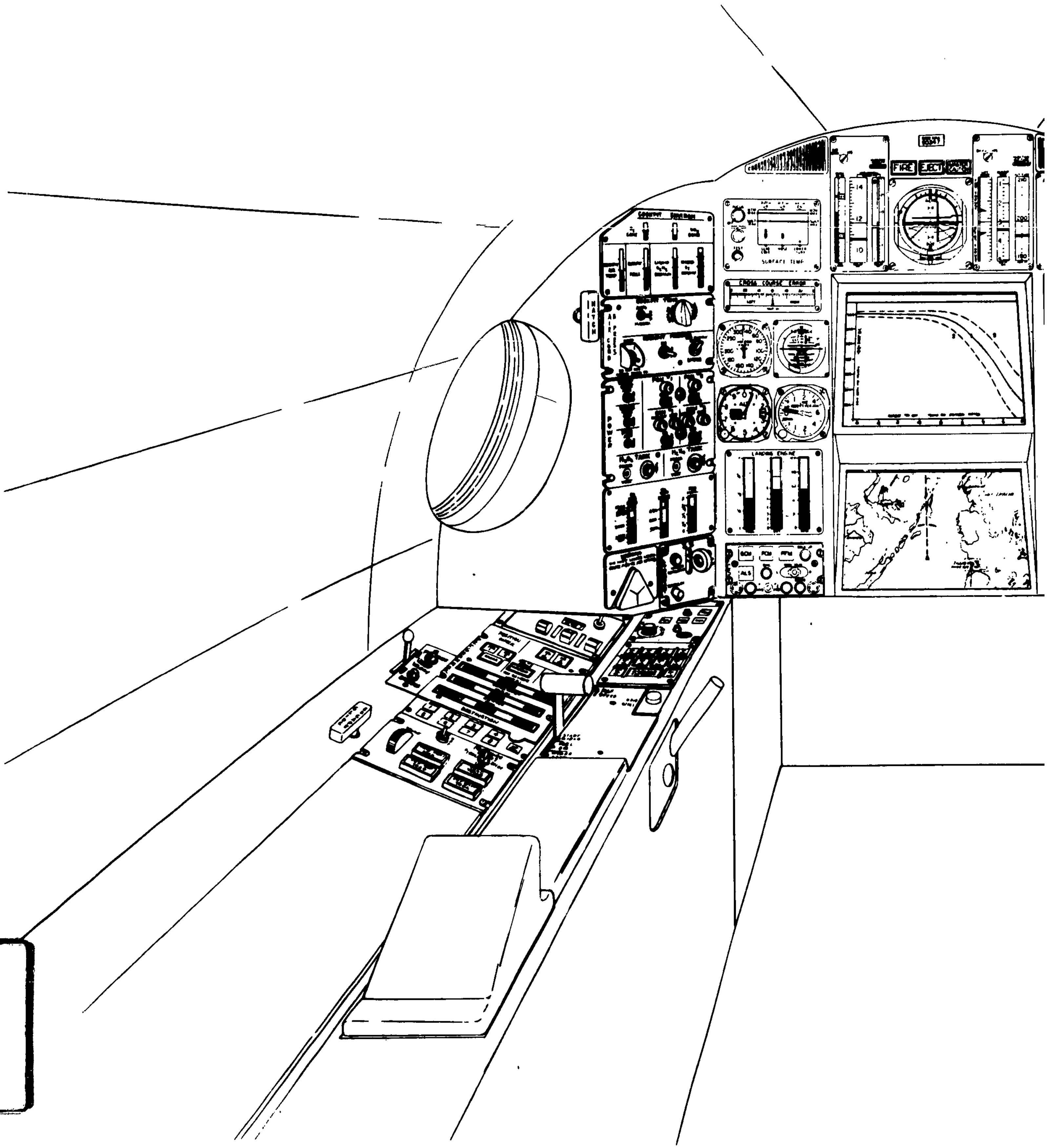


Figure III-2. Cockpit Area


HATCH
FIRE
EJECT
SURFACE TEMP
CROSS COURSE ERROR
LANDING ENGINE
POWER

1

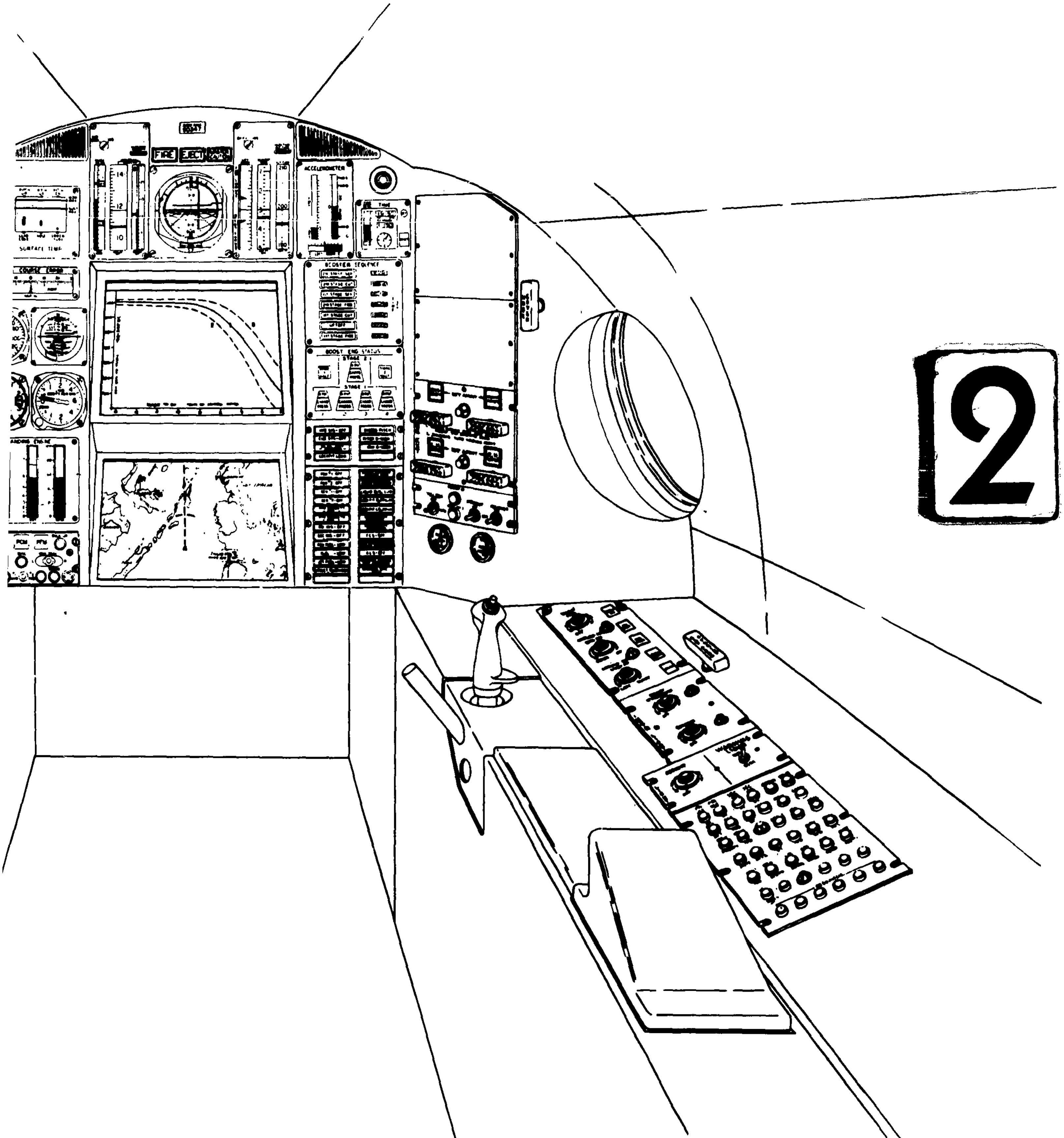
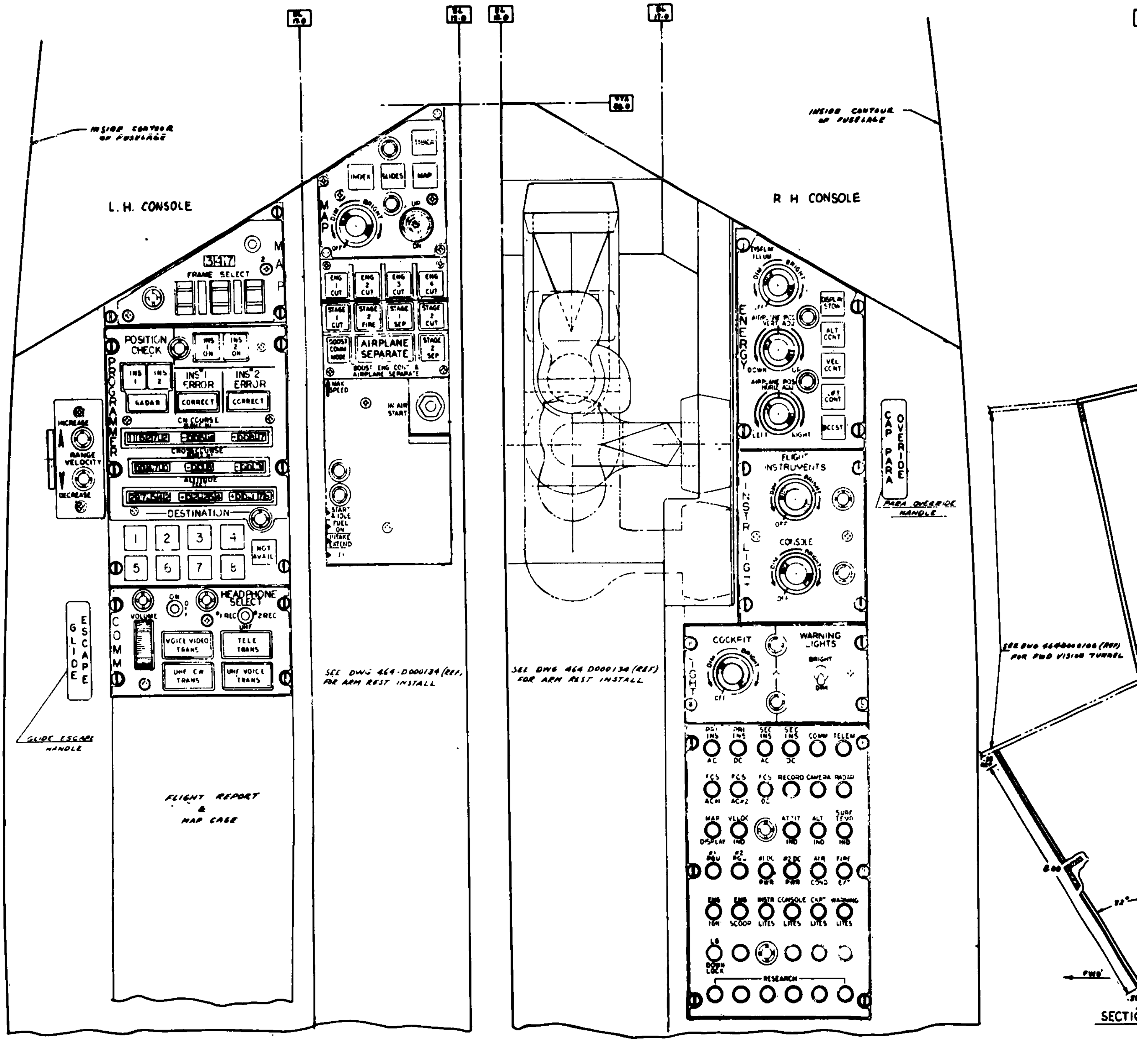
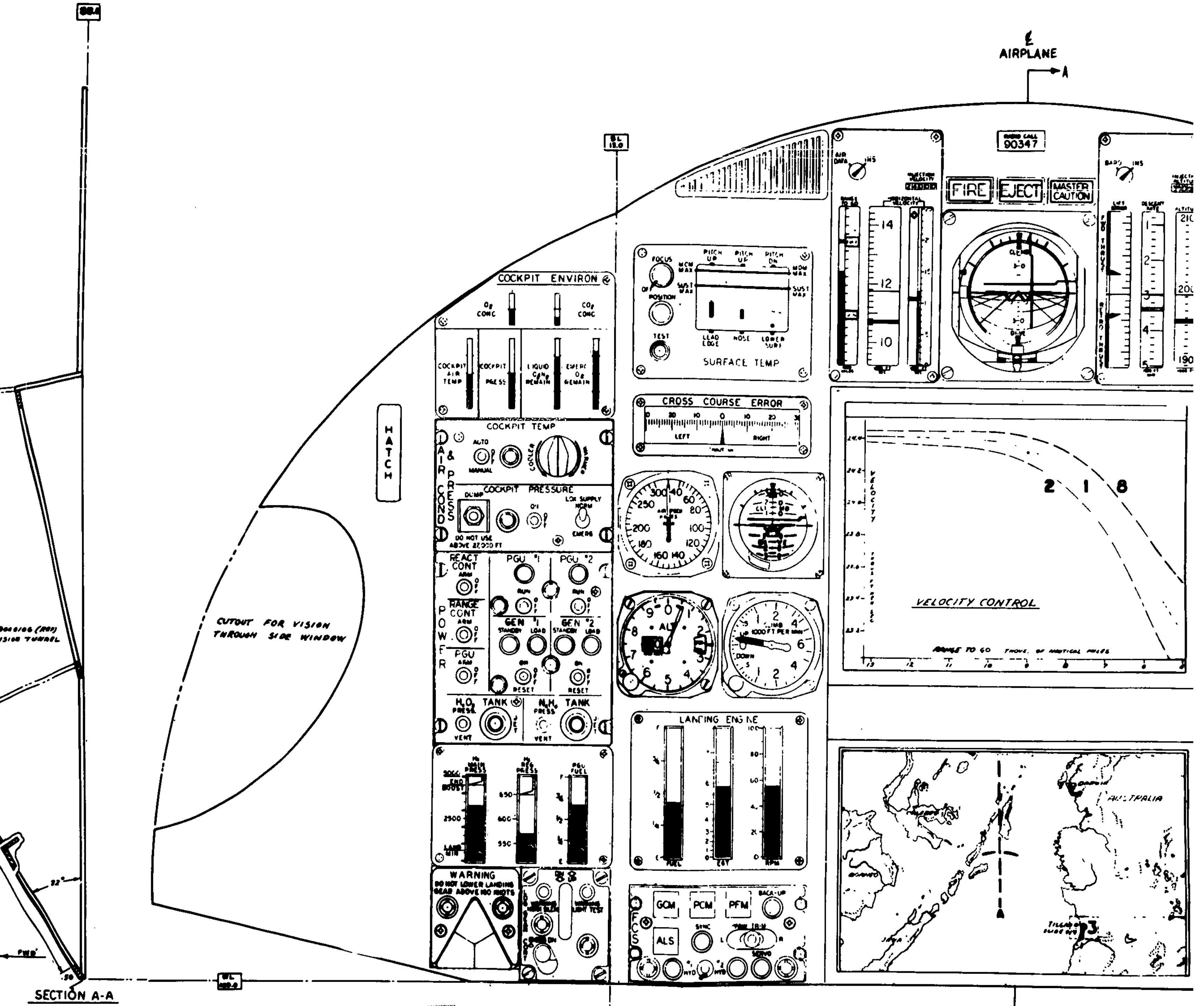


Figure III-3. Display Control Arrangement Looking Forward


INSIDE CONTOUR OF FUSELAGE
L.H. CONSOLE
R H CONSOLE
FRAME SELECT
POSITION CHECK
PROGRAMMER
INS 1 ERROR
INS 2 ERROR
RADAR
CORRECT
CORRECT
ON COURSE
CROSS COURSE
ALTITUDE
DESTINATION
NOT AVAIL
INCREASE
RANGE VELOCITY
DECREASE
GLIDE ESCAPE
GLIDE ESCAPE HANDLE
COMM
HEADPHONE SELECT
VOLUME
VOICE VIDEO TRANS
TELE TRANS
UHF CW TRANS
UHF VOICE TRANS
FLIGHT REPORT & MAP CASE
INDEX
SLIDES
MAP
ENG 1 CUT
ENG 2 CUT
ENG 3 CUT
ENG 4 CUT
STAGE 1 CUT
STAGE 2 FIRE
STAGE 1 SEP
STAGE 2 CUT
AIRPLANE SEPARATE
BOOST ENG CONT & AIRPLANE SEPARATE
MAX SPEED
IN AIR START
SEE DWG 464-D000134 (REF) FOR ARM REST INSTALL
SEE DWG 464-D000134 (REF) FOR ARM REST INSTALL
ENERGY
DISPLAY STOW
ALT CONT
VEL CONT
BOOST
FLIGHT INSTRUMENTS
CONSOLE
INSTR LIGHT
CAP PARA OVERRIDE
PARA OVERRIDE HANDLE
COCKPIT
LIGHT
WARNING LIGHTS
BRIGHT
DIM
PRI INS AC
PRI INS DC
SEC INS AC
SEC INS DC
COMM
TELEM
FCS AC#1
FCS AC#2
FCS DC
RECORD
CAMERA
RADAR
MAP DISPLAY
VEL LOC IND
ATTIT IND
ALT IND
SURF TEMP IND
#1 PGU
#2 PGU
#1 DC PWR
#2 DC PWR
AIR COND
FIRE EXT
ENG IGN
ENG SCOOP
INSTR LITES
CONSOLE LITES
CKPT LITES
WARNING LITES
LG DOWN LOCK
RESEARCH
FOR FWD VISION TUNNEL
FWD
SECTI

1



AIRPLANE
A
HATCH
CUTOUT FOR VISION THROUGH SIDE WINDOW
SECTION A-A
COCKPIT ENVIRON
COCKPIT TEMP
COCKPIT PRESSURE
DO NOT USE ABOVE 82,000 FT
AIR COND & PRESS
REACT CONT
RANGE CONT
PGU
POWER
GEN #1
GEN #2
H_2O TANK
N_2H_4 TANK
WARNING
DO NOT LOWER LANDING GEAR ABOVE 180 KNOTS
SURFACE TEMP
CROSS COURSE ERROR
LEFT
RIGHT
LANDING ENGINE
FCS
GCM
PCM
PFM
ALS
SYNC
BACK-UP
90347
FIRE
EJECT
MASTER CAUTION
VELOCITY CONTROL
RANGE TO GO
2 1 8
AUSTRALIA

2

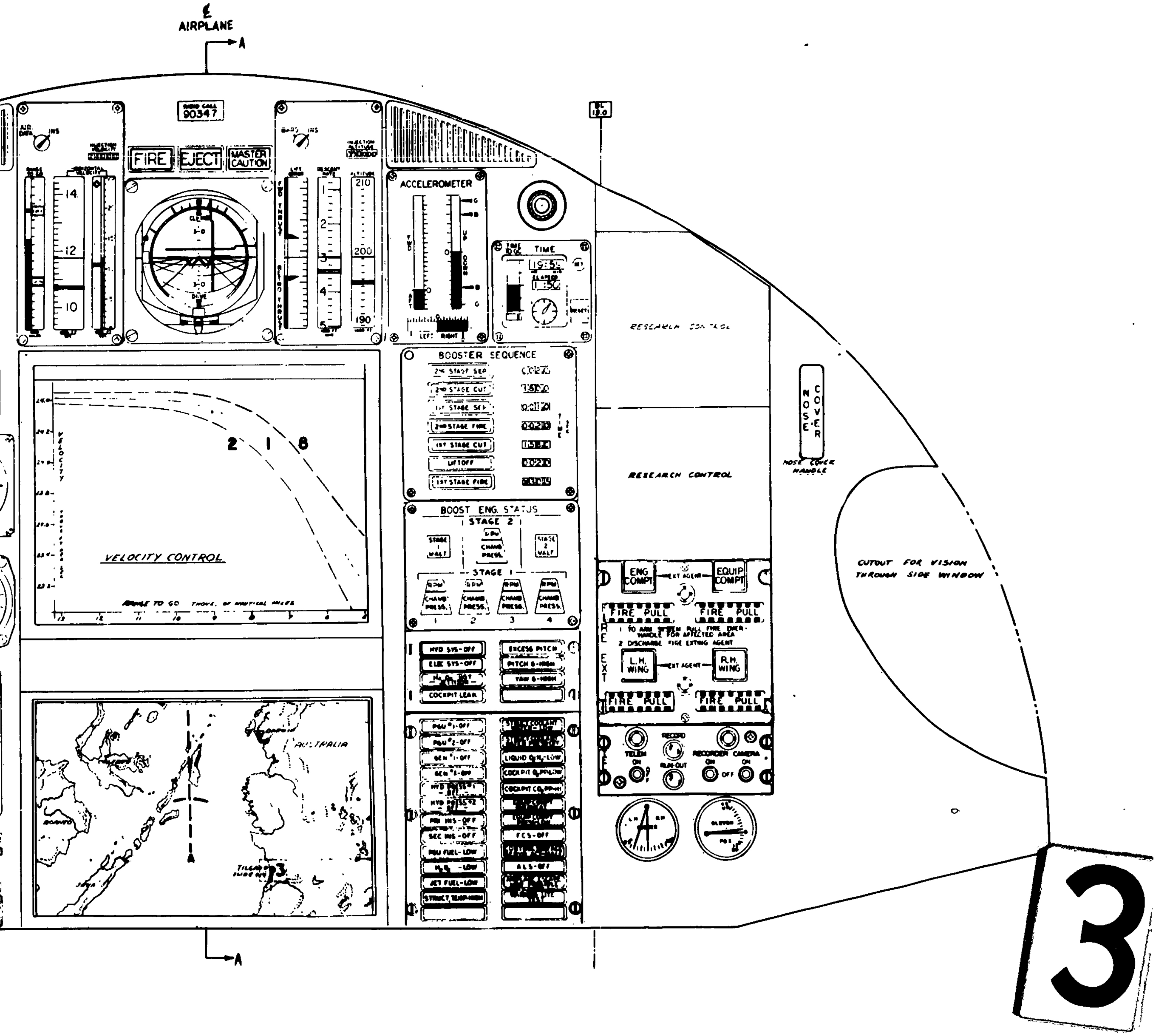
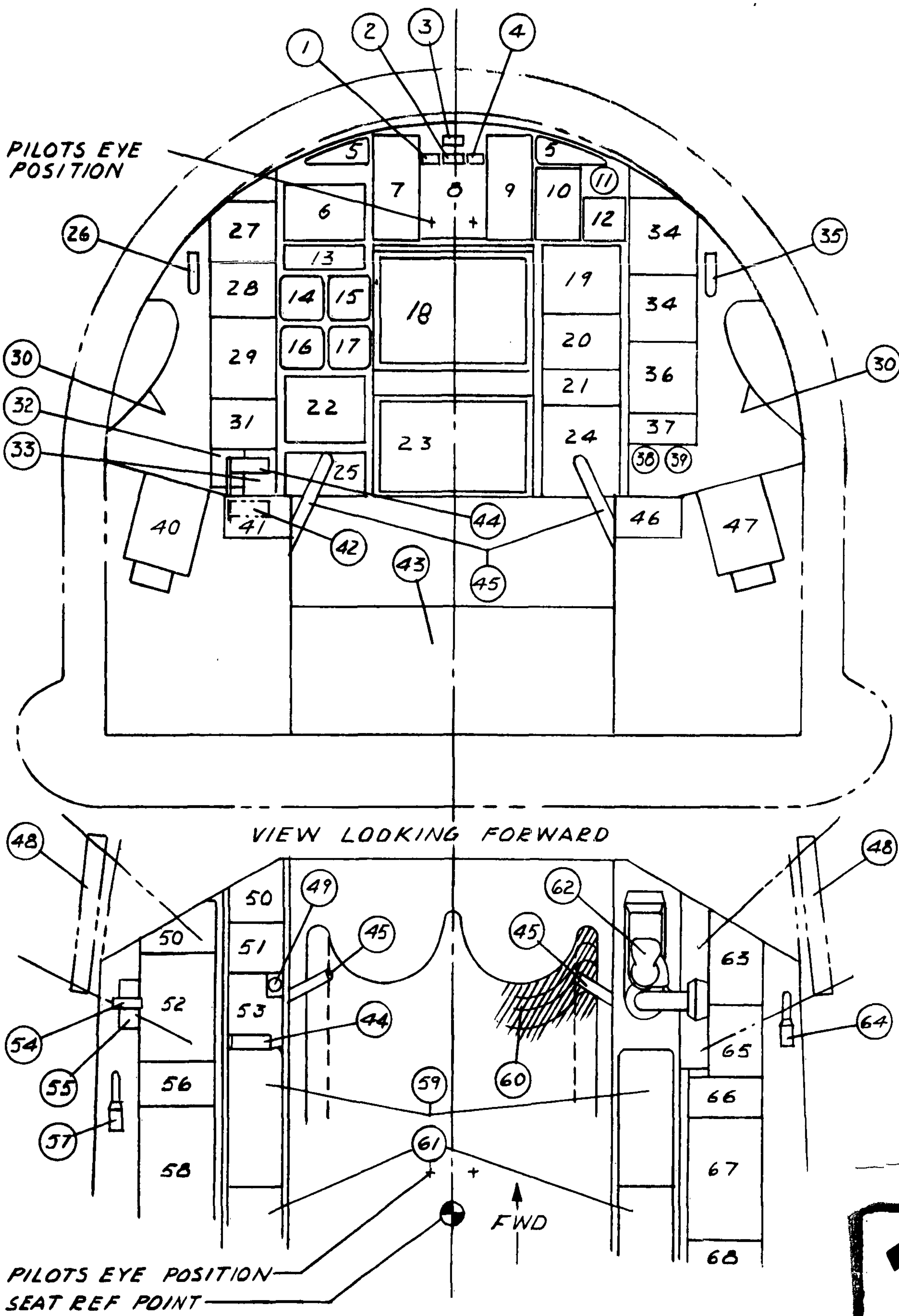


Figure III-4. Display and Control Panel Arrangement



1. "Fire" Warnir
2. "Eject" Warni
3. Radio Call Pla
4. "Master Cauti
5. Cockpit Air Du
6. Surface Temp
7. Velocity and F
8. Attitude and F
9. Altitude, Desc
   Lift Error In
10. Acceleromete
11. Cockpit Air No
12. Clock
13. Cross Course
14. Airspeed Indic
15. Attitude Indica
    Landing)
16. Altimeter (Lan
17. Rate of Climb
18. Instrument Pa
    Forward Lan
    Graphic Scre
19. Booster Seque
20. Booster Status
21. Warning Light
22. Landing Engin
23. Map Display
24. Caution Indica
25. Flight Control
    Panel
26. Hatch Control
27. Cockpit Enviro
28. Cockpit Enviro
29. Power Control
30. Side Panel Cut
    Through Side
31. Helium Pressu
    Quantity Indic
32. Landing Gear
33. Landing Gear
34. Research Cont
35. Nose Cover Ha
36. Fire Extinguis

1

1. "Fire" Warning Light
2. "Eject" Warning Light
3. Radio Call Plate
4. "Master Caution" Warning Light
5. Cockpit Air Duct (2)
6. Surface Temperature Indicator
7. Velocity and Range Indicator
8. Attitude and Flight Director
9. Altitude, Descent Rate and Lift Error Indicator
10. Accelerometer
11. Cockpit Air Nozzle
12. Clock
13. Cross Course Error Indicator
14. Airspeed Indicator (Landing)
15. Attitude Indicator (Backup and Landing)
16. Altimeter (Landing)
17. Rate of Climb Indicator (Landing)
18. Instrument Panel Aperature for Forward Landing Vision and Graphic Screen Displays
19. Booster Sequence Indicator
20. Booster Status Indicator
21. Warning Light Panel
22. Landing Engine Indicator
23. Map Display
24. Caution Indicator Panel
25. Flight Control System Control Panel
26. Hatch Control Handle
27. Cockpit Environment Indicator
28. Cockpit Environment Control Panel
29. Power Control Panel
30. Side Panel Cutout (For Vision Through Side Window)
31. Helium Pressure and PGU Quantity Indicator
32. Landing Gear Warning Lights
33. Landing Gear Operating Control
34. Research Control Panel (2)
35. Nose Cover Handle
36. Fire Extinguisher Control Panel
37. Telemetry Control Panel
38. Rudder Surface Position Indicator
39. Elevon Surface Position Indicator
40. L. H. Console Control Panel Installation
41. L. H. Arm Rest
42. Throttle - Jet Engine (Stowed Position)
43. Footrest
44. Throttle - Jet Engine (Operating Position)
45. Escape Capsule Handle and Trigger
46. R. H. Arm Rest
47. R. H. Console Control Panel Installation
48. Side Window (With Angle of Vision Shown)
49. "In Air Start" Button
50. Map Control Panel (2)
51. Boost Engine Control and Airplane Separate Buttons
52. Programmer Control Panel
53. Throttle Nameplate
54. Range Velocity Control
55. Range Velocity Control Nameplate
56. Communication Control Panel
57. Glide Escape Handle - Escape Capsule
58. Flight Report and Map Case
59. Arm Rest Pads
60. Nylon Net Seat
61. Elbow Restraints - Retractable
62. Side Stick Flight Controller - 3-Axis
63. Energy Control Panel
64. Parachute Override Handle - Escape Capsule
65. Instrument and Console Light Control Panel
66. Cockpit Light Control Panel
67. Circuit Breaker Control Panel
68. Data Case

2

Figure III-5. Display and Control Location Numbers

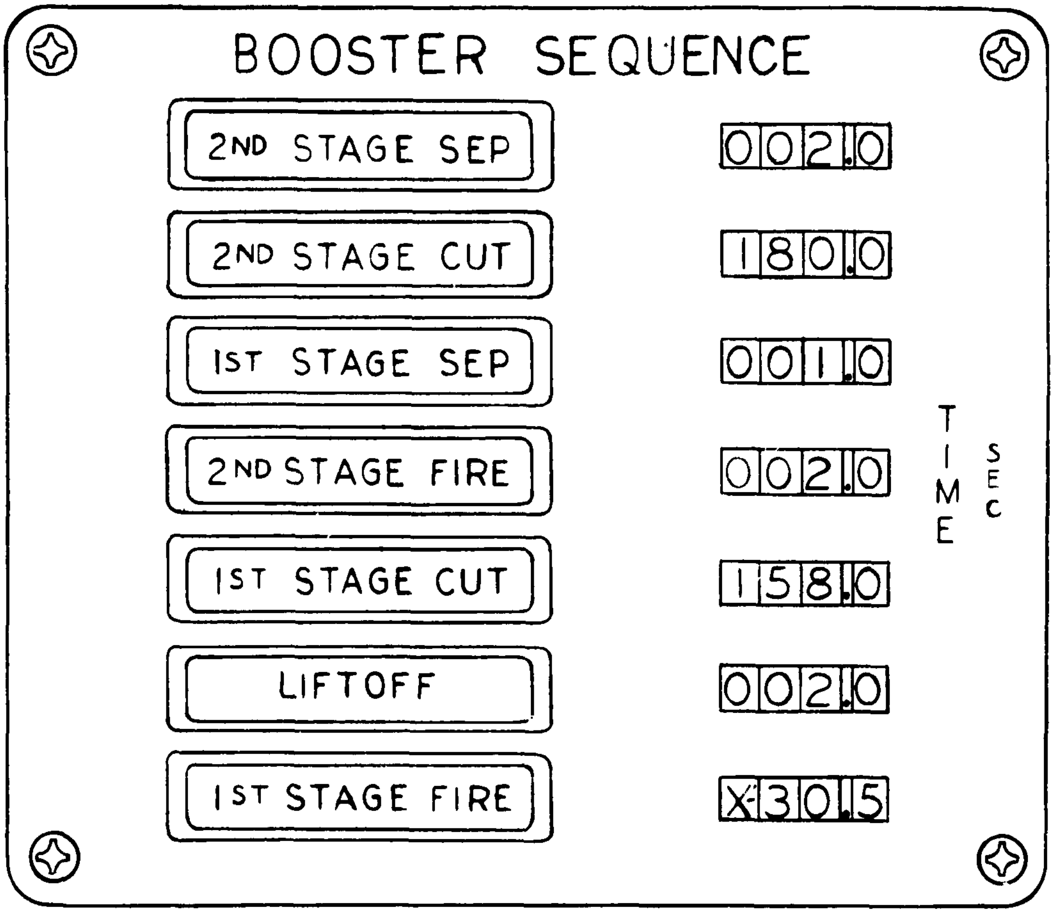


Figure IV-1. Booster Sequence Panel

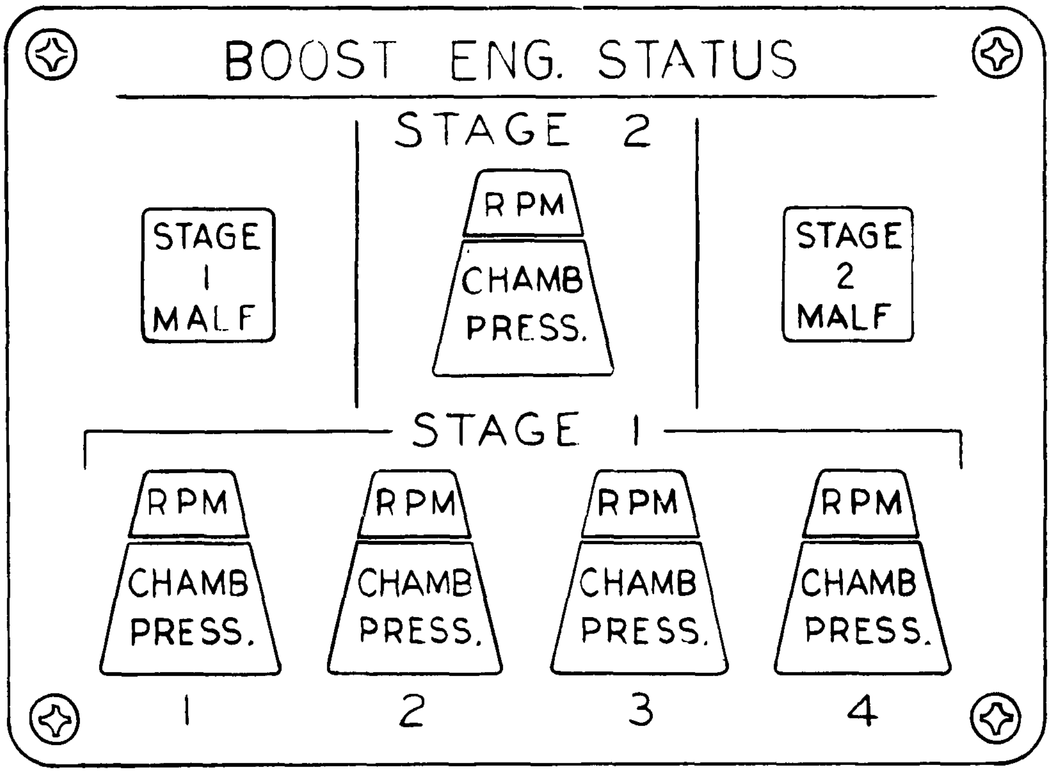


Figure IV-2. Booster Engine Status Indicator

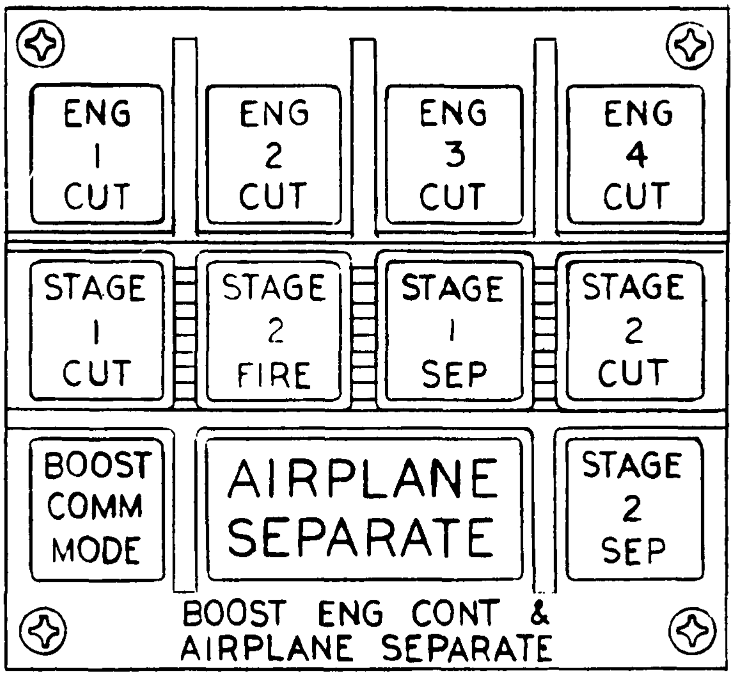


Figure IV-3. Boost Engine Control and Airplane Separation

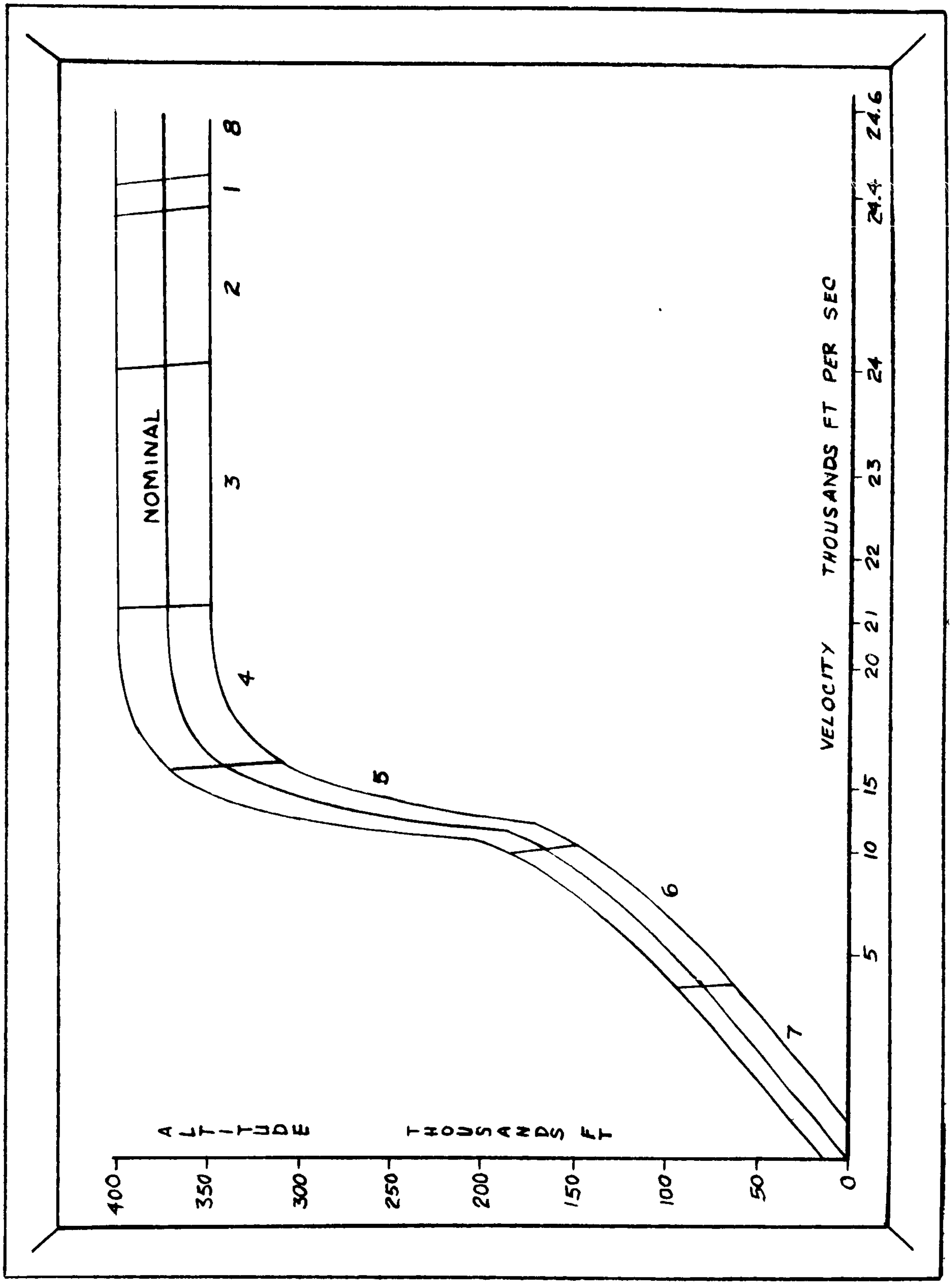


Figure IV-4. Boost Zone

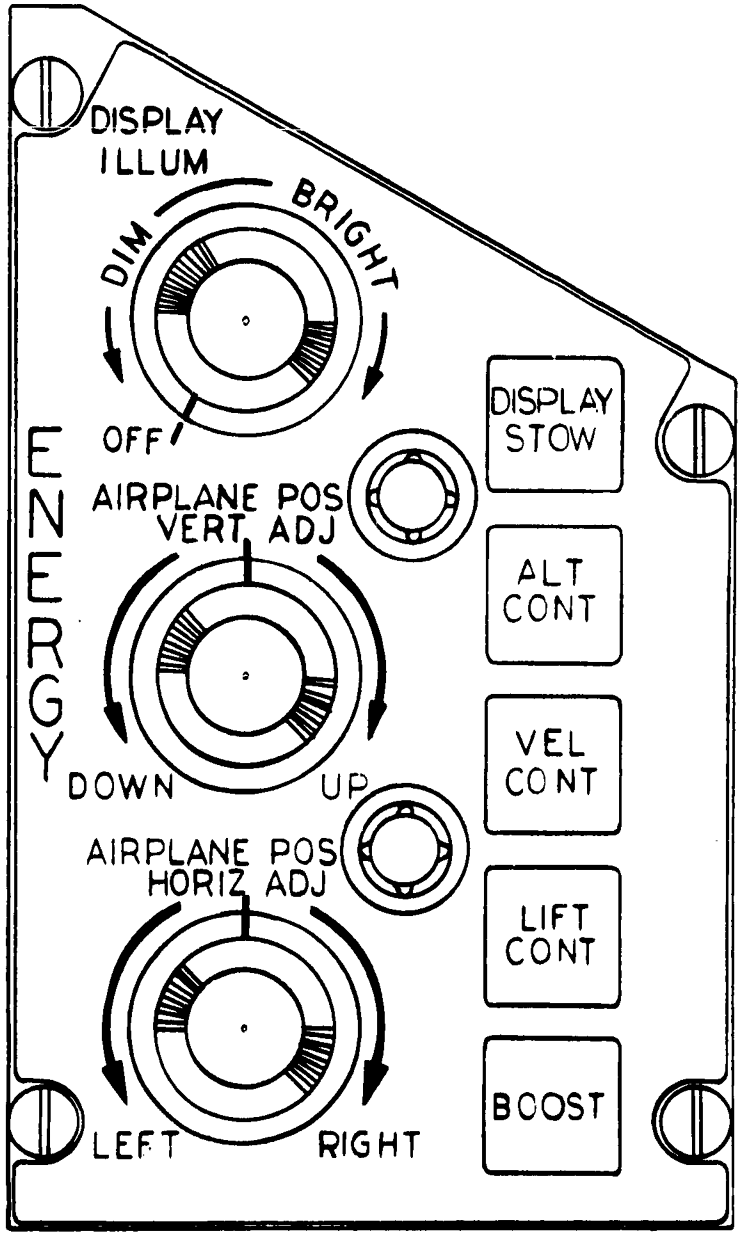


Figure IV-5. Energy Display Control Panel

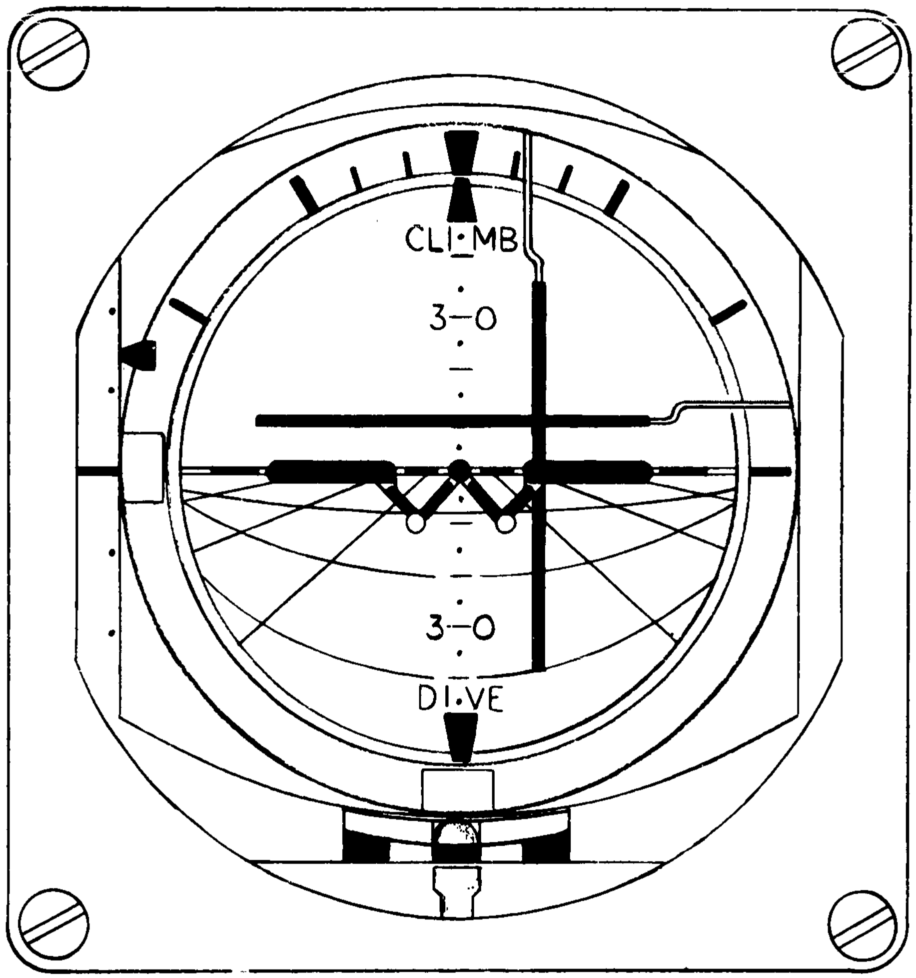


Figure V-1. Attitude Indicator/Flight Director

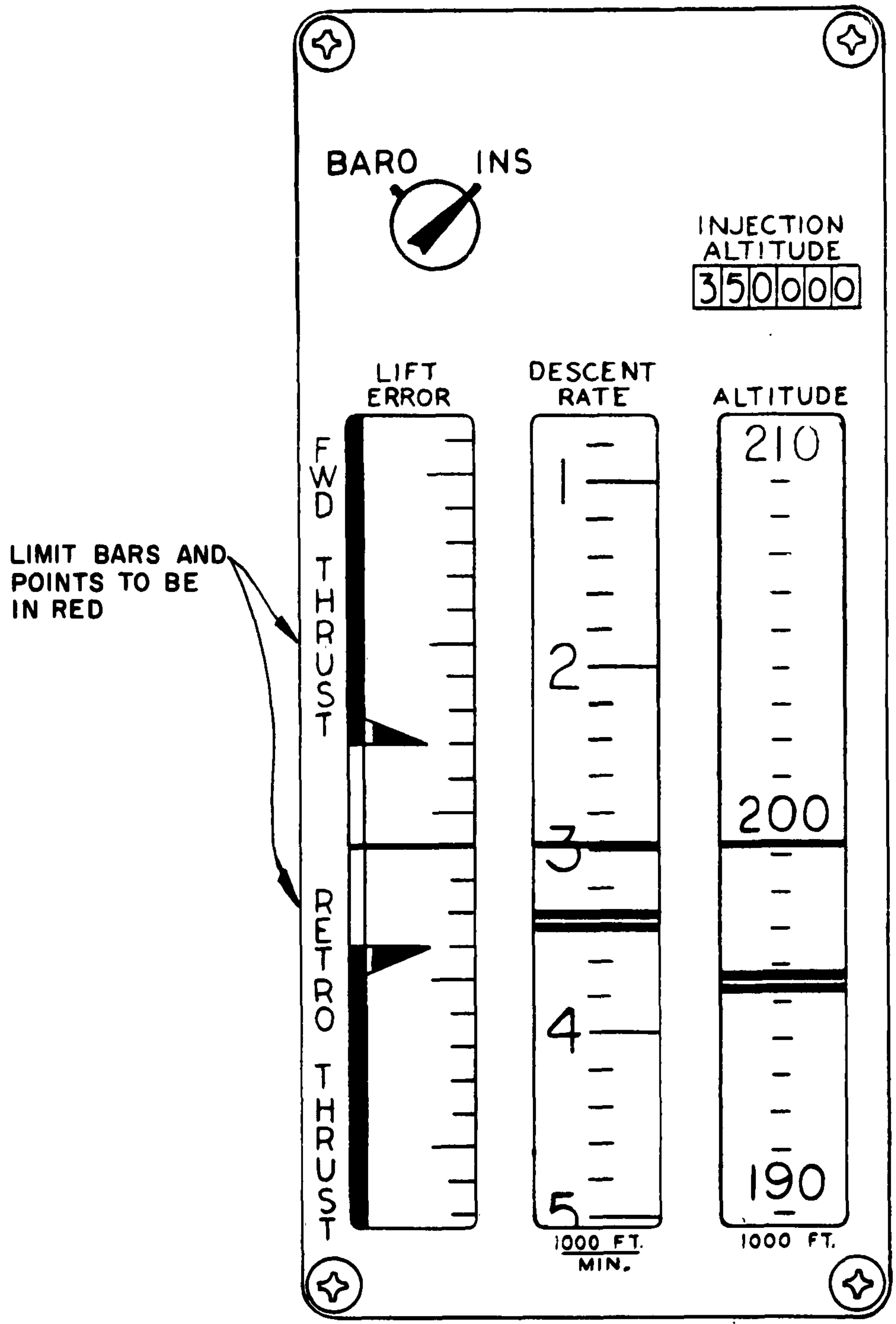


Figure V-2. Altitude, Descent Rate, and Lift Error Indicator

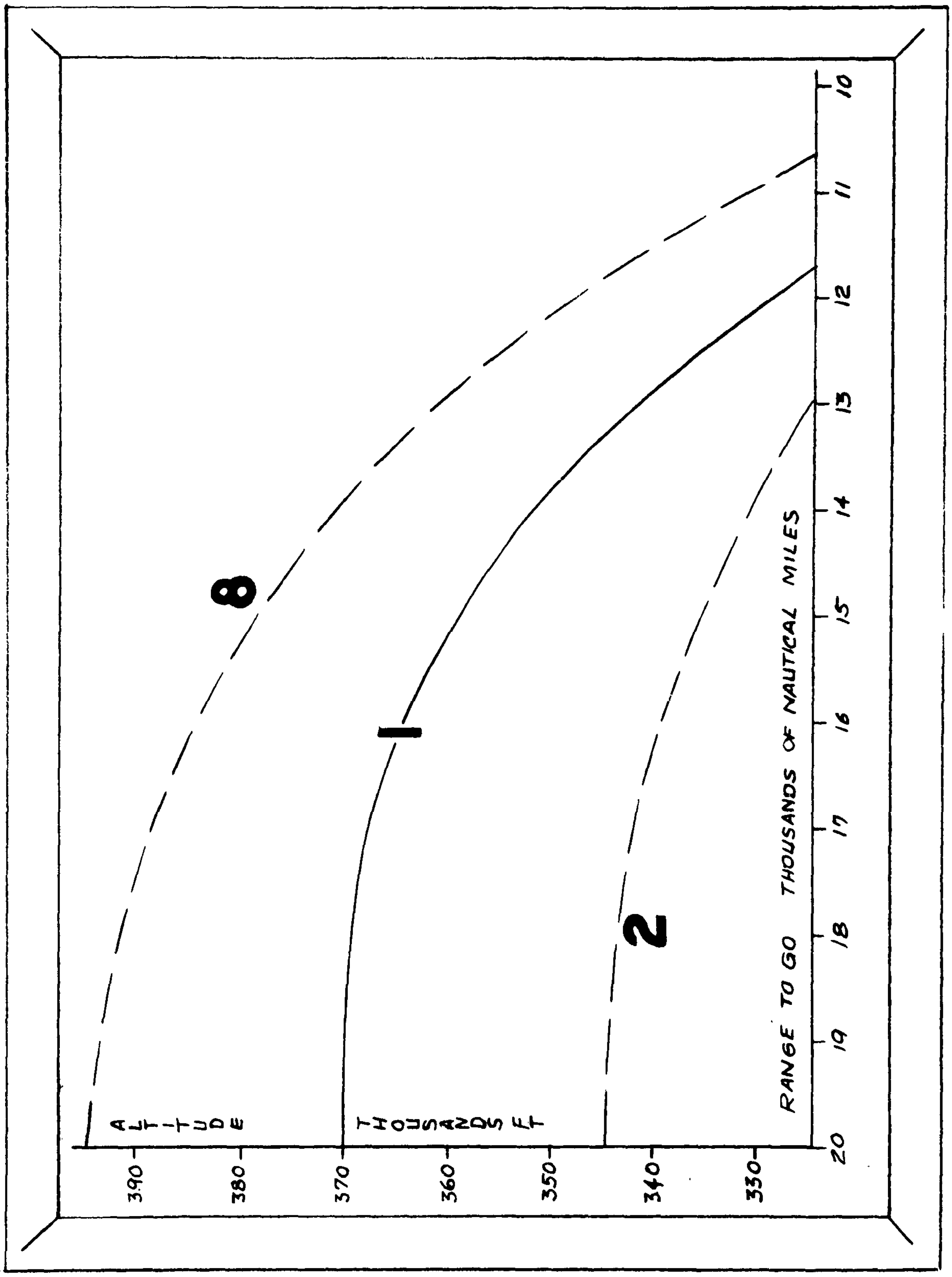


Figure V-3. Lift Control Graphic Screen Display

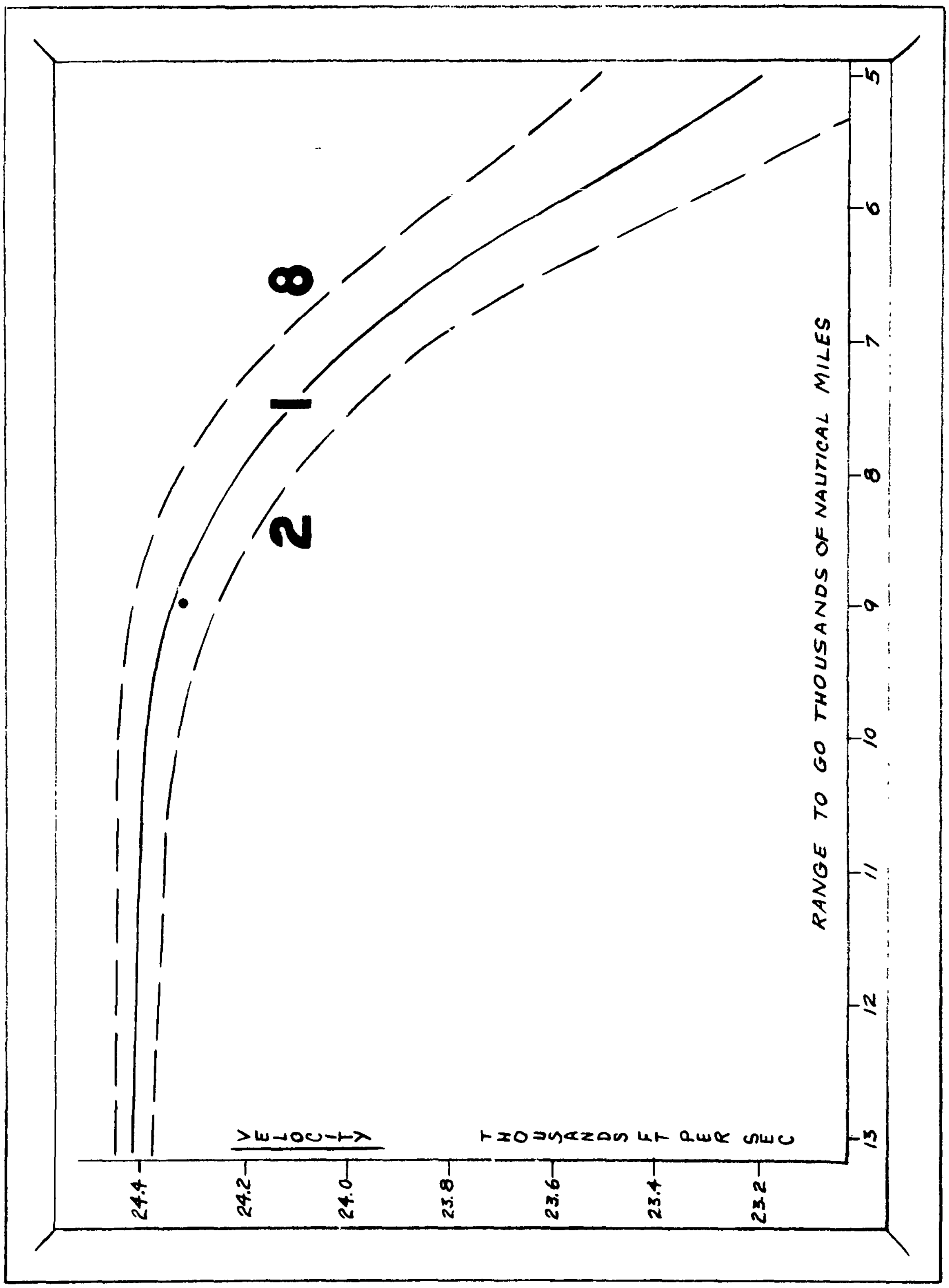


Figure V-4. Velocity Control Graphic Screen Display

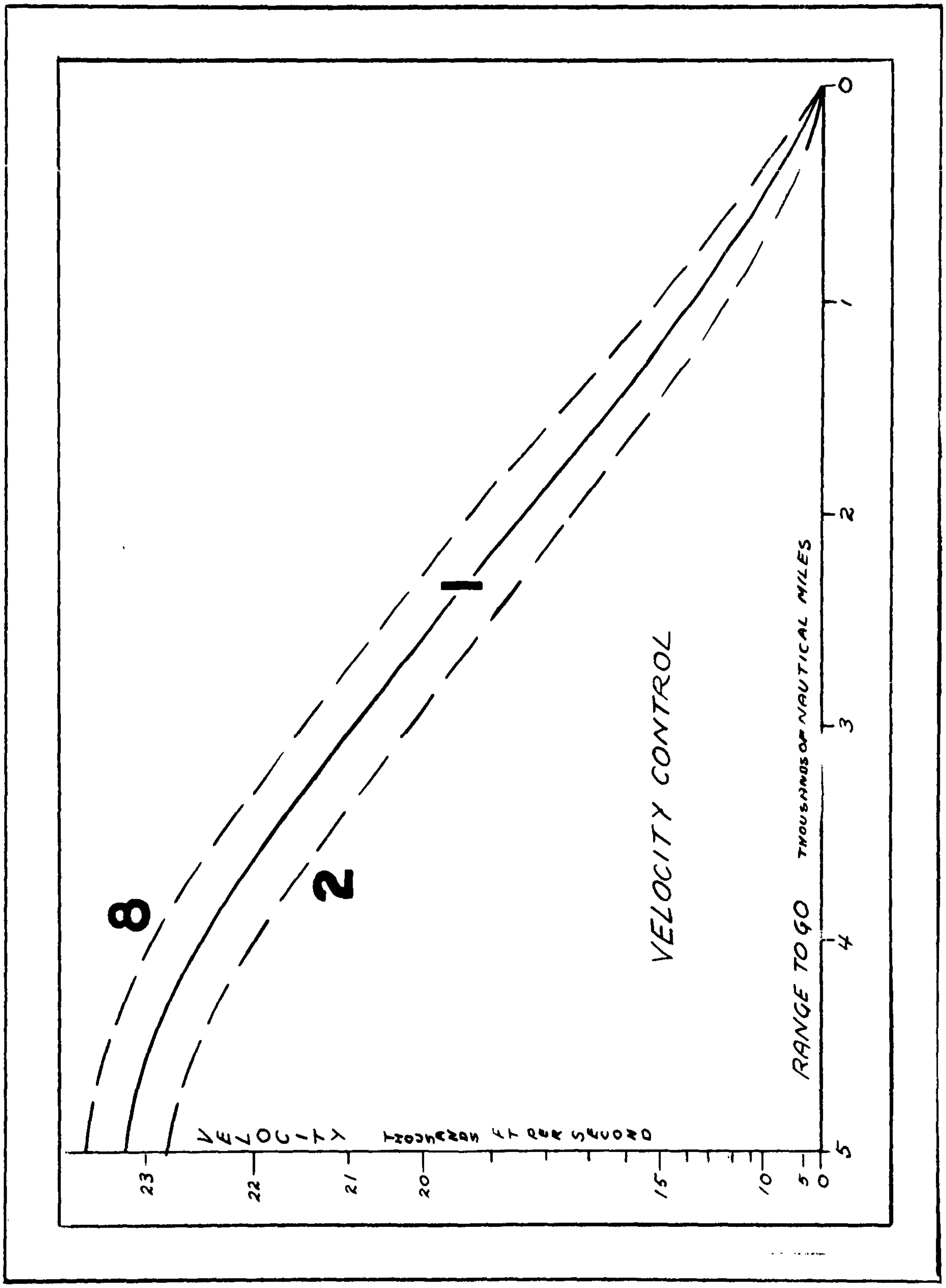


Figure V-5. Velocity Control Graphic Screen Display (Low Range)

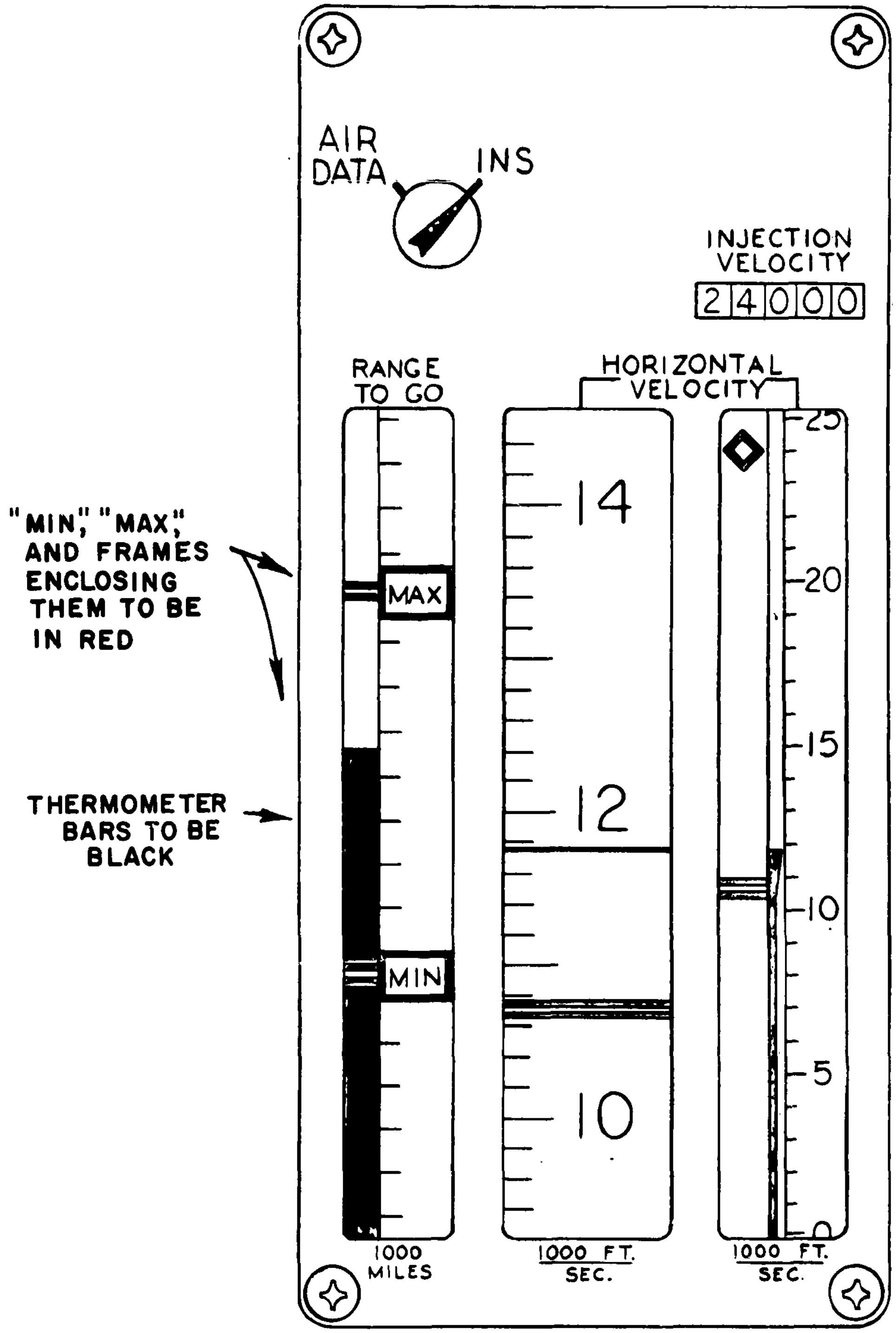


Figure V-6. Velocity and Range Indicator

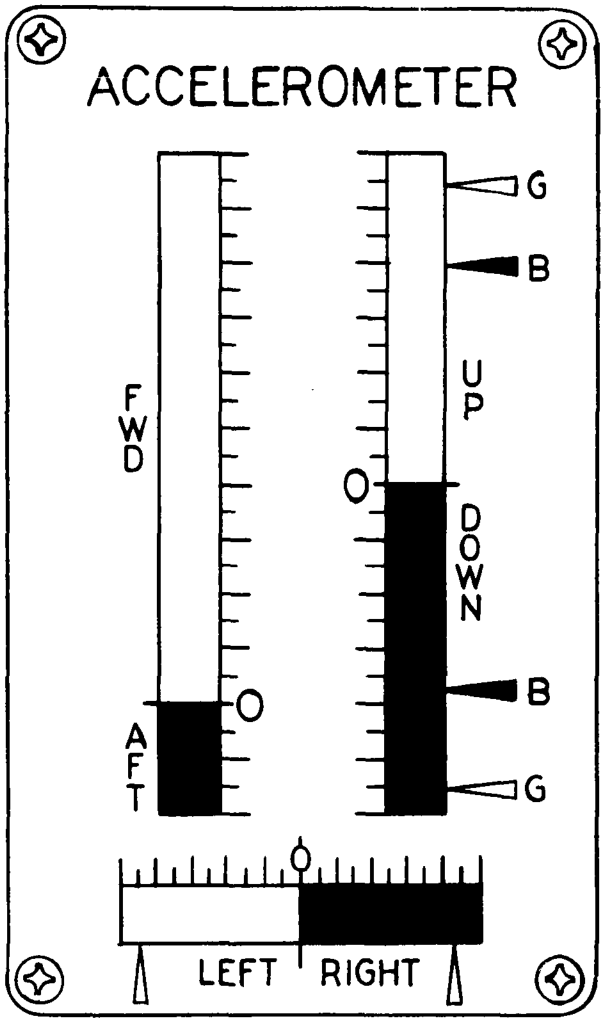


Figure V-7. Accelerometer

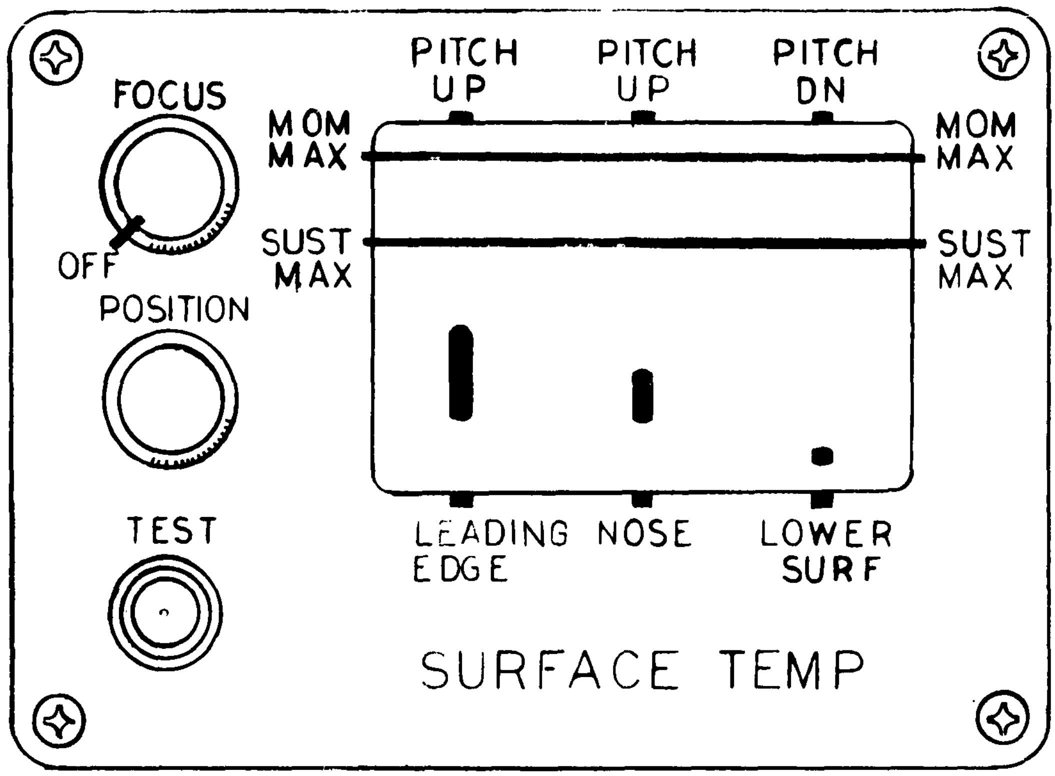


Figure V-8. Surface Temperature Indicator

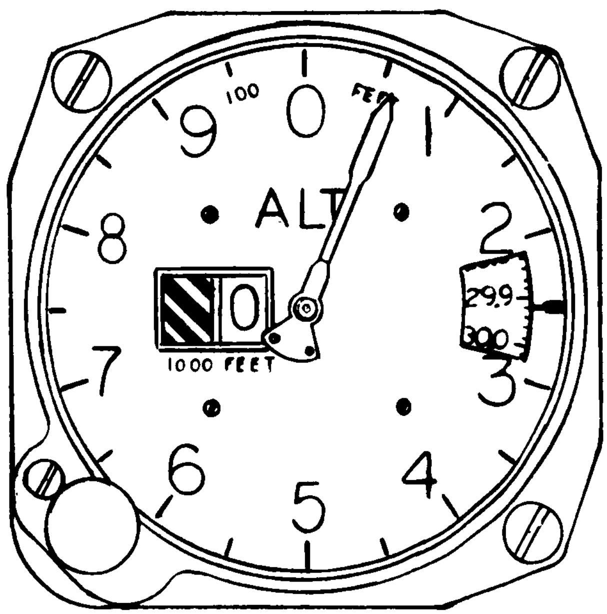


Figure V-9. Back-Up Altimeter

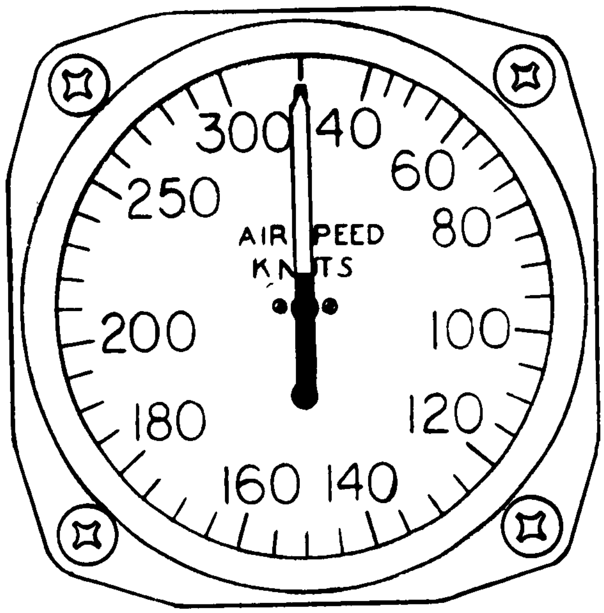


Figure V-10. Back-Up Airspeed Indicator

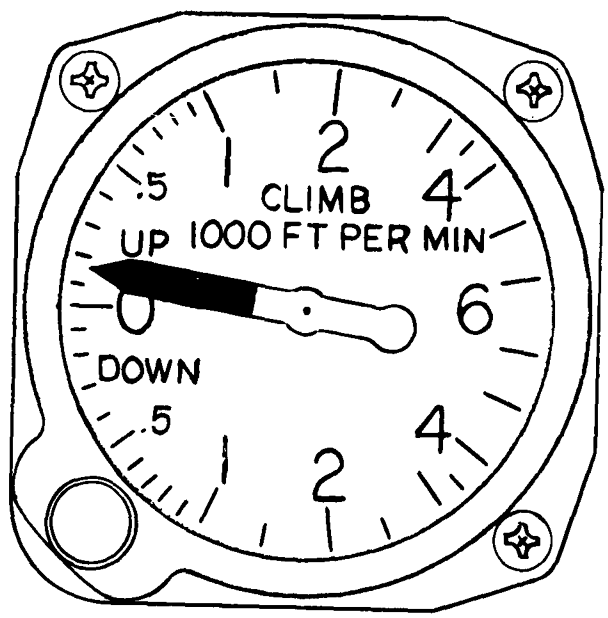


Figure V-11. Back-Up Rate of Climb Indicator

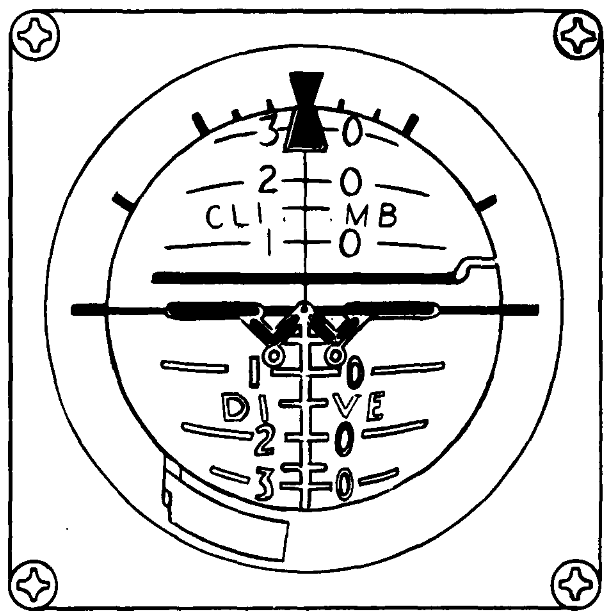


Figure V-12. Back-Up Attitude Indicator

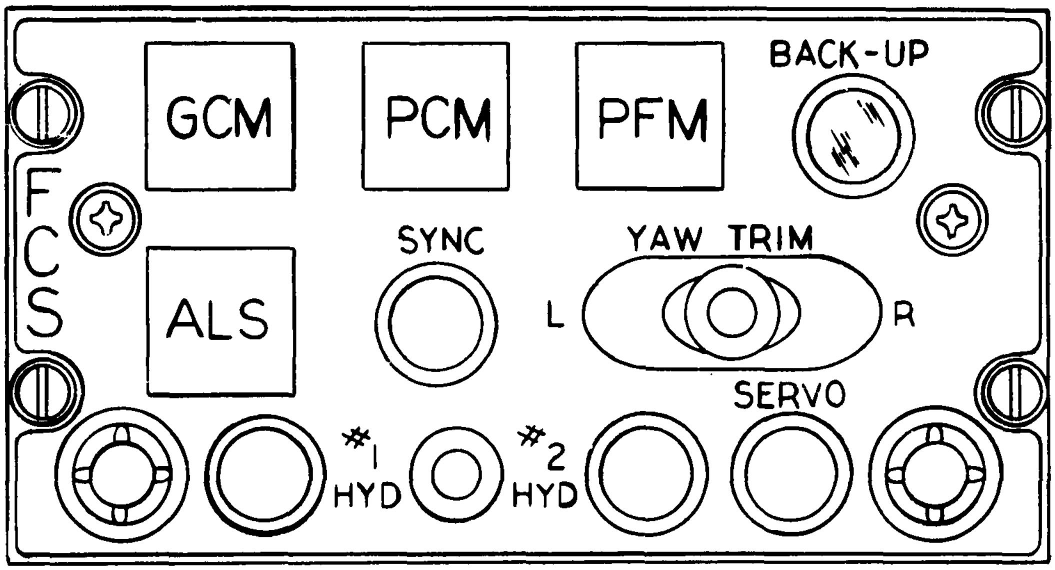


Figure V-13. Flight Control System Panel

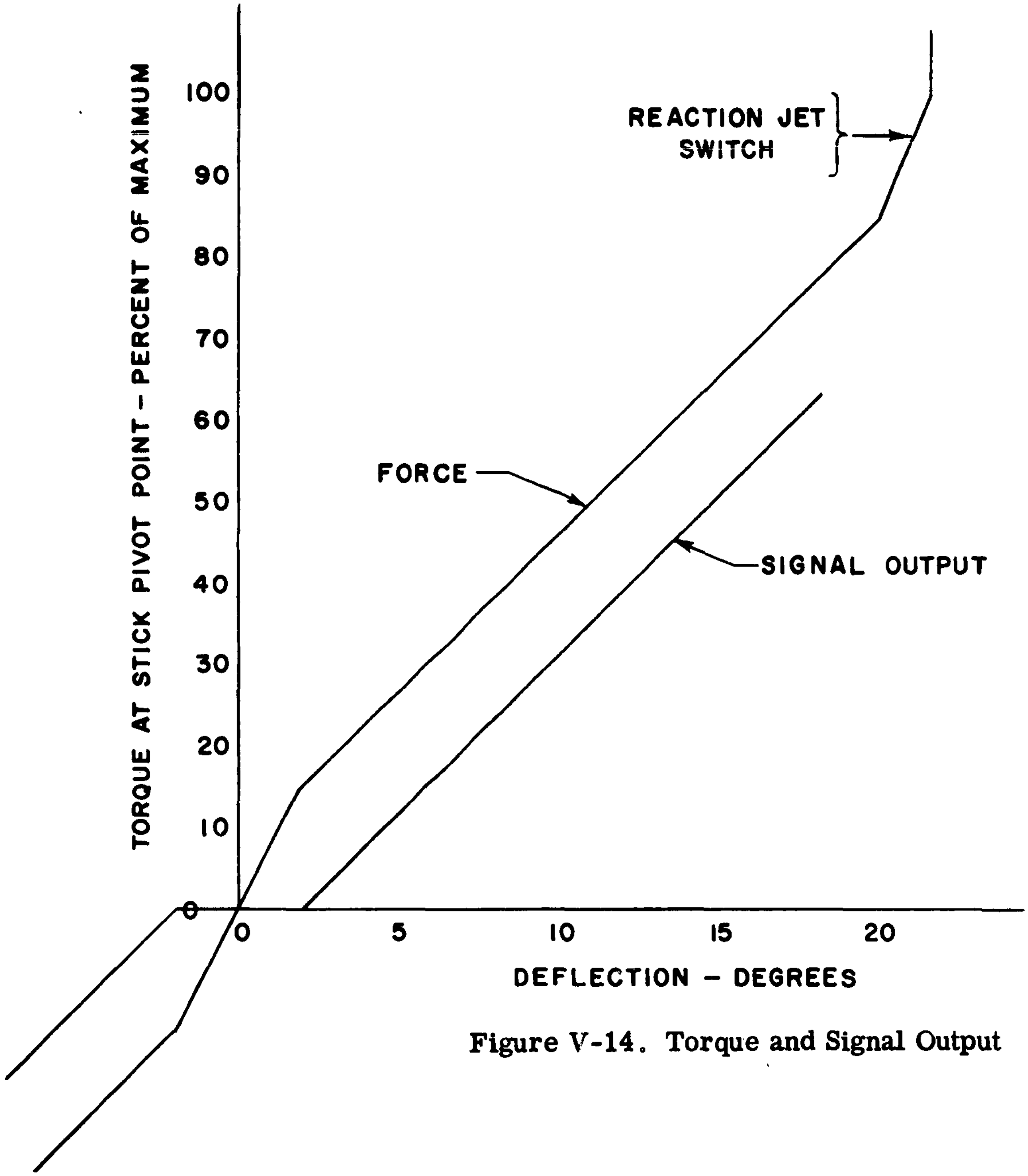


Figure V-14. Torque and Signal Output

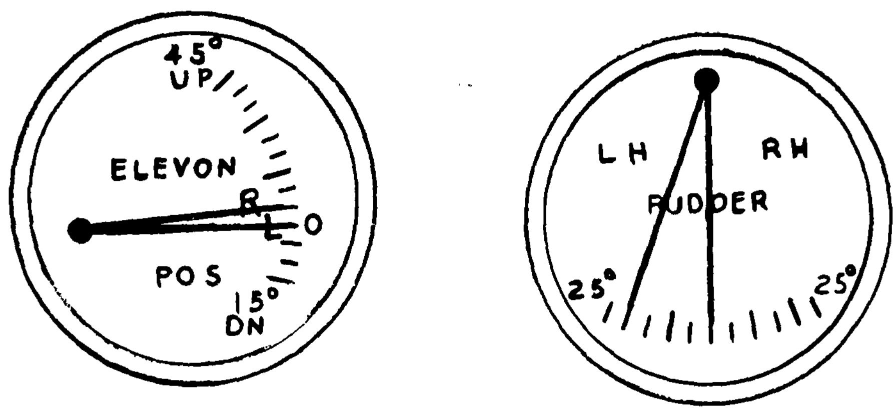


Figure V-15. Surface Position Indicator

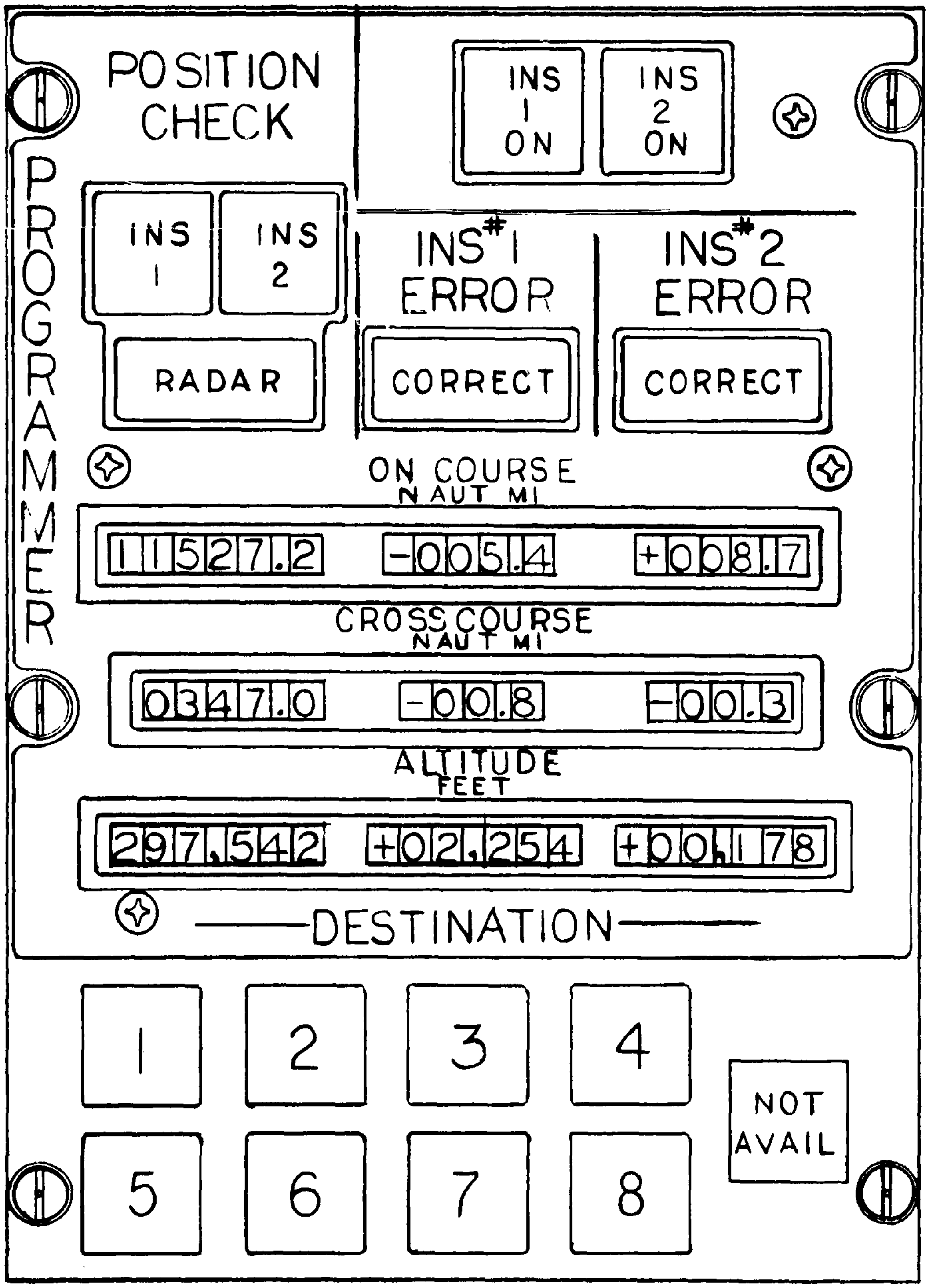


Figure VI-1. Programmer Panel

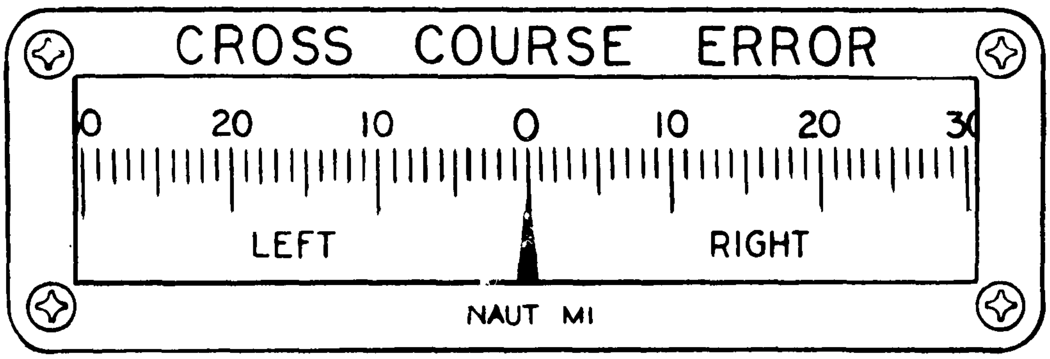


Figure VI-2. Cross Course Error

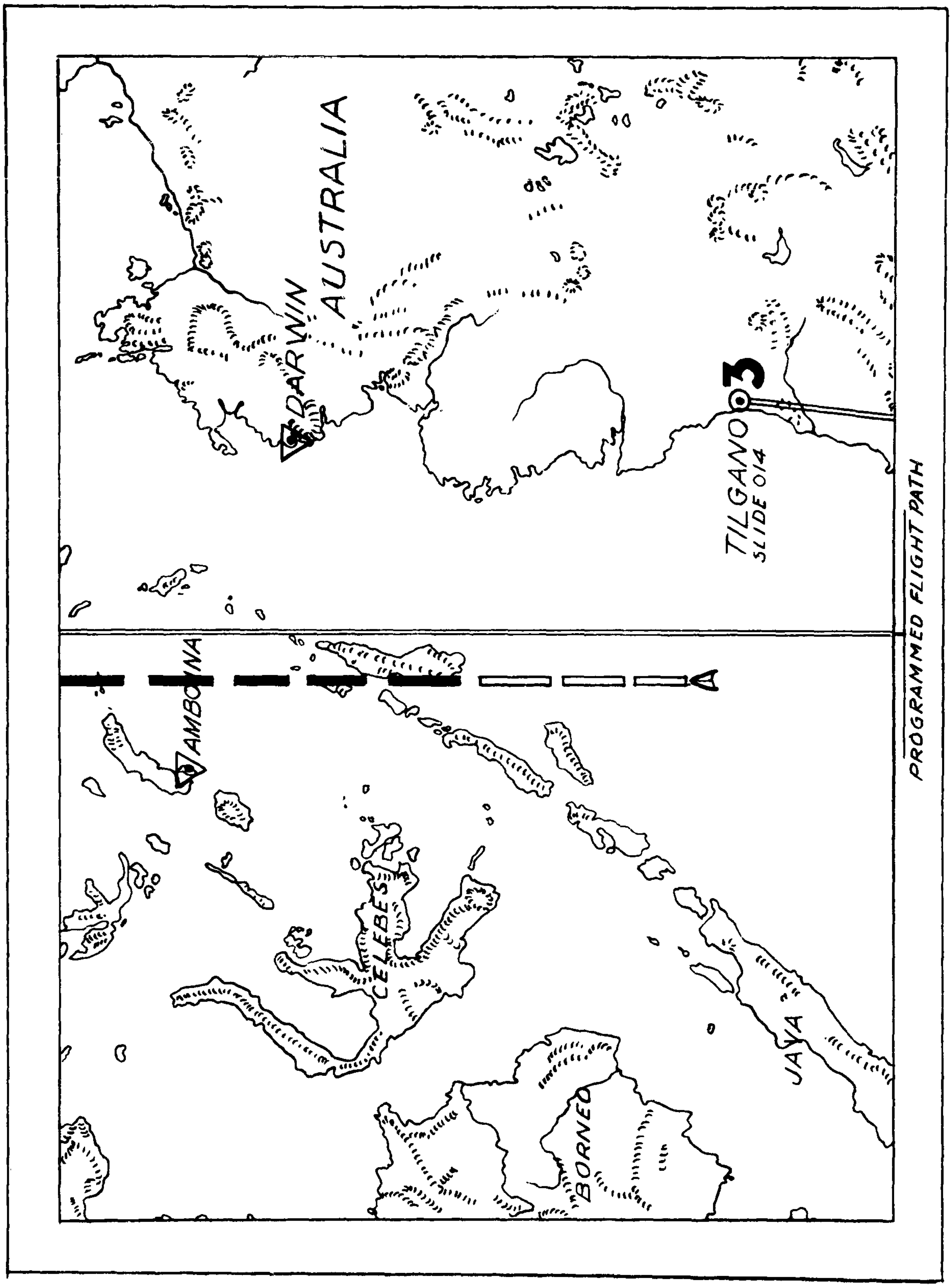


Figure VI-3. Strip Map

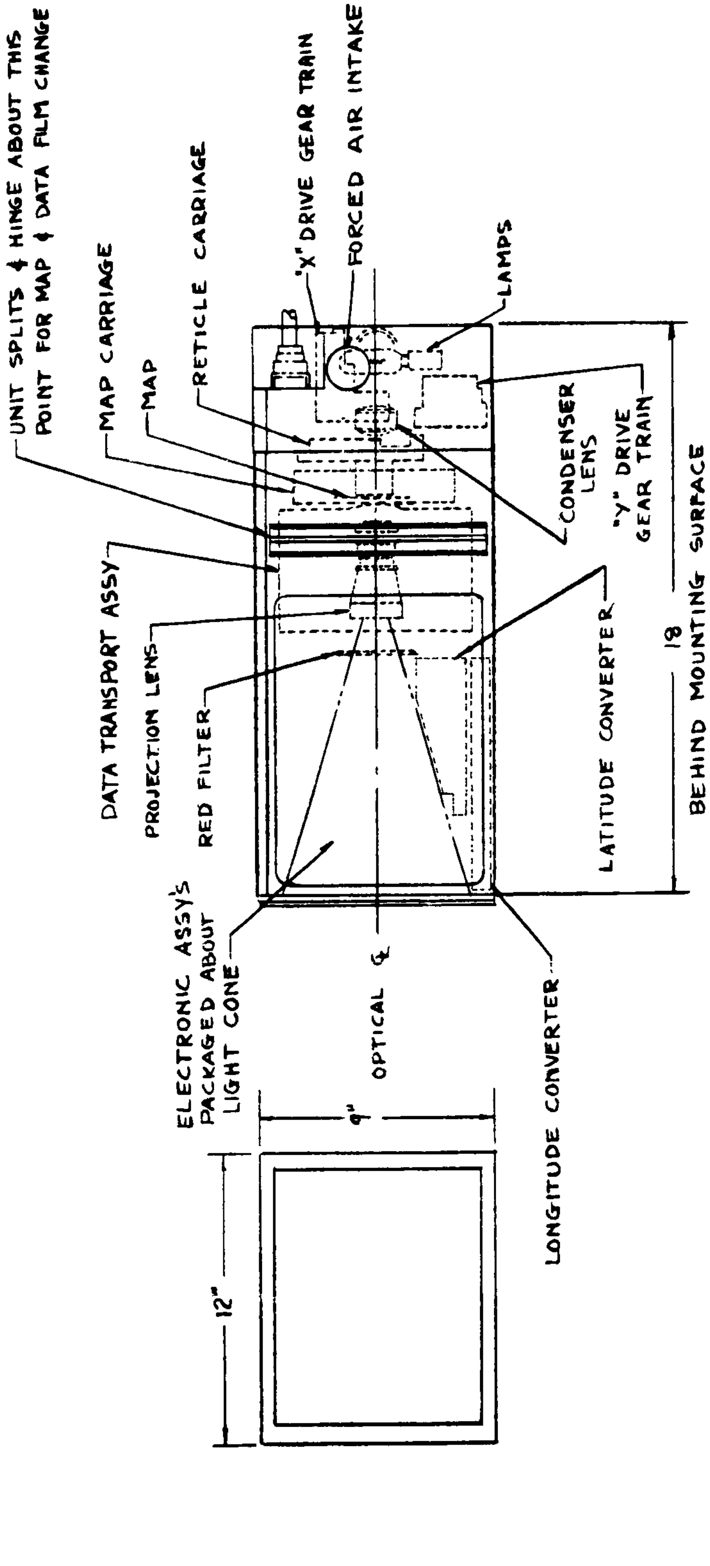


Figure VI-4. Map Display Apparatus

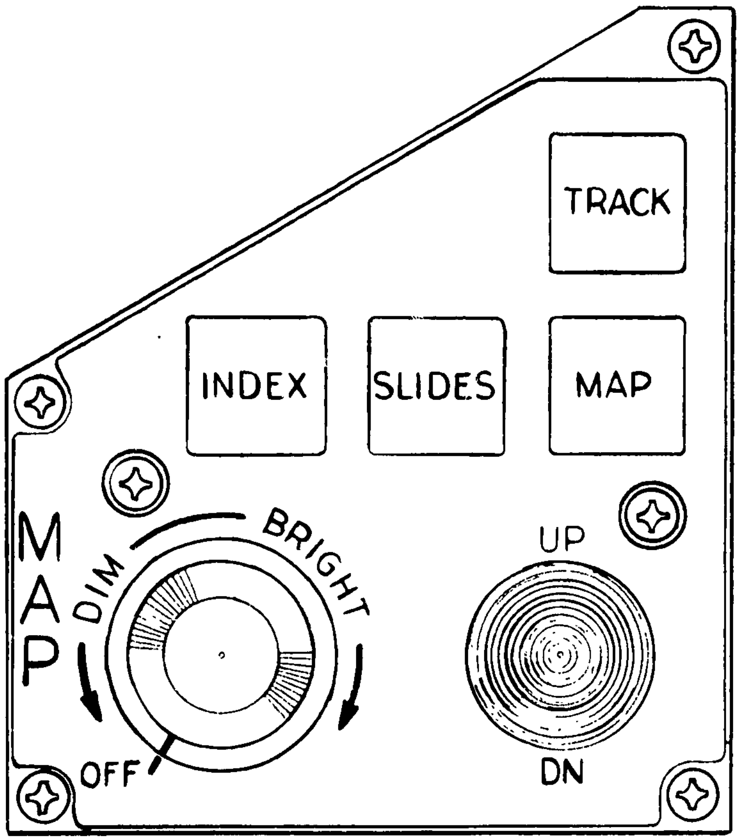


Figure VI-5. Map Control Panel

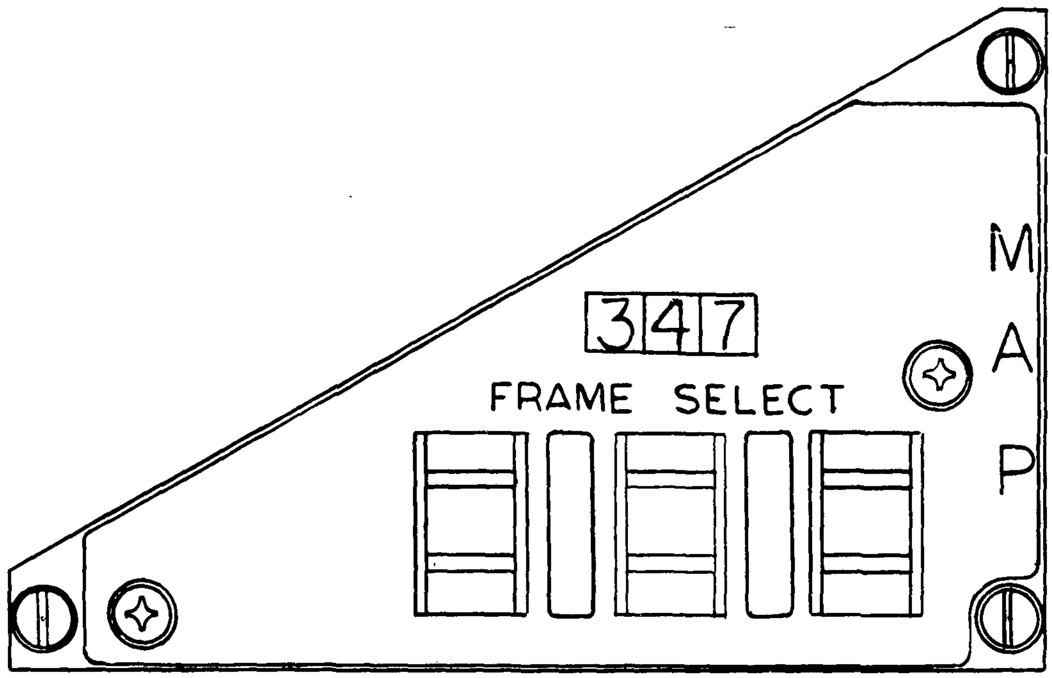


Figure VI-6. Frame Select Pan

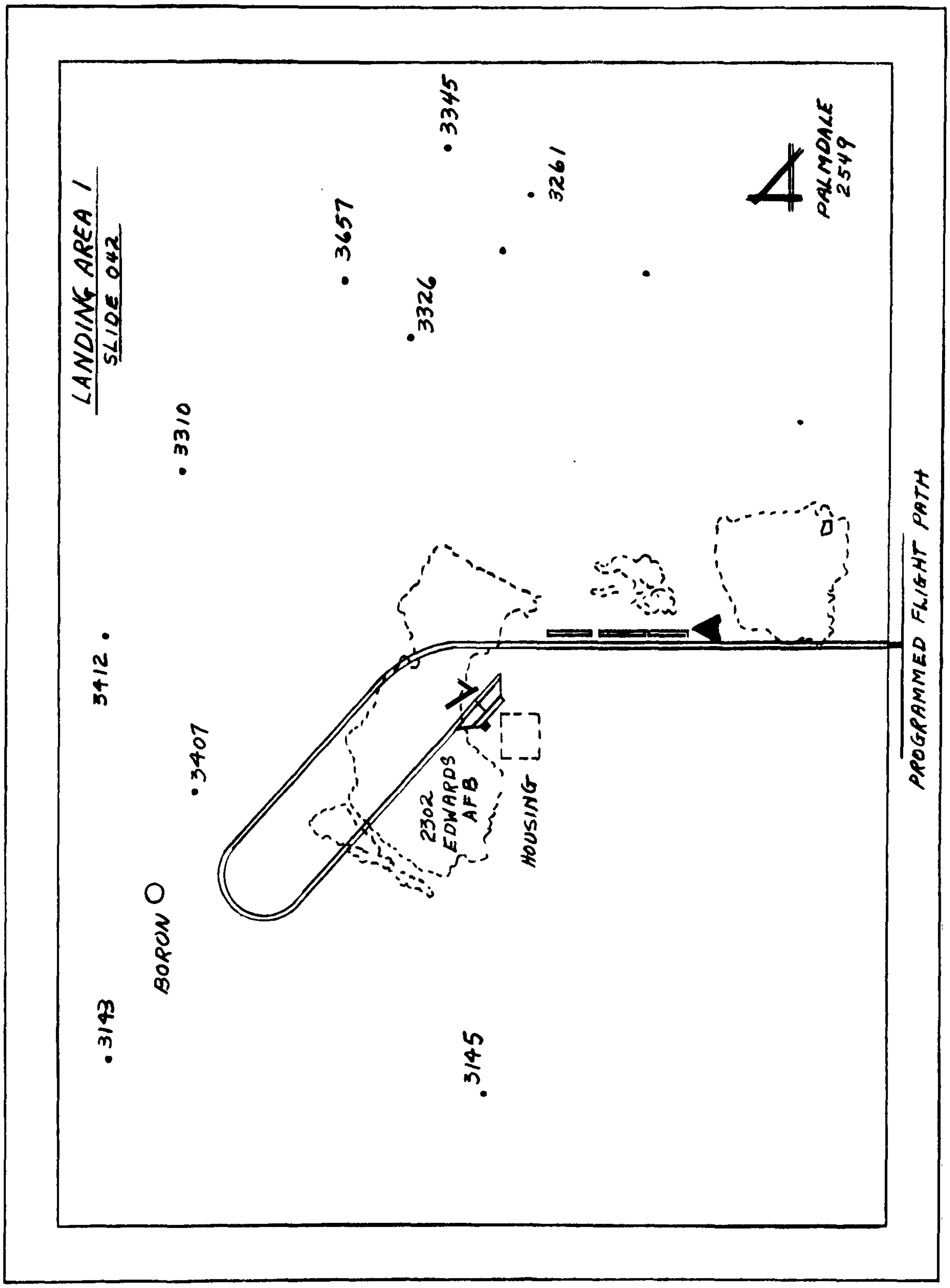


Figure VI-7. Landing Area Map

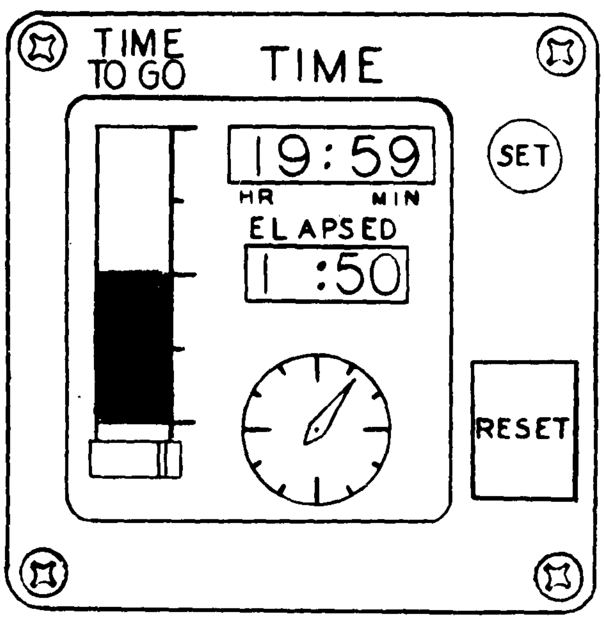
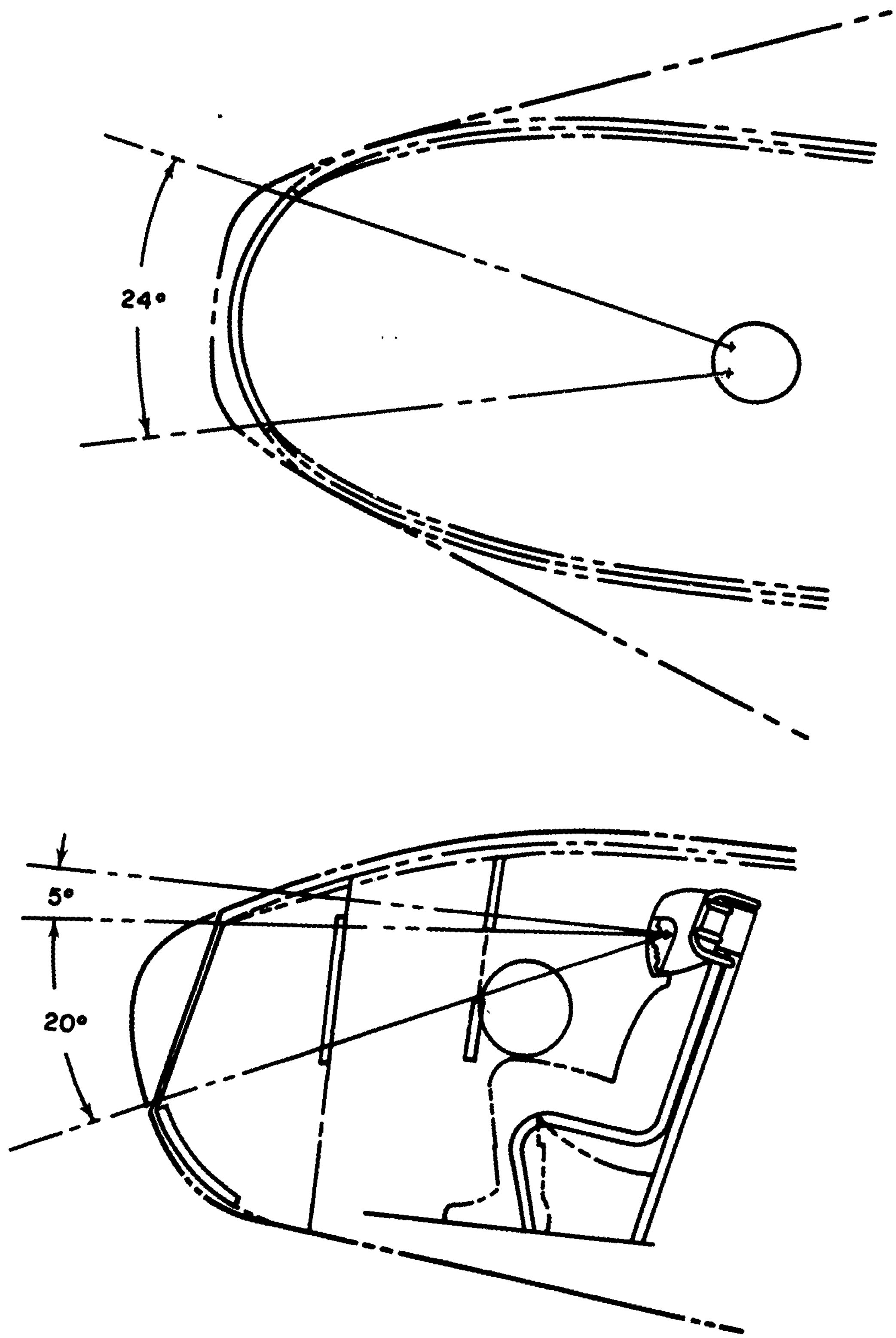
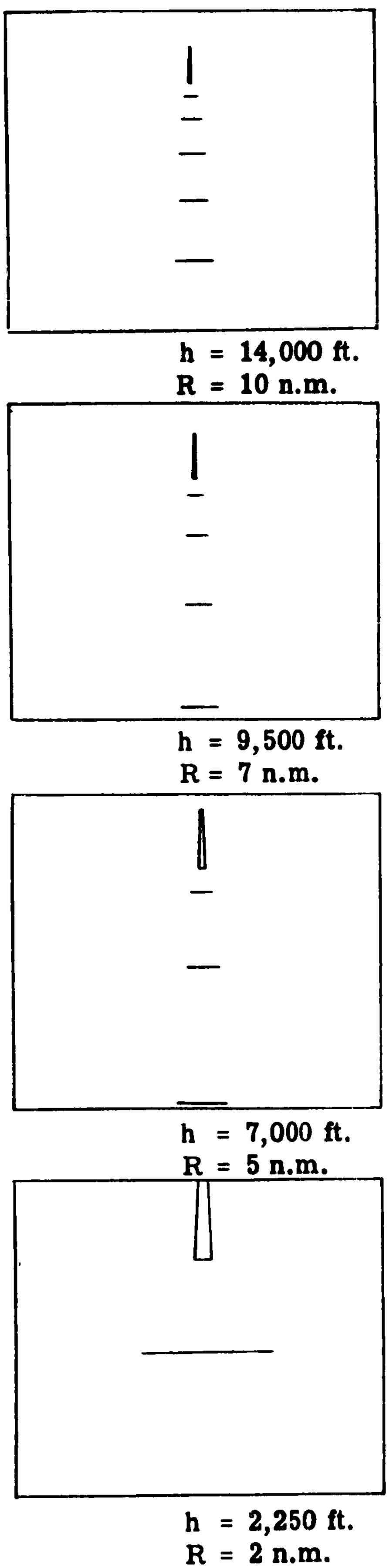


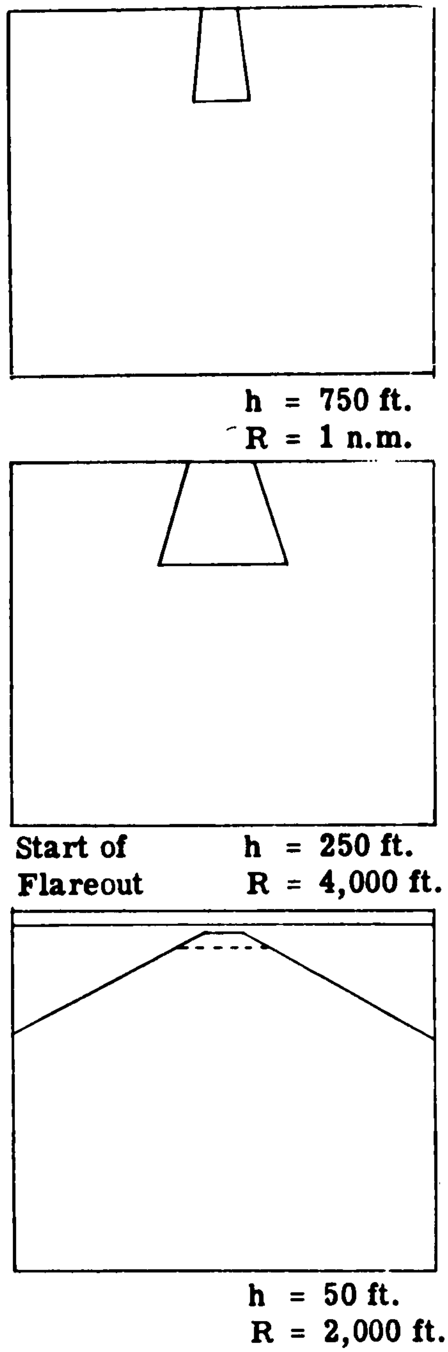
Figure VI-8. Time Indicator



ER 10390

Figure VI-9. Nose Window





Forward vision during final approach and flareout.

Glide Path 14°
Velocity 330 ft/sec.
Touchdown at runway - 3000 ft.
Runway shown 10,000 ft. long x 150 ft. wide
Approach markers 1,000 ft. wide, spaced at 1 n.m. intervals

Figure VI-10. Landing Approach Visibility

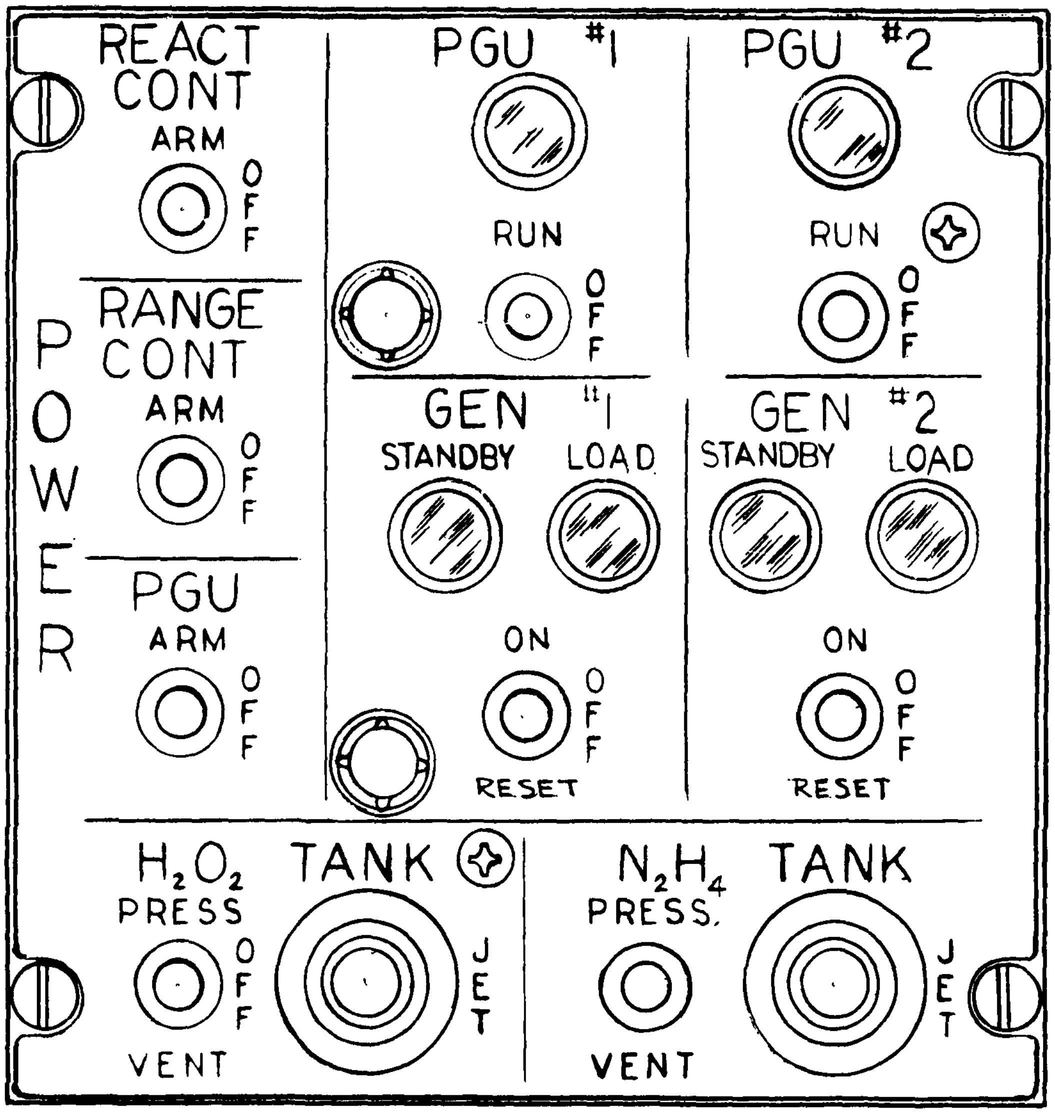


Figure VII-1. Power Control Panel

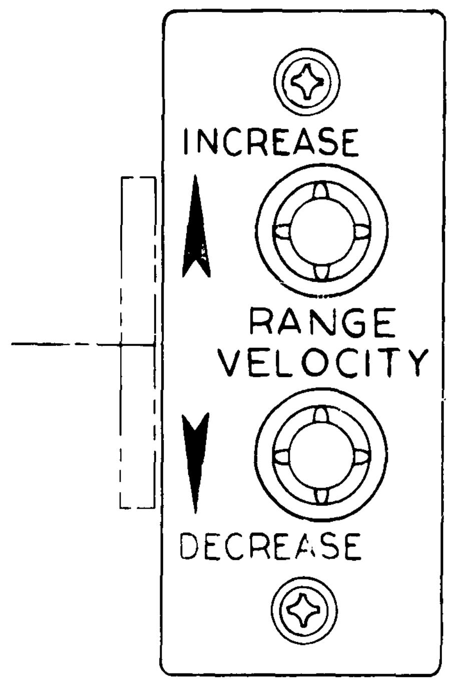


Figure VII-2. Range Velocity Control Panel

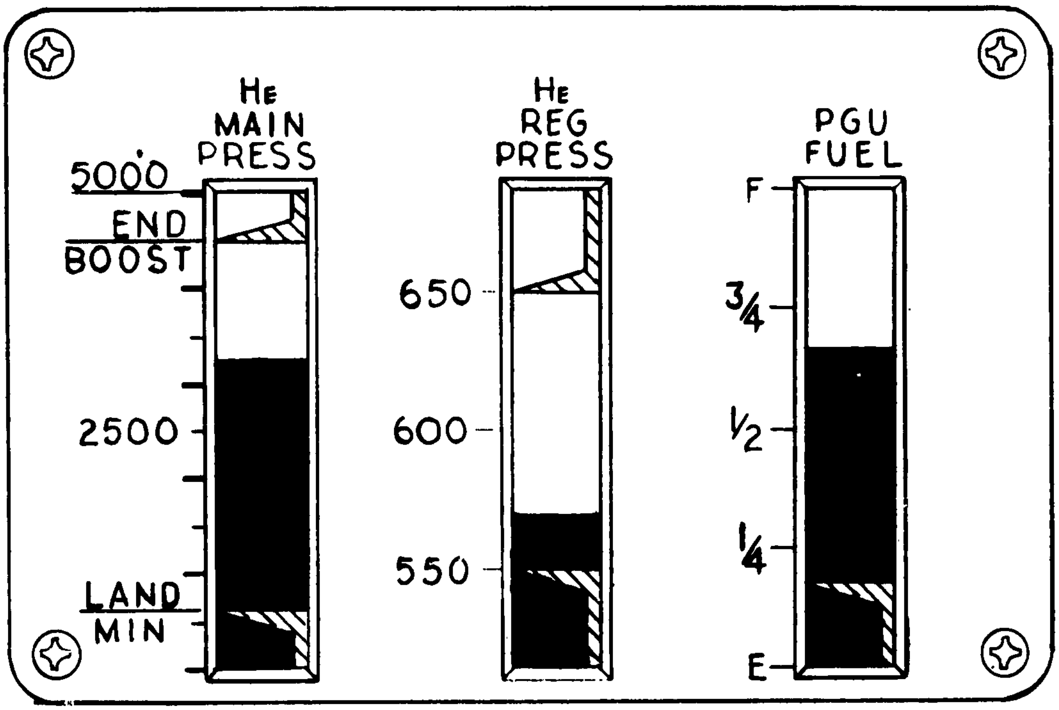


Figure VII-3. Helium and PGU Propellant Quantity Indicator

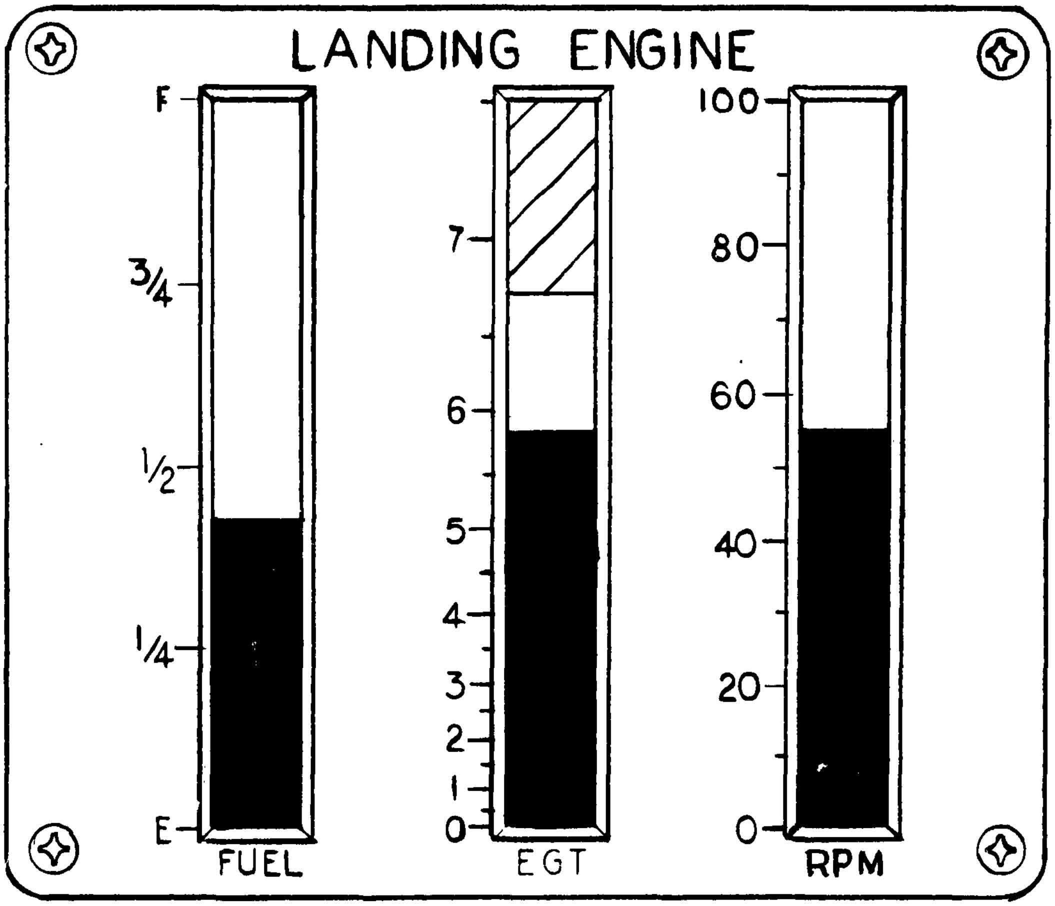


Figure VII-4. Landing Engine Instruments

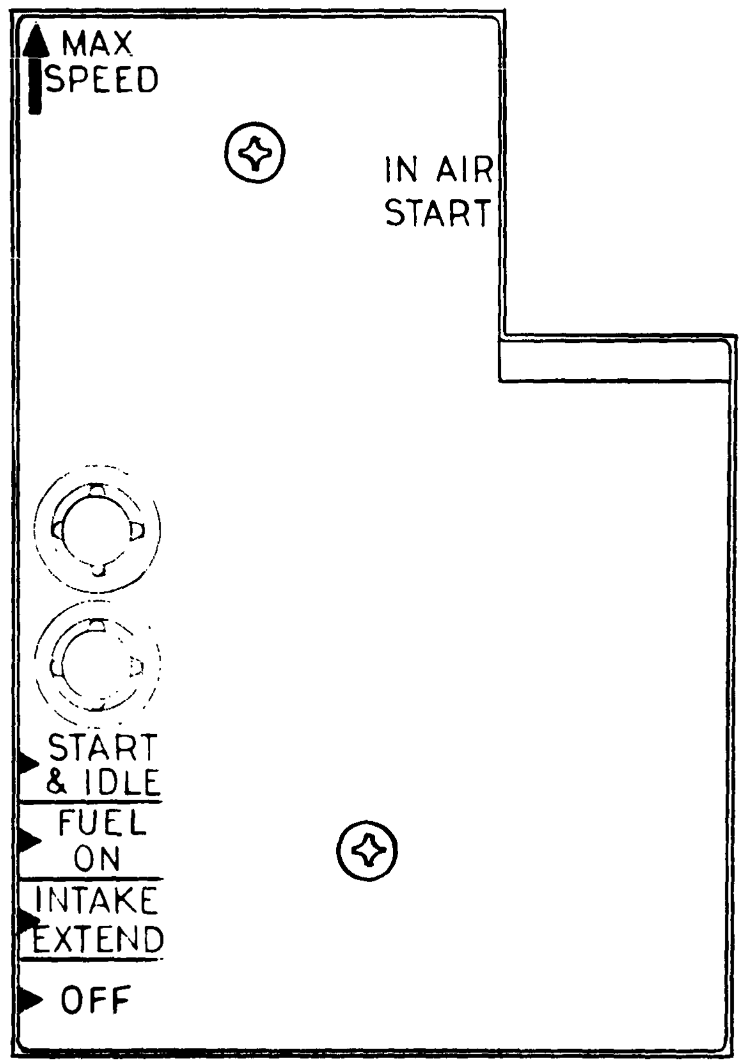


Figure VII-5. Throttle - Jet Engine

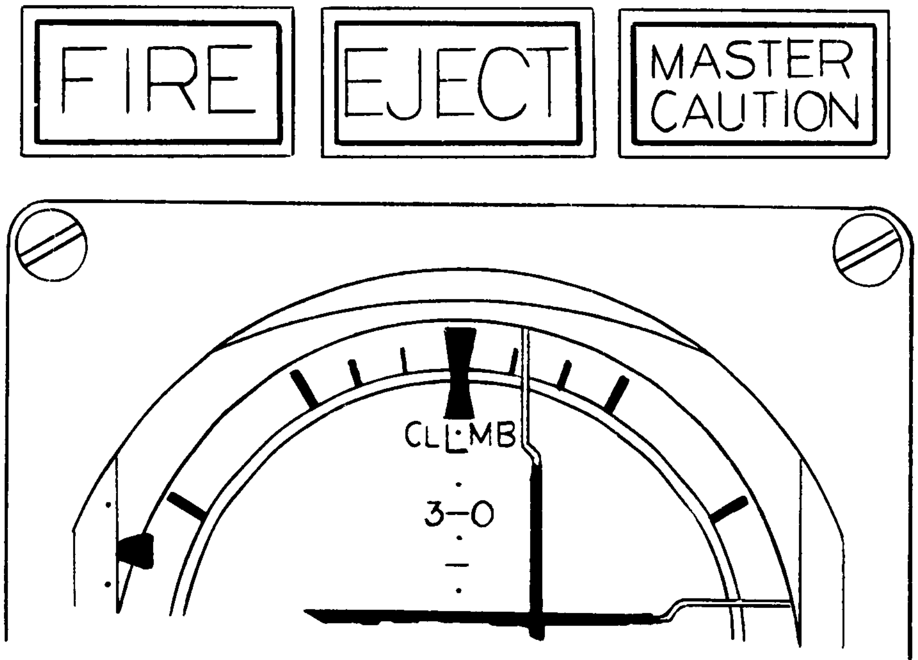


Figure VIII-1. Central Warning Indicators

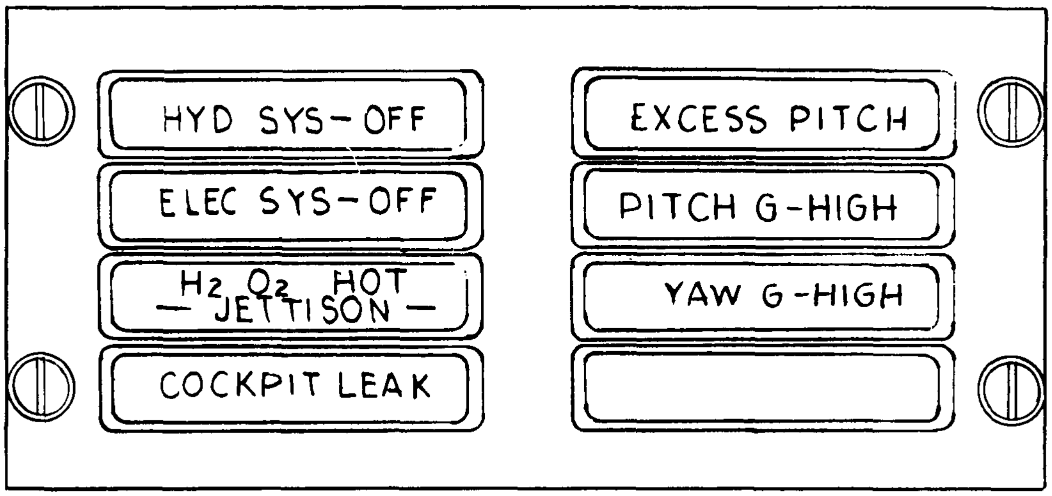


Figure VIII-2. Warning Indicator Panel

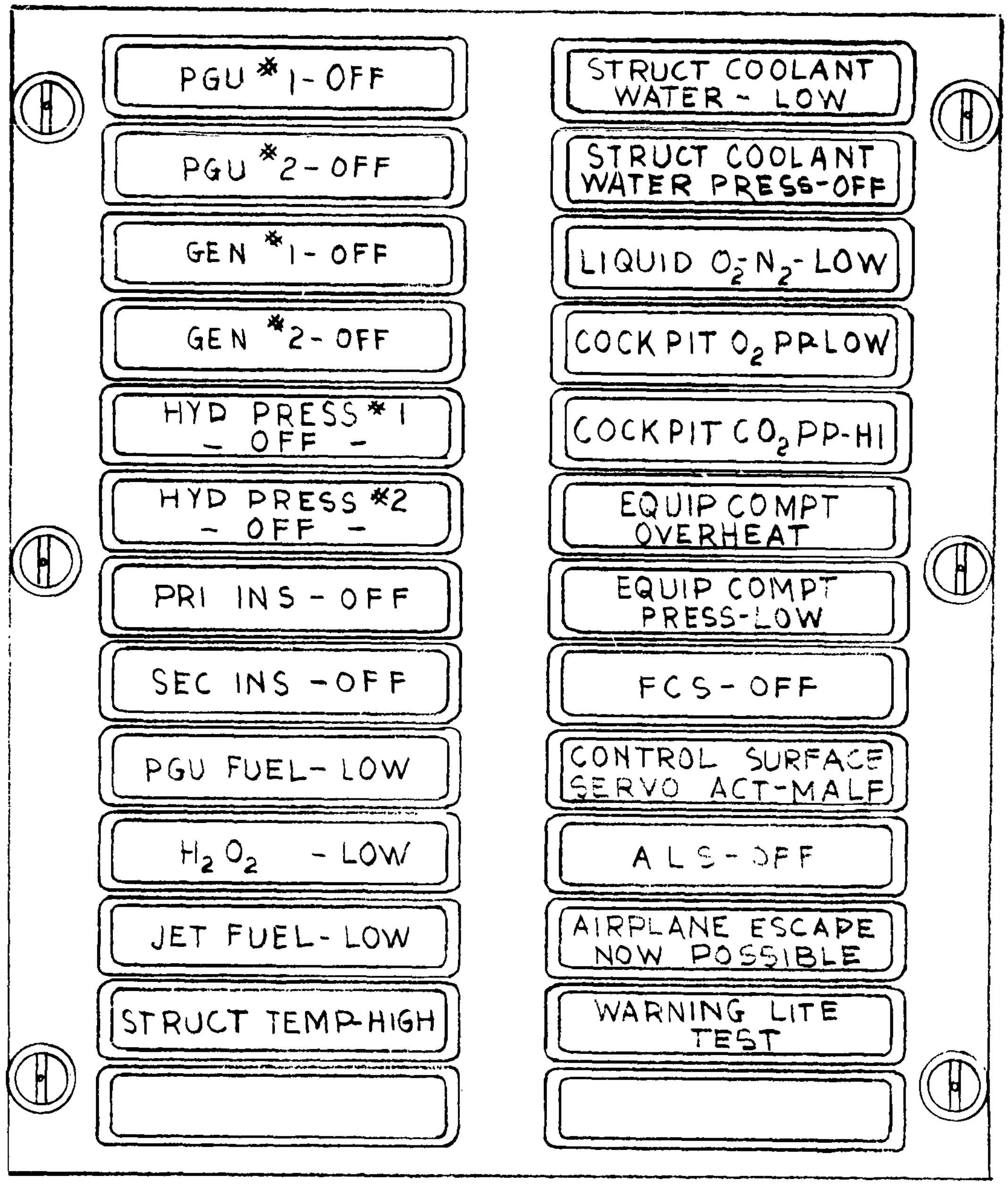


Figure VIII-3. Caution Light Panel

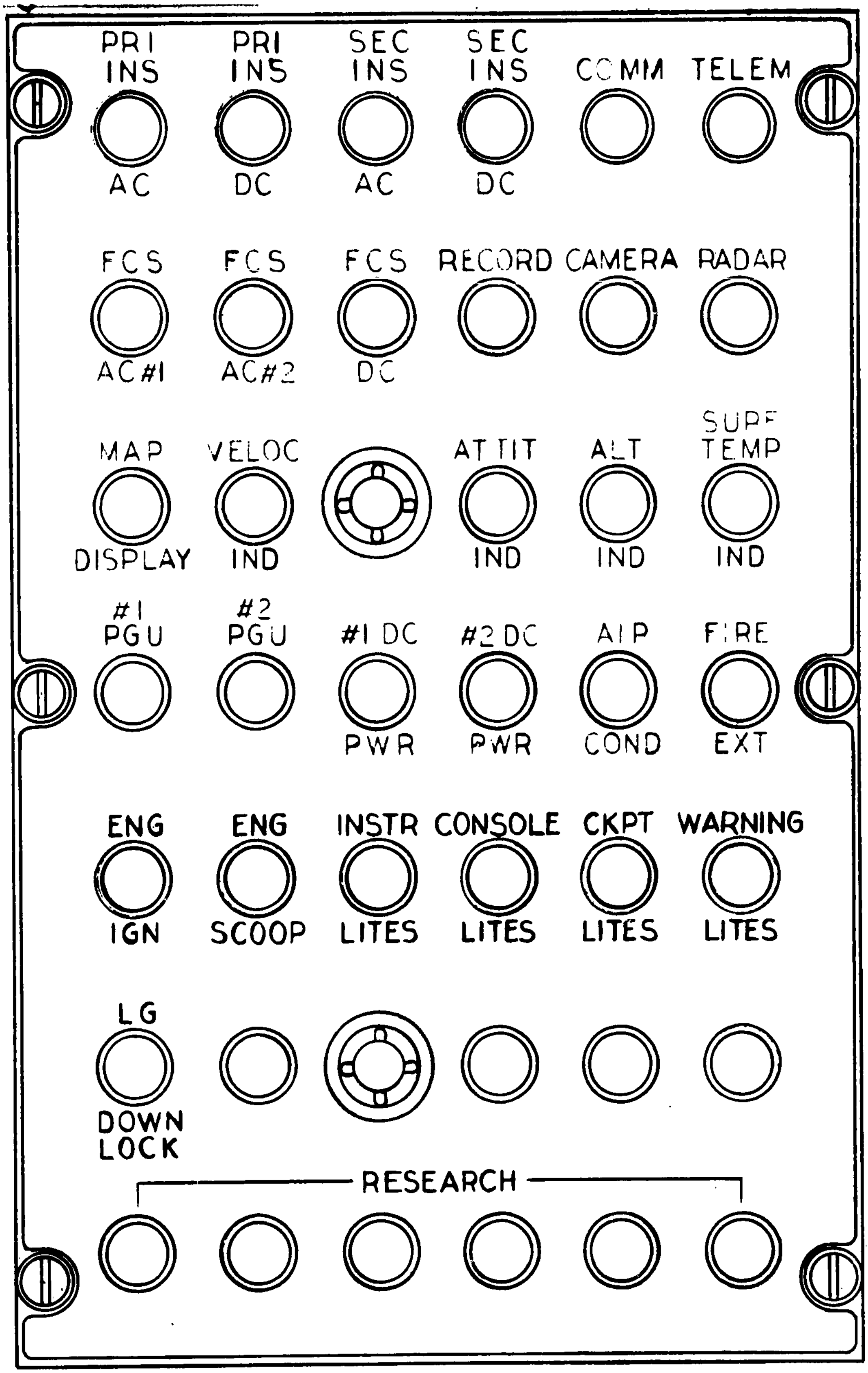


Figure VIII-4. Circuit Breaker Panel

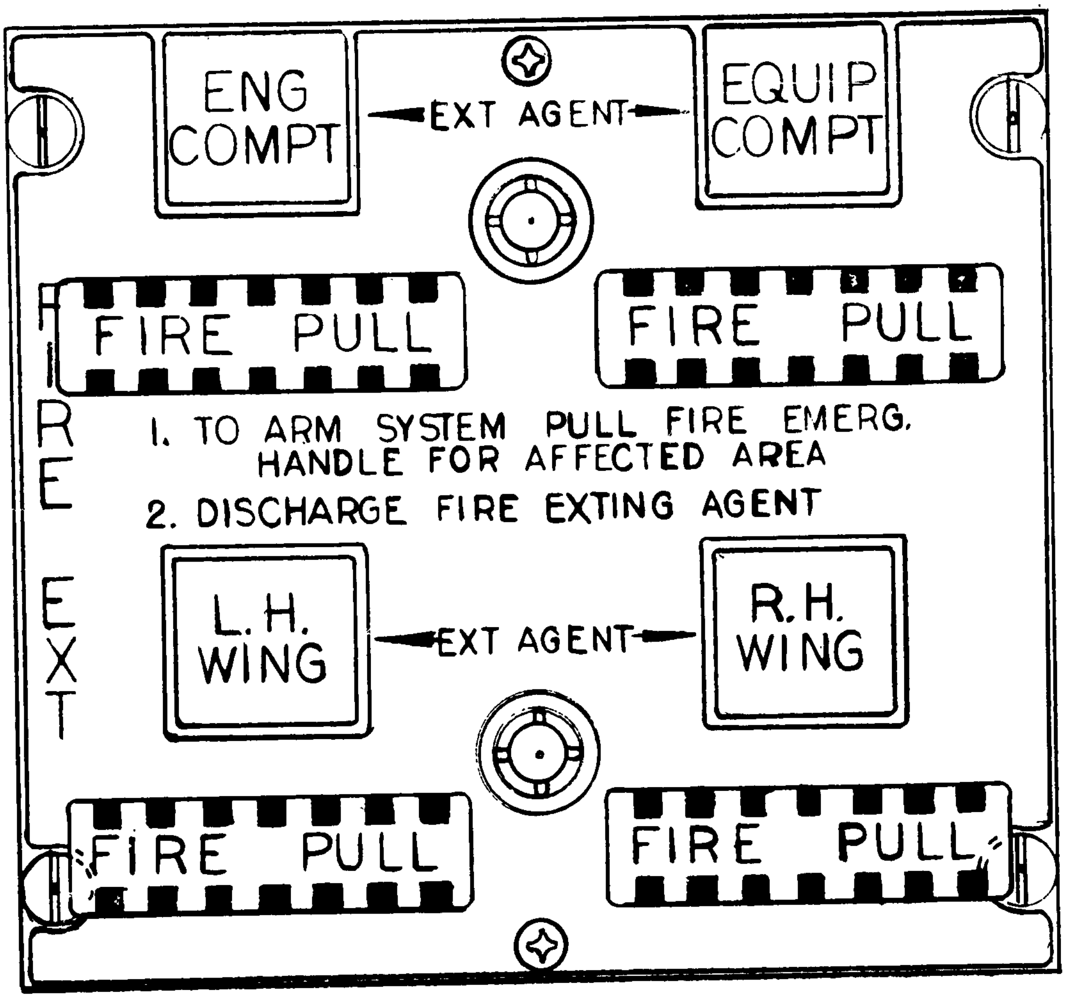


Figure VIII-5. Fire Extinguisher Panel

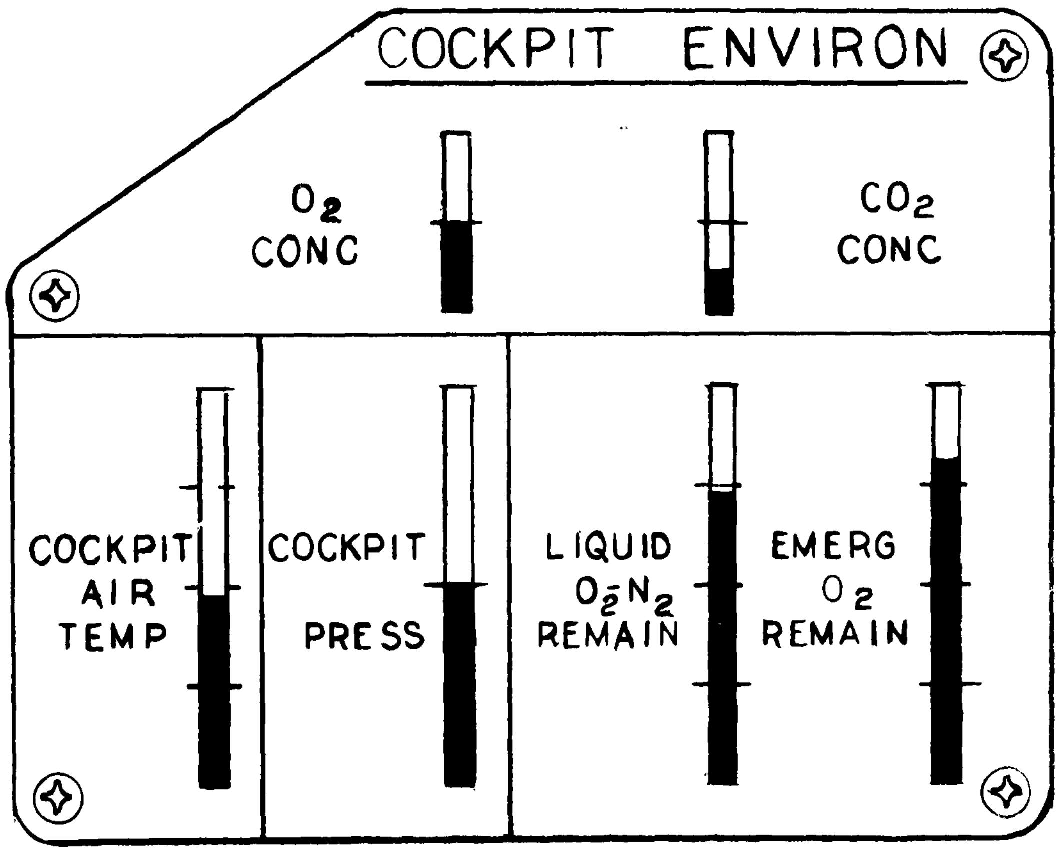


Figure IX-1. Cockpit Environment Indicator

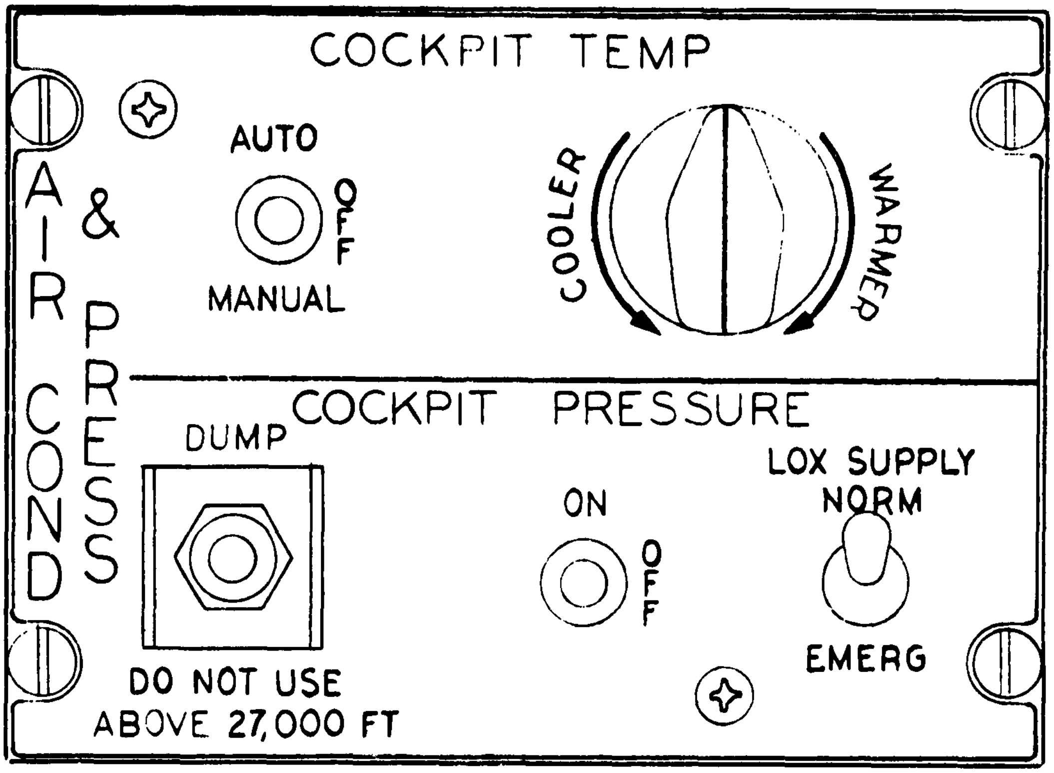


Figure IX-2. Cockpit Air Conditioning and Pressure Control Panel

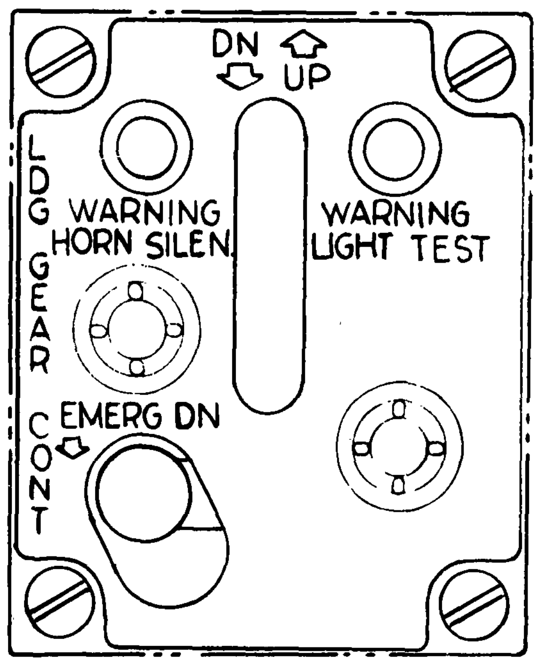


Figure IX-3. Landing Gear Control

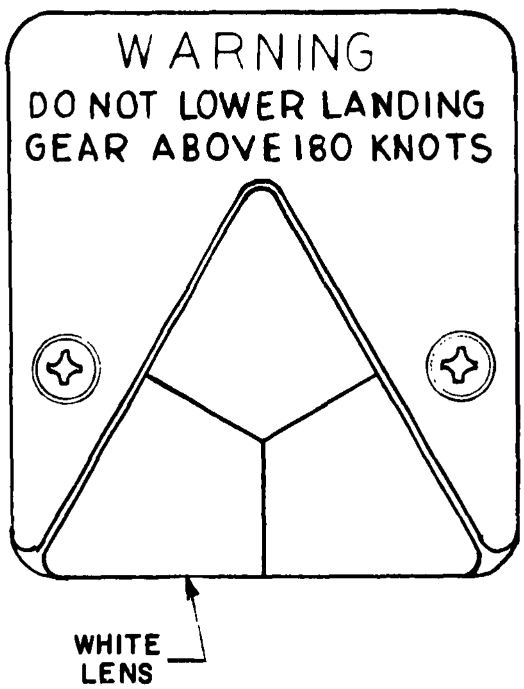


Figure IX-4. Landing Gear Indicator Light

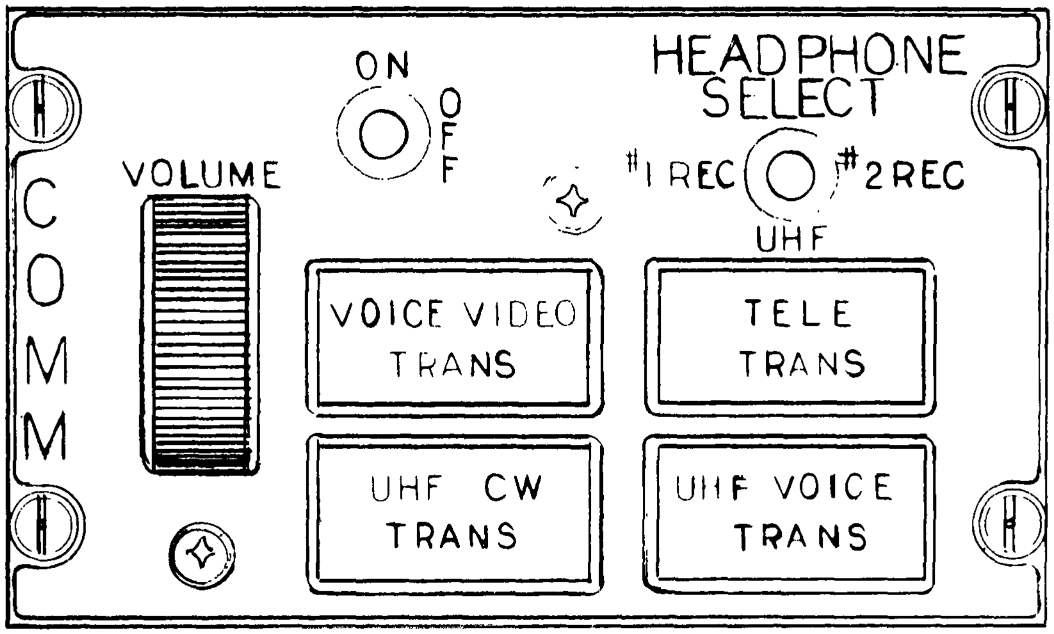


Figure IX-5. Communication Panel

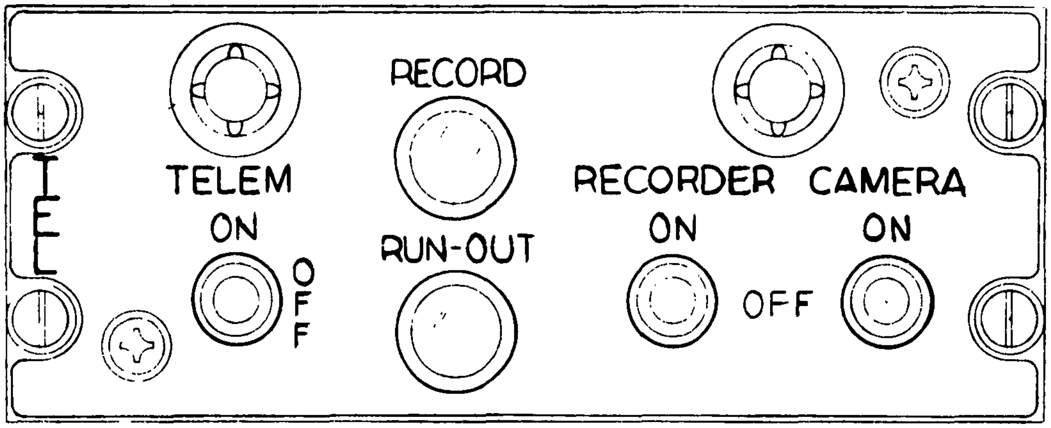


Figure IX-6. Telemetering Control Panel

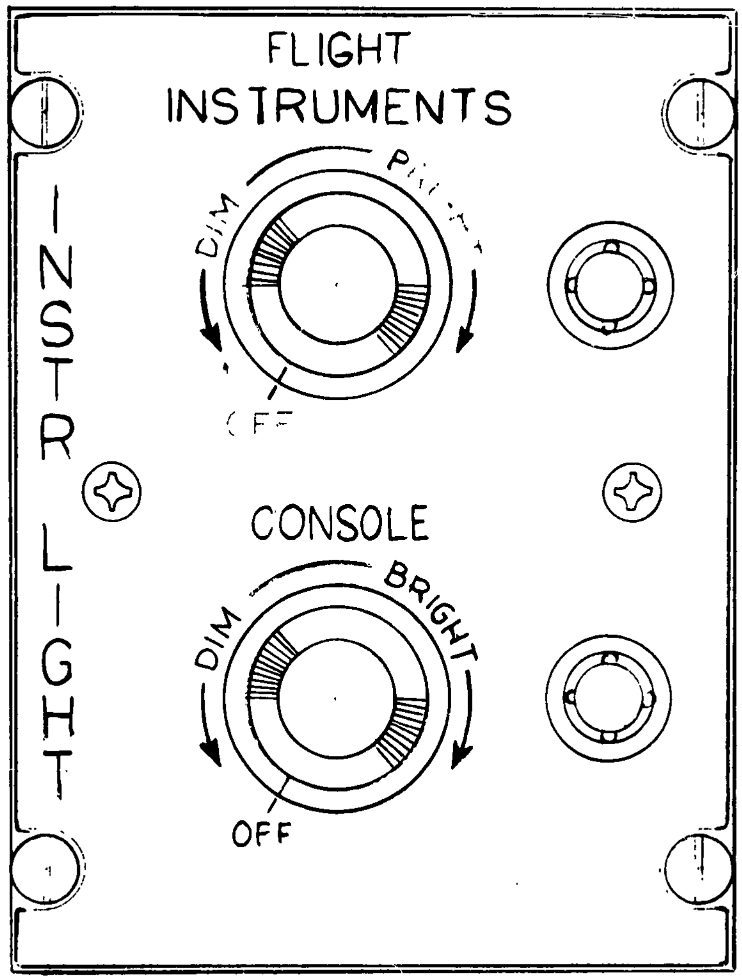


Figure IX-7. Flight Instrument Light Control Panel

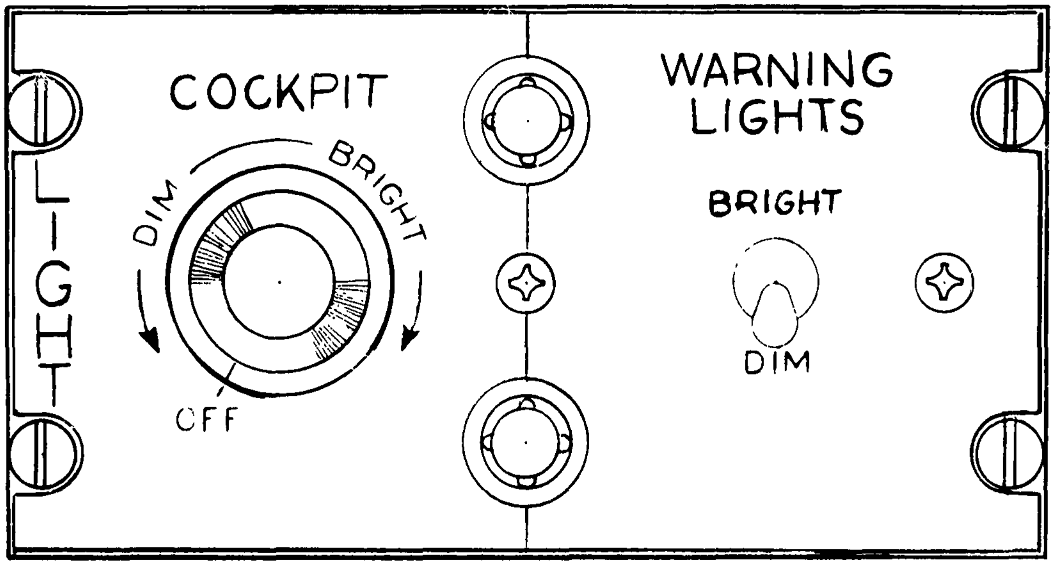


Figure IX-8. Cockpit/Warning Light Control Panel

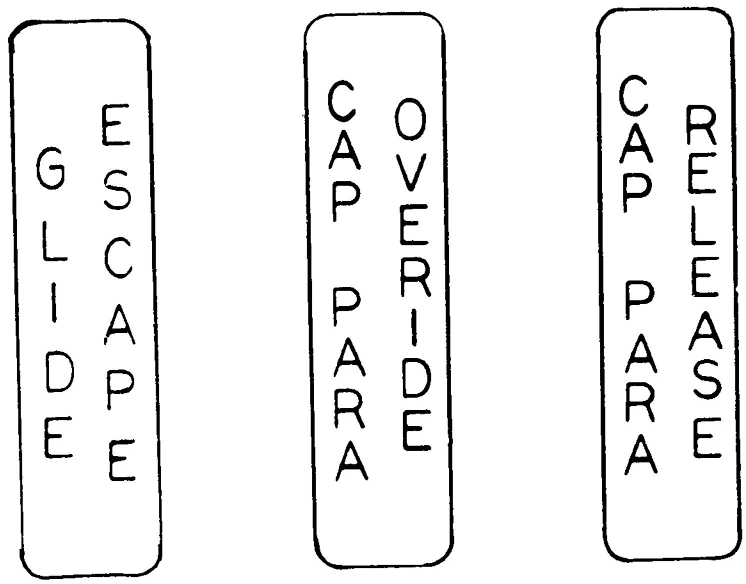


Figure IX-9. Release Handle

APPENDIX A

DESCRIPTION OF FLIGHT CONTROL SYSTEM OPERATION

A simplified schematic of the over-all flight control system is shown in Figure A-1. This diagram indicates the various ways in which the flight control signals are generated by the pilot and the guidance system and the alternate paths by which these signals are processed and transmitted to the surface actuators and reaction control thrust chambers.

1. Modes of Control

The airplane may be controlled in any of three automatic and one non-automatic modes. The automatic modes (Guidance Command Mode, Pilot Command Mode, and Piloted Flight Mode) are designed to operate the surfaces and reaction control thrust chambers through an automatic control channel. The non-automatic Backup Flight Mode bypasses the automatic control channel and is designed to provide the simplest possible control of the airplane.

a. Guidance Command Mode

In this mode, which is expected to be the nominal mode of operation, the FCS is tied directly to the guidance system and provides stability augmentation, automatic trimming, and signal limiting. The pilot's three-axis stick controller is de-energized to preclude inadvertent inputs. The inertial platform of the guidance system is used as the basic attitude reference for this mode. Thus, the inputs to the FCS are attitude and attitude command signals for pitch and roll axis control. The FCS controls the yaw axis in such a way as to provide turn coordination or to keep the airplane pointed in the general direction of the velocity vector. This is normally sensed by an accelerometer, except in the high altitude flight regime where the aerodynamic forces are small. In this case, the guidance system provides the difference between the heading and bearing which is nulled by the yaw channel of the FCS.

b. Pilot Command Mode

The function of the Pilot Command Mode is to provide the pilot with what is essentially a pilot relief autopilot with control stick steering. This mode might be used at the pilot's discretion during those phases of the flight when automatic flight path control is not available either by plan, through failure of the digital computer, should unanticipated flight conditions beyond the range of the computer be encountered, or at the pilot's option. The FCS provides the following features for this mode of operation.

(1) Rate of change of airframe pitch and roll proportional to side stick controller force.

(2) Pitch and roll attitude hold when the stick force is zero.

(3) Rate of change of yaw proportional to controller yaw axis force in the high altitude regime. Rate of change of yaw and lateral acceleration proportional controller yaw axis force in the low altitude regime.

(4) Stability augmentation.

(5) Automatic trimming.

This mode will be automatically synchronized to prevent transients at engagement.

c. Piloted Flight Mode

The function of the Piloted Foight Mode is to provide the pilot with a manual control system exhibiting optimized flying qualities. The FCS receives no direct signals from the guidance system and the pilot flies using visual information for attitude and flight path control.

This is the simplest of the automatic modes and may be utilized in the event of certain control system malufunctions, if there is a loss of dependable guidance information from the tie-in, or at the pilot's option.

The Piloted Flight Mode will have all the features of Pilot Command Mode with the following exceptions:

(1) The system will not automatically hold attitude when the controller is released.

(2) Trimming will be accomplished manually through momentary contact switches rather than automatically.

d. Backup Flight Mode

The Backup Flight Mode provides the pilot with a simplified basic manual control system. This mode is available at the pilot's option and provides a redundant path of controlling the surfaces and thrust chambers. The features provided include:

(1) Elevon and rudder displacement proportional to side stick controller force.

(2) Reaction control thrust chambers operated when controller is moved full deflection.

(3) Manual trim

2. Channels of Control

As shown in Figure A-1, there are two available control channels between the sources of the flight control signals and the actuation mechanisms, the automatic control channel and the backup channel, only one of which may be used at any time.

a. Automatic Control Channel

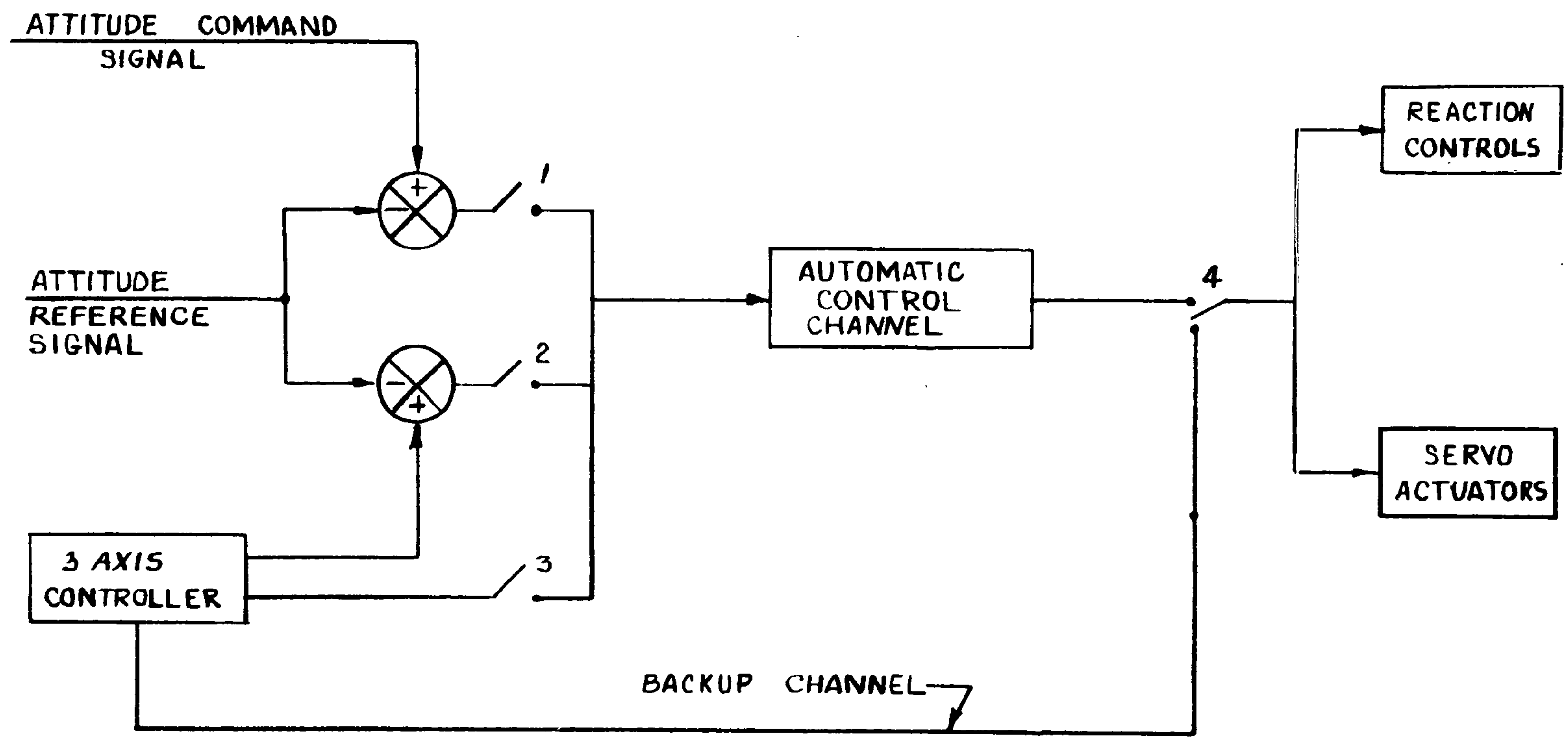
When any of the three automatic modes are engaged, the automatic channel must be utilized. It is the primary function of this channel to process the signal inputs such as to optimize the airplane's flying qualities despite wide variations in flight conditions. An additional function is to coordinate the utilization of the aerodynamic control surfaces and reaction controls so as to obtain an optium compromise between airplane dynamic performance and propellant consumption.

Flying qualities are optimized in two ways. First, the automatic channel, by use of self-adaptive control techniques, keeps the airplane's apparent dynamic stability constant. This is done by varying the gain of the stability augmentation system so that the airplane natural frequency and damping ratio are held at optimum values. And second, the inputs from the pilot's controller are connected such that rates of change in roll, pitch, and yaw are kept proportional to the forces applied to the controller.

During flight, the output of the automatic channel is normally connected to the servo actuators and the reaction controls. When the reaction control system is armed, in the high altitude flight regime, it is important that the reaction controls are operated only when necessary so that the propellant consumption can be kept low. The automatic channel, therefore, is designed such that all trimming moments are supplied by the control surfaces. The reaction controls will not operate until a limited error in rate and/or attitude is developed. Within these limits the surfaces, if sufficient effectiveness exists, will supply the control moments.

b. Backup Control Channel

The backup channel is utilized only in the Backup Flight Mode and serves to connect the cockpit controller directly to the servo actuators and reaction control thrust chamber valves. It is the purpose of this channel to provide the pilot with a simplified, highly reliable, redundant control path. To assure high reliability, the backup channel is designed to obviate the need for electronic amplifiers and their attendant complexity. Except for electrical trim provisions and necessary relay contacts, the backup channel consists essentially of wires between the cockput and the actuation devices.



SWITCH	FUNCTION
1	GUIDANCE COMMAND MODE
2	PILOT COMMAND MODE
3	PILOTED FLIGHT MODE
4	BACKUP FLIGHT MODE

Figure A-1. Flight Control System Simplified Schematic Diagram

APPENDIX B

DESCRIPTION OF MOVING TAPE INSTRUMENTS

1. Reason for Choice

The synthesis of the instrument panel for the Dyna-Soar I airplane must take into account the total mission profile from take-off to landing. Analysis of the information required by the pilot revealed that this information generally falls into two categories:

(1) Information that conveys knowledge to the pilot as to what he is doing (actual performance).

(2) Information that indicates what he should be doing (desired performance).

In semi-automatic or fully automatic modes of flight, the pilot's job becomes that of monitoring the automatic performance, and assuming manual control when the indicated performance warrants such action. The display requirements for monitoring differ for those of manual control because of the readout accuracies required and the dynamics of the situation. Display emphasis must, therefore, be given to that information necessary to achieve manual control of the airplane. It indicates the desirability of achieving display dynamics consistent with the control actions required.

The next step in the layout of the instrument panel was the determination of some natural frames of reference for the panel which are consistent with external information available to the pilot. The information necessary for the pilot to control the aircraft falls logically into two categories:

(1) Those which combine to provide the pilot with a forward looking or elevation view such as pitch altitude, altitude, and rate of climb.

(2) Those which combine to provide the pilot with a downward looking or plan view such as heading, bank, and turn rate,

The fundamental problem posed by these two instrument groups entails the selection of a display technique which would effectively integrate command information with actual performance and to provide for an instinctive control-display interaction.

For example, the display of command information on a standard three pointer altimeter would require the addition of three more needles rendering that instrument worthless because of the complicated readout problem. The use of digital type displays alone must be ruled out because these devices would require a continuous cross check by the pilot. In addition, these devices do not permit an instinctive realization (both quantitative and qualitative) of the control action required to follow the given command, especially under dynamic conditions.

The choice of moving tape instruments for the display of these parameters is nothing more than the end result of a process of elimination. The forward looking group has vertically moving tapes and those of the downward looking group move horizontally. Digital readouts are generally provided to convey to the pilot a precise indication of actual conditions. References 1 and 2 describe in detail recent work done in the selection and integration of displays.

2. Description of a Typical Instrument

Figure B-1 shows the face of a typical altitude and rate of climb indicator of this type. The required expanded scale is printed on a tape, stored on suitable spools, and moves vertically behind a fixed bar at the center of the instrument. This arrangement permits the reading of the instrument to a close tolerance, and the rate of motion of the tape conveys to the pilot an instinctive assessment of the dynamic situation. Only a portion of the tape, in general 3 to 25 percent will be visible at any one time. The exact amount depends on the required reading accuracy and the total expected range of the displayed parameter.

Command or programmed information is displayed by means of a double bar across the face of the tape moving parallel to the tape motion. It should be noted that the actual commanded value is indicated by the relationship between the double bar and the tape beneath it, and the error between commanded and actual performance by the distance between the fixed readout needle and the moving double bar.

This class of instruments are remote indicators, that is the sensors, amplifiers, and the like are remotely located and the tape and command bar are driven by servo mechanisms. For this reason they are often referred to as servoed tape instruments. The Lear, Inc., Model 3905A Altimeter is a typical instrument of this class.

Figure B-2 is a simplified block diagram of an Altimeter.

Flight test results and pilot comments on these instruments are given in detail in Reference 2. In general, pilot comments were very favorable.

Two of the problems associated with these instruments have been tape stability and the precision gearing required to eliminate back lash. Recent tape developments (Mylar Tape) appear to have, to a large extent, alleviated the tape problem. The use of electromagnetic clutches may also eliminate many of the presently required precision gears.

Reference 1 - WADC Technical Report 56-582, The Air Force Program for Improved Flight Instrumentation.

Reference 2 - WADC Technical Report 58-431, The Air Force Integrated Flight Instrument Panel.

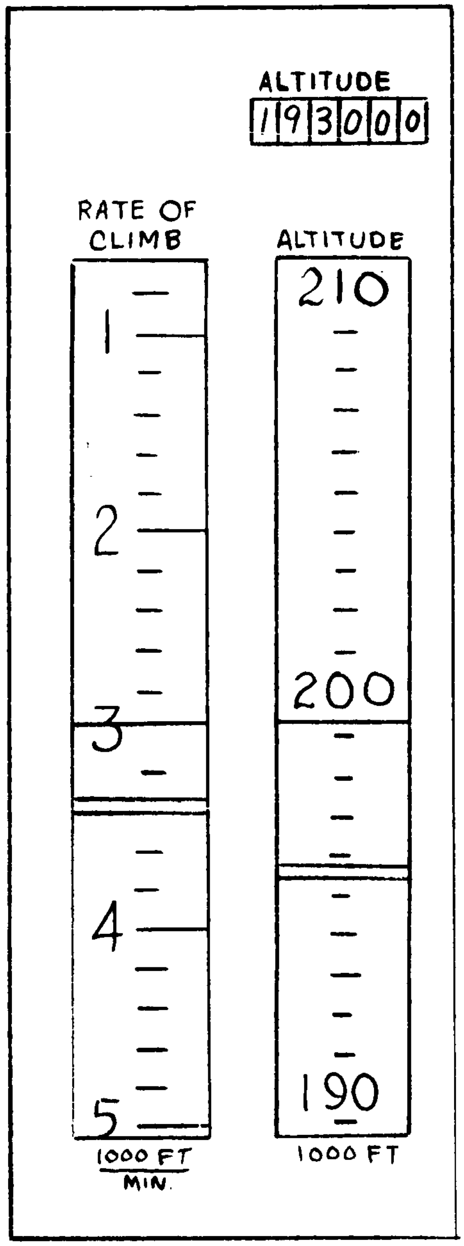


Figure B-1. Altitude-Rate of Climb Indicator

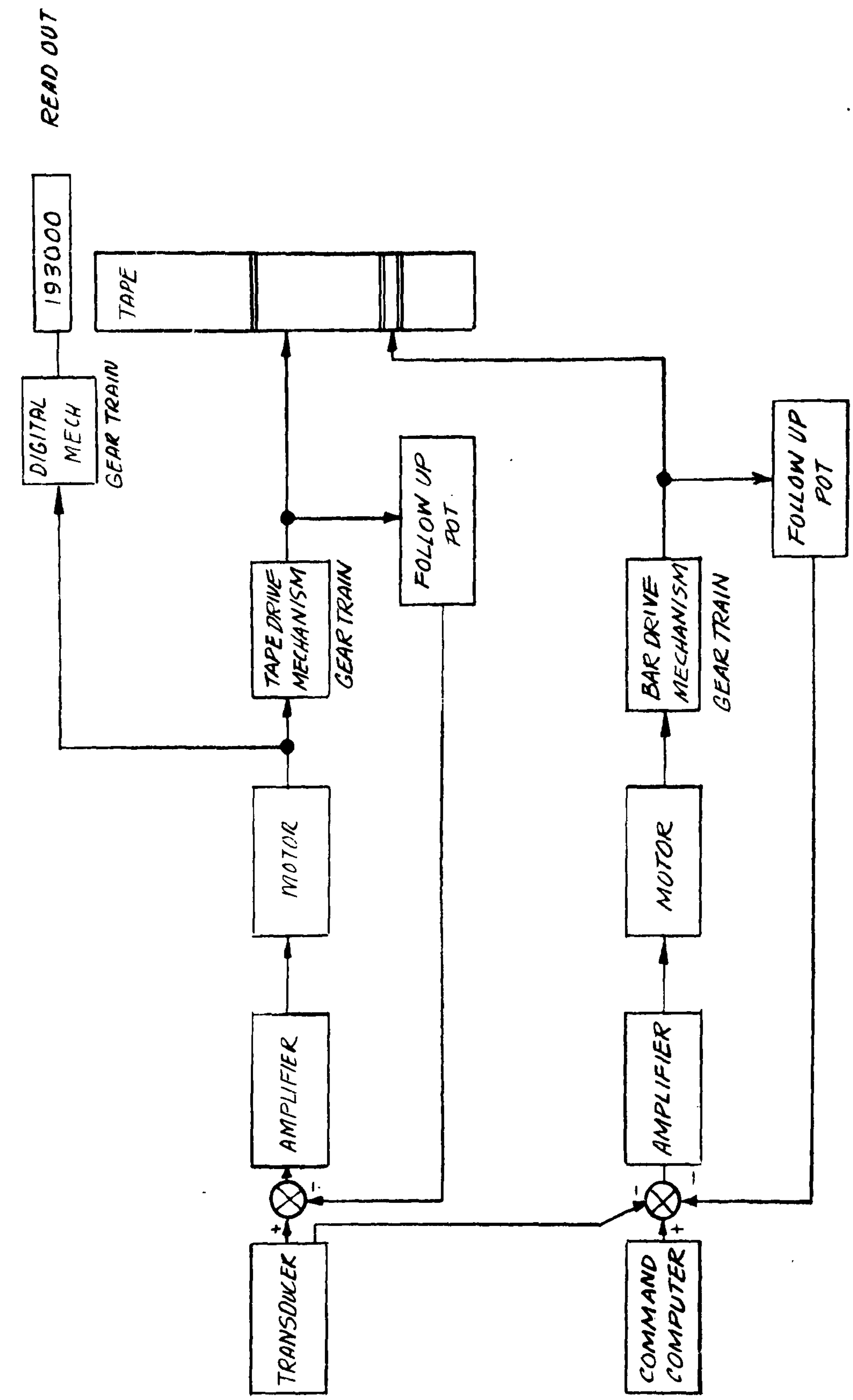


Figure B-2. Block Diagram - Moving Tape Instrument

APPENDIX C

DESCRIPTION OF GRAPHIC SCREEN DISPLAYS

The graphic screen display panel incorporated in the cockpit provides the following displays to the pilot:

(1) Boost corridor graphic screen display.

(2) Lift control graphic screen display (Phase I of Glide).

(3) Velocity control graphic screen display (Phase II of Glide).

(4) Altitude control graphic screen display (Phase III of Glide).

All these displays are presented during appropriate portions of the flight on a ground glass screen located at position 18 in Figure III-5. The screen is located in the aperture opening on the forward looking window and the pilot may fold the screen out of the way to make use of the window. The controls for this display are located at position 63 in Figure III-5.

Utilization of the graphic screen display technique offers the following advantages:

(1) The pilot is presented with advance information of required flight path changes for energy management purposes.

(2) The display shows allowable limits at all times independent of the INS computer. This permits the pilot to

monitor the performance of the computer.

(3) In the absence of the computer the pilot is provided with the necessary information to control his energy during glide.

(4) The screen display serves to focus the pilot's attention on those guidance parameters which are of particular importance during the various phases of flight.

(5) The screen display permits optimum scheduling of the limited display space available. That is, when one phase of flight is completed, the display associated with that phase can be replaced by the next applicable display.

The graphic screen display mechanism is similar to that used for the map display. (Figure VI-4.) Since the space utilized by the graphic screen is also used for viewing through the window, it is necessary to locate the mechanism remotely from the screen. The mechanism therefore is mounted just forward and below the map mechanism and a stowable mirror is used to project the image to the rear of the ground glass display screen.

The chart to be displayed is engraved on a glass slide mounted on a carriage which is moveable in an X and Y direction. The X axis is driven by velocity in the boost phase and by range-to-go in the glide phase. The Y axis is driven in the consecutive phases by geometric altitude, density altitude, velocity, and the geometric or barometric altitude. The slide is moved through the optical system so that only a small portion, suitably expanded, is displayed on the screen at any instant.

A reticle carriage is provided in the optical system which causes the image of a marker to be superimposed on the image of the chart. This marker is driven in X and Y coordinates such that its position relative to the chart represents the present flight condition. The marker may also be rotated so that it can be used to indicate the relative direction in which conditions are changing with respect to the curves on the chart.

Glass slides suitable for each phase are fed into the slide carriage by operating the Energy Panel shown in Figure IV-5. Prior to take-off the boost button is depressed. At separation, the lift control phase is entered and the Lift Control button is depressed to insert the proper slide and reset the mechanism. The slides are suitably keyed to insure proper alignment. As the succeeding phases are entered the proper buttons are used. Finally when the pilot wishes to use the window, the Display Stow button is depressed causing the screen and mirror to fold out of the way. The buttons are provided with integral lighting to indicate the display in use.

The airplane position knobs are used for initial alignment of the marker and are normally locked during flight. The display illumination knob adjusts the intensity of the projection light source. Two filaments are provided in the projection lamp as a reliability measure. The auxiliary filament may be energized by depressing the illumination knob.

APPENDIX D

ELECTROLUMINESCENT DISPLAYS

Technical literature shows much work in progress in the field of electroluminescence. Recent work at an advanced level by many firms clearly indicates the feasibility of employing electroluminescent (EL) panels for aircraft data display. An EL panel is basically a solid state device consisting of a phosphor material sandwiched between two conducting electrodes. The electrodes are normally fabricated from a transparent material and bonded to glass plates. Such a device may be likened to a large condenser whose insulator glows when an alternating current is applied. By use of a system of cross wires the panel may be excited to light at any point and show images with gray scale contrast, outlines, etc. Thus data readout, image formation, symbol presentation, etc., are now possible with present panels of this type. Input data form may be electrical, such as time-based video, digital computer outputs etc., or may be optical, from projected film or slides. EL panels may also be made semi-transparent, to see through, and can store images for short periods, as well as serve as storage light amplifiers.

A flat screen EL panel is extremely attractive as an aircraft display device from several standpoints. The screen and auxiliaries are lightweight, draw low power, are environment resistant, and have inherently high reliability. The screen requires only an inch or so of panel depth, is non-implosive, and can be made shatter proof. From a standpoint of instrument panel usage efficiency the device has great potentiality, since different data could be displayed during different phases of flight. Thus, only the important parameters required during a particular phase of flight need be displayed while other parameters required for other phases may be displayed when they are required.

1. Utility

In the instrument panel as presently conceived, EL panels could be used to replace the glass screens now used for the map display and graphic screen display. In addition they could be used to replace the failure warning panel, and many of the moving tape instruments.

For future applications in Dyna-Soar I, EL panels could be used for new integrated display concepts:

(1) To present groups of data only when needed during a given flight phase.

(2) To keep important data always in most valuable center section of displays.

(3) To eliminate all electromechanical instruments, except a very few backups, in interests of high reliability and reduced weight.

(4) To show data in forms most efficient and understandable from a psychological standpoint, implementing the best new display concepts. Its image storage feature will aid ground pattern interpretations.

(5) To present data not displayed by any other system e.g., maneuver possibilities as read out from a computer.

(6) To provide a common final viewing device for all physical reconnaissance system for Dyna-Soar I, whether research or operational gear; for active microwave, passive microwave, infra-red, visible, etc., including moonlight-starlight image amplification.

2. History of Activity to Date

(1) A survey was made of industry activity in the field. This survey showed over 35 vendors actively engaged in EL development in various fields, i.e. aircraft instruments,

lighting devices, X-ray, etc.; with marked progress over the last two years.

(2) A preliminary performance specification was formalized listing electrical mechanical and optical requirements which would be required for the development of a desired EL display system, to be flight testable by June 1960.

(3) Requests for proposals with the specification attached were submitted to over 30 vendors; this yielded proposals from seven responsible companies.

(4) A comparative analysis of the seven proposals was conducted; then two of these were selected for final consideration since both vendors showed plausible evidence of meeting the specification requirements. In addition, both vendors appear well qualified technically and would be able to produce demonstrable working hardware (in mockup form) in the near future.

3. Performance Requirements

Paragraphs a through n list performance requirements as thought to be necessary for an EL display system for Dyna-Soar I.

a. General

The EL screen shall be designed as a flat rectangular panel approximately one foot by one foot with straight edges. Nonfunctional borders shall be less than 1/2 inch at all margins and over-all panel depth shall be less than two inches. The system should lend itself to easy installation and removal in the Dyna-Soar vehicle and have inherently high reliability and maintainability. The puncture of the display by fragments from various sources should result in no hazard to crew or equipment attributable to display disintegration. If possible, the design should ensure that the image generated in the display be lost only in the immediate area of the damage. Consideration shall be given in the design to the use of miniaturized, transistorized, modularized, ruggedized components wherever possible. The design shall minimize corona, emission of electromagnetic or acoustic noise, and other undesirable characteristics which could disturb the crew or other airborne equipment.

b. Resolution

The EL screen shall have a minimum resolution of 40 lines per inch for electronically inserted data and 200 line per inch for optically inserted data. A resolution of 50 lines per inch shall be a design goal for electronically inserted data.

c. Frame Rate

The EL screen should accept and present images at frame rates up to 24 frames per second without flicker, blur, or undesirable overlap. Interlace schemes may be used.

d. Brightness

The time averaged highlight brightness shall be uniform across the screen and continuously adjustable to a maximum of 20 foot-lamberts. (It is recognized that this may demand non-stored instantaneous brightness peaks an order of magnitude higher). This requirement shall not, however, be obtained at the expense of resolution loss or low frame rates and where this is the case, may be compromised to a minimum of 10-foot lamberts.

e. Contrast

A contrast ratio of 10:1 maximum (subjective) with nominal cockpit illumination and no external illumination, with at least 10 gray scale steps, as in RETMA test charts, shall be a design goal. (Reflection reducing coatings may be used.) This requirement shall not be obtained at the expense of resolution loss and frame rate and where this is the case may be compromised to a minimum of five gray scale steps.

f. Persistence and Storage

The EL screen should be capable of retaining as desired, any single frame image for times up to 30 seconds, preferably by means contained within the screen (such a photo conductive regeneration) and acceptably by storage in the screen support devices. Storage for one minute shall be a design goal.

g. Color

This requirement is not considered critical and peaks in green near 5600 A shall be acceptable.

h. Character Display

The EL screen should adequately display symbols, characters, and closely grouped patterns without interaction between adjacent screen elements (within the resolution units defined). Ghost images and unwanted "spot blooming" shall be eliminated.

i. Data Insertion

The screen adjunct devices shall accept time based electrical or video signals or readout from a digital computer, prepare these for insertion and apply signals to the screen electrode matrix. Equipment utilized for data insertion shall not limit the over-all ability of the system to meet the requirements of frame rate, resolution, and other performance requirements. Optical data insertion, by means of images projected on the rear face of the screen and viewed by a "see through" method or by light amplification techniques shall also be possible.

j. Optical Quality

The minimum desirable total light transmission shall be 70 percent (flat over the visible spectrum) for a semi-transparent screen. The optical distortion under these conditions shall not exceed that of present day fighter windshields.

k. Spectral Conversion

Other than visible light, it is desirable that radiation incident on the rear surface of the screen such as microwave, ultra-violet, far infra-red, near infra-red, etc., be converted within the screen (or a phosphor panel adjunct) into a moderate brightness visible image. Information on the spectral conversion frequency spectrum shall be specified by the contractor.

l. Duty Cycle

The EL system shall be capable of a four-hour duty cycle without the need for major adjustments.

m. Life

As a design goal, the EL system shall have an operating life of 5,000 hours and a storage life of five years minimum. A 20 percent loss in brightness shall be used as a criterion to determine operating life. Components and other parts subject to deterioration due to the effects of humidity and other environmental stresses shall be protected by hermetic sealing, pressurizing, etc., where such techniques will prolong the life of the system.

n. Electrical

The EL system shall meet its performance requirements when supplied with electrical power in accordance with Specifications MIL-E-26144 and MIL-E-7894A.

A summary in tabular form of the comparative analysis conducted on the EL panel proposals is presented in Table D-1.

TABLE D-1

COMPARATIVE ANALYSIS SUMMARY OF ELECTROLUMINESCENT DISPLAY PROPOS

BIDDERS	IMAGE AREA 1' x 1'	DEPTH < 2"	WT. 3#/Ft^2	BRIGHTNESS 20 FT-LAMB.	CONTRAST > 10:1 10 GRAY STEPS	RESOLUTION 50 LINES/IN. & 200/IN OPT.	PERSISTANCE AND STORAGE	FRAME RATE 24/SEC.	DATA INSERT
COMPANY "A"	Yes	Yes	Not Given But Seems Fea-sible	20 Ft-Lamb at 10 Frames Per Sec. (Flicker-less) (Unknown at 24 Frames Per Second)	10:1 5 Gray Scale Steps Mini-mum; Possi-bly More	Can Obtain 40 Lines/In (50 is Objec-tive)	Indefinite Storage in Screen Via PC Feedback	Can obtain 24 Frames Sec. at Some Brightness Sacrifice	Electr ically Optica
COMPANY "B"	Yes	Yes	Not Given But Seems Fea-sible	Sufficient Brightness for Satis-factory Visual Presentation	Sufficient for Satis-factory Visual Presentation	Appears Feasible to Obtain 50 Lines/In.	Storage in Switching Tube – No Screen Storage	Scan at 24 Frames/ Sec.	Electr ically Switch Tube a Optica Via Ba Lighti
COMPANY "C"	Yes	Yes	Approx. 1-1/2 Lbs. per Ft^2	> 8-12 Ft. Lamberts in Storage Mode	10:1 Constant Only 4 Gray Scale Steps	30 Lines/In. 50 Lines/In. May be Attained Later in Program	Storage in Screen Using Photo-Cond. Present Maximum is 10 Seconds	10 Frames Per Second 24/Sec. May be Attained Later in Program	All Inp Optica Electr Data C verted Optica CRT S
COMPANY "D"	Yes	Yes	Yes	Goal of 20 Ft. -Lamberts	10:1 for Optical None for Matrix	Goals as Stated	No Storage for Matrix Storage on Optical Screen Via PC Regenera-tion	Goal of 24 Per Sec. for Optical Not Stated for Matrix	Optica to Opti Screen Electr Data to Matrix
COMPANY "E"	Yes	Yes	3 to 5 Lbs.	Yes (has been demonstrated)	Yes (has been demonstrated)	No 16 Lines/In. Appears Max. and Present Screens have only 5-10	Yes Indefinite Storage in Screen	Yes (Has been demon-strated)	All Inp Electr Optical Must b verted Electr Info.
COMPANY "F"	No Data	No Data	No Data	No Data	No Data	No Data	No Data	No Data	Optical from C
COMPANY "G"	Yes	6"	Yes	Yes	Yes	Yes	Not Given	Yes	Optical Using U Light

TABLE D-1

RY OF ELECTROLUMINESCENT DISPLAY PROPOSALS (IN ORDER OF RANKING)

SOLUTION LINES/IN. 200/IN OPT.	PERSISTANCE AND STORAGE	FRAME RATE 24/SEC.	DATA INSERTION	REMARKS
n Obtain Lines/In) is Objec- ve)	Indefinite Storage in Screen Via PC Feedback	Can obtain 24 Frames Sec. at Some Brightness Sacrifice	Electron- ically and Optically	New approach is used and system shows a great deal of promise. An EL panel with a light amplifier and a storage light amplifier is proposed. Switching accomplished via CRT which can be located remotely. Panel semi-transparent. Appears to have good growth potential.
pears asible to tain Lines/In.	Storage in Switching Tube - No Screen Storage	Scan at 24 Frames/ Sec.	Electron- ically to Switching Tube and Optically Via Back Lighting	Semi-Transparent Screen. Optical data presented on back of screen via "see through" method. Screen not capable of light amplification. Switching CRT built as "frame" around screen using "line scan". It appears that average brightness and contrast (gray scale) will be biggest problems.
Lines/In. Lines/In. y be tained ter in ogram	Storage in Screen Using Photo-Cond. Present Maximum is 10 Seconds	10 Frames Per Second 24/Sec. May be Attained Later in Program	All Inputs Optical. Electrical Data Con- verted to Optical Via CRT Screen	Optical data presented on rear face, screen is essentially a light amplifier modulated via CRT. A folded optical system could present superimposed images simultaneously with com- puter or video images. An advanced system proposed-screen is incorporated in a flat CRT and all inputs would be electrical. It appears major problem is frame rate due to PC cells.
als as ted	No Storage for Matrix Storage on Optical Screen Via PC Regenera- tion	Goal of 24 Per Sec. for Optical Not Stated for Matrix	Optical Data to Optical Screen. Electrical Data to Matrix	Two parallel adjacent screens; an EL-PC system to receive optical info and a EL matrix screen for electrical data. Two images could be displayed simultaneously. No info on how problems of brightness will be solved on the EL screen or how problem of frame rate on the EC-PC screen will be solved. NOTE: Very doubtful that specifications of brightness, contrast, resolution, or frame rate can be met with screens displayed.
Lines/In. pears Max. l Present reens have ly 5-10	Yes Indefinite Storage in Screen	Yes (Has been demon- strated)	All Inputs Electrical. Optical Info. Must be Con- verted to Electrical Info.	Screen not transparent, optical data would require projec- tion on face of screen or conversion to electrical signals. Screen not capable of light amplification directly. Switch- ing performed via pulse circuitry (no CRT). This screen appears to be further along in development than other types. Problem of resolution is one of fabrication techniques.
No Data	No Data	No Data	Optical Inputs from CRT	An optical system is proposed using a light amplifier type EL screen and CRT tube. System could accept direct optical image. Also the EL screen would be subcontracted. Consider- able detail on data insertion and function generating tech- niques presented. No info on screen presented.
Yes	Not Given	Yes	Optically Using U.V. Light	Proposal not on electroluminescent panel. Panel proposed is fluorescent display device. Bidder not considered.

2

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

MEMORANDUM FOR: DTIC – OQ
Attn: Larry Downing
8725 John J. Kingman Rd.
Ft. Belvoir VA 22060

23 October 2006

FROM: 88 CG/SCCM (FOIA Office)
Bldg 1455
3810 Communications Blvd
WPAFB OH 45433

SUBJECT: Freedom of Information Act (FOIA) Cases, WPAFB FOIA Control # 06-443AB and 06-444AB

1. On 17 May 2006, we received a FOIA request for document:

   AD125726, "MX-2276 Reconnaissance Aircraft Weapon System. System Design," Bell Aircraft Corporation, 1 Dec 1955, 157 pages.

   AD318479, "Dyna-Soar. Cockpit Display Report," Bell Aerospace, March 1959

2. The documents have been reviewed by the Aeronautical Systems Center (ASC/XR), and it has been determined that the distribution statement should be statement A (publicly releasable).
3. I am the point of contact and can be reached at (937) 522-3092 or DSN 672-3092.

ABBY L BOGGS
Freedom of Information Act Analyst
Management Services Branch
Base Information Management Division

Attachment
AFMC Form 559
AFMC Form 556
FOIA Request